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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 4: AIRFRAME - LIST OF CHANGES

The following table details the changes applied to this revision:

ATA NAME AND NUMBER	CMUI NUMBER	CHANGES APPLIED
Front Page	CS130-21.05-06.00-121418	Version number and CMUI number
Acronyms and Abbreviations	CS130-21.05-06.00-121418	Deleted BCT (bus tie contactor current transformer) and added MKP (multifunction keyboard panel) (pges ii and viii)
List of Changes	CS130-21.05-06.00-121418	New CMUI
25 Equipment and Furnishing	CS130-21.05-06.00-121418	Moved Tests before CAS Messages and updated CAS Messages text (pges 25-58 to 25-63)
27 Flight Controls	CS130-21.05-06.00-121418	Updated Sidestick Controller section; added Interface subsection and updated graphic (pges 27-14, 27-15)
		Updated Primary Flight Controller Computer section (text and graphics) (pges 27-20 to 27-25)
		Updated Sidestick Controller section; deleted Main Body subsection; added Mobile Assembly subsection and Pitch and Roll Axis Assembly subsection; updated graphic accordingly (pges 27-32, 27-33)
		Updated graphic (pges 27-41, 27-43)
		Updated Remote Electronic Unit section (text and graphic) (pges 27-44, 27-45)
		Updated Primary Flight Control Computer Control Interfaces section (text and graphic (pges 27-60, 27-61)
		Added Flight Control Rigging Reference Marks section with graphic and Flight Control Rigging section with graphic before Onboard Maintenance System Functions section (pges 27-90 to 27-93)
		Added Aileron, Elevator, and Rudder Surface Rigging subsection (pge 27-94)
		Updated graphic (pges 27-95, 27-97)
		Added NVM Clear and Latched Fault Clearing section with graphic (pges 27-100, 27-101)

ATA NAME AND NUMBER	CMUI NUMBER	CHANGES APPLIED
27 Flight Controls (cont)	CS130-21.05-06.00-121418	Updated graphic (pge 27-103)
		Added NOTE (pge 27-120)
		Updated Horizontal Stabilizer Trim System - General Description section (pge 27-150)
		Updated figure title (pge 27-151)
		Added Upper Gimbal Assembly subsection and Lower Gimbal Assembly subsection in Horizontal Stabilizer Trim Actuator section; updated graphic accordingly (pges 27-152, 27-153)
		Changed ‘Ballscrew Assembly’ heading to ‘Ballscrew Assembly and No-Back Device’ and updated text; added Primary Load Path subsection; updated figure title and graphic accordingly (pges 27-154, 27-155)
		Replaced Secondary Nut Assembly and Tie Rod reset section with Secondary Load Paths section (text and graphic) (pges 27-156, 27-157)
		Added Secondary Nut Assembly and Tie Rod Reset section with graphics (pges 27-172 to 27-175)
		Added NOTE and WARNING (pge 27-214)
29 Hydraulics	CS130-21.05-06.00-121418	No change
32 Landing Gear	CS130-21.05-06.00-121418	Added NOTE (pge 32-32)
		Updated graphic (pge 32-69)
		Updated Synoptic Page (pges 32-87, 32-93, 32-95)
35 Oxygen	CS130-21.05-06.00-121418	No change
38 Water and Waste	CS130-21.05-06.00-121418	No change
50 Cargo Compartments	CS130-21.05-06.00-121418	No change
51 Standard Practices - Structures	CS130-21.05-06.00-121418	No change
51A Standard Practices and Structures (Damage Classification, Assessment and Repair) - Example	CS130-21.05-06.00-121418	No change
52 Doors	CS130-21.05-06.00-121418	Updated Cargo Compartment Doors Practical Aspects section (pges 52-132 to 52-135

ATA NAME AND NUMBER	CMUI NUMBER	CHANGES APPLIED
53 Fuselage	CS130-21.05-06.00-121418	Changed ‘five additional frames’ to ‘four additional frames’ (pge 53-34)
		Added NOTE (pge 53-66)
54 Nacelles and Pylons	CS130-21.05-06.00-121418	Updated thrust reverser (pges 54-3, 54-5, 54-7, 54-9, 54-11, 54-15)
55 Stabilizers	CS130-21.05-06.00-121418	No change
56 Windows	CS130-21.05-06.00-121418	No change
57 Wings	CS130-21.05-06.00-121418	No change

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ATA 25 - Equipment and Furnishing

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

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EQUIPMENT AND FURNISHING - CHAPTER BREAKDOWN

25-10 FLIGHT DECK

GENERAL DESCRIPTION

The flight deck can accommodate three occupants: a pilot, copilot, and observer. The aft bulkhead of the flight deck contains miscellaneous stowage compartments, the observer seat and the flight deck door. The flight deck has an overhead hatch for emergency exit.

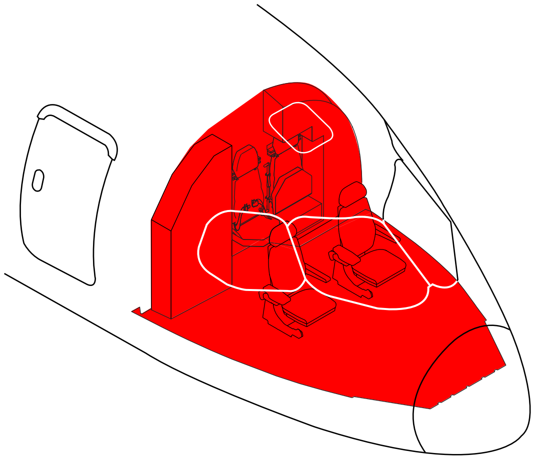
The pilot and copilot seats are floor mounted, and mechanically and electrically actuated. The seat position can be adjusted to accommodate a wide range of occupant sizes. The seats have a five-point restraint system to restrain the occupants during takeoff, turbulence, and landing.

The observer seat accommodates a third person in the flight deck. The stowed seat is located on the aft bulkhead, behind the pilot seat. When deployed, the seat sits in front of the flight deck door, providing the observer with an unobstructed view of the flight deck controls and displays. The seat has a five-point restraint system.

COMPONENT LOCATION

The flight deck contains the following:

- Pilot and copilot seats
- Observer seat



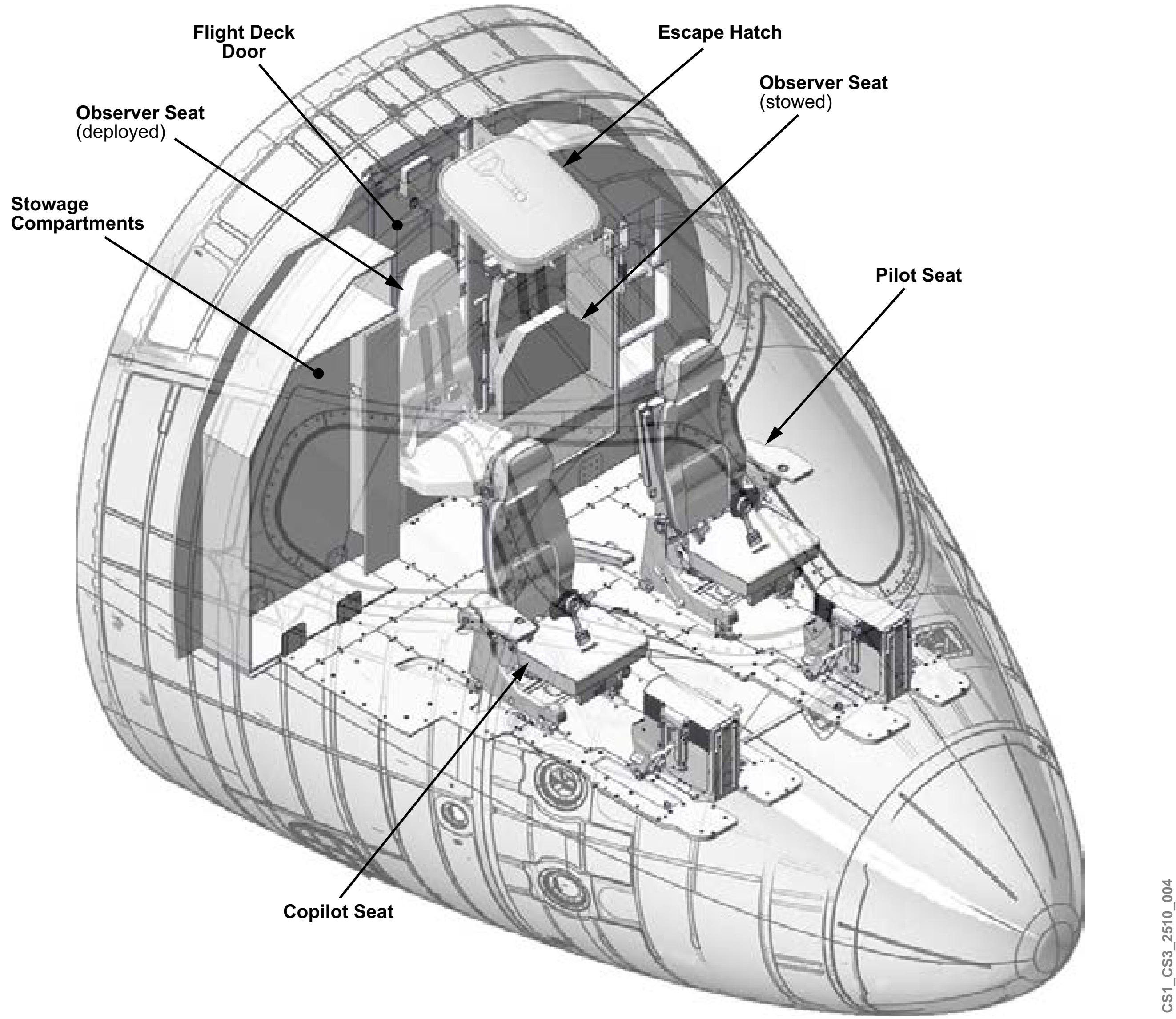


Figure 1: Flight Deck

COMPONENT INFORMATION

PILOT AND COPILOT SEATS

Each seat is installed on two F tracks that provide 22.9 cm (9.0 in.) of fore and aft adjustment. Ten locking pin holes, spaced 2.5 cm (1.0 in.) apart, allow the occupant to move the seat forward and rearward, and lock the seat in the desired position.

There is one lateral lock position 10.2 cm (4.0 in.) outboard at the rear most portion of the travel. This permits lateral seat movement and stowage of the seat to allow for occupant exit. The seat has 17.8 cm (7.0 in.) of electrically powered vertical seat movement, with a mechanical backup adjustment system.

The seats have the following adjustment features:

- Lumbar adjustment
- Recline adjustment
- Manual and electrical vertical adjustments
- Fore, aft, and lateral adjustments
- Headrest adjustment
- Armrest tilt and vertical adjustments
- Seat pan pitch adjustment

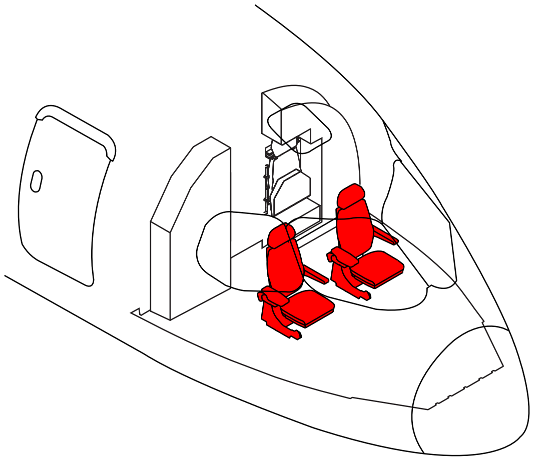
A power cutoff switch disables the seat electronic control unit in the event of an electrical failure in the seat.

The outboard armrest supports the use of the sidestick controller and is mounted to the seat assembly.

The seat assembly incorporates a five-point safety restraint system consisting of a lap belt connector, a rotary buckle, two shoulder belts, and a crotch belt.

The seats also have electrical connectors for the seat electronic control unit, and crew life vest located under the seat.

A step mounted to the upper portion of the inboard side of the seat back permits emergency exit through the overhead escape hatch.



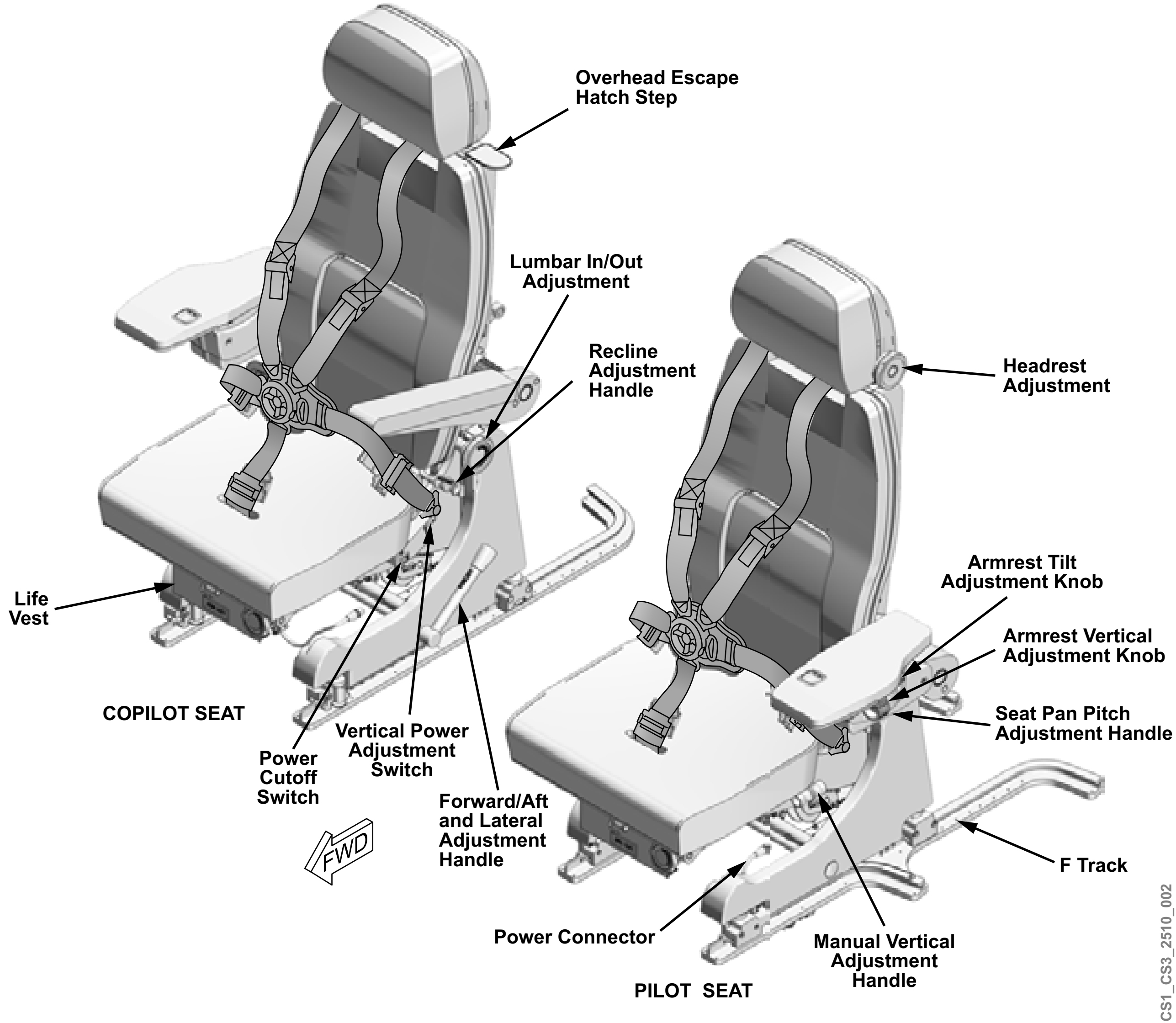


Figure 2: Pilot and Copilot Seats

OBSERVER SEAT

The observer seat assembly is located at the aft bulkhead, forward-facing, behind the pilot seat. The assembly is mounted on upper and lower slide rails. When stowed the seat is held in place by a restraint latch.

To deploy the seat, the restraint latch is lifted and the seat is pulled out of the stowed position using the seat handle.The seat engages a support bracket mounted on the bulkhead across the isle. The observer seat is fully deployed once it engages the support bracket.

When the occupant stands the seat base folds up automatically.

To stow the seat, the release handle on the support bracket is lowered. The seat can then be moved to the stowed position behind the pilot’s seat. The restraint latch is then lowered to secure the seat in place.

The seat is furnished with a five-point restraint system similar to the pilots seats.

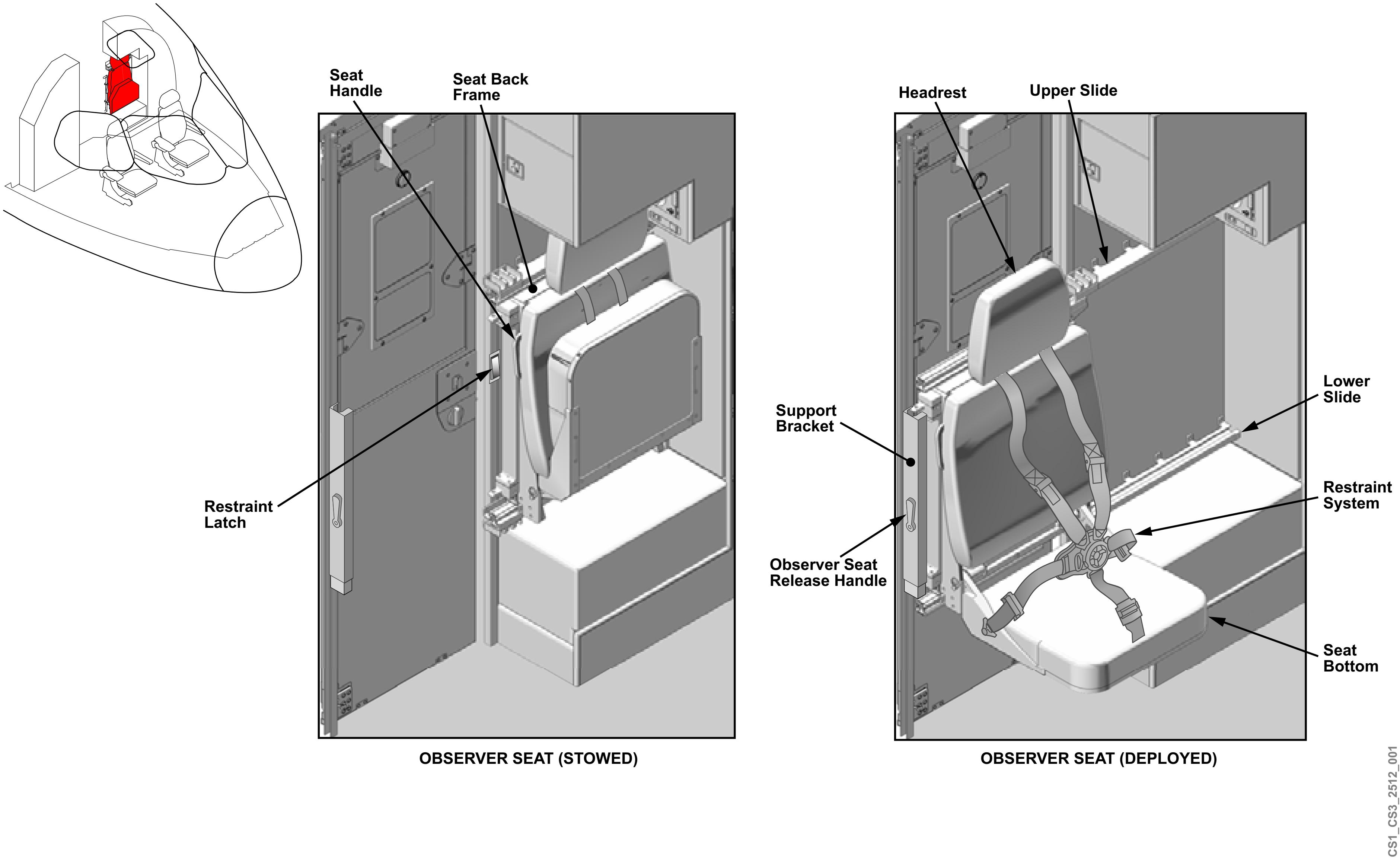


Figure 3: Observer Seat

25-20 PASSENGER CABIN

GENERAL DESCRIPTION

The passenger cabin extends from the flight deck aft bulkhead to the aircraft aft pressure bulkhead. It includes the galleys and utility areas, lavatories, and passenger seating areas. The galley and utility areas are separated from the passenger area by partitions and curtains.

The passenger cabin has overhead storage bins for passenger carry-on items, as well as passenger service units for passenger comfort, emergency oxygen masks, and lighting. The complete compartment is fully insulated with thermal and acoustic insulation. The upper area of the passenger cabin is enclosed with ceiling panels. The left and right sides of the cabin are enclosed with sidewall and dado panels. The passenger floor areas are carpeted, while the utility and lavatory floors are covered with rubberized flooring.

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Figure 4: Passenger Cabin

COMPONENT LOCATION

The following primary components are located in the passenger cabin:

- Passenger seats
- Flight attendant seats
- Overhead storage bins
- Wardrobe, stowage compartments, and partitions
- Finishing panels and linings
- Passenger service units

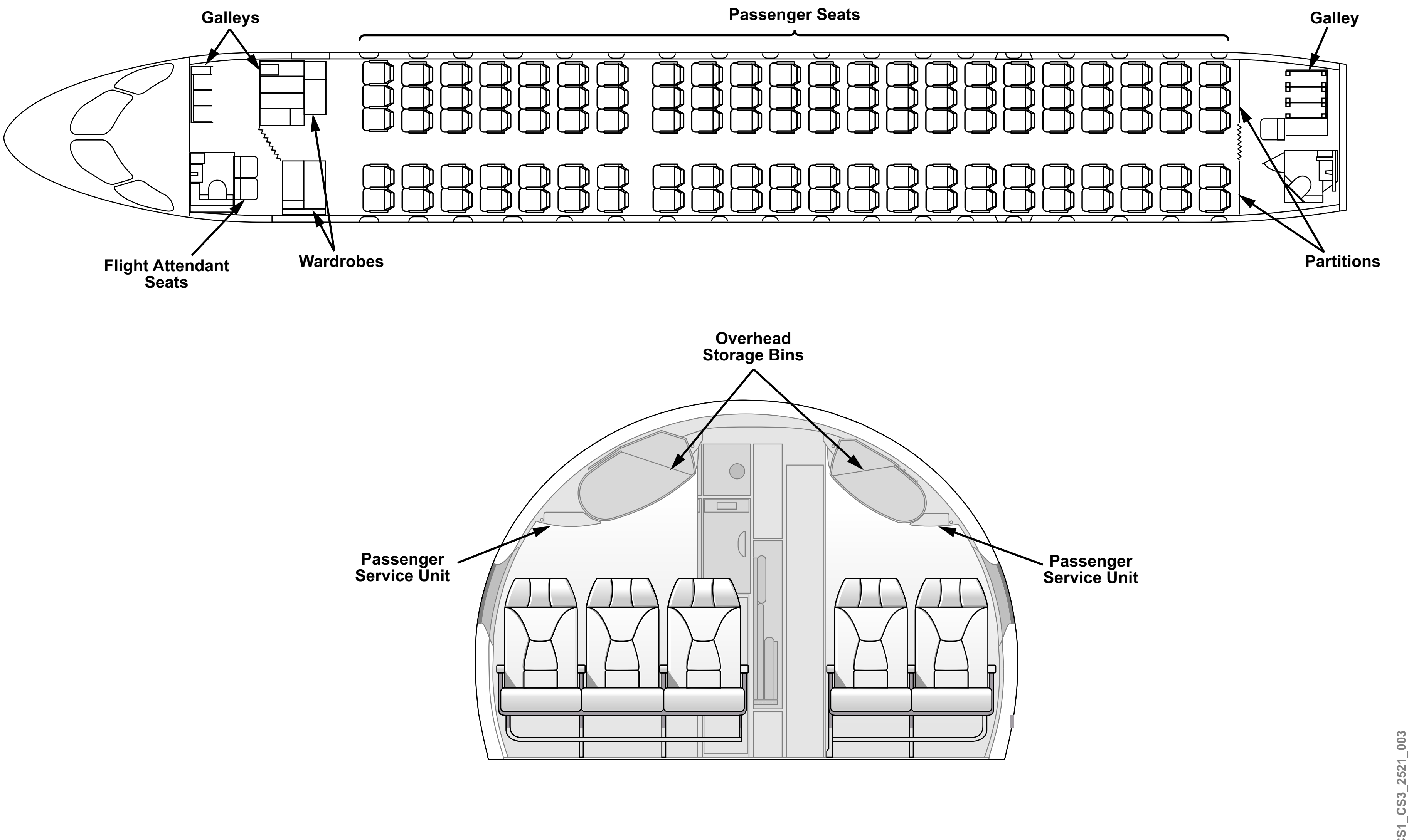


Figure 5: Passenger Cabin Components

COMPONENT INFORMATION

PASSENGER SEATS

The passenger seats are forward-facing, installed three and two abreast with an aisle. Each seat consists of a bottom cushion, seat back, and headrest. The left and right group of seats are attached to two fuselage rails by extruded track fittings and studs with anti-rattle track locks.

The seat leg stud fits into matching notches in the floor tracks. Seat track covers are installed on the seat rails to protect the rails from damage, dirt, or fluid spillage.

The passenger lap belts are secured to the seat frame by bolts.

The passenger seat was designed to accommodate three passengers on the aircraft right side and two on the left side. All passengers have access to a folding food tray, literature pocket, baggage restraint bar, seatbelt restraint, and life vest compartment.

The different types of passenger seats are:

Passenger Seat - Front Row

Standard features include an in-arm food tray, folding aft mounted food tray, literature pocket, cart bumper, life vest pouch, baggage restraint bar, seatbelt restraint, and a recline mechanism.

Passenger Seat - Standard Row

Standard features include a folding aft mounted food tray, literature pocket, cart bumper, life vest pouch, recline button, foldable armrest, baggage restraint bar, and seatbelt restraint.

Passenger Seat - Forward of Overwing Exit Row

Standard features include a literature pocket, cart bumper, life vest pouch, foldable armrest, baggage restraint bar, and seatbelt restraint. This seat does not have a fold-down aft mounted food tray and does not recline.

Passenger Seat - Aft of Overwing Exit Row

Standard features include an in-arm food tray, folding aft mounted food tray, literature pocket, cart bumper, life vest pouch, baggage restraint bar, and seatbelt restraint. This seat also includes a shortened outboard armrest in order to clear the overwing emergency exit door.

Passenger Seat - Last Row

Standard features include a literature pocket, cart bumper, life vest stowage provision, life vest pouch, an in-arm food tray, foldable armrest, baggage restraint bar, standard seatbelt restraint, and recline mechanism. This seat does not have a sliding aft mounted food tray.

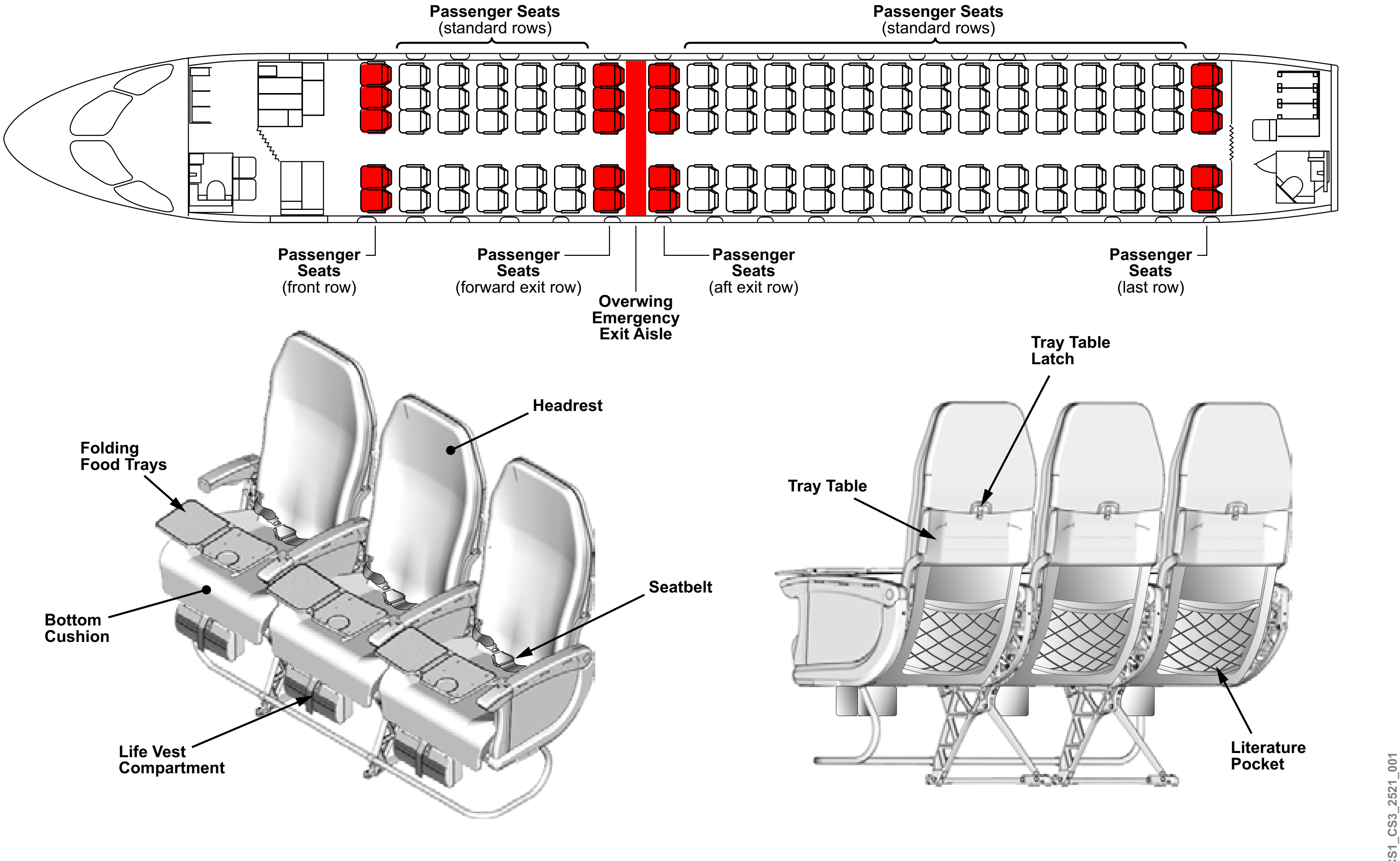


Figure 6: Passenger Seats

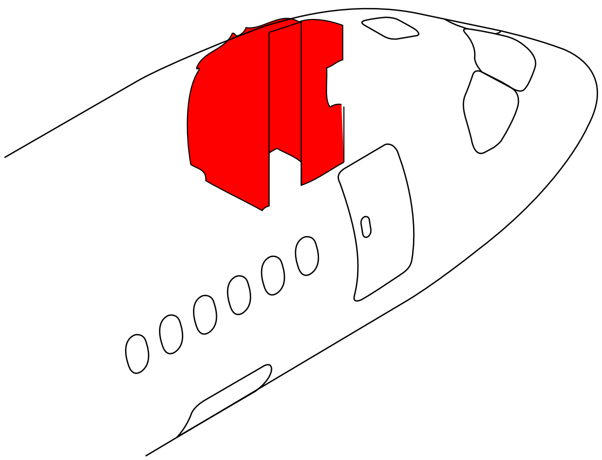
FORWARD FLIGHT ATTENDANT SEATS

The forward flight attendant double seat is attached to the aft bulkhead of the forward lavatory. The double seat assembly consists of a seat back frame, seat bottom, back cushion, headrest cushion, bottom cushion, and seatbelts.

The seat back frame provides a mounting structure for the bottom supports and for the seatbelts. The two sides of the seat back frame assembly are machined from aluminum. The seat bottom structure is provided with a spring-loaded pivot to make the bottom support structure self-stowing.

The seat cushions consist of foam headrests, backs and a single bottom cushion all secured by Velcro straps.

The seatbelt restraint consists of polyester webbing, metal end-fittings, buckles, and inertial reel shroud assemblies.



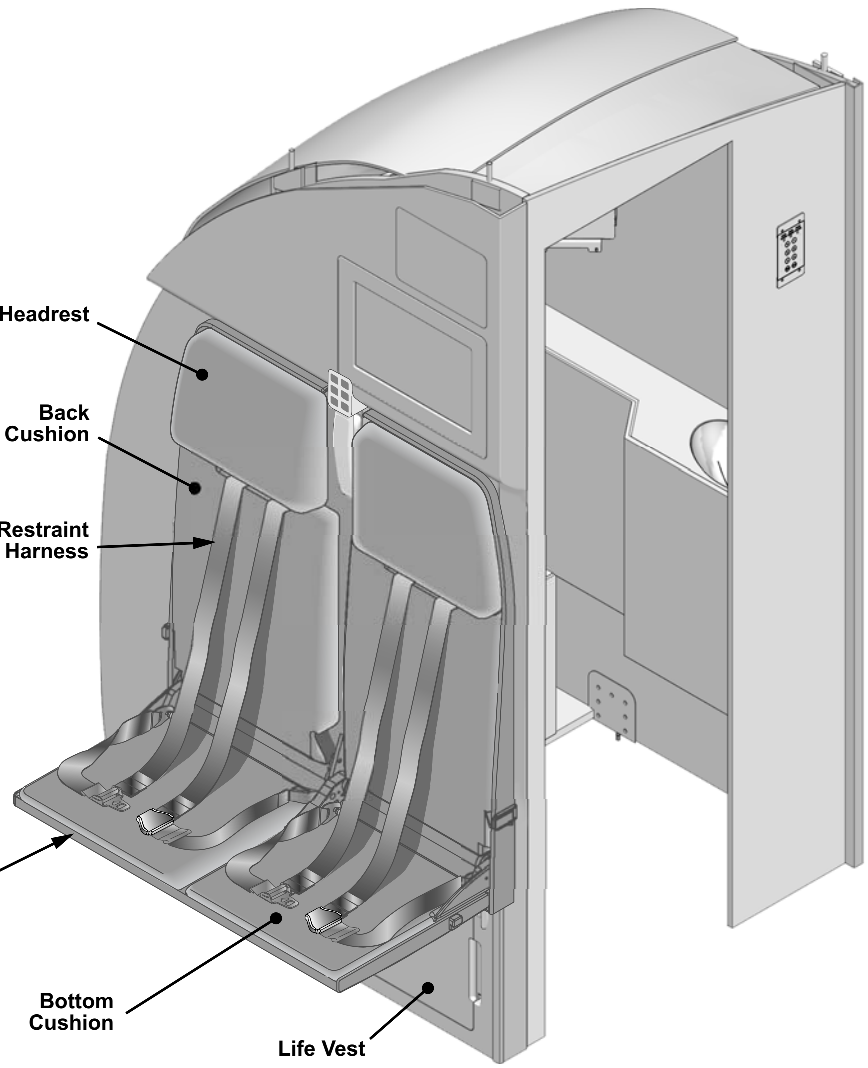


Figure 7: Forward Flight Attendant Seats

AFT FLIGHT ATTENDANT SEAT

The aft flight attendant seat is mounted to the aisle surface of the aft galley between the aft galley and aft lavatory. The seat slides on an upper and lower carriage assembly.

To deploy, the stow latch is lifted and the seat is pulled forward using the seat handle. It is then rotated 90° to engage a locking bar on galley G4.

To stow, the release latch on the side of the seat is pushed down and the seat is returned to the stow position.

The attendant seat is latched in place both in the stowed and in-use positions.

The seat consists of a seat back structure, seat bottom, back cushion, headrest cushion, bottom cushion, and seatbelts.

The bottom support structure is bolted into the side frames that provide a spring-loaded pivot to make the bottom support structure self-stowing.

The restraint harness consists of polyester webbing, metal end-fitting buckles, and an inertia reel assembly.

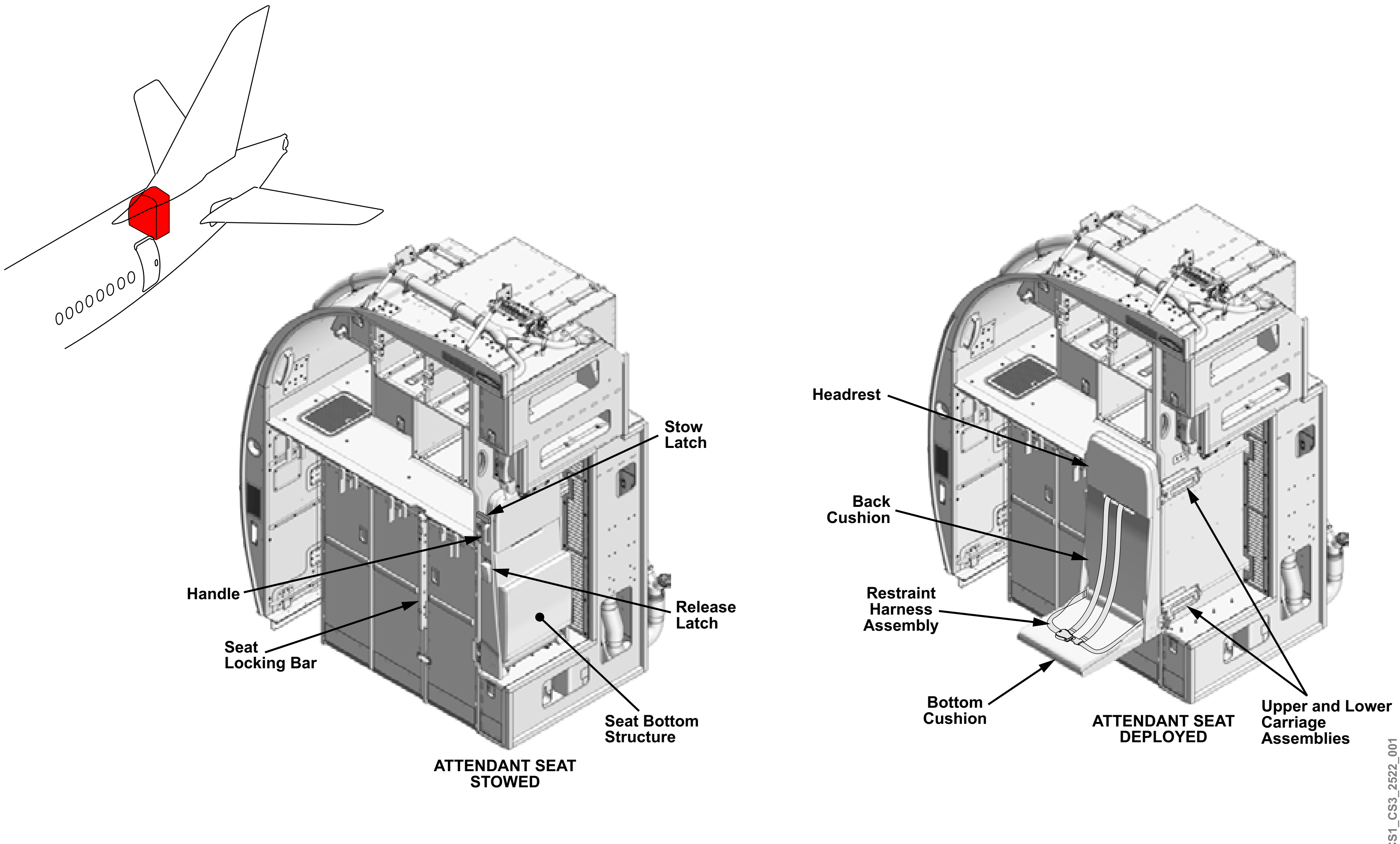


Figure 8: Aft Flight Attendant Seat

OVERHEAD STORAGE BINS

There are seven 254 cm (100 in.) long storage bins in the passenger cabin. Each unit has two buckets for the overhead stowage of passenger carry-on items or other items designated by the airline. Each of the bins is supported by two inboard blade style fittings, a center tie rod that connects to the bin structure between the two buckets, and a 9 G tie rod mounted on top of the bin. The 9 G tie rod reacts to forward inertial loads in case of rapid deceleration. There is an outboard tie rod at each end of the bin near the outboard bin structure.

The bins are opened by pulling downward on the locking latch mechanism. The bin bucket falls slowly under the control of dual-gas struts attached to the bucket and bin assemblies. To close, lift the bucket until the latch engages.

Cabin upwash lighting is located above the overhead bins. Downwash lighting is located at the upper sections of the sidewall panels and passenger service units (PSUs).

The lower outboard cavity of the stowage bin structure houses the PSUs and the oxygen generator housings. All the necessary wiring associated with these components is routed through the bins.

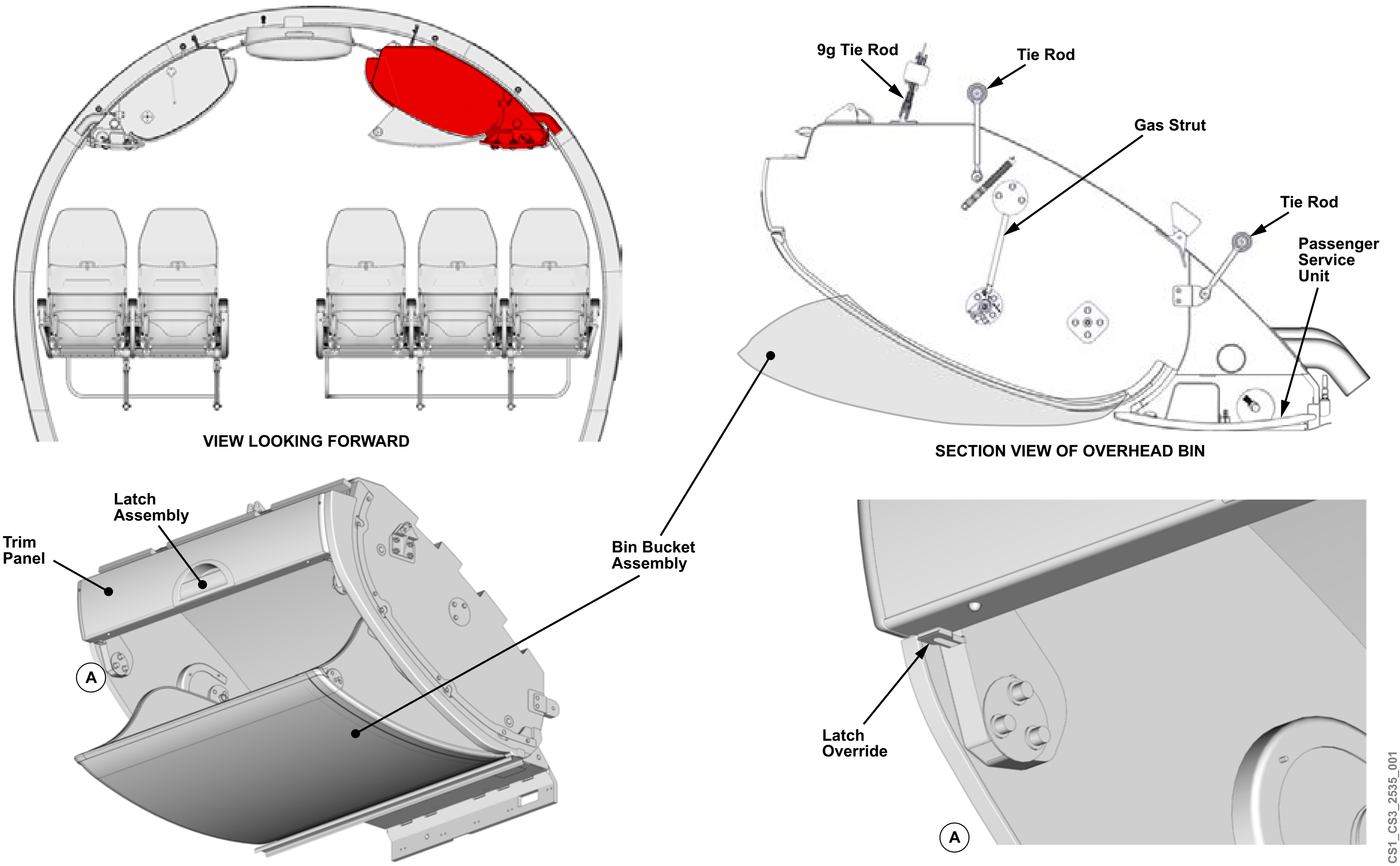


Figure 9: Overhead Storage Bins

PASSENGER SERVICE UNITS

The passenger service units (PSUs) are located in panels above the passenger seats, outboard of the overhead bins. Two additional PSUs are located in the lavatories. The PSU panels are approximately 45.2 cm x 41.4 cm (17.8 in. x 16.3 in.)

The panels contain the following:

- Reading light panel
- Air gasper panel
- Signage panel
- Oxygen dispensing panel
- Attendant call button
- Filler panel
- Speaker

The left side PSUs have two lights, and the right side PSUs have three lights, all with independent switches. The air gasper panels also have separate gaspers: two on the left side, three on the right.

When lowered for maintenance, the PSU swings down and outboard on fore and aft hinges, and hangs on a short lanyard. Disconnecting the lanyard from its hanger allows the PSU to swing fully open, for better access to the oxygen generators, gasper air hoses, electrical devices, and other PSU components.

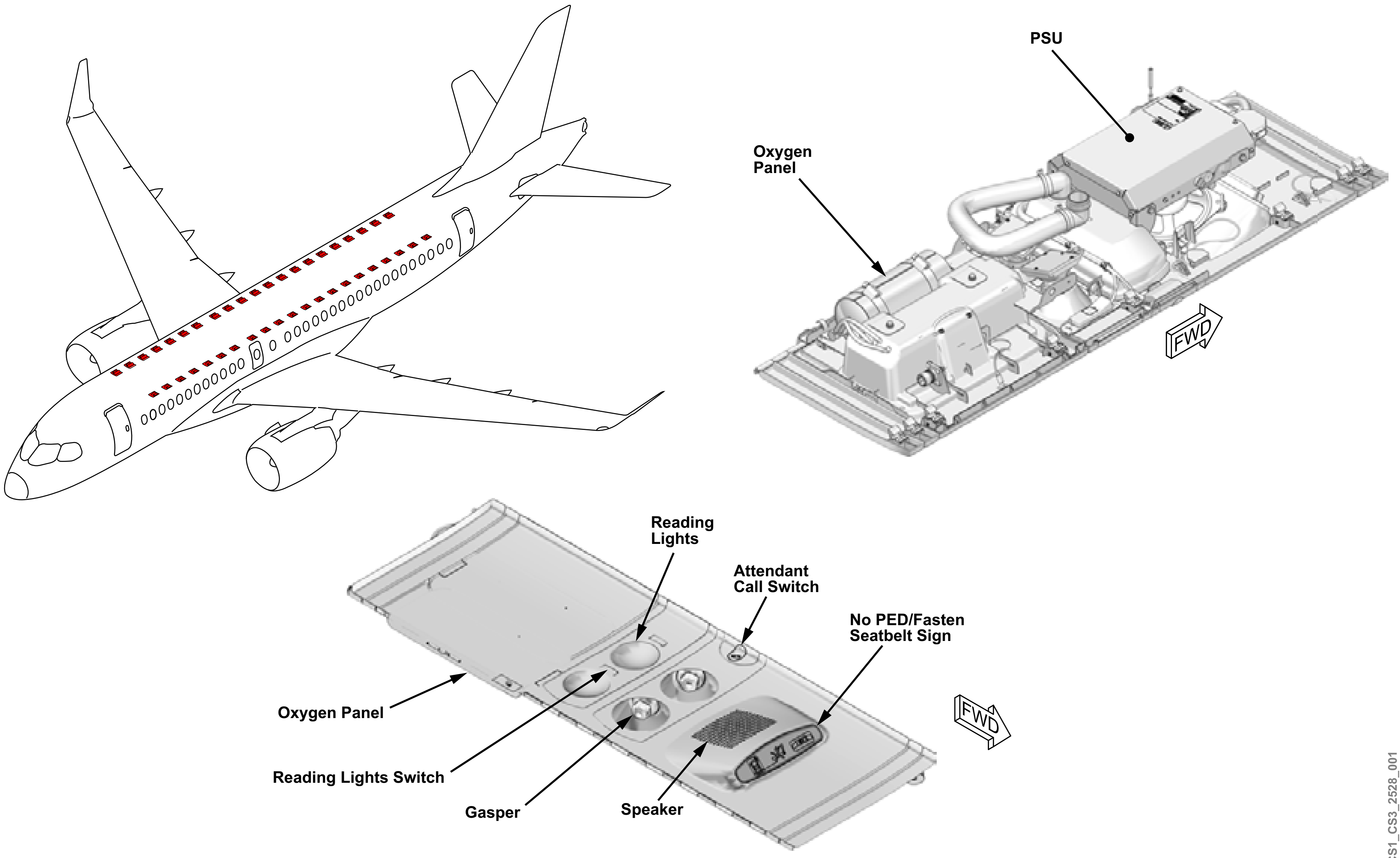


Figure 10: Passenger Service Units

CLOSETS AND STORAGE

The closets and storage bins installed on the aircraft are dependent on the options selected by the airline. Closets are located near the forward and aft passenger doors.

The closets lights are controlled by a microswitch actuated when the closet door is opened. The closet lights are powered from passenger service unit (PSU) controllers.

The storage bins are located in galley area or attached to the windscreens. The storage bins are mainly used for the stowage of emergency equipment.

CLOSET

STORAGE BIN (AFT WINDSCREEN)

STORAGE BIN (AFT GALLEY)

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Figure 11: Closets and Storage

FINISHING PANELS

The entire passenger cabin is finished with ceiling, sidewall, and dado panels. The panels provide coverings for the aircraft structures, ducting, wiring, and insulation installed throughout the passenger cabin.

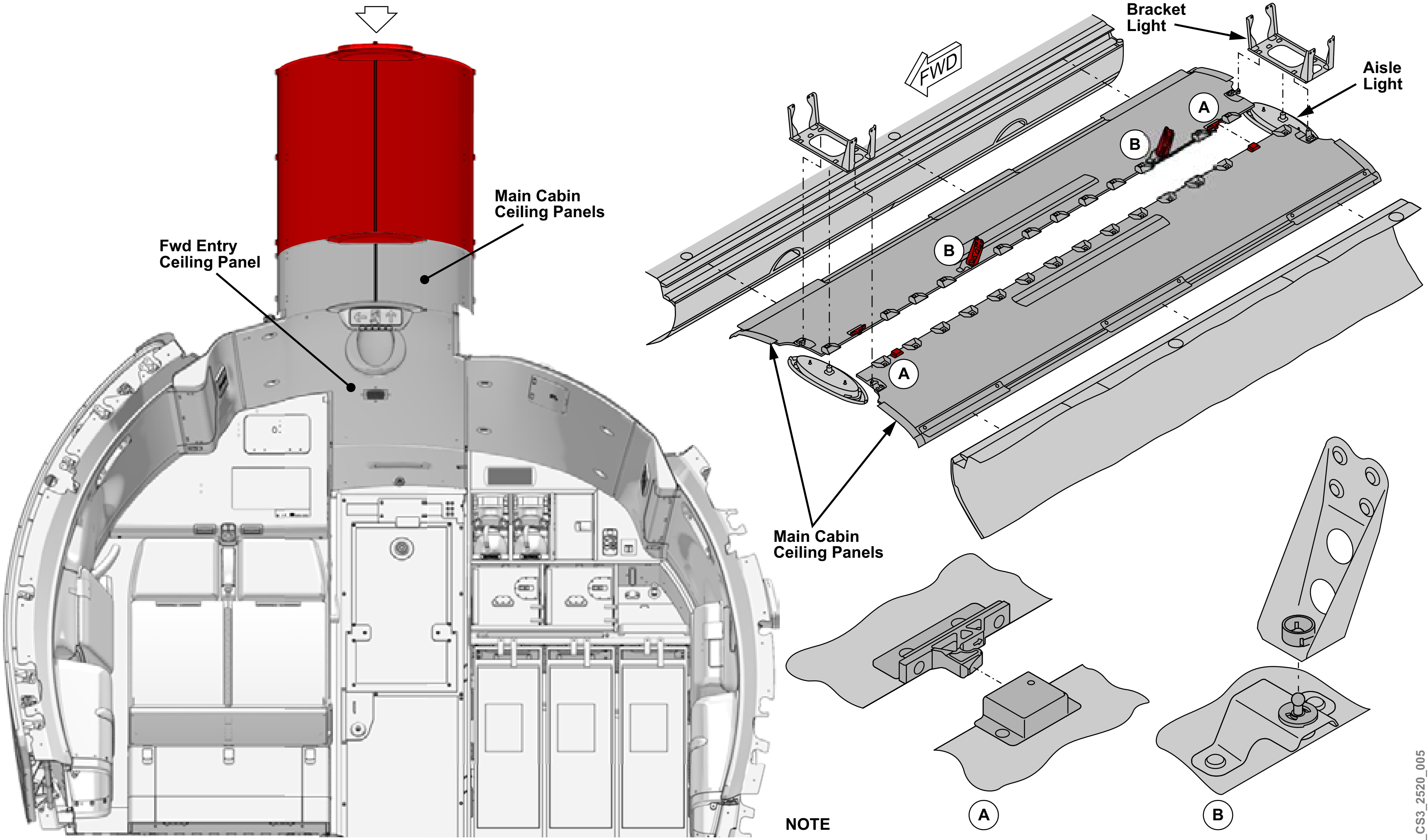
CEILING PANELS

Passenger cabin ceiling panels are divided into forward entry, main cabin, and aft entry panels.

The forward entry ceiling consists of formed panels of different sizes, attached by various press and click fasteners, and clips. The panels have cutouts to accommodate air gaspers, a galley oxygen dispensing unit, signage, and lighting.

The main cabin ceilings are 254 cm (100 in.) long. There is a left and right ceiling panel between each pair of overhead bins. The joints between the ceiling panels align with the separation of the bins. Each ceiling panel slides into an extrusion on the top of the bins. The ceiling panel is retained at each end by press-and-click fasteners, and screwed to a bracket at each end where the aisle accent lights are retained. The left and right ceiling panels are attached together by mechanical latches that engage when pressed together, and disengage when a force is applied to pull them apart. After the ceiling panels are installed, the light lenses for the aisle lights or emergency exit signs are snapped in place.

The aft entry area has two formed ceiling panels, attached by various clips and press-and-click fasteners. The panels have cutouts to accommodate air gaspers, signs, lighting, and an attendant oxygen dispensing unit.



NOTE

Aft entry ceiling panels similar.

Figure 12: Ceiling Panels

SIDEWALL PANELS

The passenger seating area is lined with sidewall panels that cover the inner sides of the cabin between the overhead bins and the cabin floor. The panels consist of an upper sidewall panel and a lower dado panel.

The sidewalls are single frame bay width with a single window reveal, or dual frame bay in width with two window reveals. A typical sidewall panel spans two window bays. At the flexible zones the panels can be as long as 157.5 cm (62 in.) and take a non-standard shape since they are installed in the non-constant cross section of the aircraft (as it tapers inward going forward and aft of the core section).

The window reveal attaches to the sidewall. The window reveal has an outer section which can be removed so the transparent glareshield can be removed and replaced if it becomes scratched. Also attached to this outer section is a rubber seal which bridges the gap between the window reveal and the window.

The sidewall slides into, and is retained by, a U-shaped channel attached to the bin. On the lower side, push-and-click fasteners attach the sidewall to the dado panel. The sidewall installation is designed so the panel can be removed without removing the passenger seats.

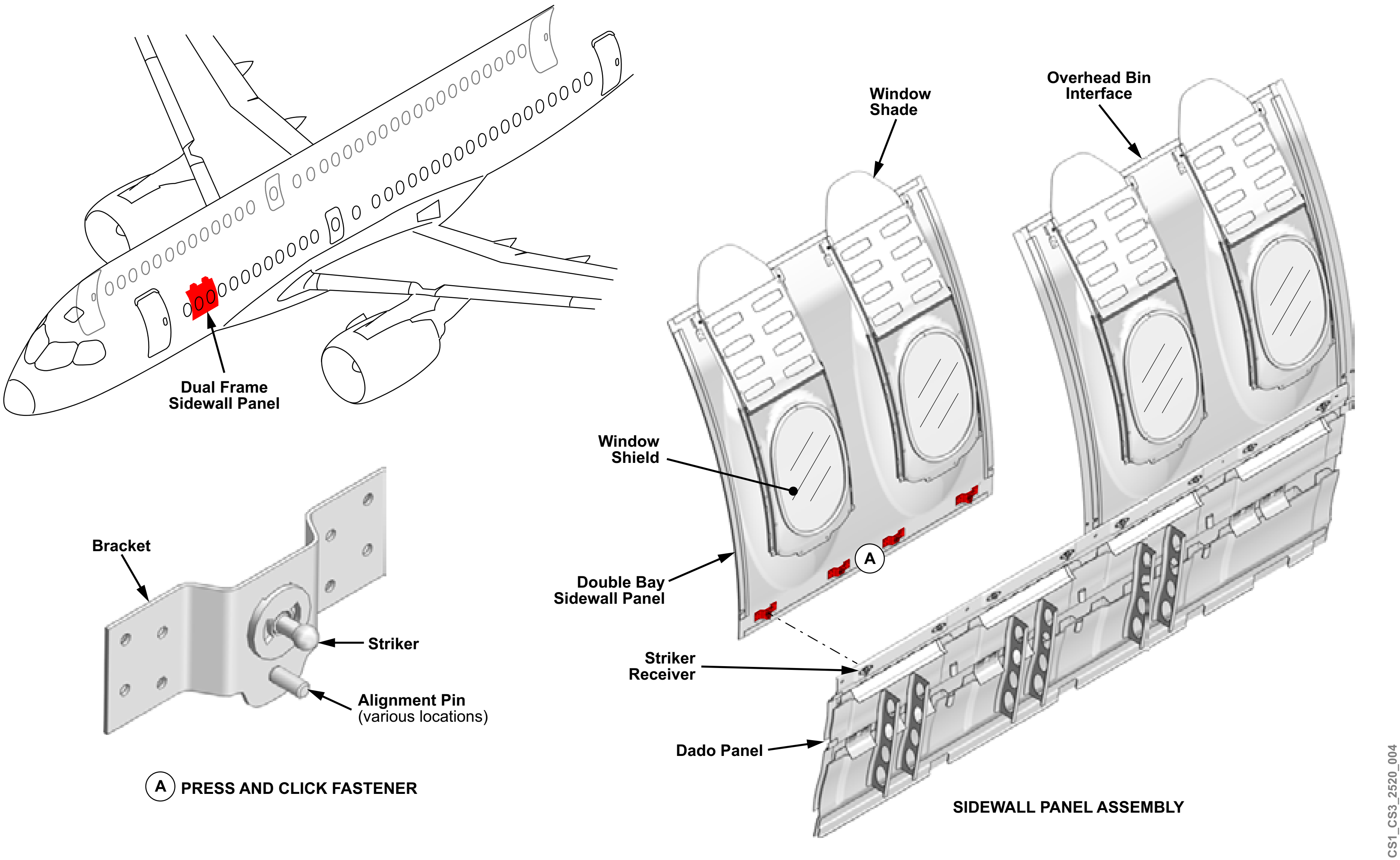


Figure 13: Sidewalls

DADO PANELS

The dado panels are pressure relief panels from the floor to sidewall junction. They are located between FR 26 and FR 41, and FR 45 and FR 70, on the left and right sides of the passenger cabin.

To allow pressure equalization, decompression flaps are installed on the outboard side of the dado panels. To minimize structural damage, the decompression flaps open at 0.5 psid in the event of rapid decompression of the aircraft.

The length and locations of the decompression flaps vary throughout the cabin by dado assembly, based on the estimated decompression rate and aircraft pressurization system capabilities.

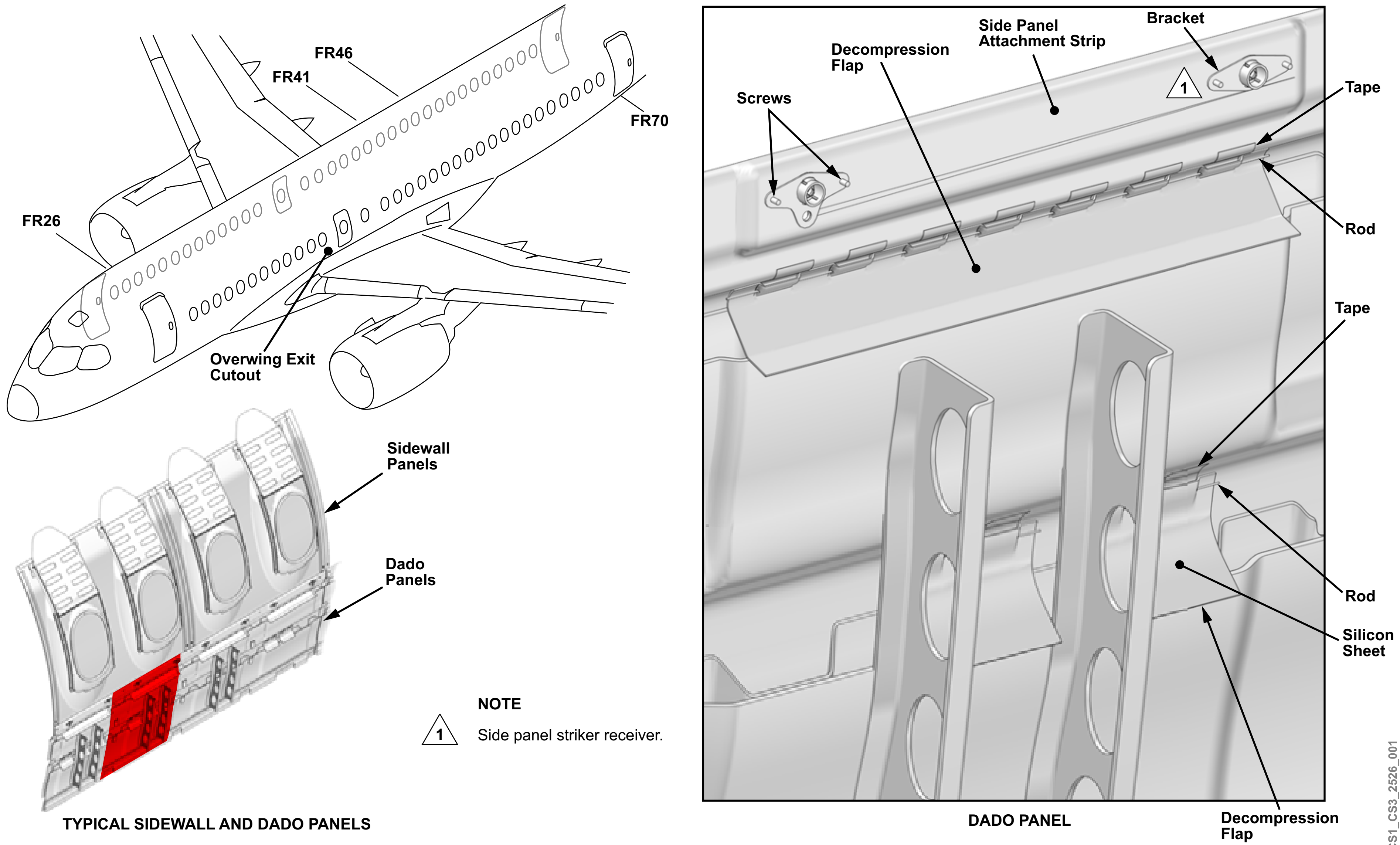


Figure 14: Dado Panels

PRACTICAL ASPECTS

PASSENGER SEAT INSTALLATION AND REMOVAL

The passenger seats are mounted on the floor track by two rear track fittings (inboard and outboard) and two front studs. The forward fitting contains an antirattle nut and the aft has a locking plunger. There are eight studs, two antirattle nuts, and two plungers per seat. The plungers and antirattle nuts secure the seat in the track and reduce noise under turbulent flying conditions, as well as during takeoff and landing.

The seat studs and track fittings are offset by 1 1/2 times the stud diameter, to the locking plungers and antirattle nuts. Once the front studs and rear studs are aligned and lowered into the seat track, the seat is moved slightly to align the locking plungers and antirattle nuts. Lock engagement is performed by turning the locking plungers and antirattle nuts with a hex wrench. Plungers and antirattle nuts are torqued as specified in the Aircraft Maintenance Publication (AMP). Seat removal is opposite of installation.

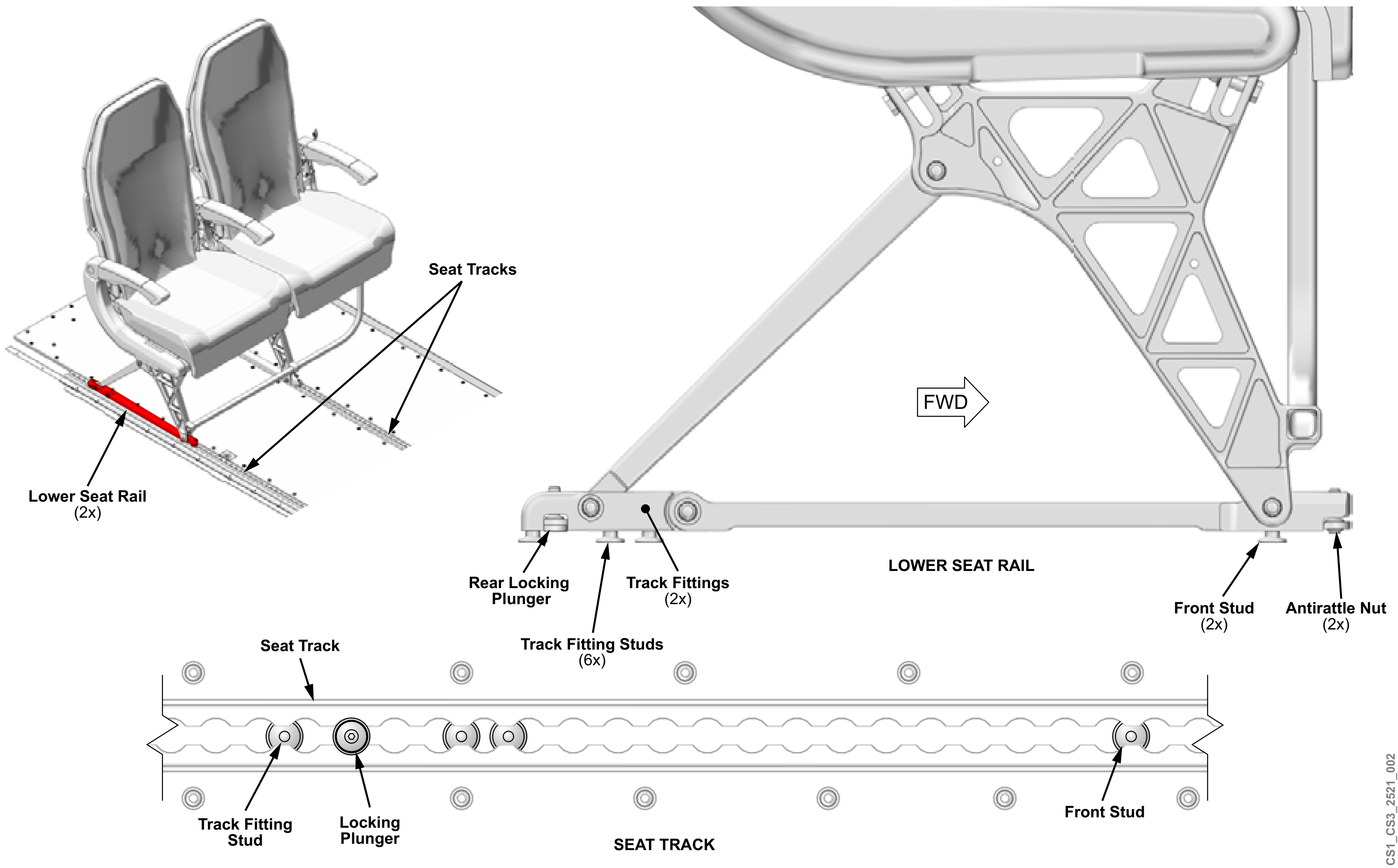


Figure 15: Passenger Seat Installation and Removal

25-30 GALLEYS

GENERAL DESCRIPTION

Galleys are installed in the forward and aft passenger cabin areas to provide in-flight food and beverage services to the passengers and crew. Galleys located forward of the two forward exits, or aft of the two rear exits, are installed with specific floor fittings and upper attachment link arms. Galleys, located between the left or right main exits, are attached using the cabin seat tracks and upper attachment links, as required.

COMPONENT LOCATION

GALLEYS

In the standard configuration, the aircraft is equipped with three galleys:

- Galley G1, located forward of forward right service door
- Galley G2, located aft of the forward service door
- Galley G4, located aft of the rear right service door

Optional items can be added to provide numerous configurations based on customer needs, including additional galleys, storage compartments, lavatories, and windscreens.

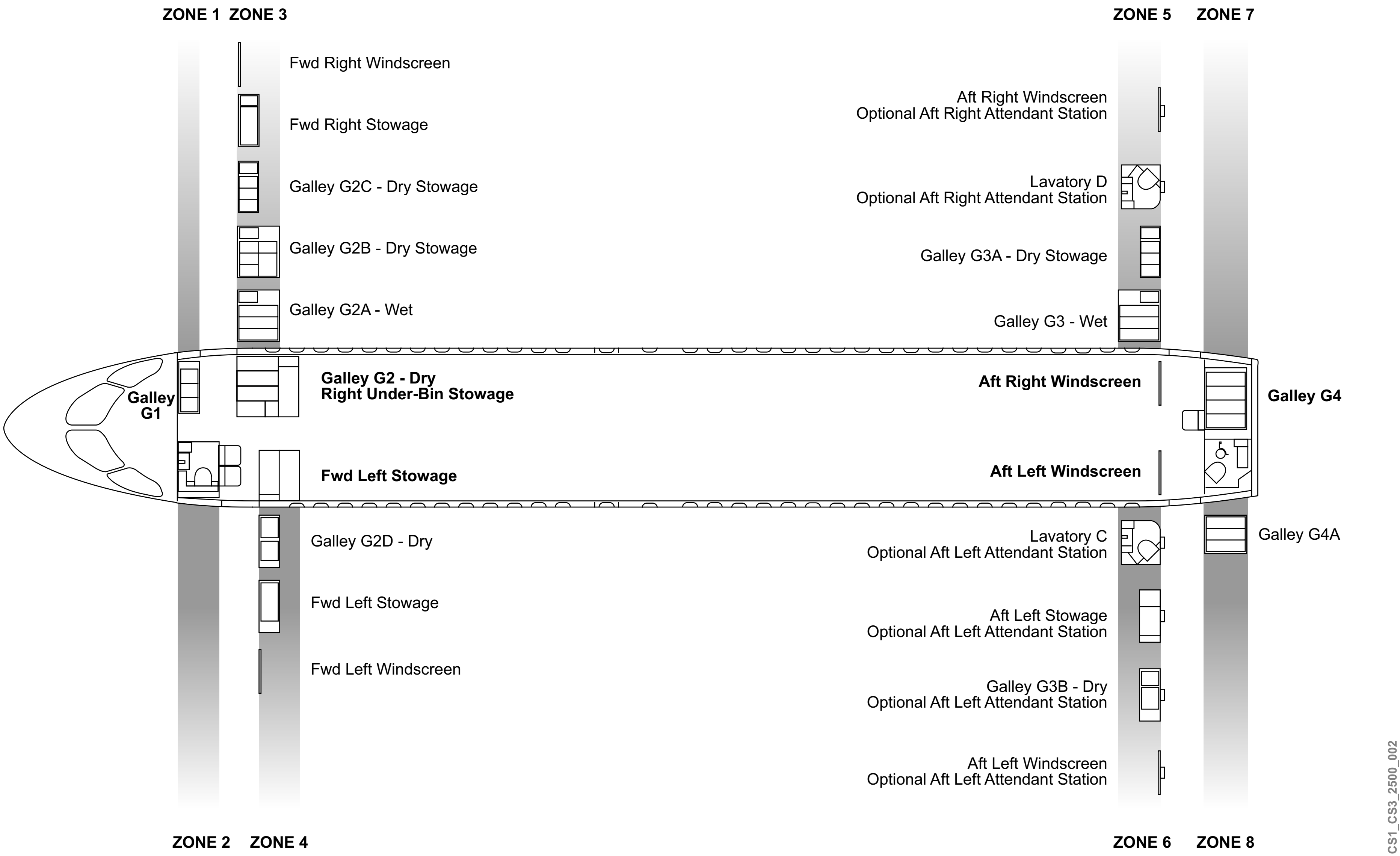


Figure 16: Optional Configurations

COMPONENT INFORMATION

GALLEY G1

Galley G1 is a wet galley, with the following features and options:

- Provisions for three half-size trolleys and two stowage units
- One coffee maker
- Space and power for up to three additional appliances (second coffee maker, water boiler, half-size trash compactor)
- Potable water spigot
- One liter sink and drain
- Provision for potable water filter
- Door area heater and outlet
- Electrical power outlet for ground use only (115/200 VAC 400 Hz/15A)

GALLEY G2

Galley G2 is a dry galley with the following features and options:

- Provisions for four half-size trolleys or two full-size trolleys
- Provisions for two stowage units
- Space and power for up to four appliances (32-meal oven, full-size trash compactor)
- Electrical harness for installed appliances
- One fixed, fireproof waste container
- Wardrobe with the following features:
  - Coat rod for passenger coats or garment bags
  - Fixed, folding, adjustable shelves
  - Hinged door
  - Compartment lighting
  - Emergency equipment closet
  - Provisions for three passenger literature pockets on the aft face
  - Provisions for baby bassinet fittings on the aft face
  - Partial provisions for magazine rack on aft face

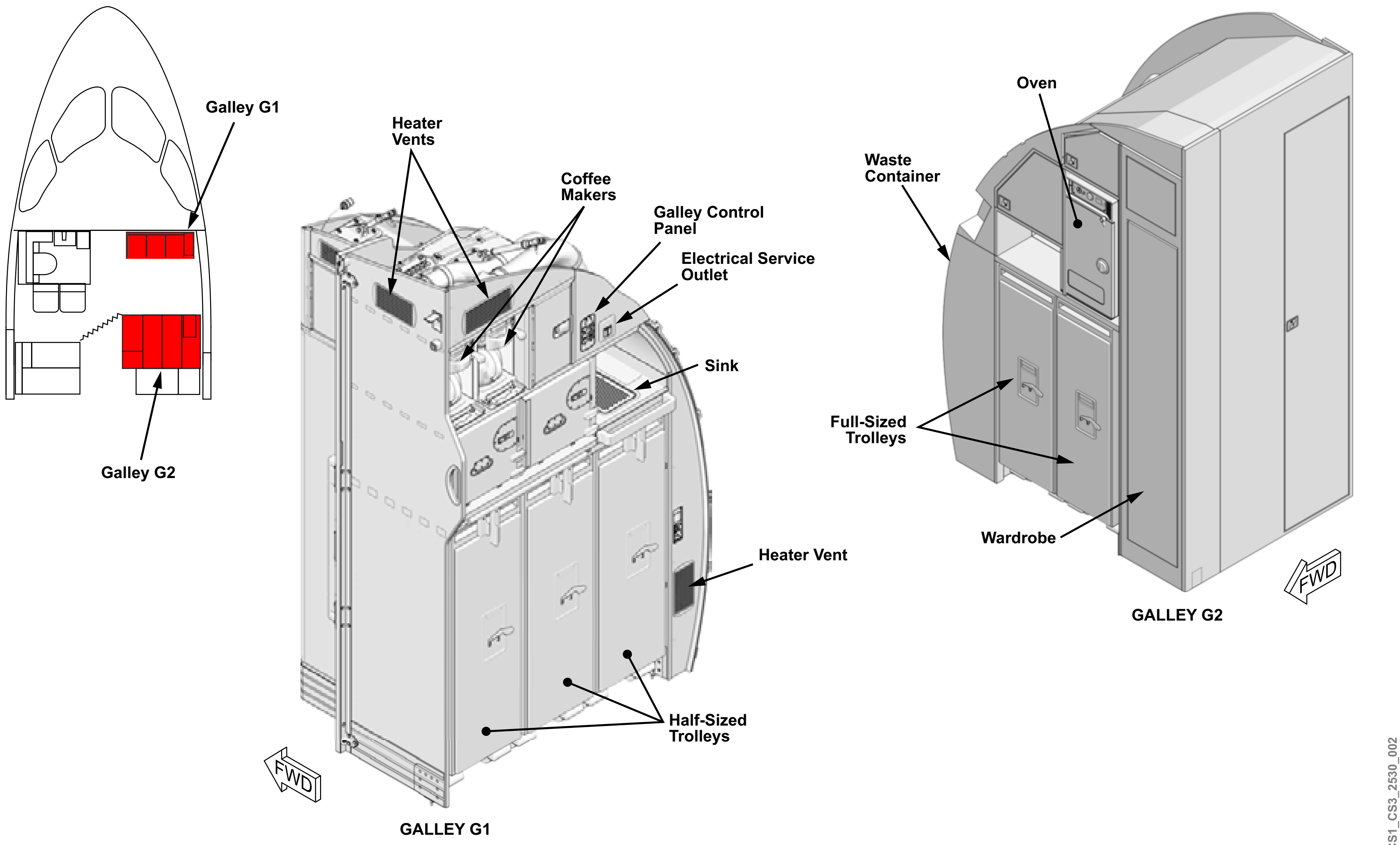


Figure 17: Galley G1 and Galley G2

GALLEY G4

Galley G4 is a wet galley with the following features and options:

- Complete provisions for eight half-size trolleys or four full-size trolleys
- Provisions for nine storage units
- Space and power for up to five appliances:
  - Coffee maker
  - Water boiler
  - Oven (32-meal)
  - Steam oven (32-meal)
  - Full-size trash compactor or trolley chiller unit
- Electrical harness for installed appliances
- Potable water supply shutoff valve
- Flush drain and gray water plumbing
- Electrical power outlet for ground use only
- Pullout attendant seat with autofold base and headrest
- Aft attendant lighting and control panel
- Aft cabin passenger address (PA) handset and cradle
- Access panel for aft pressure bulkhead access

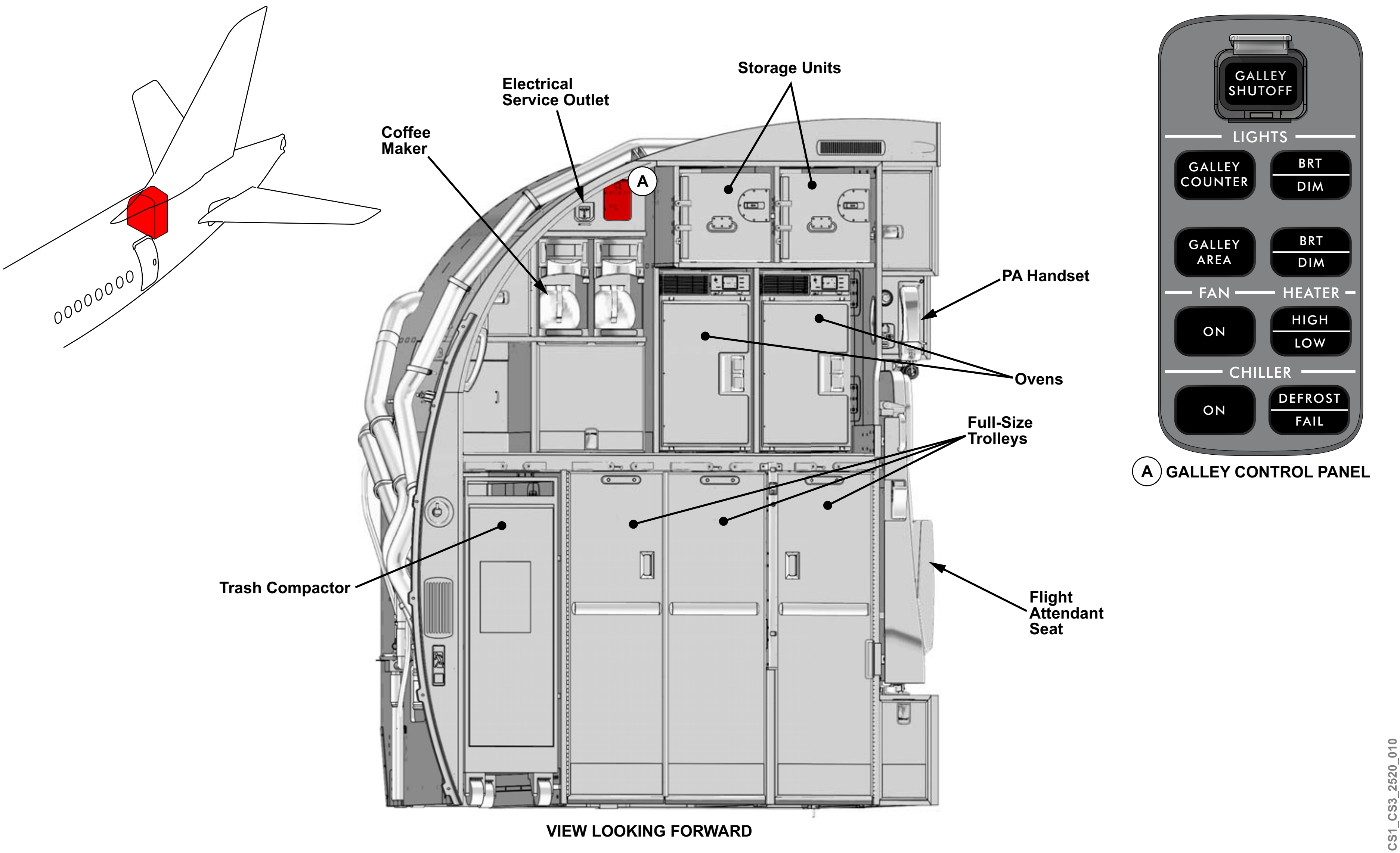


Figure 18: Galley G4

DETAILED COMPONENT INFORMATION

GALLEY ATTACHMENT FITTINGS

The galleys are installed with floor fittings and upper attachment link arms. Galleys, located between the left or right main exits, are attached using the cabin seat tracks and upper attachment links, as required.

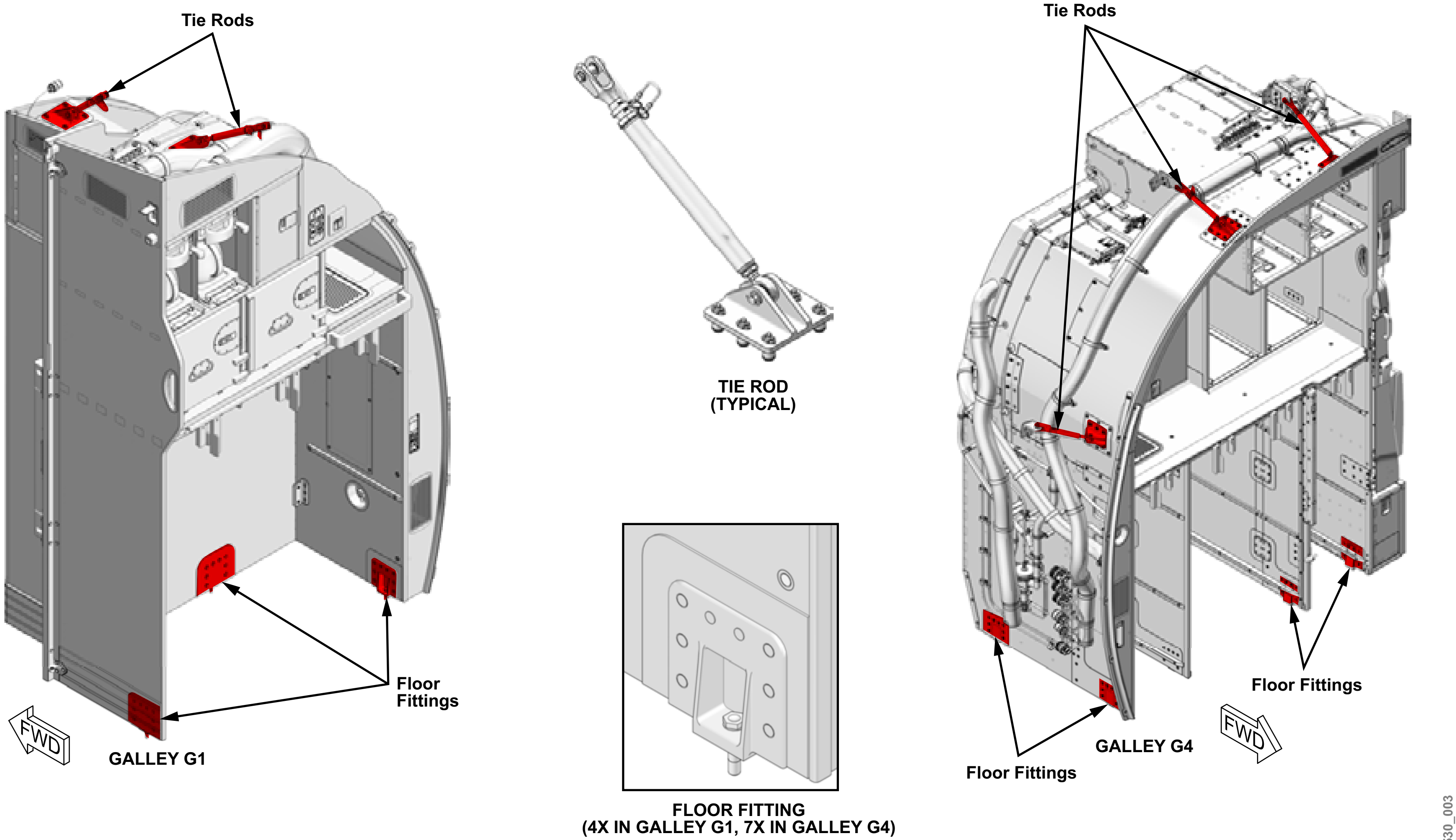


Figure 19: Galley Attachment Fittings

25-40 LAVATORIES

GENERAL DESCRIPTION

Lavatories located in the passenger cabin area provide passenger comfort and convenience.

There are two lavatories for the baseline configuration on C Series aircraft. Lavatory A is located in the left side of the passenger cabin, forward of the forward passenger door. Lavatory E is located on the aft left side of the passenger cabin, beside galley G4, and aft of the rear passenger door.

Other lavatory configurations can be provided, based on operator requirements:

- Lavatory C can be installed in the rear left side of the aircraft, forward of the aft passenger door
- Lavatory D can be installed in the rear right side of the aircraft, forward of the aft service door

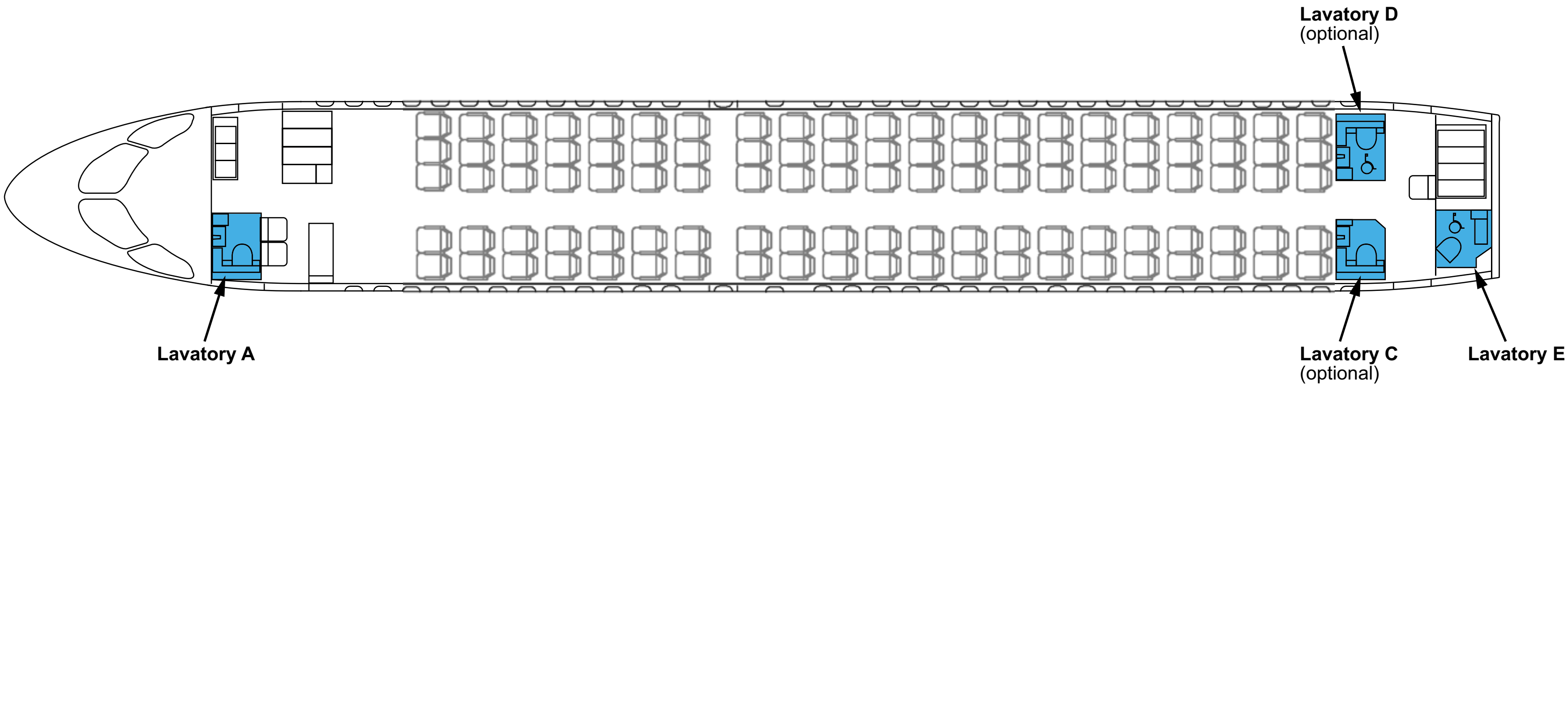


Figure 20: Lavatories

COMPONENT INFORMATION

Each lavatory is equipped with the following:

- Tamper-proof smoke detector
- Tamper-proof oxygen panel
- Sink and faucet
- Mirror
- Water shutoff valve
- Electrically operated flush toilet
- Vanity items
- Trash container

The side unit door above the sink provides access to the oxygen dispenser, as well as compartments for hand towels and tissues.

Lavatory communication equipment consists of:

- Ordinance signs
- A ceiling mounted passenger address speaker
- A flight attendant call button used for summoning assistance

The return-to-seat ordinance sign illuminates in the lavatory when the passenger cabin fasten seatbelt signs are activated.

The lavatory floor is designed to contain spillage. Ventilation is provided for each lavatory and air exhausts through overboard vents.

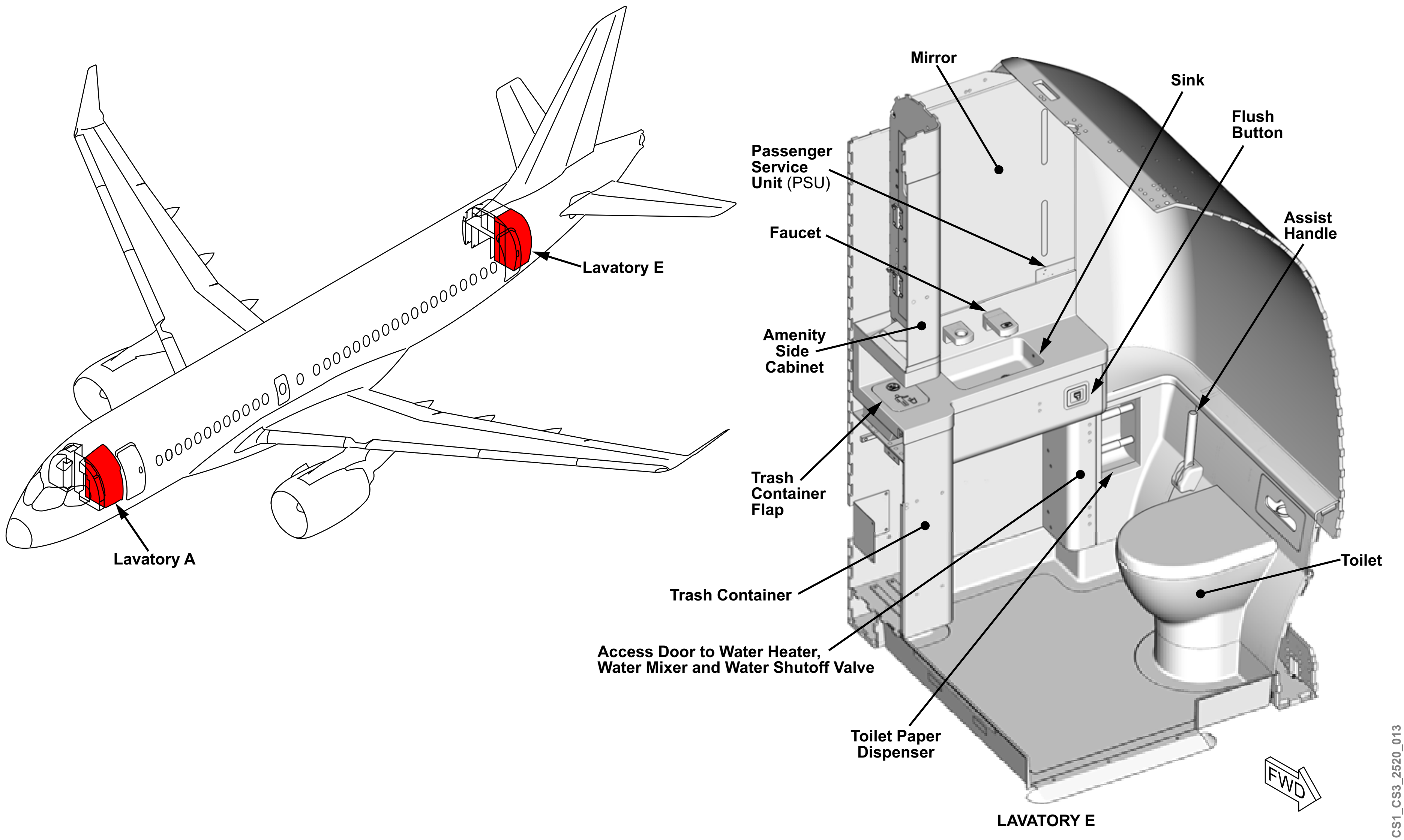


Figure 21: Lavatory Components and Features

25-60 EMERGENCY EQUIPMENT

GENERAL DESCRIPTION

Emergency equipment is installed in the flight deck and passenger cabin for use by the flight and cabin crew, in the event of emergency situations. The equipment is located for easy access and is kept fully prepared for immediate use.

COMPONENT INFORMATION

FLIGHT DECK EMERGENCY EQUIPMENT

Emergency equipment is provided in the flight deck. One rechargeable flashlight is installed forward of the circuit breaker panels, behind each pilot seat. A crash ax is mounted behind the pilot seat. A portable fire extinguisher is mounted behind the copilot seat. A protective breathing equipment (PBE) unit for the observer is located behind the pilot seat.

Crew life vests are installed beneath the outer thigh support of each pilot seat. An escape rope is installed in the upper headliner. The flight deck overhead emergency exit must be opened to access the escape rope.

The escape rope is used by the crew to exit the aircraft through the overhead emergency exit during an emergency situation.

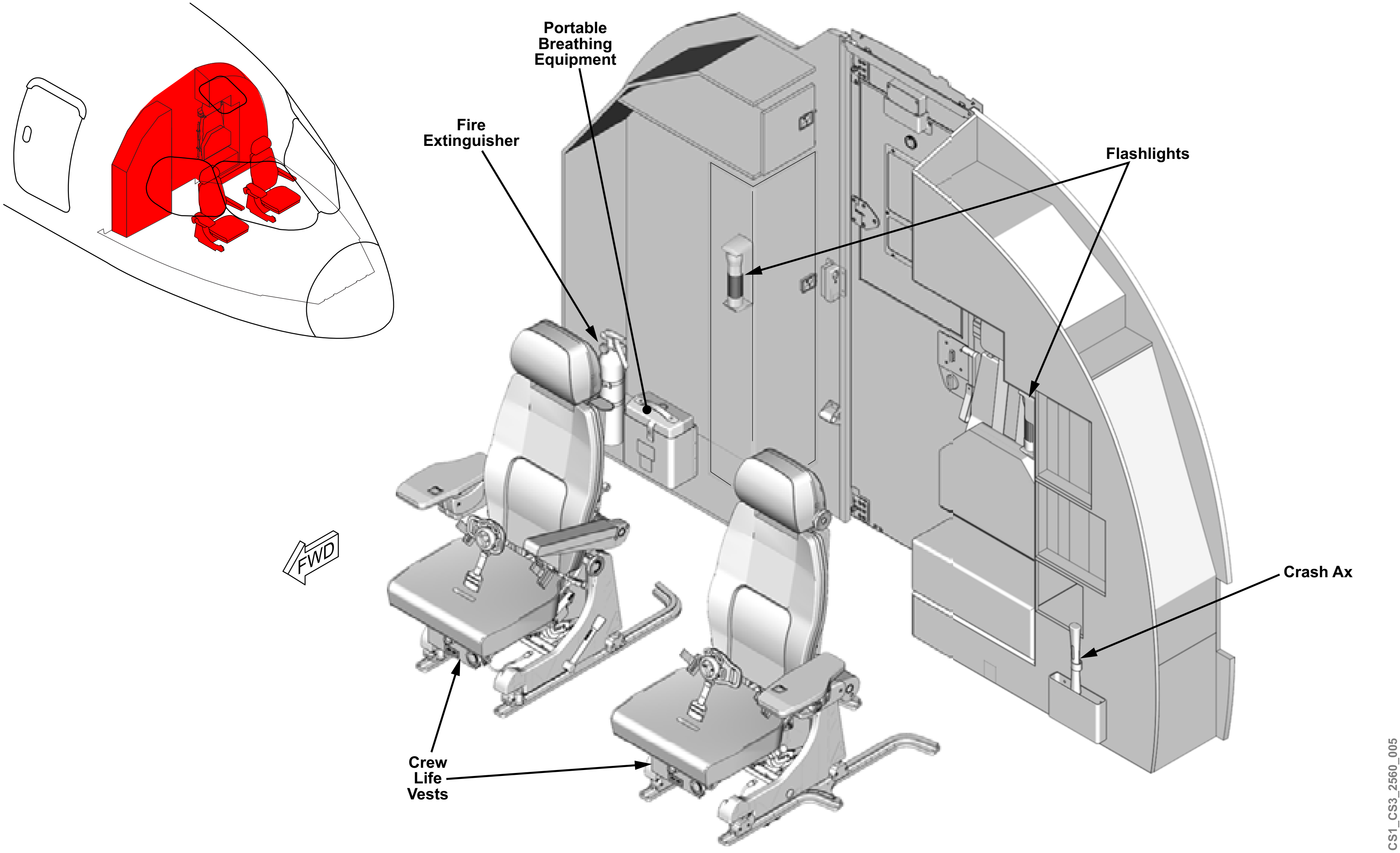


Figure 22: Flight Deck Emergency Equipment

PASSENGER CABIN EMERGENCY EQUIPMENT

First aid kits are located in the forward galley G2 and at the aft flight attendant station.

Two flashlights are installed on the forward flight attendant station facing the main passenger entrance. Another flashlight is located at the aft flight attendant station.

Portable fire extinguishers are located in the forward galley G2 and at the aft flight attendant station.

Portable oxygen bottles, oxygen mask, and tote bag are located in the forward galley G2 and on the aft left side windscreen.

Crew life vests are stowed at the forward flight attendant station. Another life vest is installed at the aft flight attendant station. Protective breathing equipment (PBE) units are installed in the forward galley G2 and at the aft flight attendant station.

Megaphones are located in the forward galley G2 and at the aft flight attendant station. One portable emergency locator is installed in the passenger cabin. Location depends on the customer.

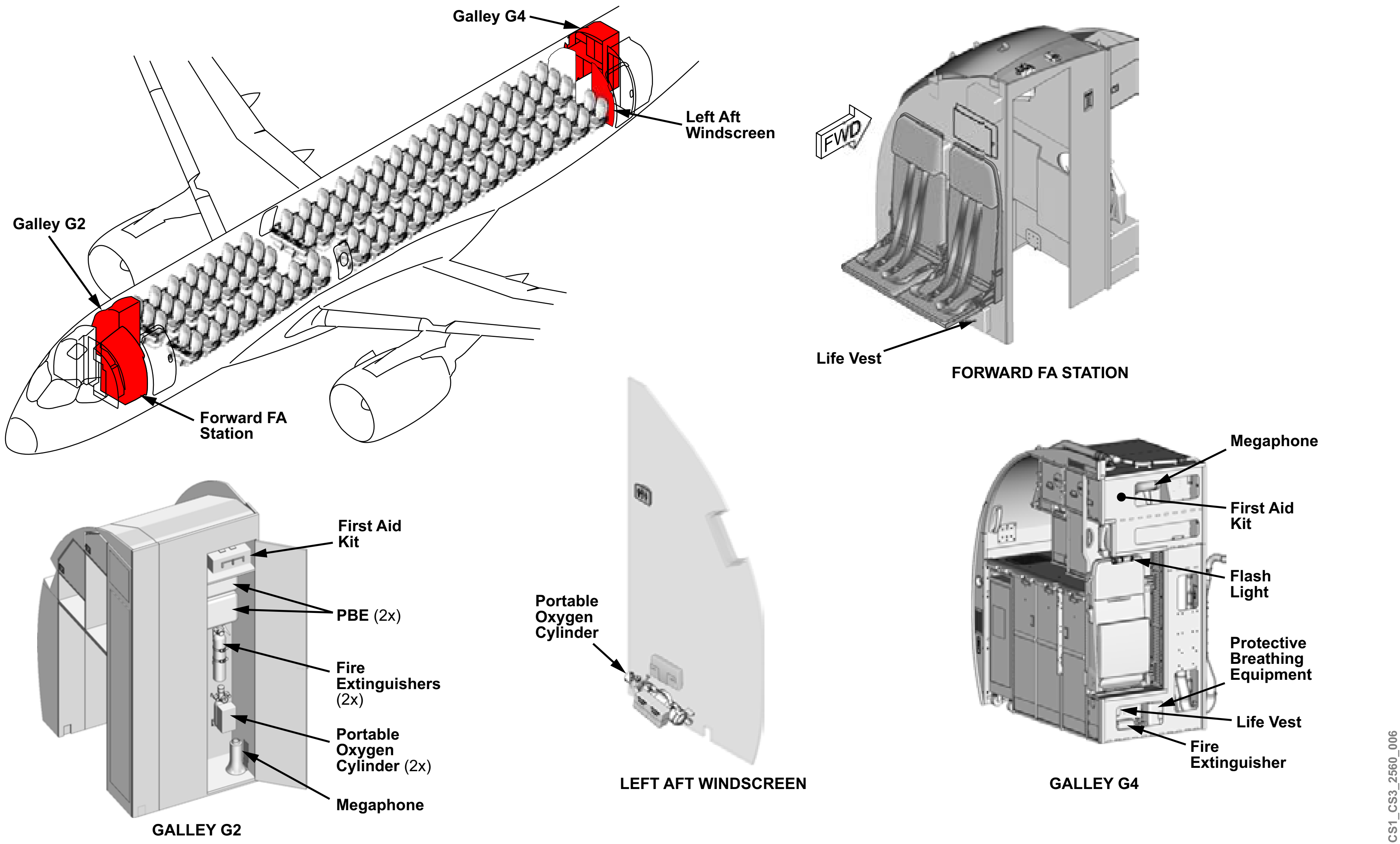


Figure 23: Passenger Cabin Emergency Equipment

25-62 EMERGENCY LOCATOR TRANSMITTER

DETAILED COMPONENT INFORMATION

TRANSMITTER UNIT

The transmitter unit has a single axis G-switch for detection of a crash, which automatically activates and transmits emergency signals. The unit is capable of transmitting the following short messages:

- 24 bit aircraft address number
- Aircraft operator designation number and serial number
- Aviation user protocol
- Serial number protocol
- Serialized aviation user protocol

The ELT is powered by two cell lithium battery pack connected to the ELT transmitter unit by a quick-release connector. The battery has a service life of 12 years. Battery replacement can be done without removing the ELT from the aircraft.

The ELT front panel has a locking three position OFF/ARM/TX switch. The switch is always set to ARM when it is installed in the aircraft, and selected OFF if removed for service.

An indicator light on the front of the unit is on when the unit is transmitting.

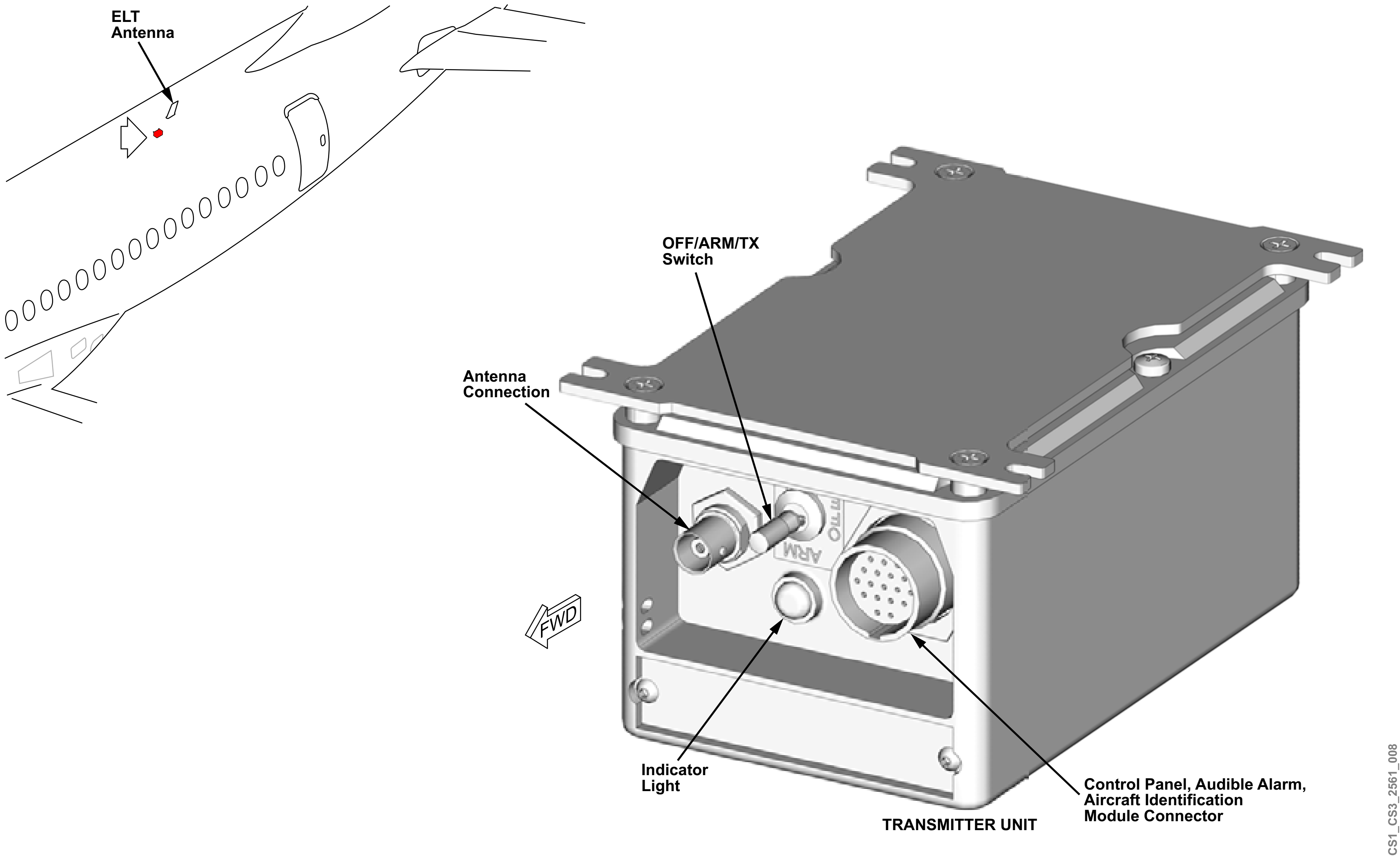


Figure 24: Transmitter Unit

AIRCRAFT IDENTIFICATION MODULE

The aircraft identification module (AIM) is used to provide the transmitter unit with aircraft specific data. The AIM is attached to the aircraft structure and remains with the aircraft if the ELT transmitter unit is removed or replaced. The AIM is not connected to the transmitter unit during normal aircraft operation.

The AIM is connected to the transmitter unit momentarily when the transmitter unit is first installed and any time the transmitter unit is replaced. After the initial transfer of aircraft data from the AIM to the transmitter unit, the AIM is unplugged from the transmitter unit and secured to the aircraft near the transmitter unit.

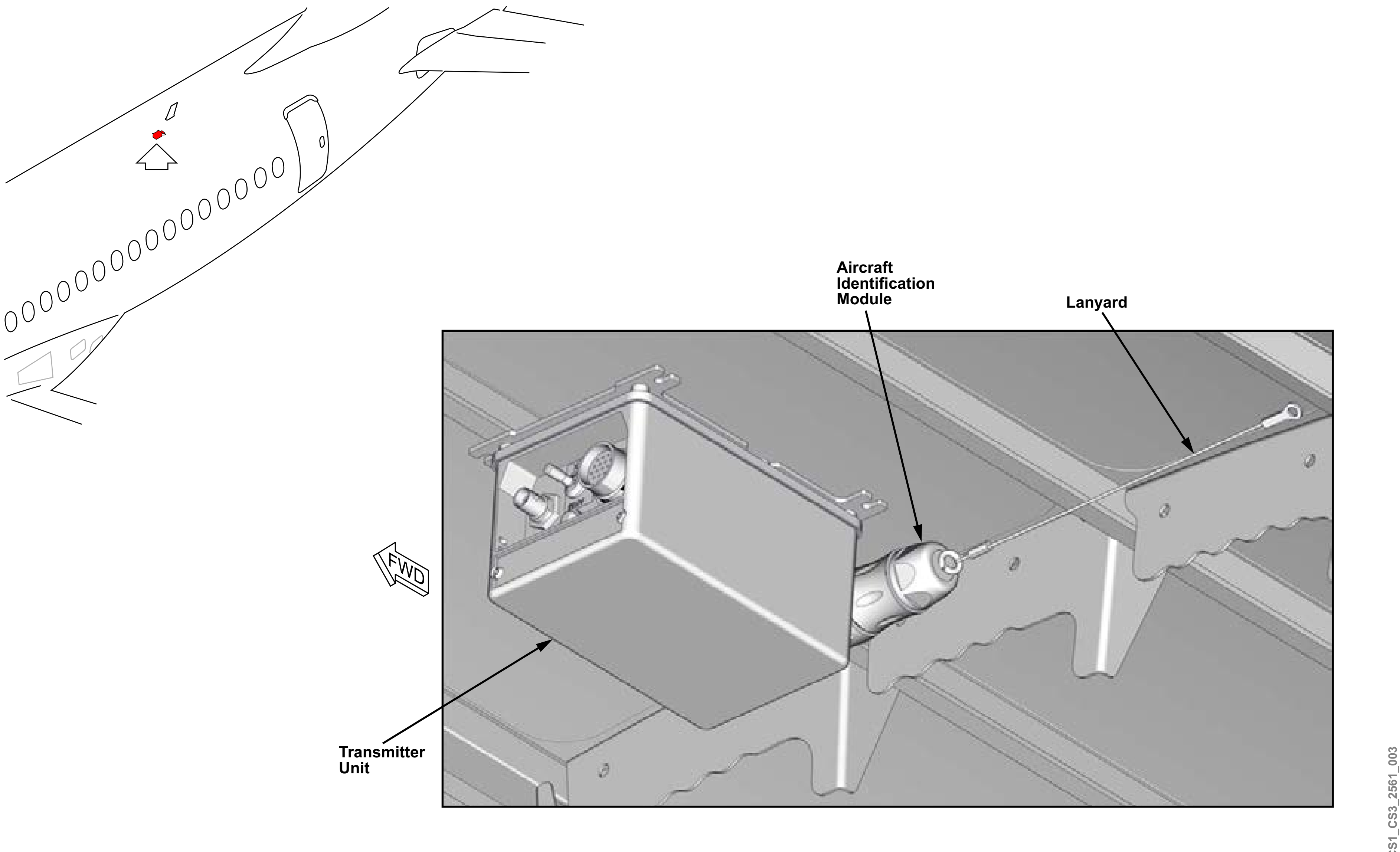


Figure 25: Aircraft Identification Module

CONTROLS AND INDICATIONS

ELT PANEL

The ELT panel has a three-position pull-to-turn rotary knob labeled TEST/RESET, ARM, and ON. The TEST LED illuminates when the knob is selected to TEST/RESET.

Figure 26: ELT Panel

DETAILED DESCRIPTION

INTERFACE

The ELT panel is hardwired to the ELT transmitter unit. The reset interface is used to determine the operating mode of the ELT. The transmitter unit provides a discrete output to the TEST light on the ELT panel indicating it is transmitting. The ELT rotary switch is monitored for faults and failure by the data concentrator unit module cabinet (DMC). The DMC provides a service message for a switch fault.

The DMC also receives a discrete signal from the ELT transmitter unit indicating it is transmitting. An ELT ON caution message is displayed when the unit is transmitting.

The audible alarm is located in the aft cabin ceiling and beeps if the ELT is activated. The alarm is powered by the transmitter unit.

The battery pack provides 121.5 MHz and 406 MHz transmission for the first 24 hours after activation and an additional 26 hours of 121.5 MHz transmission only.

The transmitter unit has a single axis G-switch for detection of in impact in the direction of flight. If the G-switch activates, the unit starts transmitting.

The aircraft identification module remains attached to the aircraft and is available to reprogram the ELT whenever it is replaced.

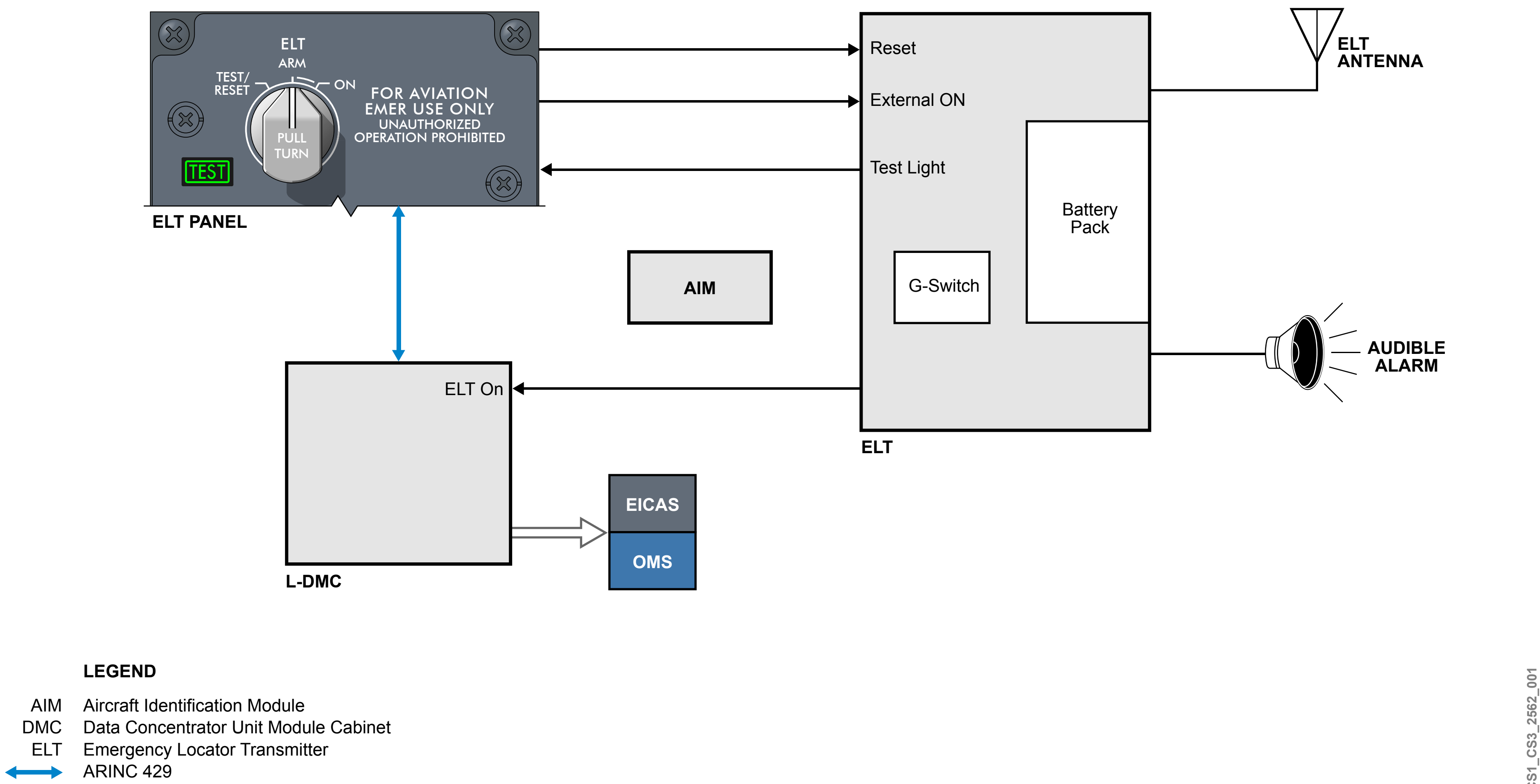


Figure 27: ELT System Interface

ELT MODES

OFF Mode

In the OFF mode, all circuits in the transmitter unit are disabled. This mode is only available on the transmitter unit front panel.

ARM Mode

During normal operation, the ELT control panel knob is set to the ARM position. In the event of a sudden deceleration, the G-switch will automatically activate the ELT transmission.

During ELT transmission, a tone is heard in the crew headsets, or the speaker when the emergency frequency is set on the VHF radio. An ELT ON caution message is displayed and the TEST LED indicator illuminates. The ELT transmitter unit light also illuminates when the unit transmits.

The ELT transmits continuously until the battery is discharged or the ELT is reset.

Manual Mode

For manual activation of the ELT, either the control panel switch can be selected ON or the transmitter unit toggle switch can be set to TX.

If the ELT control panel switch is selected ON, the ELT transmits continuously until the battery is discharged or the switch is momentarily turned to the TEST/RESET position.

If the transmitter unit toggle switch is set to the TX position, the ELT transmits continuously until the battery is discharged or the switch is returned to the OFF position.

In both cases, an ELT ON caution message is displayed while the ELT transmits.

Test/Reset Mode

To reset following ELT G-switch activation, turn the control panel switch from the ARM position to the TEST/RESET position for less than 3 seconds, and then return to the ARM position. The ELT transmission deactivates but remains armed. The reset procedure can also be accomplished by an equivalent OFF-ARM-OFF selection of the transmitter unit toggle switch.

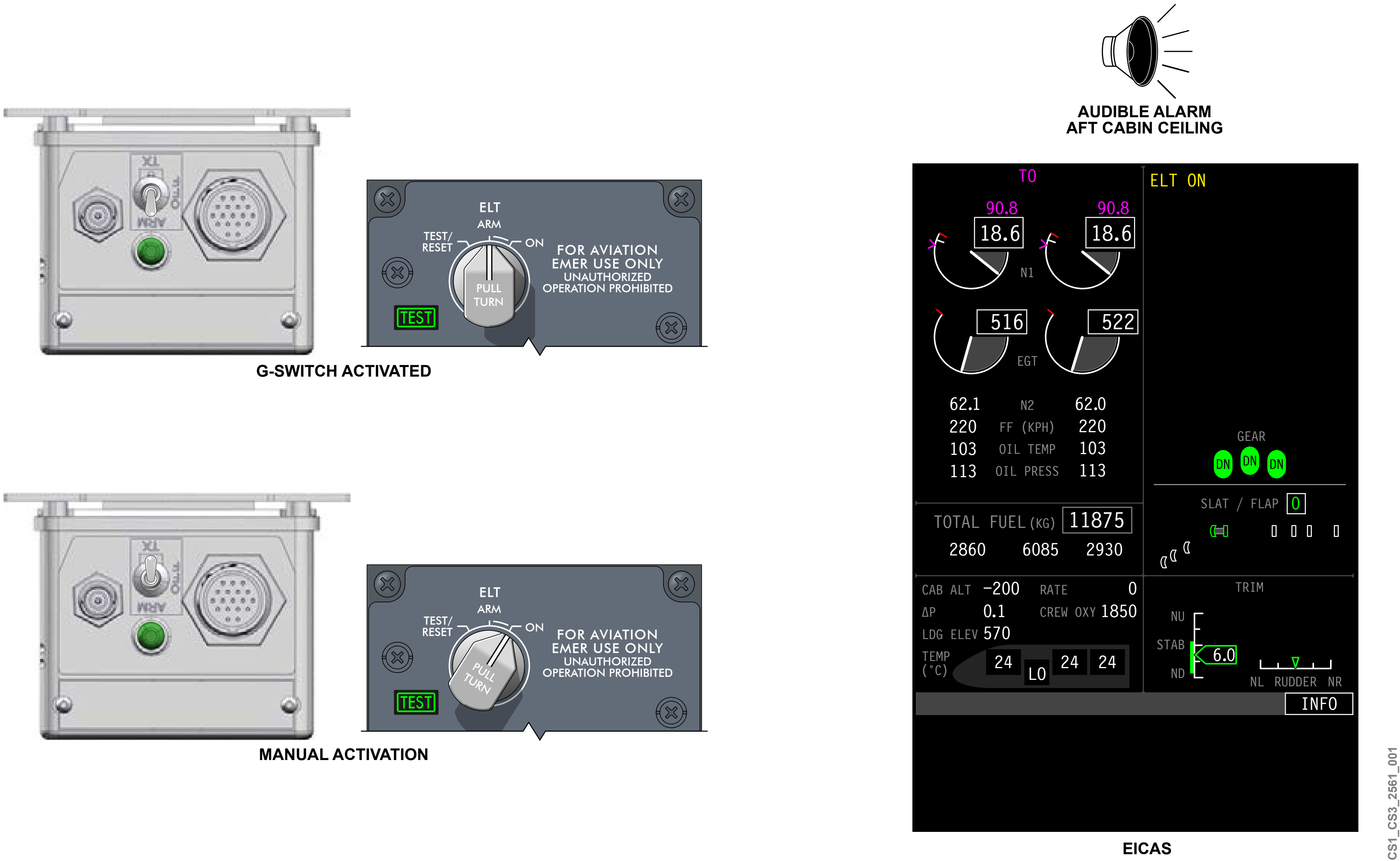


Figure 28: Emergency Locator Transmitter Modes

MONITORING AND TESTS

TESTS

The operational test checks the transmitter, the ELT battery and can be performed from:

- ELT transmitter unit
- ELT panel

Operational Test Setup

On the transmitter unit, ensure the toggle switch is in the ARM position.

On the ELT panel in the flight deck, ensure the rotary switch is in the ARM position.

On the audio control panel (ACP):

- Push the VHF 1 listen switch in
- Set the SPKR volume control knob to the center position
- On the control tuning panel (CTP), tune the VHF 1 to 121.5 MHz

CAUTION

ELT operational tests must be carried out only during a time approved by the National Regulatory Authority. If a test signal is transmitted at a different time, it can initiate a search and rescue operation.

NOTE

Two people are required for the operational test.

Operational Test from ELT

Perform the operational test from the transmitter unit as follows:

- On the transmitter unit, set the test switch momentarily from the ARM position to the OFF position and back to the ARM position
- On the transmitter unit, ensure the LED flashes once if the aircraft identification module (AIM) is installed on the ELT, and twice if the AIM is not installed
- Ensure the audible alarm in the aft cabin ceiling is heard

In the flight deck:

- Ensure the ELT signal is heard on VHF 1
- On the ELT panel, ensure the LED flashes once if the AIM is installed on the ELT, and twice if the AIM is not installed

Figure 29: Self-Test from ELT

Operational Test from ELT Panel

Perform the operational test from the ELT panel as follows:

- On the ELT panel in the flight deck, set the rotary switch to the TEST/RESET position and back to the ARM position in less than 3 seconds

NOTE

Holding the switch at the TEST/RESET position for more than 3 seconds activates the ELT.

- On the ELT panel, ensure the LED flashes once if the AIM is installed on the ELT, and twice if the AIM is not installed
- Ensure the ELT signal is heard on VHF 1
- Ensure the audible alarm in the aft cabin ceiling is heard

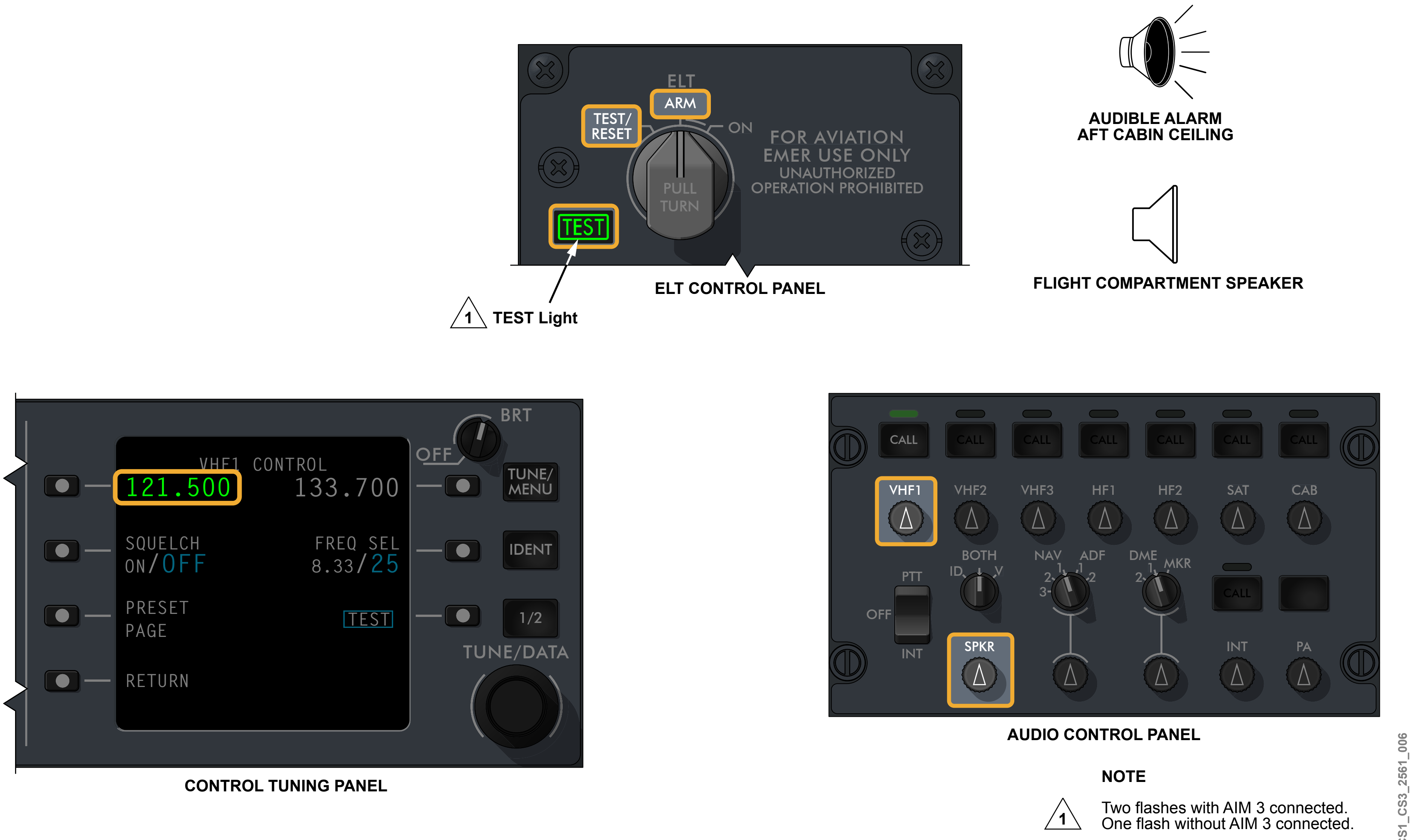


Figure 30: ELT Self-Test From Control Panel

CAS MESSAGES

The following page provides the crew alerting system (CAS) associated with the ELT.

Table 1: CAUTION Message

MESSAGE	LOGIC
ELT ON	ELT is transmitting.

25-63 EMERGENCY ESCAPE SLIDE SYSTEM

GENERAL DESCRIPTION

The emergency escape slide system consists of four door-mounted slides and two overwing emergency exit (OWEE) door slides.

COMPONENT LOCATION

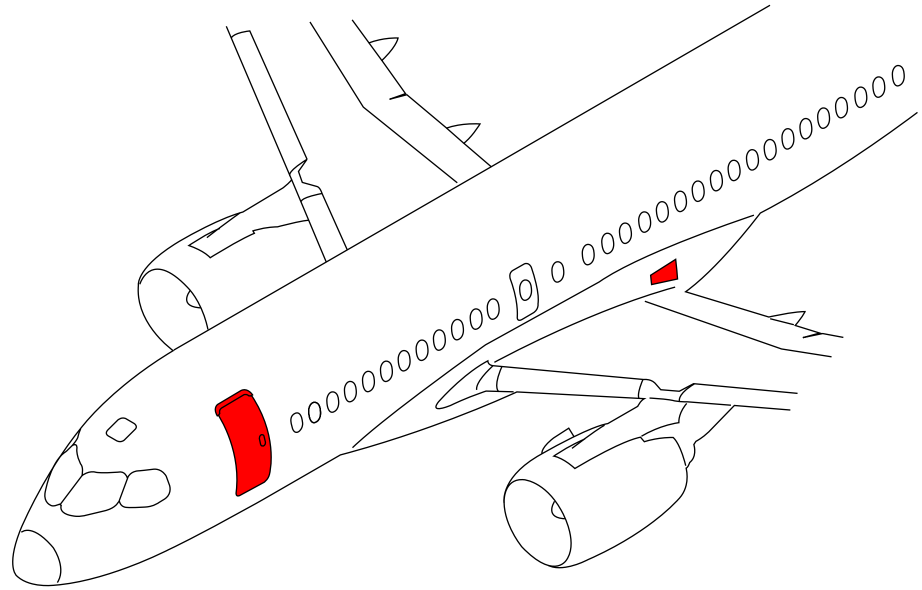
DOOR MOUNTED SLIDES

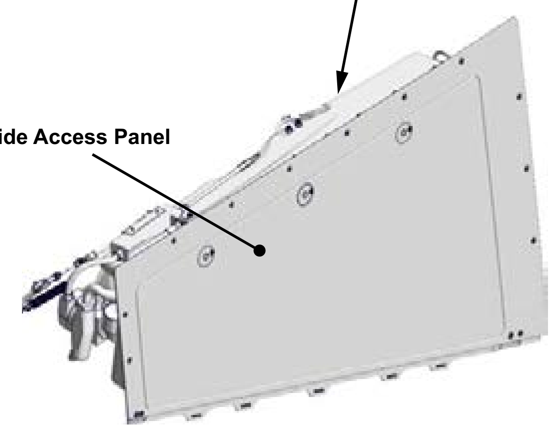
The door mounted slides are located under the bustle on each passenger and service door.

OVERWING EMERGENCY EXIT DOOR SLIDES

The slides for the OWEE doors are located in the wing-to-body fairing (WTBF) above the wing trailing edges.

FORWARD PASSENGER DOOR





OVERWING EMERGENCY EXIT DOOR SLIDE

NOTE

Same configuration for all passenger and service doors.

CS1_CS3_2566_002

Figure 31: Slide Locations

COMPONENT INFORMATION

DOOR MOUNTED SLIDE

The door-mounted slides consist of:

- Packboard
- Inflatable slide
- Reservoir
- Girt bar and release mechanism

The packboard is the primary structure which mounts the slide system to the interior of the door via two upper and two lower mounting brackets. The inflatable slide and reservoir are secured to the packboard with a lacing cover. The reservoir gauge is visible through a viewing window in the bustle that covers each door-mounted slide system.

The slide is protected by a bustle installed on the lower part of the door. The bustle attaches to the packboard via two upper hinges and two side clips.

The girt bar is attached to the girt which connects to the slide. The girt bar is engaged into floor fittings when the door is placed in the armed mode. The slide release mechanism, part of the door mounted slide, frees the lacing cover allowing the slide to drop from the door and deploy.

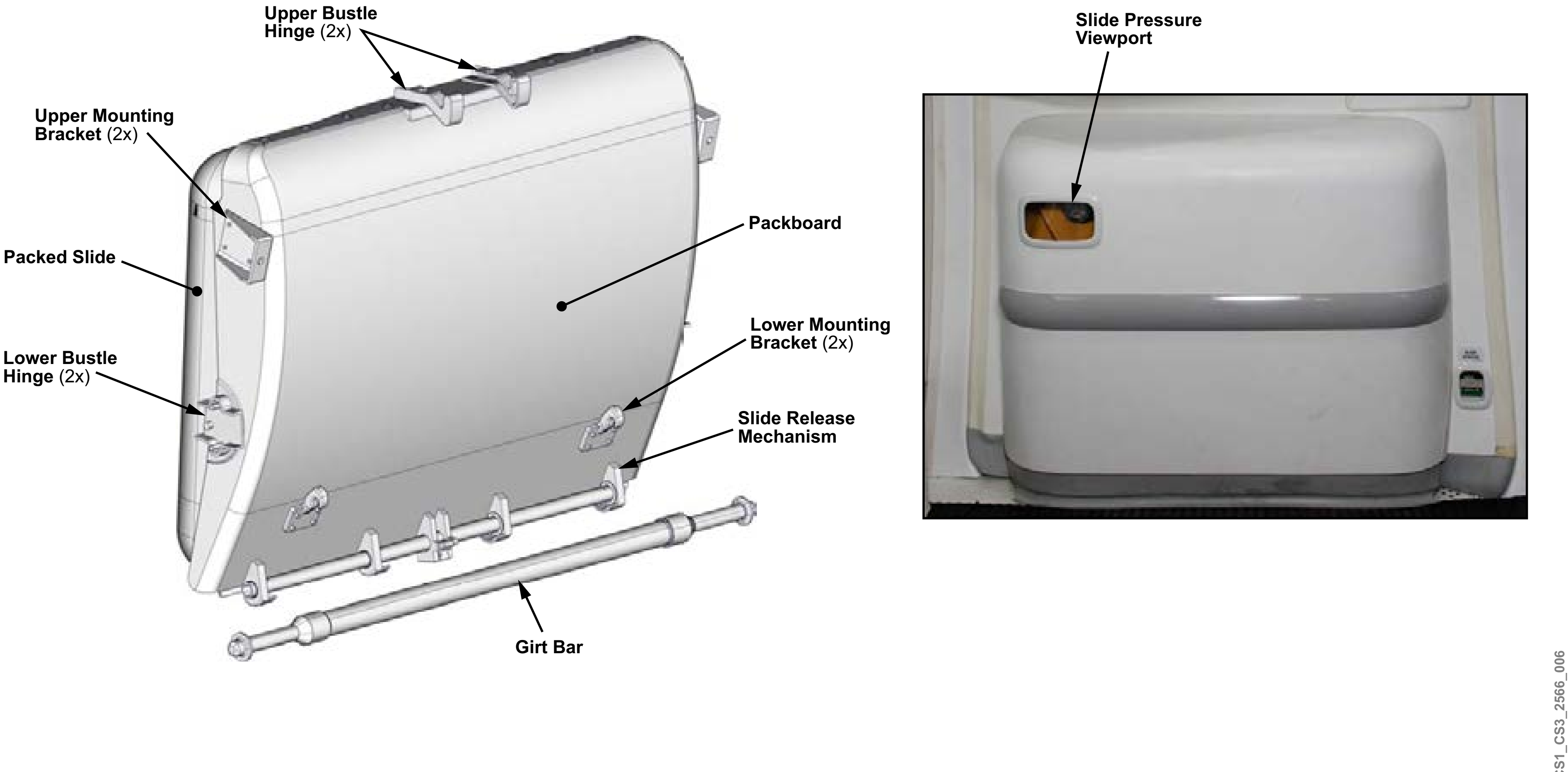


Figure 32: Door Mounted Slide

Door Mounted Slide Assembly

The single-lane inflatable slide has the following components mounted to it:

- Reservoir
- Aspirator
- Girt bar
- Mooring and life lines
- Release handle
- Lighting system
- Manual inflation handle

The reservoir is installed in a fabric sling under the inflatable slide. Each reservoir is a cylindrical composite structure comprised of a seamless aluminum cylinder reinforced with overlapping carbon filaments bonded with epoxy resin. The reservoir is charged to 3,000 psi.

A direct reading, go/no-go temperature-compensated pressure indicator gauge on the reservoir displays the reservoir pressure.

The girt bar holds the slide to the aircraft during slide deployment.

The reservoir has a valve and regulator assembly. The valve and regulator assembly provides a high-pressure flow of gas to the aspirator inlet. The high-pressure gas induces a secondary flow of ambient air into the aspirator to assist the reservoir in inflating the slide.

A mooring line is provided for use when the slide is deployed in water. The mooring line can be released by pulling a release lanyard attached to the slide end of the mooring line. Lifelines attached to both sides of the slide provide hand holds when the slide is used as a flotation device.

The slide has light-emitting diode (LED) lights powered by an battery pack. The battery pack is activated when the slide deploys. A pin, attached to a lanyard, is pulled from the battery pack. When the pin is pulled out, a switch turns on the lights.

If the slide fails to deploy automatically, a manual inflation handle can be used to deploy the slide.

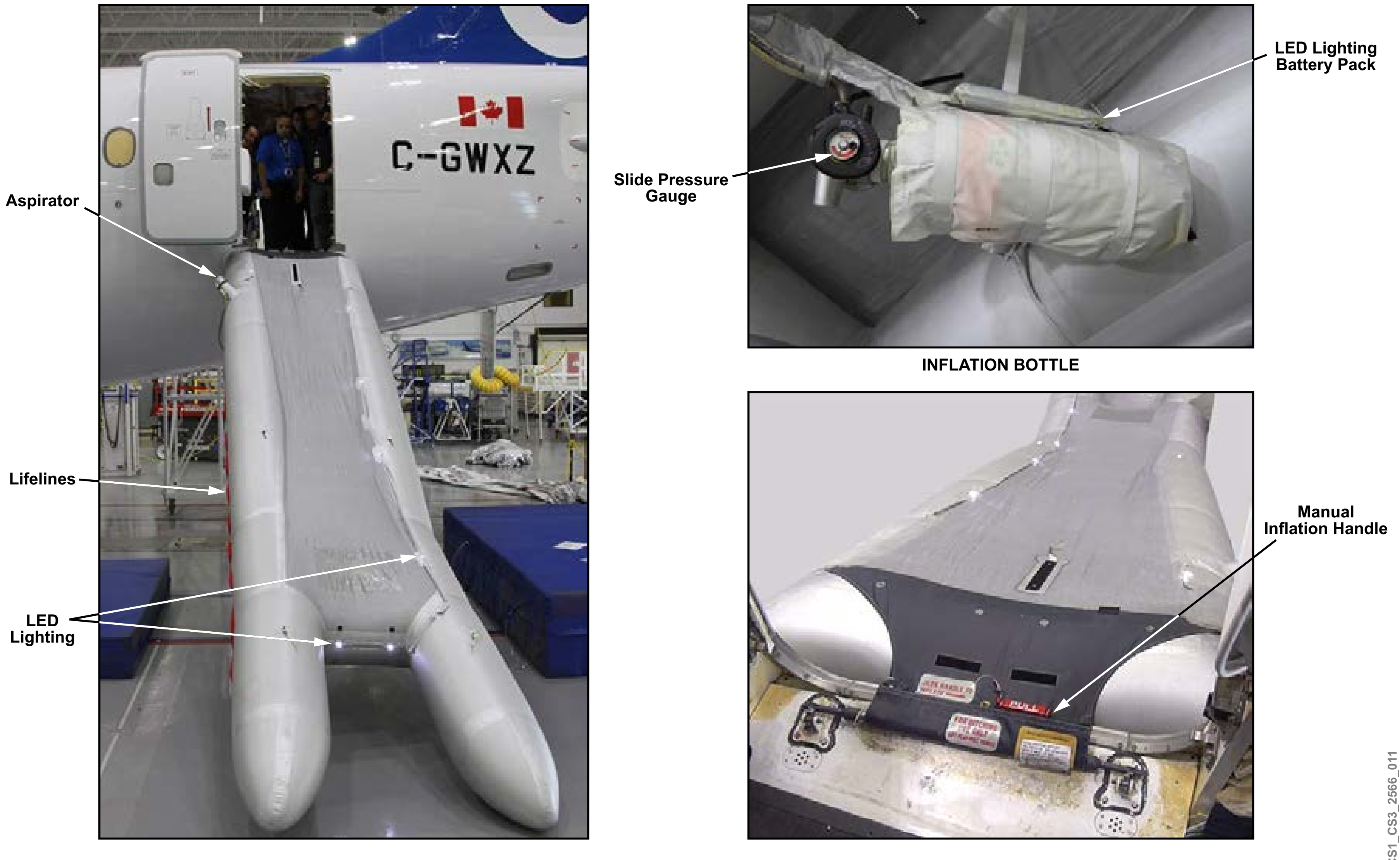


Figure 33: Door Mounted Slide Assembly

Girt Bar Attachment Fittings

Two girt bar fittings hold the slide girt bar in position when the passenger and service doors are armed. A spring loaded pawl is acted on by the door girt mechanism (see ATA 52).

When the doors is closed, and the door slide mode select handle is in the DISARM position, the door girt mechanism contacts the pawl and rotates it forward providing an opening to drop the girt bar into the girt fittings.

When the door slide mode select handle is moved to the ARM position, the girt mechanism rotates to drop the girt bar into the fittings. As the girt mechanism rotates it moves away from the pawl. The pawl spring moves the pawl up to lock the girt bar in the girt fitting.

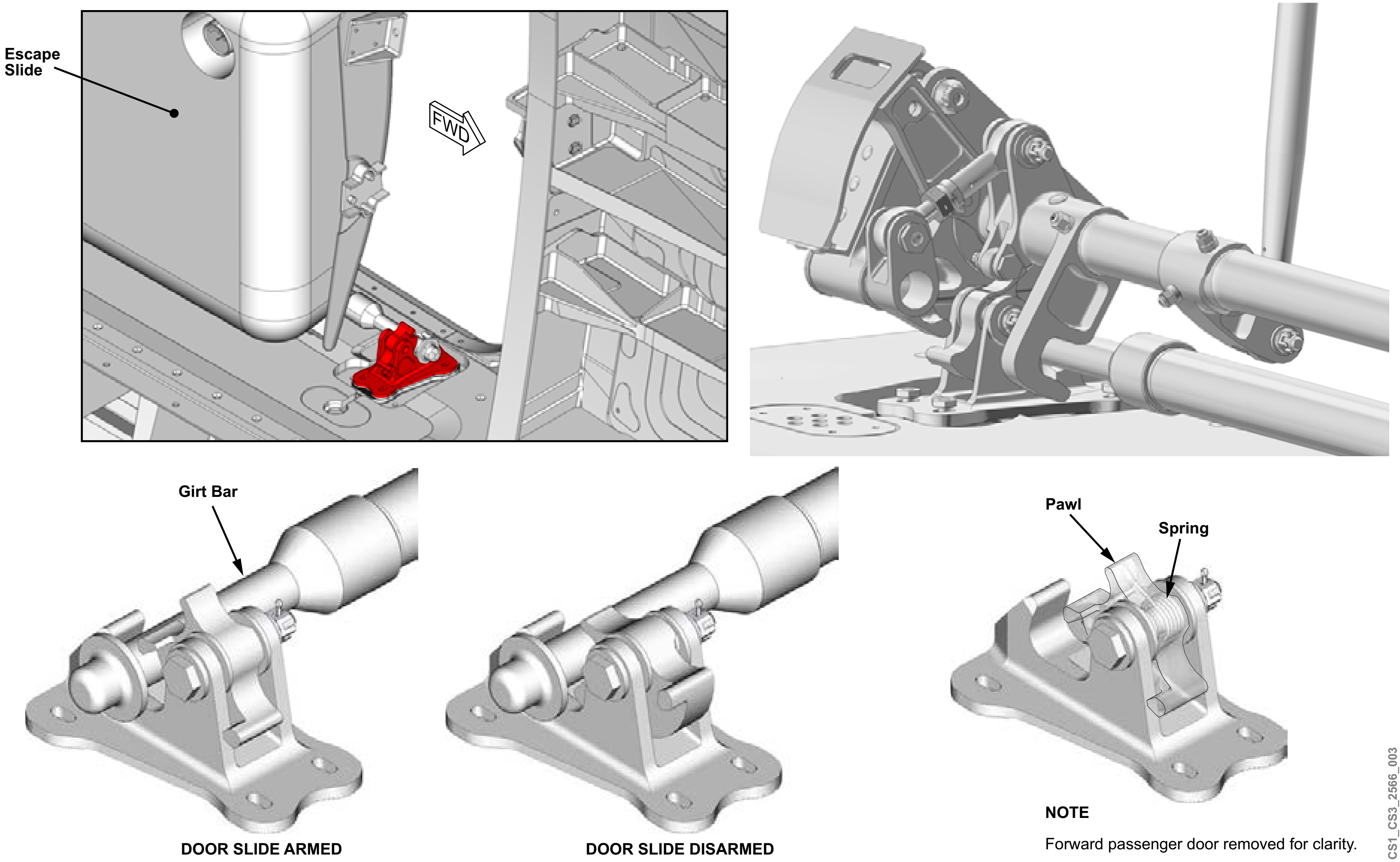


Figure 34: Girt Bar Attachment Fittings

OVERWING EMERGENCY EXIT DOOR SLIDES

Enclosure

The overwing emergency exit (OWEE) door slide is installed in an enclosure located on the wing-to-body fairing (WTBF). The enclosure has a door panel that opens when the slide is deployed.

Pneumatic door latches and ball locks secure the door panel to the enclosure until the slide inflation is initiated. In the event one of the latches fails closed, the enclosure door will break to allow the slide to inflate.

The enclosure can be opened to access the slide inflation bottle and the aircraft slide inflation actuation cable. After removing the mounting screws, the enclosure can pivot on the lower mounting flange on the WTBF. A support strap holds the enclosure in position while maintenance is being performed.

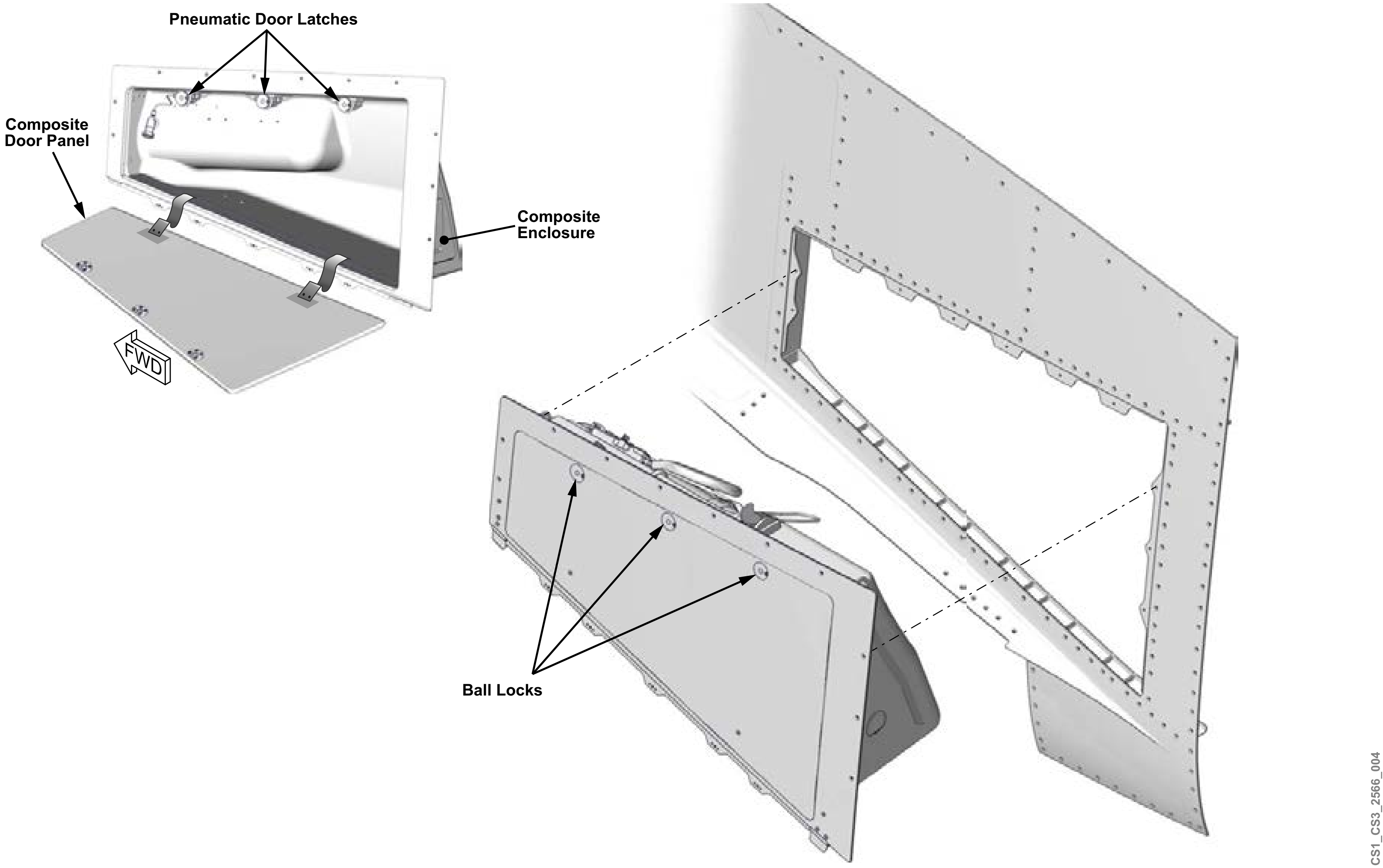


Figure 35: Overwing Emergency Exit Door Slide Enclosure

Inflation Assembly

The inflation assembly consists of a reservoir and a valve and regulator assembly. The reservoir is an aluminum cylinder reinforced with overlapping carbon filaments bonded with epoxy resin. The reservoir is charged to 2,500 psi.

The valve and regulator includes a temperature-compensated pressure gauge which indicates the status of the gas charge, a frangible burst disc which prevents overpressure of the reservoir, and a filler valve for charging the reservoir.

The direct reading pressure gauge cannot be viewed with the slide installed so a pressure switch is installed to monitor the reservoir pressure.

The cable connecting block has a hinged door that allows access to the cable connection between the aircraft slide inflation actuation cable and the inflation assembly. The connecting block door is held closed by a safety pin that is used to disarm the inflation assembly before maintenance is performed on the off-wing slide.

The safety pin is inserted into the valve and regulator assembly to prevent actuation of the inflation unit. The safety pin must be returned to its storage position on the cable connection block at the completion off maintenance in order to lock the cable connection block closed.

NOTE

The aircraft slide inflation actuation cable is connected to the overwing emergency exit (OWEE) door and is described in ATA 52.

WARNING

1. **BEFORE PERFORMING MAINTENANCE ON THE OFF-WING SLIDE, DISARM THE SLIDE USING THE EMERGENCY KEY LOCATED AT THE EMERGENCY EXIT SIGN NEXT TO THE OWEE DOOR.**
2. **INSTALL THE SAFETY PIN INTO THE VALVE PIN HOLE, RIGHT AFTER REMOVING IT FROM THE CABLE CONNECTION BOX. NOT INSTALLING THE SAFETY PIN INTO THE VALVE PIN HOLE, RISKS ACTIVATING THE SLIDE, WHICH CAN CAUSE INJURIES TO PERSONS AND/OR DAMAGE TO EQUIPMENT.**

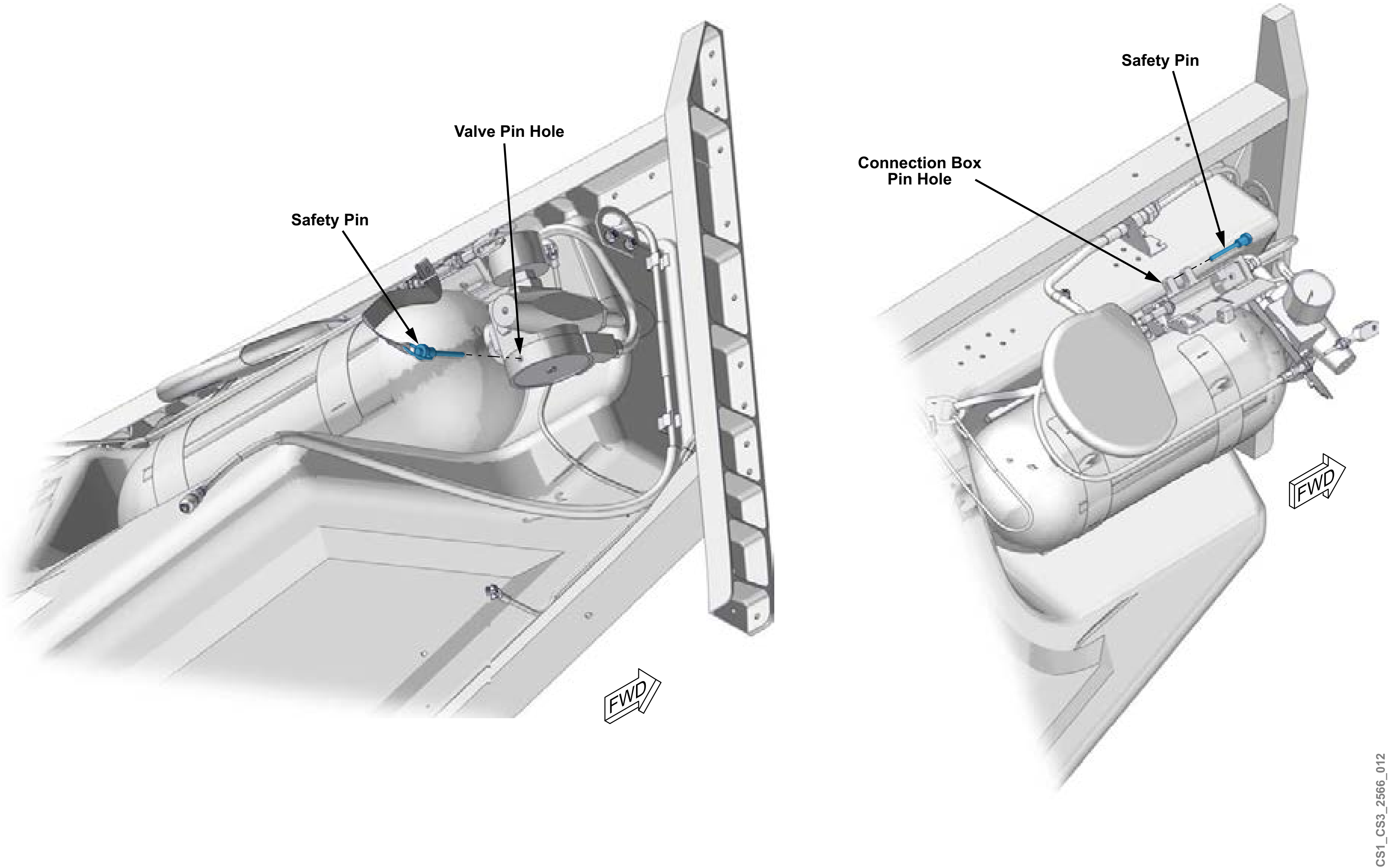


Figure 36: Inflation Assembly

Slide Assembly

The slide assembly consists of a ramp and sliding surface. An outboard rail acts as a handhold when the slide is inflated. The slide is directly connected to the enclosure by two quick-disconnect girts.

The valve and regulator assembly provides a high-pressure flow of gas to the aspirator inlet. The high-pressure gas induces a secondary flow of ambient air into the aspirator to assist the reservoir in inflating the slide.

The slide has light-emitting diode (LED) lights powered by a battery pack. The battery pack is activated when the slide deploys. A pin, attached to a lanyard, is pulled from the battery pack. When the pin is pulled out, a switch turns on the lights.

NOTE

The slides differ if dual OWEE are installed. The slides have two ramps instead of the standard single lane, and the inflation bottle is larger.

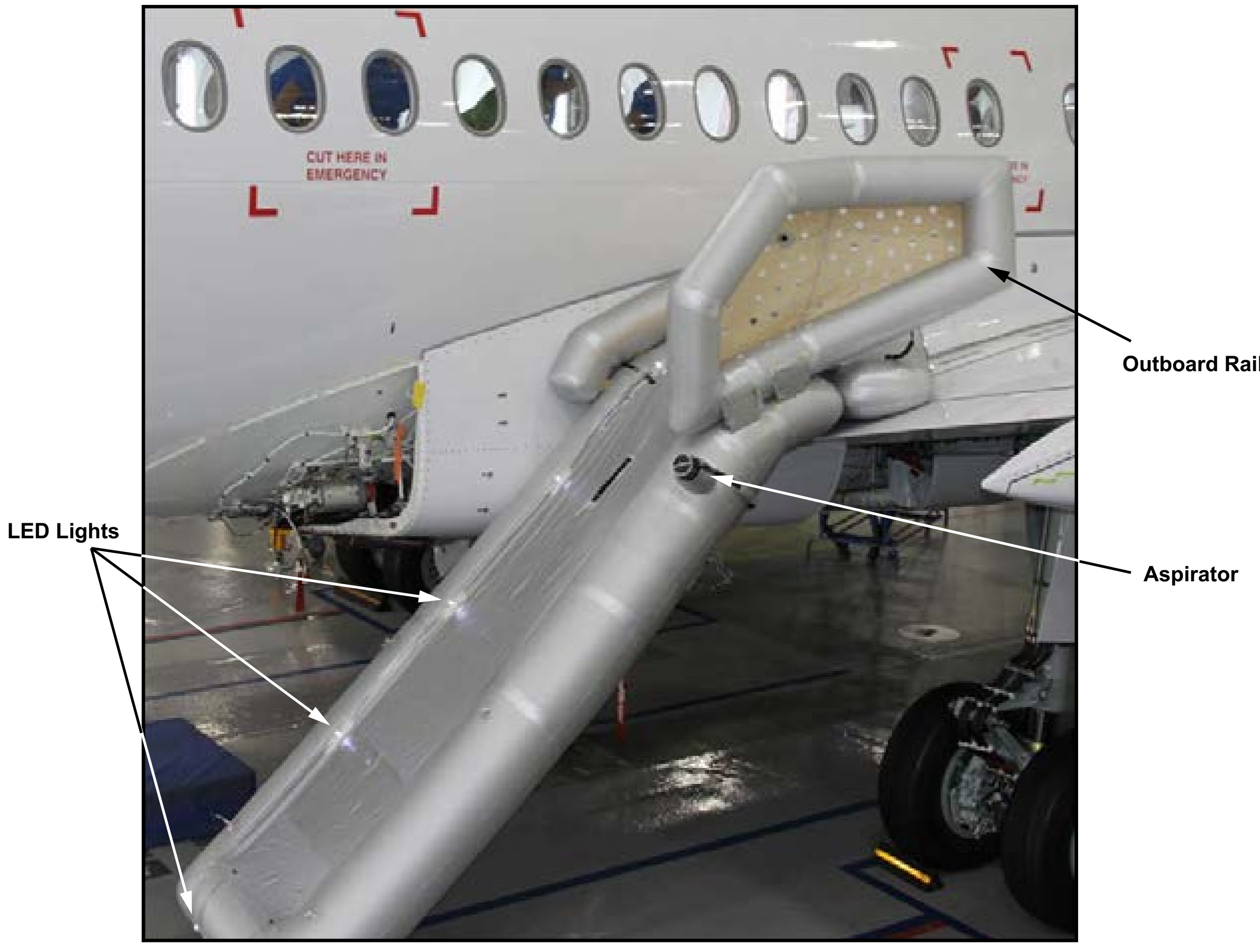


Figure 37: Slide Assembly

DETAILED DESCRIPTION

PASSENGER AND SERVICE DOOR SLIDE DEPLOYMENT

The slide can be deployed automatically or manually. The door-mounted slide is deployed and inflated whenever the door is opened with the mode select handle in the armed state.

If the automatic activation fails, the slide pack is left hanging from the girt. The manual activation PULL handle on the slide can be used to inflate the slide.

The girt bar is automatically engaged in the floor fittings when the door mode select handle is placed in armed mode. When the door opens, the pack release cable is pulled free, and the packed slide drops from the door. The packboard and lacing cover remain attached to the aircraft door.

The slide falls from the door to the full length of the girt and the automatic inflation cable tensions and opens the valve and regulator assembly, initiating inflation. The pull handle on the girt assembly manually actuates the same cable to open the valve and regulator assembly in the event that automatic inflation does not occur.

As regulated stored gas flows through the hose to the aspirator, ambient air is drawn in, increasing the rate of slide inflation. During inflation, the slide is restrained by frangible links in a semi-extended position. As the slide pressure increases, the link separates and the slide extends to full length within 5 seconds.

MODE SELECT HANDLE

MANUAL INFLATION

Figure 38: Passenger and Door Slide Deployment

OVERWING EMERGENCY EXIT DOOR SLIDE AUTOMATIC AND MANUAL DEPLOYMENT

The slide can be deployed automatically or manually. The overwing emergency exit (OWEE) door slide is always armed. It deploys and inflates automatically whenever the OWEE door is opened.

If the automatic activation fails, the off-wing enclosure remains closed by the door panel. The manual activation PULL handle, located in the overwing emergency exit door surround initiates the slide inflation.

The manual slide mechanism bypasses the automatic slide deployment mechanism and operates the common slide cable to release the enclosure panel and inflate the off-wing slide.

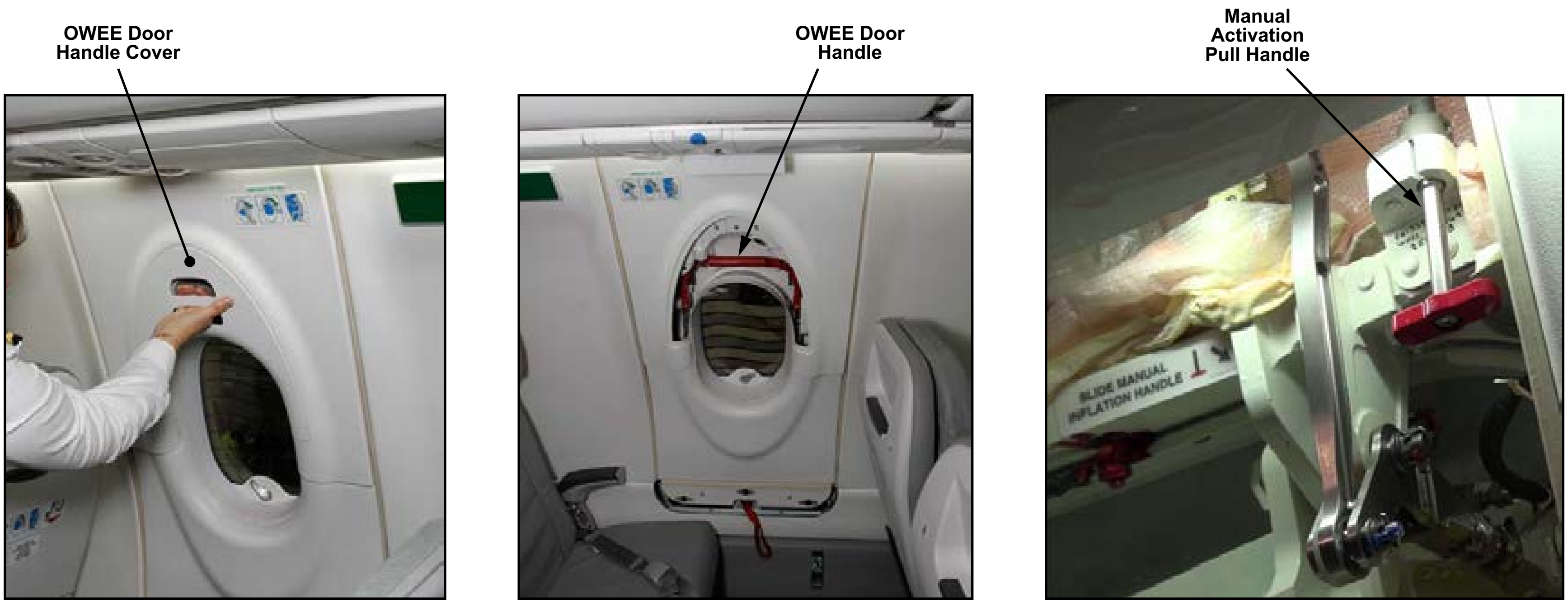


Figure 39: Overwing Emergency Exit Door Slide Automatic and Manual Deployment

OVERWING EMERGENCY EXIT DOOR SLIDE DEPLOYMENT

The slide can be deployed automatically or manually. The overwing emergency exit (OWEE) door slide is always armed. It deploys and inflates automatically whenever the OWEE door is opened.

If the automatic activation fails, the enclosure remains closed. The manual activation PULL handle, located in the overwing emergency exit door surround initiates the slide inflation.

The enclosure door is held closed by pneumatic pinball lock mechanisms, releasing the door from the enclosure when the OWEE door is opened. At the start of the slide inflation, the gas from the reservoir operates the pneumatic ball-lock mechanism releases, which allow the top of the enclosure door to open. After pivoting downward, the hinge at the bottom of the enclosure door disengages, and the door is released from the enclosure. Two retaining straps keep the door attached to the enclosure.

The ramp section of the slide inflates first and then the frangible links break allowing the slide section to deploy over the trailing edge of the wing.

The manual slide mechanism bypasses the automatic slide deployment mechanism and operates the common slide cable to release the enclosure panel and inflate the off-wing slide.

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Figure 40: Overwing Emergency Exit Door Slide Deployment

CONTROLS AND INDICATIONS

PASSENGER AND SERVICE DOOR SLIDES

The passenger and service door slides are armed using the escape slide mode select handle.

A mechanical flag at the bottom of the door shows the actual girt bar status.

A direct reading pressure gauge is installed on the reservoir and can be viewed through a window in the slide bustle.

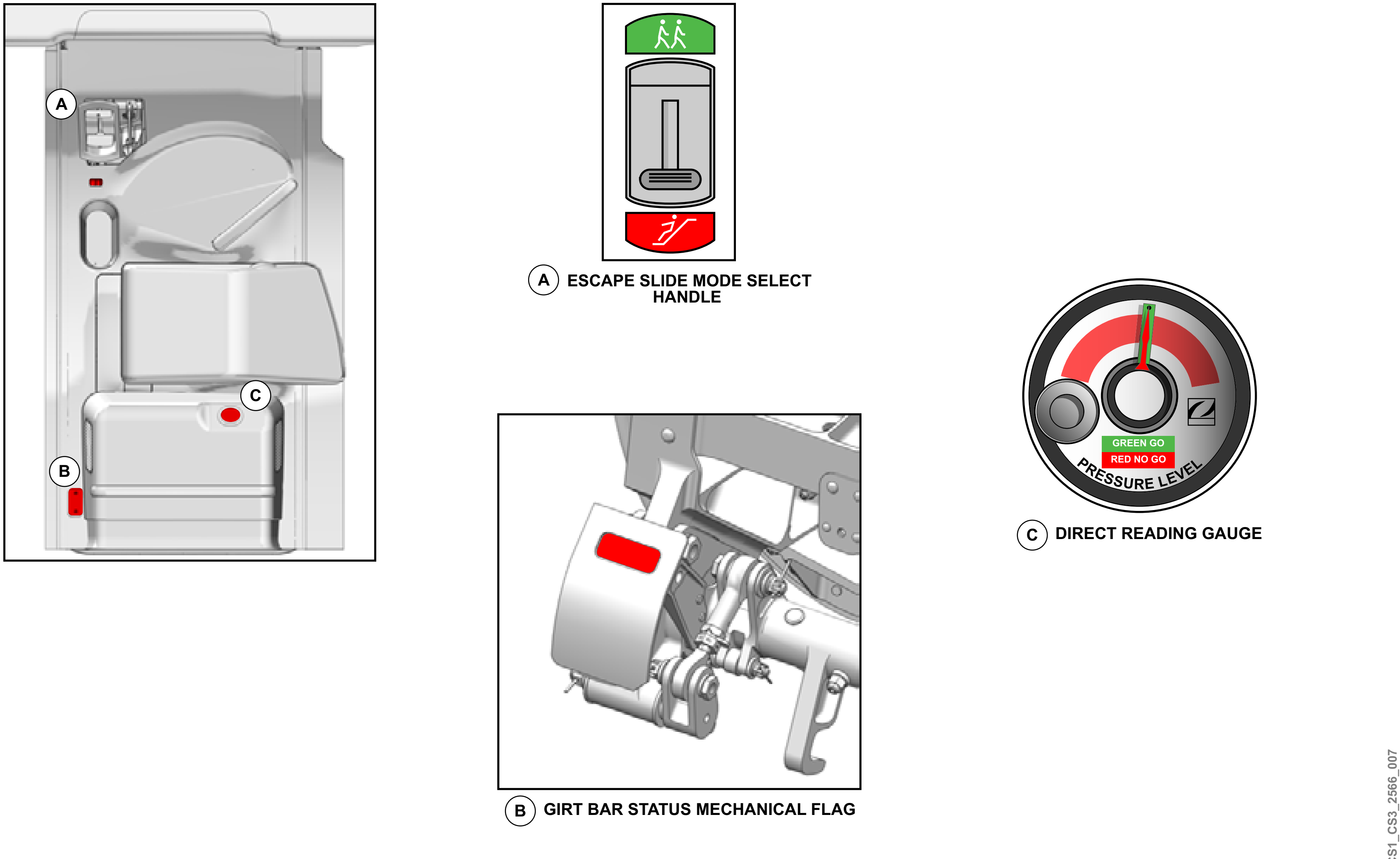
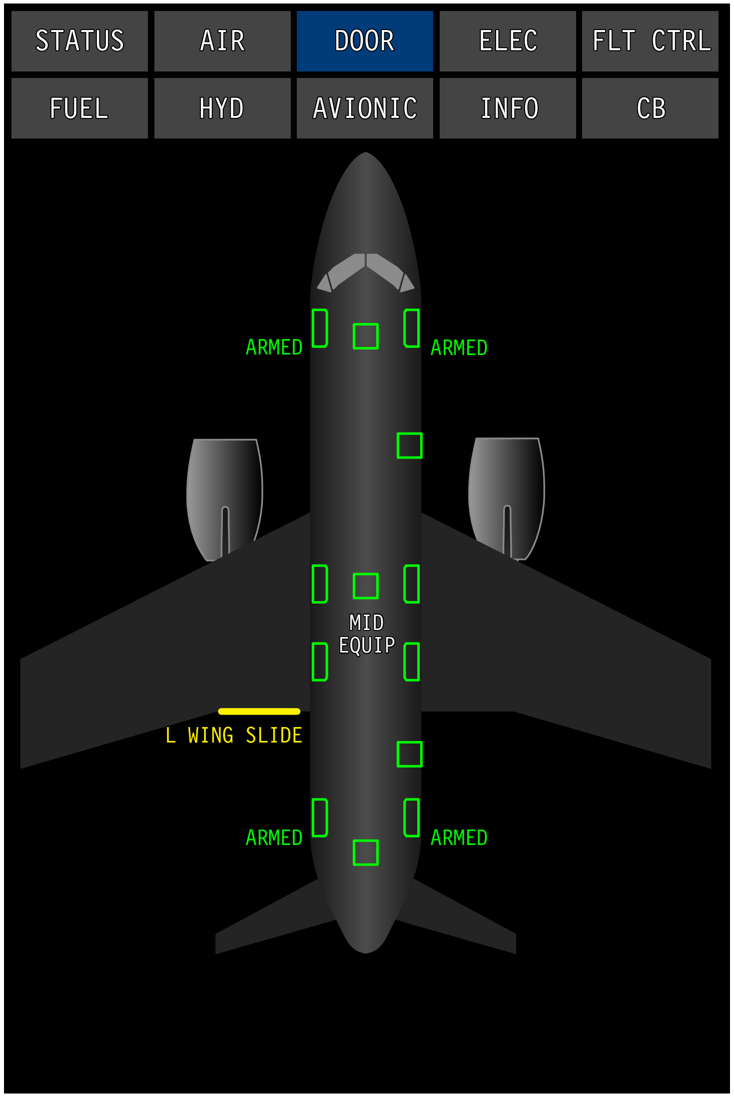


Figure 41: Passenger and Service Door Slide Indications

SYNOPTIC PAGE

The DOOR synoptic page provides the slide status for the passenger, service, and overwing emergency exit doors.



SYNOPTIC PAGE - DOOR

DOOR SLIDE TABLE	
Symbol	**Condition**
DISARMED	Door closed / latched / locked but Slide Lever Disarmed (removed when door opened)
ARMED	Door closed / latched / locked & Door Slide Lever Armed
SLIDE	Door SLide Fault Detected (Electrical or mechanical failures, Invalid or failed LGSCU)

WING SLIDE STATUS (ALWAYS ARMED)	
Symbol	**Condition**
SLIDE	Wing slide fault detected (low pressure or deployed).

CS1_CS3_2566_009

Figure 42: Passenger, Service, and Overwing Emergency Exit Door Slide Indications

MONITORING AND TESTS

The following page provides the CAS and INFO messages for the emergency escape slide system.

CAS MESSAGES

Table 2: CAUTION Messages

MESSAGE	LOGIC
FWD SLIDE	Forward passenger or service door slide is faulty with at least one engine running.
AFT SLIDE	AFT passenger or service door slide is faulty with at least one engine running.
WING SLIDE	Any wing slide faulty or deployed (low pressure value detected).
DOOR SLIDE DISARMED	Any access door slide is still disarmed airborne or with one engine running.

Table 3: ADVISORY Messages

MESSAGE	LOGIC
DOOR SLIDE FAULT	Any electrical or mechanical failures of any door while on ground and no engine running.
DOOR SLIDE DISARMED	Any access door slide is disarmed on ground and no engines running.

Table 4: INFO Messages

MESSAGE	LOGIC
52 DOOR SLIDE FAULT - FWD PAX DOOR SLIDE SNSR INOP	FWD passenger door slide sensor inop and weight-on-wheels (WOW).
52 DOOR SLIDE FAULT - AFT PAX DOOR SLIDE SNSR INOP	AFT passenger door slide sensor inop and weight-on-wheels (WOW).

Table 4: INFO Messages

MESSAGE	LOGIC
52 DOOR SLIDE FAULT - FWD SERV DOOR SLIDE SNSR INOP	FWD service door slide sensor inop and weight-on-wheels (WOW).
52 DOOR SLIDE FAULT - AFT SERV DOOR SLIDE SNSR INOP	AFT service door slide sensor inop and weight-on-wheels (WOW).
52 DOOR SLIDE FAULT - FWD PAX DOOR SLIDE TRGT INOP	FWD passenger door slide target disagree and weight-on-wheels (WOW).
52 DOOR SLIDE FAULT - AFT PAX DOOR SLIDE TRGT INOP	AFT passenger door slide target disagree and weight-on-wheels (WOW).
52 DOOR SLIDE FAULT - FWD SERV DOOR SLIDE TRGT INOP	FWD service door slide target disagree and weight-on-wheels (WOW).
52 DOOR SLIDE FAULT - AFT SERV DOOR SLIDE TRGT INOP	AFT service door slide target disagree and weight-on-wheels (WOW).
52 WING SLIDE - L OWEE SLIDE INOP	Left overwing emergency exit (OWEE) door slide reservoir pressure low.
52 WING SLIDE - R OWEE SLIDE INOP	Right overwing emergency exit (OWEE) door slide reservoir pressure low.

PRACTICAL ASPECTS

OVERWING EMERGENCY EXIT DOOR SLIDE ARMING/DISARMING

When the OWEE door needs to be opened for maintenance work, a disarm key is used to arm or disarm the OWEE door slide mechanism. The key is inserted and rotated to the DISARM position in the mode selector box, located underneath the emergency exit sign beside the door. This operation disables the OWEE door slide automatic deployment. Red armed markings and green disarmed markings indicate the slide status. Once the key is installed and placed in disarmed condition, it cannot be removed.

To rearm the slide, rotate the key to the arm position and remove the key.

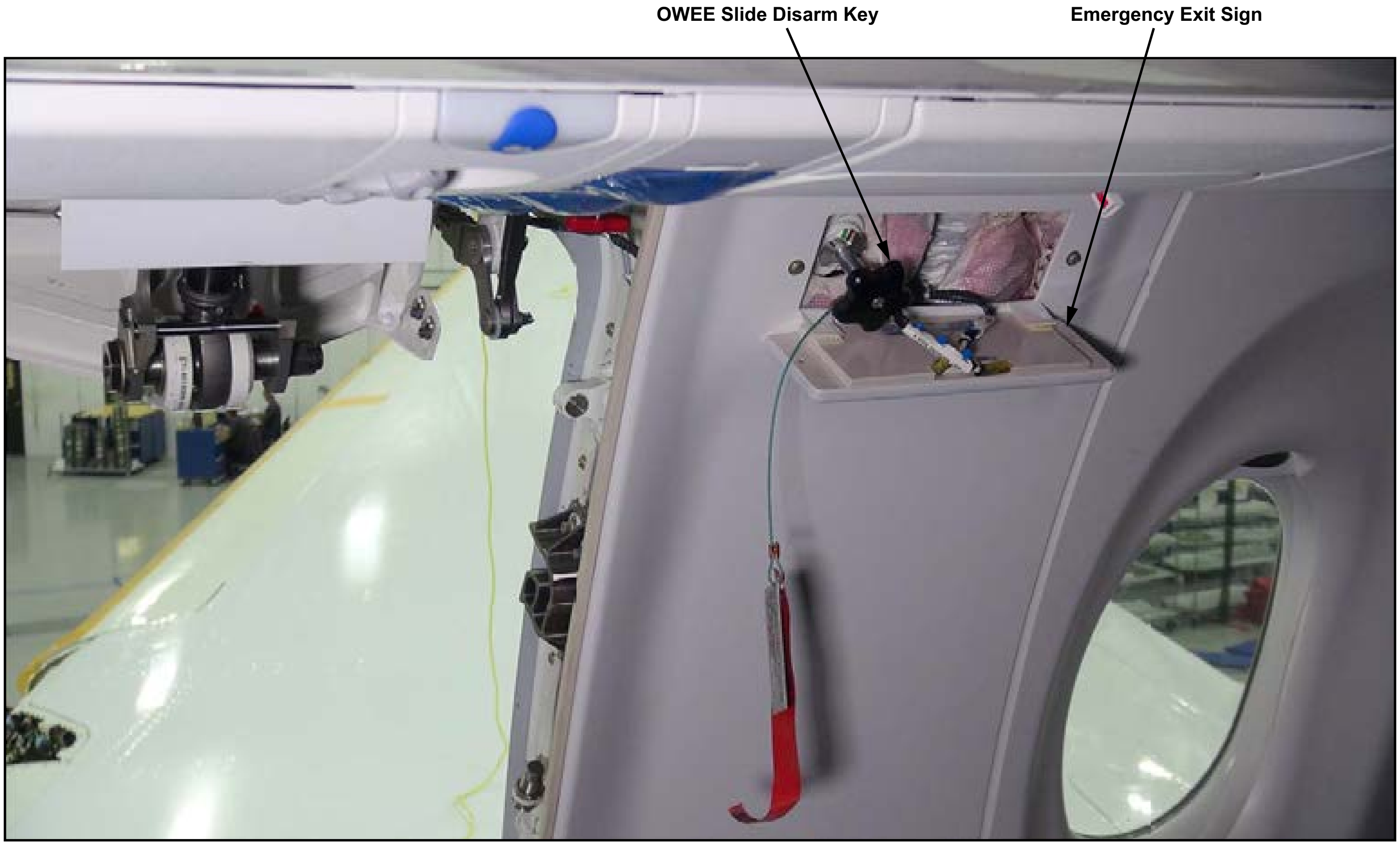


Figure 43: Overwing Emergency Exit Door Slide Lockout

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ATA 27 - Flight Controls

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

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FLIGHT CONTROLS - CHAPTER BREAKDOWN

27-00 ELECTRONIC FLIGHT CONTROL SYSTEM

GENERAL DESCRIPTION

The electronic flight control system (EFCS) is an integrated system that provides control and monitoring of all primary flight controls. The system also provides full-time augmentation of pilot commands and flight envelope protection during manual flight and autoflight.

Pilot inputs to the EFCS are provided from the following components:

- Sidestick controllers (SSCs)
- Pitch trim switches (mounted on the SSCs)
- Rudder pedal assemblies (RPAs)
- Flight spoiler control lever (FSCL)
- Aileron/rudder trim switches

The EFCS includes the following principal electronic components:

- Primary flight control computers (PFCCs)
- Inceptor interface modules (IIMs)
- Remote electronic units (REUs)
- Alternate flight control unit (AFCU)
- Motor control electronics (MCE)
- Attitude heading computer (AHC)

The EFCS contains distributed control electronics, located in the vicinity of the respective actuators, to reduce the weight of wiring.

The architecture is structured around three separated stick-to-surface control paths. Each of the three control paths consists of one IIM and its associated set of REUs and the three PFCCs, through high-performance data buses. The AFCU provides a completely independent fourth path.

By default, the EFCS operates in normal mode. However, depending on the type and nature of failures, a number of direct modes are available, including the AFCU direct mode.

In normal mode, the pilot control inputs are read by the three IIMs. The IIMs transmit this information directly to the PFCCs. Only one PFCC is in control at a time, the others are on standby. The controlling PFCC determines the required surface commands and retransmits these through the IIMs to the REUs. The REUs operate the flight control PCUs. Direct modes include the PFCC direct mode and REU direct.

The AFCU direct mode provides a command-by-wire system. In this mode, the pilot inputs are routed through the AFCU, then directly to the primary flight control power control units (PCUs) to move the surfaces.

The attitude heading computer (AHC) provides the PFCCs with attitude and heading reference inputs, as well as with aircraft acceleration, rate, and angle data

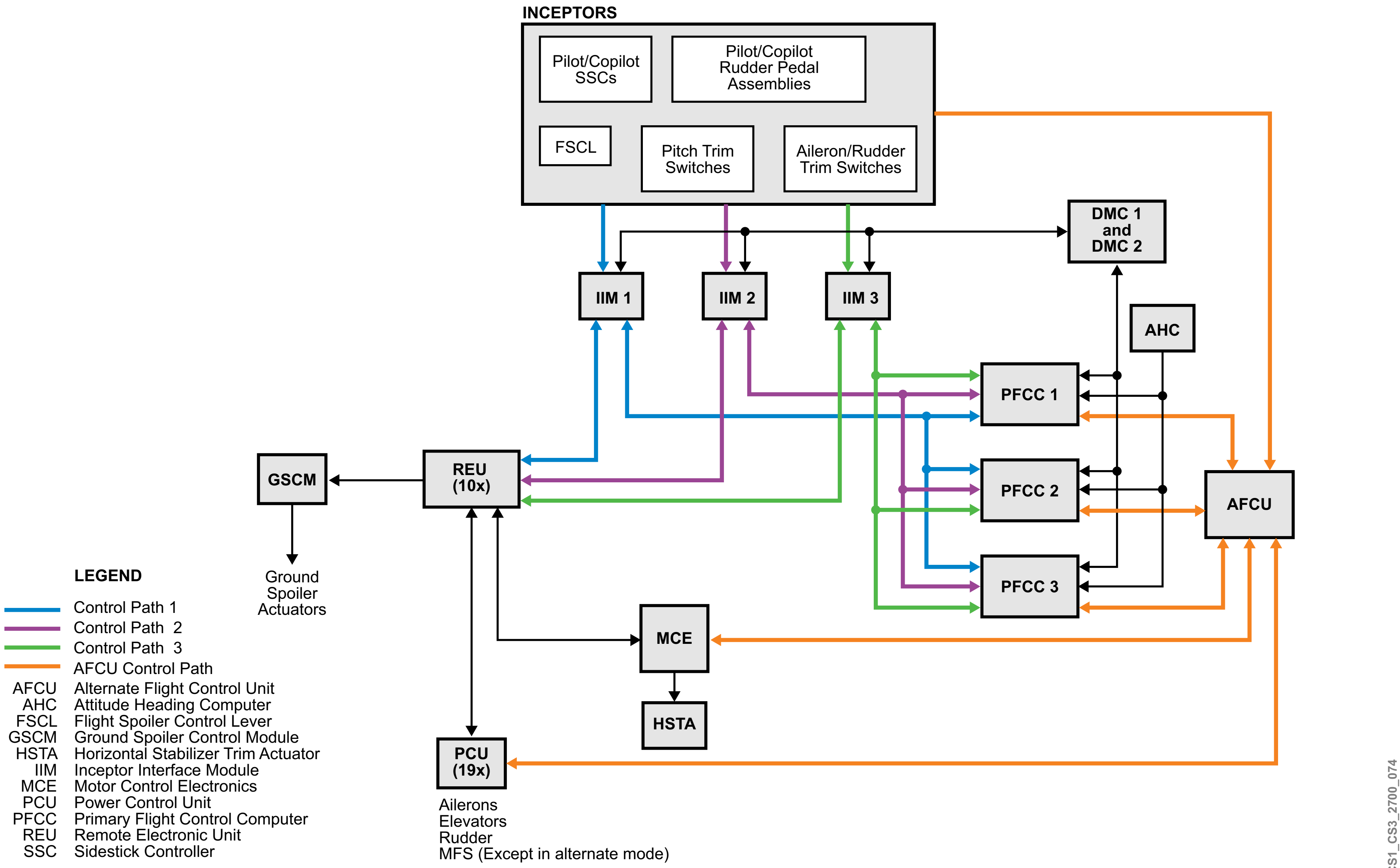


Figure 1: Electronic Flight Control System - Simplified

ELECTRICAL POWER DISTRIBUTION

The EFCS components, except the motor control electronics (MCE), use 28 VDC electrical power from the main DC BUSES (1 or 2), the fly-by-wire power converters (FBWPCs), or the DC ESS BUSES. The power supplies to the various component groups are divided into three power paths, with the AFCU power path comprising a fourth path.

Each individual circuit protection device (CPD) output, within the FBWPC, is independently protected against overload and short circuit.

In the event that all AC power is lost but the RAT generator has not yet come online, the secondary backup power supply from the DC ESS BUS provides emergency power to the FBWPC system. This secondary power source is automatically disconnected after 10 seconds to avoid draining the battery.

A diode shunt protection unit (DSPU) is installed in the DC power line to each REU and the AFCU. The DSPU protects the REU and AFCU in the event of a short circuit between the DC power line and the aircraft high voltage wiring. Each DPCU is installed near its respective REU or AFCU.

There are three power paths for the EFCS.

Power path no. 1 supplies:

- Primary flight control computer (PFCC) 1
- Inceptor interface module (IIM) 1
- Remote electronic units (REUs) (MFS 1, MFS 3, and AFT 1)
- Inertial reference unit (IRU) 1

Power path no. 2 supplies:

- PFCC 2
- IIM 2
- REUs (MFS 4, OB AIL, and AFT 4)
- AHC

Power path no. 3 supplies:

- PFCC 3
- IIM 3
- REUs (AFT 2, AFT 3, IB AIL, and MFS 2)

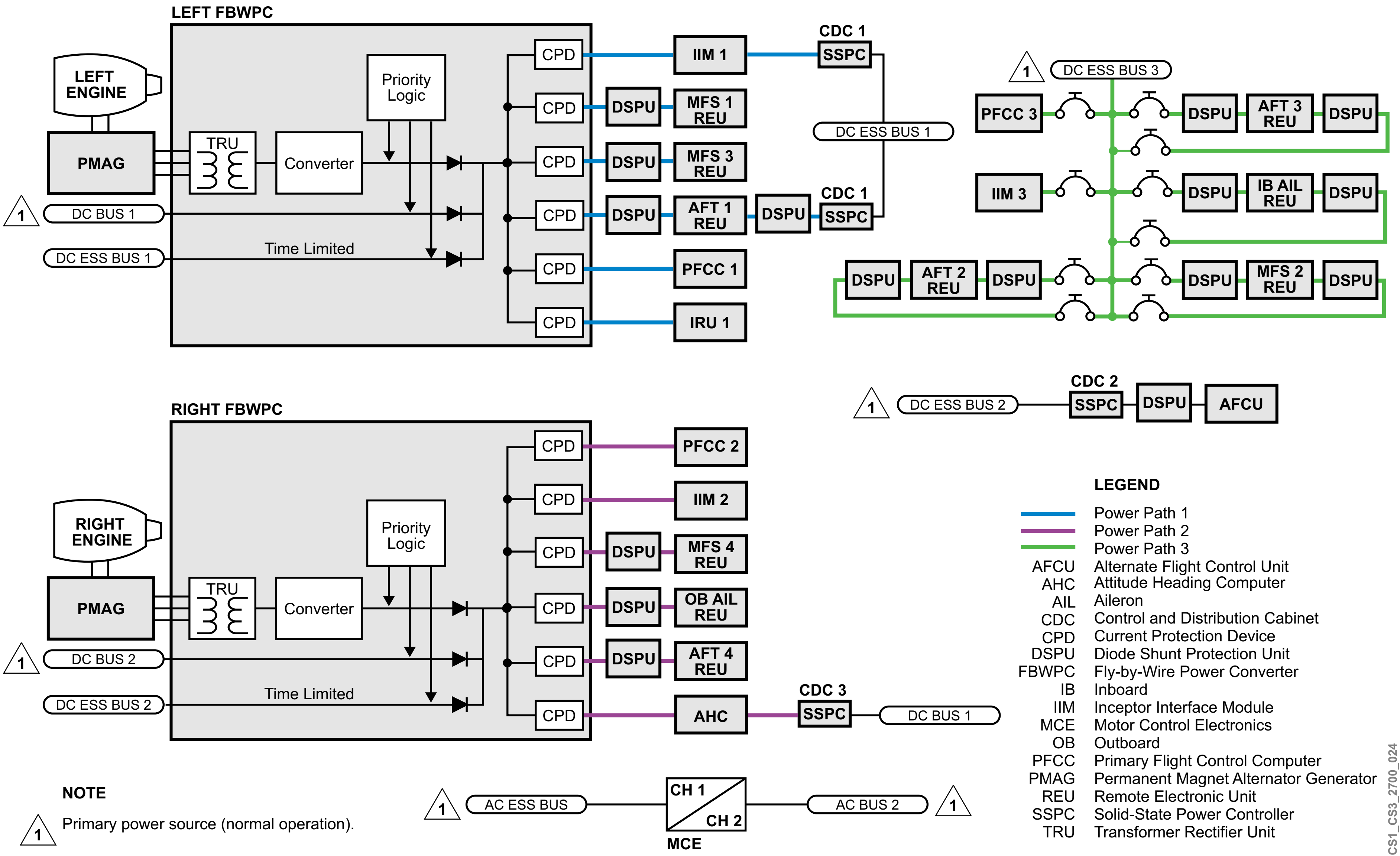
Power path no. 1 uses a secondary power source from DC ESS BUS 1 to power IIM 1 and AFT 1 REU. The reason for two separate power sources to these components is because the wires are routed through the engine rotor burst zone. The electrical power sources for IIM 1 are designated as primary power and secondary (back-up) power. These two power inputs enter the IIM through separate connectors. Automatic power switching in the IIM selects the primary power bus if its voltage is within allowable limits. Otherwise, the IIM automatically switches to the secondary power bus.

Power path no. 2 receives power from the right engine FBWPC and has similar architecture to power path 1.

Power path no. 3 is supplied from DC ESS BUS 3 only, using two separate paths for components where the wiring must pass through the rotor burst zone. The AFCU path is supplied exclusively by DC ESS BUS 2.

The MCE receives 115 VAC from AC BUS 2 and the AC ESS BUS.

The attitude heading computer (AHC) receives power from the right FBWPC and DC BUS 1.



LEGEND

	Power Path 1
	Power Path 2
	Power Path 3
AFCU	Alternate Flight Control Unit
AHC	Attitude Heading Computer
AIL	Aileron
CDC	Control and Distribution Cabinet
CPD	Current Protection Device
DSPU	Diode Shunt Protection Unit
FBWPC	Fly-by-Wire Power Converter
IB	Inboard
IIM	Inceptor Interface Module
MCE	Motor Control Electronics
OB	Outboard
PFCC	Primary Flight Control Computer
PMAG	Permanent Magnet Alternator Generator
REU	Remote Electronic Unit
SSPC	Solid-State Power Controller
TRU	Transformer Rectifier Unit

NOTE

1 Primary power source (normal operation).

Figure 2: EFCS Electrical Power Distribution

HYDRAULIC POWER DISTRIBUTION

The FBW flight controls, except the horizontal stabilizer trim system, are powered by the three hydraulic systems in different configurations. The aileron power control units (PCUs) are powered by hydraulic system no. 2 and hydraulic system no. 3. The left elevator PCUs are powered by hydraulic system no. 1 and hydraulic system no. 3. The right elevator PCUs are powered by hydraulic system no. 2 and hydraulic system no.3. The rudder PCUs are powered by all three hydraulic systems. Ground spoiler power is supplied by hydraulic system no. 1, and the multifunction spoilers (MFS) receive power from all three hydraulic systems. During AFCU direct mode, the single source of hydraulic power is hydraulic system no. 3.

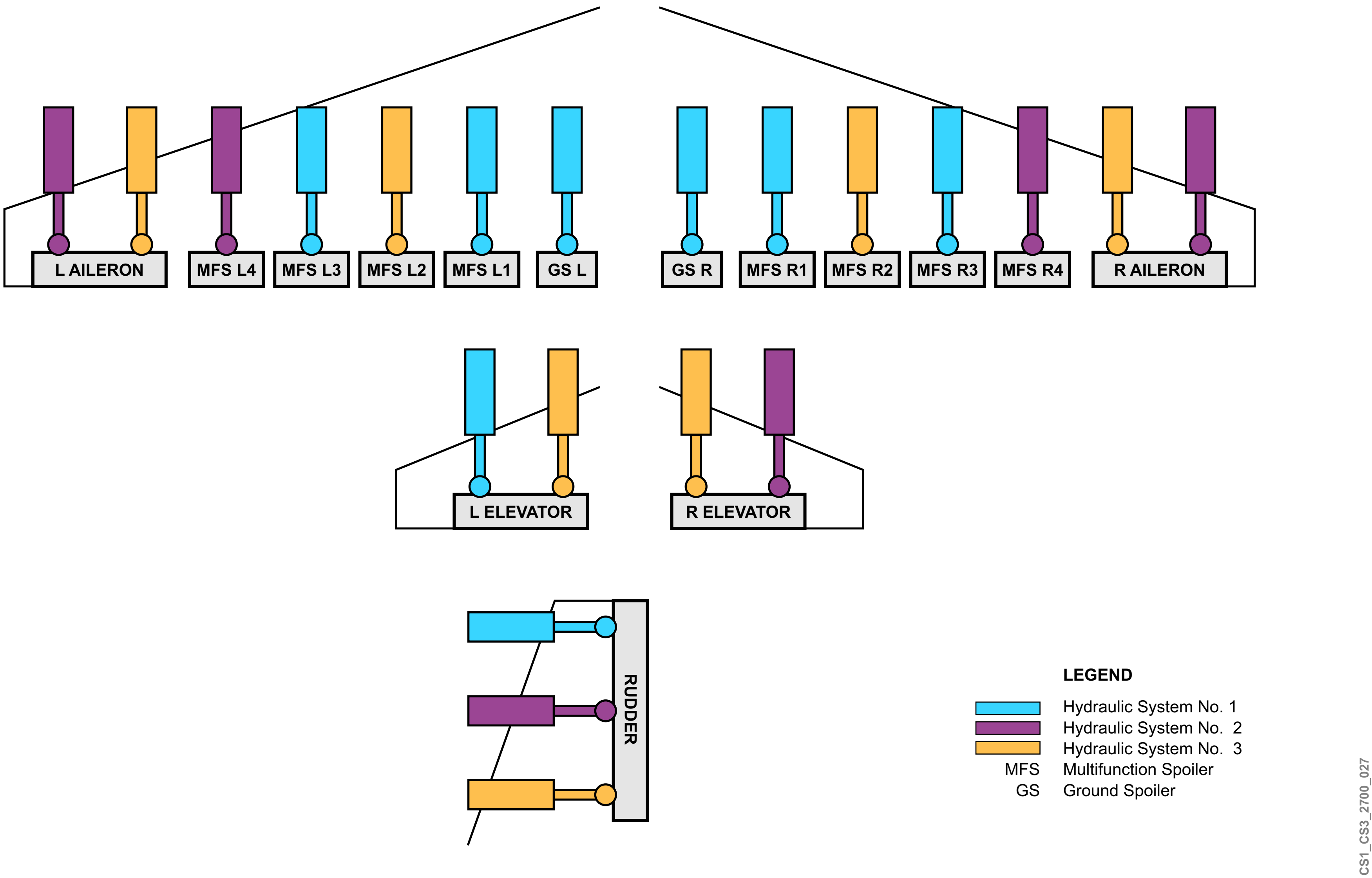


Figure 3: Hydraulic Power Distribution

COMPONENT LOCATION

The electronic flight control system (EFCS) consists of the following components:

- Primary flight control computers (PFCCs)
- Inceptor interface modules (IIMs)
- Alternate flight control unit (AFCU)
- Attitude heading computer (AHC)
- Remote electronic units (REUs) (Refer to figure 5)
- Motor control electronics (MCE) (Refer to figure 5)
- Diode shunt protection units (DPSUs) (Refer to figure 6)

PRIMARY FLIGHT CONTROL COMPUTERS

PFCC 1 and PFCC 3 are located in the forward equipment bay. PFCC 2 is located in the mid equipment bay.

INCEPTOR INTERFACE MODULES

IIM 1 and IIM 3 are located in the forward equipment bay. IIM 2 is located in the mid equipment bay.

ALTERNATE FLIGHT CONTROL UNIT

The AFCU is located in the mid equipment bay.

ATTITUDE HEADING COMPUTER

The AHC is located in the forward equipment bay.

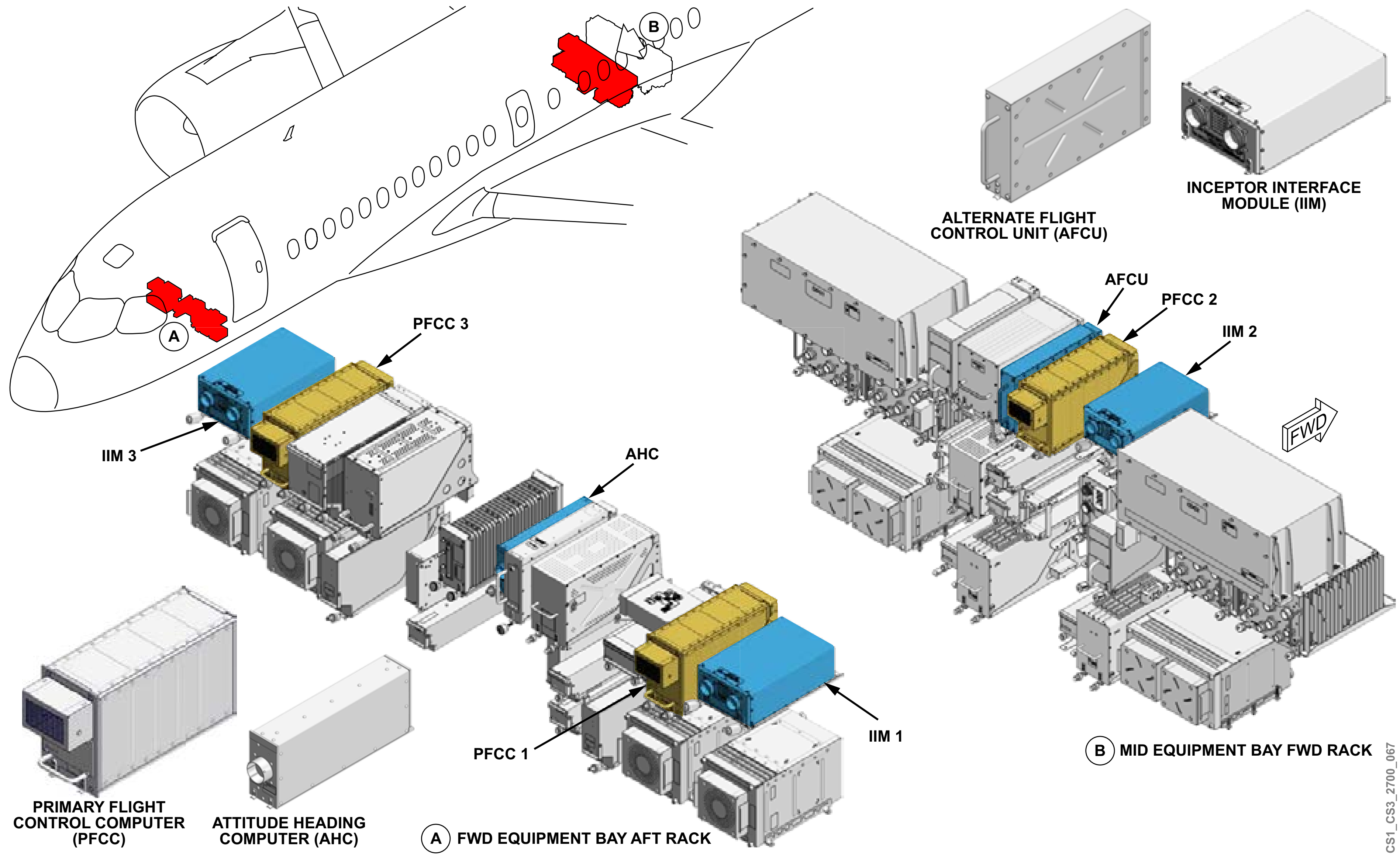


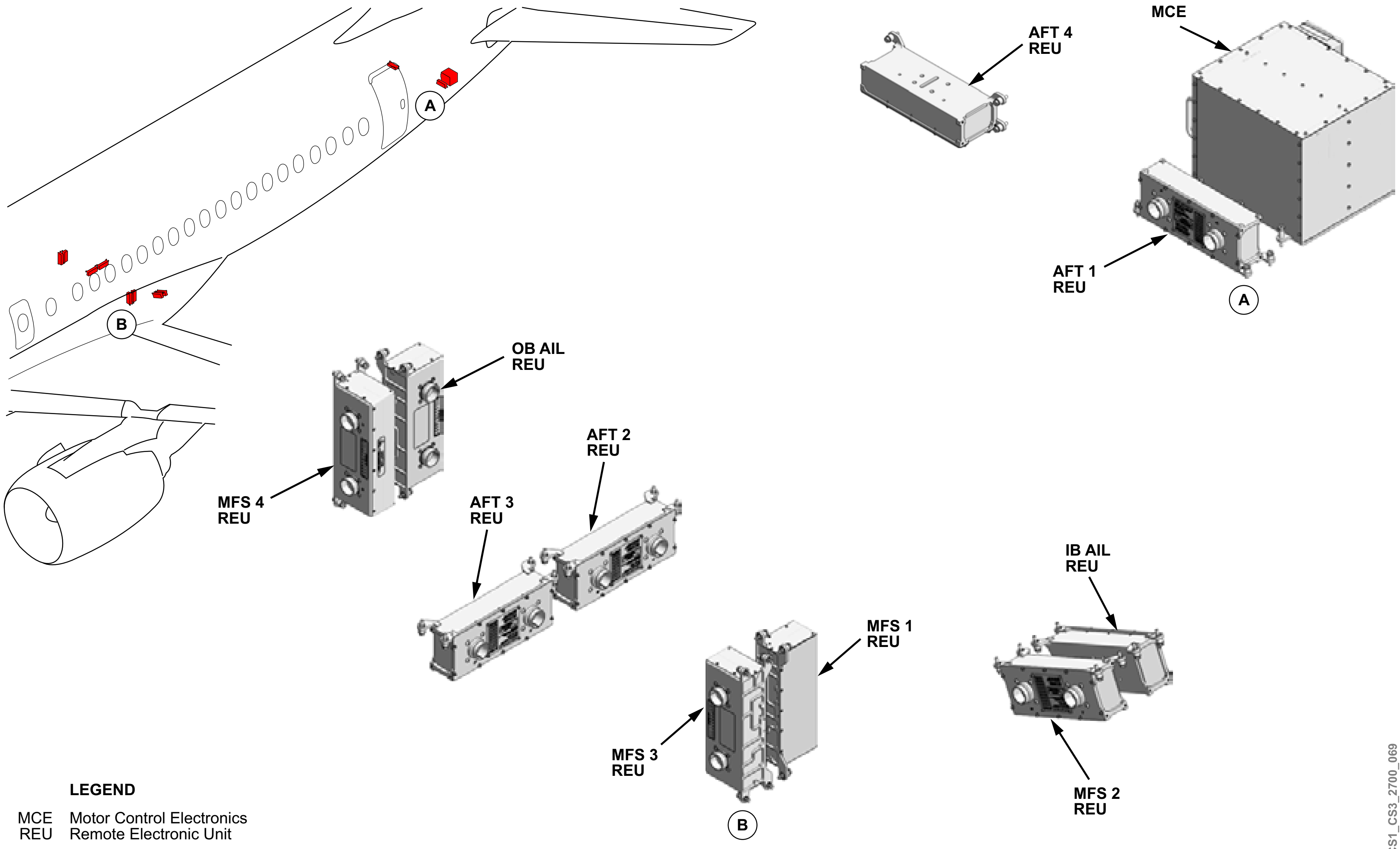
Figure 4: EFCS Component Location

REMOTE ELECTRONIC UNITS

The REUs are located in the mid equipment bay, wing-to-body fairing (WTBF), and the aft equipment compartment.

MOTOR CONTROL ELECTRONICS

The MCEs are located in the aft equipment compartment.



LEGEND

MCE Motor Control Electronics
REU Remote Electronic Unit

Figure 5: Remote Electronic Units and Motor Control Electronics Location

DIODE SHUNT PROTECTION UNITS

The diode shunt protection units (DSPUs) are located in the mid equipment bay and the aft equipment compartment.

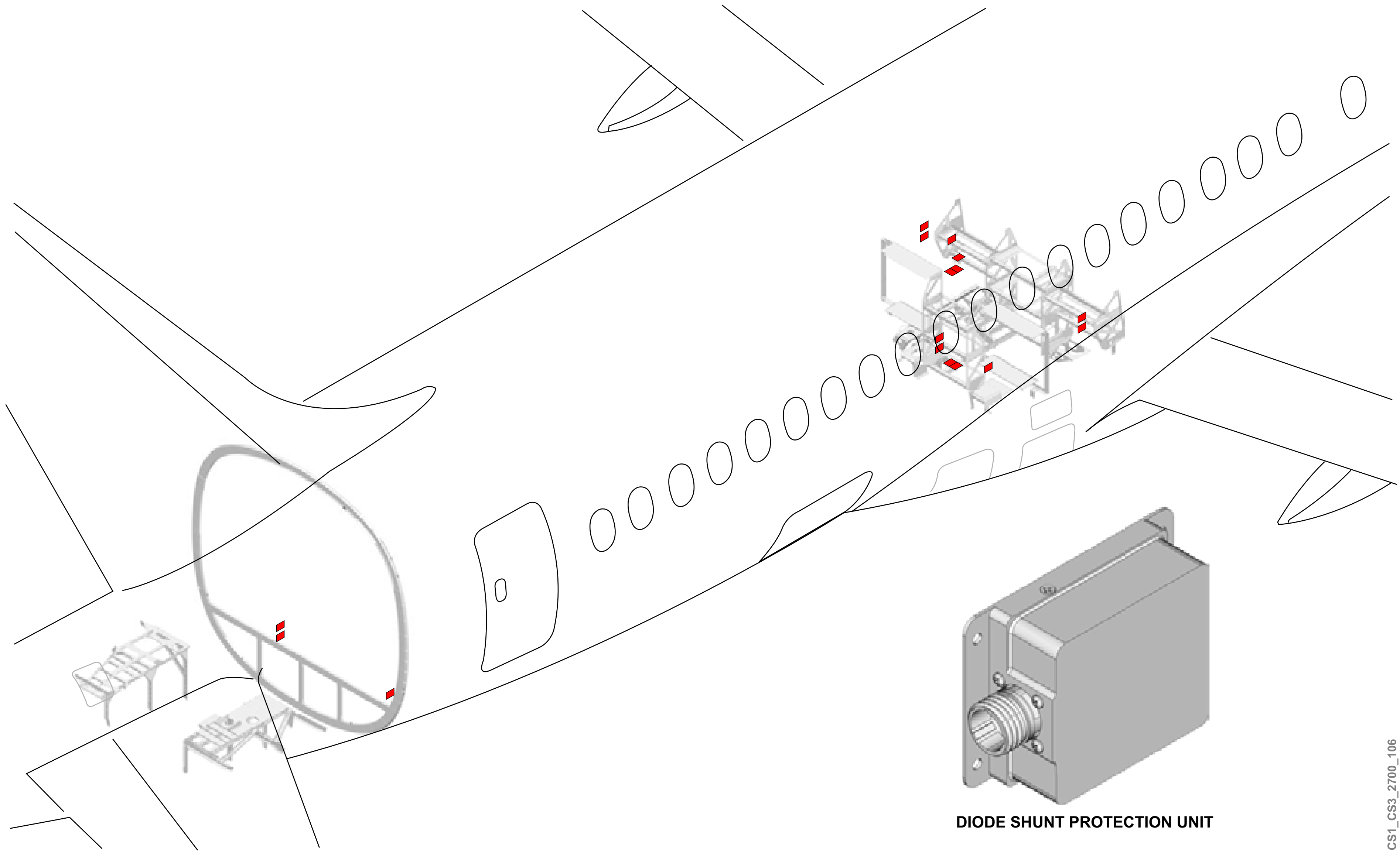


Figure 6: Diode Shunt Protection Units Location

COMPONENT INFORMATION

SIDESTICK CONTROLLER

The sidestick controllers (SSCs) are the electromechanical control devices (inceptors) for the pilot and copilot inputs for roll and pitch control. The SSCs operate independent of each other.

Each SSC consists of the following major components:

- Body
- Grip handle
- Bellows

The left and right SSCs are installed on the side consoles and are not interchangeable. A locator pin on the SSC body prevents incorrect installation.

Control inputs are made by moving the grip handle in the direction of flight. The grip handle has the following control switches installed:

- AP/PTY switch
- Pitch trim switch
- PTT/INT switch

The AP/PTY (autopilot/priority) switch performs two functions. If the autopilot (AP) is engaged, pressing the switch disconnects the autopilot (AP).

If the autopilot is not engaged, pressing and holding the switch provides the SSC with full control of the aircraft, and the inputs from the opposite SSC are ignored by the fly-by-wire (FBW) system.

The pitch trim switch is used to set a reference trim speed in normal mode or to manually trim the horizontal stabilizer in direct mode.

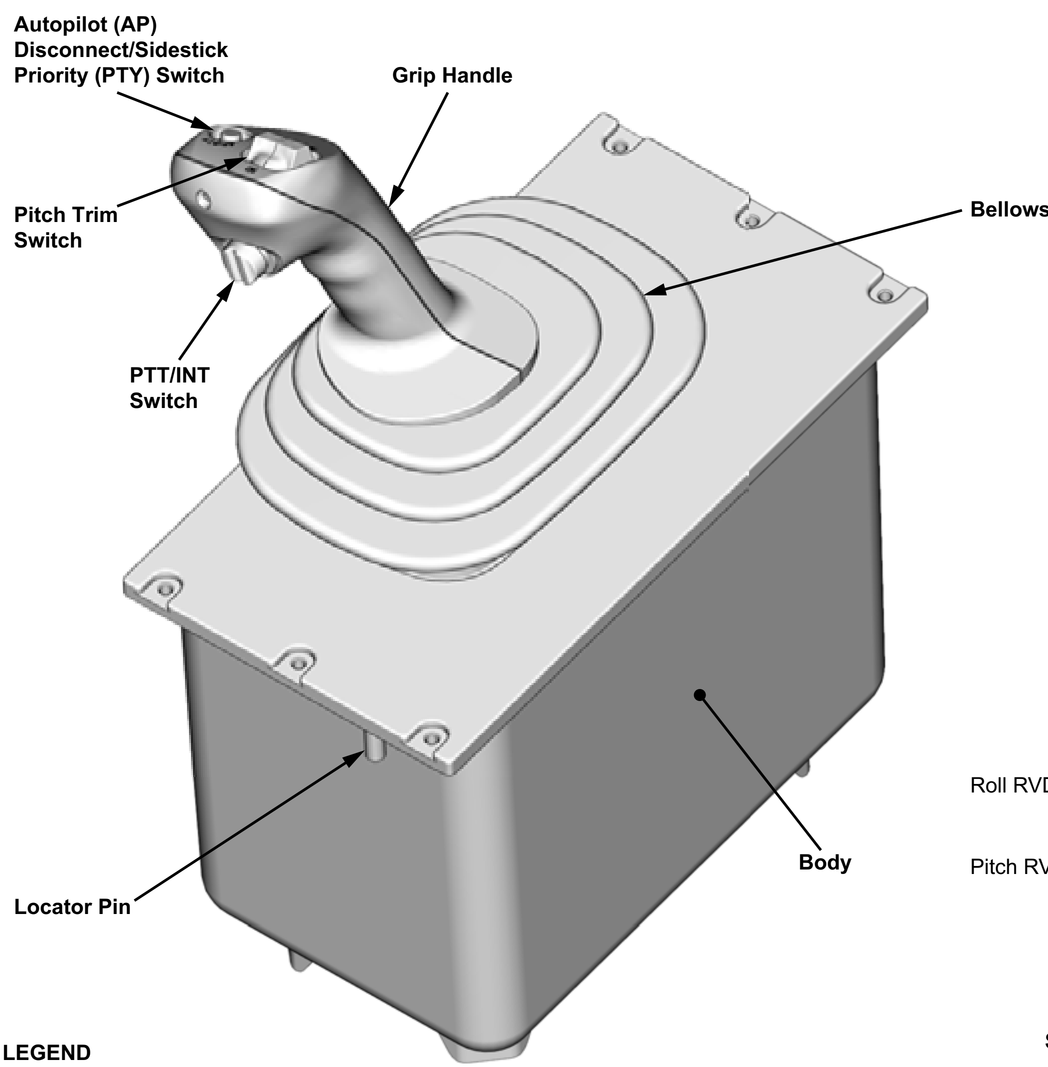
The two position PTT/INT (push-to-talk/interphone) switch provides radio or intercom communication. Refer to ATA 23 for additional information.

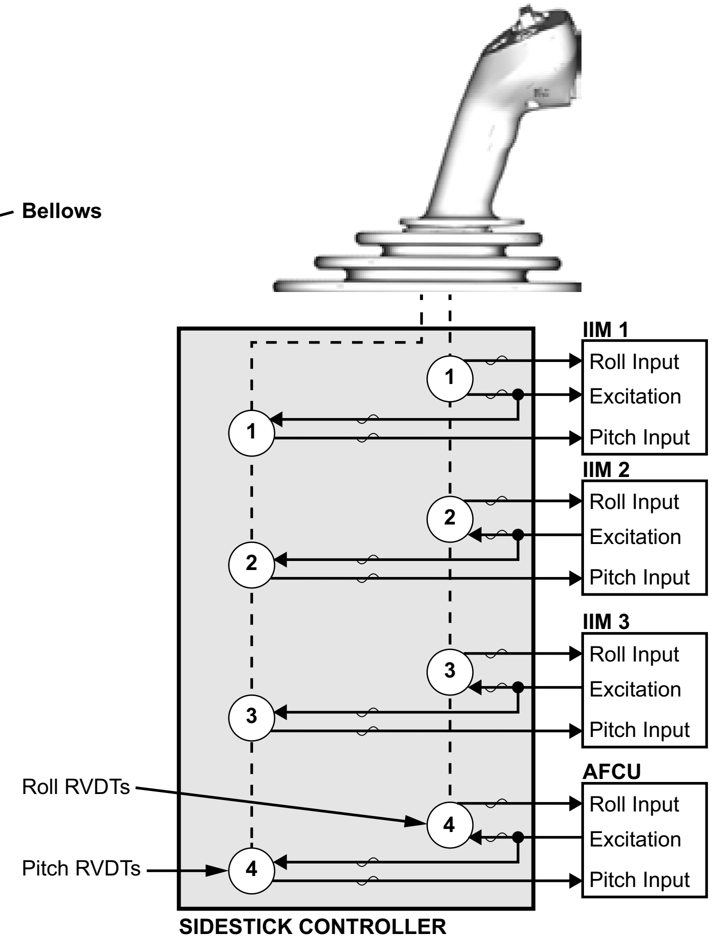
A removeable bellows prevents foreign object debris (FOD) from entering the SSC.

Interface

Sidestick position is measured by eight independent rotary variable differential transformers (RVDTs), four for each pitch axis and four for the roll axis. The SSC RVDTs receive excitation from, and are monitored by, the inceptor interface modules (IIMs) and the alternate flight control unit (AFCU).

Operation of the SSC is monitored by the inceptor interface module (IIM). Faults are reported to the engine indication and crew alerting system (EICAS) and to the onboard maintenance system (OMS).





SIDESTICK CONTROLLER ELECTRICAL INTERFACE

LEGEND

PTT/INT	Push-to-Talk/Interphone
RVDT	Rotary Variable Differential Transformer

CS1_CS3_2700_089

Figure 7: Sidestick Controller

FLIGHT SPOILER CONTROL LEVER

The flight spoiler control lever (FSCL) is used to deploy the multifunction spoiler (MFS) panels symmetrically during flight or upon landing. The angle of panel displacement is proportional to the FSCL position. The speed brake function is applied to dump lift and increase aircraft drag thus creating a steeper angle of descent. It also deploys the ground spoilers (GS) on ground in the direct mode.

The FSCL includes a control lever with a sliding foreign object debris (FOD) curtain, light plate with position indicators and gear-driven, rotary variable differential transformers (RVDTs).

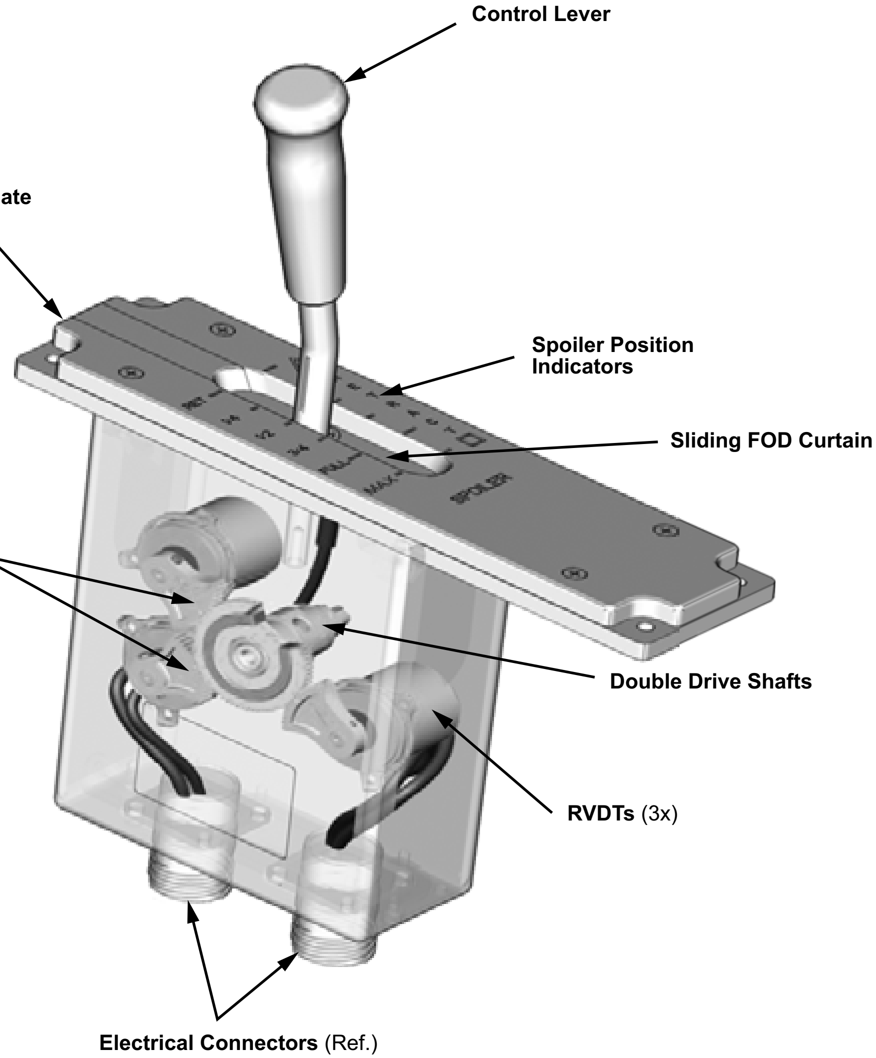
The FSCL has six detented positions:

- Retracted (RET)
- 1/4
- 1/2
- 3/4
- FULL
- Maximum (MAX)

In certain cases the control lever must be lifted in order to move from one position to the next. These are as follows:

- From RET to higher positions
- From FULL to MAX
- From FULL to lower positions

Movement of the lever drives three RVDTs through drive gears mounted around the lever pivot axis.



LEGEND

FOD Foreign Object Debris
RVDT Rotary Variable Differential Transformer

Figure 8: Flight Spoiler Control Lever

RUDDER PEDAL ASSEMBLIES

The rudder pedal system senses yaw and brake commands from the pilot and copilot rudder pedal assemblies and transmits pedal position signals to the EFCS.

The left and right rudder pedal assemblies (RPAs) are similar. The left assembly has the following:

- Pedal feel and centering mechanism (Refer to figure 18)
- Pedal rotary variable differential transformers (RVDTs) (Refer to figure 17)
- Rudder trim assembly (Refer to figure 19)
- Trim RVDTs (Refer to figure 19)

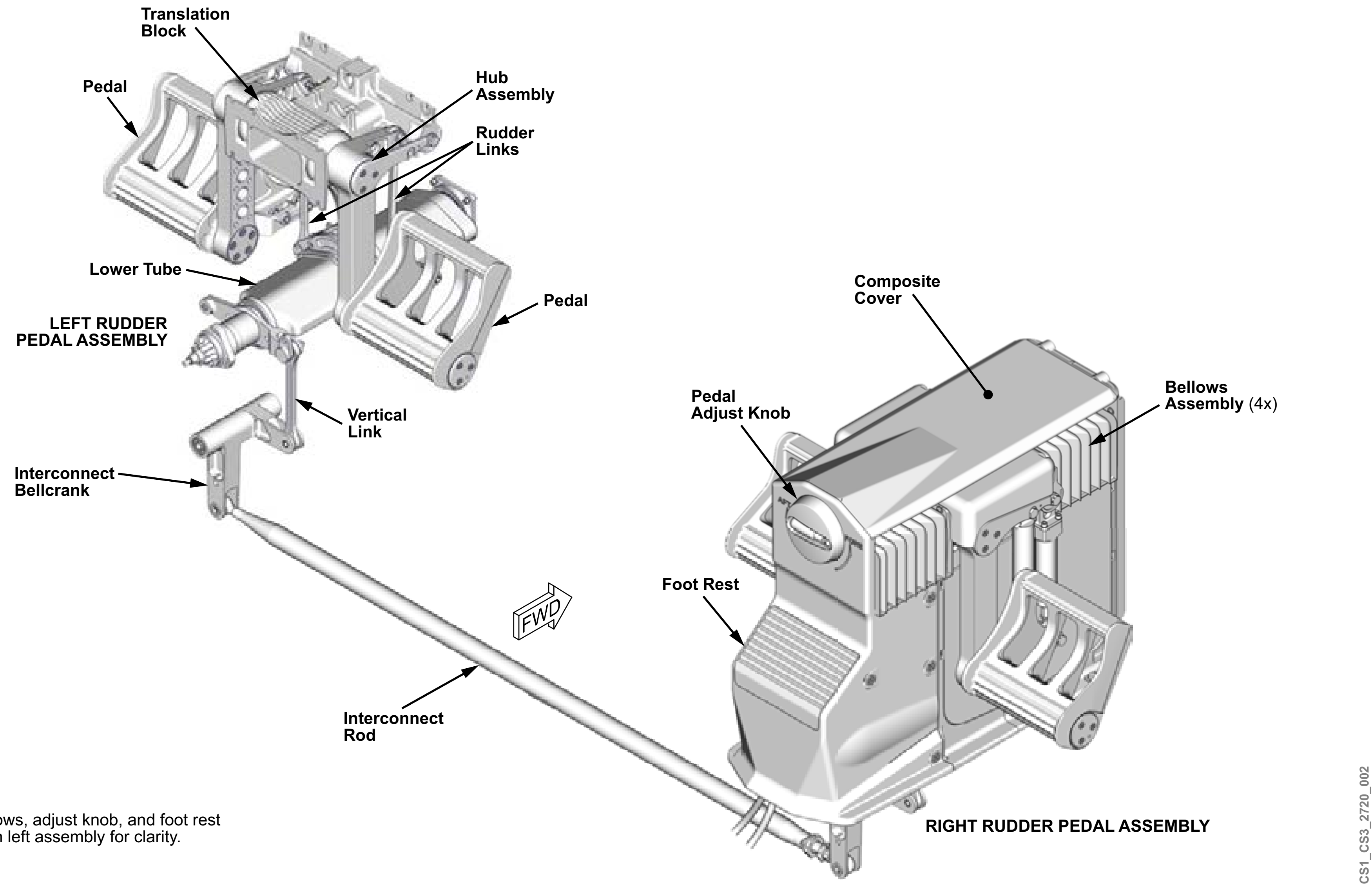
The right RPA has the damper (Refer to figure 20)

The RPAs are of the hanging pedal type. The rudder pedals pivot from a hub assembly that translates above the pedals, suspended by the unit chassis. When the pilot applies force to one pedal, the pedal assembly pivots on the translation block, forcing the rudder links to rotate the lower tube. The rotation of the lower tube is transmitted to the other rudder pedal assembly, through vertical links, interconnect bellcranks and an interconnect rod. This interconnect rod provides pedal feel and centering, and trim capability to the right assembly and damping capability to the left assembly.

Rotation of the pedal adjust knob, moves both pedals fore or aft, to accommodate different pilot leg lengths.

The bellow assemblies prevent foreign object debris (FOD) from entering the adjustment area and protect the RVDT wires while allowing translation block movement. There are four vinyl bellows per assembly.

The composite cover provides FOD protection and improves visual appearance and has a built-in footrest. The cover has to be removed to access the RPA line replaceable units.



NOTE

Cover, bellows, adjust knob, and foot rest removed on left assembly for clarity.

Figure 9: Rudder Pedal Assemblies

PRIMARY FLIGHT CONTROLLER COMPUTER

The primary flight control computer (PFCC) consists of the following independent lanes:

- A command lane
- A monitor lane

Each lane performs identical computations. The outputs from the command lane are compared with those of the monitor lane. If the outputs are identical, the command lane outputs are transmitted over time-triggered protocol (TTP) and ARINC 429 DATA BUSES. If the outputs do not agree, the output of the command lane is inhibited.

The PFCC receives flight deck control inputs through the inceptor interface modules (IIMs). Aircraft systems data, required for flight control functions, including autoflight, is received from the data concentration unit module cabinet (DMC).

The PFCC computes the necessary flight control outputs.

These outputs are modified by control laws (CLAWS) to provide operational limits, flight envelope, and structural protection functions.

Each PFCC hosts the autopilot (AP) and autoland function. The PFCC conditions and monitors flight guidance commands from the AP, and provides AP command limiting and engage/disengage logic.

The PFCCs continually communicate with each other over cross-channel data link (CCDL) to determine which PFCC is active, placing the other two in standby mode.

The PFCC is also used to provide arming and dormancy monitoring of the alternate flight control unit (AFCU). If all three PFCCs are unable to provide EFCS control, the AFCU receives an arming signal from the last available PFCC.

The PFCC can be disabled by pressing the PFCC pushbutton annunciator (PBA) on the primary flight controls panel. This PBA can also be used to reset the PFCC in the event of a fault.

A fan in the PFCC is turned on when the internal temperature reaches 63°C (145°F) and turns off when the temperature cools down to 53°C (127°F).

The fan can also be turned on for test purposes. The fan controller has overcurrent monitoring and fan speed monitoring. The fan is turned off if overcurrent, overspeed, or underspeed conditions exist.

The PFCC also performs built-in test (BIT) and system health monitoring. System status is reported to the engine indication and crew alerting system (EICAS), the onboard maintenance system (OMS), and the aircraft health monitoring system through the data concentration unit module cabinets (DMCs).

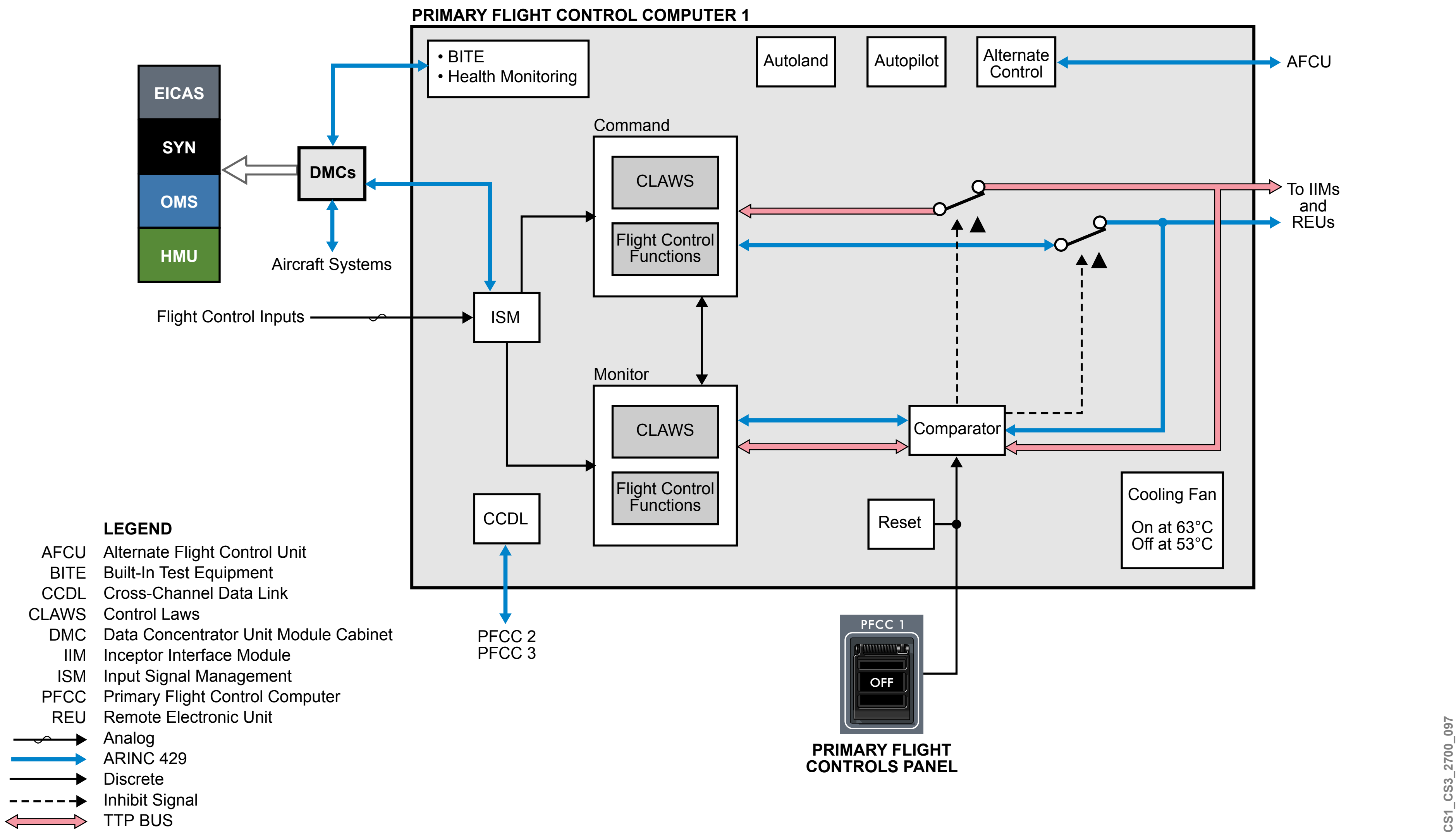


Figure 10: Primary Flight Controller Computer

INCEPTOR INTERFACE MODULE

Inceptor interface modules (IIMs) are data concentrators that provide the interface between the flight control inceptors and the electronic flight control system (EFCS). The IIMs are not interchangeable.

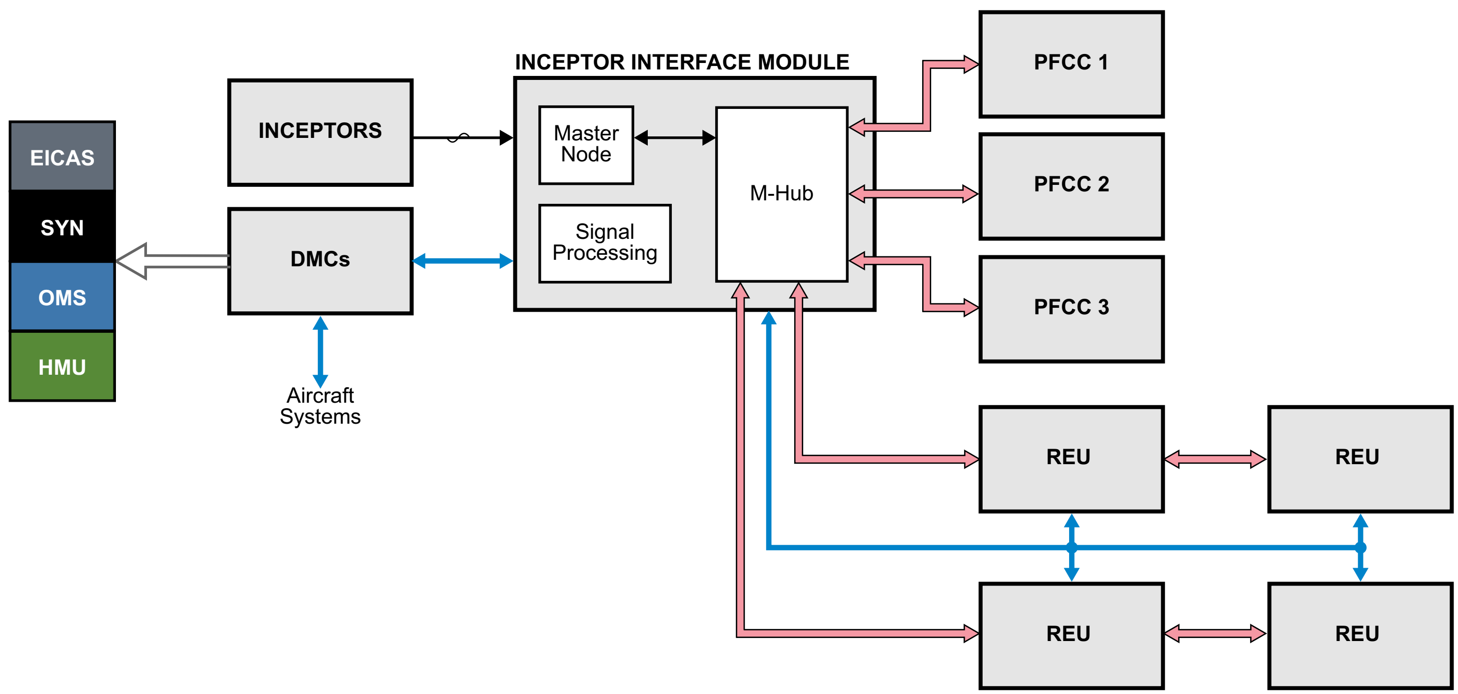
Some aircraft systems also provide inputs to the IIMs through the data concentration unit module cabinets (DMCs).

The IIMs process the flight deck inceptor inputs and distribute the information to the electronic flight control system (EFCS). The primary communication link is the dual channel time-triggered protocol (TTP) BUS. The IIM acts as a five-branch repeater for the TTP BUS.

Each IIM has a hub for one of the three TTP networks. The Mhub combines the TTP routes signals on the TTP BUS from channel 1 to channel 5. The master node synchronizes TTP transmissions from the M-hub.

For REU direct mode, the IIM provides data to the REUs over an ARINC 429 BUS as a backup to the TTP BUS.

Faults are reported to the EICAS and to the OMS through the DMCs.



LEGEND

DMC	Data Concentrator Unit Module Cabinet
PFCC	Primary Flight Control Computer
REU	Remote Electronic Unit
	Analog
	ARINC 429
	Discrete
	TTP BUS

Figure 11: Inceptor Interface Module

REMOTE ELECTRONIC UNIT

The remote electronics unit (REU) is a dual-lane, command/monitor based, actuator controller. Each REU can control up to three actuators. When operating in REU direct mode, the control laws (CLAWS) located in the REU are used for flight control.

The REUs that control the same flight control surface communicate with each other over a cross-channel data link (CCDL) BUS. The CCDL BUS allows the REUs to exchange:

- Mode and engagement status for use in engage logic
- Ram position and actuator pressure for use in monitoring and for the force-fight compensation

The bus is also used to exchange PFCC commands between REU pairs as a redundant source of PFCC command in the case of a TTP failure.

The REUs process control commands from the PFCCs and provide position control of the ground spoiler control module (GSCM), power control units (PCUs), and the motor control electronics (MCE).

Communication with the MCE is over a motor control data link (MCDL) bus.

The REUs interface with the IIMs over the TTP BUS in normal mode or over ARINC 429 in direct mode. The TTP BUS provides the primary interface between the PFCCs, IIMs, and REUs. The TTP DATA BUSES carry all flight deck commands, control surface commands, and actuator feedback between the electronic flight control system (EFCS) line replaceable units (LRUs).

The ARINC 429 BUS carries the input data required for direct mode CLAWS calculations.

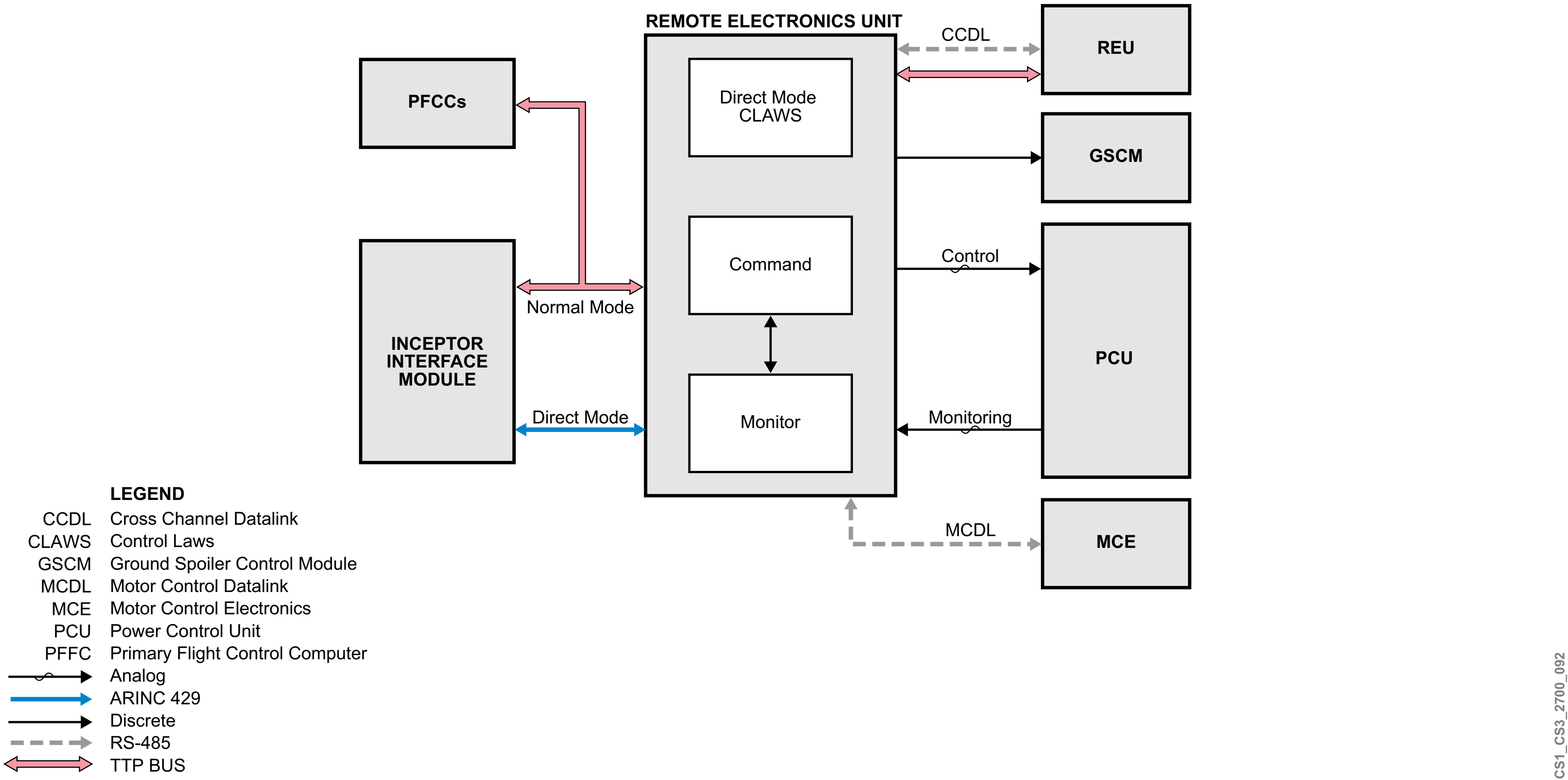


Figure 12: Remote Electronic Unit

ATTITUDE HEADING COMPUTER

The attitude heading computer (AHC) supplies attitude and attitude rate information to the primary flight control computers (PFCCs) to validate inertial reference system (IRS) data. The AHC data is only used by the

The AHC receives two sources of power. The right fly-by-wire power converter (FBWPC) is the primary source and DC BUS 1 is the secondary source.

The AHC mounting tray is a stabilized platform for the AHC. The mounting tray is leveled using shims. The AHC attitude outputs are compared to IRS 1 attitude information during leveling. The information is found on the onboard maintenance system (OMS) LRU/System Operations AHC - Config/Status page.

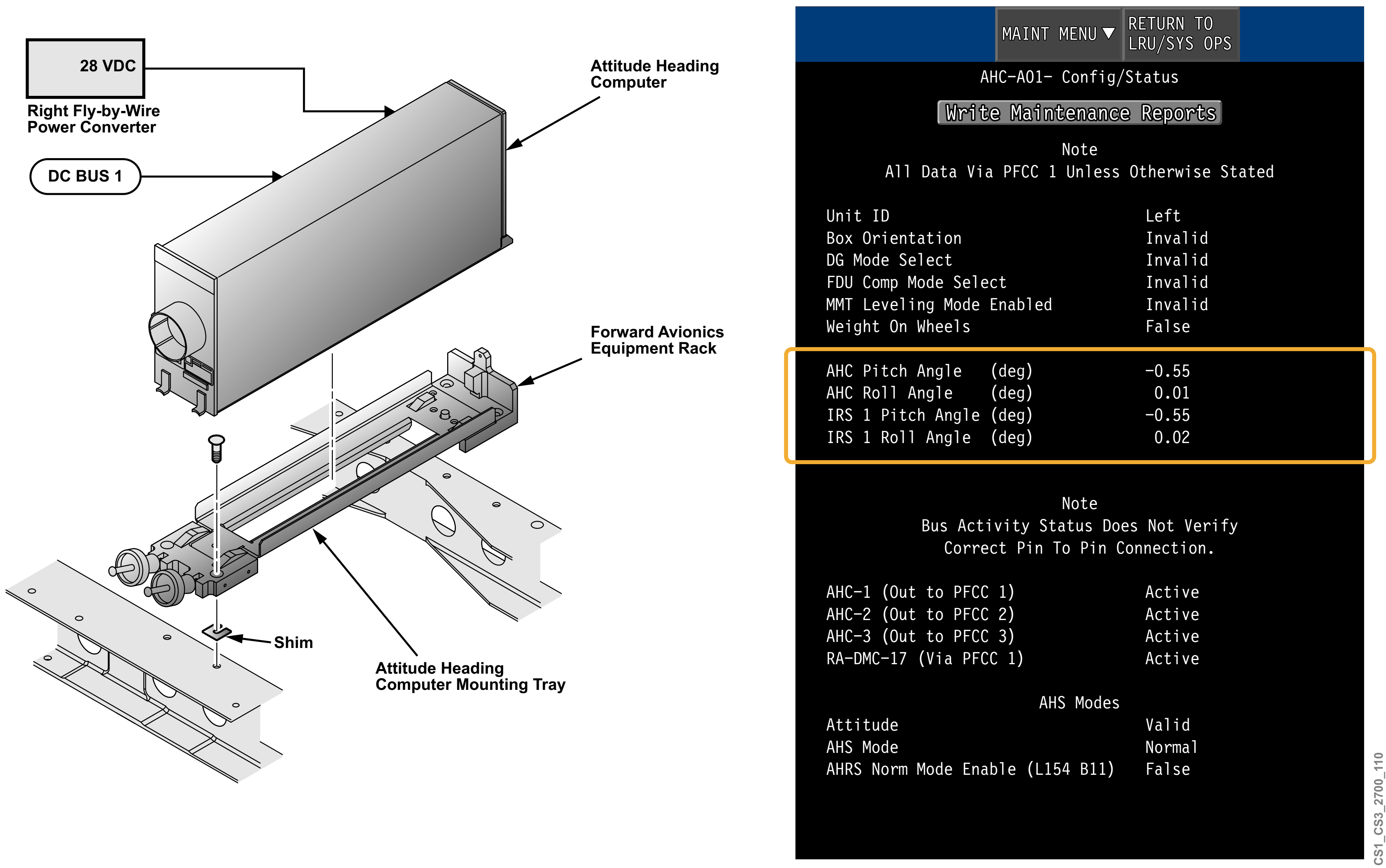


Figure 13: Attitude Heading Computer

ALTERNATE FLIGHT CONTROL UNIT

The alternate flight control unit (AFCU) provides direct control of ailerons, elevators, and rudder PCUs when normal control paths have failed. It is intended to provide a basic flight control system enabling the flight crew to land the aircraft as soon as possible.

The AFCU provides a completely independent control system. It receives inputs from the sidestick controllers, rudder pedals, pitch trim switches, and aileron trim switch.

The AFCU is engaged when the primary flight control computers (PFCCs) and REUs indicate that they can no longer meet the minimum control requirements. The AFCU can be engaged on a per axis basis, meaning that only one axis (for example, roll) can be controlled by the AFCU while the other axes are controlled by the REU or PFCC.

When in the AFCU direct mode, the AFCU directly controls the operation of a single PCU per primary control surface, while the other PCU(s) on that surface are placed into the soft damped state. The AFCU provides the direct control of the horizontal stabilizer by the pitch trim switches through the MCE.

AFCU status is reported to the EICAS, and OMS through the DMCs.

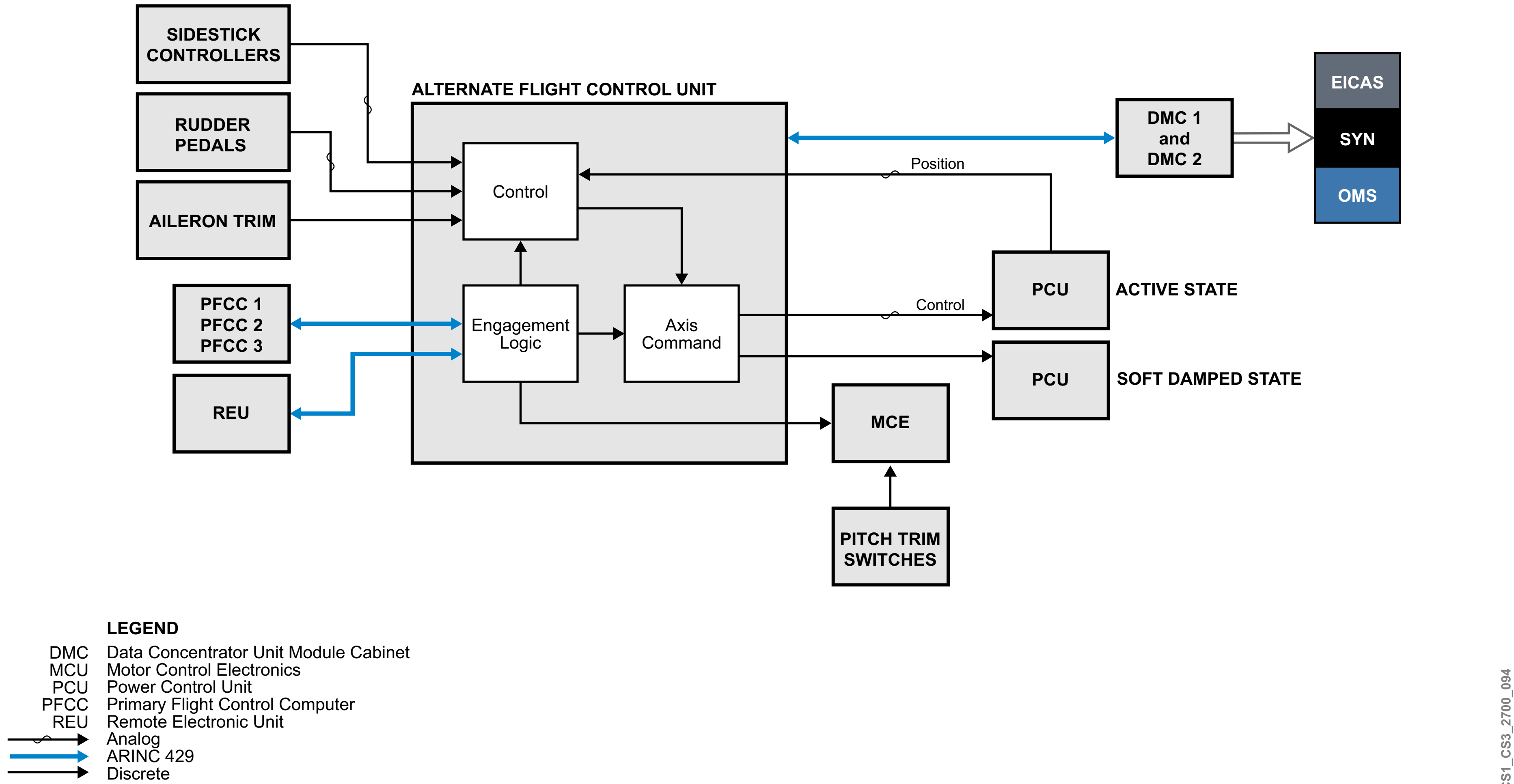


Figure 14: Alternate Flight Control Unit

MOTOR CONTROL ELECTRONICS

The motor control electronics (MCE) provides motor and brake control for the horizontal stabilizer trim actuator (HSTA). The MCE has two channels. Channel 1 is controlled by the aft 1 remote electronic unit (AFT 1 REU). Channel 2 is controlled by the aft 4 REU.

The MCE provides motor and brake commands to the HSTA. In normal mode the trim commands are provided by the PFCCs through the REUs.

If the system reverts to AFCU direct mode, the pilot and copilot trim switches are supplied to MCE channel 1. Channel 1 has an alternate control path that allows direct control of the HSTA by the trim switches.

MCE status is reported through the REUs to the inceptor interface modules (IIMs) which report to the EICAS and OMS.

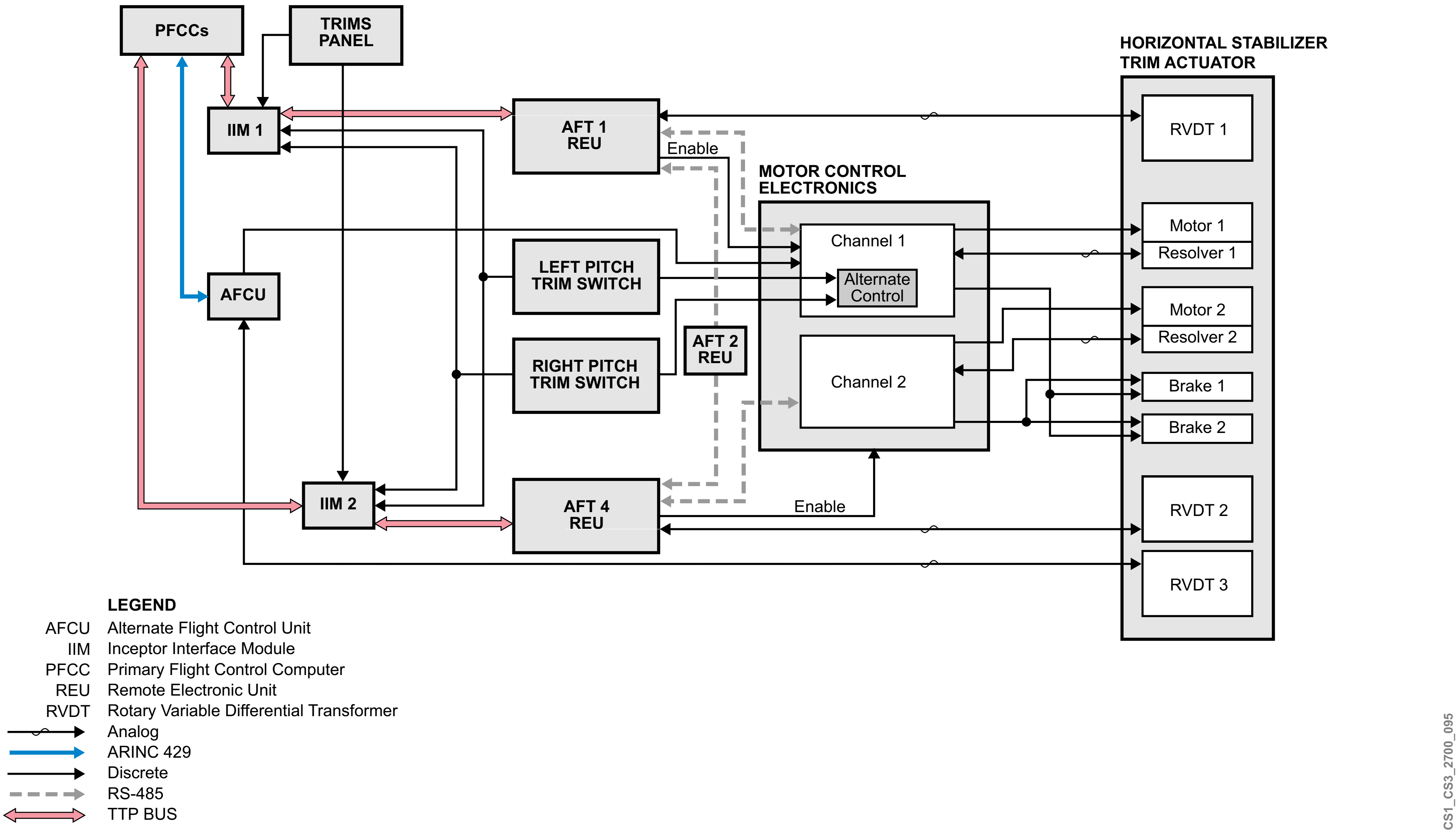


Figure 15: Motor Control Electronics

DETAILED COMPONENT INFORMATION

SIDESTICK CONTROLLER

The sidestick controller (SSC) consists of:

- Grip handle
- Mobile assembly
- Pitch and roll axis assembly

Grip Handle

The grip handle is a line replaceable unit (LRU) that is mounted to the mobile assembly. A DC motor is mounted within the grip handle that acts as a stick shaker, providing a tactile indication of the onset of a stall condition. The grip handle has a connector for the motor and switches.

Mobile Assembly

The mobile assembly is a universal joint assembly that divides combined pitch and roll inputs into independent outputs on the roll and pitch axis. The mobile assembly also includes the end stops for both pitch and roll axes.

Pitch and Roll Axis Assembly

Each pitch and roll axis assembly includes a rotary variable differential transformer (RVDT) plate and axis on which the following four subsystems are installed:

- Tension spring feel assembly
- Torsion spring feel assembly
- RVDT assembly
- Damping and autopilot locking assembly

The tension and torsion spring feel assemblies provide a feel force for each axis.

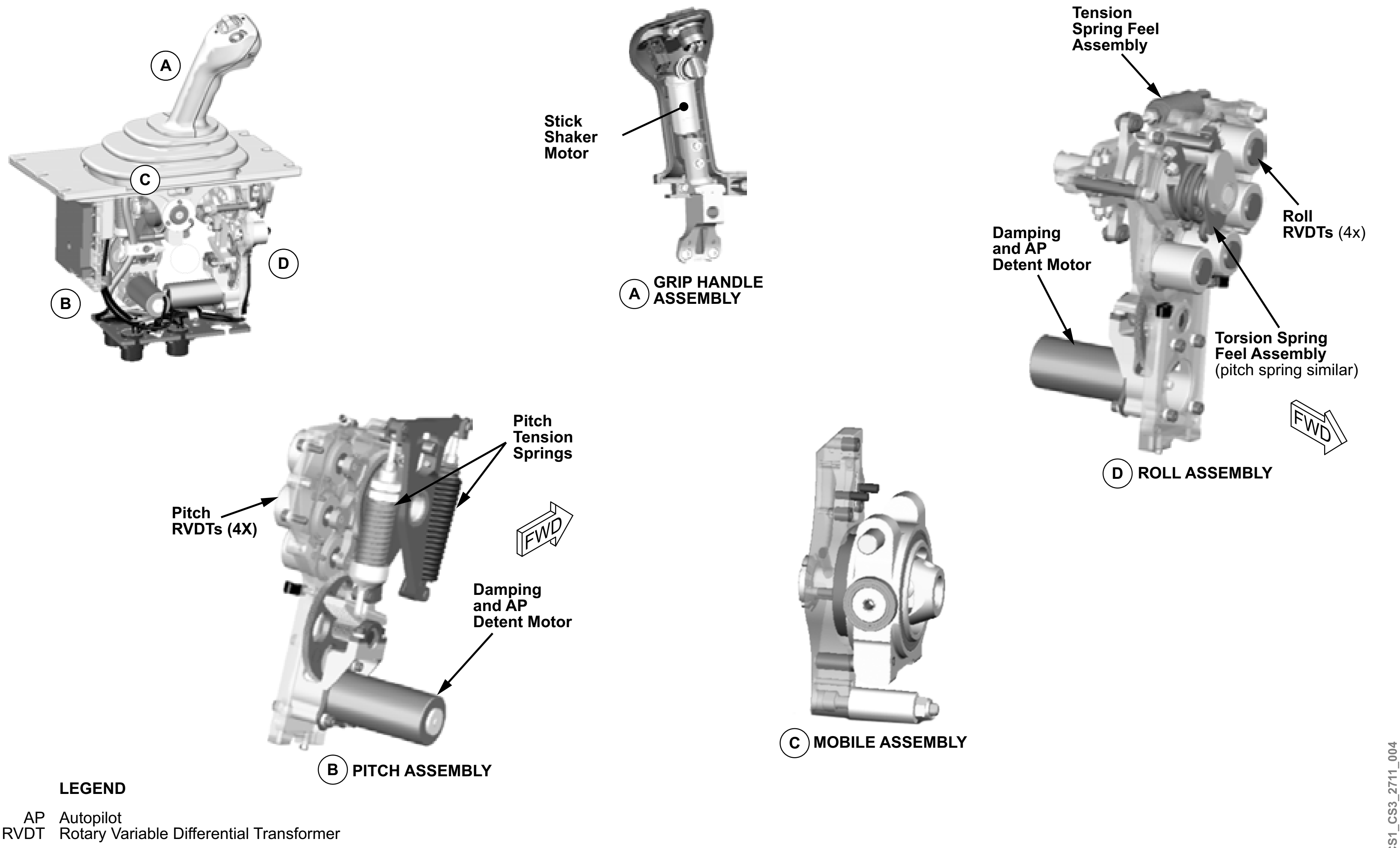
The RVDT assembly has four RVDTs that are driven by a master gear.

As a tactile cue that the operating limits have been reached, the SSC provides a soft stop in the pitch axis. The SSC can be moved past the soft stop until it contacts the hard stop using an increased pitch force. The hard stop indicates the limits for structural protection.

Each axis assembly has a DC motor and gear train. The axis assembly is used to generate a variable damping force, or an autopilot (AP) detent force at the neutral position when the AP is engaged.

For damping, the DC motor acts as a generator to produce a back electromotive force proportional to grip speed input in either the pitch or roll direction.

For AP detent, the same DC motor is powered on 2 of 3 coils, generating a resistive force at the locked position.



LEGEND

AP Autopilot
RVDT Rotary Variable Differential Transformer

Figure 16: Sidestick Controller Details

RUDDER PEDAL ASSEMBLIES

Rudder Pedal Rotary Variable Differential Transformers

Rudder pedal position is measured by four segregated RVDTs. In normal mode or PFCC direct mode, the PFCC establishes a consolidated rudder pedal position using three of the four RVDTs. In normal mode and PFCC direct mode, the position is considered lost whenever two RVDT are lost, however in REU direct mode, the remaining RVDT can be used. There is also a remaining dedicated RVDT that interfaces directly with the AFCU allowing an alternate source of rudder control if minimum control requirements (MCR) is lost and the AFCU is engaged. They are mounted on two different gears.

Rudder Pedal Feel and Centering Mechanism

The feel and centering mechanism provides the interface with the rudder and trim RVDTs. It has dual-load paths for redundancy purposes. The trim function also has a primary and secondary load path, using two separate worm wheels for redundancy. The centering is done by a cam and a roller. Dual-springs are used to create the force required to center the roller in the cam.

When the pilot moves the pedals, the centering cam (attached to the lower tube) forces a rocker arm to compress the dual-springs which create the feel force (60 lb max). The feel and centering mechanism uses a brake force feel mechanism, which provides feedback when brakes are applied. The feel force is transmitted to the right RPA through a tube assembly.

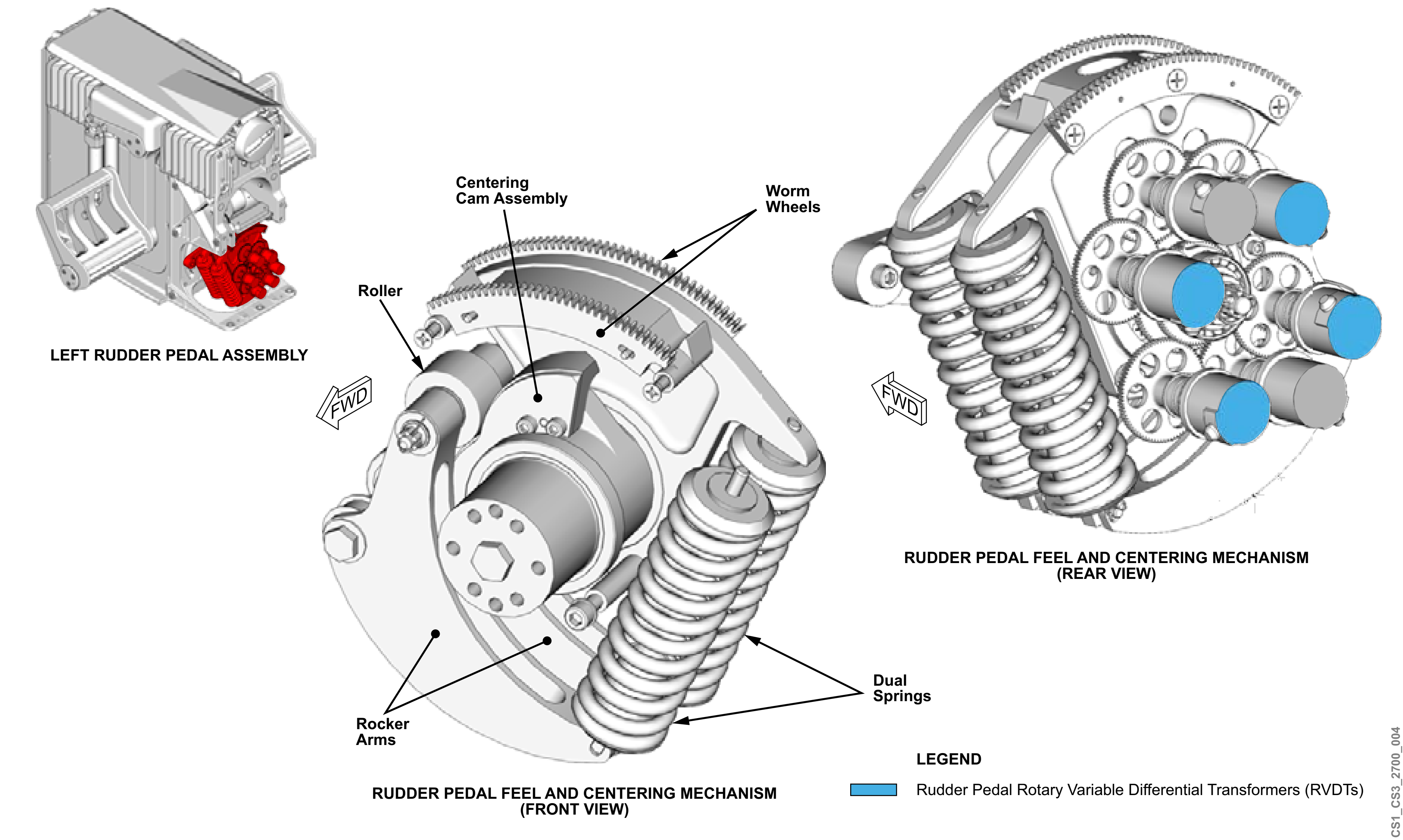
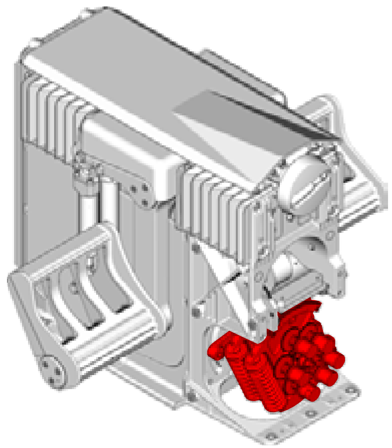


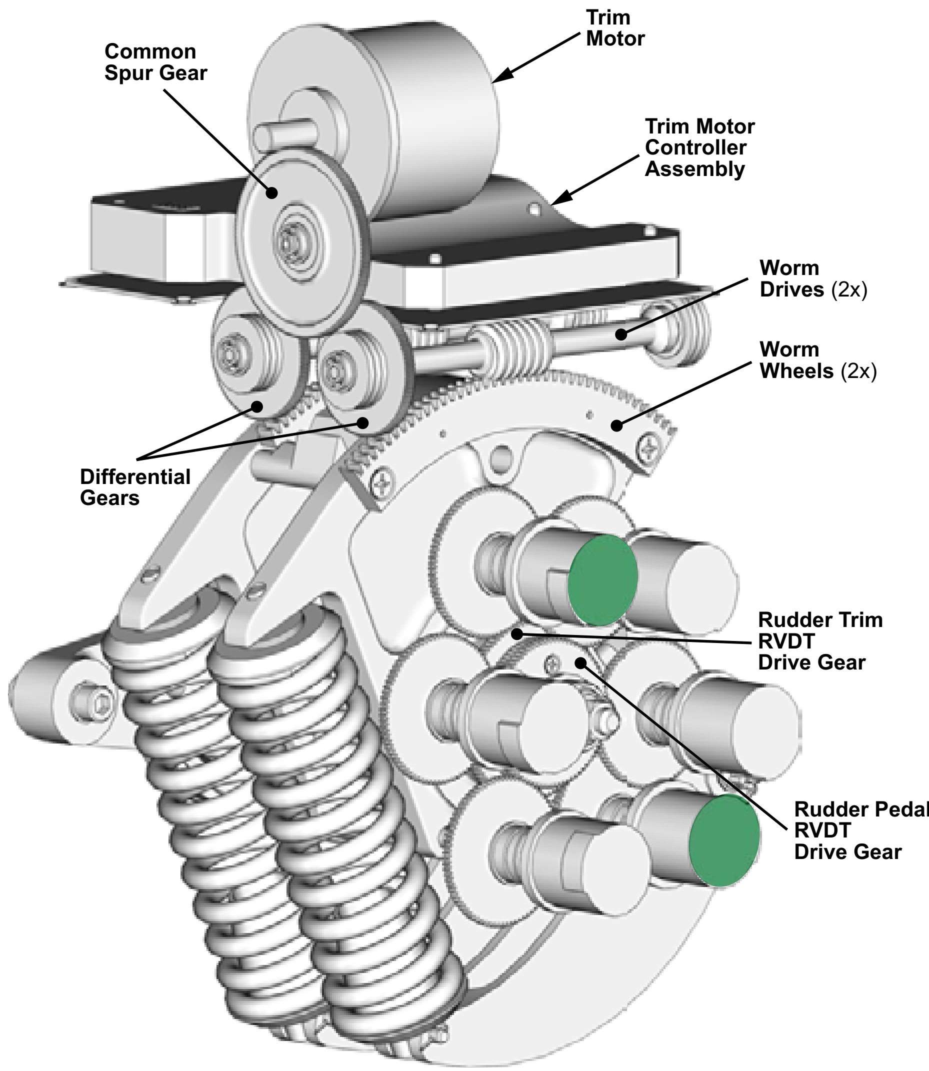
Figure 17: Left Rudder Pedal Assembly Components

Rudder Trim

Selection of the rudder trim switch sends pedal adjustment commands to the trim motor controller assembly. The controller operates the trim motor, which, through a gearing arrangement, rotates the feel and centering mechanism, back-driving the rudder pedals to a new position. The trim motor is powered by DC BUS 1.

Two rudder trim RVDTs provide the rudder trim information. They are mounted on a separate gear from the rudder RVDTs.





LEGEND

Rudder Trim RVDT
RVDT Rotary Variable Differential Transformer

Figure 18: Left Rudder Pedal Assembly Rudder Trim Components

Right Rudder Pedal Assembly Differences

The right RPA provides the damping capabilities through a damper located at the end of the lower tube, rather than a feel and centering mechanism used on the left RPA. The damping mechanism uses the internal porting of viscous oil to produce resistance and improve pilot feel.

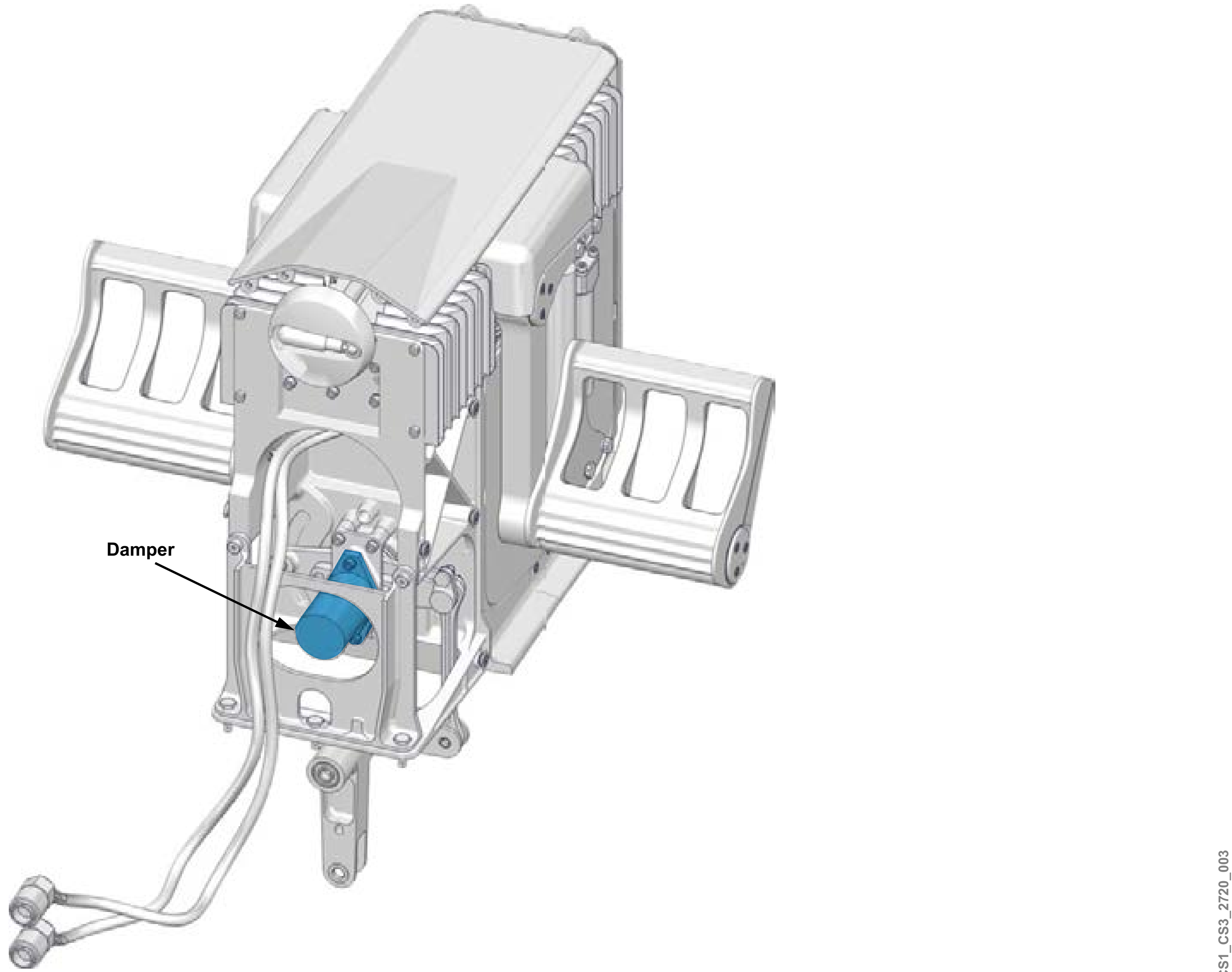


Figure 19: Right Rudder Pedal Assembly

PRIMARY FLIGHT CONTROL COMPUTERS

The PFCCs are comprised of two physically segregated, computational lanes; a command lane and a monitor lane. Each of the independent lanes is comprised of input-output functions that make up the I/O functional block, and a central processing unit (CPU) function that makes up the processing functional block. Each lane performs identical computations using identical data provided by ARINC 429 and TTP DATA BUSES, as well as identical analog and discrete inputs to both lanes.

The TTP outputs are sent to the REUs and IIMs. For invalid command lane and monitor lane computations (i.e. computations disagree), the comparator function disables the outputs from the PFCC by opening the switch in the output zone functional block. To achieve high integrity output, the PFCC monitor comparisons allow for rapid detection of failure conditions. Synchronization circuitry is used in each lane so that the comparator function performs comparisons on the same data from the two lanes.

System integrity within the PFCC monitoring architecture is achieved through the comparator function continuously testing itself. The comparator function and the output switches are also tested at power up.

Continuous self-test is performed by the comparator function to verify that it functions correctly. Also, to verify that the output switches operate properly, a functional test is performed by commanding one of the lanes to output an erroneous command. The comparator function is then expected to detect a miscompare between the two lanes and open the switches in the comparator function output path.

The PFCCs process the inceptors and avionics system inputs to compute digital control surface actuator commands. The three PFCCs exchange information cross-channel to support mode consolidation and equalization functions.

Each PFCC operates in either a command or standby role with identical processing. The PFCCs inform the REUs which of them is in command based upon their availability and validity. Each PFCC receives data from multiple buses (TTP or ARINC 429) for computation of stability augmentation and control laws.

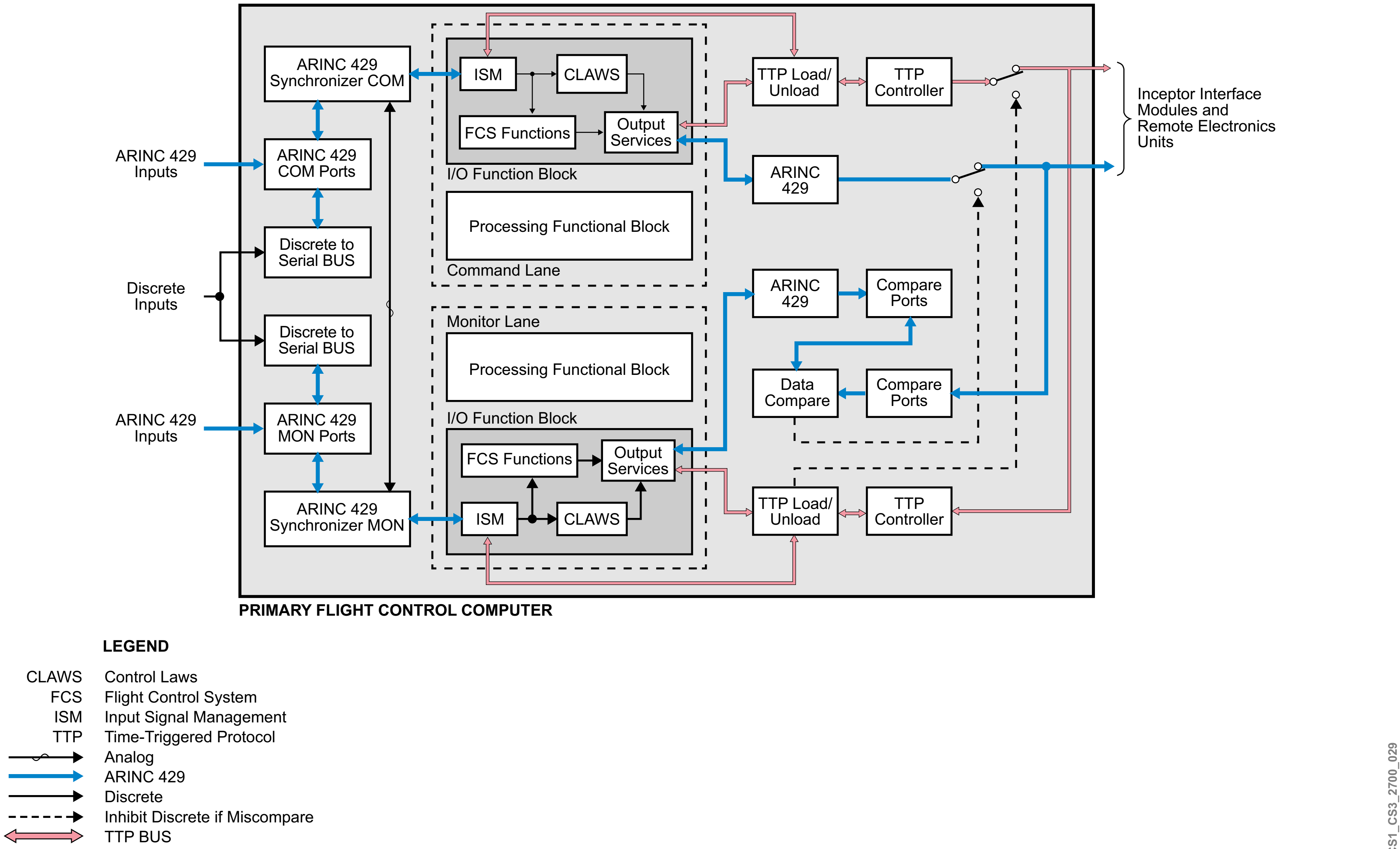


Figure 20: Primary Flight Control Computer

INCEPTOR INTERFACE MODULE

The inceptor interface module (IIM) is the primary interface between the cockpit controls and the flight control system. It is a digitizer and data concentrator for the pilot controls and flight compartment switches. There are three different IIM types used, defined as IIM 1, 2, and 3. Each type uses dissimilar complex hardware. They have dissimilar processing element hardware and firmware functions to guarantee detection and minimize generic failures.

Each IIM type has a unique mounting footprint and unique connector keying. The IIM has discrete configuration ID pins, which allows the firmware to detect the location in which the IIM is installed. The IIMs have only one operating mode. The IIMs operate the same way in normal and direct mode.

Functions of the IIM include:

- Data acquisition
- Excitation and demodulation of control sensors:
    - Pilot sidestick RVDTs
    - Copilot sidestick RVDTs
    - Rudder pedal RVDTs
    - Rudder trim RVDTs
    - Flight spoiler control lever (FSCL) RVDTs
- Digitizing aileron and pitch trim switch discrete signals
- Sensor data concentration for transmission on TTP BUSES
- Hosting of TTP BUS
- Transmission of sensor data to REUs via ARINC 429 direct path buses
- Transmission via ARINC 429 data to the data concentrator unit module cabinets (DMCs) for EICAS, flight controls synoptic page and the onboard maintenance system (OMS)
- Actuation of tactile warning and autopilot detent for pilot and copilot sidestick controllers

The IIM reports sensor and switch data to the PFCCs and REUs, which perform voting, rigging, and additional processing. In normal mode, the PFCCs use the data from the IIM. In direct mode, the REUs act upon this data.

For flight control system communications, the primary communication link is the dual-channel time-triggered protocol (TTP) BUS. The IIM serves as a five-branch repeater for the TTP BUS.

The IIM uses air data information from the ADSPs to determine the stick shaker activation point.

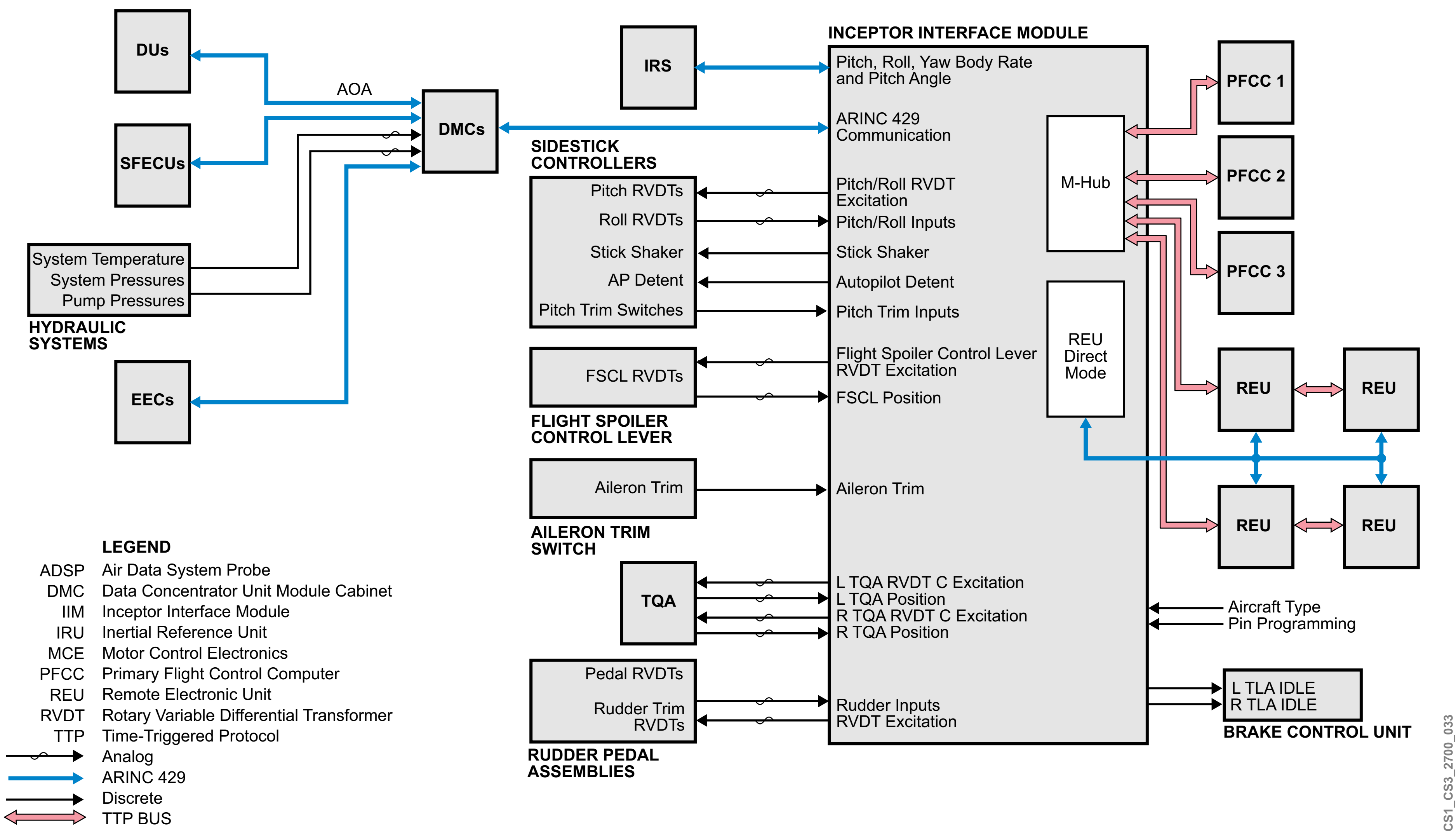


Figure 21: Inceptor Interface Module

REMOTE ELECTRONIC UNIT

The principle function of the remote electronic unit (REU) is to provide closed loop control of the ailerons, elevators, rudder, multifunction and ground spoilers, and the horizontal stabilizer. The REU also provides fault monitoring, status communication, maintenance support, and rigging. The REU is a dual-lane command-monitor based actuator controller. The REU provides actuator loop closure and mode control for up to three actuation channels and has the capability to adapt its functionality depending on its installation location.

The REU operates in the normal and PFCC direct mode when at least one primary flight control computer (PFCC) is operational and providing control surface commands to the REU. The REU transitions to the REU direct mode using its own control laws (CLAWS) if the REU is not receiving commands from a PFCC, but is receiving commands from the IIM either through the TTP BUS, or ARINC 429 DATA BUSES.

The REU receives commands from the PFCCs through the IIMs (normal mode and PFCC direct mode) or the IIMs (direct mode). Primary and secondary communications are done through the TTP BUS. The ARINC 429 BUS allows the IIMs to communicate with the REUs in the event that TTP BUS is no longer available for use.

The motor control data link (MCDL) interface provides communication between the REUs and the motor control electronics (MCE).

The cross-channel data bus (CCDL) interfaces provide communication between the two REUs controlling each pair of actuators on each surface.

The command lane can exchange data with the monitor lane over its serial port command monitor data link (CMDL).

The maintenance interlocks are satisfied when the aircraft is on the ground and stationary. When the conditions are met, the REU can be put in rigging mode or data loading to allow maintenance to be performed.

Several components can be controlled at one time by the REU. Each REU supplies DC excitation for the PCU actuator pressure monitoring and AC excitation for the linear variable differential transformers (LVDTs). The LVDTs provide position feedback to the command and monitor lanes.

Both the solenoid valves (SVs) and the electrohydraulic servovalves (EHSVs) are controlled by the command and monitoring channels. The command channel controls the power and the monitor channel provides the ground to the driver.

The REU has six high integrity, low power discrete outputs for providing discrete logic signals to external LRUs.

The REU can be powered by two independent power sources of 28 VDC. REUs that are critical to maintaining the minimum control requirements (MCR) use dual power sources.

Processing in the normal mode (PFCC state) consists of closed servo-loop control of the surface using the surface position command transmitted by the PFCC. Surface control in the normal mode consists of a position loop with the inputs of PFCC surface position command and the surface LVDT position feedback.

Processing in the REU direct mode consists of calculating the surface position command using the pitch, roll, and yaw angles, as well as trim switch commands inputs from the IIM. The position loop closure is performed by using the calculated surface position command and the surface LVDT position feedback.

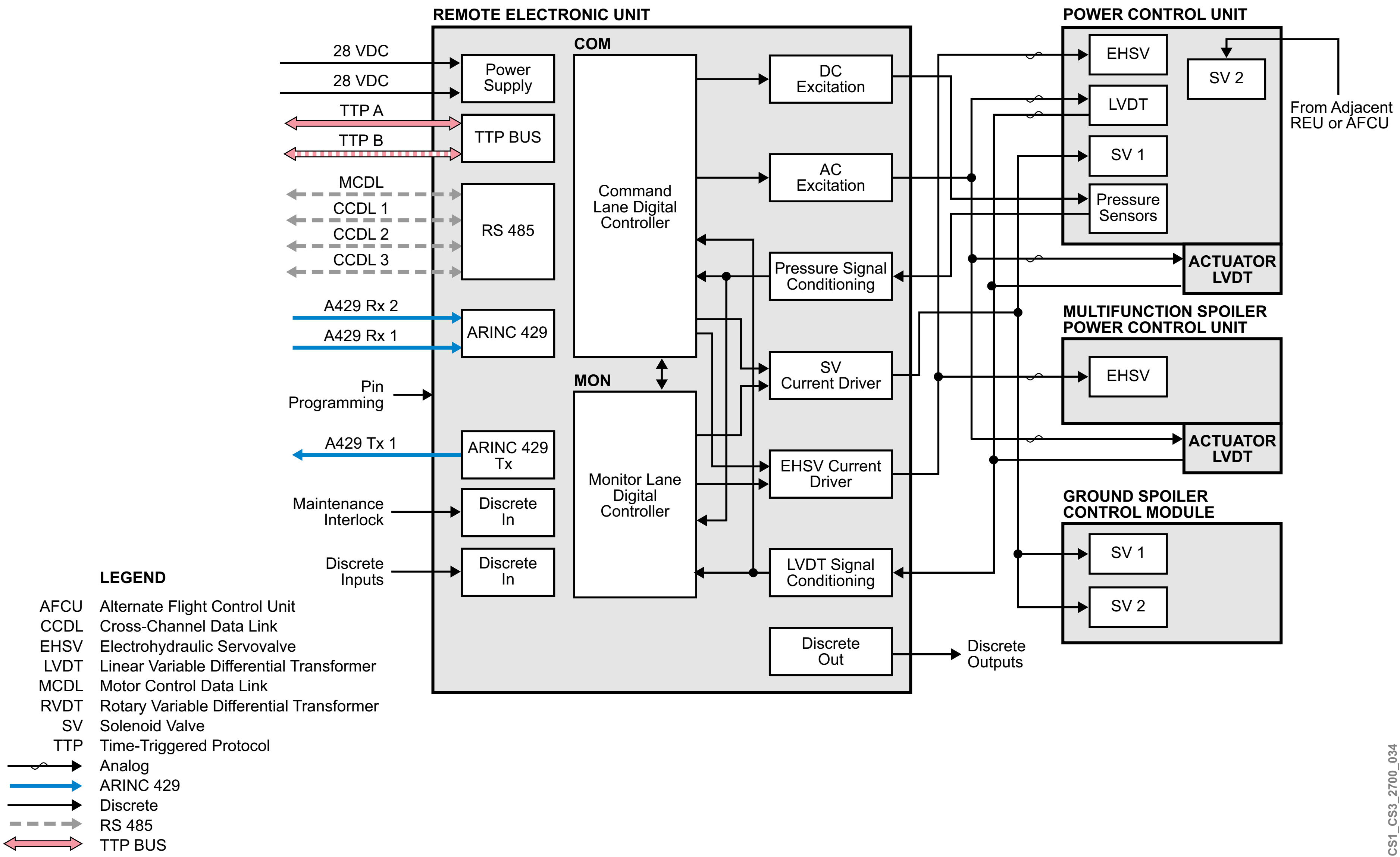


Figure 22: Remote Electronic Unit

ALTERNATE FLIGHT CONTROL UNIT

In the event that the primary flight control computers (PFCCs) or the remote electronic units (REUs) are unable to provide minimum control requirements on a particular flight axis, the system reverts to alternate flight control unit (AFCU) direct mode. This mode is more accurately a command-by-wire mode, as the pilot inputs are sent directly to the control surfaces through the AFCU. The inputs are not processed by the PFCCs or the REUs. AFCU direct mode provides degraded but sufficient control of the aircraft flight controls. Reversion to AFCU direct is considered an abnormal event and requires landing at the nearest suitable airport.

Flight control power control units (PCUs) can be directly controlled by the AFCU. The AFCU receives arm commands from the PFCCs and enable signals from the REUs. The AFCU receives commands from the independent set of sensors on each inceptor (sidesticks and rudder pedals), and from the aileron trim switch. In addition to PCU control, the AFCU also controls the horizontal stabilizer actuator through one MCE channel.

The AFCU controls PCUs via analog interfaces. To prevent force fight between PCUs on the same surface, the adjacent PCU is set to soft damp mode. Position feedback is achieved by monitoring the PCU position.

The AFCU controls the following PCUs in the AFCU direct mode:

- Left and right aileron inboard PCUs
- Left and right elevator inboard PCUs
- Rudder lower PCU

The AFCU provides soft damping commands to the following PCUs in the AFCU direct mode:

- Left and right aileron outboard PCUs
- Left and right elevator outboard PCUs
- Middle and upper rudder PCUs

Roll assist is disabled in the AFCU direct mode.

The AFCU enables the horizontal stabilizer alternate control state of operation in which the stabilizer can be operated directly by the flight crew. The pitch trim switches are interfaced directly to the motor control electronics (MCE) channel 1, which drives the HSTA motor.

The AFCU interfaces with the following systems:

- DMC - slat/flap position data
- SFIS - pitch, roll, yaw rate, and pitch attitude (IRS data)
- PFCC - engagement status, monitoring and power-up built-in test (PBIT)

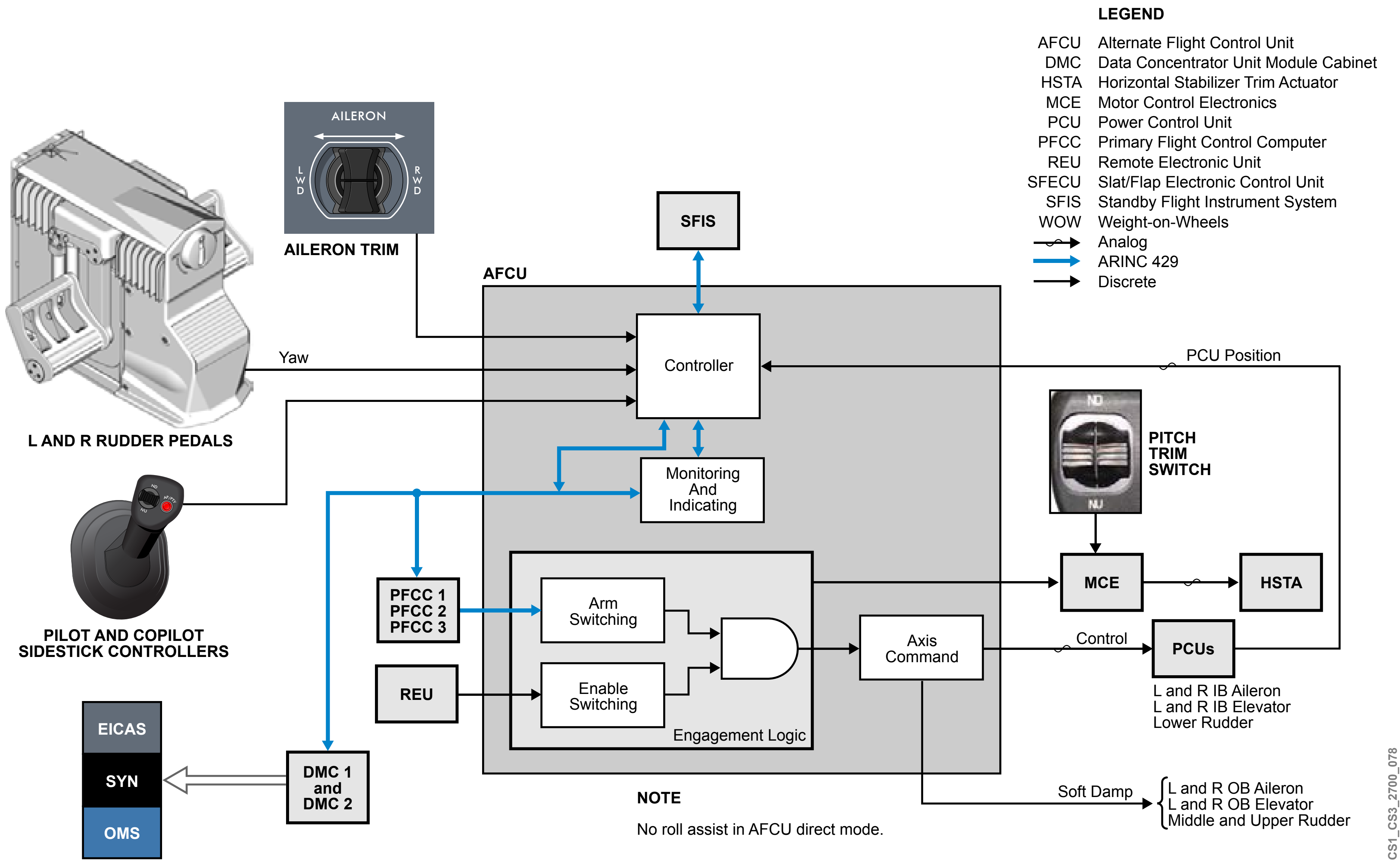


Figure 23: Alternate Flight Control Unit

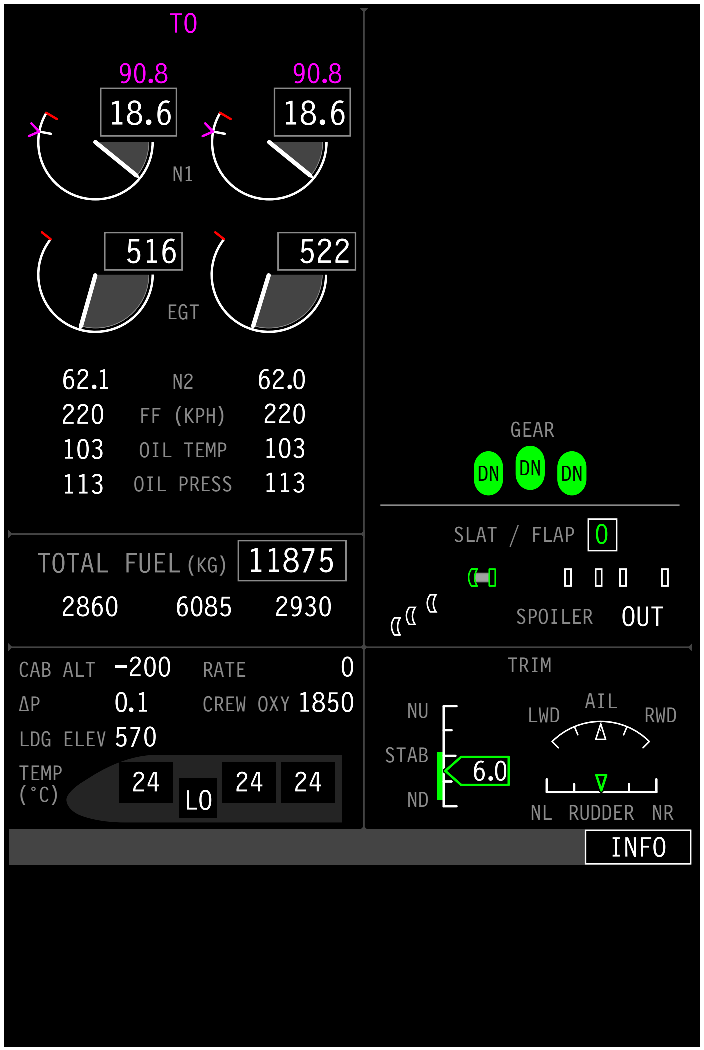
CONTROLS AND INDICATIONS

INDICATIONS

EICAS Page Flight Control Indications

The following EFCS indications are displayed on the EICAS page:

- Spoiler position
- Aileron trim position and status (in direct mode only)
- Rudder trim position and status
- Horizontal stabilizer trim position, status and takeoff range



EICAS PAGE

AILERON TRIM INDICATOR (DISPLAYED IN DIRECT MODE ONLY)	
Symbol	**Condition**
Δ	In flight or on ground
	Invalid

SPOILER INDICATOR	
Symbol	**Condition**
OUT	commanded
MAX	MAX Commanded

RUDDER TRIM INDICATOR	
Symbol	**Condition**
∇	In flight, or on ground and trim outside of takeoff range
∇	Trim in takeoff range, on ground
	Invalid

STAB TRIM INDICATOR	
Symbol	**Condition**
8.5	Pitch Trim Value (Normal and PFCC Direct mode only)
	Trim in takeoff range, on ground
	In flight, or on ground and trim outside of takeoff range
	Invalid
	Takeoff range, on ground

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Figure 24: EICAS Page Flight Control Indications

Flight Controls Synoptic Page Indications

The following EFCS information is displayed on the flight controls synoptic page:

- Multifunction spoiler position and status
- Ground spoiler position and status
- Aileron position and status
- Elevator position and status
- Flight controls in amber, with an amber border, or an amber indicating pointer, are considered failed
- Any indication with an amber x is considered invalid and requires further investigation to determine status

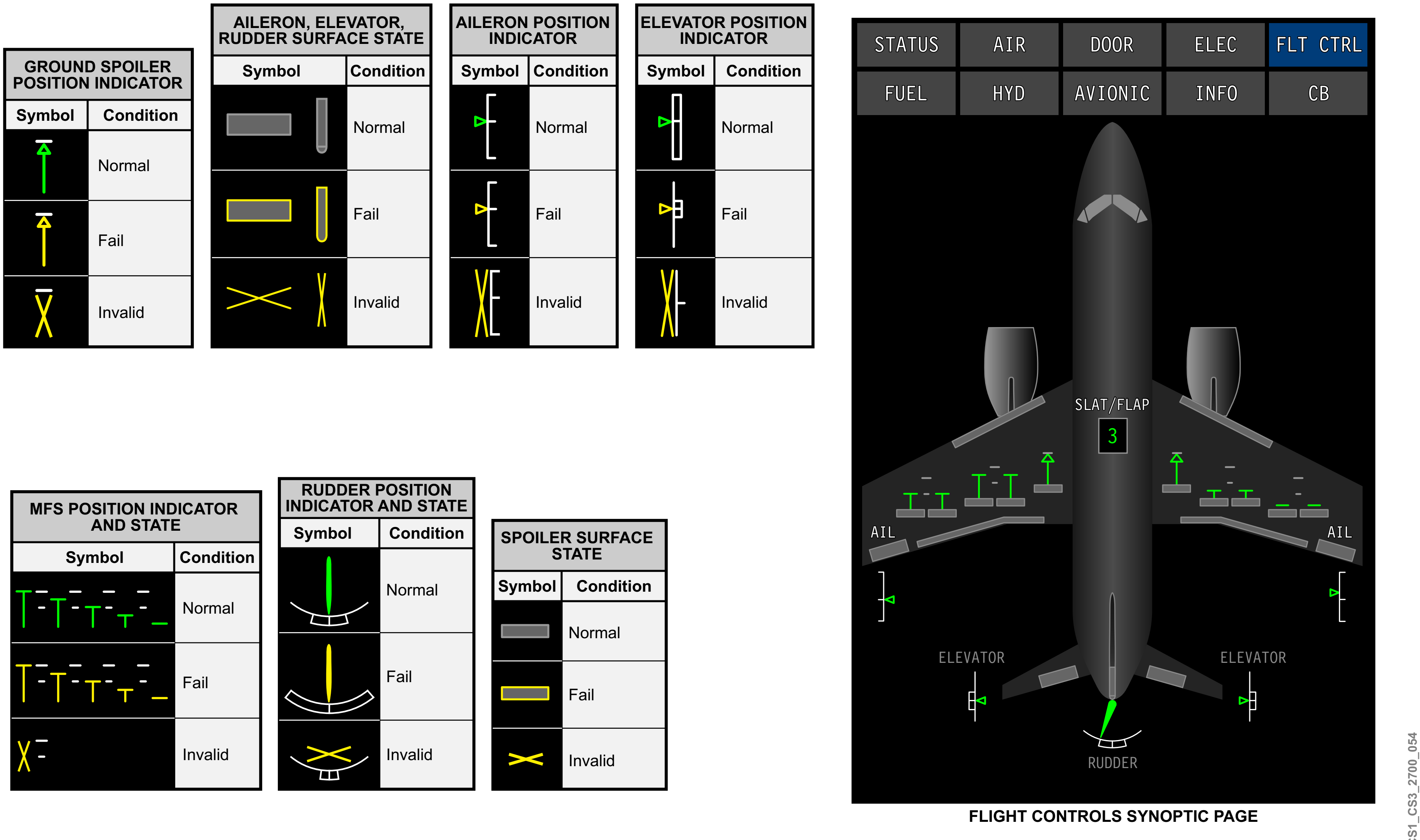
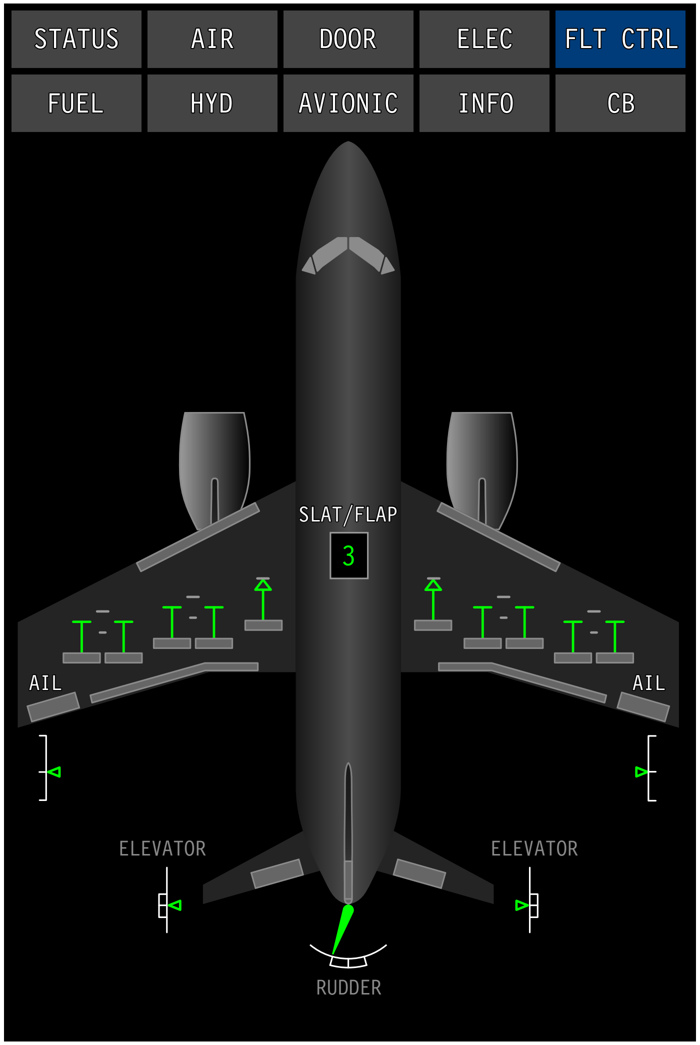


Figure 25: Flight Controls Synoptic Page Indications

Flight Controls Synoptic Page Special Indications

Several indications are displayed under specific conditions on the flight controls synoptic page as follows:

- Sidestick position and status (on the ground and no engines running)
- Primary flight control computer (PFCC) status
- Direct mode - provides crew awareness of EFCS operating mode



PFCC STATUS INDICATOR	
Symbol	**Condition**
PFCC 1 OFF	PFCC selected OFF

DIRECT MODE INDICATOR	
Symbol	**Condition**
DIRECT MODE (yellow)	Any DIRECT MODE caution message displayed
DIRECT MODE (red)	FLT CTRL DIRECT warning message displayed

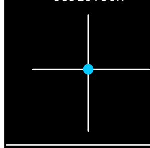
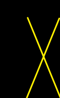
SIDESTICK INDICATOR	
Symbol	**Condition**
SIDESTICK 	On ground and no engine running. Soft stop limits.
SIDESTICK 	Invalid

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Figure 26: Flight Controls Synoptic Page Special Indications

OPERATION

ELECTRONIC FLIGHT CONTROL SYSTEM CONTROL MODES

Normal Mode

Augmentation of the pilot commands during manual flight (normal mode operation) is provided through primary flight control computers (PFCCs). Normal mode provides a fully augmented flight control capability with envelope and structural protection. In normal mode operation, pilot inputs are processed by the control laws within the PFCCs to control the aircraft response. The aircraft normally operates in this mode.

With the automatic flight control system (AFCS) engaged, the PFCC control laws provide the AFCS functions of pitch and roll axis control. When the autoland function is engaged, yaw axis control is also provided.

Direct Modes

Direct mode control of the actuators is provided as a backup to the normal mode of operation and is designed as a reversionary mode for safe flight and landing only. Direct mode operates as command-by-wire, which acts in a manner similar to a conventional mechanical system, i.e. the pilot commands surface position. While in this mode, each control channel independently generates surface commands based on its own sensor inputs and use those to control the actuators. A FLT CTRL caution message is displayed to alert the crew.

In direct mode, envelope and structural protection are not available.

PFCC Direct Mode

Transition to the PFCC direct mode occurs with loss of sensor inputs to the PFCCs (for example, data from the air data system or attitude heading computer), or integrity lost to all PFCCs. The PFCC direct mode remains active as long as the input availability or integrity required for direct mode is maintained by at least one PFCC.

REU Direct Mode

If all PFCCs are lost or there is a loss of signals from the PFCCs to the remote electronic units (REUs), the system enters the REU direct mode.

In the REU direct mode, the sidestick, rudder pedals, and trim switch positions are sent directly from the IIM to the related REUs. Control laws are located in the REUs. The REU direct mode has functionality similar to the PFCC direct mode except that sidestick priority selection is not available and there is no automatic ground lift dumping capability. The multifunction spoilers can be manually deployed but the ground spoilers will not deploy.

AFCU Direct Mode

The alternate flight control unit (AFCU) direct mode provides an independent and segregated control path from flight deck inceptors directly to control actuators. Designed for continued safe flight and landing. it provides a direct mode control comparable to the REU direct mode except for the following:

- Loss of command to all MFS surfaces
- Aileron, elevator and rudder operate with only one active PCU
- Loss of inputs from the high lift (slat/flap) system

The AFCU direct mode is indicated by a FLT CTRL DIRECT warning EICAS message.

Mode Transitions

On power-up, the system automatically transitions to normal mode when all conditions are met. Depending on the nature of the failure, the system transitions to a lower level direct mode. Direct modes can be different per axis. In some cases clearing of a fault allows the system to revert to a higher mode. In other cases, such as recovery from a TTP BUS failure, the PFCC must be reset, using the appropriate PBA on the primary flight controls panel, to allow the system to transition to the higher level mode.

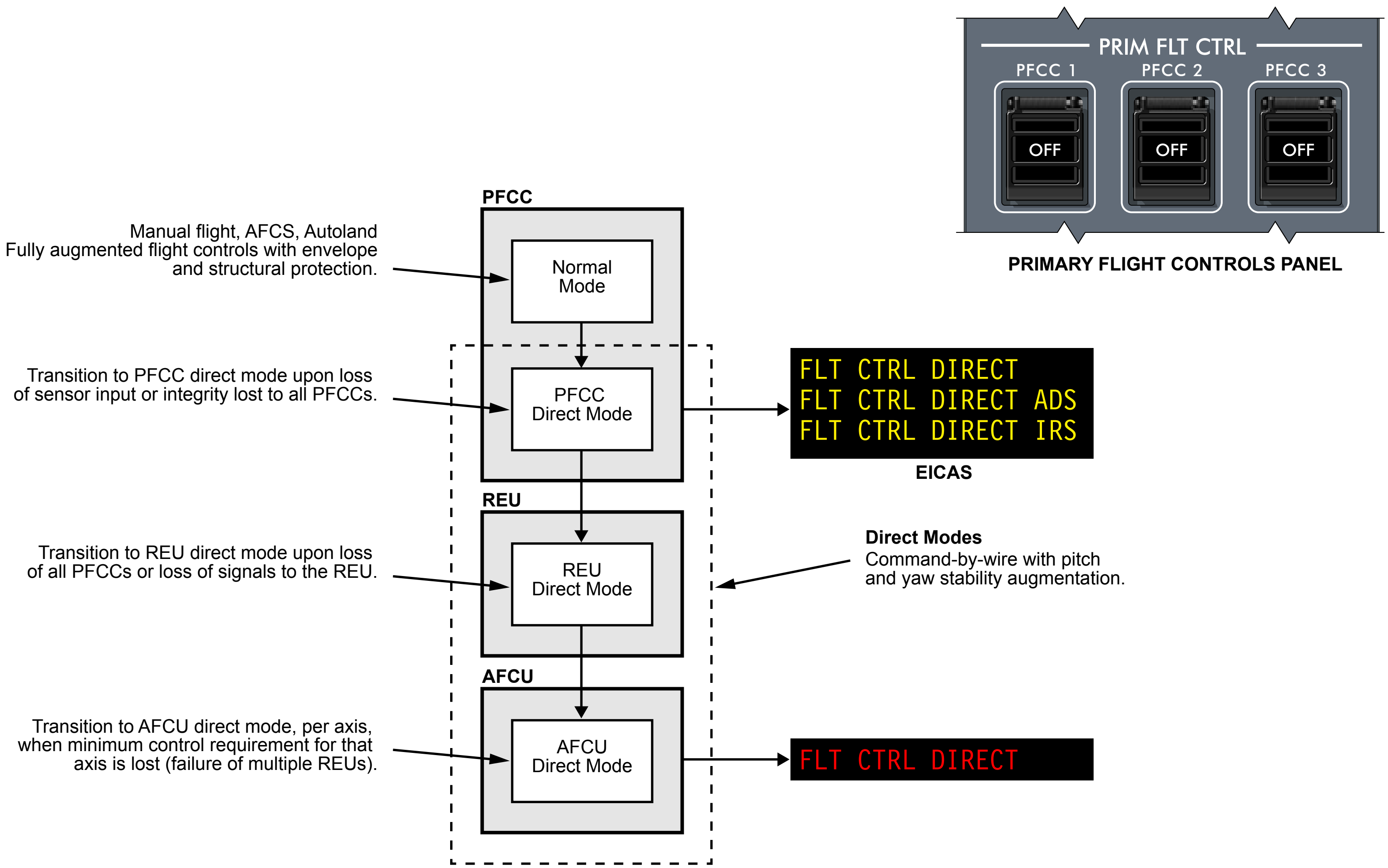

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Figure 27: EFCS Modes

SIDESTICK PRIORITY PUSHBUTTON ANNUNCIATORS

The two sidestick priority pushbutton annunciators (PBAs) give the corresponding sidestick exclusive control until deactivated.

When a priority switch is pressed, the autopilot is automatically disengaged, if active.

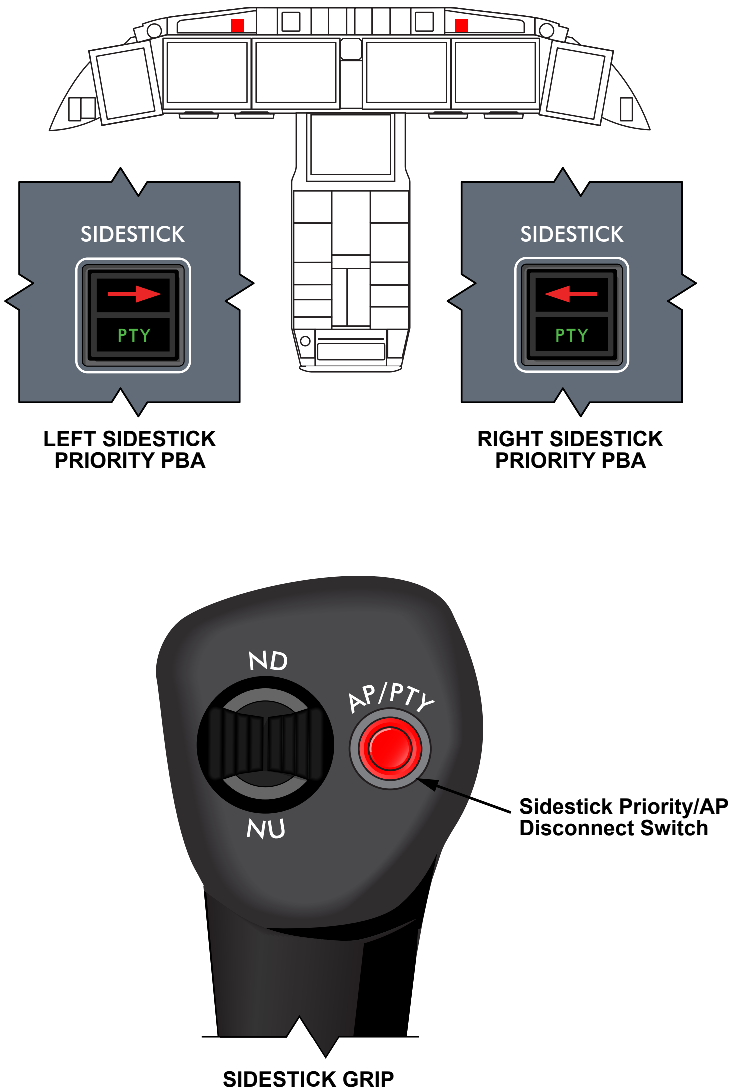
If the pilot presses and holds the PTY switch on the sidestick controller (SSC), the copilot PBA illuminates a red, left-pointing arrow. The pilot PBA flashes PTY in green, accompanied by a PRIORITY LEFT aural warning.

If the pilot presses the glareshield priority PBA, the switch latches. For the copilot PBA, a red left pointing arrow remains on. A solid green PTY is displayed for the pilot PBA, and the aural PRIORITY LEFT warning is heard again.

The opposite scenarios occur if the copilot takes priority.

A CONFIG SIDESTICK warning EICAS message, with a CONFIG SIDESTICK aural warning, is activated if a priority PBA is selected prior to takeoff.

If both pilots are making SSC inputs, a red DUAL is displayed on both primary flight displays (PFDs), accompanied by an aural DUAL INPUT warning.



STATE	GLARESHIELD PBAs STATE	VISUAL AND AURAL ANNUNCIATIONS
Normal	SIDESTICK SIDESTICK	
Dual Input	SIDESTICK SIDESTICK	DUAL on both PFDs “DUAL INPUT” aural warning
Momentary Priority Left	SIDESTICK PTY SIDESTICK	“PRIORITY LEFT” aural warning
Latched Priority Left	SIDESTICK PTY SIDESTICK	“PRIORITY LEFT” aural warning
Momentary Priority Right	SIDESTICK SIDESTICK PTY	“PRIORITY RIGHT” aural warning
Latched Priority Right	SIDESTICK SIDESTICK PTY	“PRIORITY RIGHT” aural warning

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Figure 28: Sidestick Priority

DETAILED DESCRIPTION

PRIMARY FLIGHT CONTROL COMPUTER

PFCC Interfaces

The main interface function for the PFCC is input signal management (ISM). ISM is based on multiple input voters, which receive data from independent sources and use techniques such as mid value select to determine the actual data to be used for aircraft control.

The selected and conditioned data from ISM is then used by the control laws to compute the augmented surface control commands.

The PFCCs interface with numerous external systems, which are required to support the following functionality:

- Control laws for normal and direct mode of operation
- Flight crew annunciation and interfacing (CAS/INFO messages, synoptic, maintenance, cockpit switches, data loading, etc.)
- Automatic flight control functions that are hosted on the PFCC

Main external interfaces are captured through the IIMs and the PFCCs. These interfaces can be divided into the following categories:

- Indirect input/outputs to other aircraft systems via the data concentrator unit module cabinets (DMCs)
- Direct input/outputs to other aircraft systems

The requirements that determine whether the input interface is taken directly from the line replaceable unit (LRU), or indirectly through the DMC, depends on the required availability and integrity of that input.

The other main contributor to the interface is the functionality that is hosted on the PFCC to support autoland and autopilot. These avionic functions drive external interfaces to the PFCC the same way that the EFCS does. These interfaces provide direct or indirect interconnect to other system members via the data concentration system (DCS).

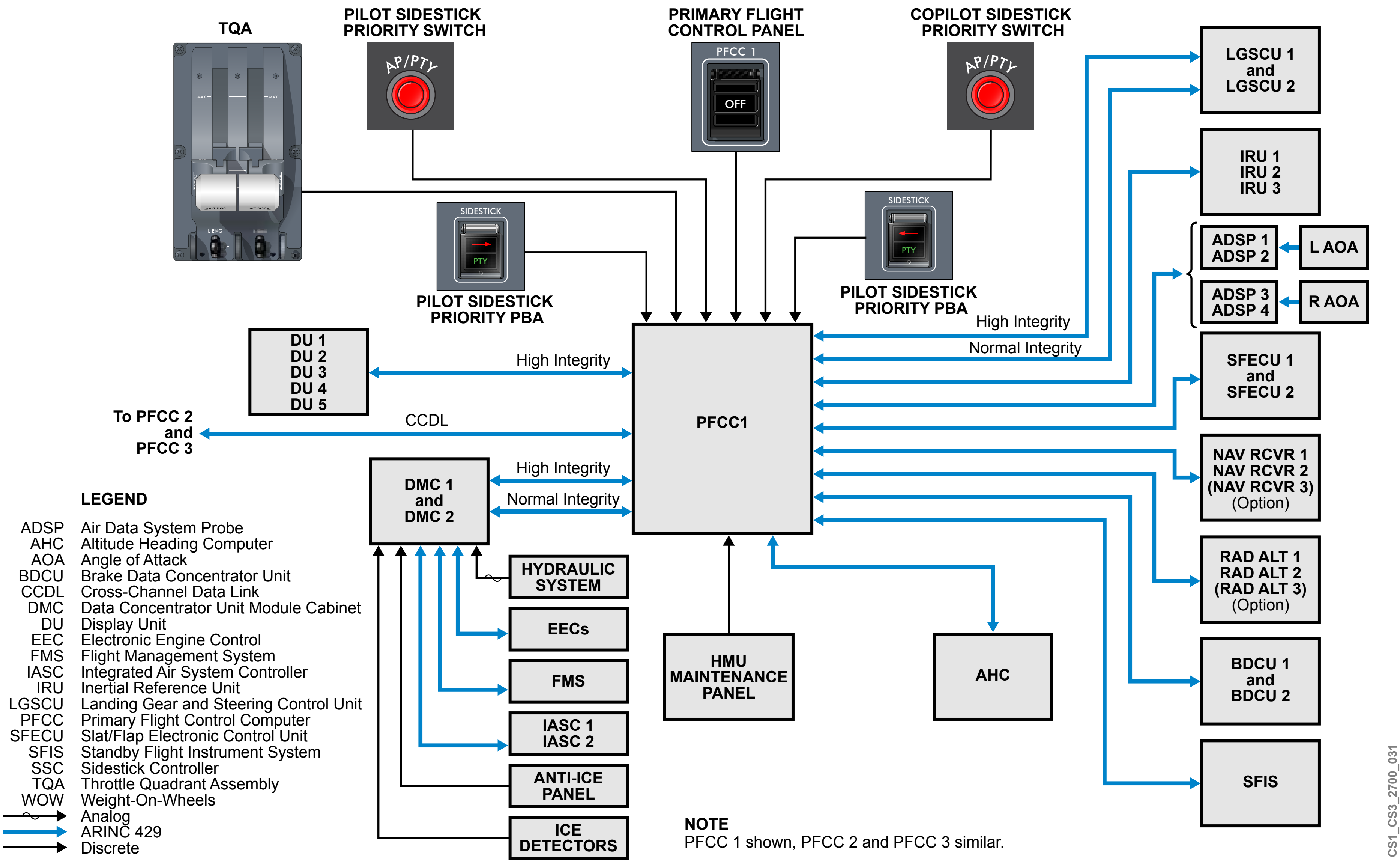


Figure 29: Primary Flight Control Computer Interfaces

Primary Flight Control Computer Control Interfaces

Communication for control of the flight controls surfaces is through time-triggered protocol (TTP) BUSES for normal mode operation. TTP BUSES are bidirectional data buses with high bandwidth.

Each PFCC receives data from multiple TTP and ARINC 429 BUSES as well as discretes for computation of control laws.

Inputs received by the IIMs are routed to the PFCCs by independent TTP BUSES. Commands to operate the surfaces are transmitted by separate TTP BUSES to the remote electronic units (REUs) through other independent TTP BUSES.

Commands are normally transmitted through the TTP BUSES with backup commands sent through independent ARINC 429 BUSES. If TTP BUS transmission fails, the direct mode ARINC 429 BUS provides the required command inputs.

The PFFCs monitor the status of the alternate flight control unit (AFCU) through an ARINC 429 BUS when the AFCU is not in command.

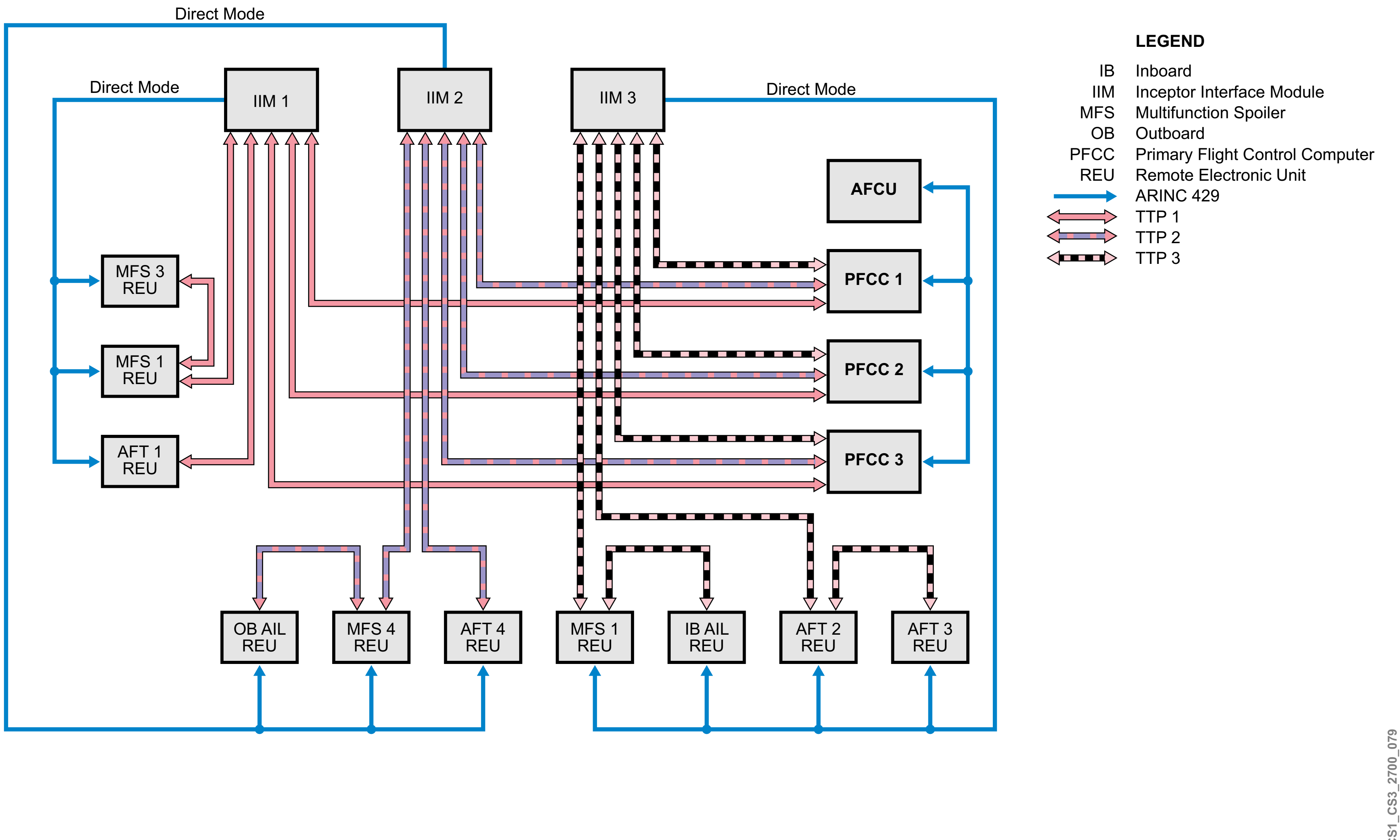


Figure 30: Primary Flight Control Computer Control Interfaces

Attitude and Heading Inputs

The primary flight control computer (PFCC) uses attitude and heading data from the inertial reference units (IRUs) as long as they are in navigation mode. When an IRU is out of navigation mode or failed, the PFCC receives attitude and heading information from the attitude and heading computer (AHC). If the AHC fails, the PFCCs receive attitude and heading information from the standby flight instrument system (SFIS).

The AHC receives air data information from the ADSP 2 through DMC 2. In the event that this information is invalid or that the ADSP heater fails, the AHC does not provide attitude and heading information to the PFCC.

The SFIS receives air data information from ADSP 3 or 4. If air data information from these probes is invalid, the attitude signal is not used by the PFCC.

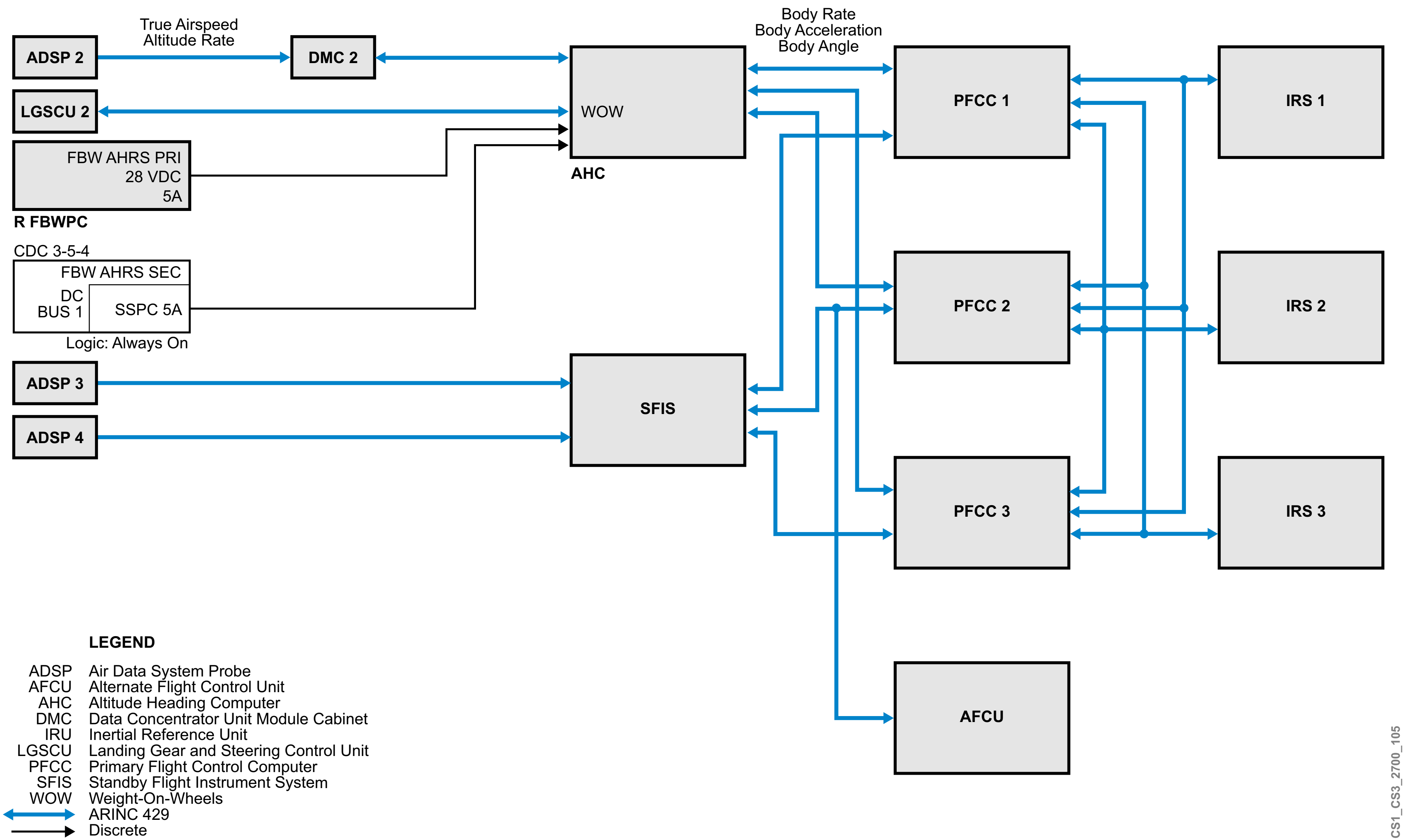


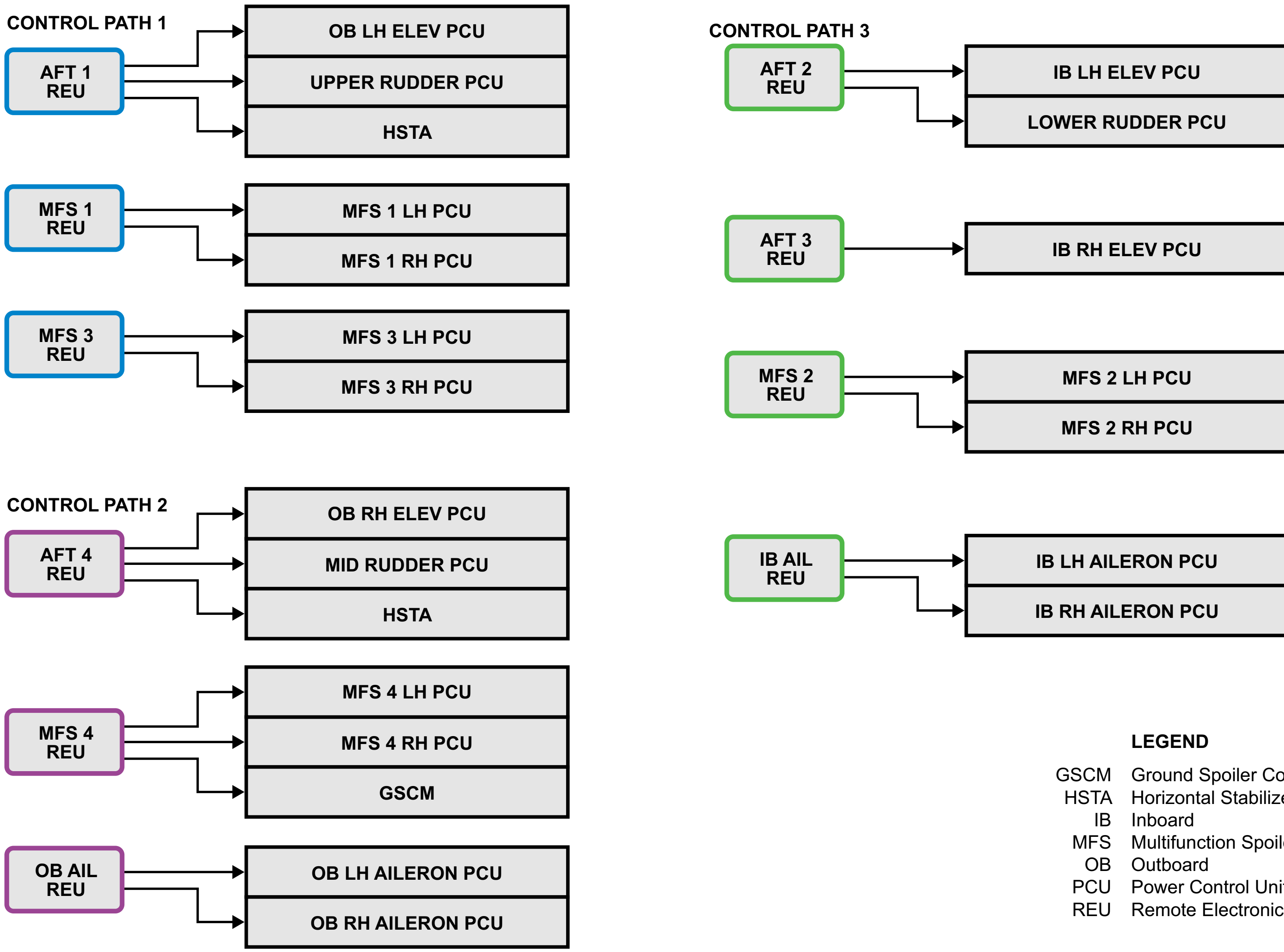
Figure 31: Attitude and Heading Inputs

REMOTE ELECTRONIC UNIT INTERFACES

Control path 1 contains three REUs that control a single PCU for the left elevator, a single PCU for the rudder and the PCUs for multifunction spoilers (MFSs) 1 and 3. In addition, control path 1 also controls one channel of the horizontal stabilizer trim actuator (HSTA).

Control path 2 also contains three REUs that control a single PCU for the right elevator, a single PCU for the rudder, MFS 4 PCUs and both outboard aileron PCUs. In addition, control path 2 also controls one channel of the HSTA and the ground spoiler control module (GSCM), which controls the ground spoiler actuators.

Control path 3 consists of four REUs that control two PCUs for the elevators, a single PCU for the rudder, the MFS 2 PCUs, and the inboard aileron PCUs.



LEGEND

GSCM	Ground Spoiler Control Module
HSTA	Horizontal Stabilizer Trim Actuator
IB	Inboard
MFS	Multifunction Spoiler
OB	Outboard
PCU	Power Control Unit
REU	Remote Electronic Unit

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Figure 32: Remote Electronic Unit Interfaces

CONTROL LAWS

The FBW system maintains the aircraft within the normal flight envelope and provides structural protection using control laws (CLAWS). CLAWS are used to improve ride quality and stability for passenger comfort. Tactile cues, such as increased force on the sidestick, warn of the limits of the normal operational flight envelope. This is known as soft envelope protection. Movement of the control stick beyond these limits is permitted, but it is limited by mechanical stops. This is known as hard envelope protection.

Normal mode control laws provide the following functions:

- Manual speed trim using the pitch trim switch
- Automatic horizontal stabilizer trim to off load the elevators
- Automatic nose up pitch compensation in turns for bank angles up to ±33°
- Nose landing gear loading – elevators pitch down at high thrust settings on takeoff roll. Pitch down signal decreases to zero at 90 kt
- Turn coordination to minimize slip or skid during aileron commanded turns
- Spiral mode stability – prevents increasing bank angles during turns
- Rudder-roll coupling – converts rudder inputs to aileron commands
- Manual directional rudder trimming using the RUDDER trim switch
- Steep approach (CS100 only) – spoiler control during approach and flare guidance cues
- Aileron lift augmentation – both ailerons deflect downward with flap extension for lift augmentation
- Proportional lift dump (PLD) retraction with high throttle lever angles
- PLD extension on aircraft overspeed
- Ground lift dumping

The following envelope and structural protections are available in the normal mode:

- Normal load factor protection depending on slats/flaps position, between -1 G and 2.75 G
- Pitch attitude protection – Limited to 30° up and 20° down. Augments the high angle-of-attack (AOA) protection
- High (AOA) protection – Prevents exceeding maximum AOA
- Overspeed protection
- Tail strike reduction - takeoff only – Reduces aircraft pitch rate during aggressive rotations
- Tail plane AOA protection – Limits the available nosedown AOA
- Elevator surface command limiting – deflection decreases with increasing airspeed
- Elevator surface split limiting as a function of airspeed
- Bank angle protection – Limited to 80° of bank
- Roll rate protection – Limited to 20° +/- 2° per second
- Wing maneuver load alleviation – both ailerons deflect up to minimize wing root bending during positive load factor maneuvers
- Gust load alleviation – both ailerons deflect up with MFS to minimize wing loads due to gusts
- In-flight partial engine out compensation – yaw assist to control the aircraft during and after engine failure
- Rudder surface command limiting – decreasing rudder travel with increasing airspeed

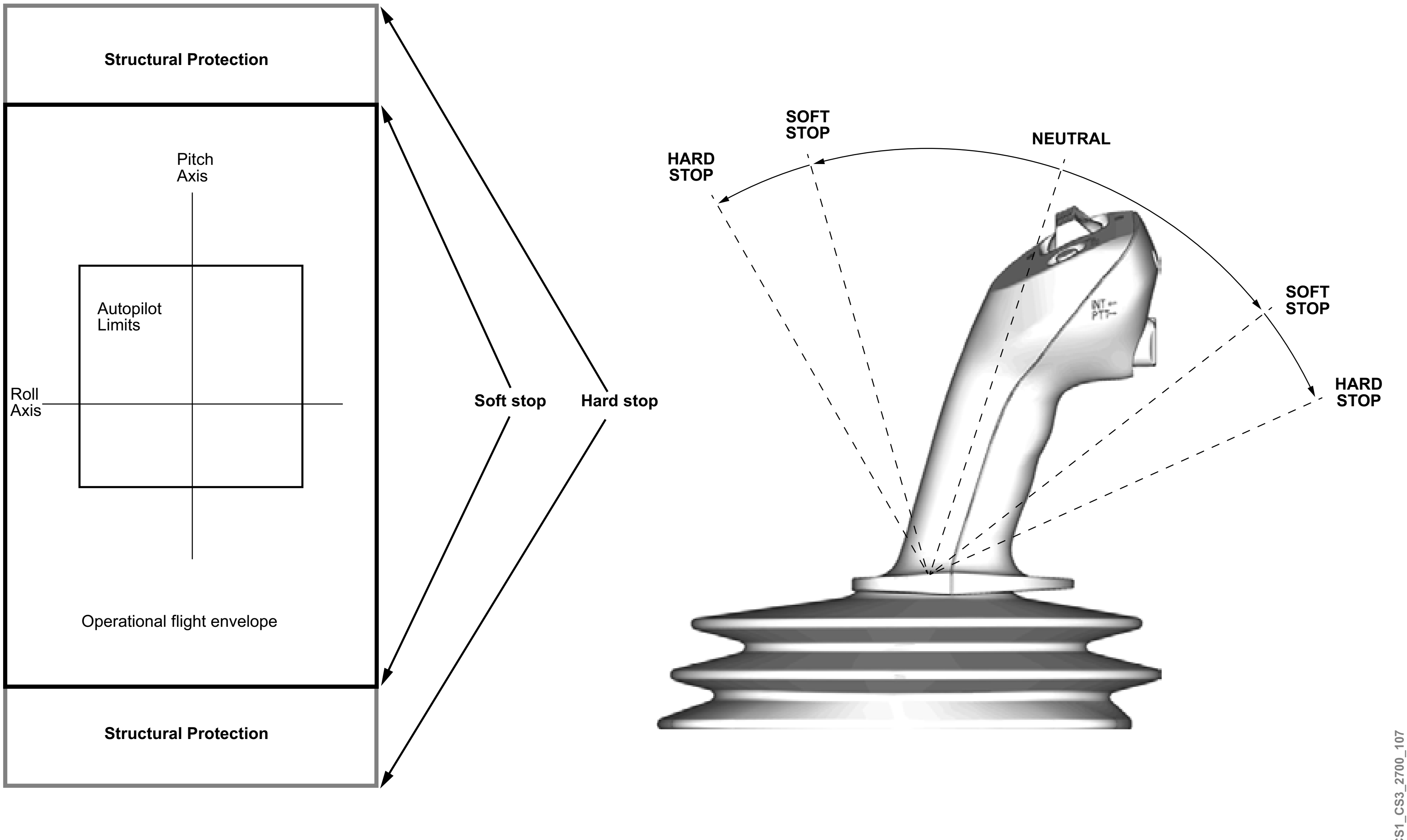

Figure 33: Control Laws

MODES OF OPERATION

The FBW system consists of three redundant control paths and an alternate independent control path that uses the alternate flight control unit (AFCU). A single control path consists of one IIM and its associated set of three or four REUs, an MCE, and three PFCCs. One PFCC is in command and is referred to as the PFCC in control (PIC). The minimum control requirements (MCRs) can be met by any single control path.

In order to provide the highest level of redundancy and system safety, the FBW system operation has the following four modes:

- Normal mode
- PFCC direct mode
- REU direct mode
- AFCU mode

NORMAL MODE

Aircraft with a FBW system require computer-controlled flight modes that are capable of determining the operational mode of the aircraft. A reduction of FBW control can be caused by the failure of a flight control computer or a sensor, such as an air data smart probe (ADSP) or inertial reference unit (IRU). Should this occur, the system reconfigures automatically.

With all FBW components operative, the FBW system operates in normal mode. In normal mode, the control laws provide full flight envelope and structural protection for pilot control inputs or autopilot commands. A number of failures can be tolerated in normal mode without affecting aircraft controllability. The FBW system switches to direct mode if too many system failures occur.

Augmentation of the flight crew commands is provided through the normal and direct mode control laws (CLAWS). In normal mode, the PFCC CLAWs also provide the autopilot (AP) inner loop control. The PFCCs receive control commands directly from the AP without the use of autopilot servos to back drive the cockpit controls. During AP engagement the sidestick controllers (SSC) are held at neutral and require a minimum breakout force to disengage the AP.

Real-time executing monitors, known as continuous built-in test (CBIT), are designed to detect faults quickly enough to allow system reversionary action to be performed before hazardous conditions occur.

Ground Operation

On the ground, the normal mode control laws are similar to the direct mode. The control surface deflections are proportional to the control inputs. During the ground roll, the direct mode yaw augmentation function is active above 60 kt, blending from zero to the yaw rate proportional signal. During landing the reverse occurs as the aircraft decelerates below 60 kt. The authority of yaw augmentation function on ground is limited to ±6° of rudder deflection.

When on ground, the sidestick controller and rudder pedals can command full aileron and rudder authority. With the high lift in takeoff configurations, the maximum function spoilers (MFSs) roll authority is reduced until the aircraft is in flight.

Ground-Air Transition

The normal control laws blend from ground mode to air mode over a 5 second period after air mode is detected. The reverse occurs during landing.

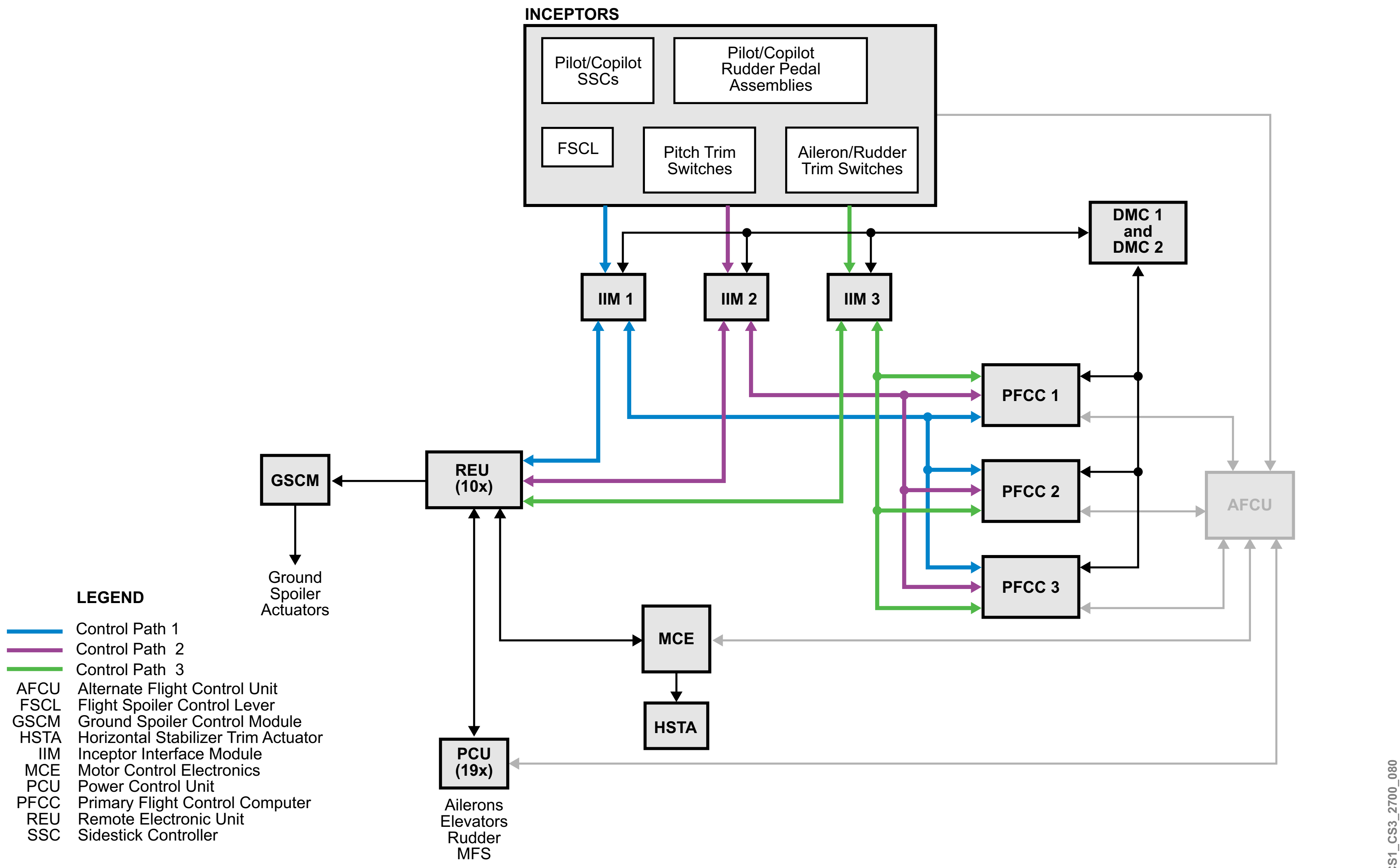

Figure 34: Normal Mode Operation

DIRECT MODE

The EFCS reverts to direct mode when normal mode can no longer be supported either as a result of the PFCC losing data from other aircraft systems essential to support normal mode, loss of all the PFCCs, or loss of multiple REUs such that minimum control requirements (MCR) can no longer be provided. The aircraft is capable of continued safe flight and landing after reversion to direct mode. The direct mode uses minimum sensor feedback in order to provide simple control laws. Direct mode is hosted in the PFCCs, the REUs, and the AFCU.

PFCC Direct Mode

The EFCS reverts to PFCC direct mode when normal mode becomes unavailable. In PFCC direct mode, the PFCCs process available aircraft sensor inputs and cockpit inputs through the same data paths as normal mode. The appropriate flight control surface position commands are however, calculated based on direct mode CLAWS, allowing stick-to-surface control of the primary aircraft control surfaces with limited augmentation.

In PFCC direct mode, full control of the spoilers is available, including ground lift dumping and horizontal stabilizer trim control. Autoflight is not available.

The EFCS remains in PFCC direct mode if the input availability or internal integrity required for direct mode is maintained by at least one PFCC. In PFCC direct mode, the PFCCs transmit the surface commands to the IIMs through the same TTP data path as normal mode.

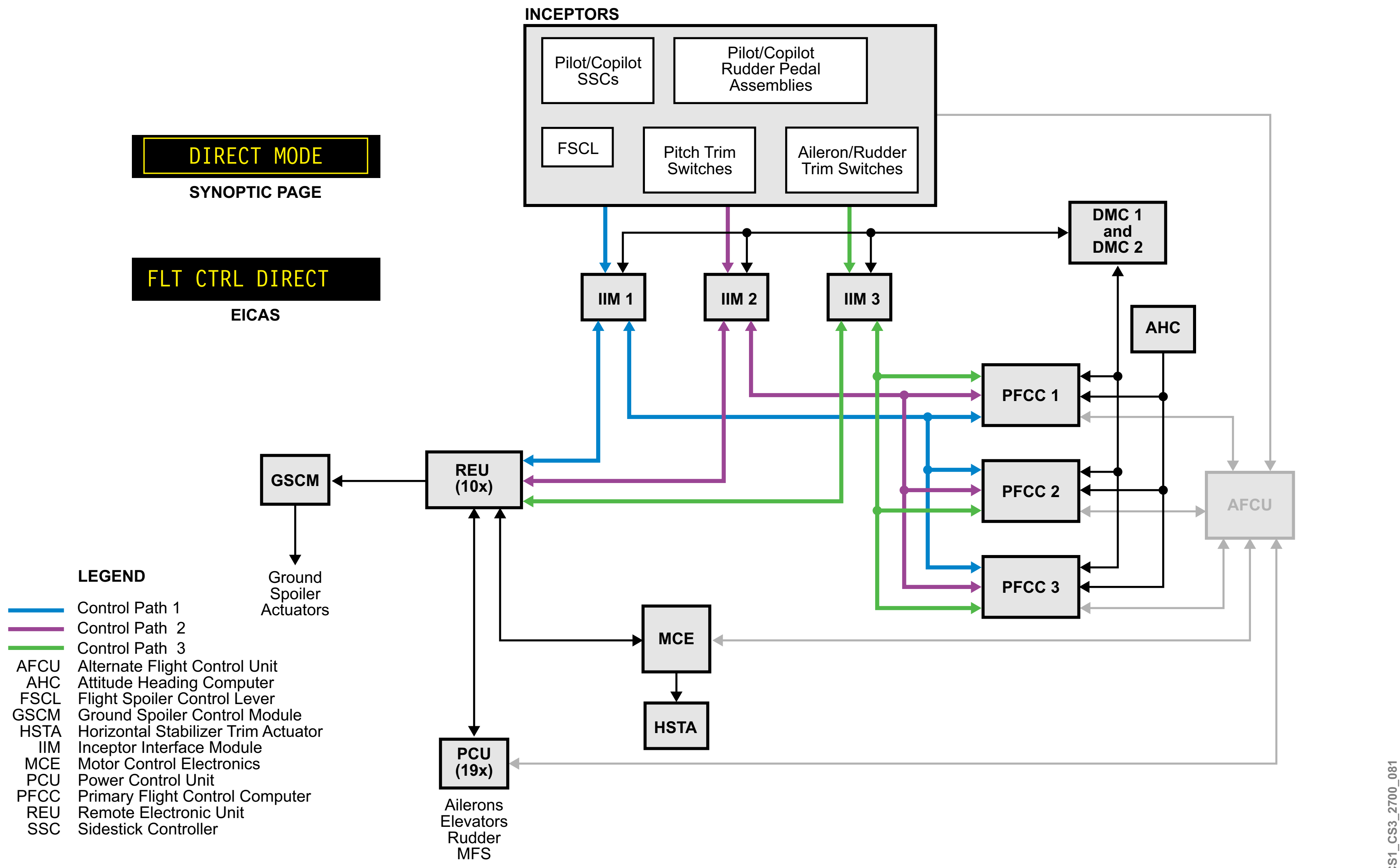

Figure 35: PFCC Direct Mode Operation

Remote Electronic Unit Direct Mode

In the event of further loss of inputs to the PFCCs or failure of all PFCCs, the flight control system reverts to REU direct mode. In REU direct mode, the REUs process input commands and aircraft sensor data received from the IIMs through its two available input sources, TTP and A429. The REUs calculate the appropriate primary flight control surface position commands, based on direct mode CLAWS (similar to PFCC direct mode). In REU direct mode automatic spoiler ground lift dumping control is not available.

REU direct mode has similar functionality to PFCC direct mode except:

- Sidestick priority selection is not available
- No automatic GLD function. MFS can be deployed manually with no ground spoiler deployment

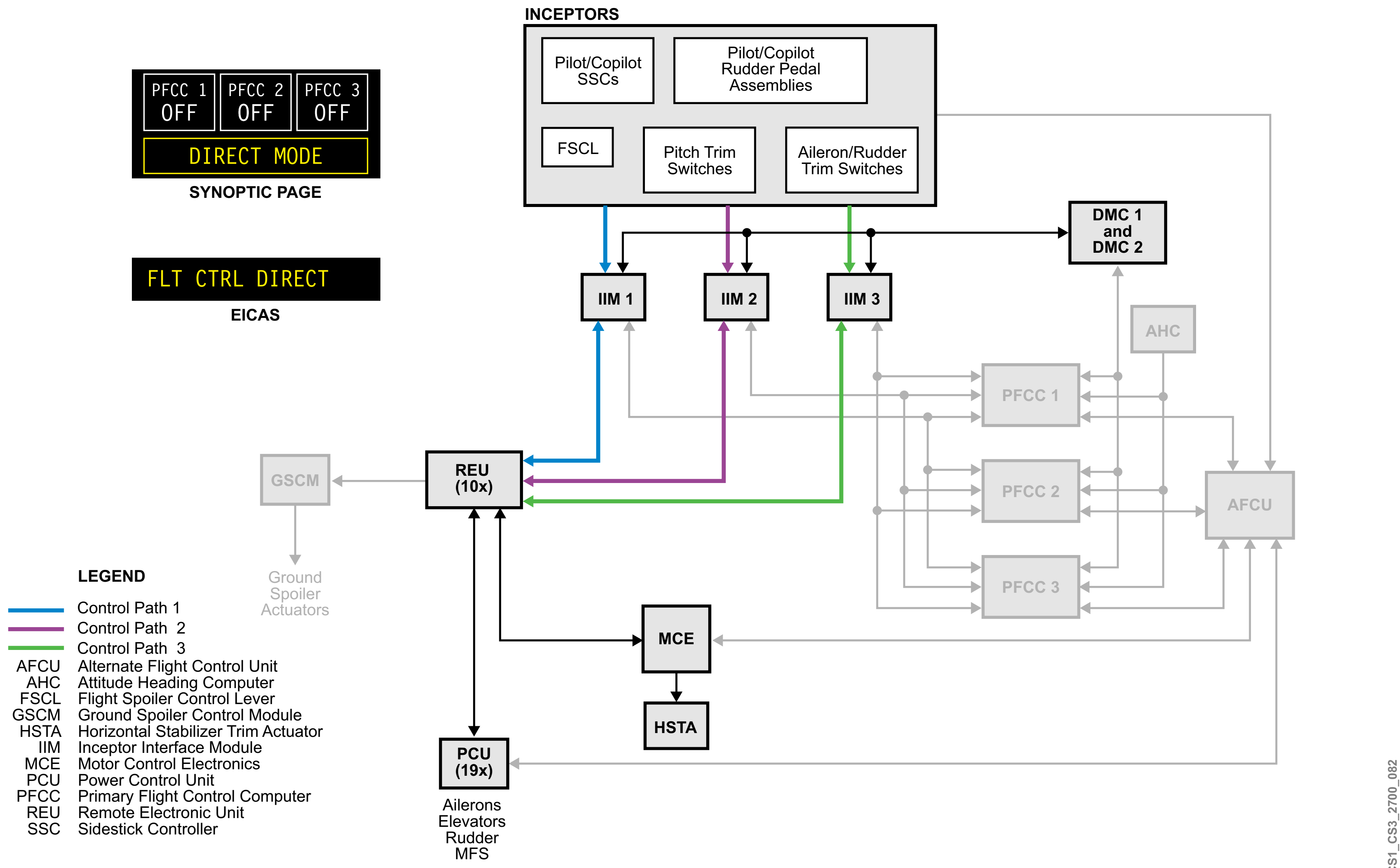

Figure 36: Remote Electronic Unit Direct Mode Operation

Alternate Flight Control Unit Direct Mode

The alternate flight control unit (AFCU) provides an independent and segregated control path from flight deck inceptors directly to control all primary surfaces. It also commands engagement of the alternate pitch trim path to the motor control electronics (MCE), allowing direct control by the pitch trim switches. The AFCU provides direct mode control comparable to the REU direct mode for continued safe flight and landing. It interfaces directly with cockpit controls and actuators using dedicated sensors/electrical channels and has no reliance on the IIMs and REUs.

The AFCU engagement logic is as follows, when the system reaches the minimum control requirements (MCR) on any axis, the processor-in-command (PIC) attempts to reset the REUs.

The AFCU gain control upon the following actions:

- Arming by the PFCC is a precondition necessary to be active
- Engagement enable by the REUs is the actual go-ahead
- Engagement is not latched. AFCU disengages when either the PFCC removes arm signal or REUs remove engagement enable

Arming upon any of the three conditions below:

- In normal condition, the PFCC provides arming for roll axis when the REU configuration is below MCR
- There is no PFCC active
- There are more than one PFCC claiming to be in control

Engagement takes place when:

- AFCU engagement on any axis when either one aileron, one elevator, or the rudder is inoperative (REUs or PCUs failure)
- AFCU activates its commands to the aileron, elevator or rudder PCUs
- AFCU forces the adjacent PCU(s) on the same surface to soft damped mode

The AFCU is dissimilar in design and complex hardware from the REU and is monitored full-time by the PFCC to minimize dormant faults during normal mode. It is not monitored when engaged.

The PFCC provides the AFCU with an arm command for each of the control axes. Engagement of the control axis is complete when the REU sends a discrete signal to the AFCU, this indicates that the number of REUs needed for minimum control for that axis has failed. The AFCU arms by default in the absence of a PFCC in control or multiple PFCCs claiming to be in control. Prior to AFCU arming in the event of multiple REU failures, the PFCC first attempts to reset the REUs. If the REUs still fail, the AFCU is armed for control engagement.

AFCU direct mode control is comparable to REU direct mode except:

- Loss of command to all MFS surfaces
- Ailerons, elevators, and rudder operate with only one active PCU per surface
- Loss of inputs from the high lift system

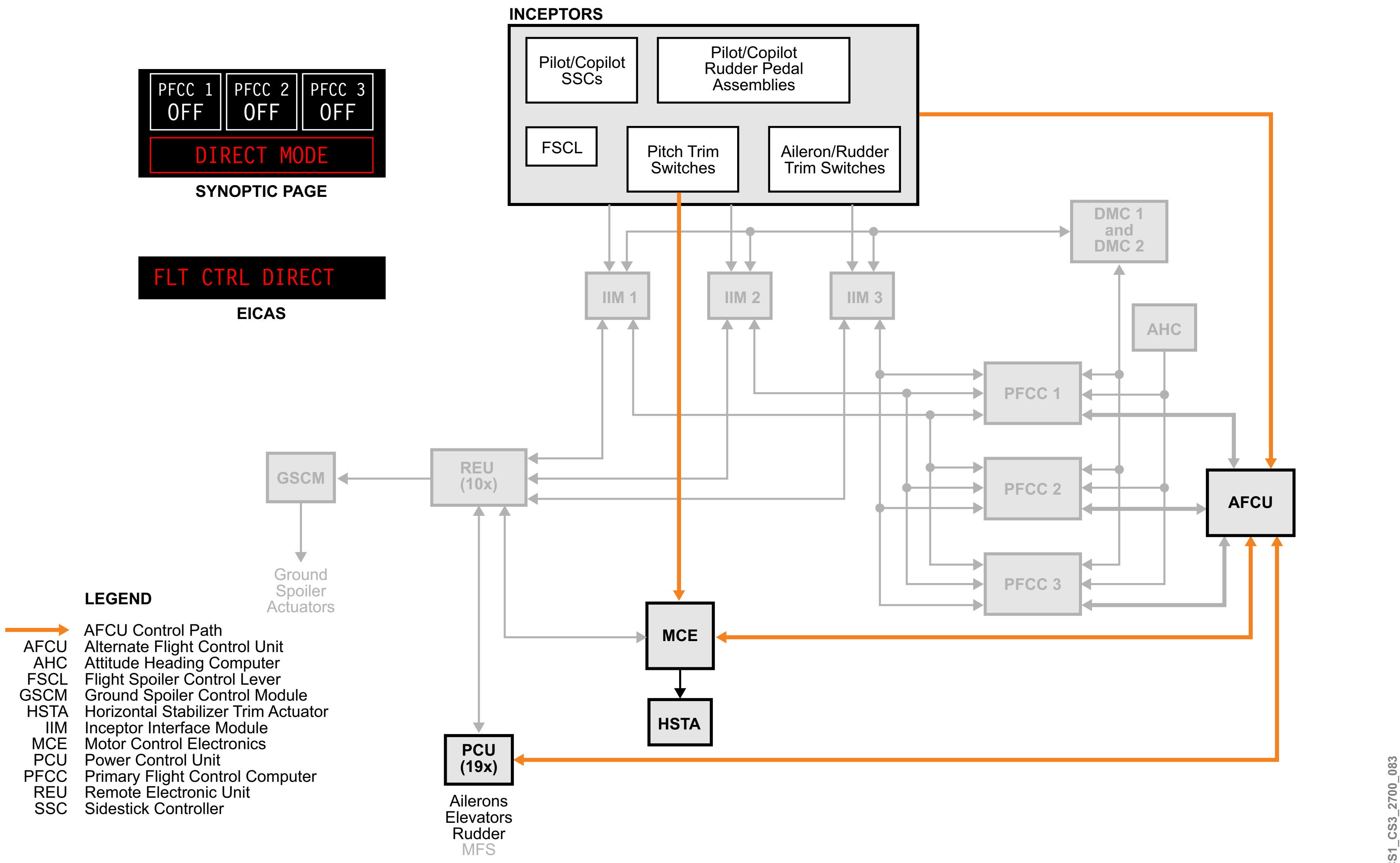

Figure 37: Alternate Flight Control Unit Direct Mode Operation

HORIZONTAL STABILIZER TRIM OPERATION

The horizontal stabilizer trim control system implements a dual-channel control architecture, operating in an active/standby configuration. The active channel selection is automatically controlled by the PFCCs and the REUs.

Each channel consists of a REU and MCE channel paired together to control one of two horizontal stabilizer trim actuator (HSTA) motors and electric solenoid driven brake. The two motors are mechanically linked; when one motor is driven the inactive motor turns.

The electric brake has two coils. Each MCE channel drives a coil. One coil is sufficient enough to disengage the electric brake and enable the active MCE channel to drive the HSTA. The brake is always energized when a channel is in control. The brake is spring-loaded to the engaged position.

The REU provides motor enable and brake enable commands to its respective MCE channel. Each REU communicates trim commands over the motor control data link (MCDL) BUS. The MCE channel supplies 270 VDC to the motor,and a 28 VDC brake release signal to the brake.

Resolvers on the motor assembly provide feedback to the MCE for motor speed control. When the brake is released, the motor drives the ballscrew assembly through the gearbox. The ballscrew assembly positions the horizontal stabilizer in response to trim commands. Three rotary variable differential transformers (RVDTs) provide horizontal stabilizer position feedback.

The REUs communicate with each other through a cross communication data link (CCDL) bus. Each REU has a monitor and control section. If a failure is detected in one channel, the motor and the brake power are removed. The horizontal stabilizer remains in the last position while command is transferred to the other REU. When the REU sends motor and brake enable commands, the brake is released and the other motor drives the horizontal stabilizer.

In normal mode, the horizontal stabilizer trim commands are generated from two sources:

- PFCCs
- Pilot and copilot trim switches

In flight, the CLAWS of the active PFCC supplies auto-trim commands to the REUs when the aircraft is more than 50 ft above the ground.

If the autopilot is not engaged, pilot and copilot pitch trim switches can be used to adjust the aircraft speed reference.

On the ground, the pilot and copilot pitch trim switches are used to trim the horizontal stabilizer noseup, or nosedown.

In direct and PFCC direct mode the pilot and copilot pitch trim switch inputs to the IIMs, and PFCCs are sent as noseup or nosedown commands to the REUs.

In flight, the pilot and copilot pitch trim switch inputs are inhibited if they are present for longer than 3 seconds. The intent of this is to prevent a runaway stabilizer. On the ground, the inhibit is not applied.

In the alternate mode, the pilot and copilot pitch trim switch inputs are provided directly to the HSTA through the alternate control path in channel 1 of the MCE.

When no trim commands are applied, a no-back device on the ballscrew assembly holds the HSTA in position. If the resolver detects any HSTA movement without a valid trim command present, the MCE channel in control powers the motor to hold the stabilizer in the last position. If the resolver and RVDT signals indicate the motor is not holding the HSTA, a pitch trim uncommanded motion fault is set, and the brake is applied. The channel in control is disabled and the standby channel takes over.

If both channels fail, the HSTA is held in the last position and a STAB TRIM FAIL caution message is displayed.

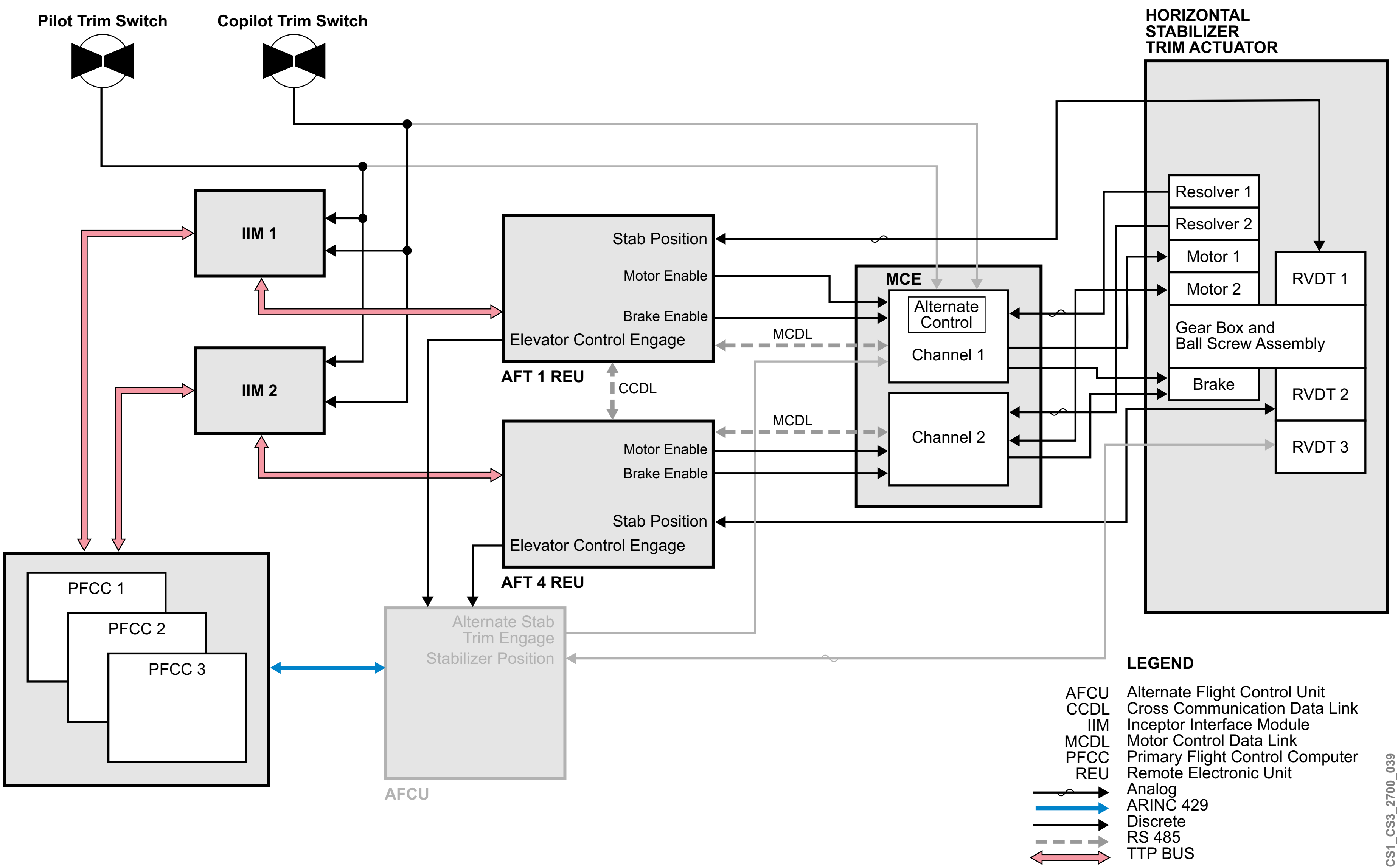

Figure 38: Horizontal Stabilizer Trim Operation

ALTERNATE HORIZONTAL STABILIZER TRIM OPERATION

In alternate horizontal stabilizer trim operation, the PFCC, IIM, and REU paths are disabled. The AFCU assumes control and sends an alternate stabilizer trim engage signal to the MCE. This allows the alternate control in MCE channel 1 to become active.

The pilot and copilot trim switches are wired directly to channel 1 of the MCE. In this mode, the trim switches provide trim up and trim down commands to relieve elevator loads and sidestick forces. MCE channel 1 provides the motor power and brake release signals to the HSTA. The motor circuit receives feedback from resolver 1 to control the stabilizer trim at a rate of 0.25° per second when the flaps and slats are retracted, and at 0.5° per second when the flaps and slats are extended.

The AFCU receives stabilizer position feedback from RVDT 3 on the HSTA.

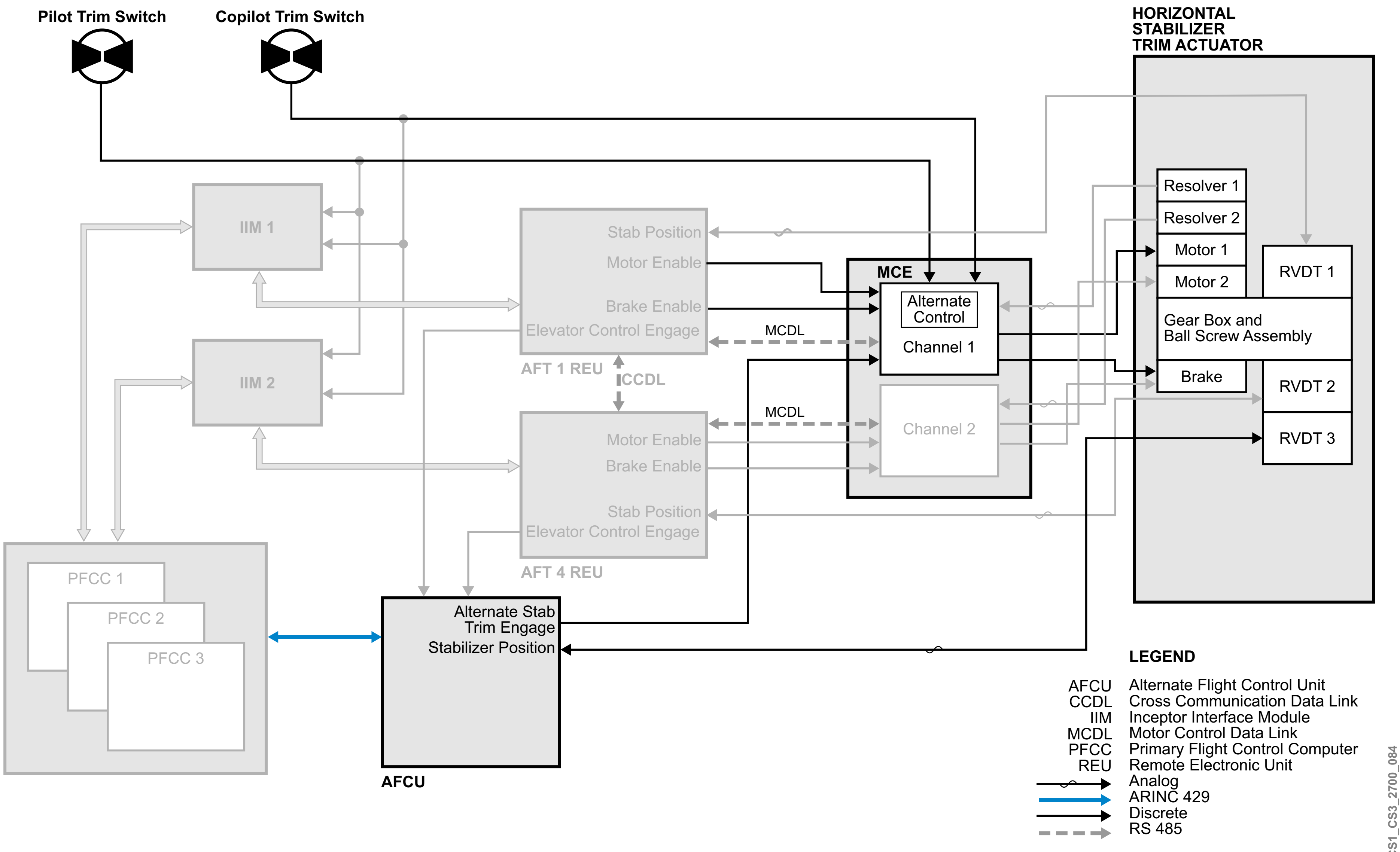

Figure 39: Alternate Horizontal Stabilizer Trim Operation

MONITORING AND TESTS

SHAKER TEST

The sidestick shaker test is initiated through the AVIO tab on the AVIONICS synoptic page. Pressing the SHAKER test button, commences the test. The pilot sidestick shaker activates for 3 seconds, followed by the copilot sidestick shaker for 3 seconds. Since the sidestick motors are not monitored by the FBWS, the sidestick shakers must be monitored for operation.

When the test is initiated, the SHAKER test button is grayed out and an IN PROG message is displayed. When the test ends, a white DONE message is displayed.

CS1_CS3_2700_103

Figure 40: SHAKER Test

EFCS FLT CTRL TEST

The automatic built-in test (ABIT) is initiated by the pilot when the FLT CTRL TEST REQUIRED advisory message is displayed.

The PFCC generates a FLT CTRL TEST REQUIRED advisory message every 20 flight hours.

The ABIT is performed to detect latent faults. The FLT CTRL test can only be conducted on ground. The hydraulics systems must be pressurized for the test which may result in movement of the control surfaces.

The test is carried out using the FLT CTRL virtual test button on the AVIONIC synoptic page. The test lasts approximately 60 seconds.

The test status and result appear beside the FLT CTRL virtual test button as follows:

- IN PROG - Test in progress
- FAULT - Test failure due to component fault
- PASS - Test successfully completed

PILOT INITIATED TEST INDICATIONS	
Symbol	Condition
FAULT	Test failure due to component fault
IN PROG	Test in progress
PASS	Test successfully completed

CS1_CS3_2700_104

Figure 41: EFCS FLT CTRL Test

CAS MESSAGES

The following page provides the crew alerting system (CAS) and INFO messages for the electronic flight control system (EFCS).

Table 1: WARNING Messages

MESSAGE	LOGIC
CONFIG RUDDER TRIM	Rudder trim position out of takeoff range.
CONFIG SIDESTICK	Left or right sidestick latched at takeoff.
CONFIG SPOILER	Spoiler (MFS or GS) deployed at takeoff.
CONFIG STAB TRIM	Stab trim out of position for takeoff.
FLT CTRL DIRECT	Aircraft in AFCU direct mode.

Table 2: CAUTION Messages

MESSAGE	LOGIC
ADS DEGRADED	PFCC is reporting errors from the air data system that could impact the input to the CLAWS.
ALPHA LIMIT	Reduction of upper pitch angle limit.
FLT CTRL DIRECT	Aircraft in direct mode.
FLT CTRL DIRECT ADS	Aircraft in direct mode due to air data input failure.
FLT CTRL DIRECT IRS	Aircraft in direct mode due to inertial data input failure.
L SIDESTICK	Left sidestick failed.
R SIDESTICK	Right sidestick failed.
STEEP NOT AVAIL	Steep approach AP mode not available, detected with system armed.
YAW AUTHORITY	Rudder command near max operational authority.
PITCH AUTHORITY	Left or right elevator command near max operational authority.

Table 2: CAUTION Messages

MESSAGE	LOGIC
ROLL AUTHORITY	Left or right aileron command near max operational authority.

Table 3: ADVISORY Messages

MESSAGE	LOGIC
FLT CTRL FAULT	Loss of redundant or non-critical functions for the primary flight control systems. Refer to INFO messages.
FLT CTRL IN TEST	Flight control fly-by-wire (FBW) system in automatic test.
FLT CTRL TEST REQ	PBIT interval time exceeded.
PFCC 1 FAIL	Loss of PFCC 1 (redundancy).
PFCC 2 FAIL	Loss of PFCC 2 (redundancy).
PFCC 3 FAIL	Loss of PFCC 3 (redundancy).
PITCH AUTHORITY	Left or right elevator near max authority.
ROLL AUTHORITY	Left or right aileron near max authority.
SPOILER MISMATCH	Spoiler autoretract engaged.

Table 4: STATUS Messages

MESSAGE	LOGIC
PFCC 1 OFF	PFCC 1 selected OFF.
PFCC 2 OFF	PFCC 2 selected OFF.
PFCC 3 OFF	PFCC 3 selected OFF.

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Table 5: INFO Messages

MESSAGE	LOGIC
27 FLT CTRL FAULT - L SIDESTICK SNSR REDUND LOSS	Loss of any pitch or roll pilot SSC sensor.
27 FLT CTRL FAULT - R SIDESTICK SNSR REDUND LOSS	Loss of any pitch or roll copilot SSC sensor.
27 FLT CTRL FAULT - L SIDESTICK SNSR DEGRADED	Loss of any two pitch or any two roll pilot SSC sensors.
27 FLT CTRL FAULT - R SIDESTICK SNSR DEGRADED	Loss of any two pitch or any two roll copilot SSC sensors.
27 FLT CTRL FAULT - PFCC 1 (2)(3) CUTOUT SW INOP	PFCC 1 (2) (3) CUTOUT SW contacts are not in the correct states.
27 FLT CTRL FAULT - L (R) GSHLD PTY SW INOP	Loss of any L (R) GSHLD PTY SW input.
27 FLT CTRL FAULT - L (R) SIDESTICK AP DISC/PTY SW INOP	Loss of any L (R) AP/DISC PTY SW input.
27 FLT CTRL FAULT - L (R) SIDESTICK SHAKER INOP	Loss of L (R) SSC shaker functionality.
27 FLT CTRL FAULT - L (R) AUTOPILOT SIDESTICK DETENT INOP	Loss of L (R) SSC AP lock functionality.
27 FLT CTRL FAULT - IIM INPUT REDUND LOSS	Loss of external system input to any IIM.
27 FLT CTRL FAULT - AFCU DMC INPUT REDUND LOSS	Loss of one DMC input to the AFCU.
27 FLT CTRL FAULT - IIM 1 (2) (3) INOP	Loss of IIM 1 (2) (3).
27 FLT CTRL FAULT - IIM INPUT REDUND LOSS	Loss of external system input to any IIM.

Table 5: INFO Messages

MESSAGE	LOGIC
27 FLT CTRL FAULT - AFCU SFECU INPUT REDUND LOSS	Loss of one DMC input to the AFCU.
27 FLT CTRL FAULT - AFCU SFECU INPUT INOP	Loss of SFECU inputs to the AFCU.
27 FLT CTRL FAULT - AFCU INOP	Loss of the AFCU.
27 FLT CTRL FAULT - DMC IIM INPUT REDUND LOSS	Loss of any input from any IIM to any DMC channel.
27 FLT CTRL FAULT - DMC AFCU INPUT REDUND LOSS	Loss of any input from any AFCU BUS to any DMC channel. NOTE: AFCU 1 and AFCU 2 data needs to be cross-talked between DMC channels to avoid potential CAS MISCOMPARE.
27 FLT CTRL FAULT - DIRECT MODE COM REDUND LOSS	Loss of partial or complete direct mode control path for only one control path.
27 FLT CTRL FAULT - REU DIRECT MODE DEGRADED	Any fault in the REU DM CLAWS.
27 FLT CTRL FAULT - MFS 1 (2) (3) (4) INOP	Loss of MFS 1 (2) (3) (4) REU or PCU.
27 FLT CTRL FAULT - SPOILER REU CCDL REDUND LOSS	Loss of CCDL communication between any two spoiler REUs.
27 FLT CTRL FAULT - PRIM REU INOP	Loss of any primary REU.
27 FLT CTRL FAULT - PRIM PCU INOP	Loss of any primary PCU.
27 FLT CTRL FAULT - PRIM PCU FAULT	Loss of soft damp SOV engagement in primary PCUs.
27 FLT CTRL FAULT - MCE CHAN 1 (2) INOP	Loss of MCE channel 1 (2).

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Table 5: INFO Messages

MESSAGE	LOGIC
27 FLT CTRL FAULT - PFCC ADS INPUT REDUND LOSS	Loss of o ADS1 or ADS2 input to any PFCC.
27 FLT CTRL FAULT - PFCC RAD ALT INPUT REDUND LOSS	Loss of 1 radio altimeter to any PFCC when only 2 radio altimeters installed or loss of 2 radio altimeters to any PFCC when 3 radio altimeters installed.
27 FLT CTRL FAULT - PFCC BDCU INPUT REDUND LOSS	Loss of one BDCU input to any PFCC.
27 FLT CTRL FAULT - PFCC IRS INPUT REDUND LOSS	Loss of one IRS input to any PFCC.
27 FLT CTRL FAULT - PFCC FADEC INPUT REDUND LOSS	Loss of one or more LGSCU inputs from one side to any PFCC.
27 FLT CTRL FAULT - PFCC LGSCU INPUT REDUND LOSS	Loss of one or more LGSCU inputs from one side to any PFCC.
27 FLT CTRL FAULT - PFCC STEEP APPR INPUT INOP	Loss of both FMS inputs to any PFCC.
27 FLT CTRL FAULT - PFCC INPUT REDUND LOSS	Loss of external system input to any PFCC.
27 FLT CTRL FAULT - PFCC 1 (2) (3) DEGRADED	Loss of PFCC 1 (2) (3).
27 FLT CTRL FAULT - AHRS INOP	Loss of AHRS or loss of AHRS input to any PFCC.
27 FLT CTRL FAULT - ISI INPUT INOP	Loss of ISI input to any PFCC.
27 FLT CTRL FAULT - OEI CONDITION DETECT	Direct mode OEI condition detect.
27 FLT CTRL FAULT - FBW FAULT	Abnormal output monitor or abnormal response monitor trips.
27 FLT CTRL FAULT - INPUT POWER REDUND LOSS	Loss of primary or secondary power input to LRUs with dual power input.

Table 5: INFO Messages

MESSAGE	LOGIC
27 FLT CTRL FAULT - L TOGA SW INOP	Loss of pilot TOGA switch input.
27 FLT CTRL FAULT - R TOGA SW INOP	Loss of copilot TOGA switch input.

PRACTICAL ASPECTS

FLIGHT CONTROL RIGGING REFERENCE MARKS

The aircraft has markings at the aileron, horizontal stabilizer, elevator, and rudder control surfaces to aid in rigging the control surfaces.

In addition to the markings, an inclinometer is installed when rigging the aileron or elevator. A goniometer is installed near the rudder power control units (PCUs). Refer to the Aircraft Maintenance Publication (AMP) for complete instructions.

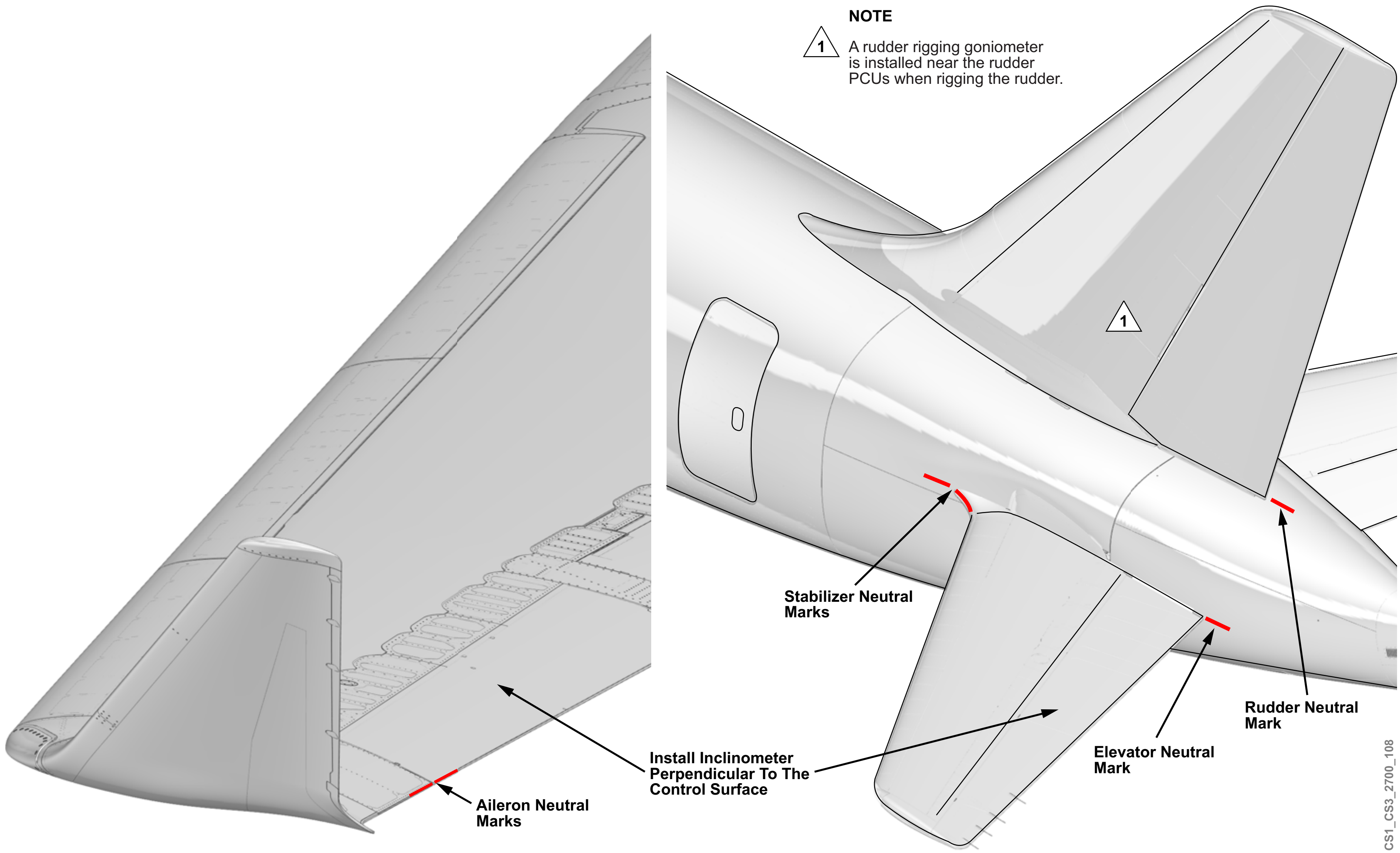

Figure 42: Flight Control Surface Rigging Reference Marks

FLIGHT CONTROL RIGGING

The electrical rigging is used to provide accurate adjusted sensor value measurements to the electronic flight control system. The aircraft must be in one of two states for rigging:

- Aircraft is on jacks without fuel
- Aircraft is weight on wheels (WOW) with 1134 to 1588 kg (2500 to 3500 lb) of fuel in each wing tank and a maximum of 907 kg (2000 lb) of fuel in center tank

The aircraft has markings at the aileron, horizontal stabilizer, elevator, and rudder control surfaces to aid in rigging the control surfaces.

In addition to the markings, an inclinometer is installed when rigging the aileron or elevator. A goniometer is installed near the rudder power control units (PCUs). Refer to the Aircraft Maintenance Publication (AMP) for complete instructions.

The rigging is carried out using a laptop with the Primary PCU Rigging Tool installed and an interface cable installed between the associated remote electronic unit (REU) and the aircraft wiring. The Primary PCU Rigging Tool program is used to operate the control surface being rigged.

The preconditions and status of the interfacing systems is displayed on the Primary PCU Rigging Tool program.

The control surface is moved to the neutral and full travel positions. At each position the control surface is adjusted to match the inclinometer or goniometer reading.

The results are stored in the REU NVM and the primary flight control computers (PFCCs).

Refer to the Aircraft Maintenance Publications for complete details.

NOTE

Stable conditions are necessary for this procedure. Do not do this test if there is noticeable wind or personnel walking in the aircraft.

WARNING

WHEN RIGGING, MAKE SURE THAT PERSONNEL AND EQUIPMENT ARE AWAY FROM THE FLIGHT CONTROL SURFACE. ACCIDENTAL FLIGHT CONTROL MOVEMENT CAN CAUSE INJURY TO PERSONNEL AND DAMAGE TO EQUIPMENT.

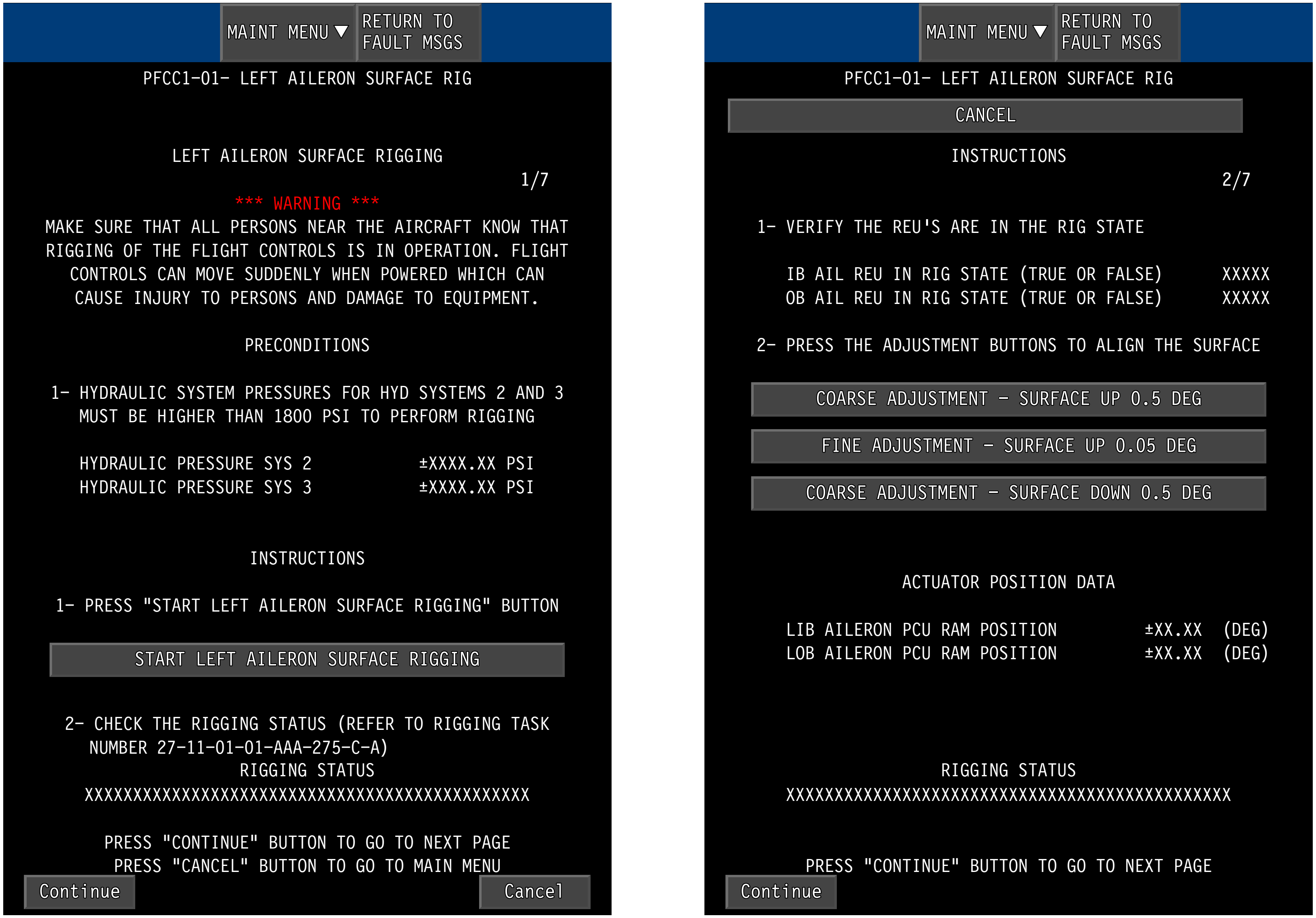

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Figure 43: Electronic Flight Control Surface Rigging

ONBOARD MAINTENANCE SYSTEM FUNCTIONS

The onboard maintenance system (OMS) provides the following functions:

- Aileron, elevator and rudder surface rigging
- Rudder pedal and rudder trim rigging
- Horizontal stabilizer trim actuator (HSTA) rigging
- Multifunction (MFS) spoiler rigging
- Autorigging of aileron, elevator and rudder PCUs
- Primary flight control computer (PFCC), inceptor interface module (IIM) and remote electronic unit (REU) quick rigging
- PFCC, IIM, REU, motor control electronics (MCEs) and non-volatile memory (NVM) clear
- Latched Fault Clearing

Aileron, Elevator, and Rudder Surface Rigging

During manual rigging on a control surface, the following rigging points are selected using the OMS:

- Null
- Full Extend
- Full Retract

At each rig point, the OMS provides coarse and fine adjustments to match the control surface position to an external reference. When the control surface is at the required position, the uncorrected position information is stored as rigging data in the non-volatile memory (NVM) of the associated REU.

A stroke verification test is run to confirm the PCU position. The stroke verification test makes sure the PCU reaches the electrical limits of travel before hitting the mechanical limits. If the test is successful, the amount of correction that was applied is stored in NVM to be used as a servo loop correction.

All of the data in the REU NVM is sent to the PFCCs for use in the quick rigging function.

Refer to the Aircraft Maintenance Publications for complete details.

NOTE

Stable conditions are necessary for this procedure. Do not do this test if there is noticeable wind or personnel walking in the aircraft.

WARNING

WHEN RIGGING, MAKE SURE THAT PERSONNEL AND EQUIPMENT ARE AWAY FROM THE FLIGHT CONTROL SURFACE. ACCIDENTAL FLIGHT CONTROL MOVEMENT CAN CAUSE INJURY TO PERSONNEL AND DAMAGE TO EQUIPMENT.

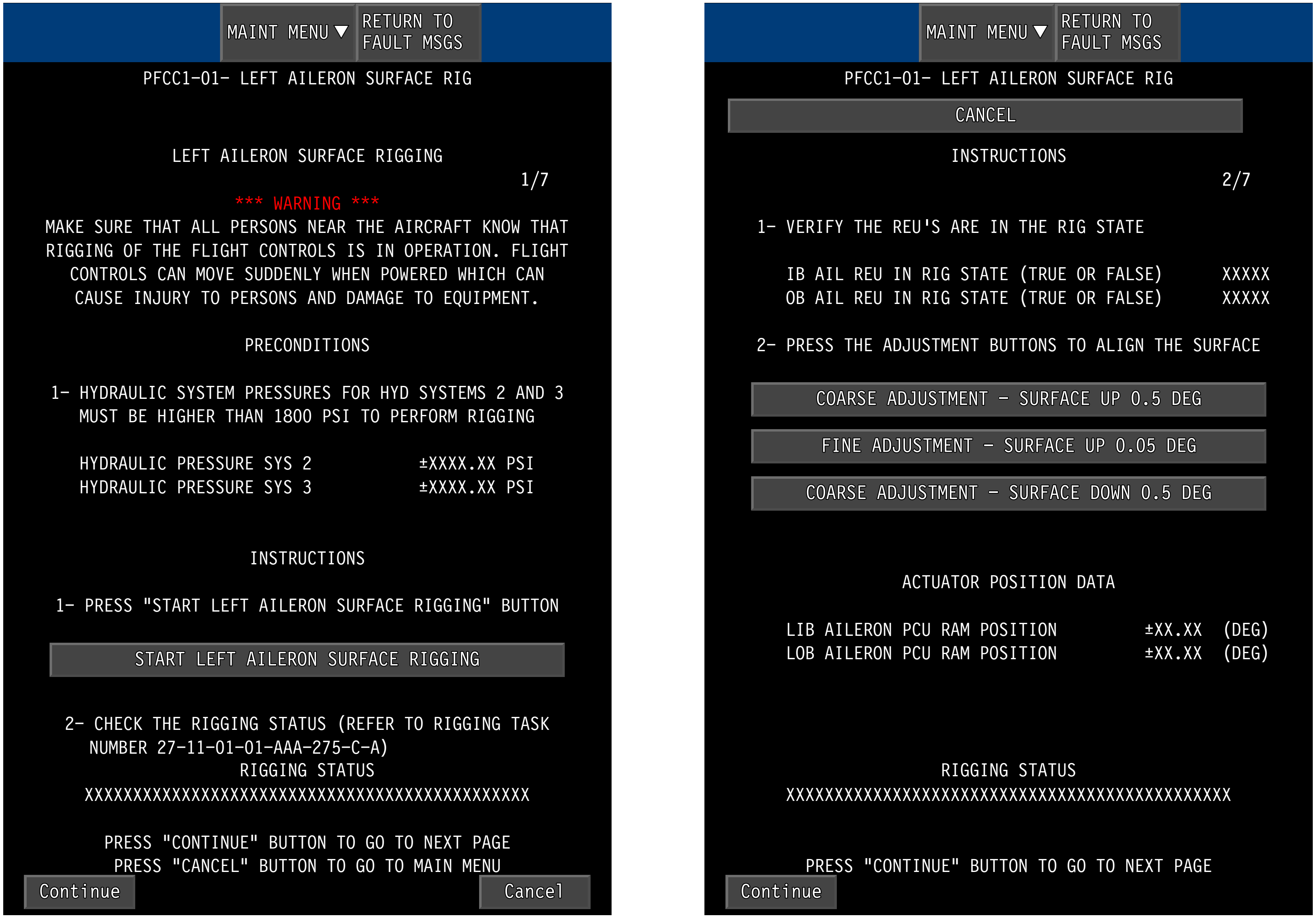

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Figure 44: Onboard Maintenance System Surface Rigging

ELECTRONIC RIGGING

The EFCS provides electronic rigging for use upon component replacement. Before beginning, the electronic rigging must be supported by either mechanical tools and/or alignment tools that establish fixed surface positions.

The electronic rigging is commanded through the OMS menu.

The OMS interfaces with the PFCC in command, which coordinates with REUs and IIMs for electronic rigging.

Electronic rigging is performed with hydraulics on.

AUTORIGGING

Autorigging is intended to avoid a full rigging following replacement of a single surface actuator. This type of rigging is performed using an automatic full stroke rigging procedure. This procedure determines when the surface has reached each of the three targeted measurement points, based on an already rigged redundant actuator LVDT sensor position.

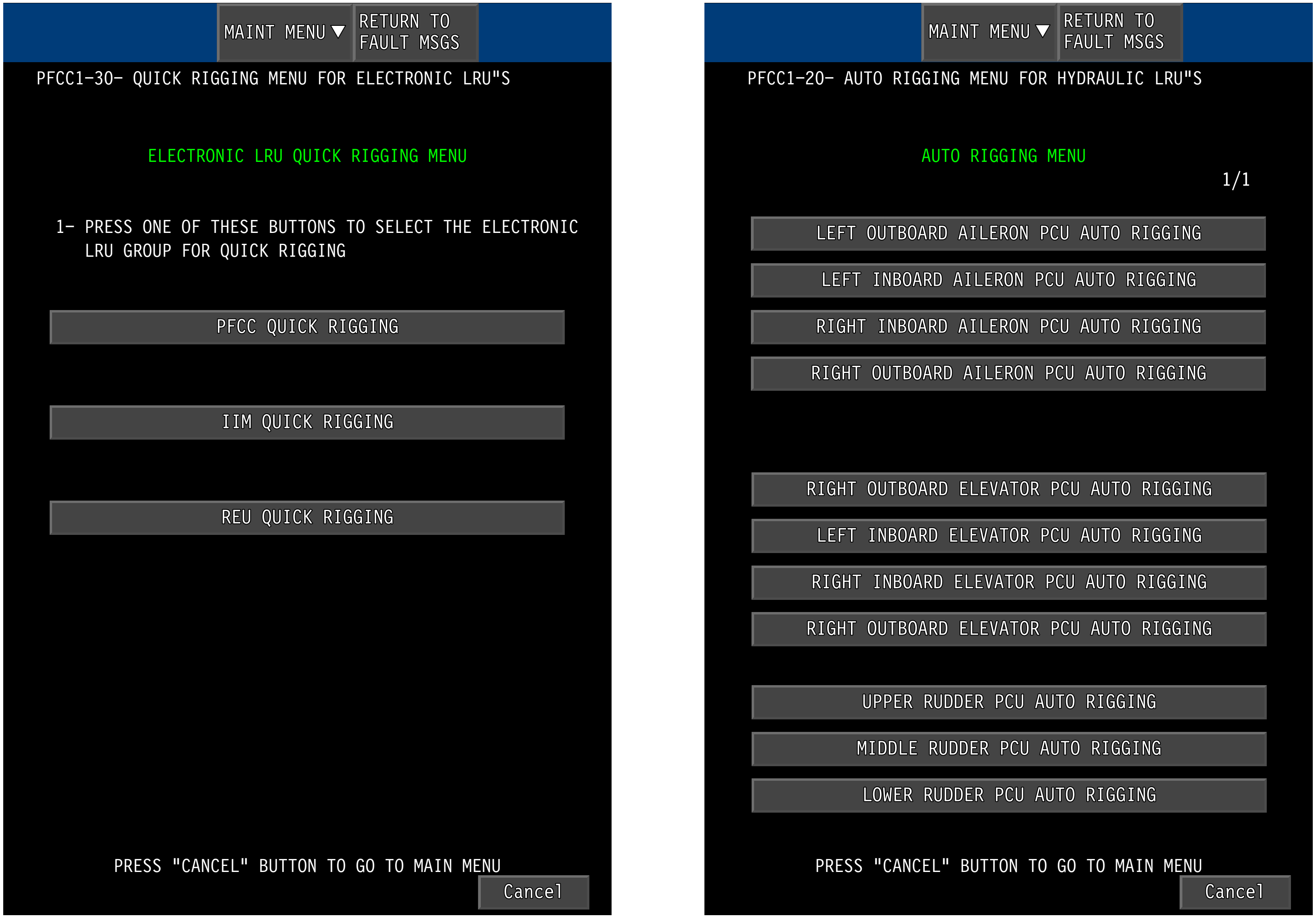

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Figure 45: Onboard Maintenance System Auto and Quick Rigging

ONBOARD MAINTENANCE SYSTEM TESTS

The onboard maintenance system (OMS) has provisions to perform the following tests:

- Automated built-in test (ABIT) manual test
- Initiated built-in test (IBIT) tests:
  - Aileron, elevator, rudder, MFS and ground spoiler actuation integrity tests
  - Sidestick controller shaker and autopilot detent test
  - All spoiler deployment test
- Power-up built-in test (PBIT) manual test

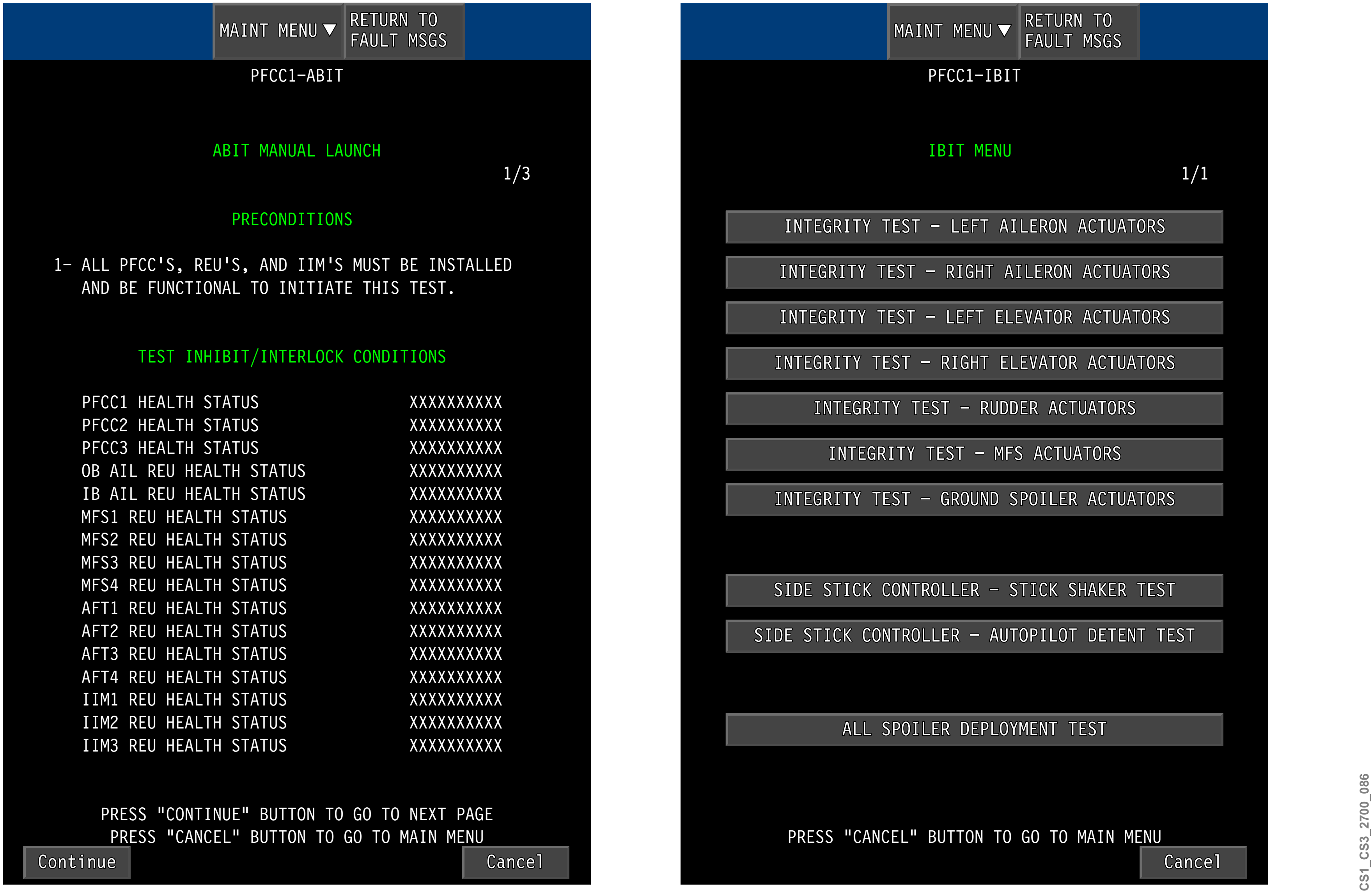

Figure 46: Onboard Maintenance System Test Functions

NVM CLEAR AND LATCHED FAULT CLEARING

NVM Clear

The NVM clearing is performed from the OMS under the Test tab. All electronic LRUs contain non-volatile memory (NVM). When the NVM is cleared, it is permanently erased.

The procedures for electronic rigging call for NVM clearing to be performed to clear out any faults that were stored, causing the component removal and subsequent re-rigging.

NVM clearing is required to be performed on the electronic LRUs associated with any rigging activity.

Latched Fault Clearing

A small subset of latched faults are retained in the reinstatement NVM.

Faults detected during ABIT, PBIT, and IBIT tests can only be cleared by rerunning the ABIT, PBIT, or IBIT test to successful completion.

Most CBIT faults can be cleared by cycling power to the electronic LRU.

Some CBIT faults are latched and can only be cleared by using the OMS Latched Fault Clearing. These faults may require additional maintenance inspections after clearing the latched fault. Latched faults in the electronic flight control system (EFCS) can only be cleared by the OMS Latched Fault Clearing.

Latched faults are identified in the specific procedures in the Aircraft Maintenance Publications.

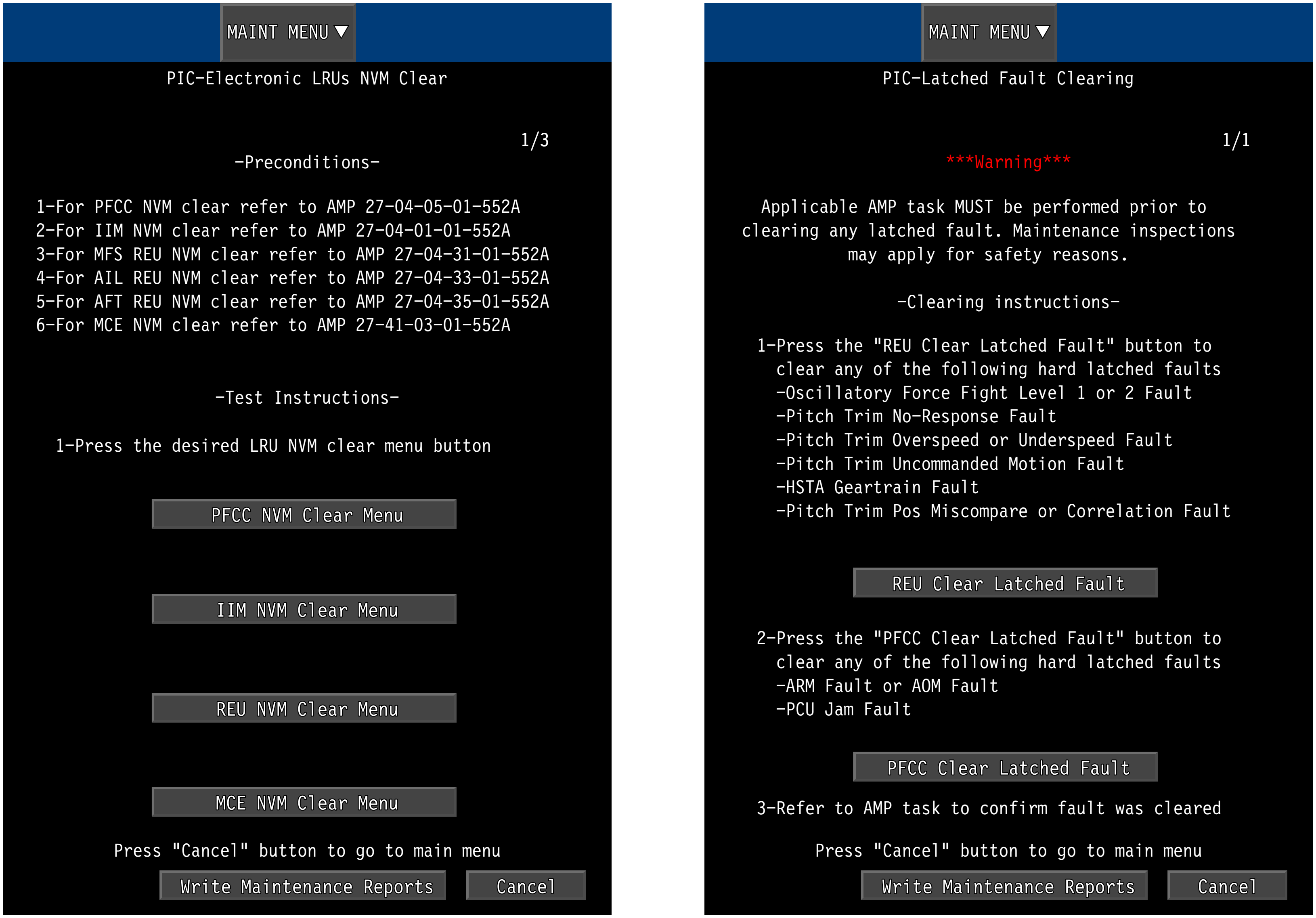

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Figure 47: NVM Clear and Latched Fault Clearing

27-10 AILERON SYSTEM

GENERAL DESCRIPTION

Primary roll control is provided by two aileron control surfaces, mounted on the outboard end of each wing rear spar. The ailerons are attached by four hinge fittings, with two additional attachments for the power control units (PCUs).

The ailerons normally operate asymmetrically, but may also operate symmetrically to perform lift modulation and load alleviation functions. In normal operation, the aileron range of movement is 25° trailing edge up (TEU), and 20° trailing edge down (TED). During flap extension, both ailerons deflect downward 10° to increase lift to assist the flaps. At touchdown, both ailerons deflect upward 10° to dump lift to assist the spoilers.

The PCUs are electrohydraulic units, and operate in an active-active mode (both PCUs are active). Hydraulic pressure for the inboard PCUs is supplied by hydraulic system no. 3, while hydraulic system no. 2 provides hydraulic pressure to the outboard PCUs.

Other than external dimensions and hydraulic sources, the aileron and elevator PCU operation and configuration is identical.

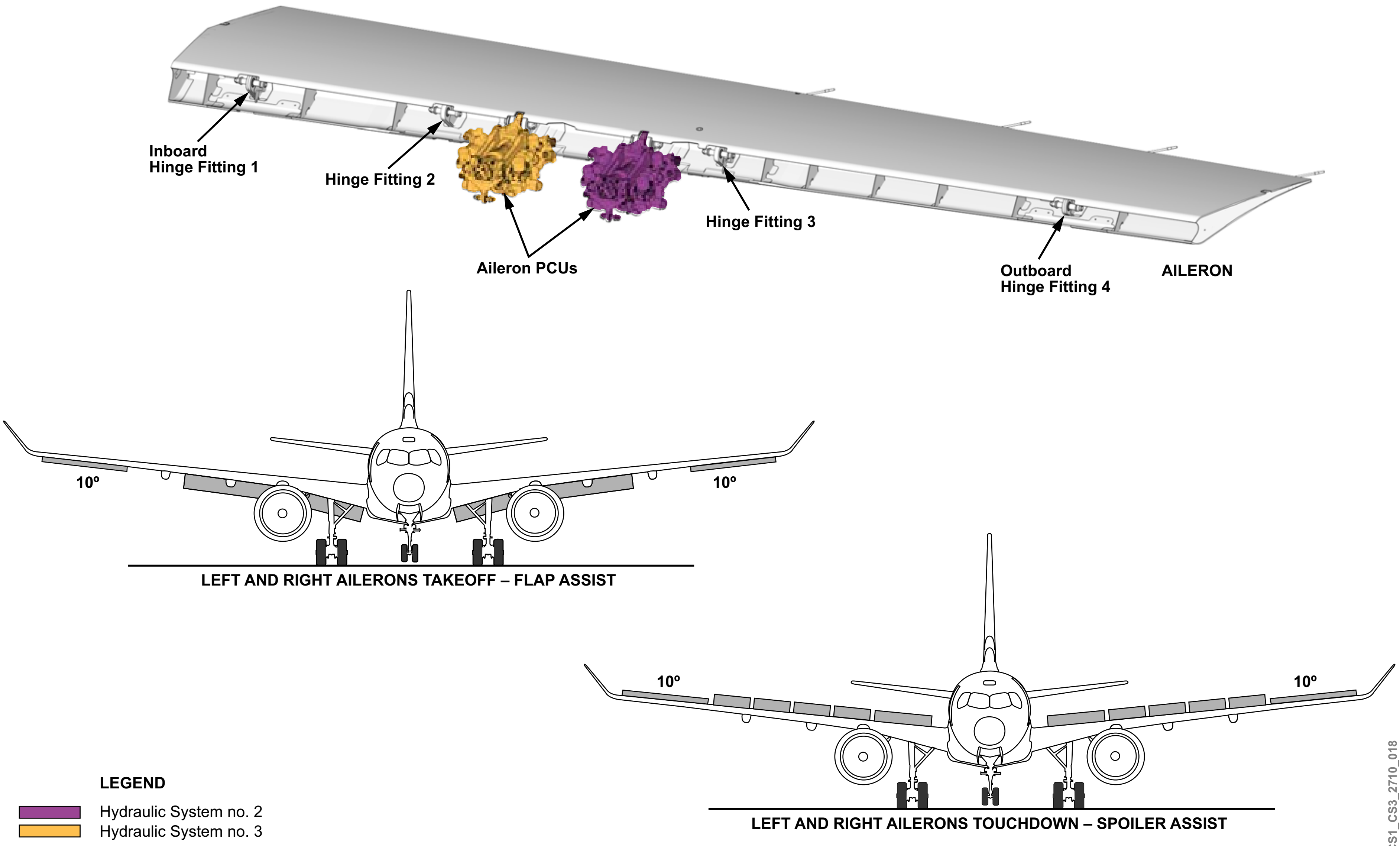


Figure 48: Aileron Overview

COMPONENT INFORMATION

AILERON POWER CONTROL UNITS

The role of the aileron power control unit (PCU) is to provide control of the aileron in response to commands. The PCU receives an electrical command from the remote electronic unit (REU), and transforms it into a hydraulic command to the control surface.

There are two PCUs per aileron. The aileron PCUs are located on the outboard portion of each wing, between the aileron and the wing rear spar. The aileron PCUs are attached to the wing rear spar by a kick link mechanism, which provides stability and prevents the PCU from moving up or down when operating. The PCU is attached to the aileron with a reaction link and an actuator. The reaction link acts as an additional pivot point for the aileron, which is deflected by the movement of the actuator.

An internal linear variable differential transformer (LVDT), connected to the PCU actuator, provides information on the position of the PCU.

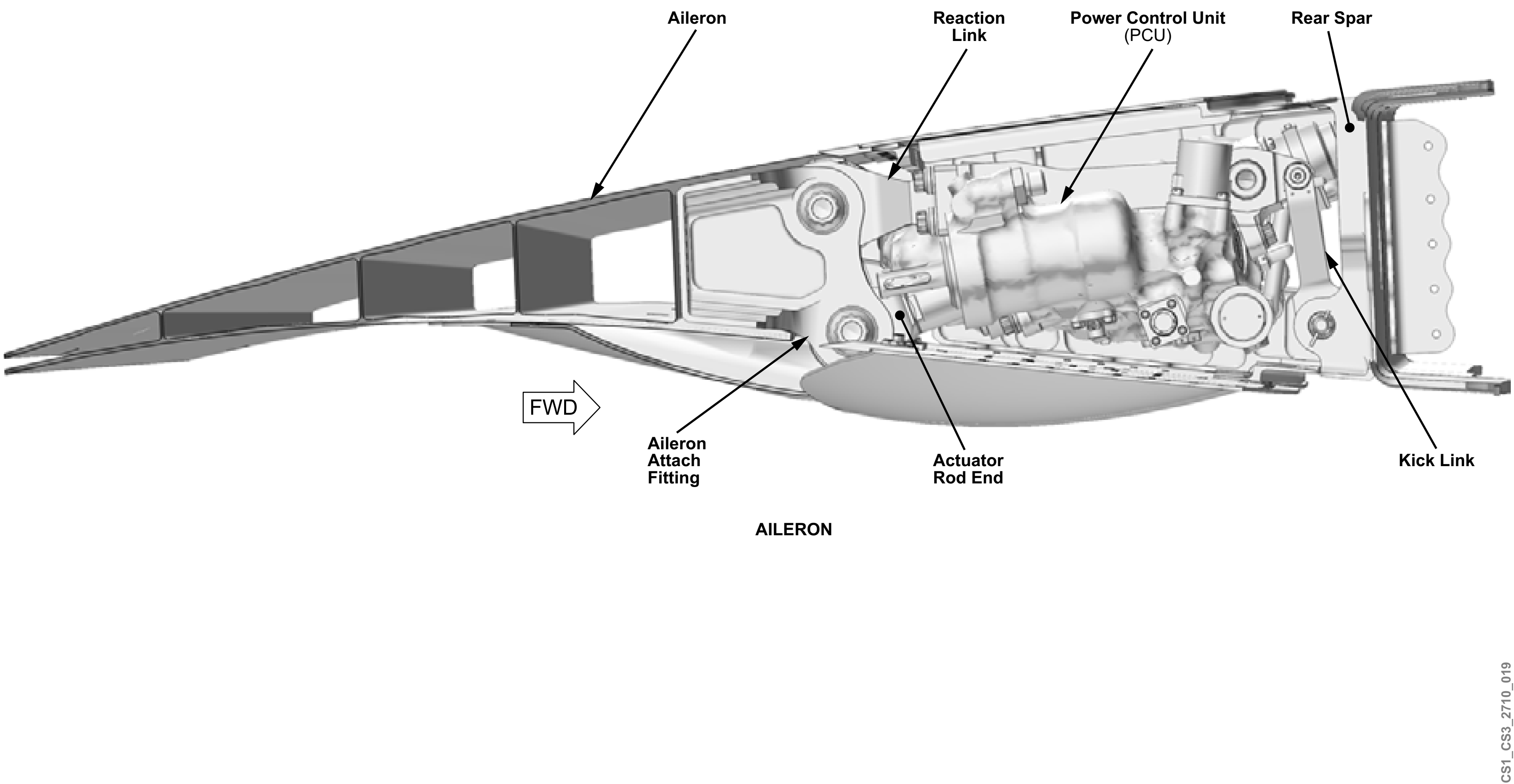


Figure 49: Aileron Power Control Unit

DETAILED COMPONENT INFORMATION

AILERON POWER CONTROL UNIT

The power control unit (PCU) in-line strainer protects the PCU from particulate contamination that is large enough to cause a malfunction of the PCU. The strainer is replaceable without removing the actuator from the aircraft.

A linear variable differential transformer (LVDT) measures the position of the piston relative to the cylinder. The LVDT information is also used for autorigging.

A spring-loaded piston compensator is provided to compensate for PCU internal leakage, external leakage, and fluid thermal contraction, ensuring primary PCU damping capability when hydraulic pressure is released. There is a level indicator for fluid level measurement.

The PCU uses a torque motor controlled, electrohydraulic servovalve (EHSV) to meter hydraulic fluid flow to the actuator in response to signals received from the REU. An LVDT provides feedback of EHSV position.

The two internal shutoff valves (SOVs) are three-way, two-position, normally closed valves. The SOVs control the PCU mode select valve (MSV) position.

The function of the MSV is to direct fluid pressure to the extend and retract ports of the actuator. In damped mode, the MSV connects the actuator extend and retract ports to each other, and to the compensator through fixed damping orifices.

The PCU has two dual-sensor pressure transducers, which are used to monitor the pressure in the chambers of the hydraulic actuator (i.e. chamber extend-CE and chamber retract-CR).

The PCU incorporates a pressure-relief valve (PRV). The pressure-relief valve ports hydraulic fluid from the extend or retract port to the system return if the pressure reaches 3900 psi.

The PCU has two inlet return check valves (IRCVs). The valves keep pressure trapped in the PCU to provide surface damping should a loss of hydraulic power occur. The internal pressure can be released by manually operating the IRCVs.

Anticavitation check valves prevent back flow of fluid while the system is active. Other check valves are used for the same purpose.

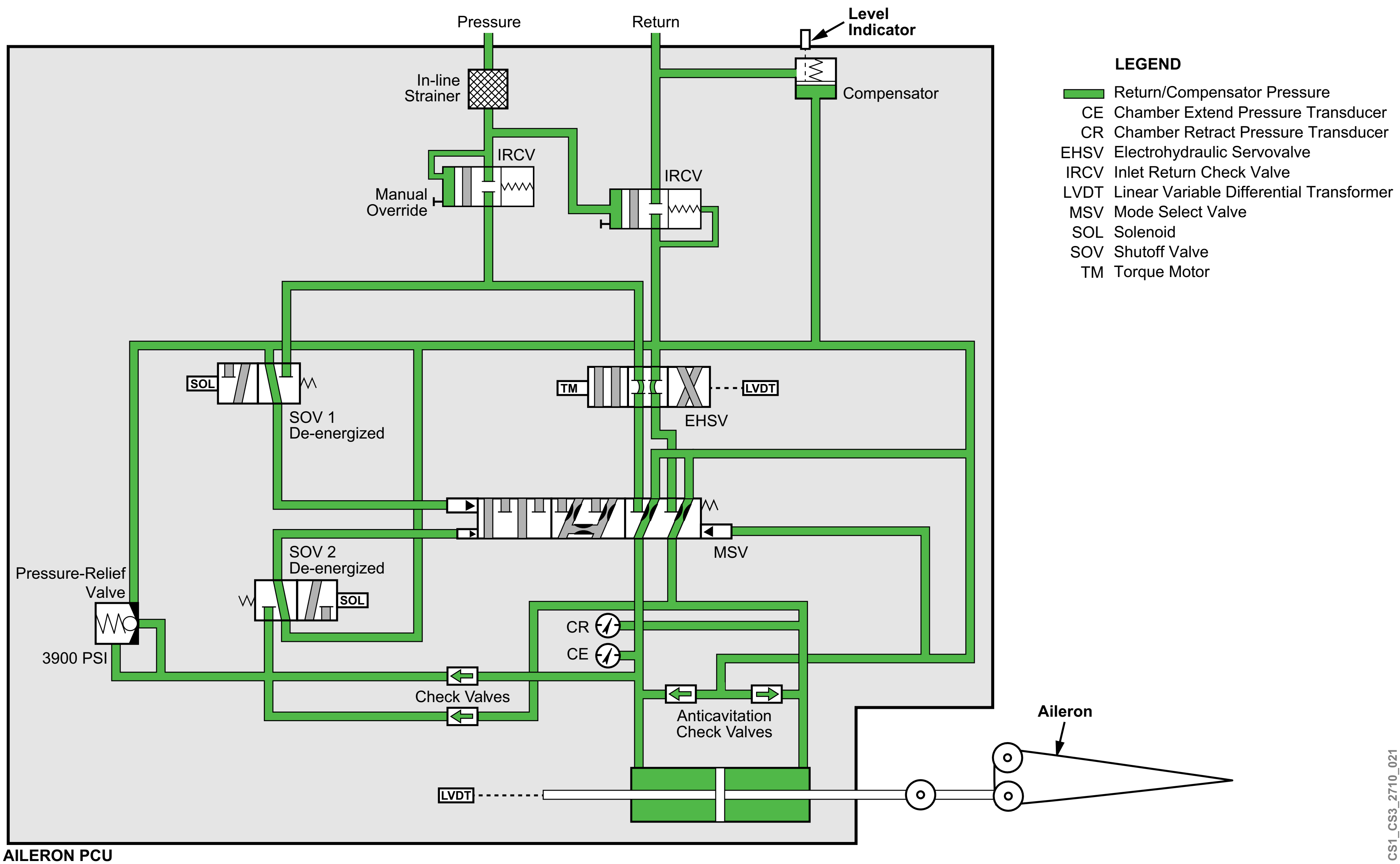


Figure 50: Aileron PCU Schematic

AILERON POWER CONTROL UNIT STATES

The aileron PCU has the following states of operation:

- Active state
- Soft damped state
- Hard damped state

In case of a malfunction, the PCU is put into the soft damped state to prevent force fighting between the other PCU on the same surface. One PCU will be in the active state, while the other is in the soft damped state.

A PCU in the hard damped state prevents the surface from moving and acts as a gust lock for that surface.

Active State

In active mode, the aileron REU sends a signal to SOV 1 of the PCU, and positions the valve to allow hydraulic pressure to act on the active solenoid of the MSV. SOV 2 may or may not be energized. At the same time, the REU signals the EHSV torque motor to allow fluid pressure to the MSV. The MSV routes fluid pressure to act upon the actuator, moving the aileron surface. Position signals from the EHSV LVDT are sent to the aileron REU, while actuator LVDT signals are sent to a separate REU. The signals are compared using the cross-channel communication link (CCDL), to ensure actuator position matches EHSV position.

Pressure transducers are used to measure extend (CE) and retract chamber (CR) pressures of the hydraulic actuator and provide feedback to the REU. The differential pressure data is used for active/active force fight mitigation between two adjacent PCUs on a given primary surface, and for oscillatory malfunction detection.

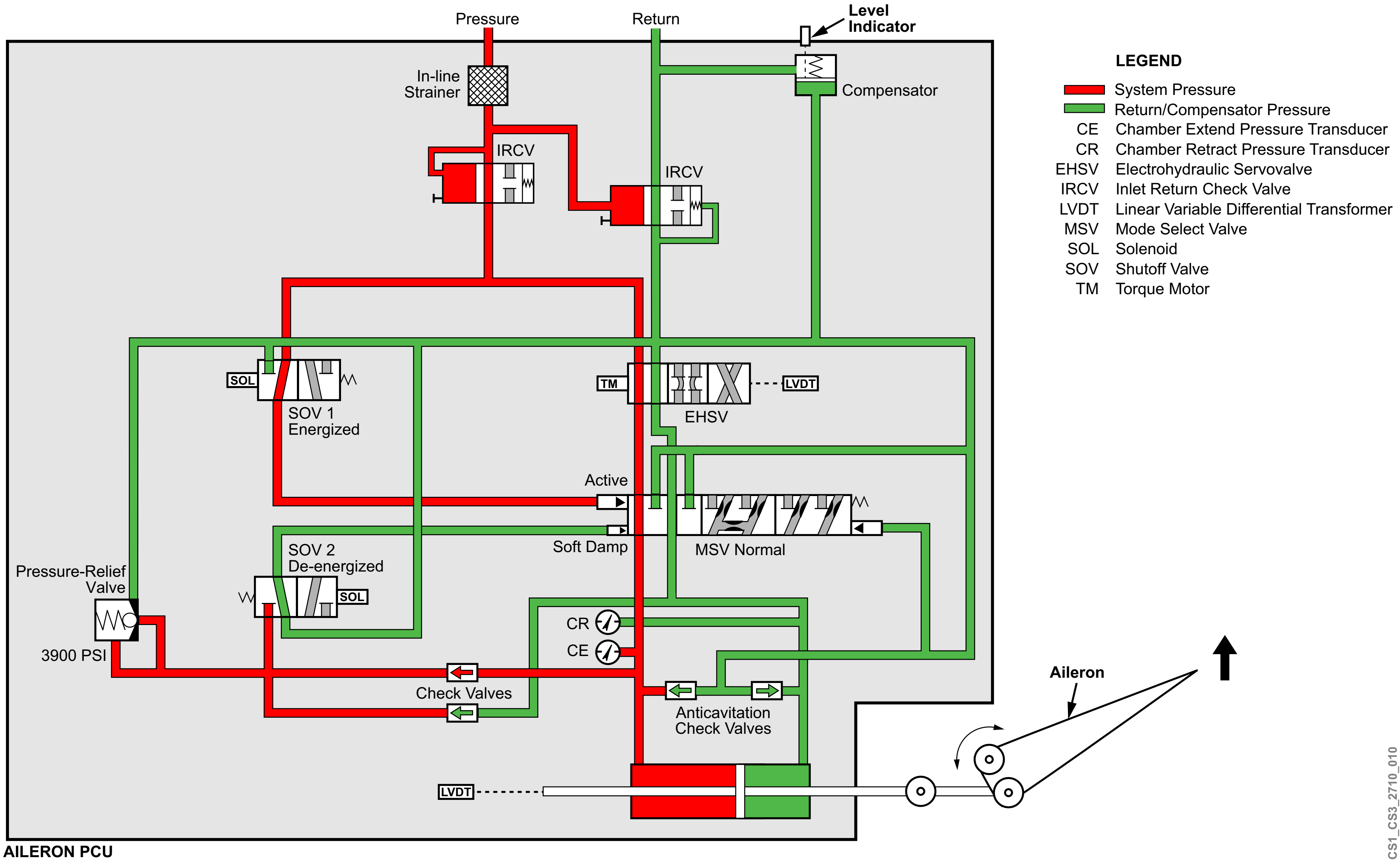


Figure 51: Aileron PCU - Active State

Soft Damped State

In the soft damped state, the REU energizes SOV 2, which positions the MSV to connect the extend and retract ports through a restricted orifice in the MSV.

In this position, fluid is allowed to transfer through the MSV through restricted orifices. EHSV and actuator position, as well as actuator hydraulic pressures, are monitored as in the active state. The damping pressure is maintained and controlled by the compensator, which provides fluid volume, and prevents hydraulic cavitation.

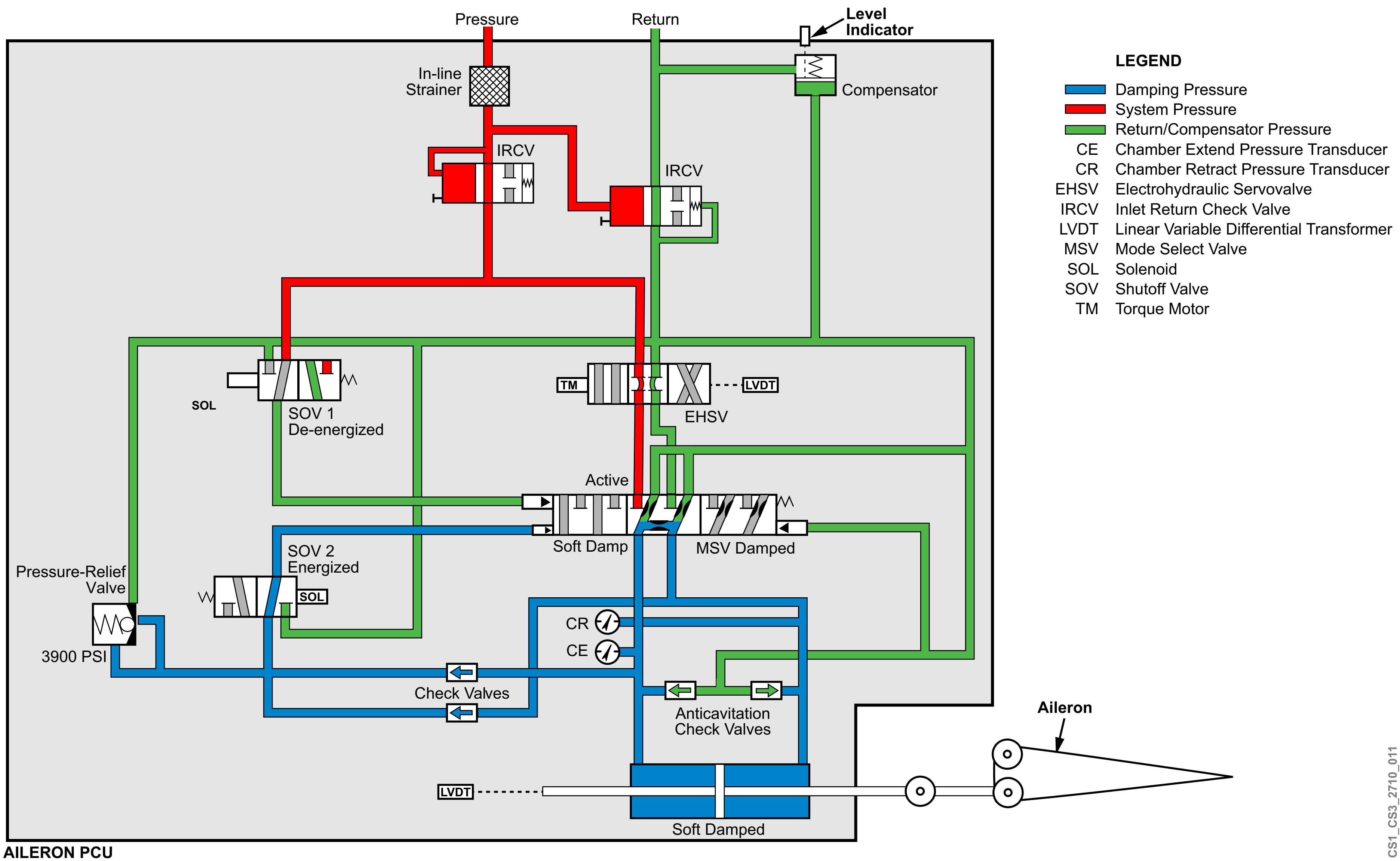


Figure 52: Aileron PCU - Soft Damped State

Hard Damped State

In the hard damped state both SOV 1 and SOV 2 are de-energized. The EHSV is commanded to a neutral position, and the MSV will move to a hard damped position, trapping fluid under pressure between the extend and retract ports of the actuator.

In order to prevent force fight, when one PCU on a given control surface enters a hard damped state, the adjacent PCU is commanded to a hard damped state.

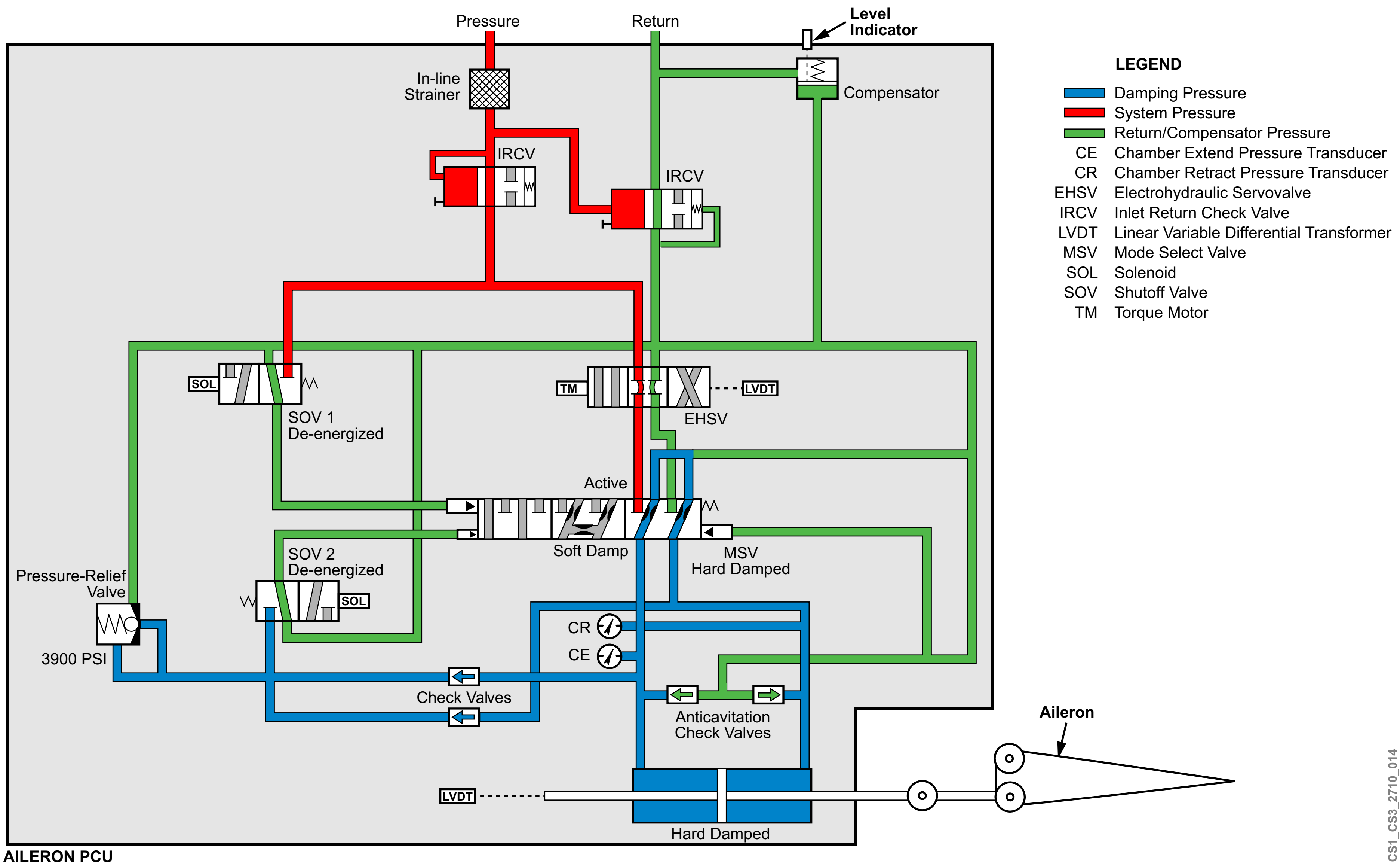
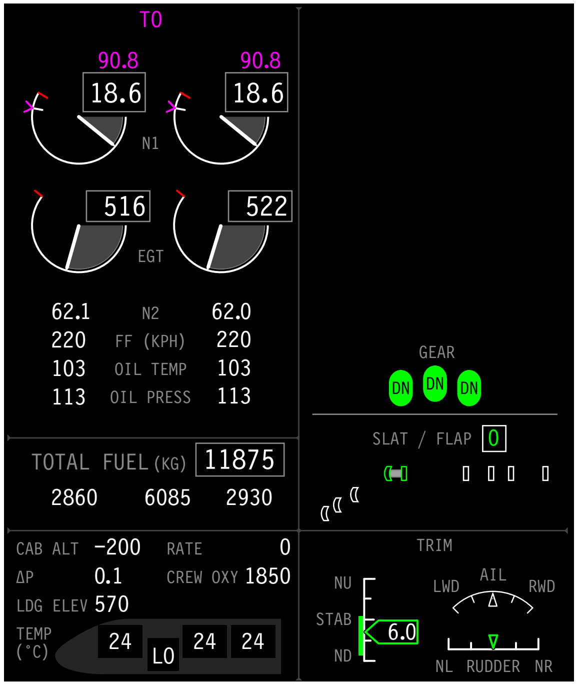


Figure 53: Aileron PCU - Hard Damped State

CONTROLS AND INDICATIONS

The aileron trim position and status in the direct mode only are displayed on the engine indication and crew alerting system (EICAS) page.

The aileron position and status are displayed on the flight controls synoptic page.



EICAS PAGE

AILERON SURFACE STATE	
Symbol	**Condition**
	Normal
	Fail
	Invalid

AILERON POSITION INDICATOR	
Symbol	**Condition**
	Normal
	Fail
	Invalid

AILERON TRIM INDICATOR (DISPLAYED IN DIRECT MODE ONLY)	
Symbol	**Condition**
Δ	In flight or on ground
	Invalid

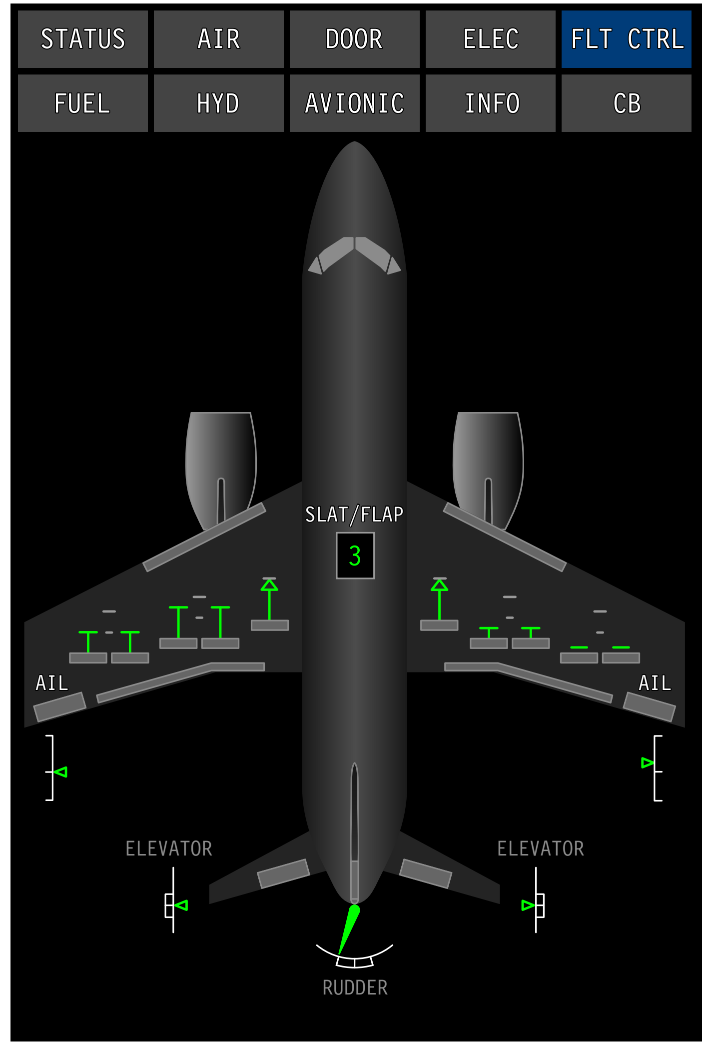


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Figure 54: Aileron Surface and Aileron Trim Indications

DETAILED DESCRIPTION

AILERON POSITION AND AILERON TRIM INDICATING

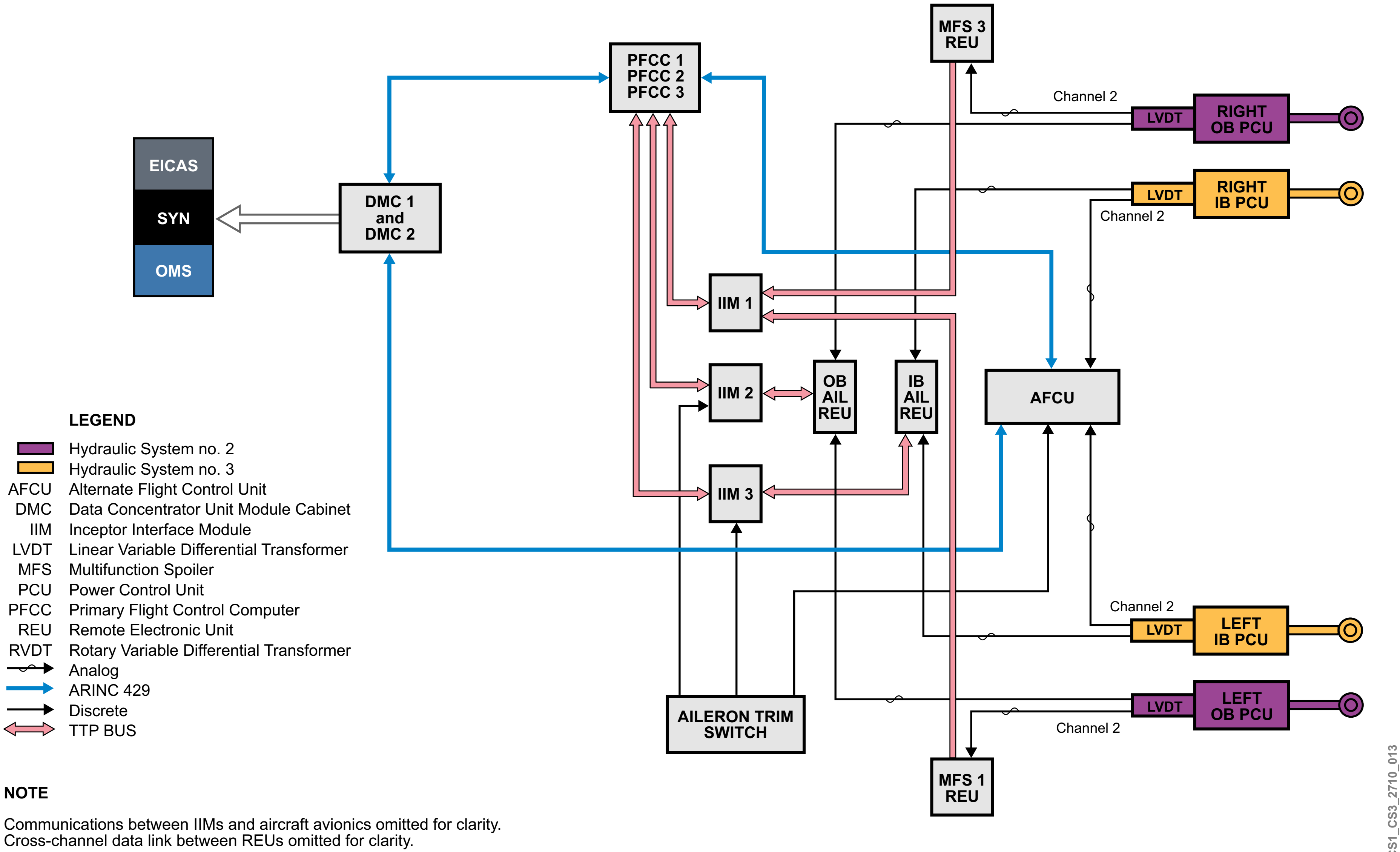
Aileron position information is derived from analog signals from the power control unit (PCU) actuator linear variable differential transformers (LVDTs). The signals are sent to remote electronic units (REUs).

The REUs transmit position data over TTP BUSES to inceptor interface modules (IIMs) which route the information to the primary flight control computers (PFCCs). The PFCCs use this information for flight control purposes.

The inboard PCUs send position information also to the alternate flight control unit (AFCU) for use during AFCU direct mode operation.

The aileron position signals are transmitted to the data concentrator unit module cabinets (DMCs) over ARINC 429 BUSES. The DMCs provide information for display on EICAS, flight controls synoptic page, and to the onboard maintenance system (OMS).

Aileron trim switch selection information is processed by the IIMs which provides trim information to the PFCCs, then to the EICAS page. Aileron trim position and status is displayed on the EICAS page when operating in direct mode.



NOTE

Communications between IIMs and aircraft avionics omitted for clarity.
Cross-channel data link between REUs omitted for clarity.

Figure 55: Aileron Position and Aileron Trim Indicating

MONITORING AND TESTS

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

CAS MESSAGES

Table 6: CAUTION Message

MESSAGE	LOGIC
AILERON FAIL	Left and right ailerons failed.

Table 7: ADVISORY Message

MESSAGE	LOGIC
FLT CTRL FAULT	Loss of redundant or non-critical function in the primary flight control system. Refer to INFO.

Table 8: INFO Messages

MESSAGE	LOGIC
27 FLT CTRL FAULT - AILERON TRIM SW REDUND LOSS	Loss of the aileron trim switch.
27 FLT CTRL FAULT - PRIM PCU INOP	Loss of any primary PCU.
27 FLT CTRL FAULT - PRIM PCU FAULT	Loss of soft damp SOV engagement in primary PCUs.

PRACTICAL ASPECTS

POWER CONTROL UNIT COMPENSATOR DEPLETION

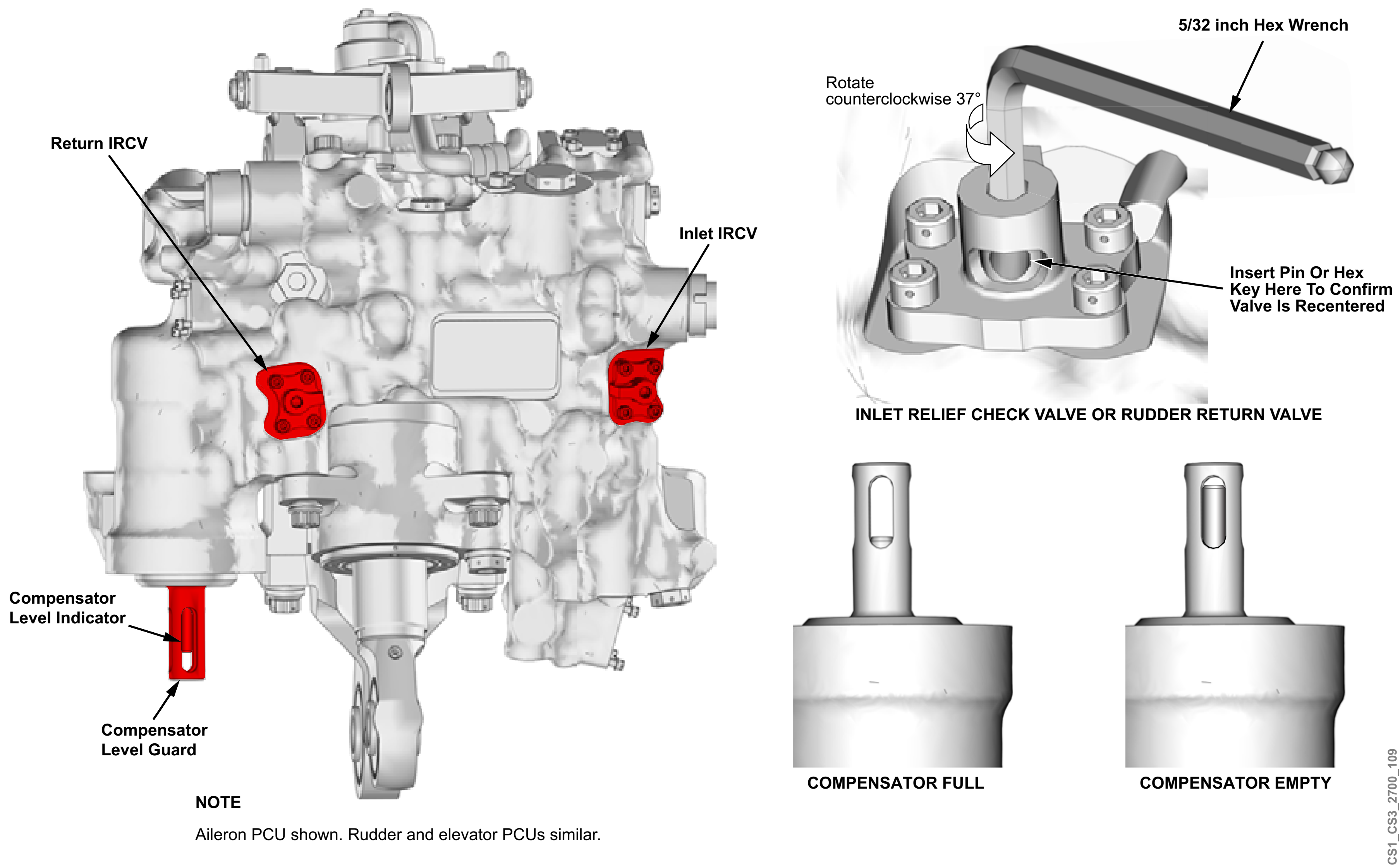
When performing maintenance on a PCU, the PCU must be depressurized. The PCU is depressurized by depleting the compensator using one of the inlet return check valves (IRCVs) on the aileron or elevator PCUs or the rudder return check valve (RRV) on the rudder PCU.

To depressurize the PCU, use a 5/32 hex wrench to rotate the inlet return check valve (IRCV) or rudder return check valve (RRV) counterclockwise until it reaches the hard stop at 37° from neutral.

The compensator is empty when the level indicator is at the bottom of the compensator level guard.

NOTE

To check that the IRCV or RRV has been recentered after being operated, a pin or hex key can be inserted from the hole on the side. If the pin or hex key can not pass through the hole, the valve is recentered.



NOTE

Aileron PCU shown. Rudder and elevator PCUs similar.

Figure 56: PCU Compensator Depletion

27-20 RUDDER SYSTEM

GENERAL DESCRIPTION

Yaw control is provided by a single rudder attached to the rear spar of the vertical stabilizer. The rudder is attached by eight hinge fittings, plus three additional power control units (PCUs) attachment points. The rudder is powered by three PCUs in an active/active/active state when operating in normal mode. The upper PCU is powered by hydraulic system no. 1, the middle PCU is powered by hydraulic system no. 2 and the lower by hydraulic system no. 3. The rudder control system senses yaw commands from the pilot and copilot rudder pedal assemblies, and transmits pedal position signals to the electronic flight control system (EFCS). Under normal conditions, the rudder range of movement is 32° either side of center.

The rudder PCUs operate in the following states:

- Active state: Actuator is actively providing position control under hydraulic power
- Damped state: Actuator is inactive and provide damping

The PCU is damped at a rate that permits one or two adjacent active PCU(s) to meet performance requirements, while providing acceptable damping to meet ground gust impact requirements.

The rudder surface operates in the following surface states:

- Active-active-active state: All three actuators are actively providing position control under hydraulic power
- Active-active-damped state: Two actuators are actively providing position control under hydraulic power, the remaining actuator is inactive and providing damping
- Active-damped-damped state: A single actuator is actively providing position control, with the remaining actuators providing damping
- Damped-damped-damped state: Three damped actuators provide surface flutter protection in flight or ground gust protection when the airplane is shut down on the ground

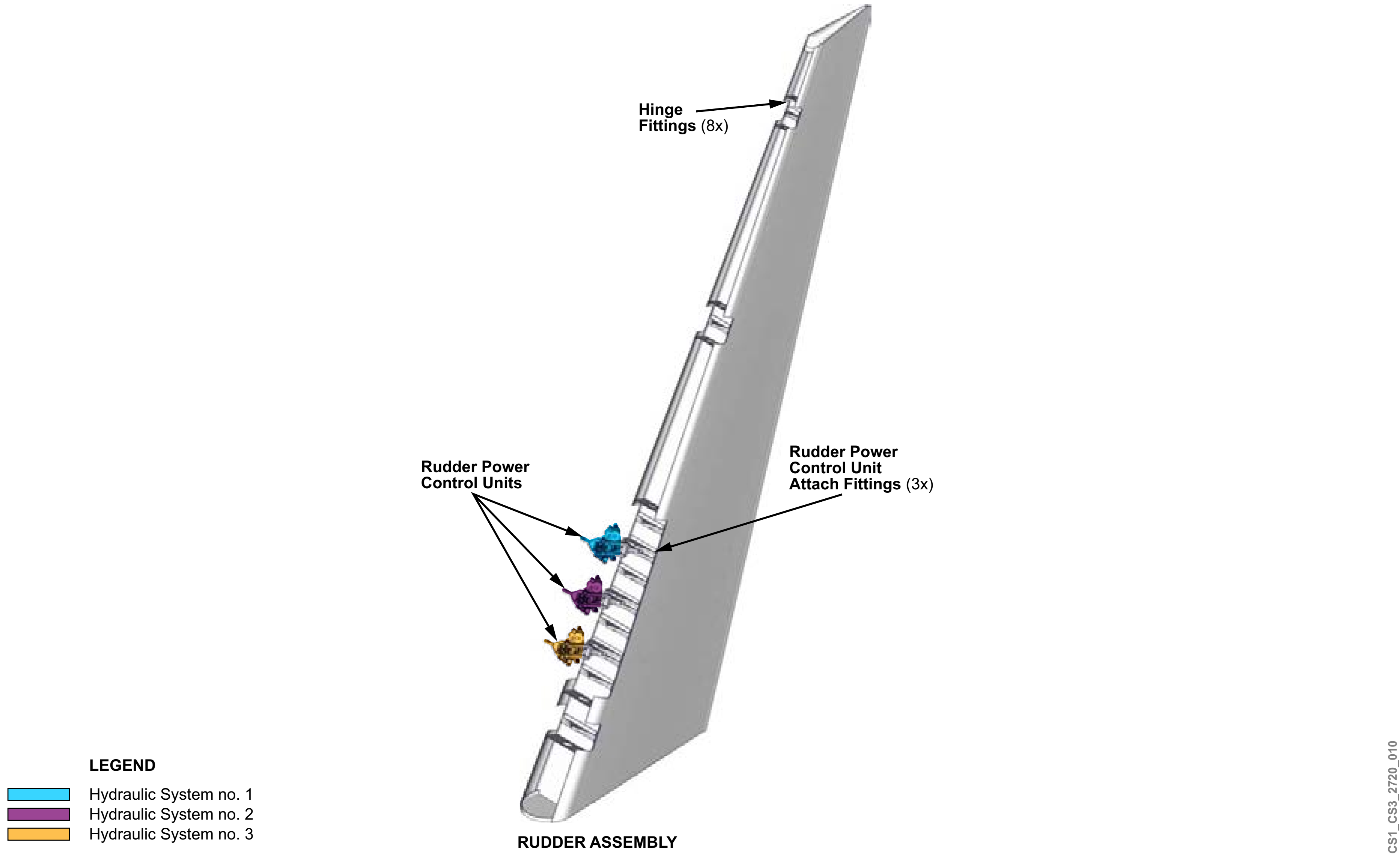


Figure 57: Rudder Overview

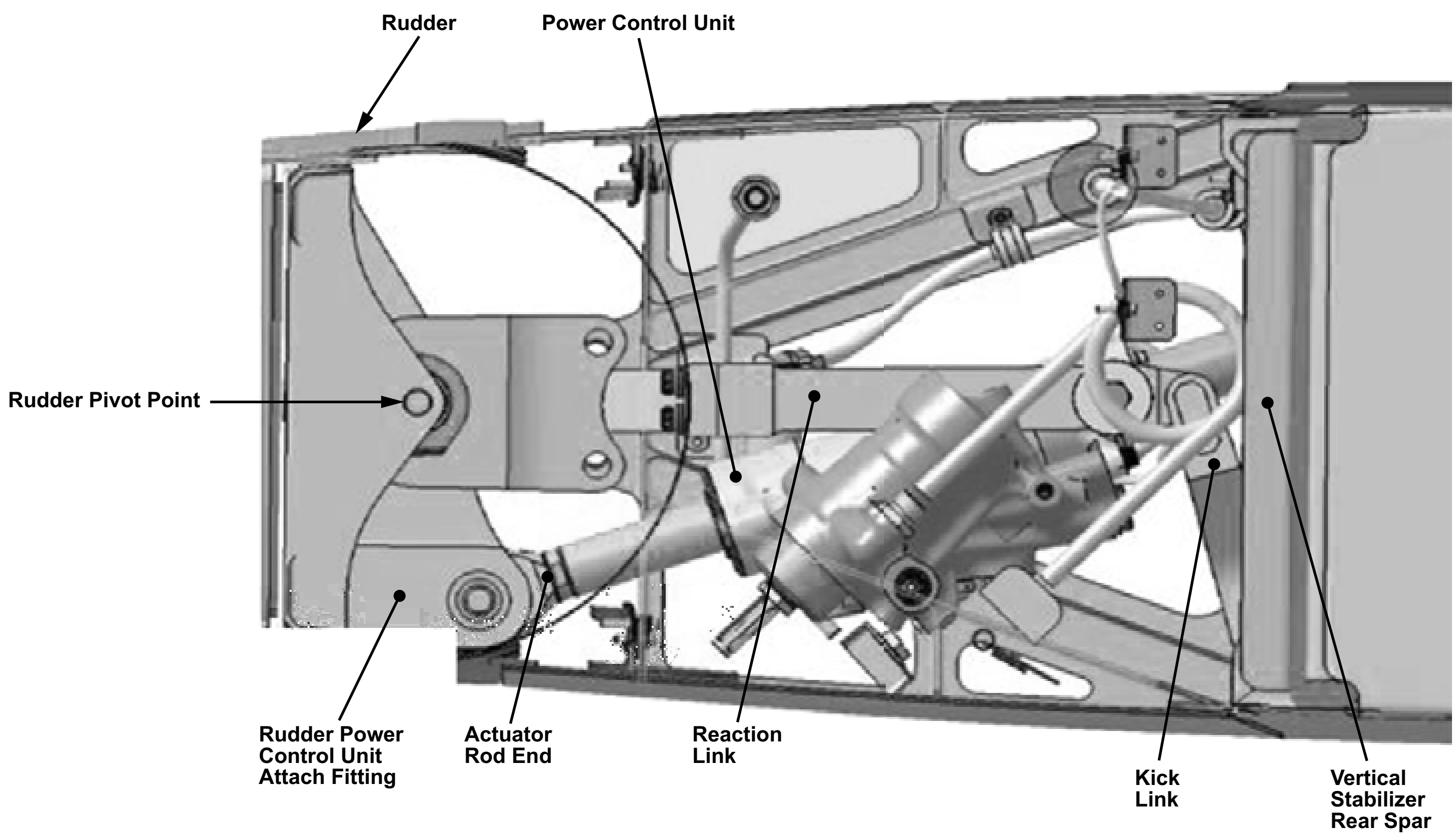
COMPONENT INFORMATION

RUDDER POWER CONTROL UNIT

The role of the power control unit (PCU) is to provide control of the rudder in response to pilot command. The PCU receives an electrical command from the remote electronic unit (REU) and transforms it into a hydraulic command to the control surface.

The rudder PCUs are attached to the vertical stabilizer rear spar by a kick link mechanism, which provides stability and prevents the PCU from moving up or down when operating. The PCU is attached to the rudder with a reaction link and an actuator. The reaction link acts as an additional pivot point for the rudder, which is deflected by the movement of the actuator.

An internal linear variable differential transformer (LVDT), connected to the PCU actuator, provides information of the actual position of the PCU.



NOTE

Looking down at upper power control unit.

Figure 58: Rudder Power Control Unit

DETAILED COMPONENT INFORMATION

RUDDER POWER CONTROL UNIT

The power control unit (PCU) in-line strainer protects the PCU from particulate contamination that is large enough to cause a malfunction of the PCU. The hydraulic in-line strainer is replaceable without removing the actuator from the aircraft.

A linear variable differential transformer (LVDT) measures the position of the piston relative to the cylinder. The LVDT information is also used for auto rigging.

A spring-loaded piston compensator is provided to compensate for PCU internal leakage, external leakage and fluid thermal contraction ensuring primary PCU damping capability when hydraulic pressure is released. There is also a level indicator for fluid level measurement.

The PCU uses a torque motor controlled, electrohydraulic servovalve (EHSV) to meter hydraulic fluid flow to the actuator in response to signals received from the REU. An LVDT provides feedback of EHSV position.

The shutoff valve (SOV) is a three-way, two-position, normally closed valve. The SOV controls the mode select valve (MSV) position.

The function of the MSV is to direct fluid pressure to the extend and retract ports of the actuator. In damped mode, the MSV connects the actuator extend and retract ports to each other and to the compensator through fixed damping orifices.

The PCU has two, dual-sensor pressure transducers, which are used to monitor the pressures in the chambers of the hydraulic actuator (i.e. chamber extend-CE and chamber retract-CR).

For load relief protection, the PCU incorporates a pressure-relief valve (PRV). The PRV will port hydraulic fluid from the extend or retract port to the compensator when extend or retract pressure reaches 3900 psi.

The PCU also has inlet, return and anticavitation check valves. The return relief valve is a spring-loaded type check valve. It keeps the pressure trapped in the PCU in case of hydraulic power loss. The return relief valve allows maintenance to release hydraulic pressure by depressing the relief valve.

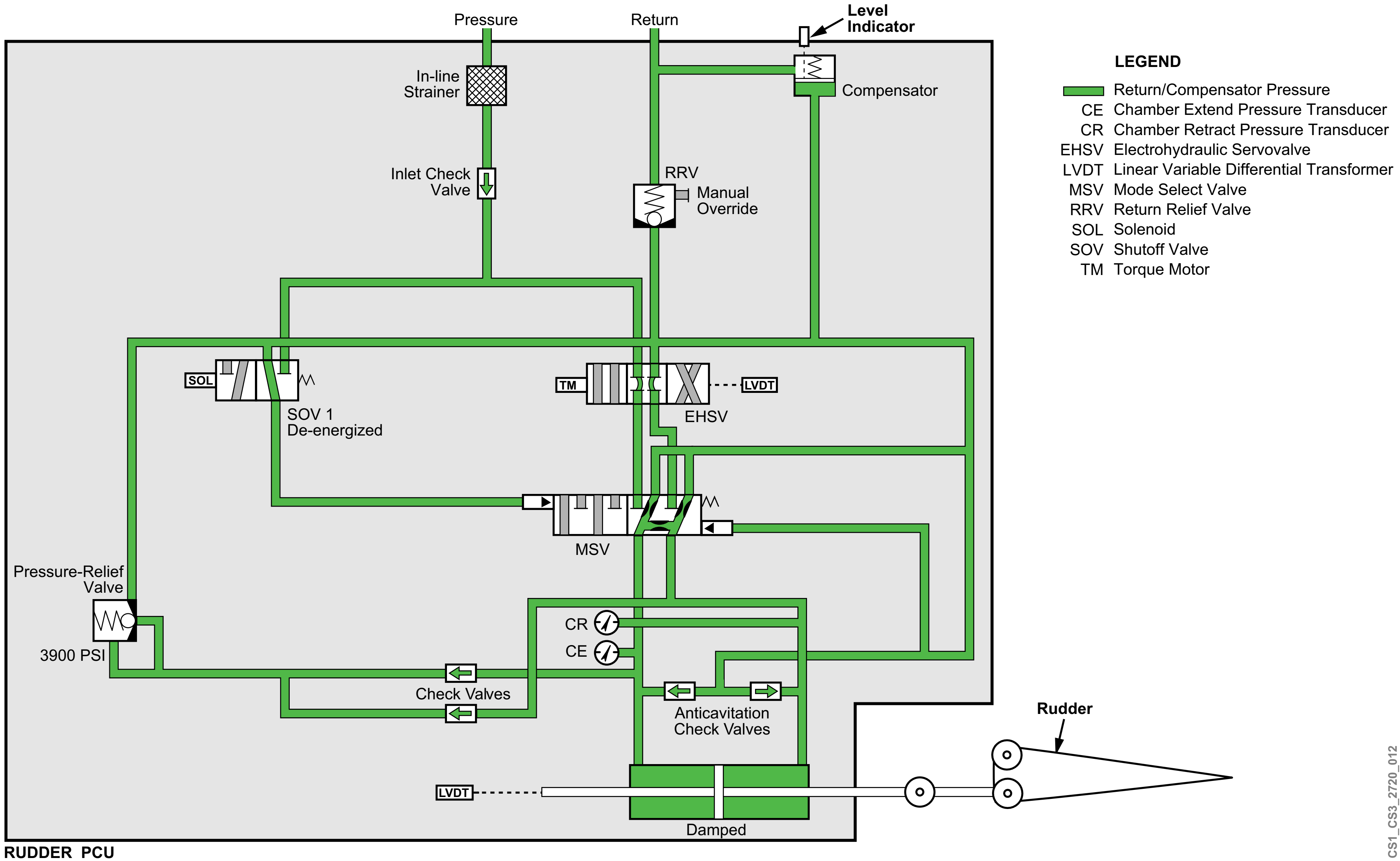


Figure 59: Rudder Power Control Unit Schematic

RUDDER POWER CONTROL UNIT STATES

The rudder PCU has the following states of operation:

- Active state
- Damped state

A PCU may be put into the damped state to prevent force fighting between PCUs on the same surface.

Active State

The rudder PCU is in the active state when the SOV is electrically energized and hydraulic supply pressure is greater than 1000 psi. In active mode, the REU sends a signal to the SOV of the PCU, and positions the valve to allow hydraulic pressure to act on the active solenoid of the MSV.

At the same time, the REU signals the EHSV torque motor to allow fluid pressure to the MSV. The MSV directs fluid pressure to the actuator to move the rudder surface. Position signals from the EHSV LVDT are sent to the rudder REU, while the actuator LVDT position signals are sent to a separate REU. The signals are compared using the CCDL BUS, to ensure actuator position matches EHSV position.

Pressure transducers are used to measure extend and retract chamber pressures of the hydraulic actuator, and provide feedback to the REU. The differential pressure data is used for active/active force fight mitigation between the PCUs.

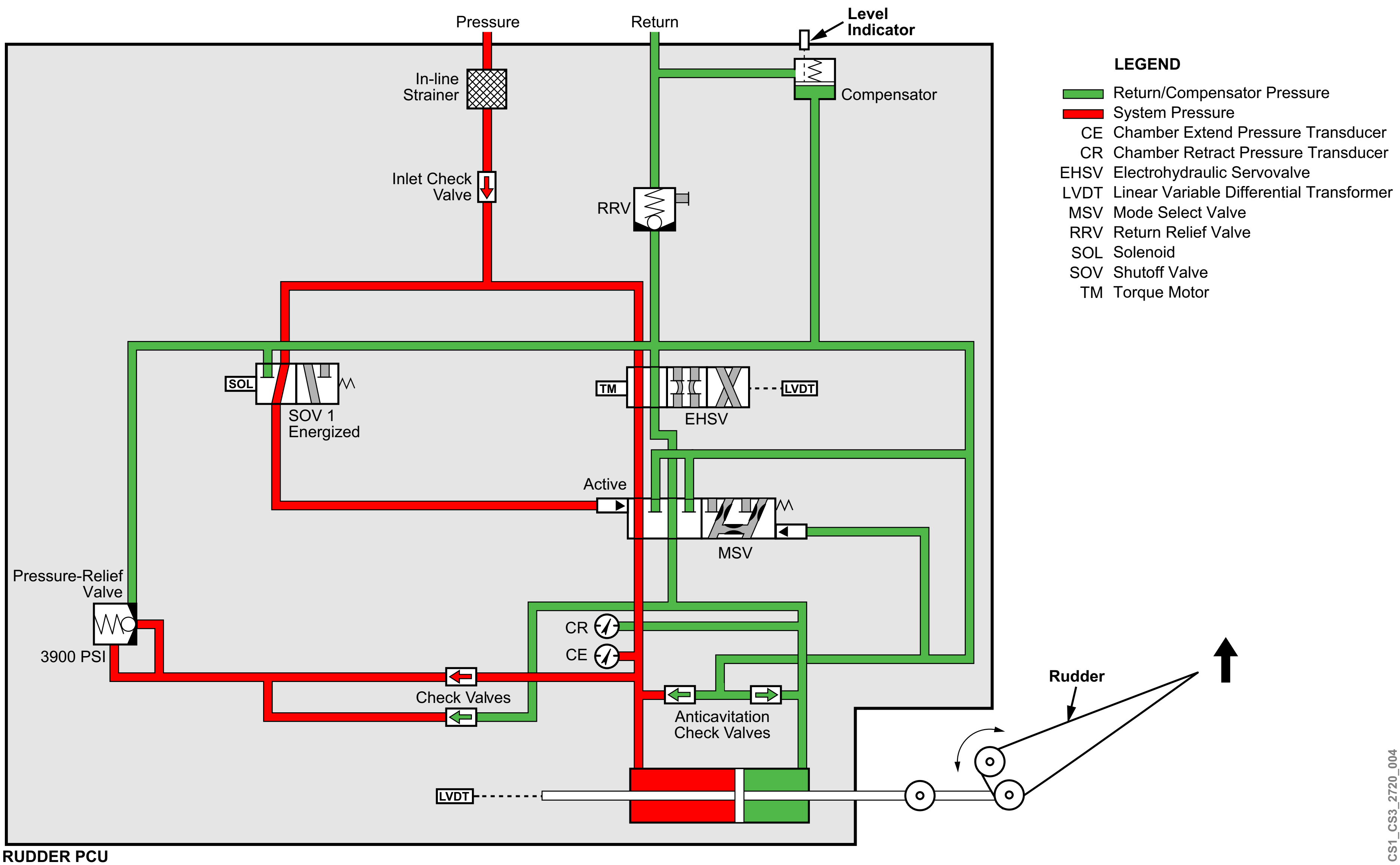


Figure 60: Rudder Power Control Unit - Active State

Damped State

In the event the REU signal is interrupted, the input to the SOV will terminate, causing the SOV to close. Once closed, hydraulic pressure to the active port of the MSV depletes, causing the MSV to move to a damped state. In the damped state, the MSV connects the extend and retract ports of the actuator through a restricted orifice. This restriction provides fluid damping of the PCU. The fluid damping is sufficient to prevent flutter, but will not impede the operation of the other PCU on the same surface. In this state, the EHSV is commanded to a neutral position.

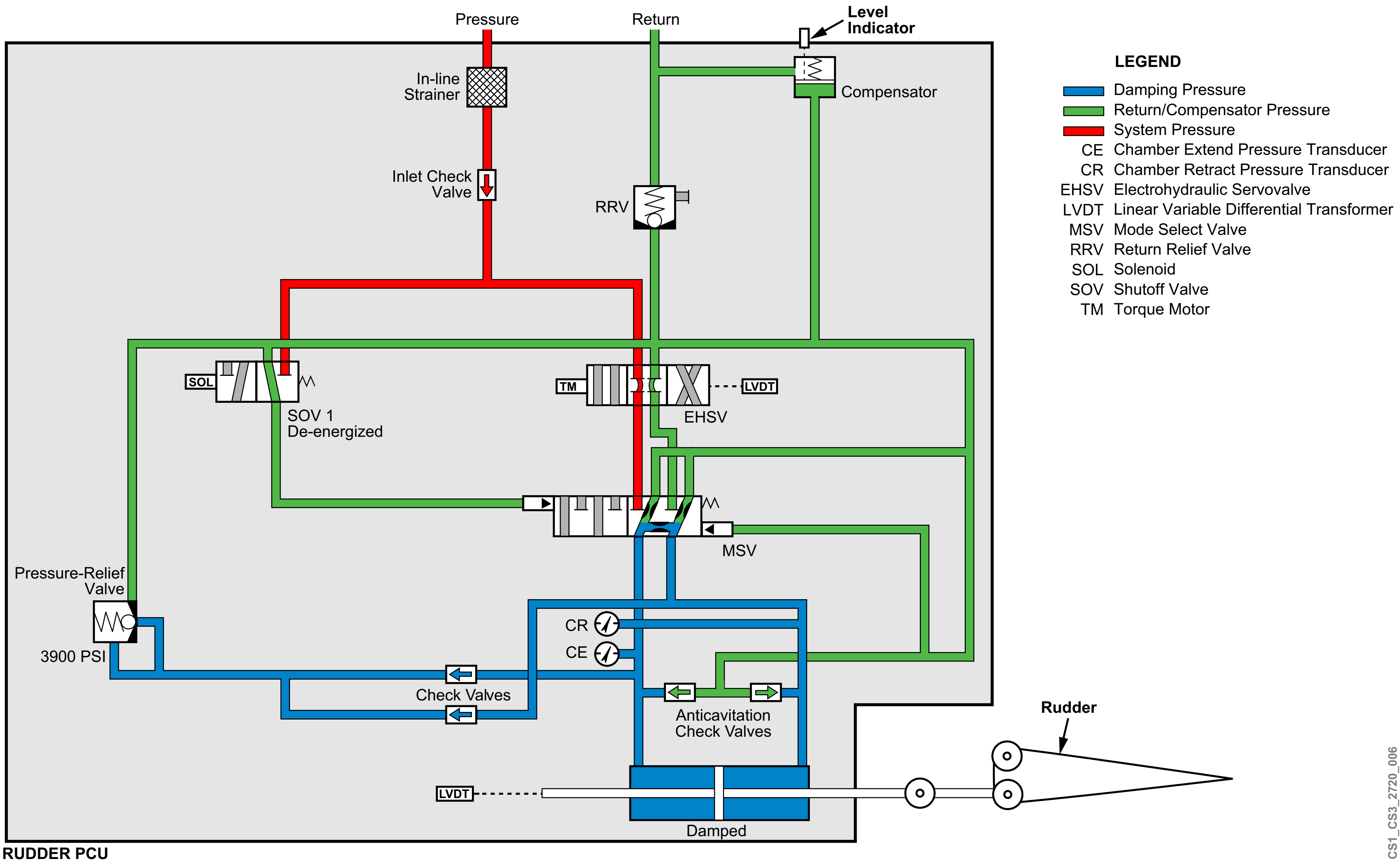
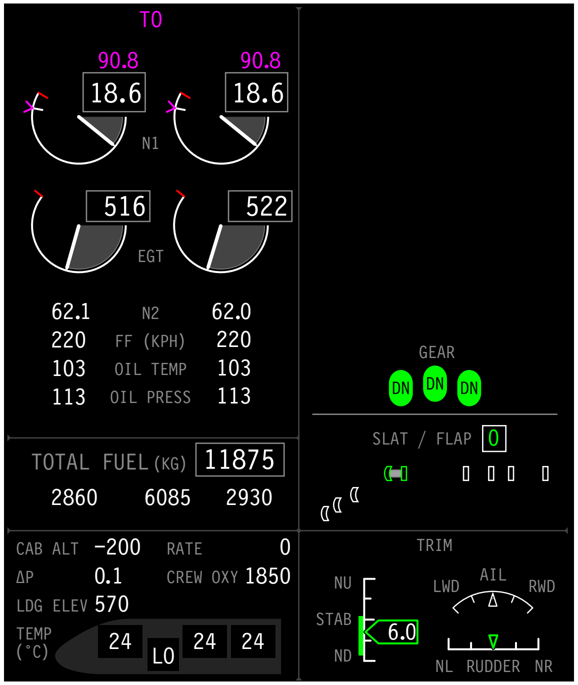


Figure 61: Rudder Power Control Unit - Damped State

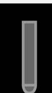
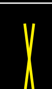
CONTROLS AND INDICATIONS

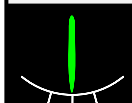
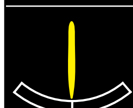
The rudder trim position and status are displayed on the EICAS page.

The rudder position and status are displayed on the flight controls synoptic page.



EICAS PAGE

RUDDER SURFACE STATE	
Symbol	Condition
	Normal
	Fail
	Invalid

RUDDER POSITION INDICATOR AND STATE	
Symbol	Condition
	Normal
	Fail
	Invalid

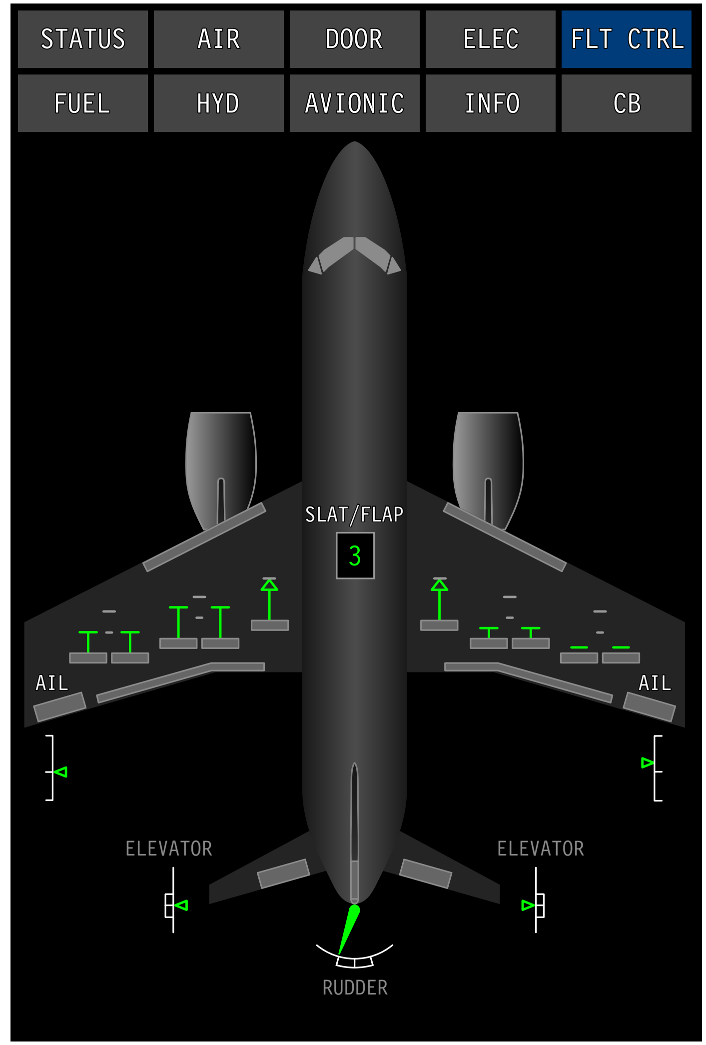


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Figure 62: Rudder Surface and Rudder Trim Indications

DETAILED DESCRIPTION

RUDDER POSITION AND RUDDER TRIM INDICATING

Rudder position and status information is transmitted by the rudder power control unit (PCU) linear variable differential transformers (LVDTs) to the remote electronic units (REUs). The lower PCU sends this data to the alternate flight control unit (AFCU). The REUs relay this information to the inceptor interface modules (IIMs), which send the signals on to the primary flight control computers (PFCCs). The PFCCs send the information to the data concentrator unit module cabinets (DMCs), which display the position and status information on the flight controls synoptic page.

Rudder position information is transmitted by the rudder trim RVDTs to the IIMs, then to the PFCCs. The PFCCs send the information to the DMCs, which display the position and status information on the EICAS page.

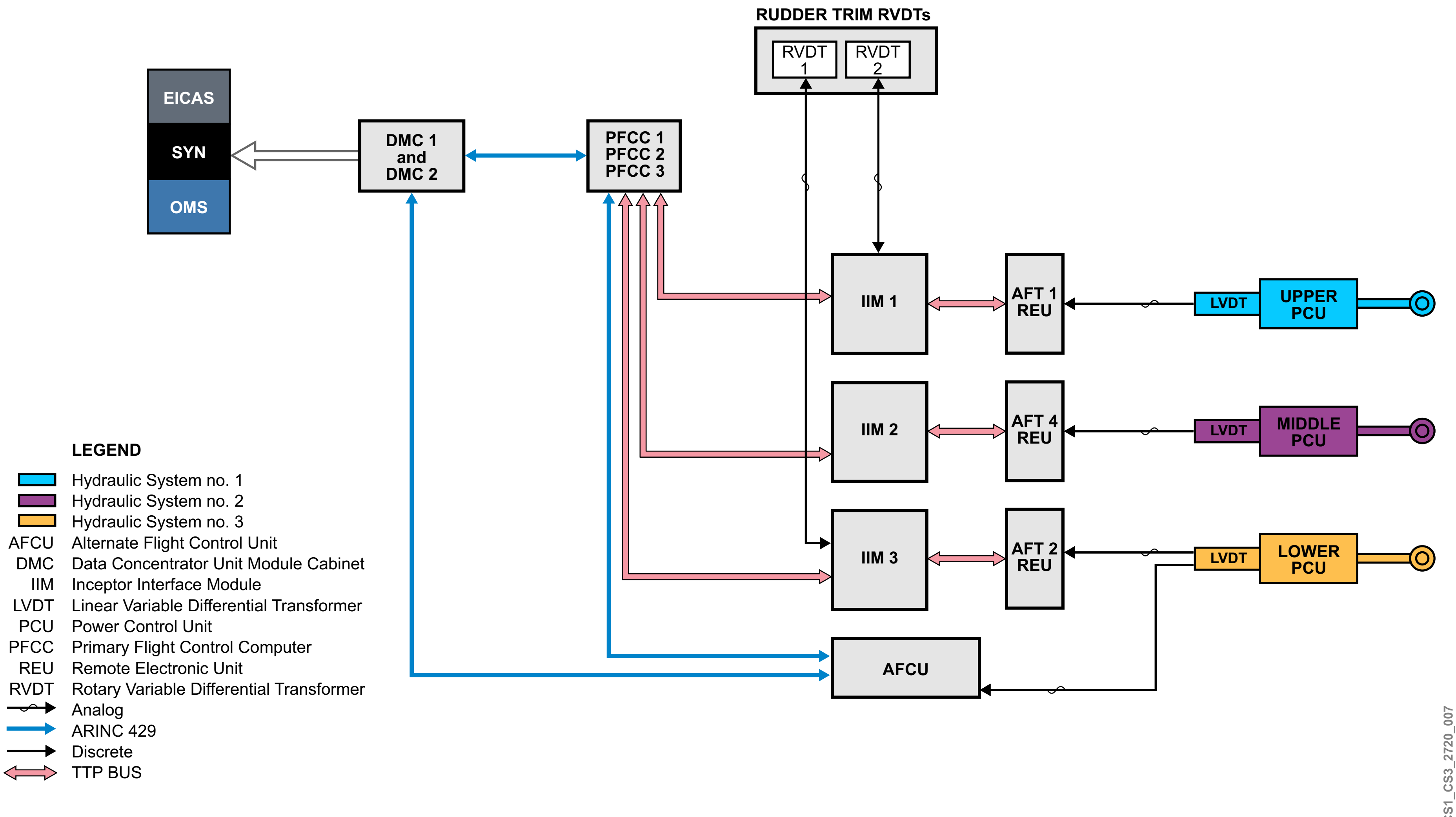


Figure 63: Rudder Position and Rudder Trim Indicating

MONITORING AND TESTS

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

CAS MESSAGES

Table 9: WARNING Messages

MESSAGE	LOGIC
RUDDER FAIL	All three rudder PCUs failed.
CONFIG RUDDER TRIM	Rudder trim position out of takeoff range.

Table 10: CAUTION Messages

MESSAGE	LOGIC
RUDDER DEGRADED	Two of three rudder PCUs failed.
YAW AUTHORITY	Rudder command near max operational authority.

Table 11: ADVISORY Message

MESSAGE	LOGIC
FLT CTRL FAULT	Loss of redundant or non critical function for the primary flight control systems.Refer to INFO messages.

Table 12: INFO Messages

MESSAGE	LOGIC
27 FLT CTRL FAULT - RUDDER PEDAL SNSR REDUND LOSS	Loss of rudder pedal sensor.
27 FLT CTRL FAULT - RUDDER PEDAL SNSR DEGRADED	Loss of any two rudder pedal sensors.
27 FLT CTRL FAULT - RUDDER TRIM SNSR INOP	Loss of both rudder trim sensors.

PRACTICAL ASPECTS

Power Control Unit Return Relief Valve Tests

The PCU return relief valve (RRV) requires scheduled testing. In addition, the compensator must be depleted for maintenance. To do this, push and hold the depletion button of the return relief valve with a pin for approximately 3 seconds.

An operational check of the compensator verifies the position of the compensator level indicator with hydraulic pressure applied, releasing hydraulic pressure, then measuring the position of the level indicator after several time periods (30 minutes each). Any significant change in position indicates excessive internal leakage, requiring maintenance.

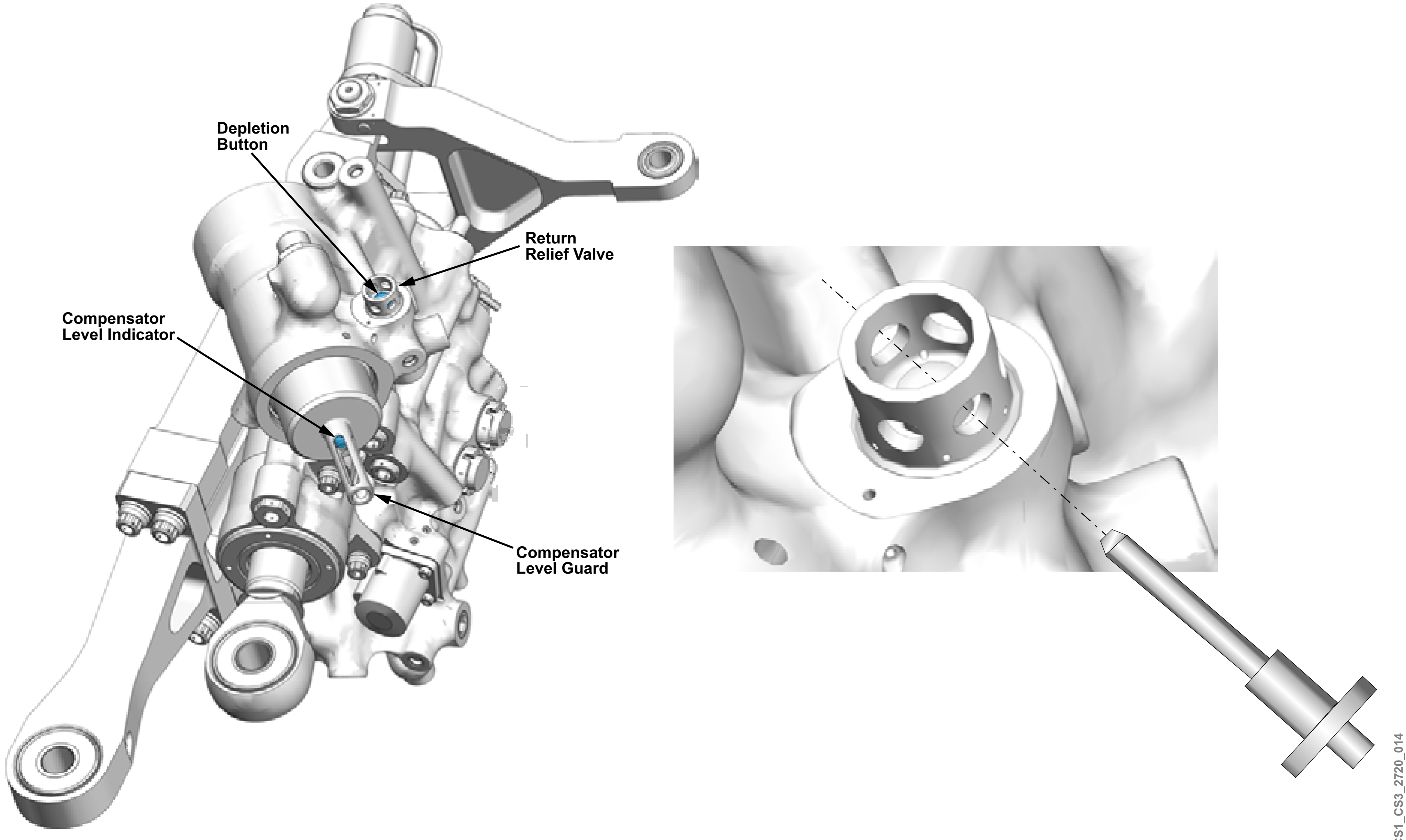


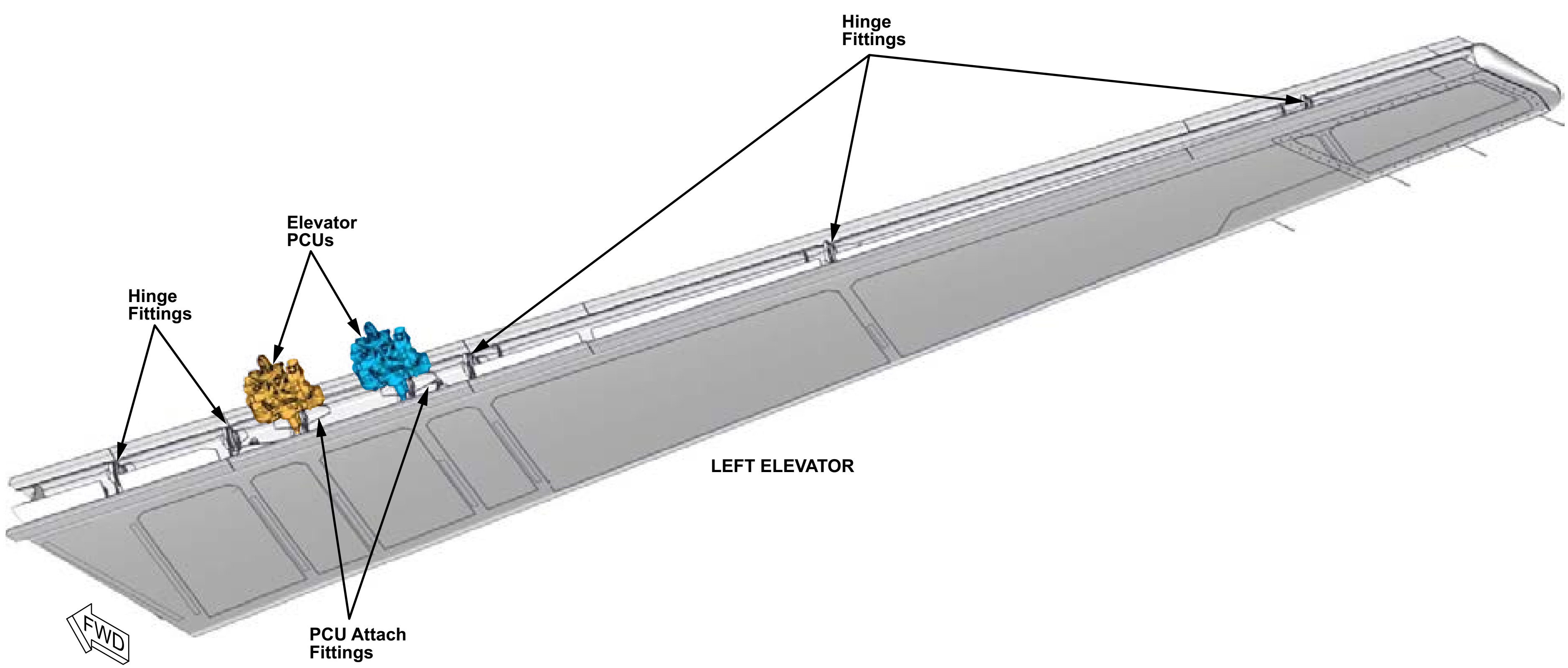
Figure 64: Rudder PCU Return Relief Valve Tests

27-30 ELEVATOR SYSTEM

GENERAL DESCRIPTION

Primary pitch control is provided by two elevator control surfaces, which move symmetrically in normal mode. They are mounted on the horizontal stabilizer rear spars.

The power control units (PCUs) are electrohydraulic units, and are identical in operation to the aileron PCUs. Hydraulic pressure for the inboard PCUs is supplied by hydraulic system no. 3, while hydraulic system no. 1 provides hydraulic pressure to the left outboard PCU, and hydraulic system no. 2 powers the right outboard PCU. Under normal conditions the elevator range of movement is 32° noseup, and 22° nosedown.



LEGEND

- Hydraulic System No. 1
- Hydraulic System No. 3

NOTE

Right elevator outboard power control unit (PCU) powered by hydraulic system no. 2, inboard PCU powered by hydraulic system no. 3

Figure 65: Elevators Overview

COMPONENT INFORMATION

ELEVATOR POWER CONTROL UNITS

The elevator PCUs are mounted on the rear spar of each horizontal stabilizer. The PCUs are similar to the aileron PCUs. Other than the external dimensional differences between the two, the operation and configuration of the two PCU types are identical. The aileron and elevator PCUs are not interchangeable.

An internal linear variable differential transformer (LVDT), connected to the PCU actuator, provides information of the actual position of the PCU.

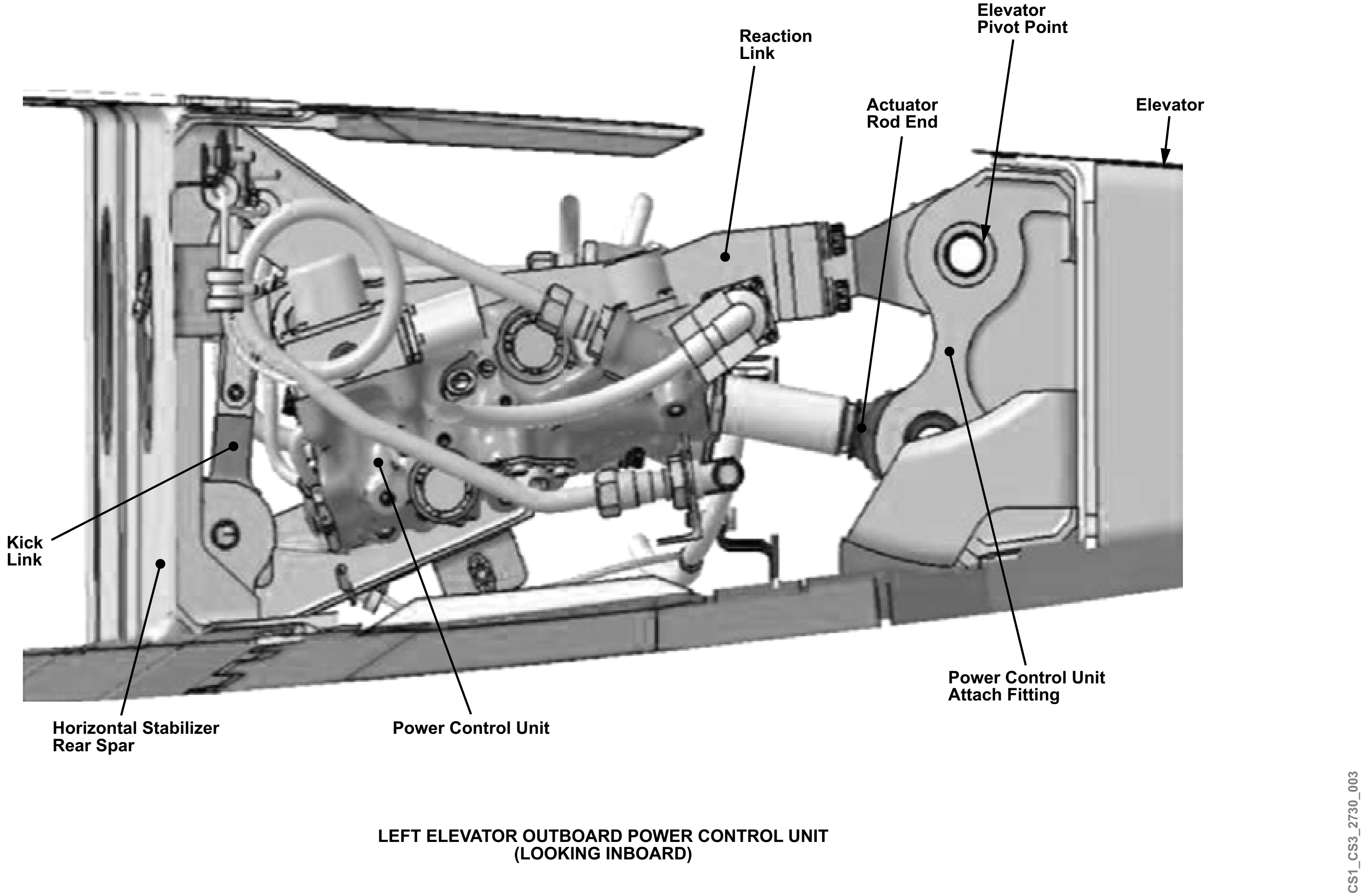
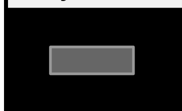
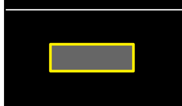
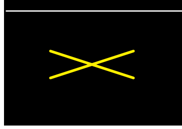
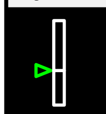
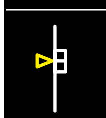
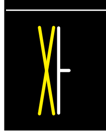


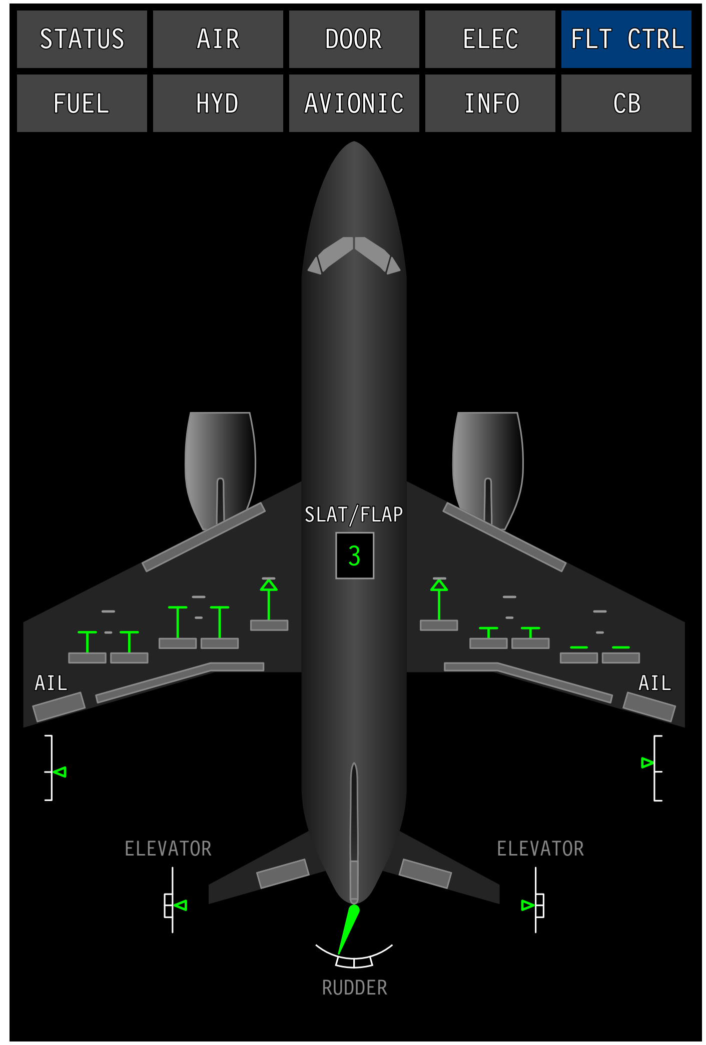
Figure 66: Elevator Power Control Unit

CONTROLS AND INDICATIONS

The elevator position and status are displayed on the flight controls synoptic page.

ELEVATOR SURFACE STATE	
Symbol	Condition
	Normal
	Fail
	Invalid

ELEVATOR POSITION INDICATOR	
Symbol	Condition
	Normal
	Fail
	Invalid



CS1_CS3_2730_005

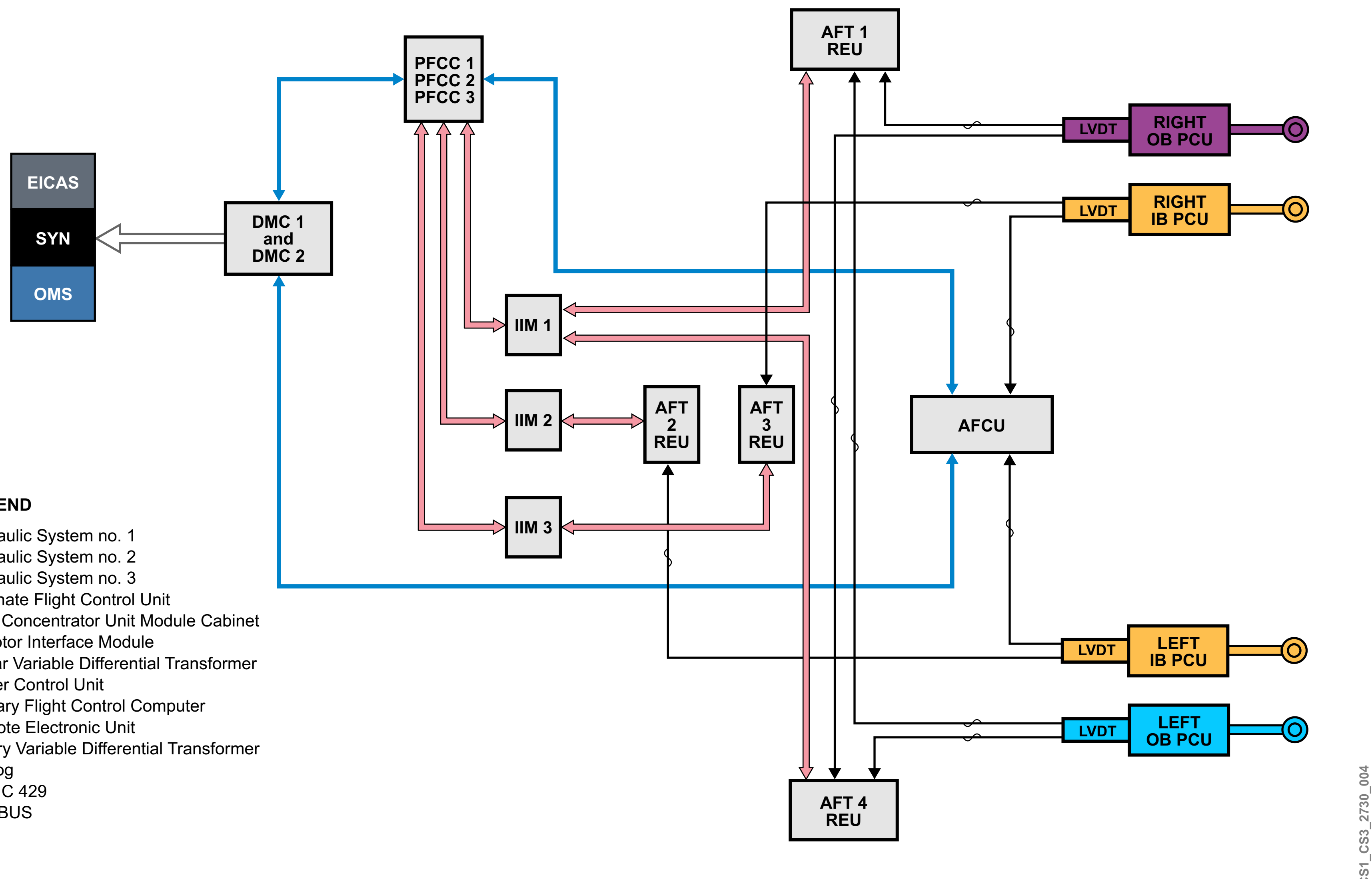
Figure 67: Elevator Surface Indication

DETAILED DESCRIPTION

ELEVATOR POSITION INDICATING

The outboard PCU position signals are transmitted to the AFT 1 and AFT 4 REUs, which relay the data to IIM 1. The data is sent to the PFCCs for transmission to the aircraft avionics. The aircraft avionics convert the information for display on the flight controls synoptic page.

The inboard PCUs send position data to the AFT 2 and AFT 3 REUs, as well as to the AFCU. The REUs transmit the data to their respective IIMs, which relay the information to the PFCCs. The data is further transmitted to the aircraft avionics for display on the flight controls synoptic page.



LEGEND

Hydraulic System no. 1
Hydraulic System no. 2
Hydraulic System no. 3
AFCU Alternate Flight Control Unit
DMC Data Concentrator Unit Module Cabinet
IIM Inceptor Interface Module
LVDT Linear Variable Differential Transformer
PCU Power Control Unit
PFCC Primary Flight Control Computer
REU Remote Electronic Unit
RVDT Rotary Variable Differential Transformer
Analog
ARINC 429
TTP BUS

Figure 68: Elevator Position Indication

MONITORING AND TESTS

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

CAS MESSAGES

Table 13: WARNING Messages

MESSAGE	LOGIC
L ELEVATOR FAIL	Left elevator failed during the takeoff phase.
R ELEVATOR FAIL	Right elevator failed during the takeoff phase.

Table 14: CAUTION Messages

MESSAGE	LOGIC
L ELEVATOR FAIL	Left elevator failed during non-takeoff phase.
R ELEVATOR FAIL	Right elevator failed during non-takeoff phase

Table 15: ADVISORY Message

MESSAGE	LOGIC
FLT CTRL FAULT	Loss of redundant or non-critical function for the primary flight control systems. Refer to INFO messages

Table 16: INFO Message

MESSAGE	LOGIC
27 FLT CTRL FAULT - PRIM PCU INOP	Loss of any primary PCU.
27 FLT CTRL FAULT - PRIM PCU FAULT	Loss of soft damp SOV engagement in primary PCUs.

27-40 HORIZONTAL STABILIZER TRIM SYSTEM

GENERAL DESCRIPTION

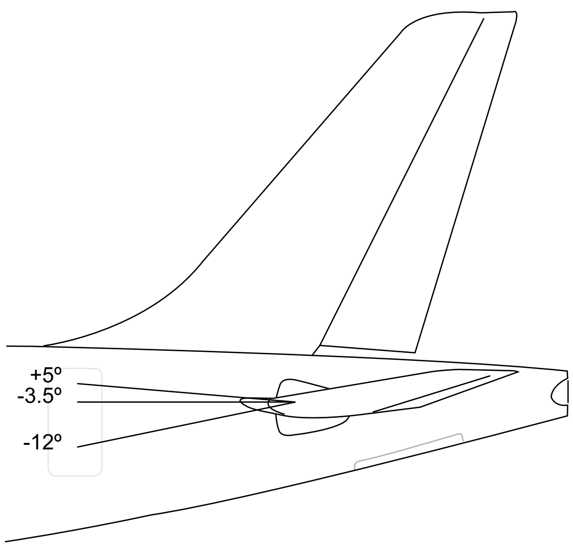
The horizontal stabilizer trim system (HSTS) allows the stabilizer to be trimmed up or down to relieve pressure on the elevator control surfaces and reduce aircraft drag.

The HSTA is a single ballscrew, electromechanical actuator, with dual-load paths for added safety. By rotating the ballscrew, the horizontal stabilizer moves up or down. The horizontal stabilizer can be trimmed from +5° to -12° (centered at -3.5°).

The horizontal stabilizer trim actuator (HSTA) is mounted in the aft fuselage. The HSTA consists of the following major assemblies:

- Upper gimbal assembly
- Lower gimbal assembly
- Ballscrew and no-back device assembly

The upper gimbal assembly has primary and secondary load path attachments connecting it to the fuselage structure. The ballscrew and no-back device connect to the upper gimbal. The lower gimbal assembly is driven by the ballscrew and has primary and secondary load path connections to the horizontal stabilizer center box.



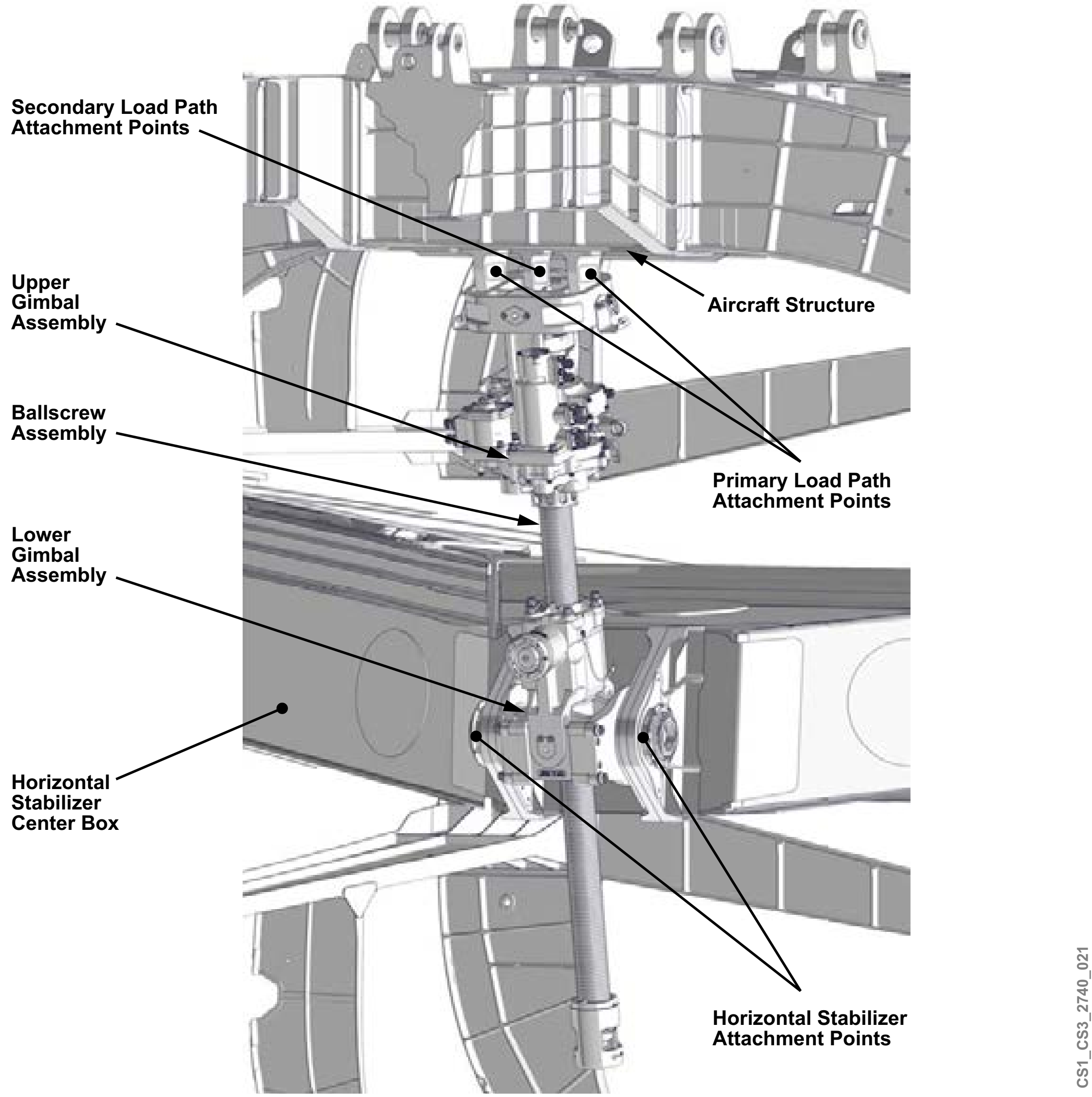


Figure 69: Horizontal Stabilizer Trim System

COMPONENT INFORMATION

HORIZONTAL STABILIZER TRIM ACTUATOR

The HSTA has the following components:

- Upper gimbal assembly
- Ballscrew assembly
- Lower gimbal assembly
- Tandem motor assembly
- Mechanical stops
- Electric brake
- Single and dual-rotary variable differential transformers (RVDT) position sensors

Upper Gimbal Assembly

The upper gimbal assembly has primary and secondary load path attachments mounted to the no-back housing which supports the ballscrew and no-back device. The ballscrew is driven by the gearbox which has mounting points for the tandem motor assembly, single and dual channel rotary variable differential transformers (RVDTs), and an electric brake.

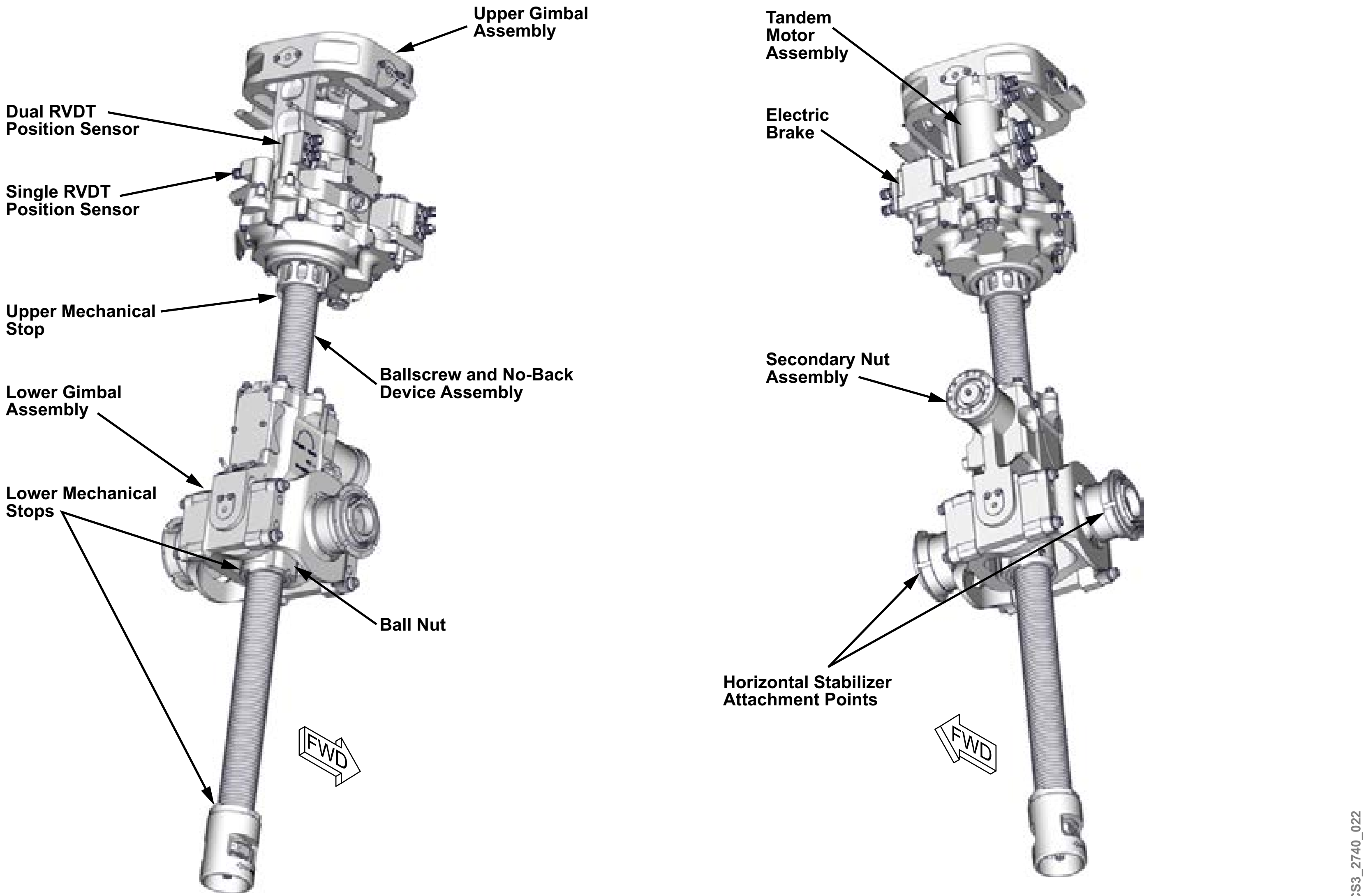
The tandem motor assembly drives the HSTA and the electric brake aids in holding the Horizontal stabilizer in position. The horizontal stabilizer position is monitored by the single and dual-channel rotary RVDTs for HSTA control and display on the engine indication and crew alerting system (EICAS).

Lower Gimbal Assembly

The ballnut is mounted within the primary gimbal assembly and is driven by the ballscrew. The primary gimbal assembly provides the attachment points to the horizontal stabilizer. A secondary nut is located within the secondary nut housing assembly and the lower gimbal housing.

The primary gimbal assembly connects to the horizontal stabilizer using primary and secondary attachment pins. The secondary gimbal assembly and gimbal plates provide a second load path in case of a primary gimbal failure.

The lower gimbal assembly travel is limited by mechanical stops at each end of the ballscrew.



LEGEND

RVDT Rotary Variable Differential Transformer

Figure 70: Horizontal Stabilizer Trim Actuator

DETAILED COMPONENT INFORMATION

BALLSCREW ASSEMBLY AND NO-BACK DEVICE

The ballscrew assembly consists of a ballscrew, ballnut, and a tie rod driven by a dual-motor assembly and gearbox. The output of the motors drives a three-stage spur gear train with a slip clutch between the first and second stage. The ballscrew is driven off the third stage gears. The slip clutch assembly protects the gear train against damage in the event of a jam. When the ballscrew is driven, the ballnut assembly moves along the ballscrew. An internal no-back device prevents back-driving the ballnut when the stabilizer is under either compression, or tension flight loads.

The no-back devices with dual-pawls and ratchets are located on the primary load path. One acts in compression loads and the other in tension loads. The no-back device is serviced with hydraulic oil for heat dissipation.

The HSTA has a primary and two secondary mechanical load paths. The primary and secondary load paths are concentric with the outer ballscrew being the primary load path. The inner secondary path is a tie-rod that maintains horizontal stabilizer attachment in the event of a primary load path failure. An additional secondary load path uses a secondary nut that clamps the lower gimbal to the ballscrew in the event of a lower primary gimbal or ballnut failure

Primary Load Path

The primary load path consists of the aircraft structure, upper primary gimbal, no-back housing, no-back device, ballscrew, ballnut, lower mechanical stop, lower primary gimbal, and the horizontal stabilizer structure.

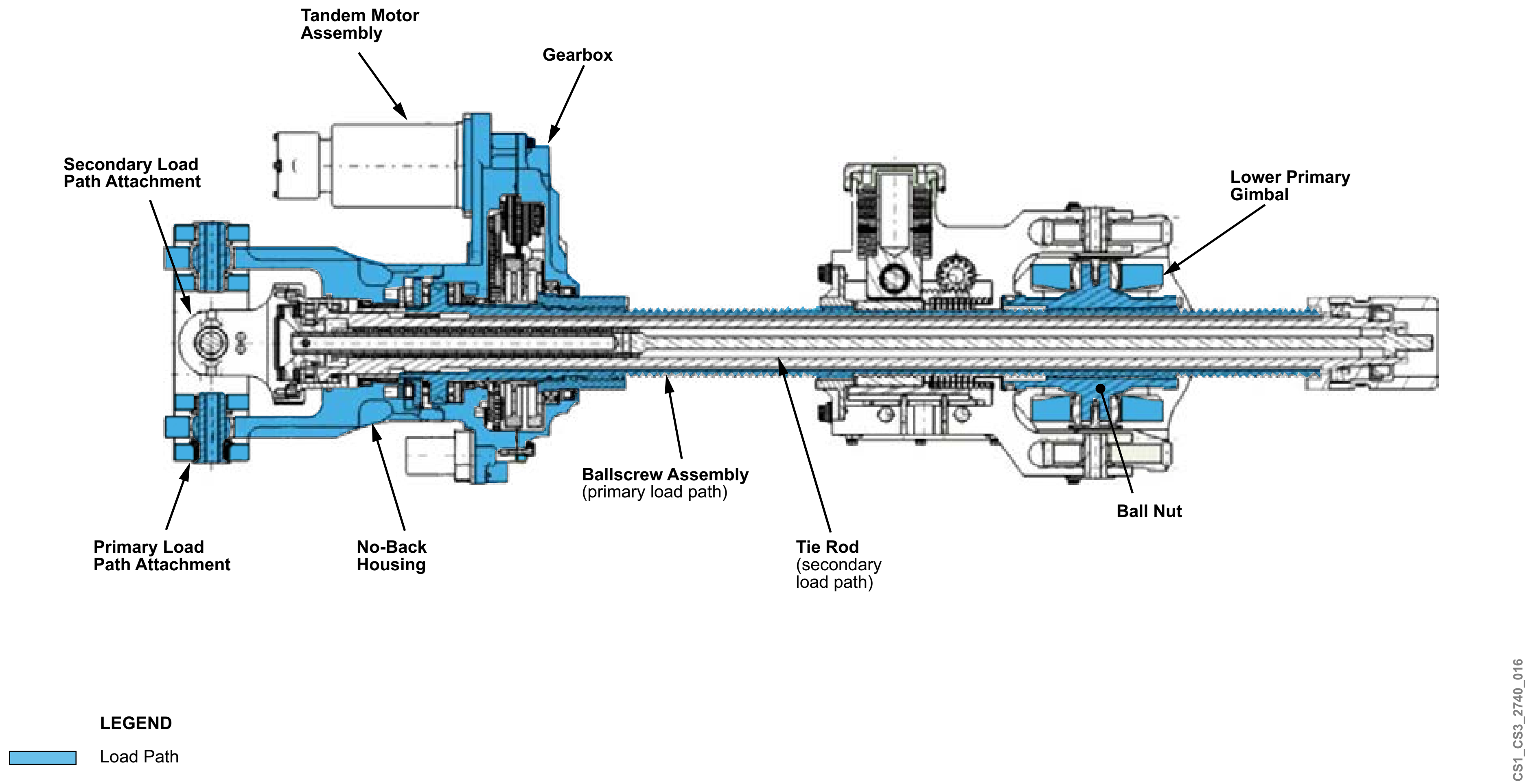


Figure 71: HSTA Ballscrew and No-Back Device Assembly

SECONDARY LOAD PATHS

The secondary load path consists of the aircraft structure, upper secondary gimbal assembly, tie-rod, tie rod latch/lock mechanism, tie-rod nut, secondary nut housing, secondary nut locking mechanism, secondary nut, lower secondary gimbal, and the horizontal stabilizer structure.

Tie Rod and Latch/Lock Mechanism

The tie rod and latch/lock mechanism provides a secondary load path to protect against disconnect failures of the primary load path between the ballscrew and the upper aircraft attachment. It will react the HSTA external load, through the tie rod and upper secondary clevis attachment, and will provide detection of the secondary load path engagement. The tie rod will travel axially, in the secondary load path clevis housing, through a controlled amount of axial play until it bottoms and carries full air load. The transient movement, through the secondary load path axial play, engages a set of latching splines in tension or compression (rotary locking). This is the latch.

A tension load will also trigger lock the mechanism, preventing any further axial movement, hence eliminating excess play and preventing disengagement of the latch. The motion will cause a set of torsionally preloaded two tooth splines, between the trigger and tie rod, to disengage, allowing the torsion spring to unwind. This rotational movement misaligns the splines, preventing re-engagement. Additionally, helixed faces on the trigger and tie rod take up axial play, removing axial backlash from the mechanism. This is the 'lock'. A tension trigger lock was chosen because a compression trigger may not actuate and detect a ballscrew fracture (since ballscrew will still carry load under compression). A stack-up analysis was performed to assure that the rotational range between the trigger and the tie rod was within acceptable limits (rotation ranges from 70° to 111°).

Secondary Nut Assembly

The secondary nut assembly provides a secondary load path to protect against disconnect failures of the ballnut and lower primary load path. In the event of an axial disconnect failure of the primary load path; the loads are transferred to the secondary load path through the secondary nut and into the secondary stabilizer support flanges. The secondary nut housing travels axially through a controlled amount of axial play, relative to the ballscrew. The loads will be taken by the secondary nut, preventing movement relative to ballscrew.

Under tension load, the secondary load path will react air loads but will not positively lock out the actuator until loads are reversed to compression. The HSTA may continue to be operable. This device triggers under compression loading only.

Under compression, the split nut will travel and push an axially spring loaded carriage. The carriage translation will trigger a spring-loaded plunger, causing the plunger to travel and releasing spring energy into secondary nut. The secondary nut is a split design. Spring energy will load both halves against the ballscrew with a large force, clamping the secondary nut to the ballscrew shaft. This will prevent operation of the HSTA and hence annunciate the failure.

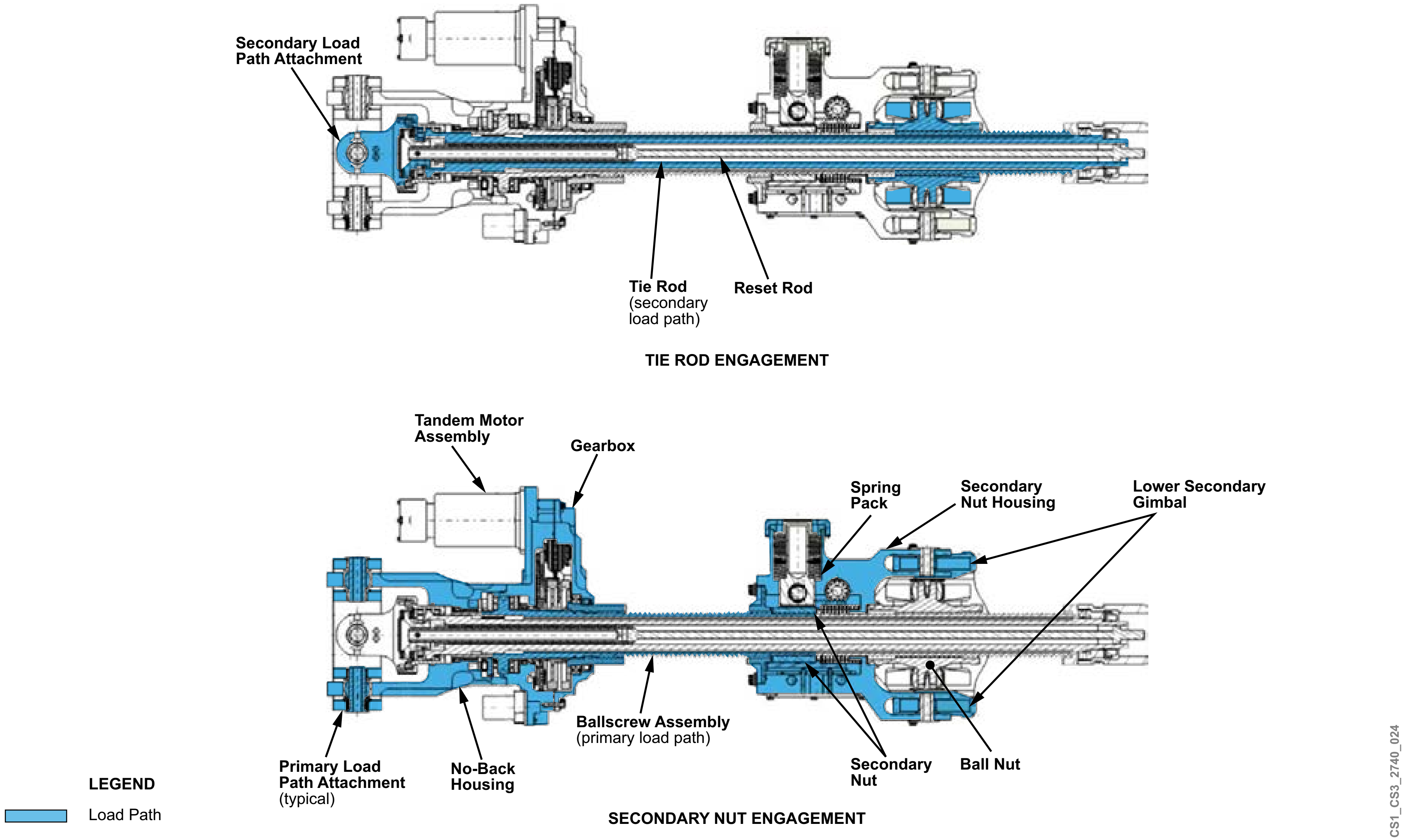


Figure 72: HSTA Secondary Load Paths

TANDEM MOTORS ASSEMBLY

Two brushless DC motors drive the actuator ballscrew in an active-standby configuration. The HSTA has two identical 270 VDC brushless motors with two integral brushless resolvers for commutation. The brushless DC motors are controlled in velocity by the REU through a velocity feedback from the appropriate motor resolver. The REU also controls the motor current, commands rate of acceleration and deceleration, and limits the HSTA output force in a jam condition.

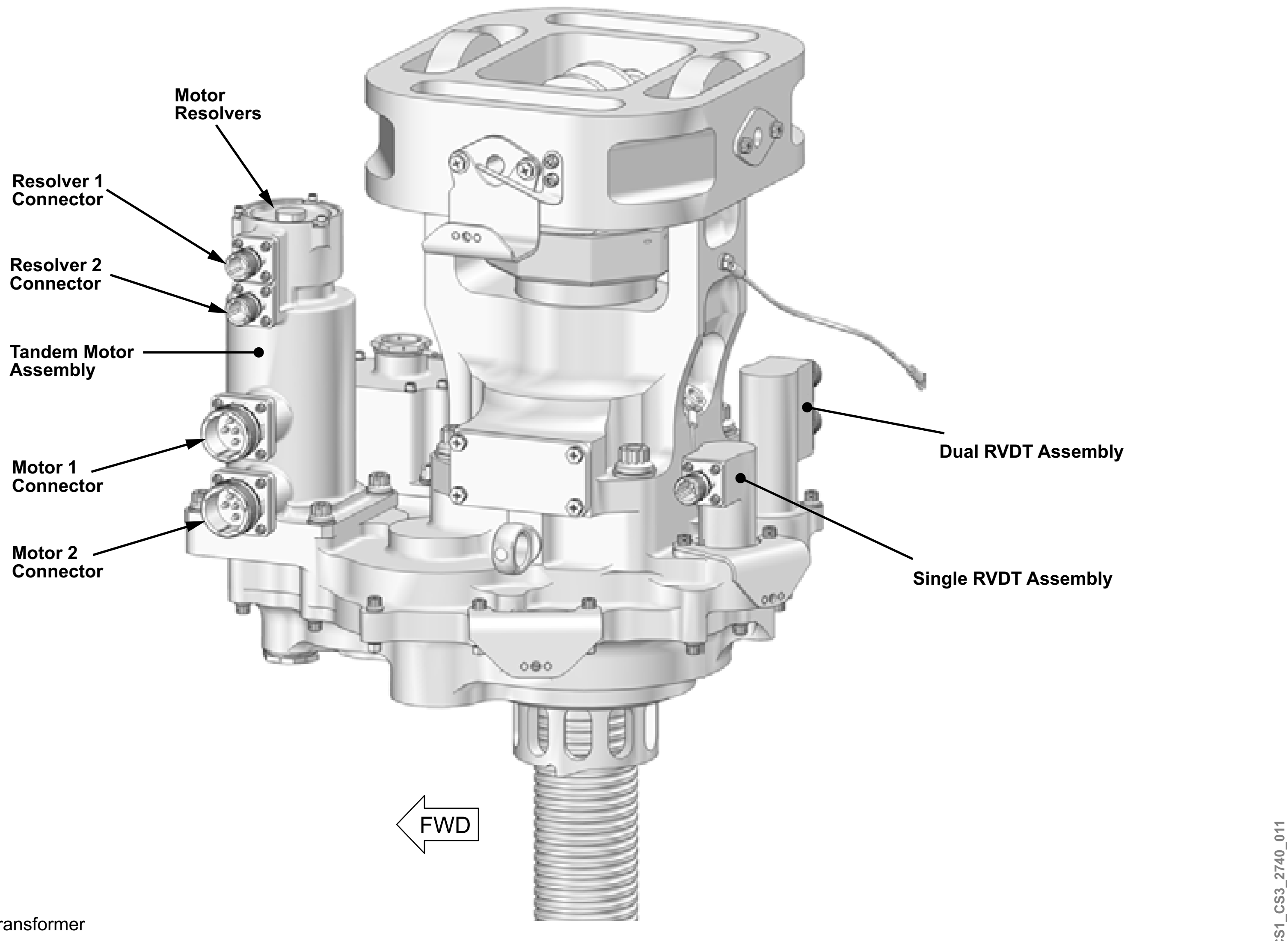
The two motors share a common rotor shaft. The motors are equipped with a manual drive input to the motor shaft to check the clutch operation during periodic maintenance tests. The motors can be removed from the HSTA while on the aircraft.

WARNING

WHEN OPERATING THE HSTA ON THE GROUND, ENSURE THAT THE DUTY CYCLE FOR THE MOTOR ASSEMBLY IS STRICTLY OBSERVED, TO AVOID OVERHEATING AND POTENTIALLY DAMAGING THE TANDEM MOTORS.

SINGLE AND DUAL RVDT POSITION SENSORS

Three RVDT position sensors are used in the HSTA system to provide actuator position feedback to the controlling REUs and AFCU. There is one single and one dual-sensor assembly. The single sensor transmits position data to channel 1 of the MCE. One RVDT from the dual-sensor transmits to channel 2 of the MCE and the second transmits to the AFCU.



LEGEND

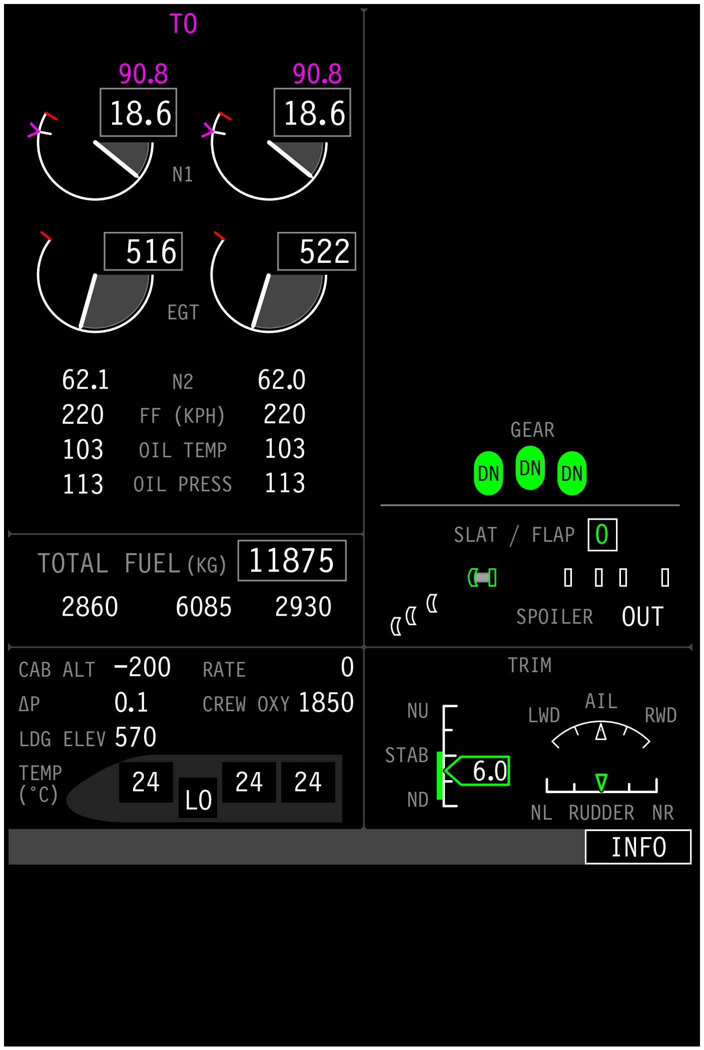
RVDT Rotary Variable Differential Transformer

Figure 73: HSTA Tandem Motor Assembly and RVDTs

CONTROLS AND INDICATIONS

The HSTA trim angle is shown in the EICAS page. Indications range from nosedown (ND) at 0° to the noseup (NU) at 17°. The scale is divided in five graduations: 0°, 4.25°, 8.5°, 12.75°, and 17°. When the aircraft is on the ground with stab trim within the takeoff range, the pointer is green.

The pointer is shown in white when the aircraft is on ground during takeoff if the horizontal stabilizer trim is trimmed outside the takeoff range. There is also a small rectangle shown beside the trim scale showing green when the aircraft is on ground or during takeoff, and the HSTA is trimmed within the takeoff range.



EICAS PAGE

STAB TRIM INDICATOR	
Symbol	Condition
8.5	Pitch Trim Value (Normal and PFCC Direct mode only)
	Trim in takeoff range, on ground
	In flight, or on ground and trim outside of takeoff range
	Invalid
	Takeoff range, on ground

CS1_CS3_2740_023

Figure 74: Horizontal Stabilizer Trim Indication

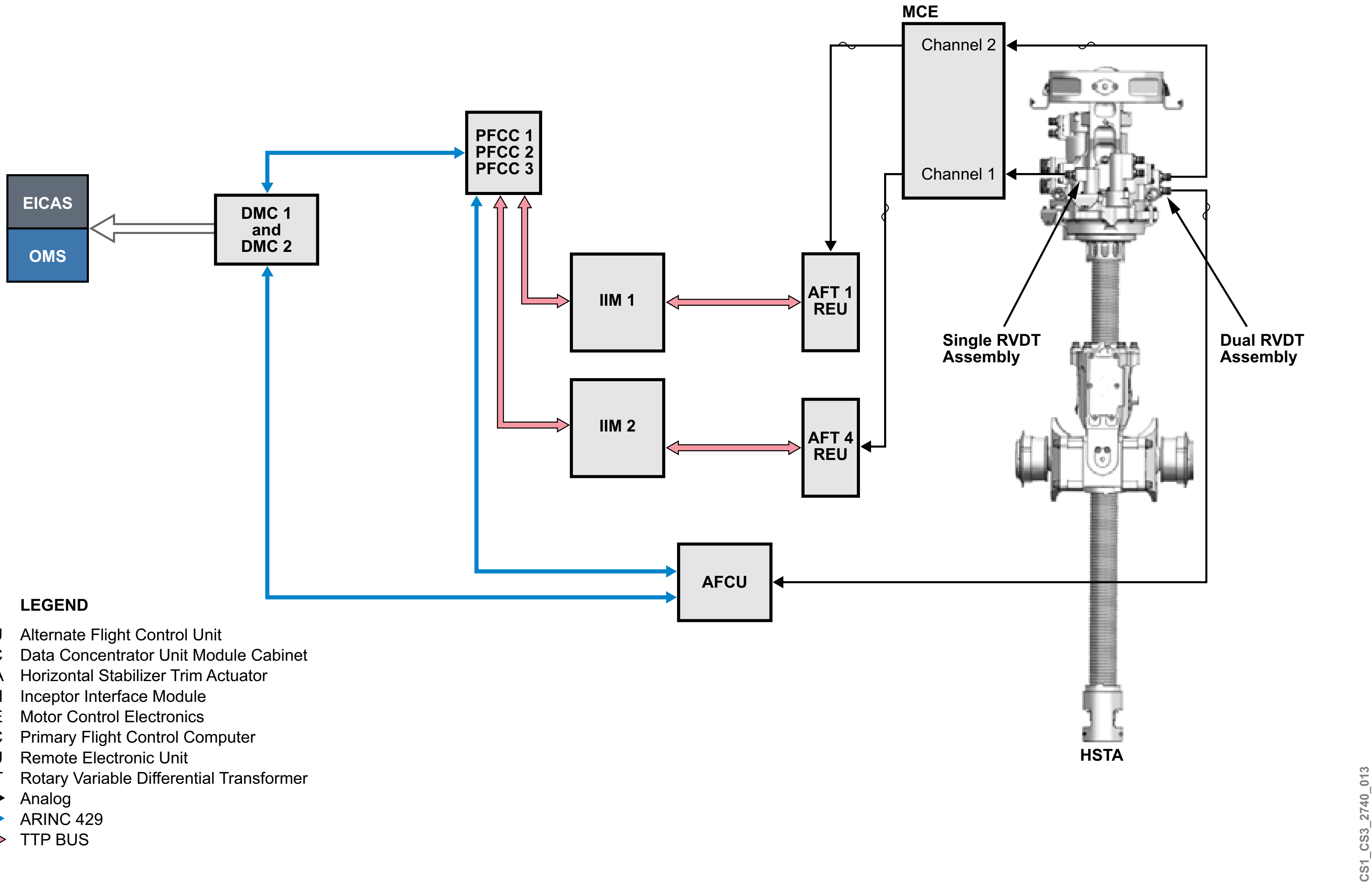
DETAILED DESCRIPTION

HORIZONTAL STABILIZER TRIM POSITION INDICATION

Rotary variable differential transformers (RVDT) provide position information of the horizontal stabilizer trim actuator (HSTA) to remote electronic units (REUs) and the alternate flight control unit (AFCU). Position information is used for system operation and for position indication on the EICAS page.

Normally, HSTA position indication is provided from the REUs to the inceptor interface modules (IIMs), which communicate over TTP BUSES to the primary flight control computers (PFCCs). The PFCCs provide the information, through the data concentrator unit module cabinets (DMCs), for EICAS display and the onboard maintenance system (OMS).

When operating in the AFCU direct mode, position indication is provided from the AFCU, over ARINC 429 BUSES, to the DMCs for EICAS display and the OMS.



LEGEND

AFCU Alternate Flight Control Unit
DMC Data Concentrator Unit Module Cabinet
HSTA Horizontal Stabilizer Trim Actuator
IIM Inceptor Interface Module
MCE Motor Control Electronics
PFCC Primary Flight Control Computer
REU Remote Electronic Unit
RVDT Rotary Variable Differential Transformer
Analog
ARINC 429
TTP BUS

Figure 75: Horizontal Stabilizer Trim System Position Indication

MONITORING AND TESTS

The following page provides the CAS and INFO messages for the horizontal stabilizer trim system.

CAS MESSAGES

Table 17: WARNING Message

MESSAGE	LOGIC
CONFIG STAB TRIM	Stab trim out of range for takeoff.

Table 18: CAUTION Messages

MESSAGE	LOGIC
STAB TRIM FAIL	Stabilizer trim function failed.
STAB DEGRADED	Selected stabilizer position is invalid.

Table 19: ADVISORY Messages

MESSAGE	LOGIC
FLT CTRL FAULT	Loss of redundant or non-critical function for the primary flight control systems. Refer to INFO messages.
L PITCH TRIM SW FAIL	Left pitch trim switch failed.
R PITCH TRIM SW FAIL	Right pitch trim switch failed.

Table 20: INFO Messages

MESSAGE	LOGIC
27 FLT CTRL FAULT - STAB TRIM CHAN 1 INOP	Loss of stabilizer trim channel 1.
27 FLT CTRL FAULT - STAB TRIM CHAN 2 INOP	Loss of stabilizer trim channel 2.
27 FLT CTRL FAULT - L (R) PITCH TRIM SW DEGRADED	Loss of the L (R) PITCH TRIM SW.

PRACTICAL ASPECTS

HORIZONTAL STABILIZER TRIM ACTUATOR NO-BACK DEVICE SERVICING

The horizontal stabilizer trim actuator (HSTA) no-back device is filled with oil to assist in dissipating heat produced by the slip clutch assemblies, and to lubricate the internal moving parts.

To service the no-back device:

1. Open the vent and fill plugs
2. Put a container below the drain plug in line with the channel machined between the motor and the brake mounting surfaces to collect the oil
3. Open the drain plug and drain the oil from the no-back device into the container
4. Install the drain plug
5. Add oil to the fill hole slowly until it flows out of the fill or vent hole. The total capacity of the HSTA no back chamber is 167cc. Wait for 3 minutes and repeat the fill if necessary
6. Install the vent and fill plugs
7. Trim the horizontal stabilizer nose-up and nose-down then set the stabilizer at the neutral position (5 units) and check that there are no leaks.

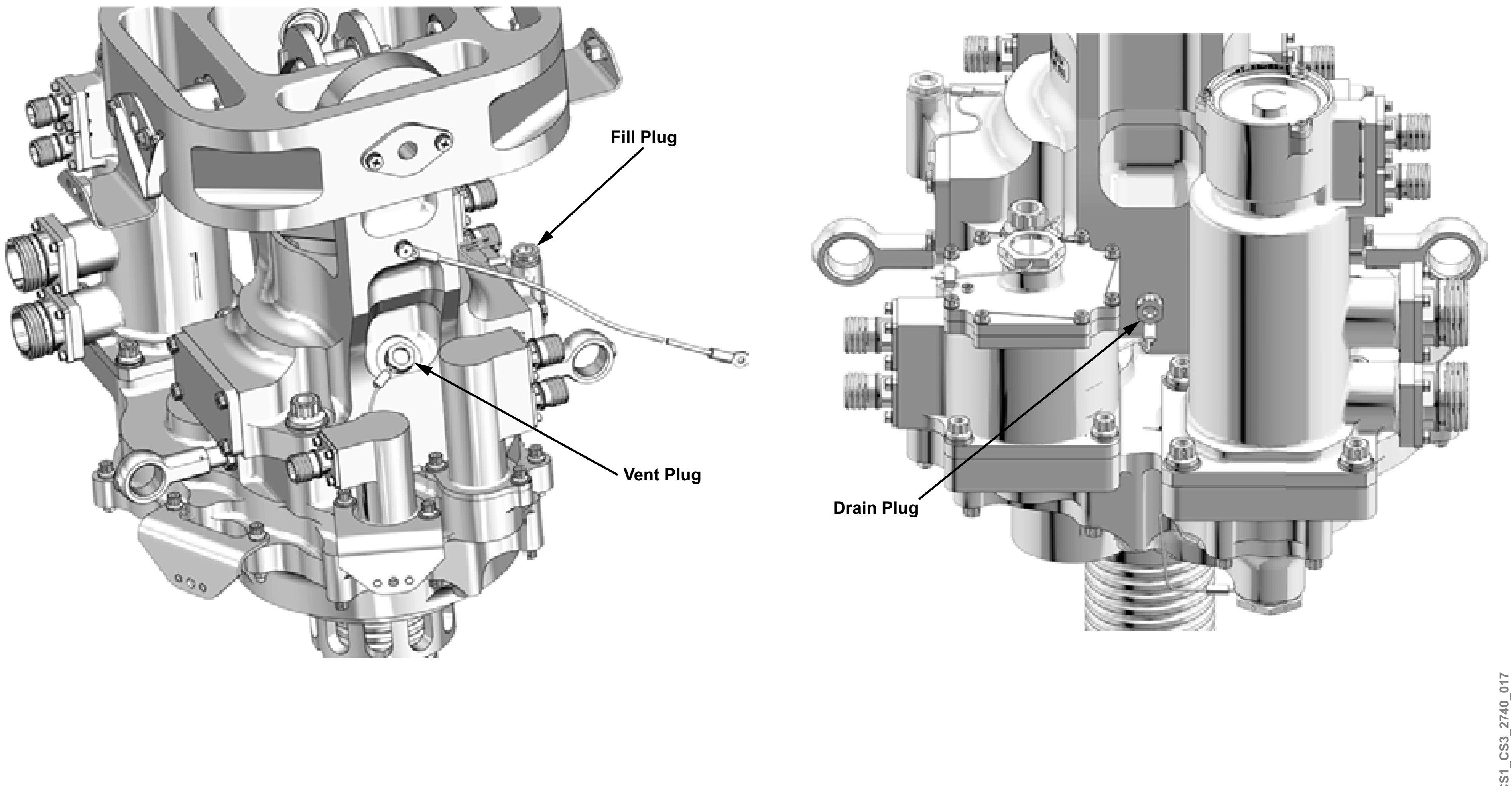


Figure 76: HSTA No-Back Device Servicing

HORIZONTAL STABILIZER TRIM ACTUATOR SECONDARY NUT AND BALL NUT LUBRICATION

The secondary nut and the ball nut have separate grease fittings. Sufficient grease has been applied when grease begins to flow from the relief valve.

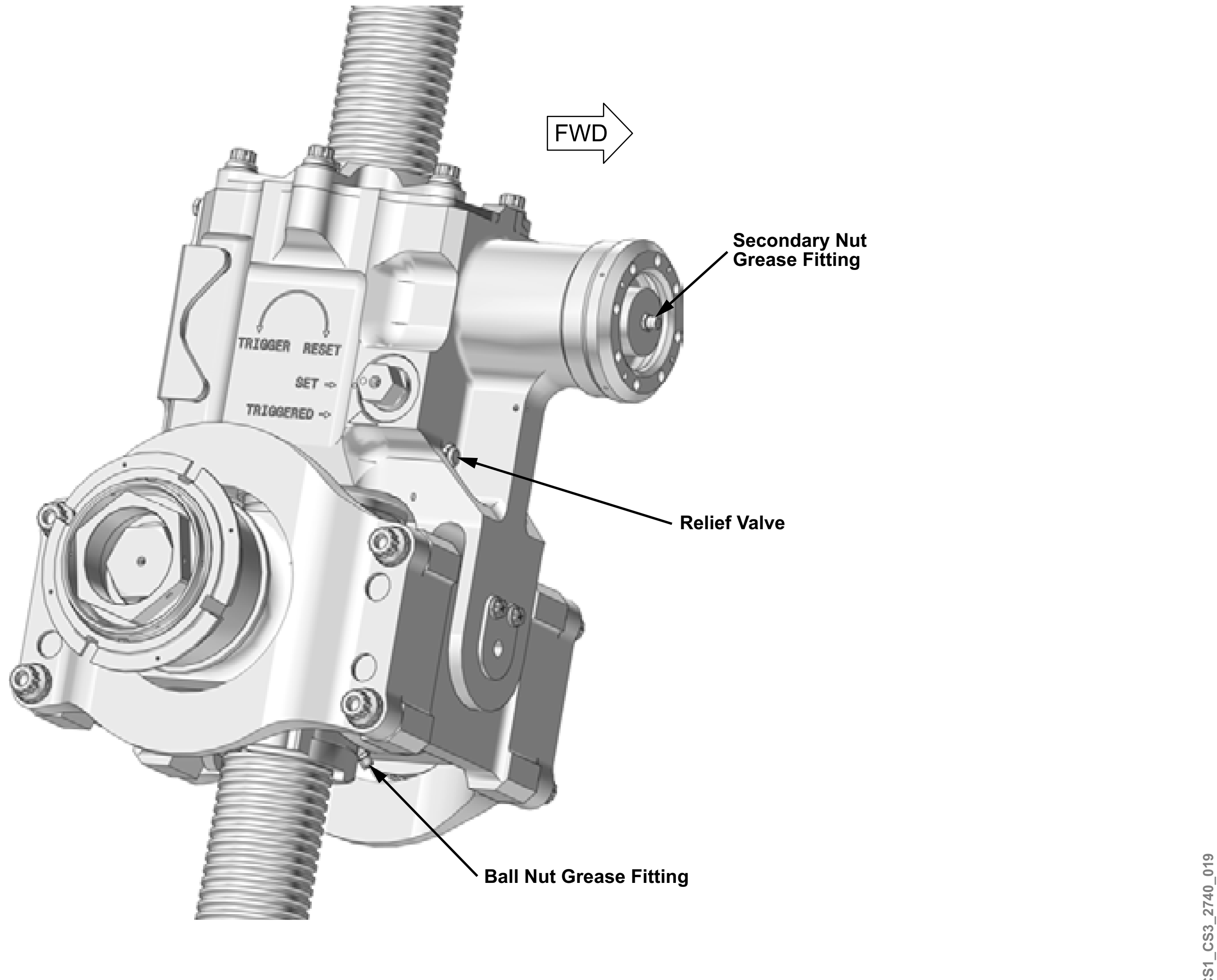
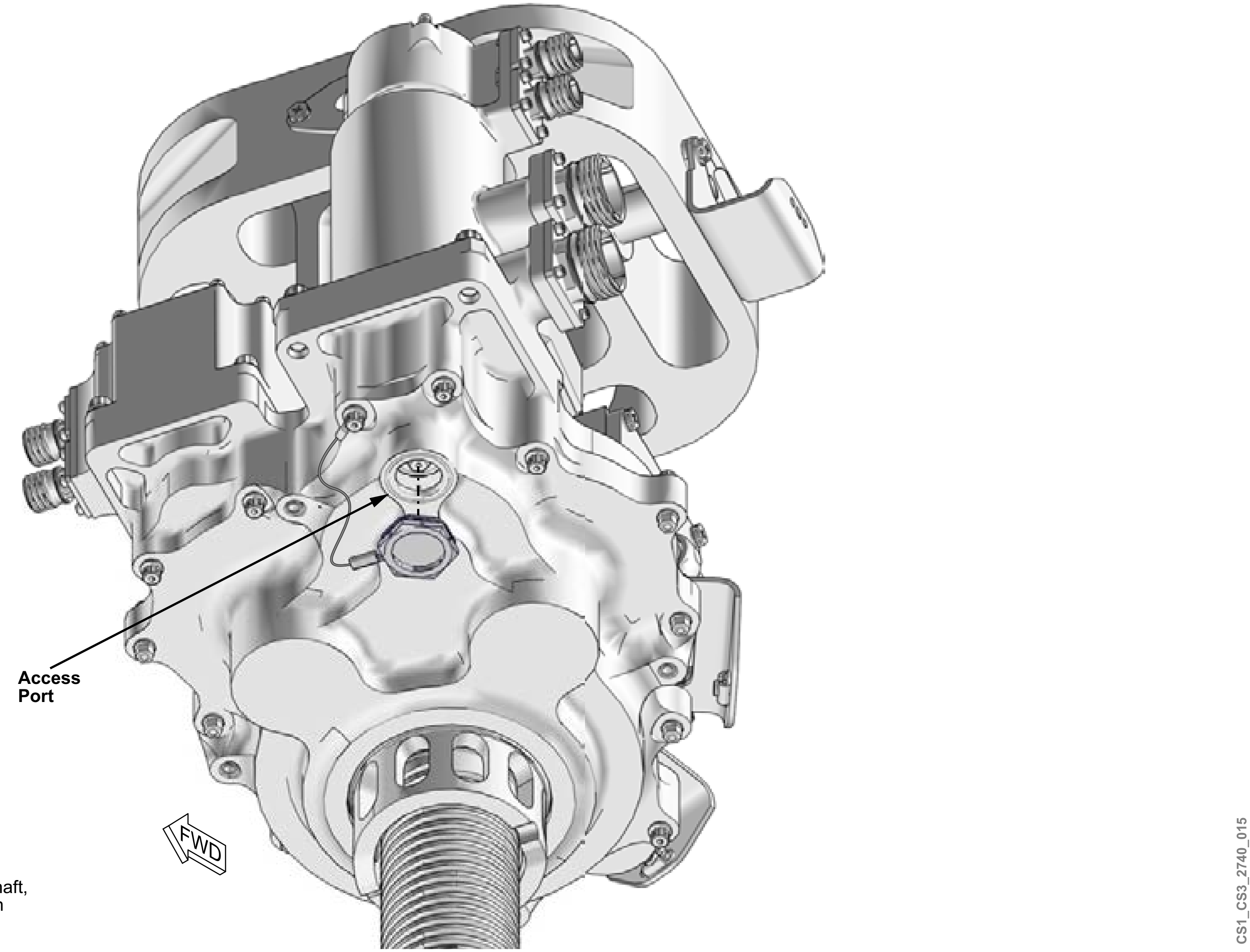


Figure 77: HSTA Secondary Nut and Ball Nut Lubrication

MANUAL DRIVE FITTING

To avoid the need to remove the tandem motors of the HSTA to rotate it manually, a manual drive fitting was added. The fitting is located directly under the tandem motor assembly, and is protected by an exterior cap.

The manual drive fitting is used to test the integrity of the HSTA brake and slip clutch components.



NOTE

Manual drive feature added to motor shaft, eliminating need to remove motor when using manual drive.

Figure 78: Manual Drive Fitting

SECONDARY NUT ASSEMBLY AND TIE ROD RESET

The secondary load path functional check is performed periodically on the tie rod latch/lock assembly and on the secondary nut latching lock mechanism. These procedures are necessary to make sure that both mechanisms operate correctly.

Horizontal Stabilizer Trim Actuator Tie Rod Latch/Lock Assembly Functional Check

The tie rod latch/lock mechanism can be check by disconnecting the upper primary load path, followed by a tensile load application.

Prior to testing, verify the secondary nut is unlocked by making sure the mark on the reset nut is aligned with the SET position. Check that the tie rod is unlocked by making sure the mark on the reset rod is aligned with the fixed index mark on the tie rod.

To test the tie rod latch/lock mechanism, ground support equipment is installed between the horizontal stabilizer and the aircraft structure. Ground support equipment (GSE) is used to support and reposition the horizontal stabilizer while one of the upper gimbal pins is removed.

Use the GSE to lower the horizontal stabilizer until an audible click is heard and there is an axial movement of 8.35 to 11.43 mm (0.25 to 0.45 in.) between primary and secondary load paths along with the rotation of the reset rod relative to the tie rod. The displacement should be between 50° to 135°. This is an indication that the tie rod latched and locked. The indication mark on the tie rod should not be in line with the indication mark on the reset rod.

To reset the latch/lock mechanism, realign the reset rod mark with the tie rod index mark using a breaker bar and ½ in socket. Continue to hold the reset rod in position while raising the horizontal stabilizer so that the gimbal pin can be reinstalled. After the gimbal pin is installed, the breaker bar and socket can be removed. To ensure the splines are reset, use a torque wrench to rotate the motor manual drive clockwise until a resistance is felt, then continue for four turns. Make sure the clutch slips and the ballscrew does not rotate during the last four turns. This confirms the splines are engaged in the locked position.

Remove the GSE and perform an operational test of the horizontal stabilizer.

Refer to the latest Aircraft Maintenance Publication for complete details.

CAUTION

1. **To prevent damage to the aircraft, ground support equipment, or injury to persons, do not operate the pitch trim switch of the sidestick controller during the test.**
2. **Do not exceed 500 in lb (56.50 Nm) of torque when turning the reset rod, otherwise damage to the horizontal stabilizer trim actuator can occur.**

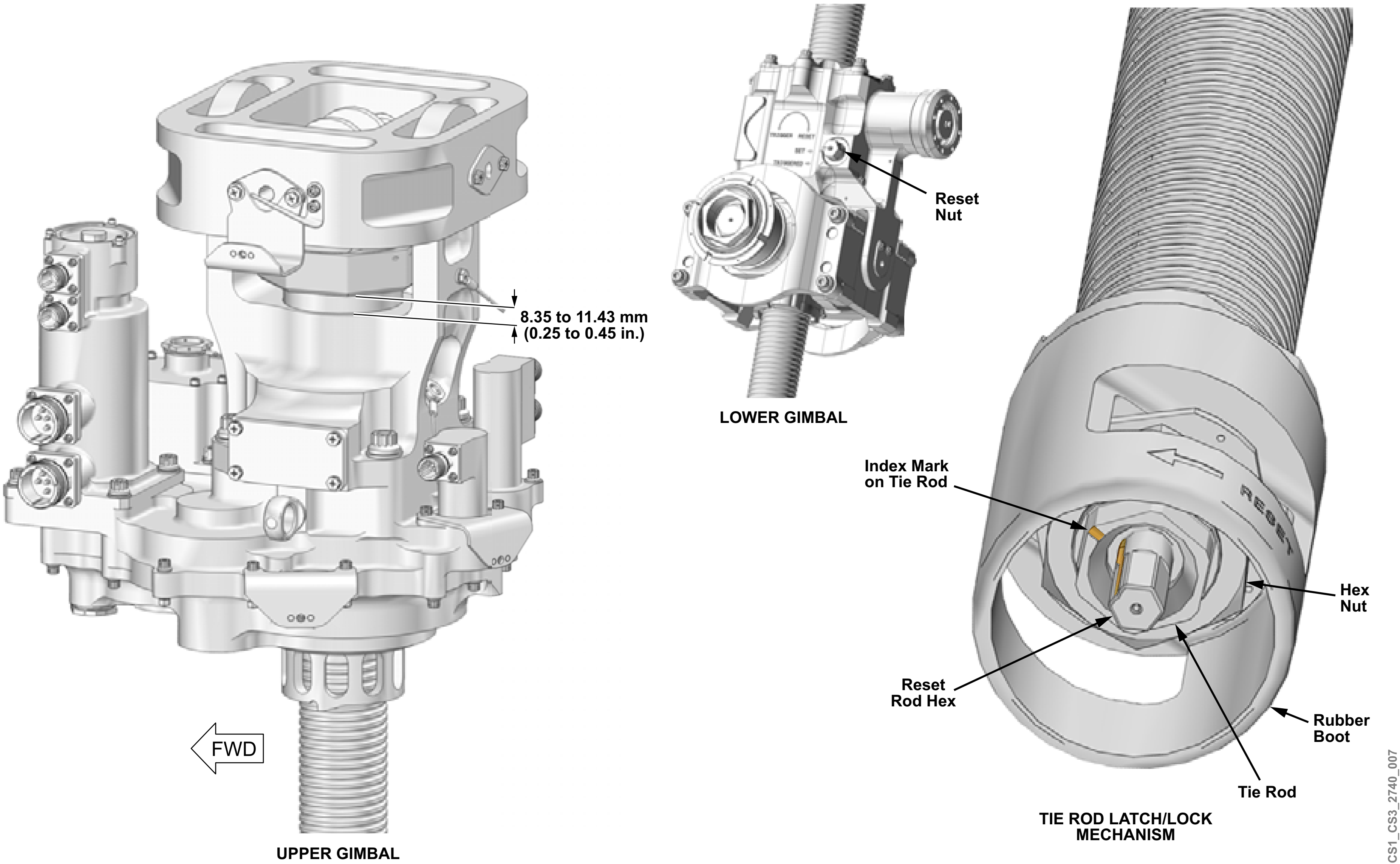


Figure 79: HSTA Secondary Nut Assembly and Tie Rod Reset

Horizontal Stabilizer Trim Actuator Torque Limiting Clutch and Secondary Nut Locking Mechanism Functional Check

This procedure is necessary to make sure that the clutch slips at the expected torque range and that the secondary nut locking mechanism is working properly.

Check that the reset rod mark at the end of the ballscrew is aligned with the index mark on the tie rod.

Rotate the manual drive at the motor shaft using a 5/16 in socket to confirm that the brake is locked. Rotate the motor manual drive between one and three full rotations, and observe that the clutch slips and the ballscrew does not rotate.

Turn the secondary nut reset drive in the counterclockwise direction to the TRIGGERED position using a 3/4 in socket and breaker bar to lock the secondary nut.

The slip clutch is checked by turning the manual drive shaft at the motor with a torque wrench and measuring the torque require to turn the HTSA. The motor shaft rotation forces the slip clutch to slip when the torque value is exceeded.

Rotate the manual drive at the motor shaft using a 0 to 30 inlb torque wrench with a 5/16 in socket in the CCW direction through one full rotation, noting the breakout torque and the running torque. Rotate the torque wrench in the CW direction through one full rotation, noting the breakout torque and the running torque. The measurements are repeated three times within a 5 minute period. The torque value should be 16.5 to 21.0 in.lbf.

To check the secondary nut locking mechanism, command the HSTA in a nose up for one second and then a nose down direction for one second. Make sure the horizontal stabilizer did not move and the following messages are displayed on the EICAS:

- STAB TRIM FAIL
- FLT CTRL FAULT
- Perform a Latched Fault Clearing on the onboard maintenance system (OMS) Perform LRU/System Operations page to clear the faults (see EFCS Practical Aspects for more details).

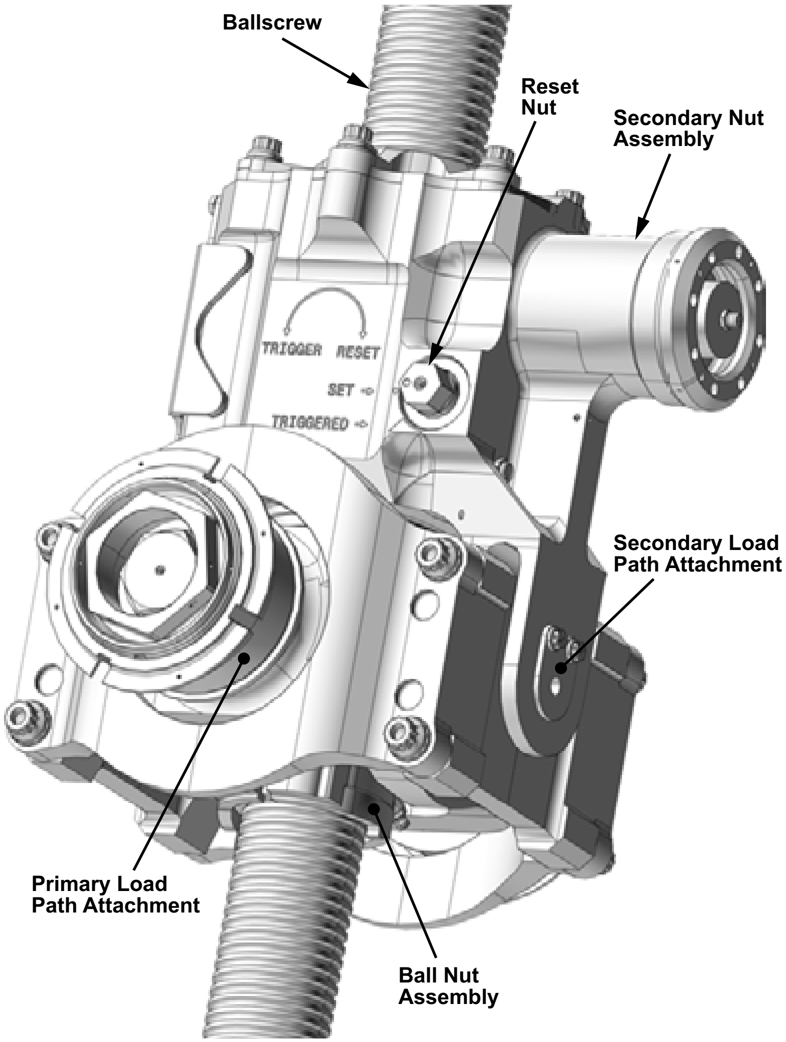
Turn the secondary nut reset drive in the clockwise direction to the SET position to unlock the secondary nut.

The secondary nut latching lock mechanism can be checked by rotating the secondary nut counterclockwise to the TRIGGERED position in order to release the latching lock mechanism, followed by a trim command. The motor will rotate, but ballscrew will remain lock due to the slip clutch slippage and a STAB TRIM FAIL caution message displayed on the EICAS.

Refer to the latest Aircraft Maintenance Publication for complete details.

CAUTION

1. **Do not use a ratcheting handle since this would allow the device to impact against its internal stop, resulting in higher internal loads than necessary.**
2. **Use care when triggering the secondary nut since pinion will release approximately 60 ft-lb of torque through more than 90° of rotation**



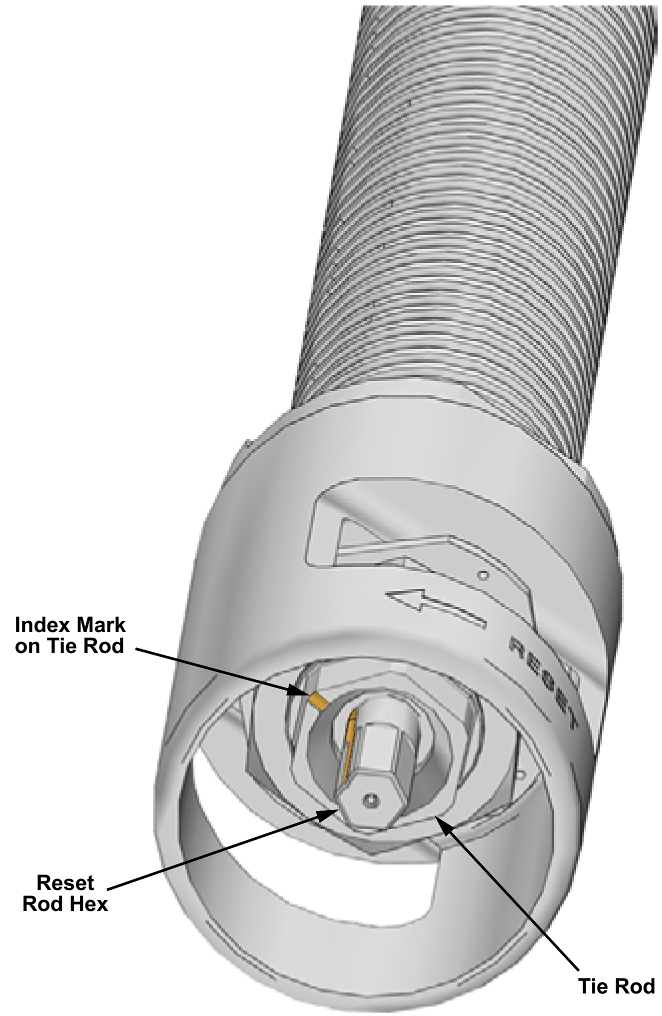


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Figure 80: HSTA Secondary Nut Assembly and Tie Rod Reset

27-60 SPOILER SYSTEM

GENERAL DESCRIPTION

There are four symmetrical pairs (left and right wings) of multifunction spoiler (MFS) that perform roll assist, proportional lift dumping, and ground spoiler functions. The MFSs are hydraulically-powered and electrically controlled. There is a single power control unit (PCU) for each spoiler. For proportional lift dumping and ground spoiler functions, the spoilers are deployed symmetrically on both sides of the aircraft. For the roll assist, the spoilers are deployed asymmetrically.

Automatic deployment of the MFS is normally controlled by the electronic flight control system (EFCS).

The crew can also use the flight spoiler control lever (FSCL) to send commands to the EFCS to extend and retract the MFSs.

The position of the MFS are monitored by linear variable differential transformers (LVDTs).

There is one dedicated ground spoiler (GS) per wing. The ground spoilers are hydraulically powered and electrically controlled. A single PCU actuates each GS. The ground spoilers operate symmetrically to provide ground lift dumping at touchdown. Automatic deployment of the ground spoilers is computed in the primary flight control computers (PFCCs) and is sent to the ground spoiler control module (GSCM) via a remote electronic unit (REU). Deployment logic is based on weight-on-wheels (WOW), wheel speed, thrust lever angle (TLA) and radio altitude data. The automatic ground lift dump function does not need to be armed by the crew.

The ground spoilers are monitored by proximity sensors, reporting to the landing gear and steering control units (LGSCUs).

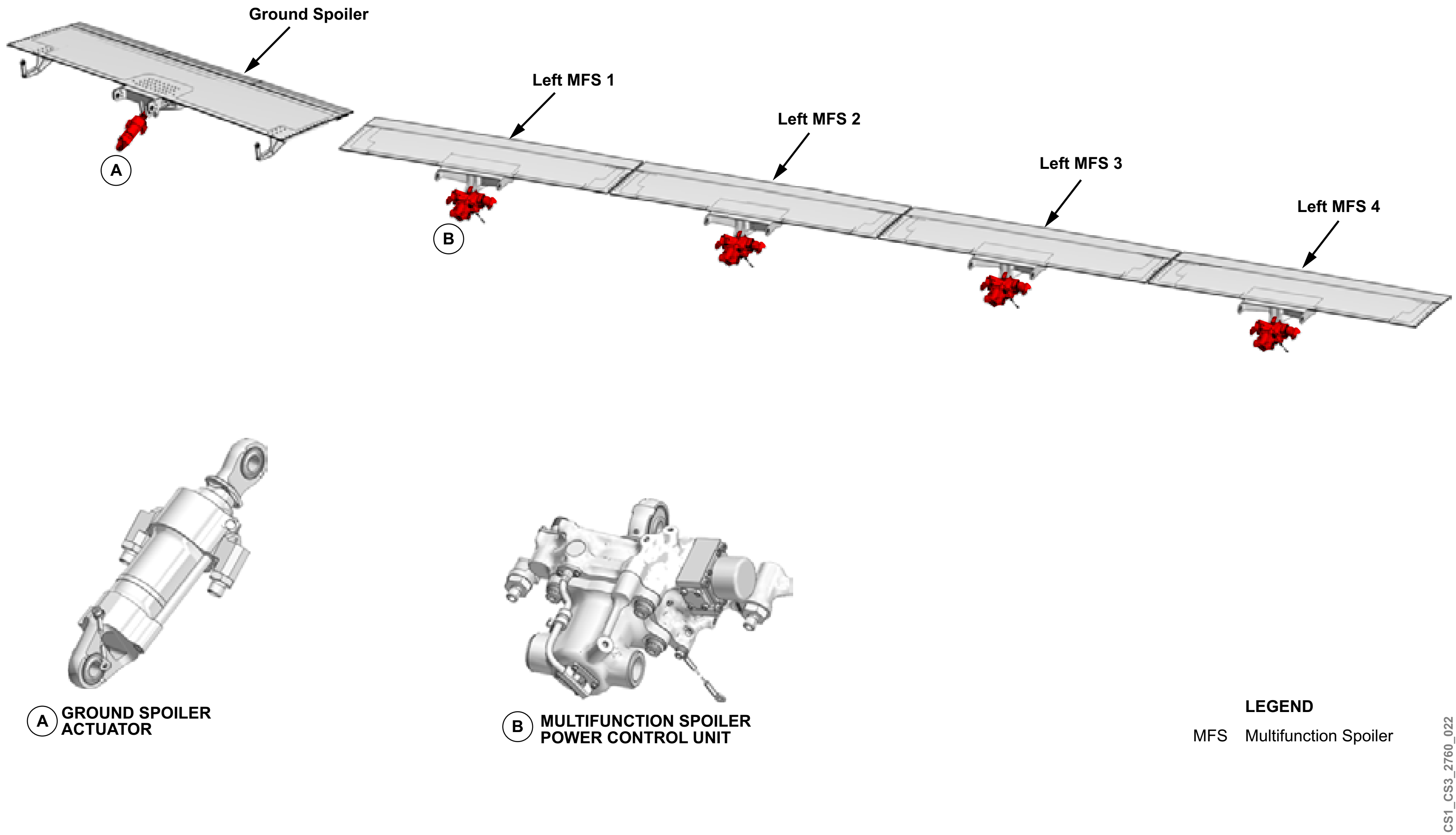


Figure 81: Multifunction Spoilers and Ground Spoilers

COMPONENT INFORMATION

MULTIFUNCTION SPOILER POWER CONTROL UNIT

The multifunction spoilers are each controlled by a single power control unit (PCU). The PCUs are attached to the wing fitting with a lined, self-lubricating bushed fitting. They are attached to the spoiler panel with a dual-row roller bearing rod end.

The PCU is an electrohydraulic servo actuator. Position and direction control of the actuator is provided by the torque motor controlled, electrohydraulic servovalve (EHSV). Each actuator also contains a linear variable differential transformer (LVDT) for position feedback. The LVDT wiring is housed externally inside a protective conduit. A single electrical connector is used for the EHSV and LVDT signals.

The PCU has thermal relief protection and a manual override. The manual override is used to manually raise the multifunction spoiler for maintenance.

The PCU has hydraulic pressure and return fittings.

LEGEND

EHSV	Electrohydraulic Servovalve
LVDT	Linear Variable Differential Transformer

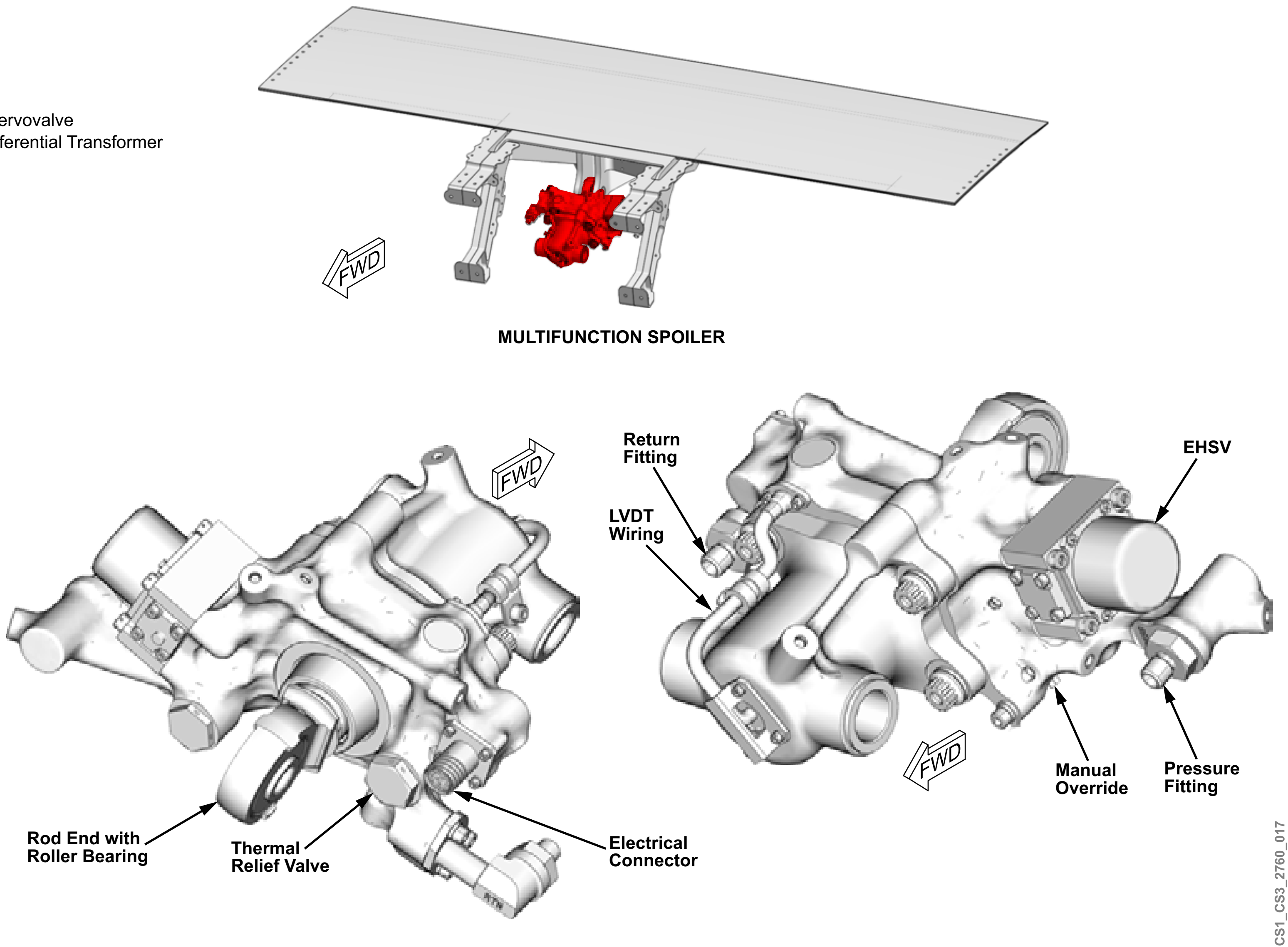


Figure 82: Multifunction Spoiler Power Control Unit

GROUND SPOILER ACTUATOR

The ground spoiler actuator (GSA) is an unbalanced area type hydraulic actuator, converting fluid pressure and flow into linear piston force and rate. It includes a cylinder, piston (ram), seals, and rod ends. The GSA is extended or retracted based on the flow from the extend and retract ports of the ground spoiler control module (GSCM). The GSA contains no valves, sensors, or electrical controls.

The forward end of the actuator attaches to the wing attach point, which is mounted on the wing rear spar. The actuator piston rod end is connected to the spoiler attach fitting.

The ground spoilers have a separate position detection system. Each ground spoiler has a single proximity sensor, which indicates whether the surface is in the extended or retracted position (near/far). This information allows for the detection of a fully retracted GS and is communicated to the EFCS via the landing gear and steering control units (LGSCUs).

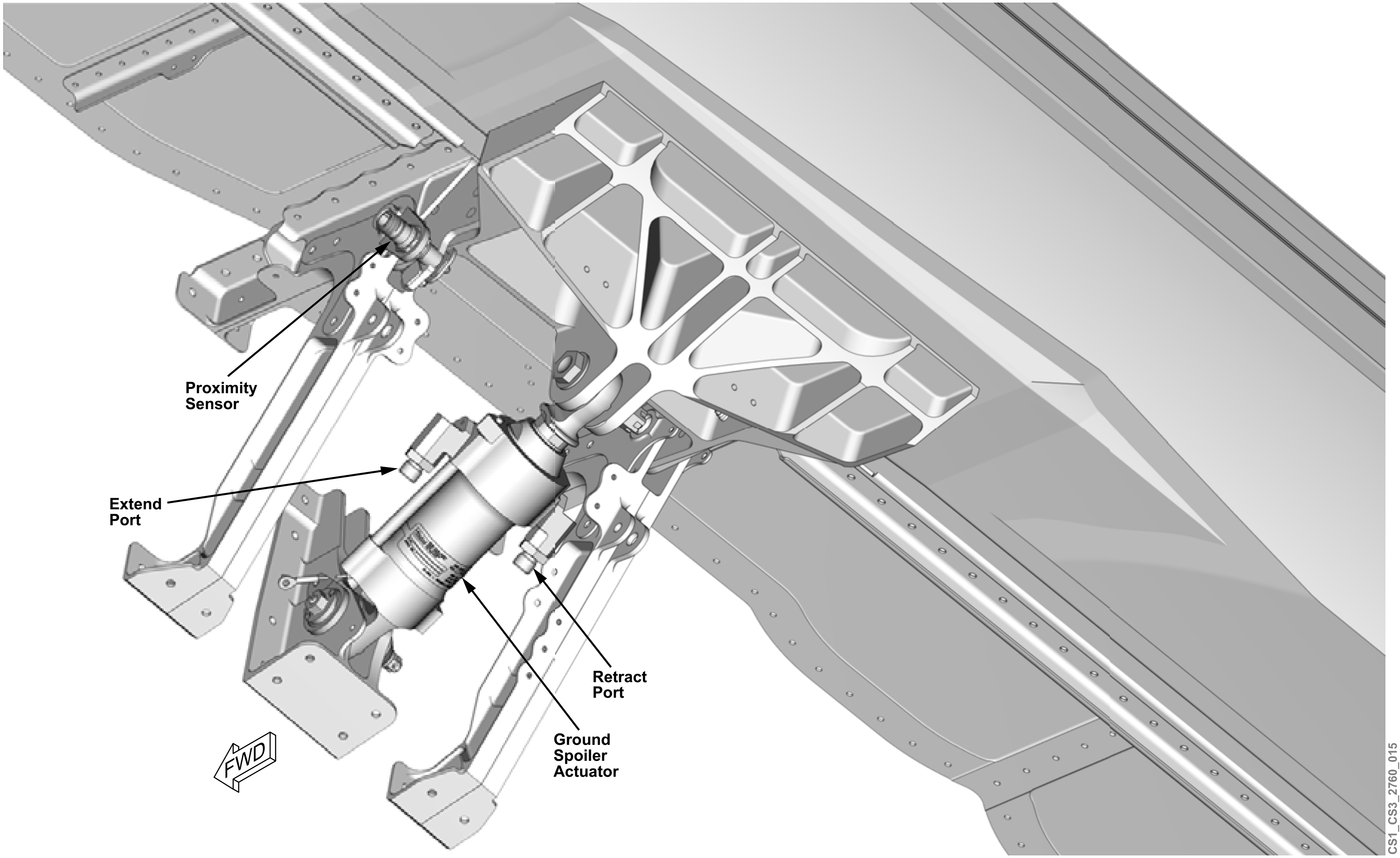


Figure 83: Ground Spoiler Actuator

GROUND SPOILER CONTROL MODULE

The ground spoiler control module (GSCM) hydraulically controls a pair of actuators connected to the ground spoilers.

The GSCM contains two shutoff valves (SOVs). Both SOVs must be energized in order for the ground spoilers to extend. The SOVs are commanded independently to prevent inadvertent deployment of the ground spoilers.

The GSCM has a manual override feature that allows maintenance personnel to raise the ground spoiler panel on the ground.

The GSCM is located in the belly fairing, forward of the left main landing gear wheel well.

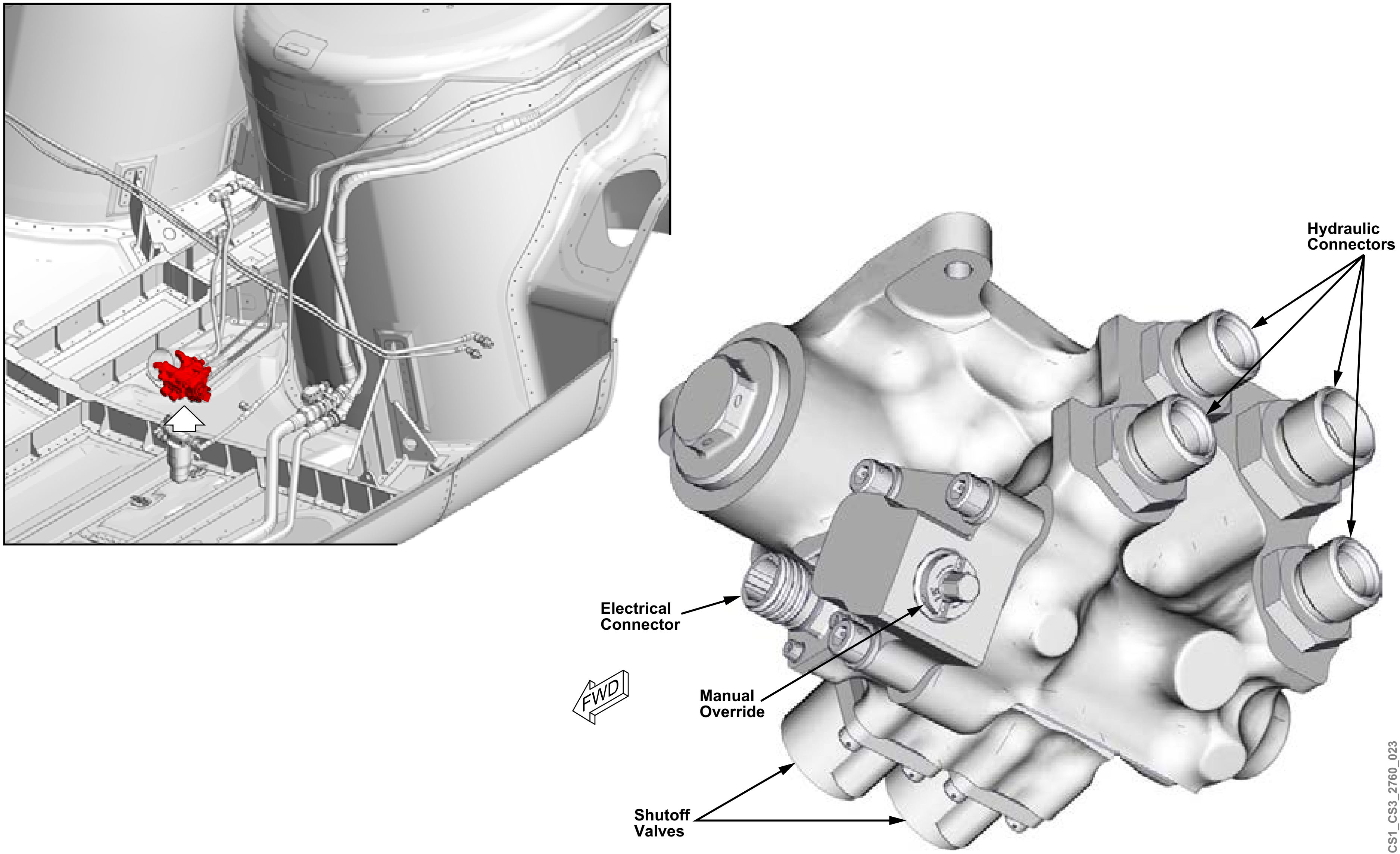


Figure 84: Ground Spoiler Control Module

DETAILED COMPONENT INFORMATION

MULTIFUNCTION SPOILER POWER CONTROL UNIT

The multifunction spoiler power control unit (PCU) consists of the following components:

- Electrohydraulic servovalve
- Thermal relief valve
- Upfloat check valve with manual override
- Actuator assembly

The pressure inlet fitting has a line replaceable in-line strainer to prevent particulate contamination from causing a malfunction of the PCU.

The PCU uses a torque motor-controlled, electrohydraulic servovalve (EHSV), to meter hydraulic fluid flow to the actuator in response to signals received from the remote electronic unit (REU). The EHSV is spring-biased to the retract position.

A thermal relief valve is installed on the retract chamber between the actuator and the upfloat check valve. It prevents overpressurization of the fluid trapped by the upfloat check valve after thermal expansion. If the pressure of the trapped fluid exceeds 3600 ± 100 psi, the thermal relief valve opens.

An upfloat check valve prevents hydraulic fluid from exiting the retract chamber of the actuator when there is a loss of system pressure. This prevents the multifunction spoiler panel from upfloating in flight. The EHSV allows the multifunction spoiler to retract under air loads.

The manual override allows maintenance personnel to deplete hydraulic pressure to manually raise the spoilers for maintenance.

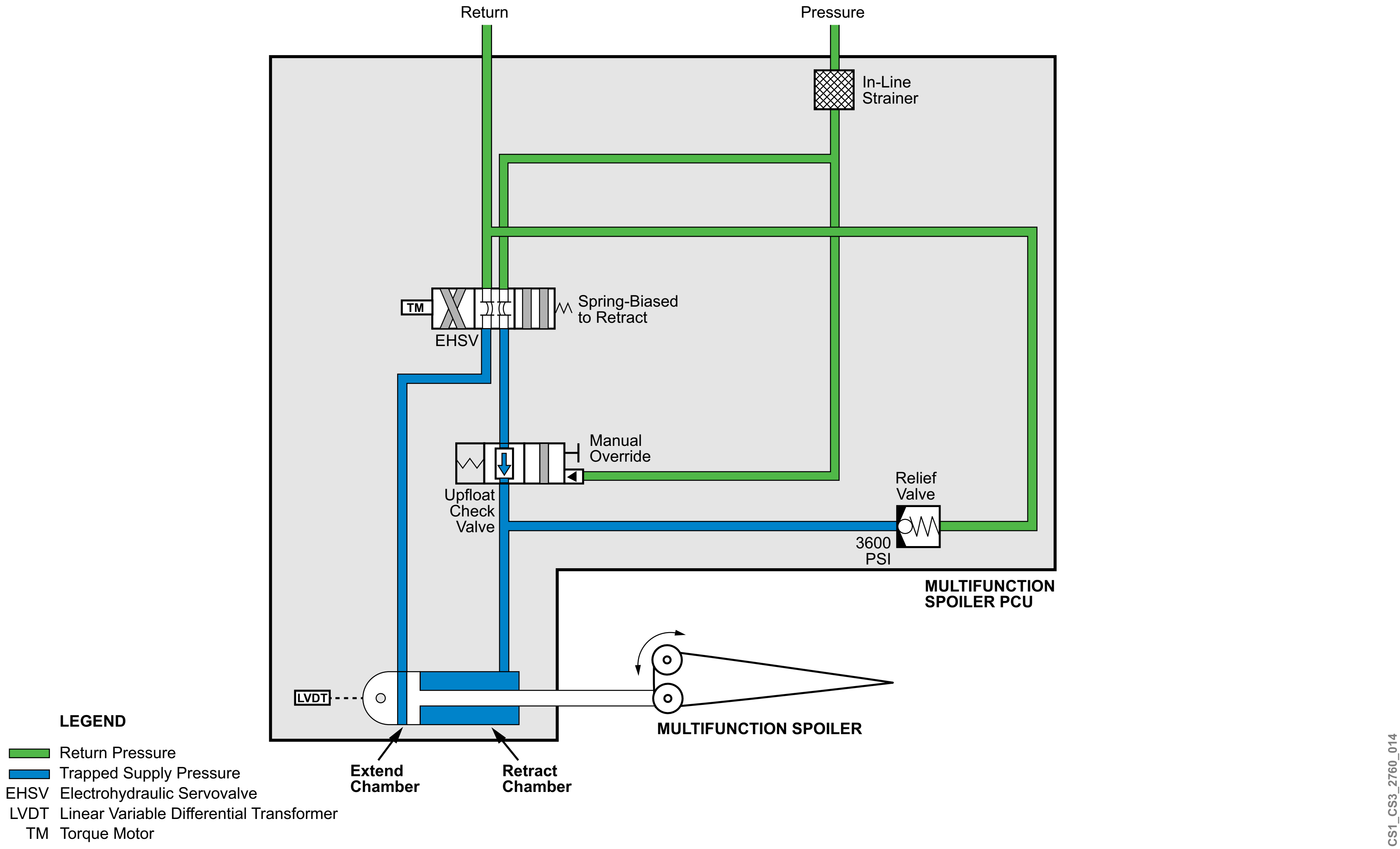


Figure 85: Multifunction Spoiler Power Control Unit Schematic - Hydraulic System Off

Multifunction Spoiler Deploy Command

With hydraulic pressure supplied, the upfloat check valve is positioned to allow fluid to flow from the retract side of the actuator. To deploy the multifunction spoiler, the EHSV torque motor is biased to the extend position. The EHSV spool ports hydraulic fluid to extend the PCU actuator. Fluid returns through the upfloat check valve to the hydraulic system return.

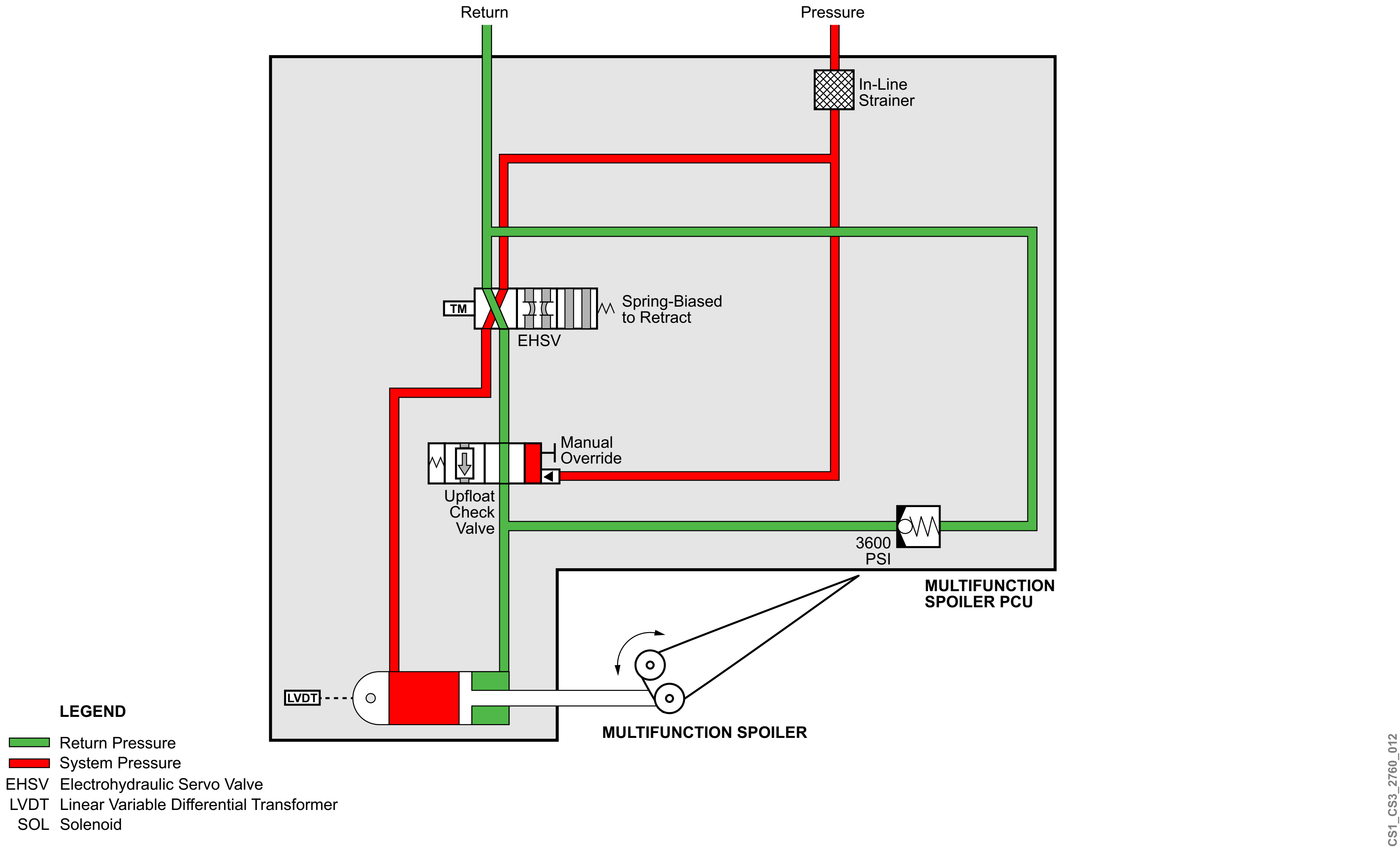


Figure 86: Multifunction Spoiler Deploy Command

Multifunction Spoiler Deployed

When the spoiler reaches the deploy position, the EHSV torque motor is biased to neutral, the EHSV spool centers, and the spoiler maintains its position. If the air loads become excessive, the relief valve provides a fluid path to the system return to allow the spoiler to blow down.

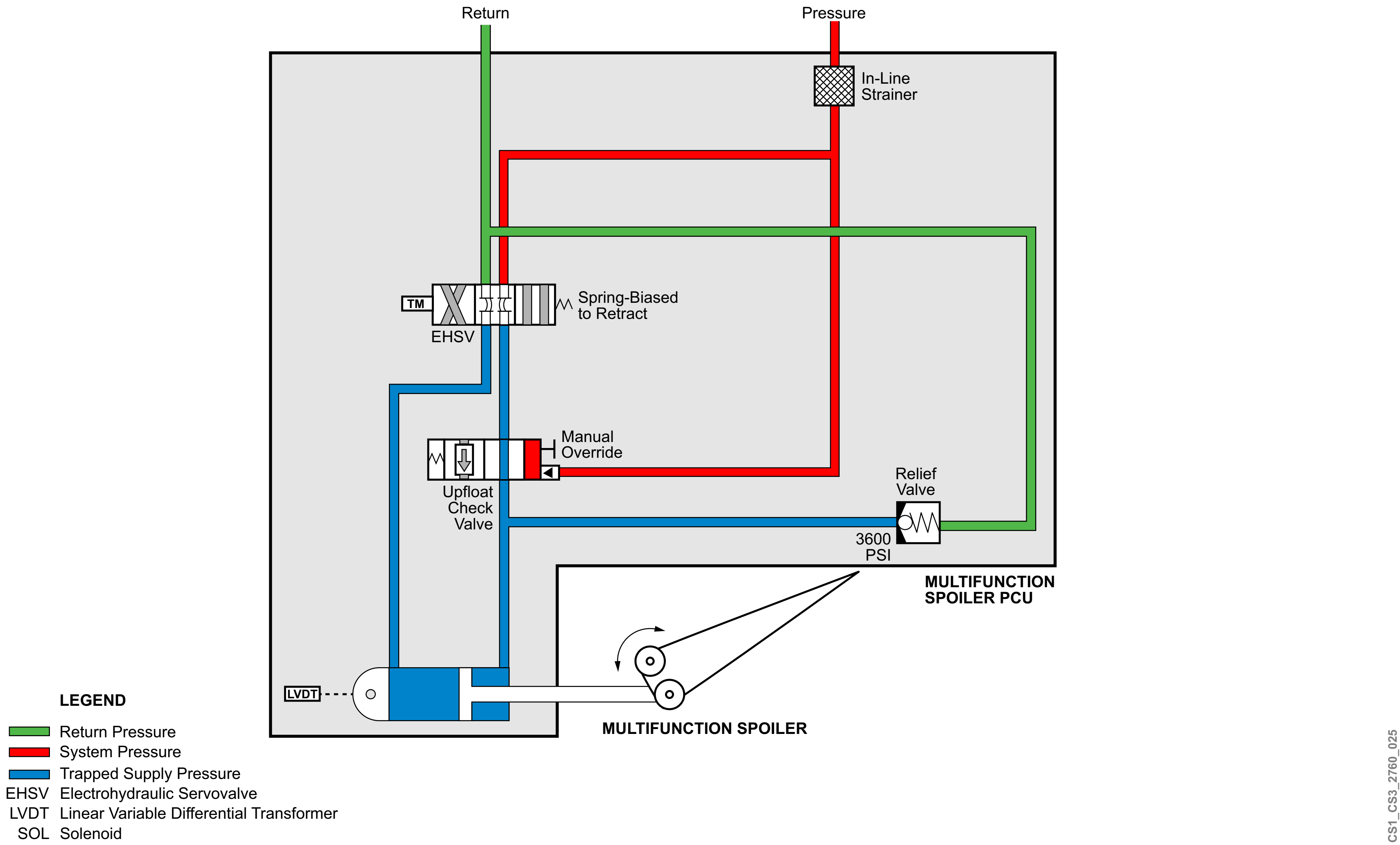
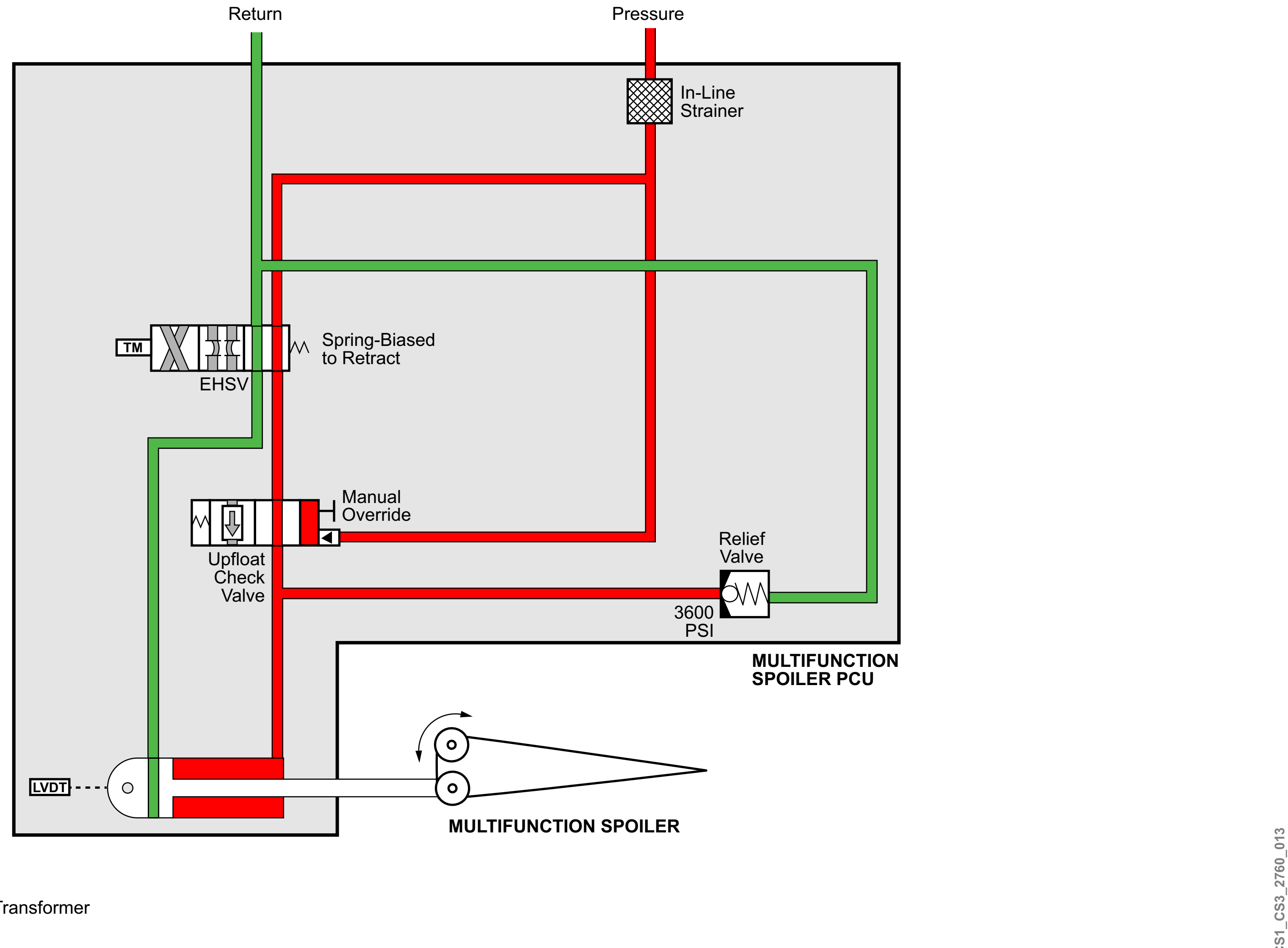


Figure 87: Multifunction Spoiler Deployed

Multifunction Spoiler Stowed

When the spoiler is commanded to stow, the EHSV torque motor is biased to the retract position. The EHSV spool ports hydraulic fluid to retract the actuator. Fluid returns through the EHSV to the hydraulic system return.

After the spoiler is stowed, the EHSV torque motor bias command is reduced, but is maintained in the retract direction. The spool, which is spring-biased to the retract position, is assisted by the EHSV retract bias command to keep the spoiler in the stowed position.



LEGEND

Return Pressure
System Pressure
EHSV Electrohydraulic Servovalve
LVDT Linear Variable Differential Transformer
SOL Solenoid

Figure 88: Multifunction Spoiler Stowed

GROUND SPOILER CONTROL MODULE

The ground spoiler control module (GSCM) contains two solenoid-operated, electrohydraulic shutoff valves (SOVs). The SOVs are three-way, two-position, normally-closed, solenoid-operated valves. The SOVs are connected in series, and control the position of the mode select valve (MSV). When both SOVs are energized, hydraulic pressure moves the MSV to extend both ground spoiler actuators. If either SOV is de-energized, the MSV moves to retract both ground spoiler actuators.

The MSV spool is spring-biased to retract when the control pressure is low. This results in the actuators retracting the ground spoiler (GS) surfaces when there is a significant drop in hydraulic pressure.

The upfloat check valve is used to trap hydraulic fluid in the retract chamber of the actuators following a loss of system pressure, preventing ground spoiler upfloat in flight. A manual override allows the release of hydraulic pressure by maintenance personnel to permit raising of the GS panel.

The thermal relief valve is installed between the upfloat check valve and the fluid line to the actuator retract chamber. Its purpose is to prevent overpressurization of the fluid trapped by the upfloat check valve after thermal expansion. If the pressure of the trapped fluid exceeds the threshold (3600 ± 100 psi), the thermal relief valve opens.

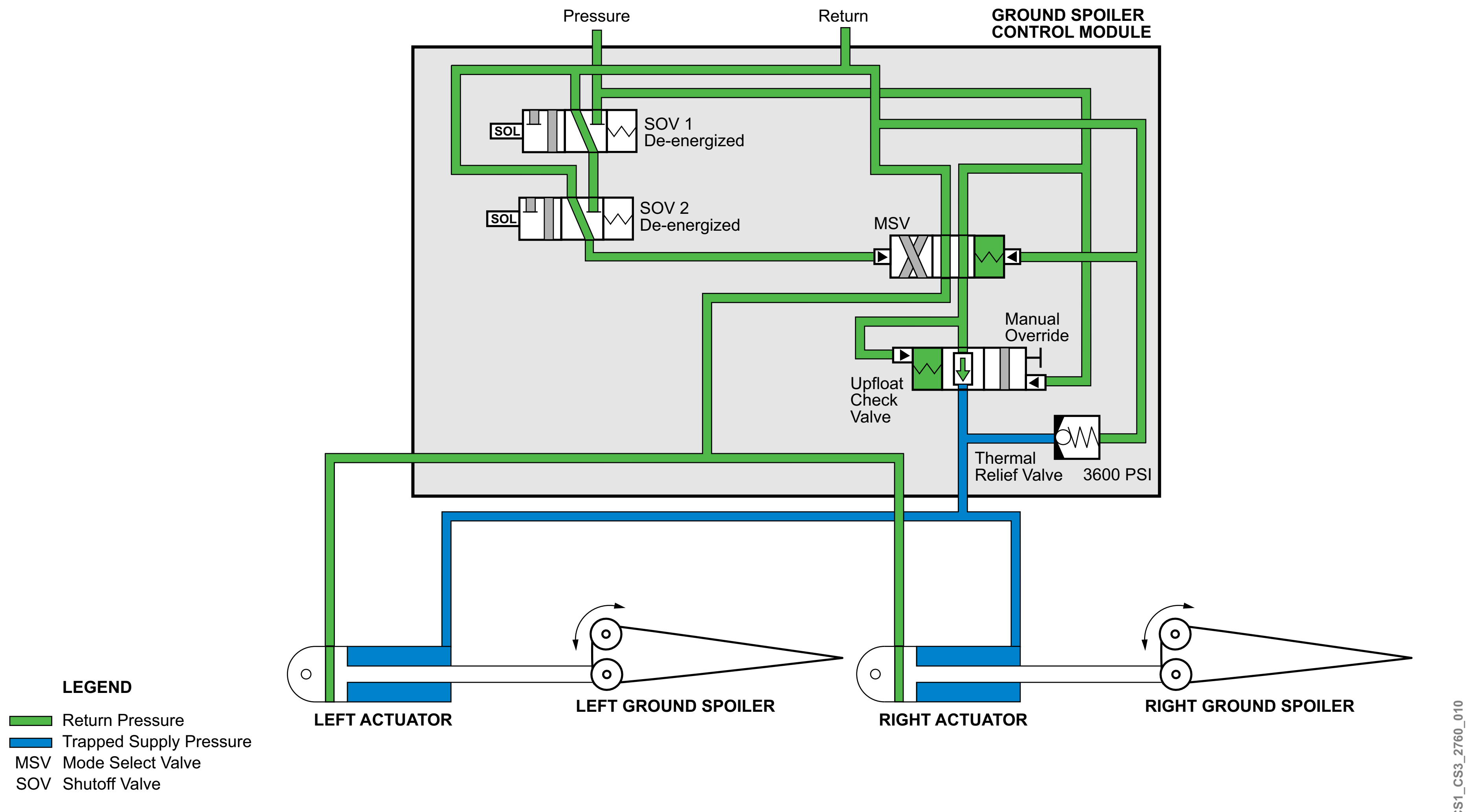


Figure 89: Ground Spoiler Control Module

Ground Spoiler Deployed

Commands to deploy the ground spoilers are received from REU 4. To deploy, the REU energizes SOV 1 and SOV 2. This routes hydraulic pressure to the mode select valve, which routes hydraulic pressure to the deploy side of both actuator chambers, and return flow back to system return.

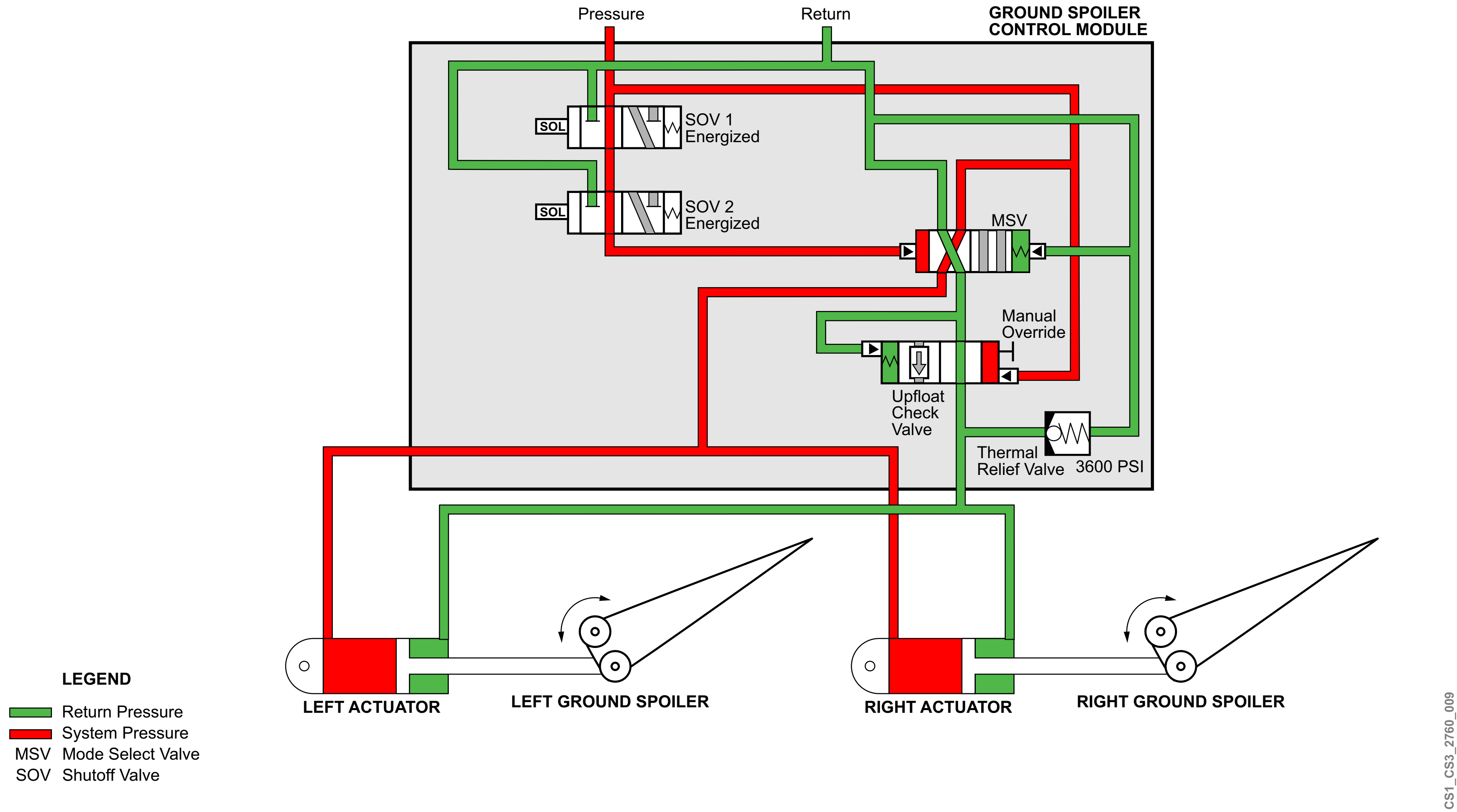


Figure 90: Ground Spoilers Deployed

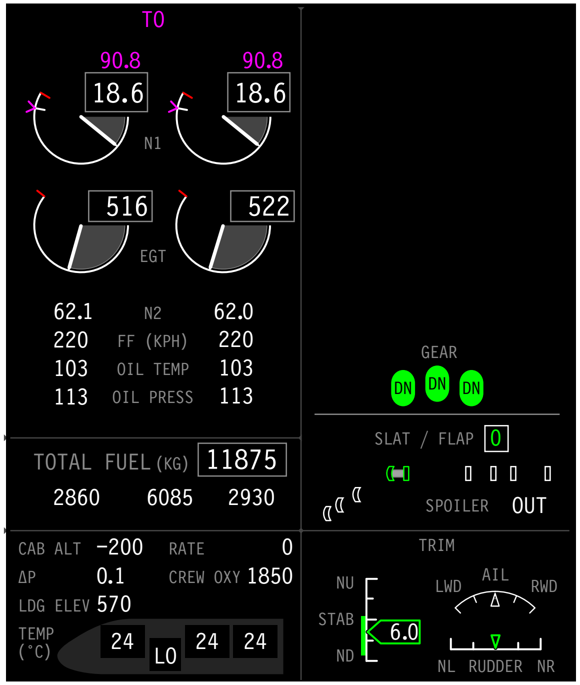
CONTROLS AND INDICATIONS

The engine indication and crew alerting system (EICAS) page provides a spoiler indicator.

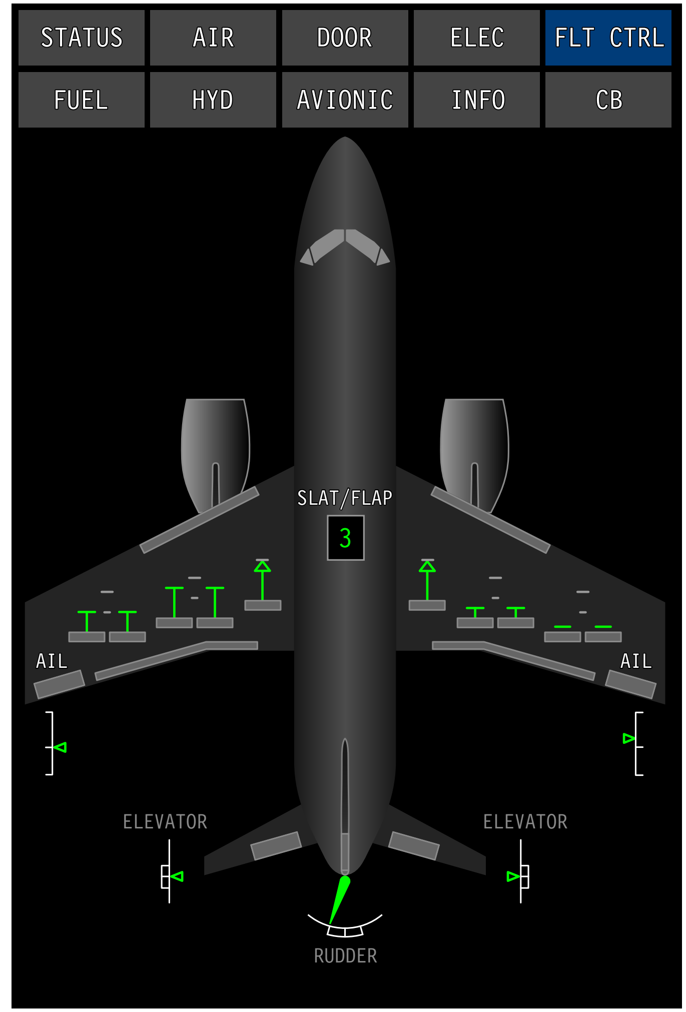
The following are displayed on the flight controls synoptic page:

- Multifunction spoiler (MFS) position and state
- Ground spoiler (GS) position and state

SPOILER INDICATOR	
Symbol	**Condition**
SPOILER OUT	Commanded
SPOILER MAX	MAX Commanded
SPOILER OUT	Commanded (Landing gear down, flaps/slats position 4 or 5)
SPOILER MAX	MAX Commanded (Landing gear down, flaps/slats position 4 or 5)



EICAS PAGE



GROUND SPOILER POSITION INDICATOR	
Symbol	**Condition**
	Normal
	Fail
	Invalid

SPOILER SURFACE STATE	
Symbol	**Condition**
	Normal
	Fail
	Invalid

MFS POSITION INDICATOR AND STATE	
Symbol	**Condition**
	Normal
	Fail
	Invalid

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Figure 91: Spoiler Controls and Indications

OPERATION

In normal mode, control of the spoiler system is automatic. The electronic flight control system (EFCS) provides automatic roll assist, as well as automatic ground lift dumping. In the event that lift dumping is required in flight, the flight spoiler control lever (FSCL) can be selected to the desired position.

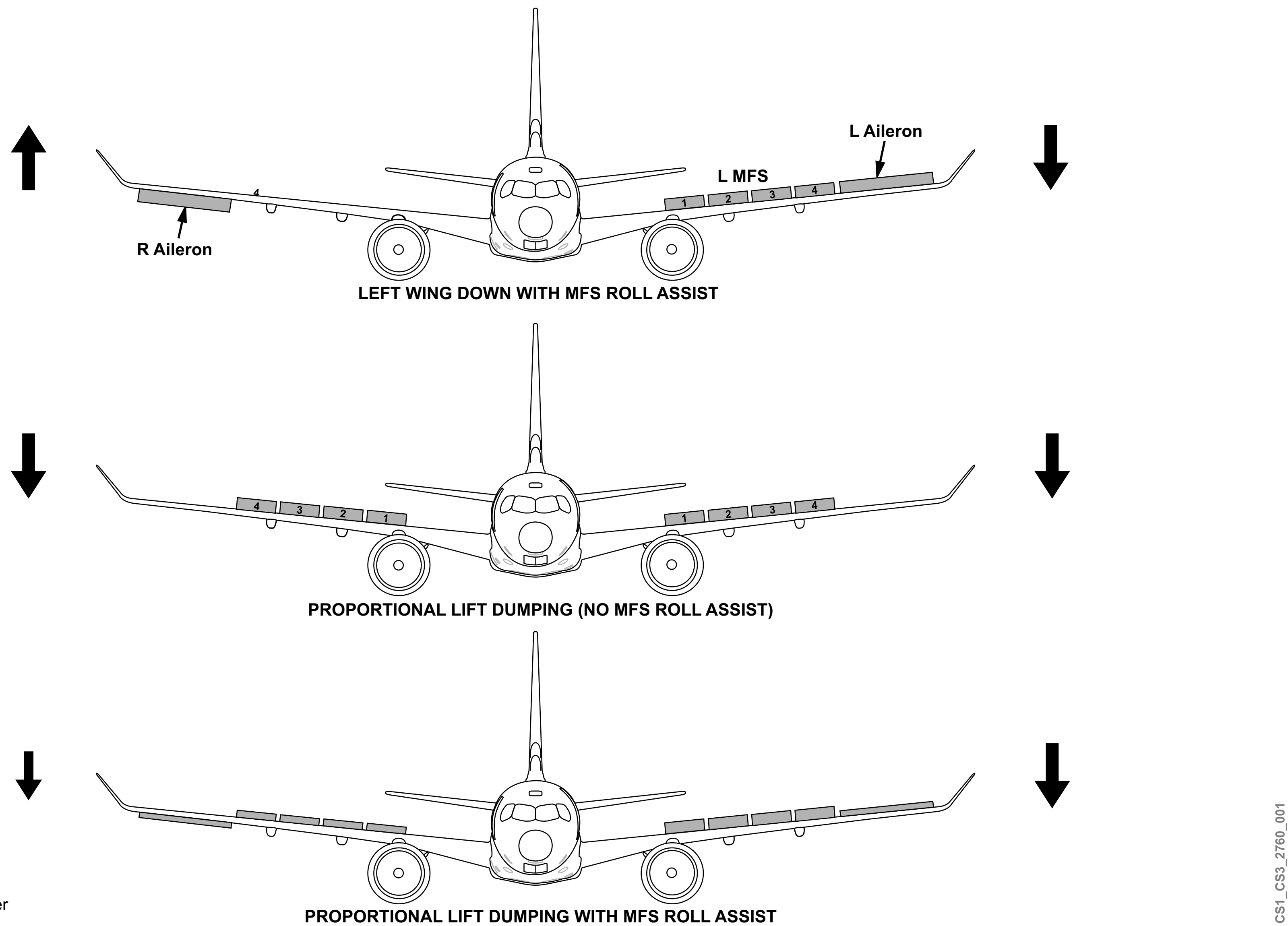
SPOILER OPERATION IN FLIGHT

During flight, the EFCS automatically controls the MFSs as required to assist in roll operation. The MFSs that are located on the wing with the upward moving aileron move up proportionally to the degree of roll required. The MFS roll assist function is not available during operation in alternate flight control unit (AFCU) direct mode.

A proportional lift dumping function allows the MFS to deploy symmetrically according to the FSCL selection. Roll assist remains available during proportional lift dumping.

If the spoilers are deployed during approach, when the landing gear is down and the slats/flaps are at position 4 or 5, a SPOILER DPLY advisory message is displayed. If the aircraft descends below 300 ft with the spoilers deployed, the SPOILER DPLY message is displayed as a caution.

In normal mode only, when the left or right thrust lever is set greater than 24° or during high angle-of-attack conditions, the MFSs retract automatically to the stowed position and the SPOILER MISMATCH advisory message displays on the EICAS page. To redeploy the MFS, the SPOILER lever must be placed to the RET position, then back to the desired setting.



LEGEND

MFS Multifunction Spoiler

Figure 92: Spoiler Operation in Flight

SPOILER OPERATION ON LANDING

The ground spoilers only deploy on touchdown. Each ground spoiler surface extends to 50° to dump lift during the landing rollout. Activation is automatic in normal and primary flight control computer (PFCC) direct modes. Deployment of the ground spoilers is not available in REU direct or AFCU modes. The PFCC requires input from the flight spoiler control lever (FSCL), weight-on-wheels (WOW) sensor, radio altimeter, thrust lever and wheel speed in order to deploy the ground spoilers.

The ground spoilers (GSs) only deploy on touchdown. Each ground spoiler surface extends to 50° to dump lift during the landing rollout. Activation is automatic in normal and primary flight control computer (PFCC) direct modes.

The multifunction spoilers (MFSs) automatically extend to 50° to dump lift during the landing rollout in normal mode and to 5° in PFCC direct mode. Further extension of the MFS in PFCC direct mode can be done using the flight spoiler control lever (FSCL).

The PFCC requires the following inputs to deploy the ground spoilers:

- Both main landing gear (MLG) weight-on-wheels (WOW)
- Radio altimeter
- Thrust lever position
- Engine running signal
- Wheel speed
- Pitch attitude (GS only)
- Airspeed

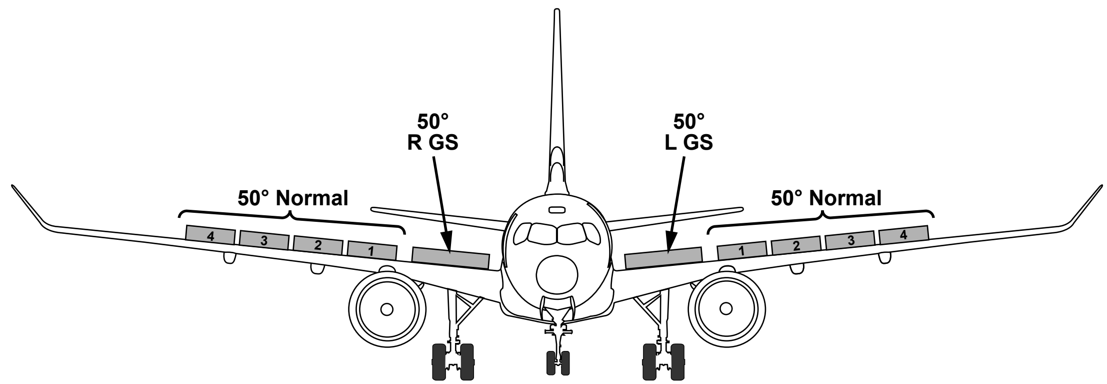
The spoilers arm for deployment on the ground with wheel speed greater than 60 kt or the thrust levers advanced or when the aircraft is in flight.

Deployment occurs when the WOW condition is sensed, rad alt is less than7 ft, and the wheel speed is greater than 16 kt. The GSs also require the pitch attitude to be less than 2.5°.

The spoilers stow automatically if the thrust levers are advanced above forward idle thrust, or when any of the arming conditions are no longer valid.

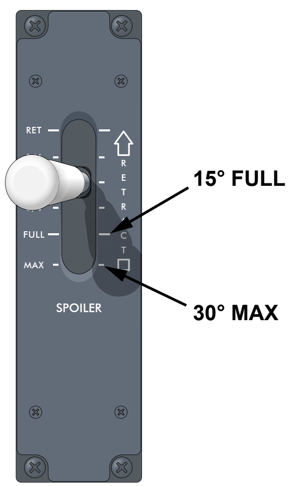
Automatic deployment of the MFS is not available in REU direct or AFCU modes. In the REU direct mode, the MFSs can be deployed manually using the FSCL. The GSs do not deploy in REU direct mode.

Neither GSs nor MFSs are available in AFCU mode.

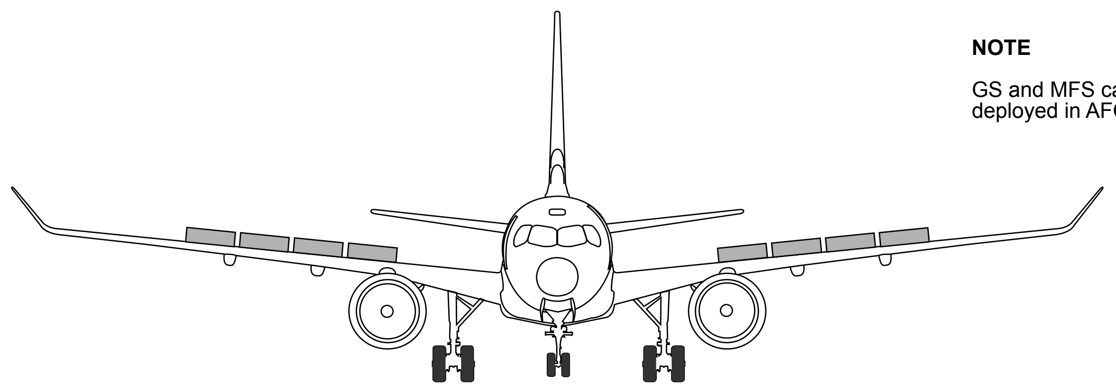


NORMAL AND PFCC DIRECT MODE

All spoilers deploy automatically on ground for lift dumping.



FLIGHT SPOILER CONTROL LEVER (REU DIRECT MODE)



REU DIRECT

No ground spoilers, MFS must be selected by pilot.

NOTE

GS and MFS cannot be deployed in AFCU mode.

LEGEND

AFCU	Alternate Flight Control Unit
GS	Ground Spoiler
MFS	Multifunction Spoiler
PFCC	Primary Flight Control Computer
REU	Remote Electronic Unit

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Figure 93: Spoiler Operation on Landing

DETAILED DESCRIPTION

SPOILER INDICATING SYSTEM

Position information for the multifunction spoilers (MFSs) is provided by the PCU LVDTs, which provide this information to remote electronic units (REUs). The REUs provide this information over time-triggered protocol (TTP) BUSES to inceptor interface modules (IIMs), then to the primary flight control computers (PFCCs). The PFCCs communicate with the data concentrator unit module cabinets (DMCs) over ARINC 429 BUSES.

The DMCs provide spoiler position information for use in the engine indication and crew alerting system (EICAS), flight controls synoptic page and the onboard maintenance system (OMS).

Position information for the ground spoilers is provided by a single proximity sensor for each ground spoiler panel. The proximity sensors report to the landing gear and steering control units (LGSCUs) which forward information to the EICAS, flight controls synoptic page, and OMS through the DMCs.

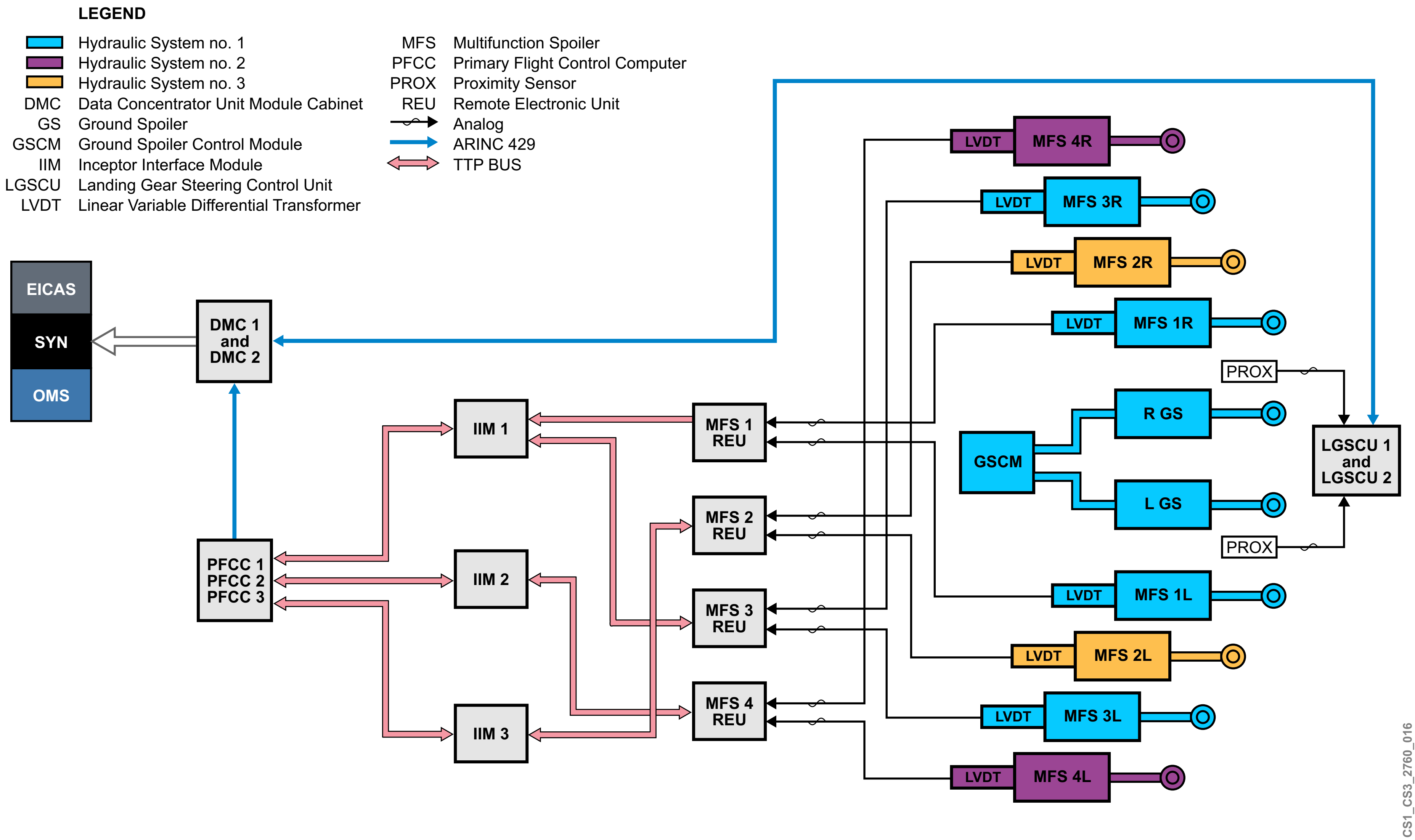


Figure 94: Spoiler Indicating System

MONITORING AND TESTS

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

CAS MESSAGES

Table 21: WARNING Message

MESSAGE	LOGIC
CONFIG SPOILER	Spoiler (MFS or GS) deployed at takeoff.

Table 22: CAUTION Messages

MESSAGE	LOGIC
GND LIFT DUMP FAIL	Auto ground lift dump failed.
GND SPOILER FAIL	Ground spoilers failed retracted.
SPOILER DEGRADED	Two pairs of MFS failed.
SPOILER DPLY	MFS deployed below 300 ft.
SPOILER FAIL	Three or more MFS failed.
SPOILER LEVER FAIL	Spoiler lever manual control failed.

Table 23: ADVISORY Messages

MESSAGE	LOGIC
FLT CTRL FAULT	Loss of redundant or non-critical function for the primary flight control systems. refer to info messages.
SPOILER MISMATCH	Spoiler autoretract function engaged and spoiler lever does not match the actual spoiler position.
SPOILER DPLY	Triggered when spoilers are deployed with either landing gear down or slats/flaps at position 4 or 5.

Table 24: INFO Messages

MESSAGE	LOGIC
27 FLT CTRL FAULT - SPOILER LEVER SNSR REDUND LOSS	Loss of any spoiler lever sensor.
27 FLT CTRL FAULT - SPOILER LEVER SNSR DEGRADED	Loss of any two spoiler lever sensors.
27 FLT CTRL FAULT - SPOILER REU CCDL REDUND LOSS	Loss of CCDL communication between any two spoiler REUs.
27 FLT CTRL FAULT - GND SPOILER INOP	Loss of ground spoilers.
27 FLT CTRL FAULT - GND SPOILER SNSR INOP	Loss of ground spoilers proximity sensors input.
27 FLT CTRL FAULT - MFS 1 (2) (3) (4) INOP	Loss of MFS 1 (2) (3) (4) REU or PCU.

PRACTICAL ASPECTS

MULTIFUNCTION SPOILER POWER CONTROL UNIT MANUAL DUMP VALVE

The MFS PCUs have a spring-loaded manual override cam, which may be used to deplete pressure in the PCU for maintenance purposes. Rotating the external hex head 15° clockwise will allow actuator retract pressure in the upfloat check valve to relieve to the return port of the PCU.

GROUND SPOILER CONTROL MODULE MANUAL RELIEF VALVE

The GSCM has a spring-loaded manual override cam to allow maintenance personnel to release the hydraulic lock on the ground spoiler actuators. This allows the panels to be opened during ground maintenance activities. Rotating the external hex head 15° clockwise will allow retract pressure in the upfloat check valve to relieve to the return port of the GSCM.

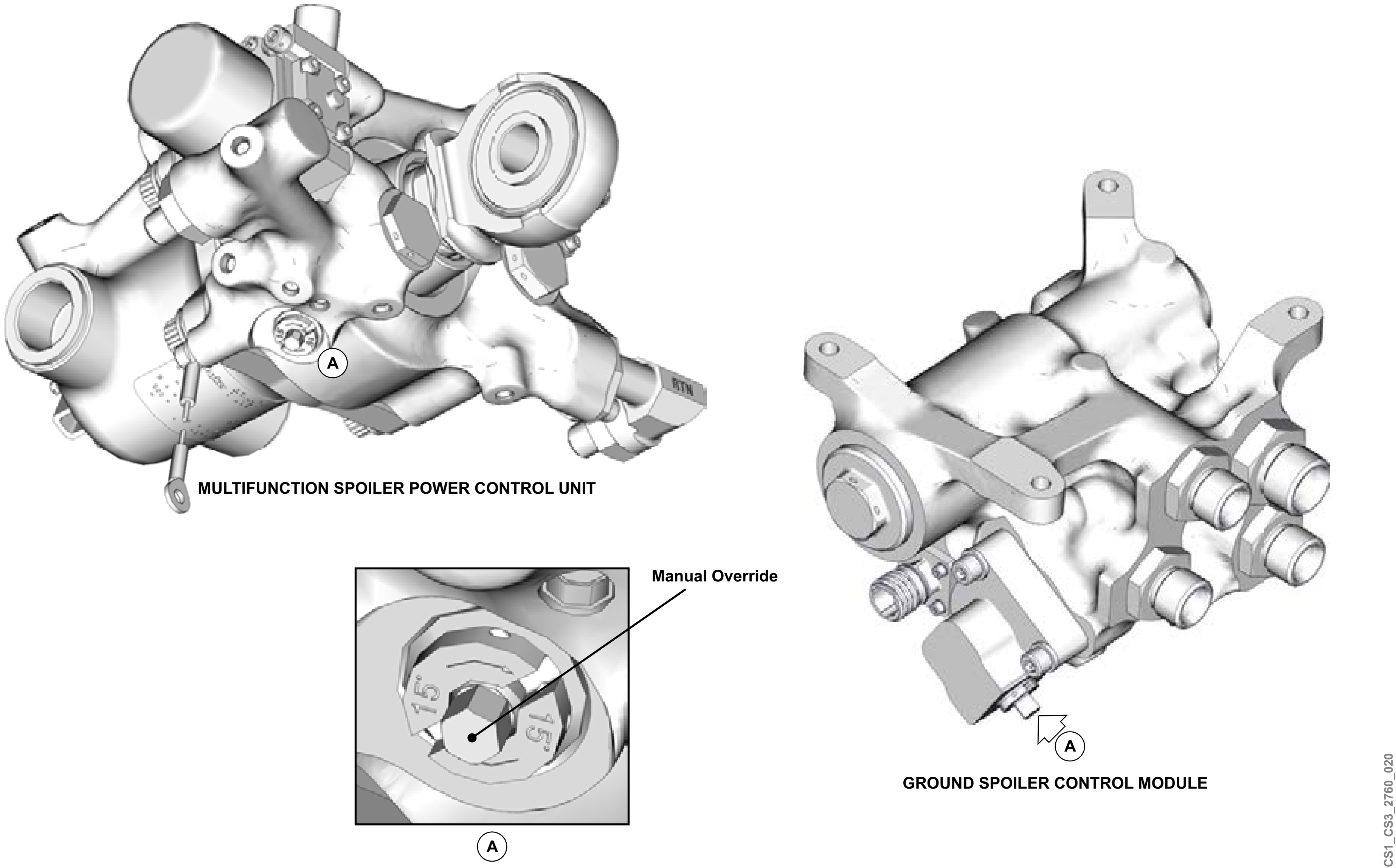


Figure 95: Multifunction Spoiler, Power Control Unit, and Ground Spoiler Control Module Manual Override

27-50/80 HIGH LIFT SYSTEM

GENERAL DESCRIPTION

The high lift system consists of leading edge slats and trailing edge flaps. The slats and flaps surface actuation systems incorporate many common components and are controlled together from the slat/flap control panel (SFCP), which operates through two identical but independent slat/flap electronic control units (SFECUs). The left SFECU is designated as SFECU 1: the right SPECU is designated as SFECU 2.

The SFECUs have two channels each: one for slats, and the other for flaps. The channels work on an active/active basis. For example, the slat channel of SFECU 1 and the slat channel of SFECU 2 both control the operation of the slats. If one SFECU channel fails, the system will operate at half speed.

The high lift system uses hydraulic pressure from all three aircraft systems which provide the force to move the surfaces through the surface actuation systems.

The surface actuation system uses drive shafts and gearboxes to move the surfaces. Each surface is extended and retracted by rotary geared actuators (RGAs).

Upon selection of a slat/flap position, the two SFECUs release the hydraulically-operated slat and flap outboard brakes (OBB) and send drive commands to the slat and flap power drive units (PDUs). These commands cause the PDUs to release their pressure off brakes (POBs) and operate the PDU hydraulic motors to drive the slats and flaps to the selected position.

The slats and flaps move in the following sequence:

- When extending, the slats move first. The flaps do not move until the slats have reached the selected position
- When retracting, the flaps move first. The slats do not move until the flaps have reached the selected position

Located within each PDU is a feedback position sensor (FPS). The FPS provides feedback to the SFECUs on the position of the surfaces. Upon reaching the selected position, the SFECUs stop the supply of hydraulic pressure to the motor and apply the brakes.

Surface movement is monitored by skew sensors. The slat skew sensors are monitored by the LGSCUs, which provide the SFECUs with skew information. The flap skew sensors report directly to the SFECUs. Any skew beyond the allowable limits will shut down the associated system.

Outboard position asymmetry sensors (OPAS), located at the end of each surface actuation system, ensure that the movement detected at the PDU (by the FPS), matches the movement at the OPAS.

The systems continually monitor for left and right asymmetry, drive disconnect, drive system speed (including runaway conditions), surface skew, drive system overload, and surface overspeed (aural warning only). In addition, each of the system components are monitored for failures.

Depending on the nature of the failure the system may continue to operate normally at half speed, or completely shut down. The SFECUs provide information to the engine indication and crew alerting system (EICAS), the flight controls synoptic page, and the onboard maintenance system (OMS).

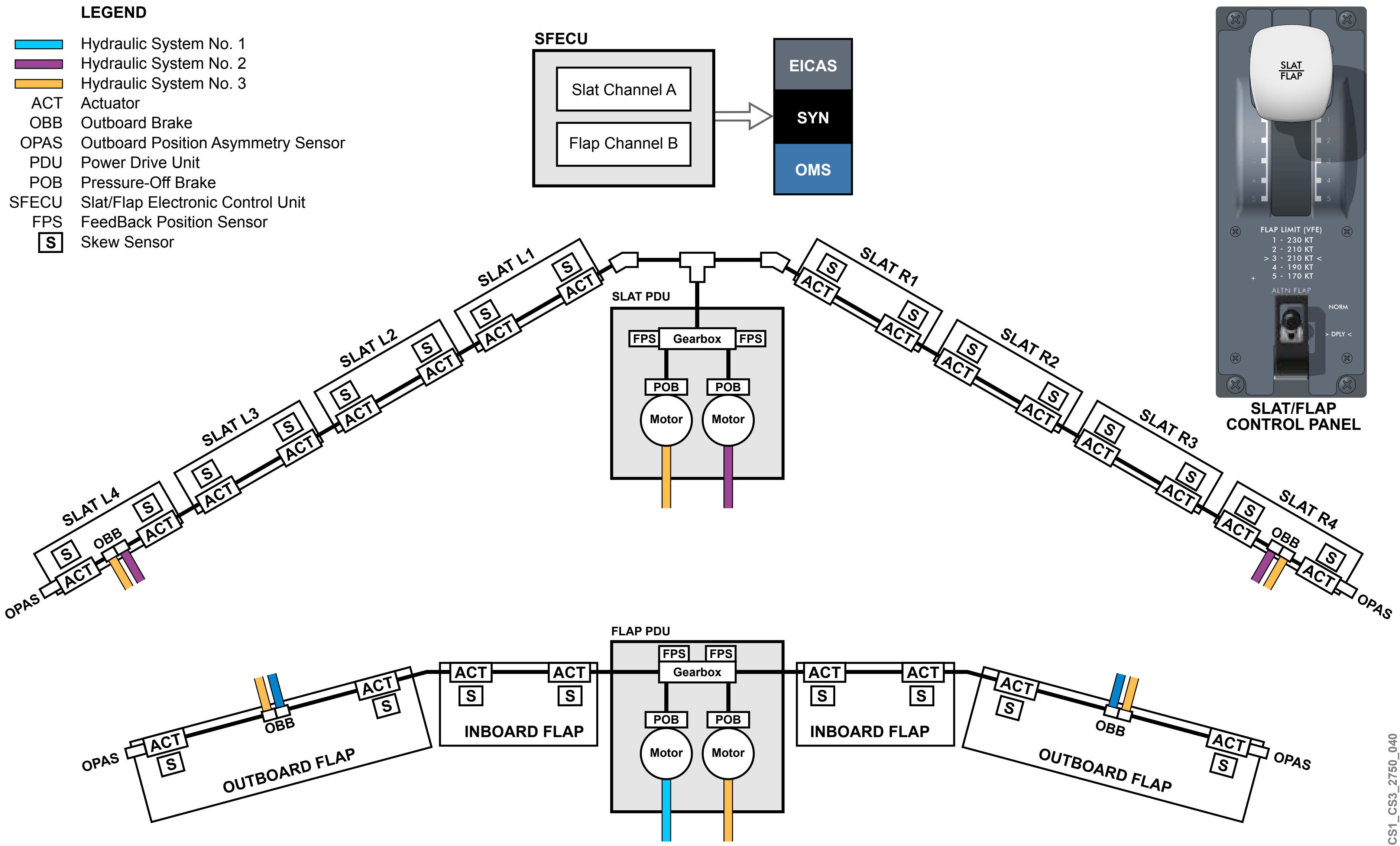


Figure 96: High lift System

COMPONENT LOCATION

The principal control component of the high lift system consists of the following:

- Two slat/flap electronic control units (SFECUs)

SLAT/FLAP ELECTRONIC CONTROL UNITS

The slat/flap electronic control units (SFECUs) are located on the forward and aft shelves of the mid equipment bay.

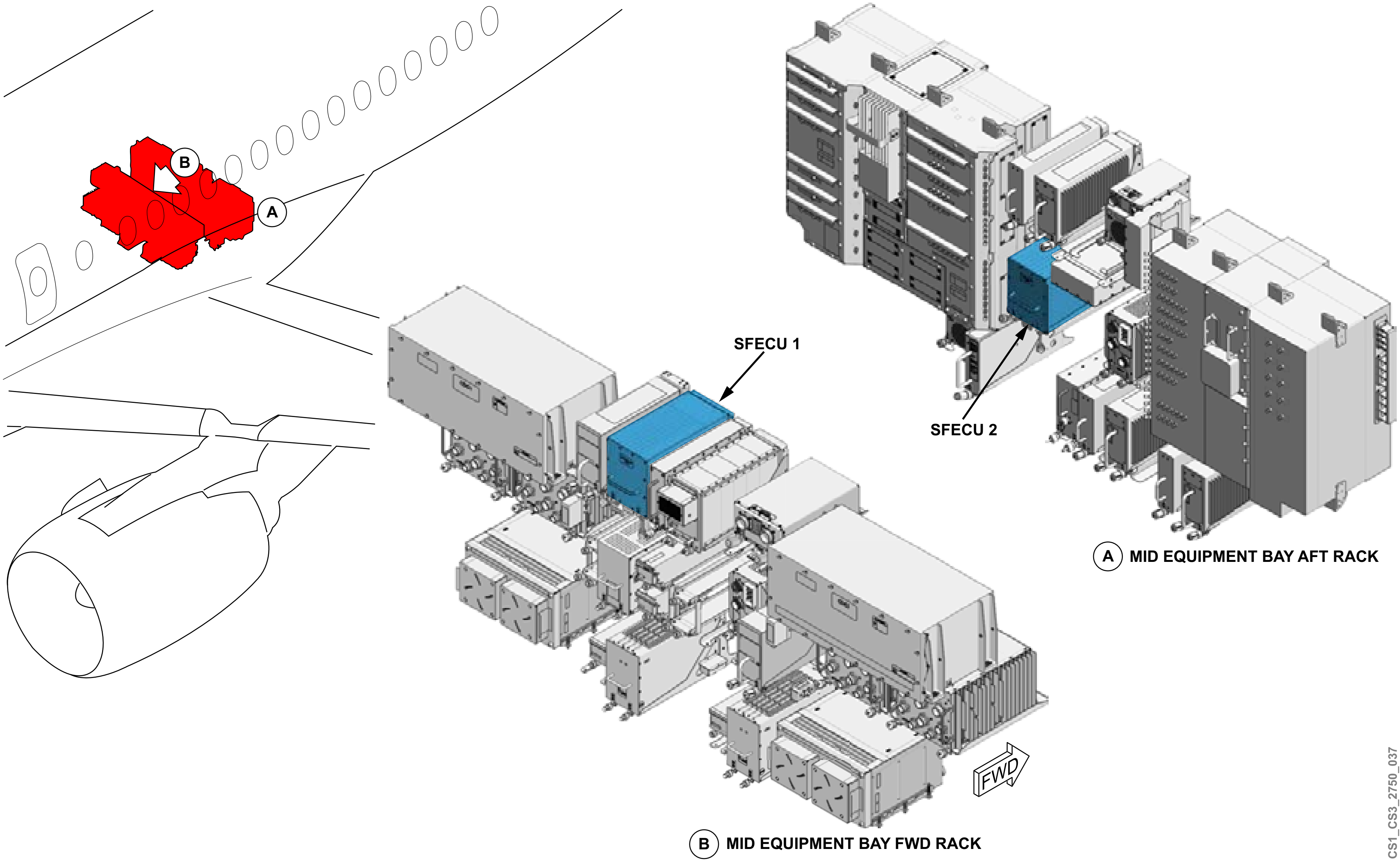


Figure 97: High Lift System Control Component Location

COMPONENT INFORMATION

SLAT/FLAP CONTROL PANEL

The slat/flap control panel (SFCP) consists of a slat/flap control lever, finger lift mechanism, an illuminated scale, guarded alternate flap (ALTN FLAP) switch and internal mechanical and electrical components. The lever is designed to prevent inadvertent selection by incorporating a trigger latch-type lever (finger lift mechanism). The selection of a new slat/flap position is made by a two-part action that involves lifting the spring-loaded finger lift mechanism combined with a fore or aft lever movement.

The SFCP also features a guarded ALTN FLAP switch to provide emergency position selection of the high lift system in case of a mechanical failure of the lever.

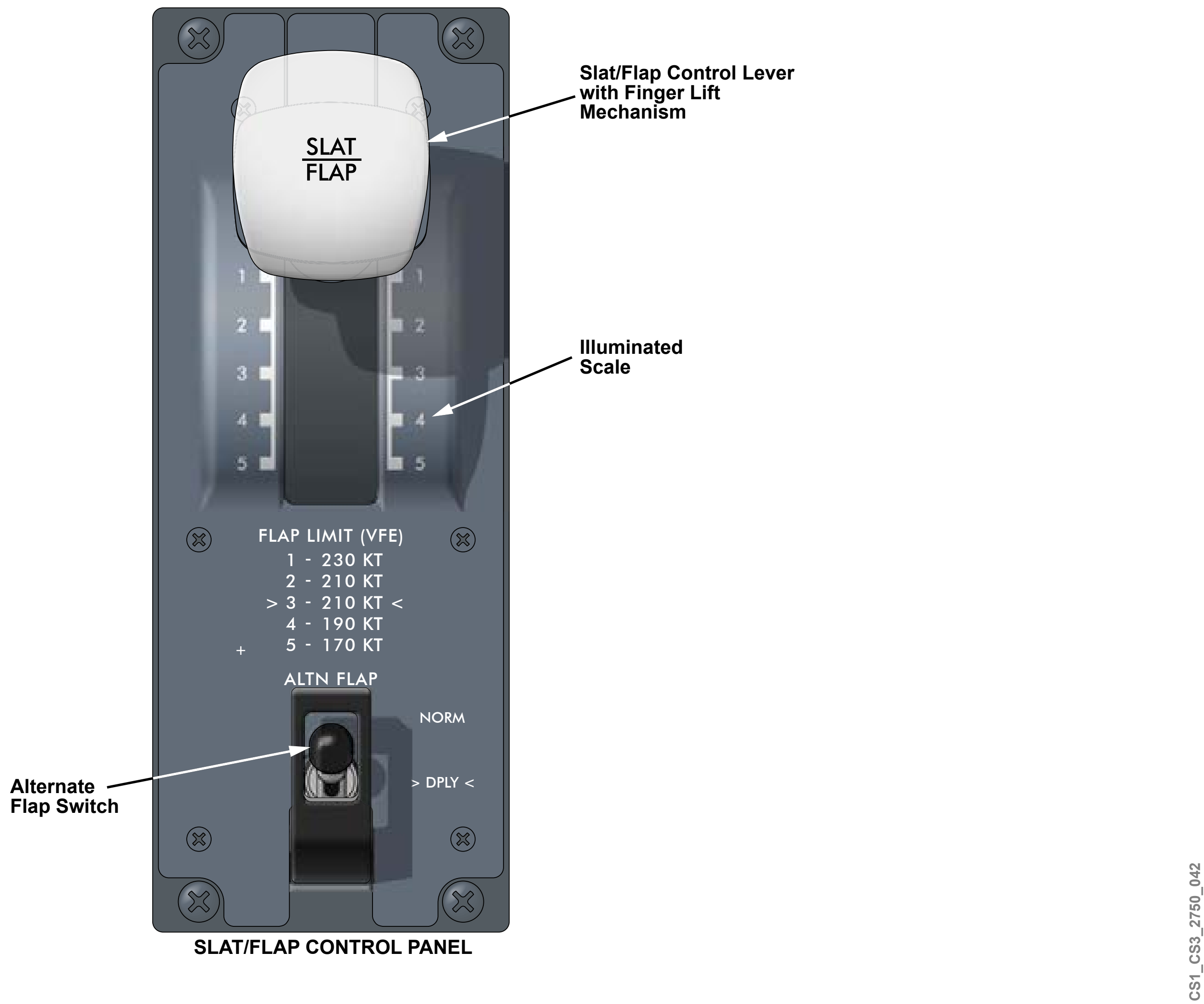


Figure 98: Slat/Flap Control Panel

SLAT/FLAP ELECTRONIC CONTROL UNIT

The two slat/flap electronic control units (SFECUs) are completely independent of each other, with separate power sources, and either unit can operate the slats and the flaps. The SFECUs are divided into two channels.

The SFECUs operate using 28 VDC. The SFECU 1, slat channel 1 (SFECU 1A) is powered from DC ESS BUS 3, while flap channel 1 (SFECU 1B) is powered by DC BUS 1. The right SFECU slat channel 2 (SFECU 2A) is powered by DC BUS 1, while flap channel 2 (SFECU 2B) is powered by DC ESS BUS 3.

When a position is selected from the slat/flap control panel, the SFECUs drive the actuation systems to the selected position.

The SFECUs report slat/flap status and position to the engine indication and crew alerting system (EICAS), flight controls synoptic page and the onboard maintenance system (OMS) through the data concentrator unit module cabinets (DMCs). The DMCs also receive slat skew sensor information through the landing gear and steering control units (LGSCUs). The LGSCUs also provide weight-on-wheels (WOW) information to both SFECUs.

Each channel of the SFECUs provide slat/flap position information to the primary flight control computers (PFCCs) for the operation of the electronic flight control system (EFCS).

Flap skew is monitored directly by the SFECUs.

NOTE

Removal of both slat flap electronic control units (SFECUs) at the same time is not allowed. When a SFECU is replaced, the operational test must only be carried out on the unit replaced. After replacement of a SFECU, some fault messages may be posted. These fault messages should be disregarded and considered normal until the slat flap electronic control unit test and the electronic flight control system test are performed through the onboard maintenance system (OMS). If the messages do not clear, further trouble shooting may be required.

WARNING

MAKE SURE THE AREA AROUND THE FLAPS IS CLEAR WHEN PERFORMING A TEST ON THE SFECU. THE FLAPS MOVE AUTOMATICALLY WITH A MANUAL LEVER INPUT.

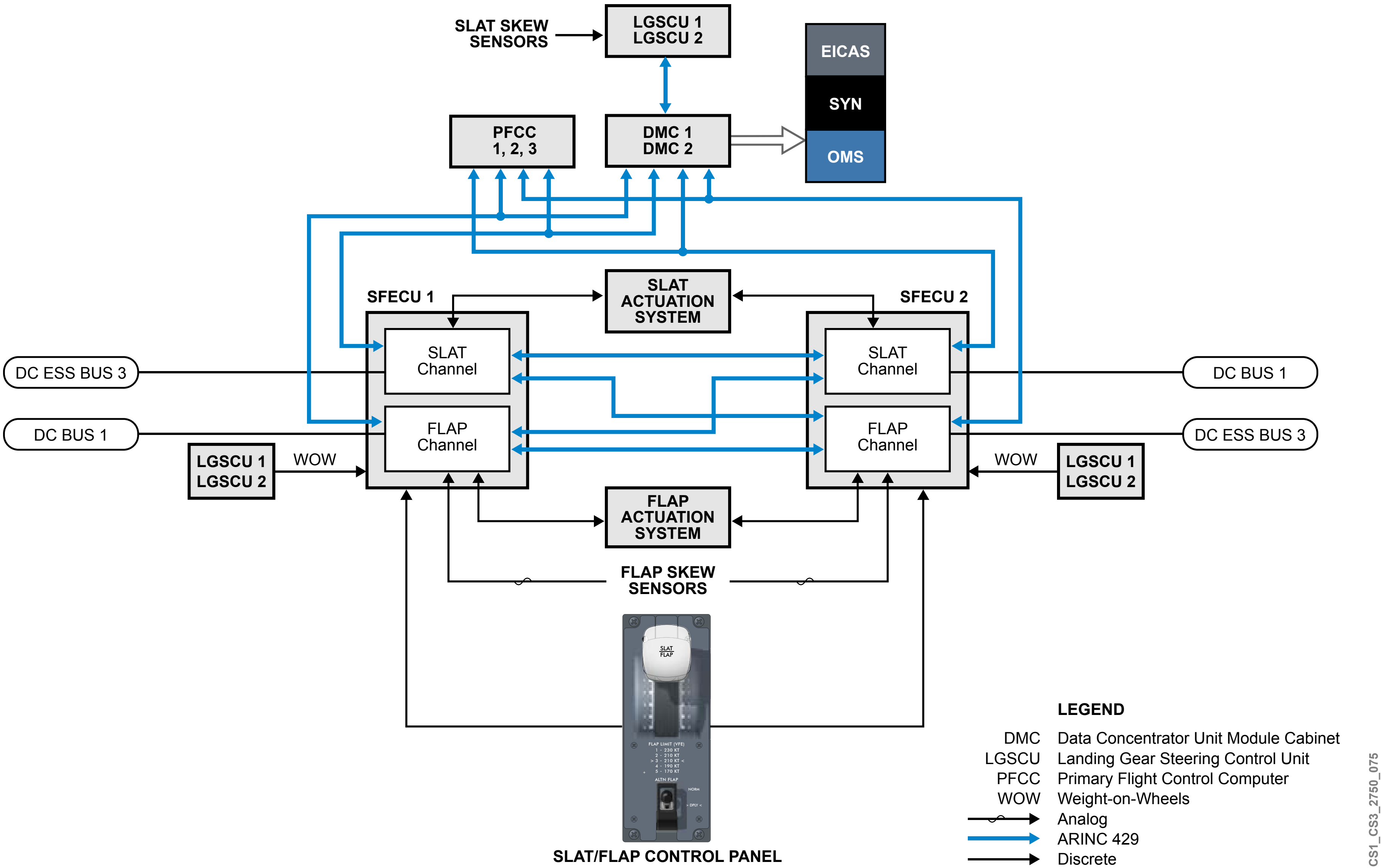


Figure 99: Slat/Flap Electronic Control Unit

DETAILED COMPONENT INFORMATION

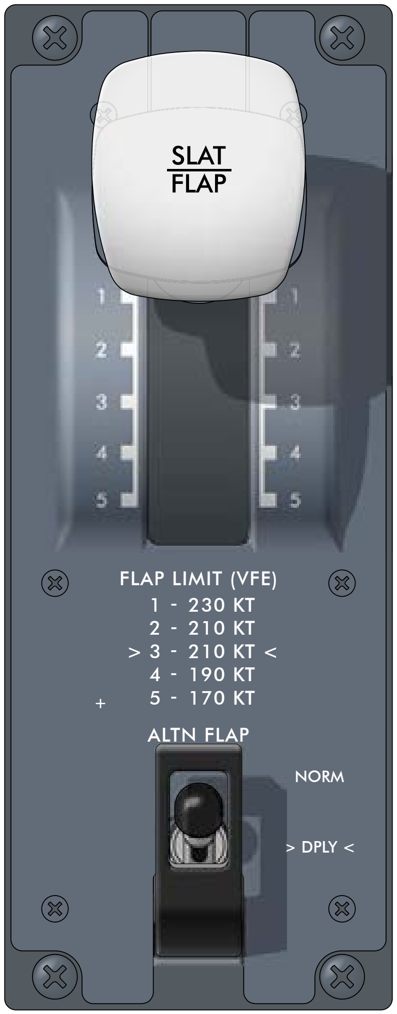
SLAT/FLAP CONTROL PANEL

Each of the slat/flap control lever (SFCL) positions is nominally spaced 10° apart. The lever has a positive detent in each of the gated positions by a spring-loaded gate pin.

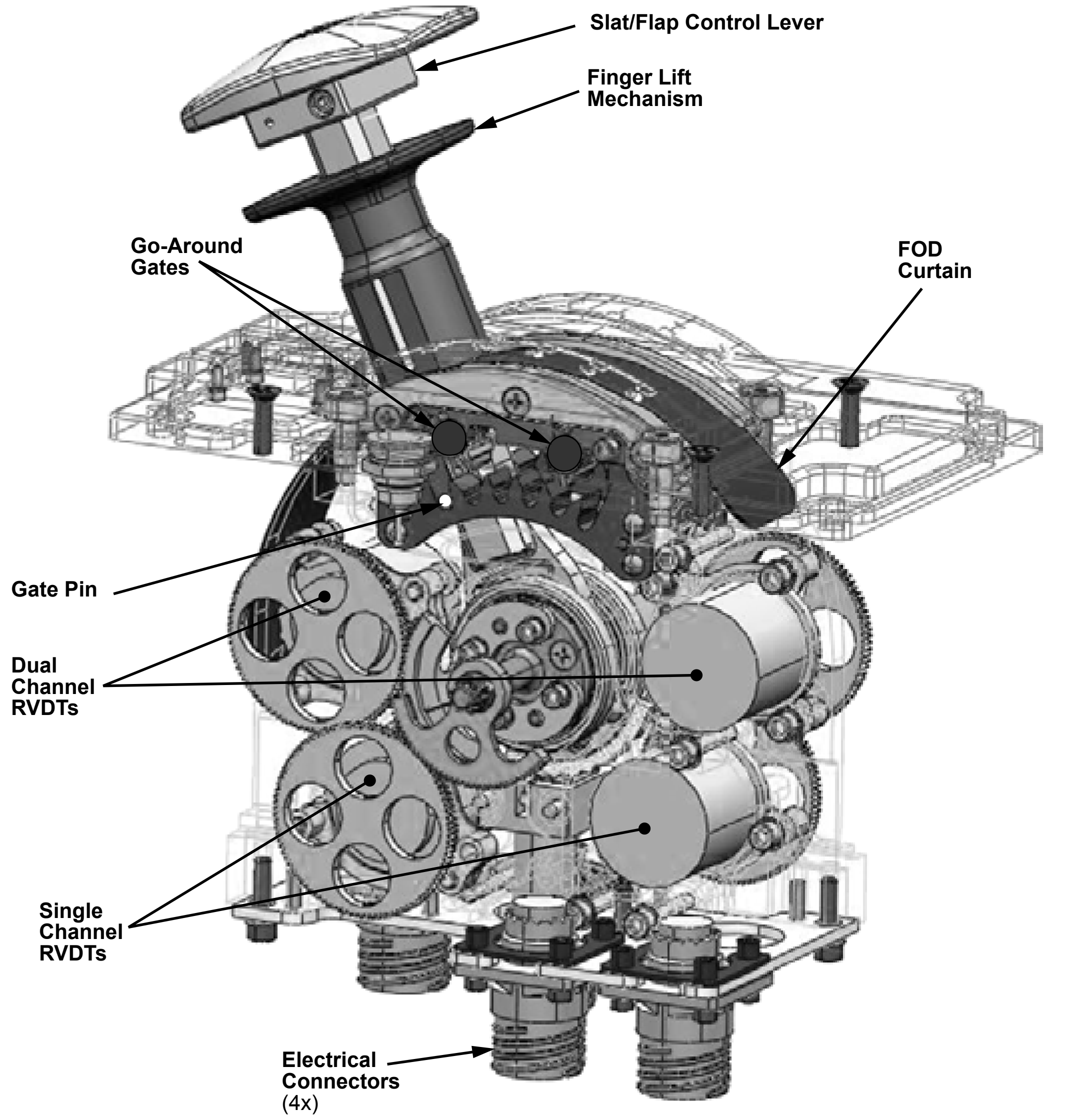
Go-around gates are designed to function only in the forward direction during normal operation. The go-around gates are spring-loaded and obstruct lever movement when going from a higher numbered to a lower numbered gated position. Moving the lever from position 5 will force a stop at position 4. The crew must release the finger lift in that position and press it again if another selection in the retract position is required. The same will happen at position 2.

The SFCL provides position information to the slat/flap electronic control units (SFECUs). The position of the SFCL is provided by four rotary variable differential transformers (RVDTs), two dual-channel, and two single channels. These RVDT signals are read by the SFECU to set the desired flap and slat positions.

A sliding foreign object debris (FOD) curtain prevents debris from entering the internal mechanisms.



SLAT/FLAP CONTROL PANEL



LEGEND

FOD	Foreign Object Debris
RVDT	Rotary Variable Differential Transformer

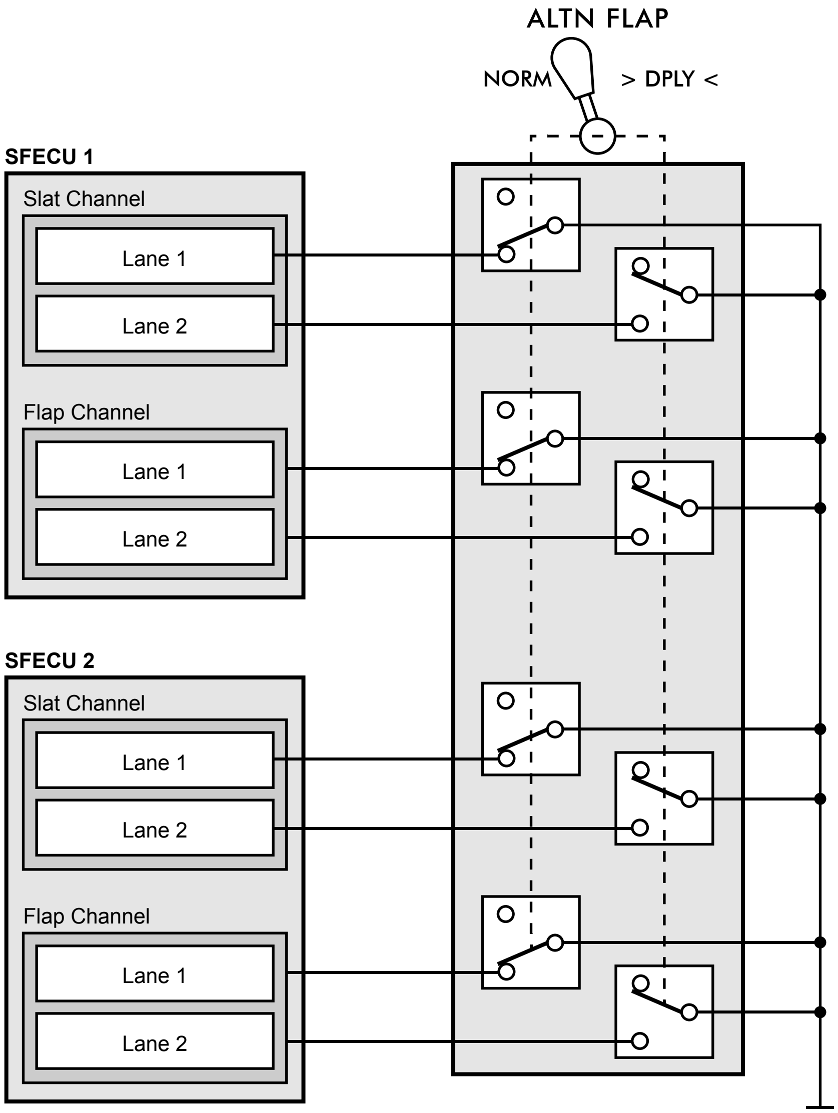
Figure 100: Slat/Flap Control Panel

ALTERNATE FLAP SWITCH

The alternate flap (ALTN FLAP) switch ensures that no single mechanical failure of the lever handle (e.g. detachment or jam) will result in loss of operation. Elements within both the lever and alternate switch are electrically segregated such that a single failure cannot lead to the inability to command the high lift system from both the lever and the alternate flap switch. The switch includes eight independent microswitches to prevent erroneous activation. All microswitches are electrically monitored, which means failing open or closed has no effect on the system due to redundancy. A message will be displayed on EICAS in case of any failure (closed or open).

Each lane of the SFECU channels requires interface to a different microswitch in the ALTN FLAP switch. The alternate switch contains eight double-throw, single-pole switches that are mechanically interlinked. Lane 1 switches are connected normally in opposite states to the switches connected to lane 2, this prevents common mode failures.

When selected to DPLY, the alternate switch will move the slats and flaps to positions S2 and F3 respectively, regardless of the position of the slat/flap control lever (SFCL).



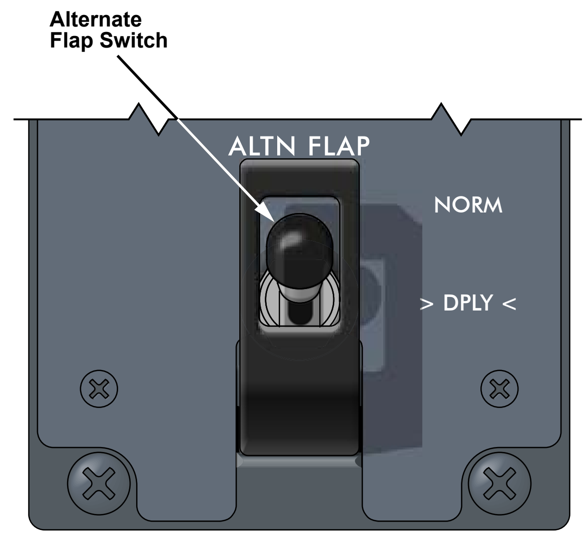


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Figure 101: Alternate Flap Switch

CONTROLS AND INDICATIONS

EICAS PAGE INDICATIONS

The position of the slats and flaps are shown on the engine indication and crew alerting system (EICAS) page while on the ground, during takeoff and landing. This indication is normally removed in flight unless the spoilers are deployed or the landing gear indication is displayed. During any malfunction the indications are also displayed.

The EICAS indications consist of the following:

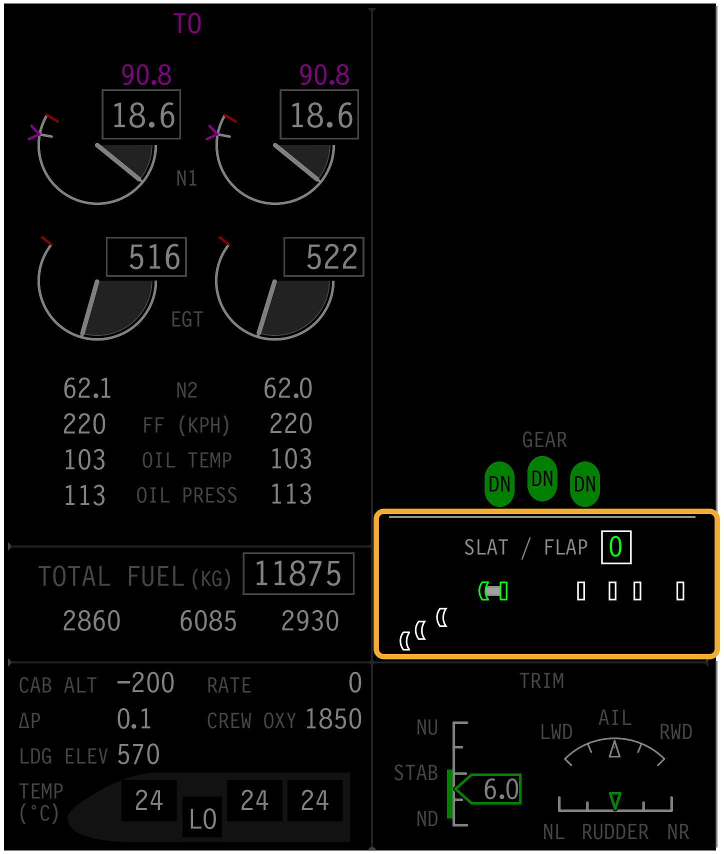
- Commanded slat/flap position
- Achieved and intermediate (in transit) slat surface position
- Achieved and intermediate flap surface position
- Failed position

The slat and flap surface positions are indicated in a continuous fashion allowing smooth progression of the indicator toward the selected position.

The commanded position indicator (cyan) is shown when a new surface position command is received, and is removed when the surface position matches the commanded position.

The color coding applies to the combined slat/flap system and indicates the response of the system as long as it is working normally. As soon as a failure is detected in either the slats or flaps, the indications are separated (one for flaps related to lever position, the other for slats using words instead of numbers). These indications are used to provide awareness to the pilot of the smallest last achieved flap or slat position.

In the event that the position data is lost or invalid, the last known valid position is displayed on EICAS.



EICAS PAGE

SLAT AND FLAP INDICATOR	
Symbol	**Condition**
	Achieved position
	Commanded position
	Normal
	In transit
	Fail
$\boxed{3}$	Slat/flap deployed at command position
$\boxed{0}$	Slat/flap retracted
$\boxed{2}$	Slat/flap in transit, number indicates lowest achieved position (combination of slats + flaps)
3	Flap deployed at command position but slat failed or invalid
FULL	Slat deployed at command position but flap failed or invalid (in, out, mid, full)
2	Flap in transit but slat failed or invalid number indicates lowest achieved position
OUT	Slat in transit but flap failed or invalid state indicates lowest achieved position (in, out, mid)
2	Flap failed deployed (last known position indicated)
0	Flap failed retracted
MID	Slat failed deployed (out, mid, full)
IN	Slat failed retracted

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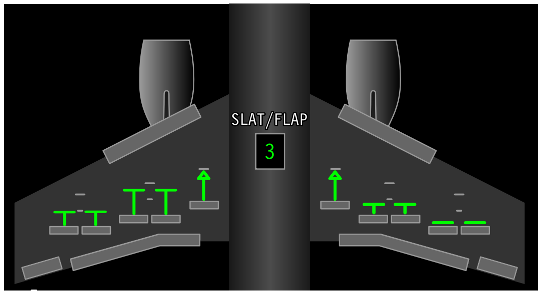
Figure 102: High Lift Indications on EICAS Page

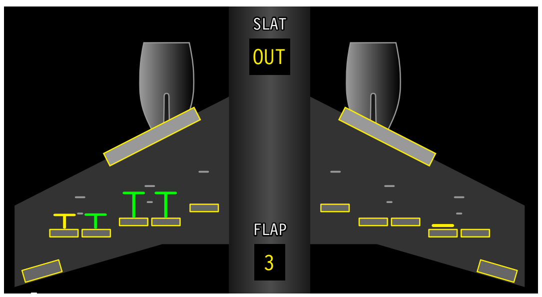
FLIGHT CONTROLS SYNOPTIC PAGE INDICATIONS

The status and position of the slats and flaps are displayed on the flight controls synoptic page.

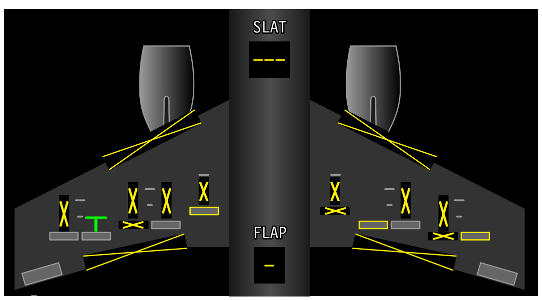
Symbols are provided indicating the surface state (in, out, normal, failed, invalid). Numbers are displayed under normal conditions (slats/flaps normal). In failure cases, numbers are provided for flap position, and text for slat position.



NORMAL OPERATION



FAIL OPERATION



INVALID OPERATION

FLIGHT CONTROLS SYNOPTIC PAGE

FLAP AND SLAT SURFACE STATE	
Symbol	**Condition**
	Normal - In
	Normal - Out

FLAP AND SLAT SURFACE STATE	
Symbol	**Condition**
	Failed - In
	Failed - Out
	Invalid

FLAP ONLY INDICATION	
Symbol	**Condition**
FLAP 3	In commanded position
4	In transition
5	Failed
-	Invalid

SLAT/FLAP INDICATION	
Symbol	**Condition**
SLAT/FLAP 4	In commanded position
4	In transition

SLAT ONLY INDICATION	
Symbol	**Condition**
SLAT IN OUT MID FULL	In commanded position
IN OUT MID	In transition
IN OUT MID FULL	Failed
---	Invalid

CS1_CS3_2750_071

Figure 103: High Lift Indications on Flight Controls Synoptic Page

OPERATION

The slat/flap electronic control units (SFECUs) control and monitor the high lift system. The SFECUs receive slat and flap position inputs from the SLAT/FLAP control lever (SFCL) or the ALTN FLAP switch, and provide drive commands to position the slat and flaps accordingly.

The SFECUs determine the required system response and command for each channel of the power drive unit (PDU) independently. The resulting drive from each PDU is summed at the differential gearbox to provide a single output to the transmission system. The feedback position sensor (FPS) provides positional feedback of the transmission system to the SFECUs.

Electrohydraulic servovalves (EHSV), within the PDU valve blocks, provide directional control, and control the acceleration and deceleration of the system.

When a new slat/flap position is requested, the EHSV is commanded to low speed to ensure the system does not back drive when the brakes are released. When the brakes release, the SFECUs position the EHSV to high speed. The system brakes are applied when approaching the commanded position. One channel stops operating and the speed is reduced in order to slow the slats or flaps to minimize the braking effort.

The outboard position asymmetry sensors (OPASs) provide input to the SFECUs to monitor transmission system failures.

SLAT/FLAP POSITION SELECTION

The system sequencing is as follows:

- For extension, slats move before the flaps
- For retraction, flaps move before the slats
- If the slat and flap surfaces are out of sequence, and the SFCL selection causes one actuation system to extend and the other to retract, the surface closest to the commanded position moves first

The table provides the actual commanded slat position and the referenced commanded flap position for each SFCL position for each flight phase.

Table 25: SLAT/FLAP Control Lever Positions

LEVER POSITION	SLAT POSITION		FLAP POSITION (REF ONLY)	FLIGHT PHASE
	INBOARD	OUTBOARD		
0	0	0	0	Cruise
1	18.1	21	0	Slats only
2	18.1	21	10	Takeoff (best climb)
3	18.1	21	15	Takeoff
4	20.7	24	25	Takeoff (short field) Alternate landing
5	23.2	27	37	Landing

CS1_CS3_TB27_003

NOTE

During maintenance activities, if the slat and flap systems are operated without all three hydraulic systems pressurized or with low hydraulic pressure, the SLAT SLOW or FLAPS SLOW advisory messages could be displayed during the next engine start.

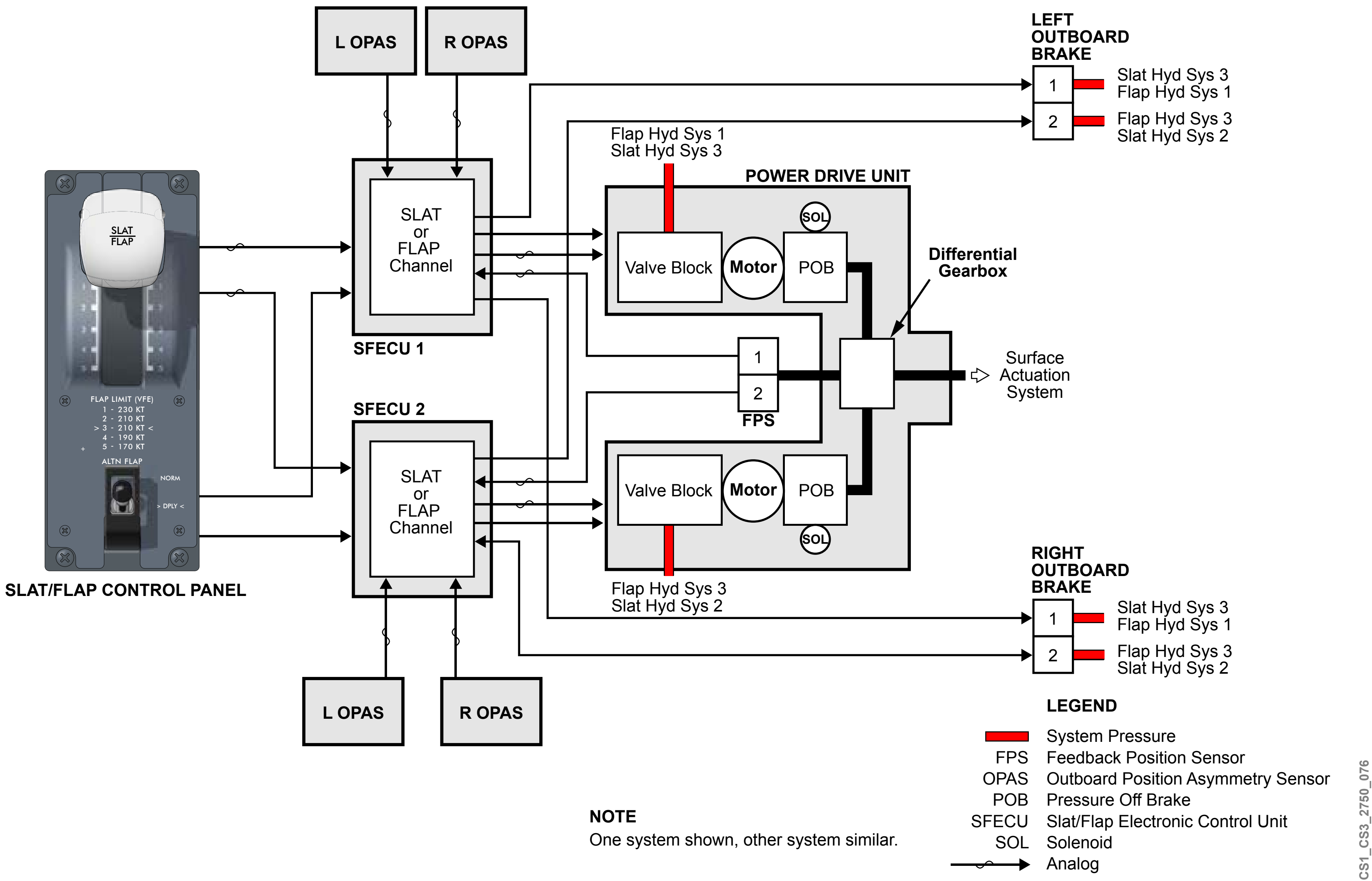


Figure 104: High Lift System Operation

NORMAL OPERATION

During normal operation the system (slat and/or flap) runs dual-channel. There are two speed demands during normal operation: full speed (the EHSVs are fully open and the system runs as fast as possible, limited by flow limit valves in the PDU), and low-speed (the EHSVs are partially closed to allow for slower system movement during brake release and engagement).

SINGLE CHANNEL OPERATION

During single channel operation, the system commands the functioning channel to work as in normal (dual-channel) operation. The affected PDU will then only have a single drive channel functional and the system operates at half of the normal operating speeds. Monitor thresholds are updated to reflect the change in operating speed and time.

RAM AIR TURBINE MODE

During ram air turbine (RAT) mode, only hydraulic system no. 3 and DC ESS BUS 3 are available. Both slat and flap systems revert to single channel operation as power is only available to one channel of each system. Since only one channel is operative, the feedback position sensor (FPS) signal is used to allow a constant speed control.

SINGLE ELECTRICAL MOTOR PUMP (HYDRAULIC SYSTEM NO. 3) OPERATION

In the event of operation with a single electric motor pump in hydraulic system no. 3, there is reduced capability to provide hydraulic power. The SFECUs receive information concerning hydraulic pump status for the hydraulic systems over ARINC 429. The slat system will operate as follows:

Aircraft In-Flight

If both SFECU channels are available, the system operates in normal single channel mode, using the channel that does not derive its hydraulic power from hydraulic system no. 3. The channel on hydraulic system no.3 is forced into the monitor-only state.

This means that although the channel is receiving data from system sensors and providing monitoring outputs, it does not actuate any system components but leaves the system in the passive, braked condition. The other channel continues to operate the system in a single channel mode.

Aircraft On-Ground

The SFECUs ignore the single pump operating information and try to operate as normal.

MAINTENANCE MODE

The maintenance mode state is entered from the normal operation state when a request to enter the state from the onboard maintenance system (OMS) and the safety interlocks are satisfied. The purpose of the interlocks is to ensure that the aircraft is in a safe condition to allow maintenance. All system maintenance (when the system is powered) is carried out in the maintenance mode state. Maintenance mode can only be exited by a system reset, which can be initiated by a request from the OMS.

Reprogram State

The reprogram state is entered if a request to enter the state is received and the interlocks are satisfied, provided the channel was previously in maintenance mode. This is to ensure that the reprogram is entered while the aircraft is in a safe, controlled state. While in reprogram, the channel can have its software reprogrammed. Reprogram is exited once the reprogramming is complete and the channel will enter boot.

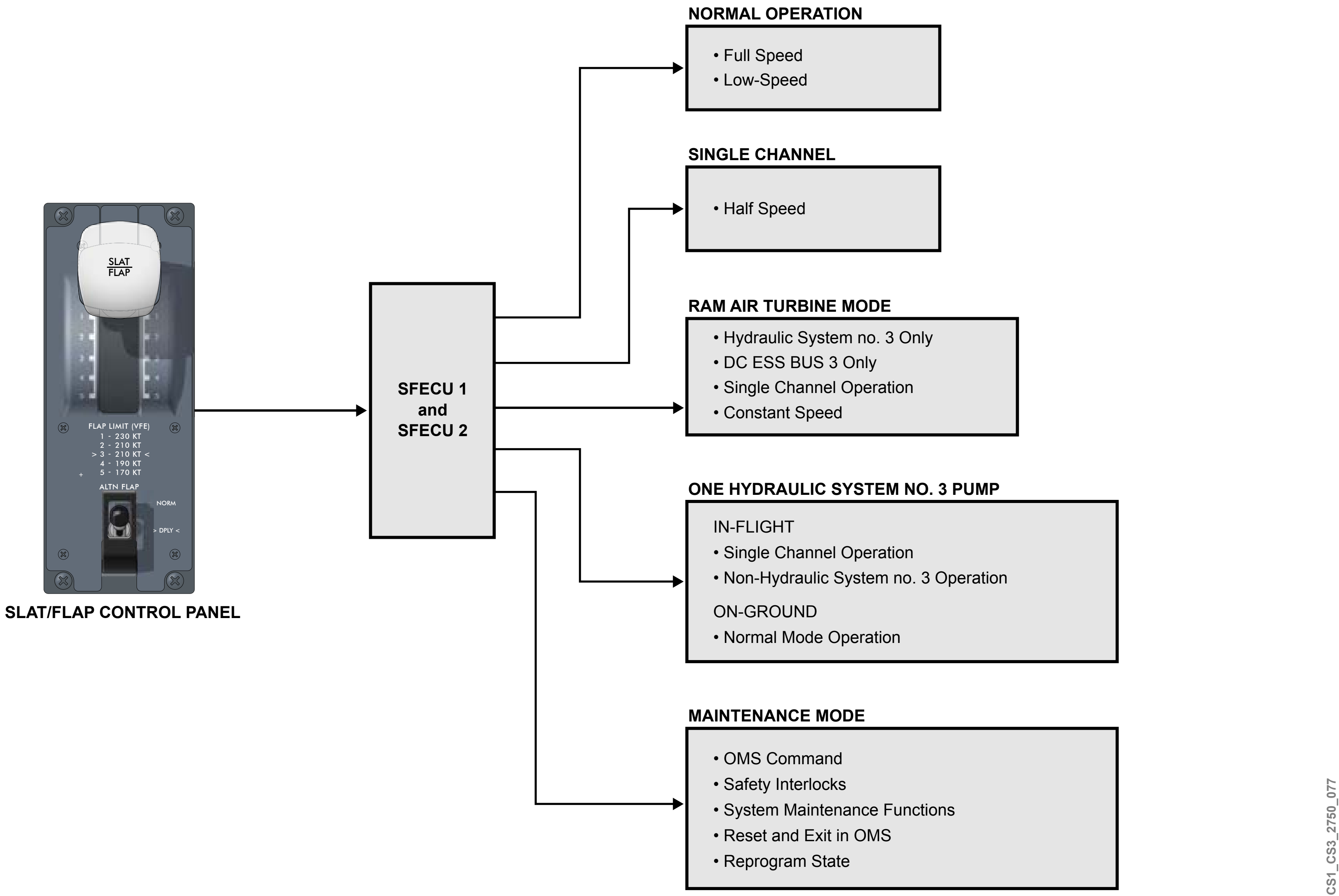


Figure 105: High Lift System Operating Modes

DETAILED DESCRIPTION

SLAT/FLAP ELECTRONIC CONTROL UNIT

SFECU Built-In Test and System Monitors

Power-Up Built-In Test

Power-up built in test (PBIT) includes passive checks of equipment connected to the SFECU to determine system integrity without un-commanded operation of the equipment.

Continuous Built-In Test

Continuous built-in test (CBIT) uses system monitors to detect system component failures. Depending on the type of failure, the system enters the monitor-only state. When in the monitor-only state, the SFECU channel closes the associated enable valve, commands 0 speed EHSV, and applies the system brakes. When a monitor is confirmed, it only affects the channel in which it is confirmed, the other channel operates independently.

System Monitors

The system monitors detect system component failures. When a channel enters the monitor only state, it closes the associated enable valve, commands 0 speed (EHSV), and applies the system brakes. When a monitor is confirmed, it only affects the channel in which it is confirmed; the other channel continues to operate independently. Each channel monitors the following external components:

- Outboard position asymmetry sensors (OPAS)
- Slat skew sensors
- Flap skew sensors
- Power drive unit (PDU)
- Slat/flap control panel
- Outboard brakes

In addition, the SFECU monitors internal operation, including:

- Channel availability
- Speed - measures extension/retraction rates against limits
- Runaway - prevents under or overshooting of commanded position
- Overload - actuators report jams or near jam conditions
- Aircraft overspeed - compares aircraft speed (from air data sensors) to surface position - if excessive aural warning is provided
- Transmitters and receivers - monitors ARINC 429 DATA integrity
- SFECU discrete inputs - correct identification of components

End-of-Flight Built-In Test

End-of-flight (EOF) built-in test (BIT) is performed automatically during single channel operation. In order to check both channels of each system, the channel used to retract the surfaces is alternated every flight. The EOF BIT checks the OBB, PDU brakes, EHSV, speed monitor, and tests the system operation of the RAT mode.

Initiated Built-In Test

Refer to the practical aspects section of this ATA for information on tests that can be initiated from the onboard maintenance system (OMS).

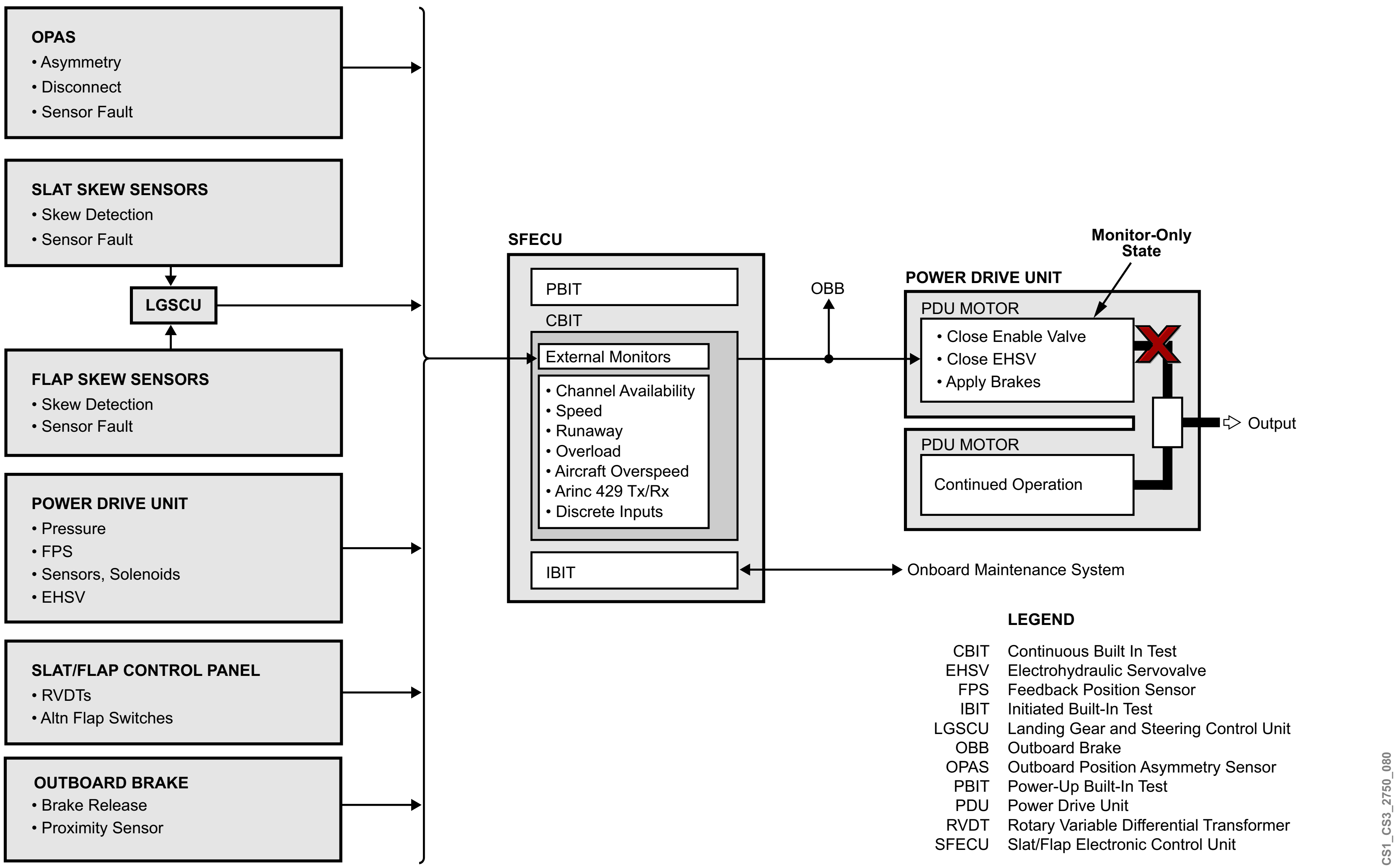


Figure 106: SFECU Monitoring

Digital Interface and Power Supply

In addition to control and monitoring functions, the slat/flap electronic control units (SFECUs) have the following functions:

- Provide status and warning inputs to the avionics system for display in EICAS
- Provide and receive data to and from other aircraft systems
- Provide aids to installation and troubleshooting for maintenance purposes through the onboard maintenance system (OMS)

Each SFECU contains two channels, and each channel contains two lanes and a power supply module.The SFECUs communicate with each other in order to control the sequencing of deployment and retraction. Each SFECU contains an electrically separated dual-channel architecture. Each channel has its own power supply. This dual-channel architecture enables the slat/flap systems to be driven (at half speed) by one SFECU channel, should the other SFECU channel fail or enter the monitor-only state.

Dual-lane architecture is employed within each channel whereby each lane performs system demand calculations independently and each lane must agree before the system responds.

Both SFECUs and their associated channels communicate over ARINC 429 DATA BUSES with the data concentrator unit module cabinets (DMCs) for engine indication and crew alerting system (EICAS) display, the flight controls synoptic page, and OMS. Aircraft speed information is provided from the air data sensing system for monitoring of the slat/flap extension speed. Excessive speed for the commanded or actual position generates an aural warning.

The SFECUs also provide data to the three primary flight control computers (PFCC) over ARINC 429 DATA BUSES. The landing gear and steering control units (LGSCUs) provide weight-on-wheels (WOW) to both channels independently.

Each SFECU receives power from a thermal circuit breaker on one channel (slat or flap) and from a solid-state power controller (SSPC) to the other channel (flap or slat).

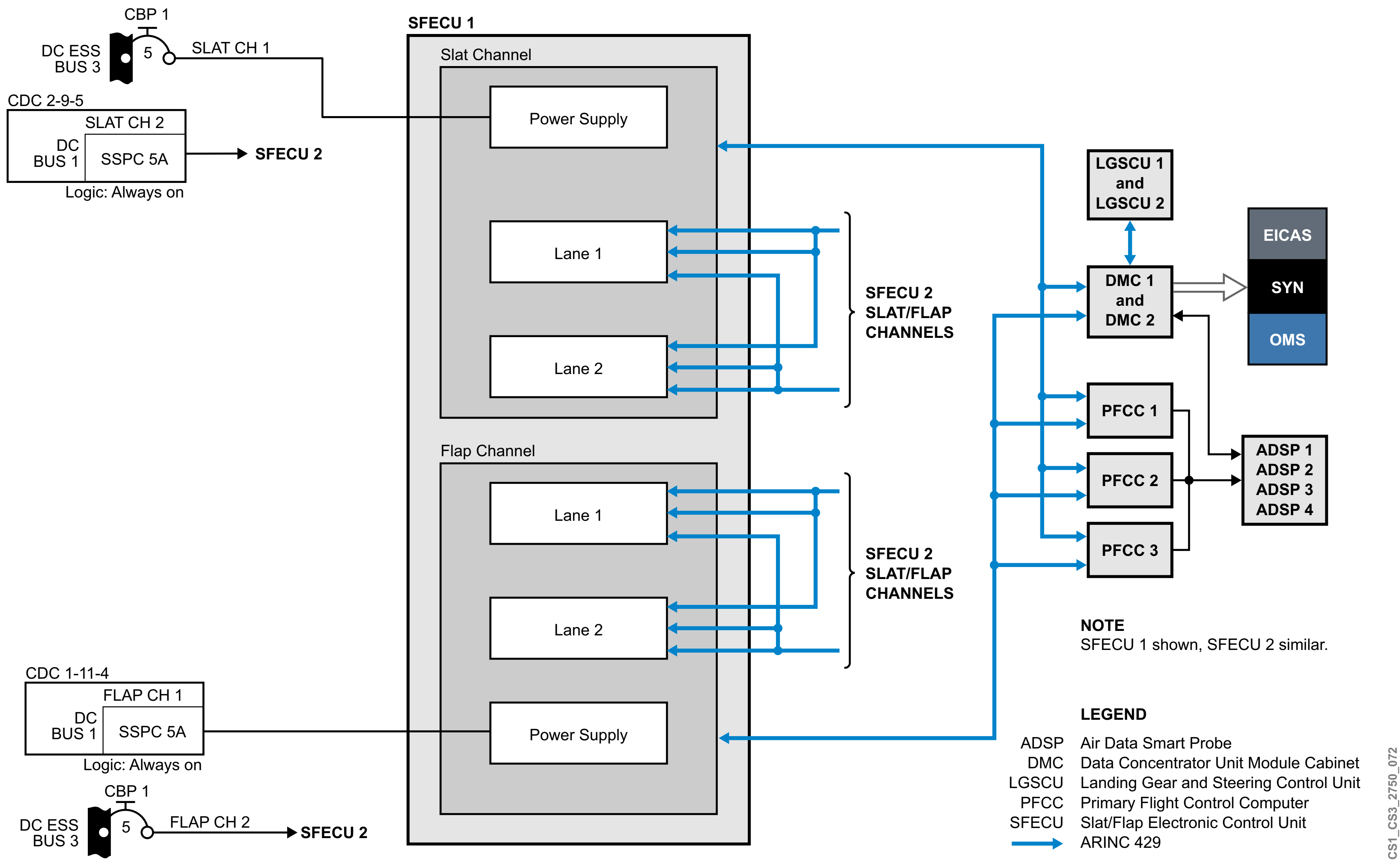


Figure 107: SFECU Digital Interface and Power Supply

Analog/Discrete Interface

Rotary variable differential transformers (RVDTs) within the slat/flap control panel (SFCP) provide analog position signals to their respective lane within the slat or flap channel. The alternate flap (ALTN FLAP) switch provides independent discrete signals to each lane. Each SFECU lane calculates the position requested. If both agree, then both lanes energize the enable solenoid which applies hydraulic pressure to the electrohydraulic servovalve (EHSV). Both lanes also energize the brake solenoid which releases the pressure off brake (POB). The EHSV applies an analog signal to the EHSV to control motor speed by controlling hydraulic pressure to the motor.

The other SFECU performs the same functions simultaneously on the other motor of the PDU. Both motors drive the slat or flap actuation system to position the surfaces to the requested position.

Each SFECU channel receives a signal from the feedback position sensor (FPS) of the PDU, and the outboard position asymmetry sensors (OPASs). The SFECUs use these signals to detect any abnormal condition in the slat/flap systems.

Each SFECU channel controls one of the two solenoids in the outboard brake (OBBs) at each end of the surface actuation system.

The landing gear and steering control units (LGSCUs) provide weight-on-wheels (WOW) to both channels (slat and flap) independently.

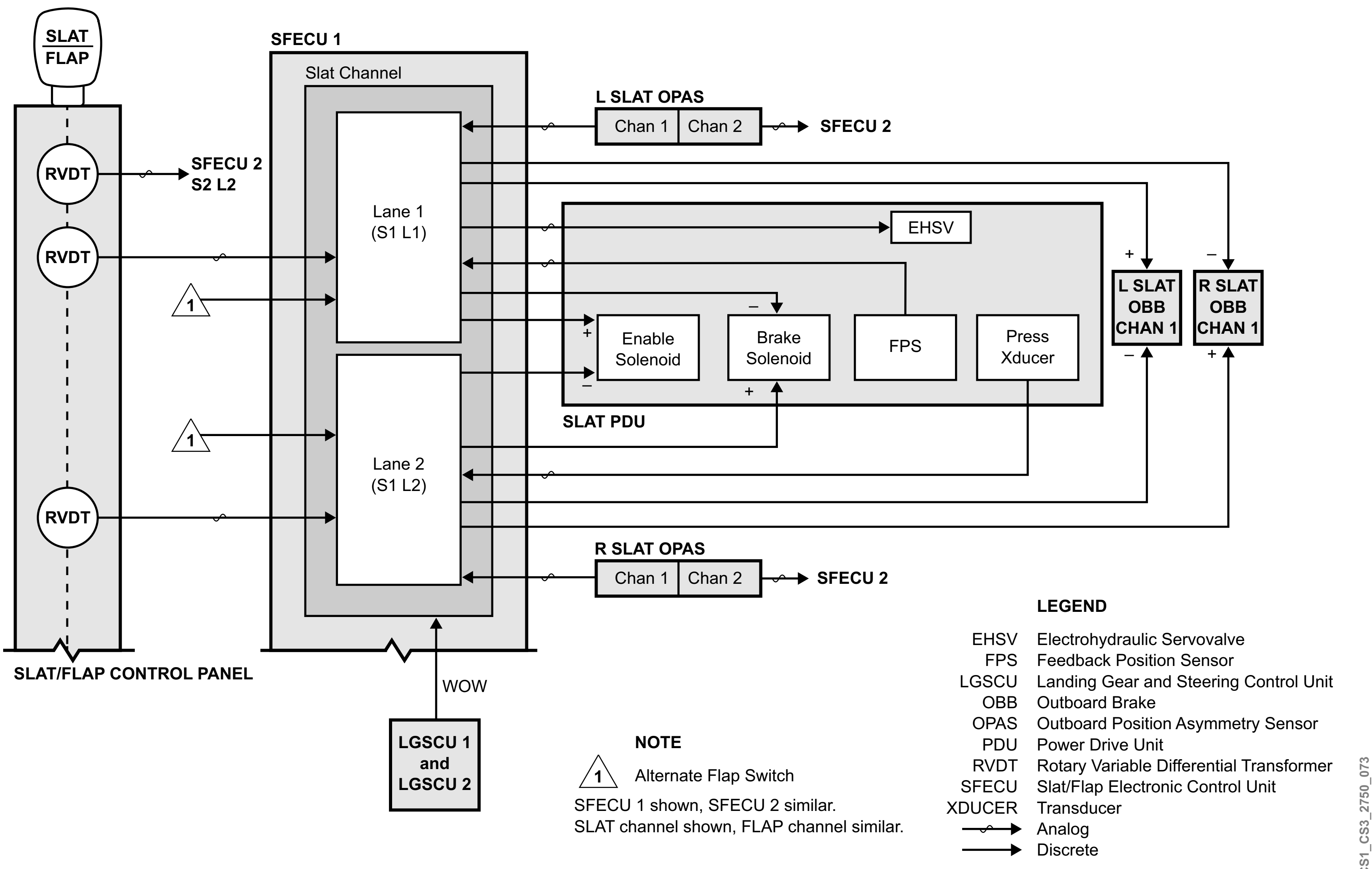


Figure 108: SFECU Analog and Discrete Interfaces

SLAT/FLAP SKEW SENSING

Slat skew sensing is accomplished by two proximity sensors, one located at the inboard and outboard slat tracks of each surface.

The proximity sensors receive excitation from the landing gear and steering control units (LGSCUs). One half of the proximity sensors (A) report to LGSCU 1, the other half (B) reports to LGSCU 2. The LGSCUs report skew data to the slat/flap electronic control units (SFECUs) through ARINC 429 BUSES through the data concentrator unit module cabinets (DMCs).

The flap skew sensors use a combination of rotary variable differential transformers (RVDTs) and linear variable differential transformers (LVDTs). Flap skew sensor 1 is an LVDT. The remainder are RVDTs. The flap system RVTDs and LVDTs report directly to the SFECUs.

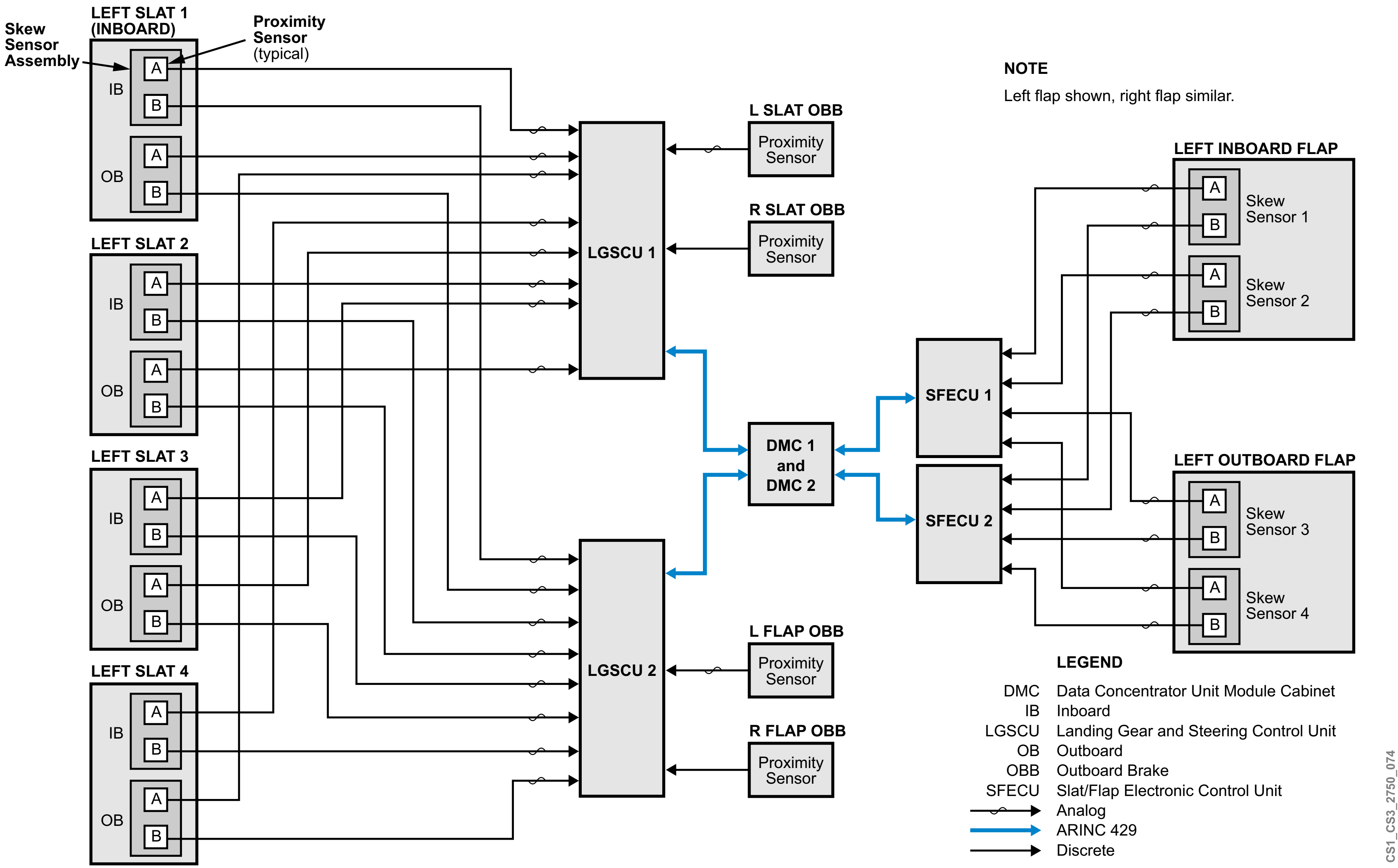


Figure 109: Slat/Flap Skew Sensing and Outboard Brakes

MONITORING AND TESTS

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

CAS MESSAGES

Table 26: WARNING Message

MESSAGE	LOGIC
CONFIG FLAP	Slat/flap not configured for takeoff.

Table 27: CAUTION Messages

MESSAGE	LOGIC
FLAP FAIL	Any failure that results in the flap system being inoperable. Both flap channels unavailable due to monitor failure.
SLAT FAIL	Any failure that results in the slat system being inoperable. Both slat channels unavailable due to monitor failure.
SLAT-FLAP FAIL	Any failure that results in the flap and slat systems being inoperable. All channels unavailable due to monitor failure.
SLAT-FLAP LEVER FAIL	Failure to command flap through the flap lever.
SLAT SKEW	Slat skew condition detected or slat asymmetry.
SLAT SLOW	Slat speed less than 30% of dual-channel nominal speed.
FLAP SLOW	Flap speed less than 30% of dual-channel nominal speed.

Table 28: ADVISORY Messages

MESSAGE	LOGIC
SLAT SLOW	Only one slat channel available to drive system at approximately half-speed (30% - 60%).
FLAP SLOW	Only one flap channel available to drive system at approximately half speed (30% - 60%).
SLAT-FLAP SLOW	One flap and one slat channel available to drive system at slow speed (RAT deployment) (30% - 60%).
SLAT FAULT	Loss of redundancy or loss of non-critical functions within the slat system. Refer to INFO messages.
FLAP FAULT	Loss of redundancy or loss of non-critical functions within the slat system. Refer to INFO messages.

Table 29: STATUS Message

MESSAGE	LOGIC
ALTN FLAP DPLY	Alternate flap setting deploy selected (setting 3).

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Table 30: INFO Messages

MESSAGE	LOGIC
27 SLAT FAULT - DATA CONFIG INPUT REDUND LOSS	Loss of ARINC 429 DATA from DMC or error with the WOW discrete or ARINC 429 transmitter fault. Inhibited if the fault is within the DMC.
27 SLAT FAULT - OUTBD BRAKE INOP	Either slat outboard brake failed in released state.
27 SLAT FAULT - OUTBD BRAKE PROX SNSR INOP	Either slat outboard brake proximity sensor failed.
27 SLAT FAULT - PDU FAULT	Either slat channel detects internal PDU fault.
27 SLAT FAULT - PDU BRAKE DEGRADED	PDU brake degraded if either slat channel detects a failed PDU brake.
27 SLAT FAULT - SKEW SNSR REDUND LOSS	Either slat channel detects a failed skew sensor.
27 SLAT FAULT - SLAT SKEW	Any slat actuator disconnect monitor is confirmed.
27 SLAT SLOW - CHAN 1 INOP	Slat channel 1 inoperative.
27 SLAT SLOW - CHAN 2 INOP	Slat channel 2 inoperative.
27 SLAT SLOW - SLAT SYSTEM INOP	Either slat channel detects that the slat system is operating at slow speed due to mechanical failure that affects both control channels. Message inhibited on the ground and both engines are off.
27 FLAP FAULT - ALTN SWITCH INOP	Two or more slat/flap channels do not have sufficient alternate switch inputs to generate a valid command.
27 FLAP FAULT - ALTN SWITCH REDUND LOSS	A flap channel detects one alternate flap switch contact has failed.

Table 30: INFO Messages

MESSAGE	LOGIC
27 FLAP FAULT - LEVER SNSR REDUND LOSS	Either flap channel detects a failed lever sensor.
27 FLAP FAULT - DATA CONFIG INPUT REDUND LOSS	Loss of ARINC 429 DATA from DMC or error with the WOW discrete or ARINC 429 transmitter fault. Inhibited if the fault is within the DMC.
27 FLAP FAULT - OUTBD BRAKE INOP	Either flap outboard brake failed in released stat.
27 FLAP FAULT - OUTBD BRAKE PROX SNSR INOP	Either flap outboard brake proximity sensor failed.
27 FLAP FAULT - PDU FAULT	Either flap channel detects internal PDU fault.
27 FLAP FAULT - PDU BRAKE DEGRADED	PDU brake degraded if either flap channel detects a failed PDU brake.
27 FLAP FAULT - SKEW SNSR REDUND LOSS	Either channel detects a failed skew sensor.
27 FLAP FAULT - FLAP SKEW	Either flap channel detects a flap skew that is below the flap shutdown threshold.
27 FLAP SLOW - CHAN 1 INOP	Flap channel 1 inoperative.
27 FLAP SLOW - CHAN 2 INOP	Flap channel 2 inoperative.
27 FLAP SLOW - FLAP SYSTEM INOP	Either flap channel detects that the flap system is operating at slow speed due to mechanical failure that affects both control channels. Message inhibited on the ground and both engines are off.
27 FLAP FAULT - PDU BRAKE DEGRADED	PDU brake degraded if either flap channel detects a failed PDU brake.
27 SLAT FAULT - PDU BRAKE DEGRADED	PDU brake degraded if either slat channel detects a failed PDU brake.

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM FUNCTIONS

The slat/flap electronic control units (SFECUs) monitor all of the system sensors and log the data when unexpected results occur. This information is used to isolate failures to line replaceable units (LRUs) where possible and is available to the maintenance personnel. The actuation system incorporates mechanical features to allow the system to be driven manually without aircraft power.

The onboard maintenance system (OMS) provides the following testing functions:

- Slat rigging test
- Flap rigging test
- High lift selector lever (slat/flap control lever) position simulation
- Alternate flaps switch return to service (RTS) test
- Slat/flap electronic control unit (SFECU) RTS test
- Slat dynamic brake performance test
- Slat feedback position sensor (FPS) RTS test
- Slat outboard brake (OBB) RTS test
- Slat outboard position asymmetry sensor (OPAS) test
- Slat power drive unit (PDU) brake RTS test
- Slat static brake performance test
- Slat system RTS test
- Flap dynamic brake performance test
- Flap feedback position sensor (FPS) RTS test
- Flap outboard brake RTS test
- Flap outboard position asymmetry sensor (OPAS) test
- Flap power drive unit (PDU) brake RTS test
- Flap static brake performance test
- Flap skew RTS test

Autorigging

The slats and flaps are autorigged, which means that no specific rigging procedure is required after the system is disturbed for maintenance.

Whenever any system sensor has been replaced, the rigging test through the OMS must be performed. The SFECUs store information on the actual position of the surfaces from the FPS. This information is used to calibrate the system.

The tests for the slats and flaps are performed separately, and for each channel. Tests are done with the surfaces in the fully retracted position.

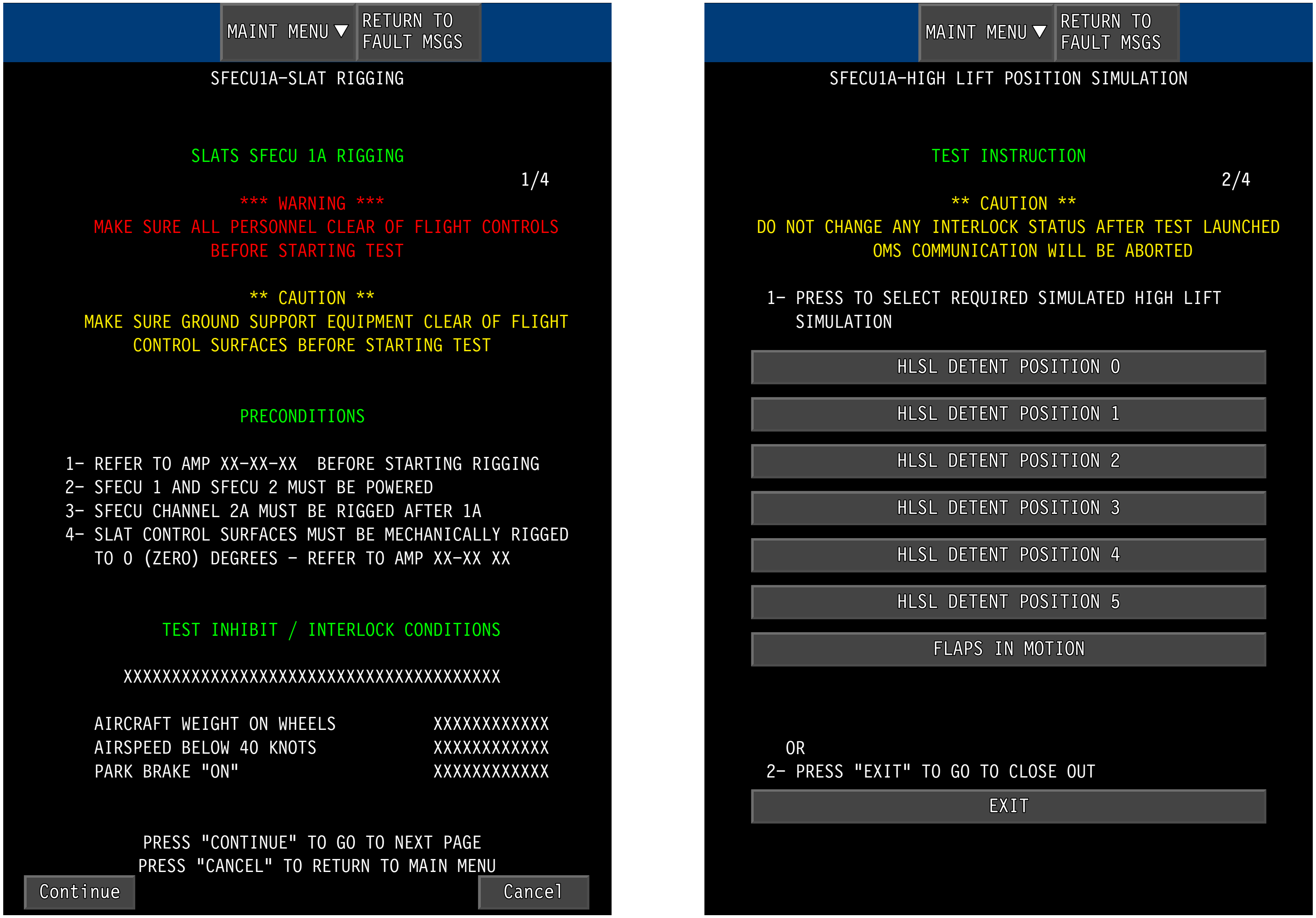

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Figure 110: Onboard Maintenance System High Lift System Functions

ONBOARD MAINTENANCE SYSTEM DATA

In addition to the testing functions, the onboard maintenance system (OMS) provides the following high lift system data and reset functions:

- SFECU hydraulic data (per SFECU channel)
- System component data (surface angles, status, etc.)
- Flap skew position data
- Non-volatile memory (NVM) fault log reset

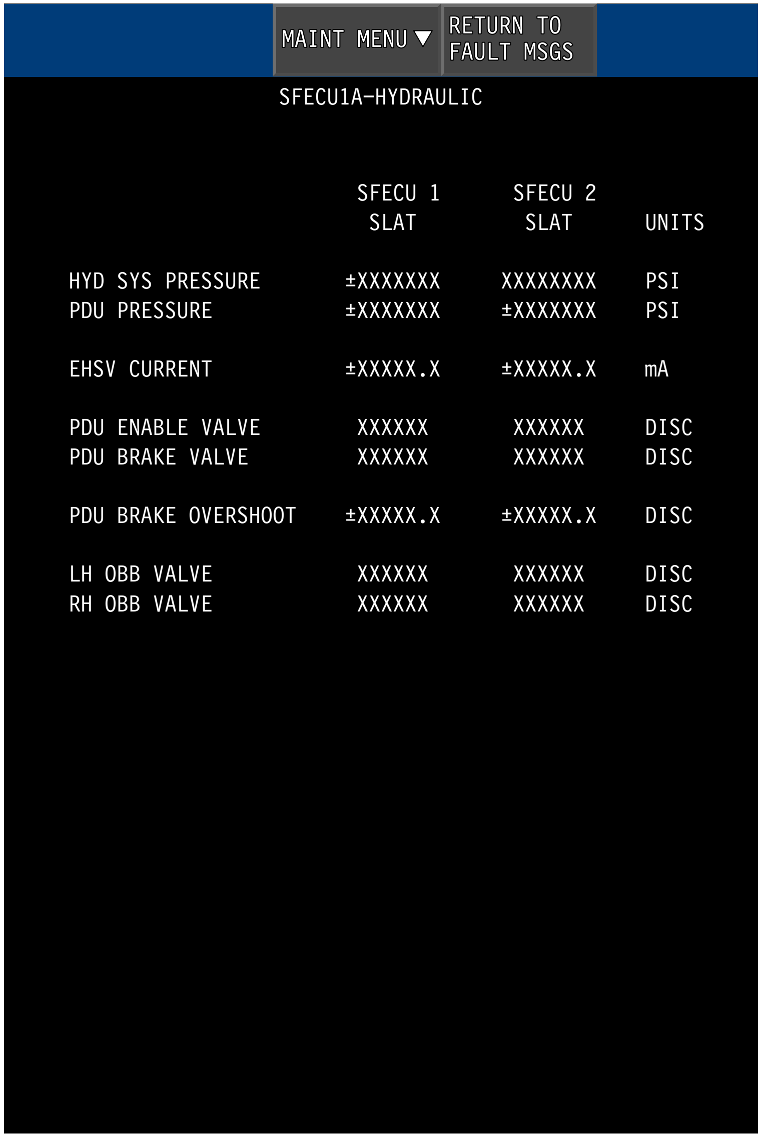

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Figure 111: Onboard Maintenance System High Lift System Data

27-80 SLAT ACTUATION SYSTEM

GENERAL DESCRIPTION

The mechanical slat actuation system begins at the power drive unit (PDU). Torque is transmitted to the slat T gearbox by a torque shaft. The T gearbox delivers the torque to the two angle gearboxes, which transfer the torque through a series of torque shafts, universal joints, steady bearings, and torque discs to the slat actuators, the outboard brakes (OBBs), and the outboard position asymmetry sensors (OPAS).

COMPONENT LOCATION

The slat actuation system consists of the following principal components:

- Power drive unit (PDU) and ecology bottle
- Slat T gearbox
- Angle gearboxes
- Torque shafts and universal joints
- Torque discs (also known as torqdiscs)
- Steady bearings
- Slat actuators (rotary geared actuators (RGAs))
- Outboard brakes (OBBs)
- Skew sensor assemblies
- Outboard position asymmetry sensors (OPASs)

POWER DRIVE UNIT AND ECOLOGY BOTTLE

The PDU is located in the belly fairing forward of the RH main landing gear wheel well. An ecology bottle is located adjacent to the PDU.

SLAT T ANGLE GEARBOX

The slat T angle gearbox is located forward of the PDU, in the wing-to-body (WOW) fairing.

ANGLE GEARBOXES

The two angle gearboxes are located in the forward wing root areas.

TORQUE SHAFTS AND UNIVERSAL JOINTS

There are 36 torque shafts and 21 universal joints located along the left and right wing front spars.

TORQUE DISCS

There are 11 torque discs located throughout the transmission system.

STEADY BEARINGS

There are 17 steady bearings at various locations along the left and right wing front spars, with one located between the PDU and the T gearbox.

SLAT ACTUATORS

The 16 slat actuators (RGAs) are located along the left and right wing forward spars.

OUTBOARD BRAKES

An outboard brake (OBB) is located on both outer wings.

SKEW SENSOR ASSEMBLIES

The 16 skew sensor assemblies are located on the wing leading edge slat track ribs.

OUTBOARD POSITION ASYMMETRY SENSORS

An OPAS is located at the outboard ends of the actuation system.

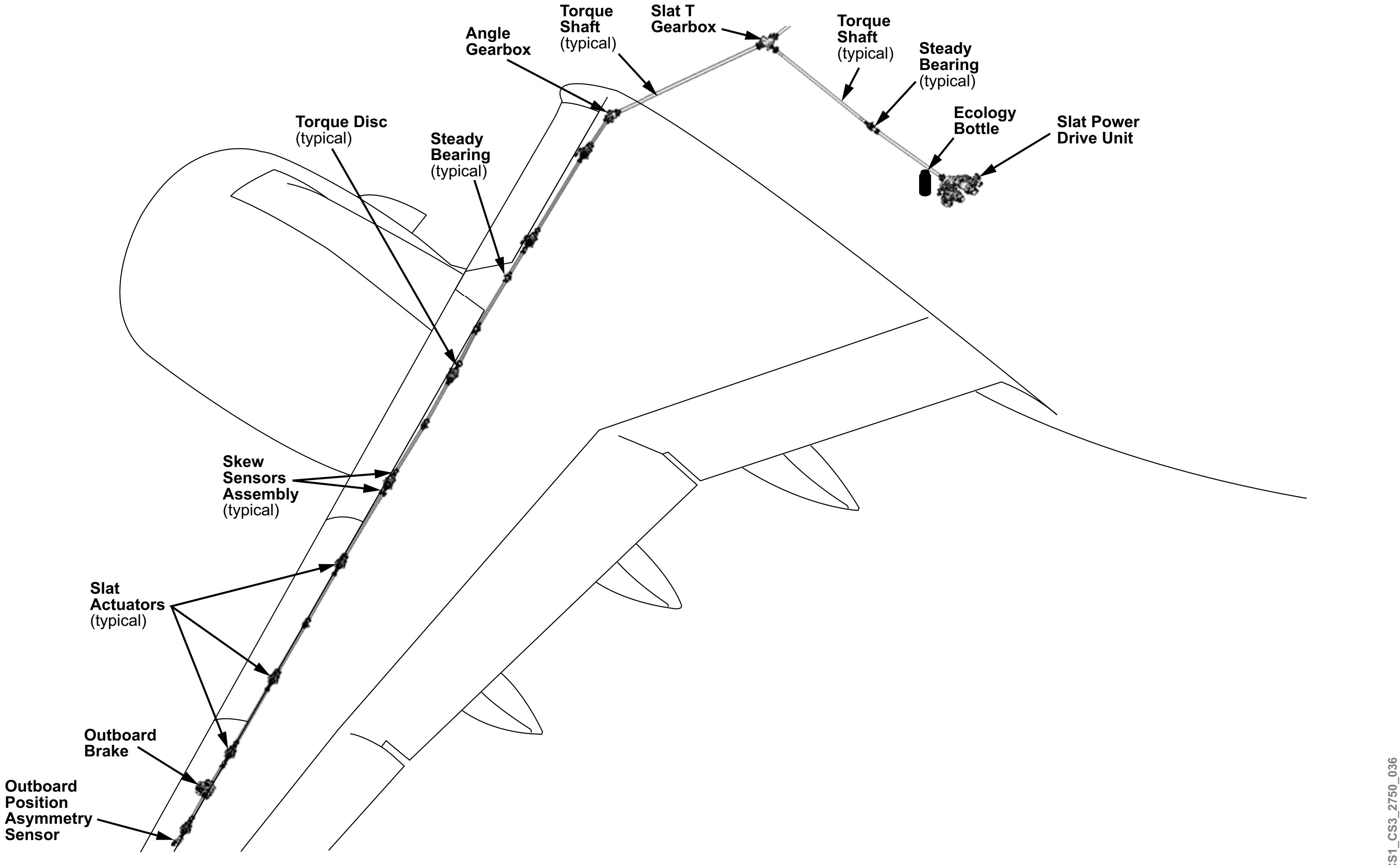


Figure 112: Slat Actuation System Components

COMPONENT INFORMATION

SLAT POWER DRIVE UNIT

The slat power drive unit (PDU) provides power to the slat transmission and actuators. The PDU is a multispeed, bidirectional unit mounted in the belly fairing. The PDU consists of two independent valve blocks coupled to a differential gearbox. The valve blocks receive hydraulic system no. 2 and hydraulic system no. 3 pressure. The differential gearbox combines the torque from the two hydraulic motors and outputs it at a speed equal to the mean speed of the two motors.

Each hydraulic motor supplies power to the output shaft through reduction gearing and a differential gear assembly.

Each PDU block has a pressure off brake (POB) downstream of the motor assembly. If the SFECU senses a failure in either valve block, the POB in the affected valve block will be depressurized and lock the affected motor. This provides an earth for the differential gear to operate at half speed.

The PDU also has a feedback position sensor (FPS) which gives continuous positional feedback to the SFECU.

A fill plug is provided for servicing.

The following items on the PDU are line replaceable units:

- Pressure sensors
- Enable solenoid valves
- Electrohydraulic servovalves (EHSVs)
- Feedback position sensor (FPS)
- Brake solenoid valves

ECOLOGY BOTTLE

An ecology bottle is mounted adjacent to the PDU to collect any fluid coming from the PDU hydraulic motors. Any fluid collected in the ecology bottle may be an indicator of hydraulic motor degradation.

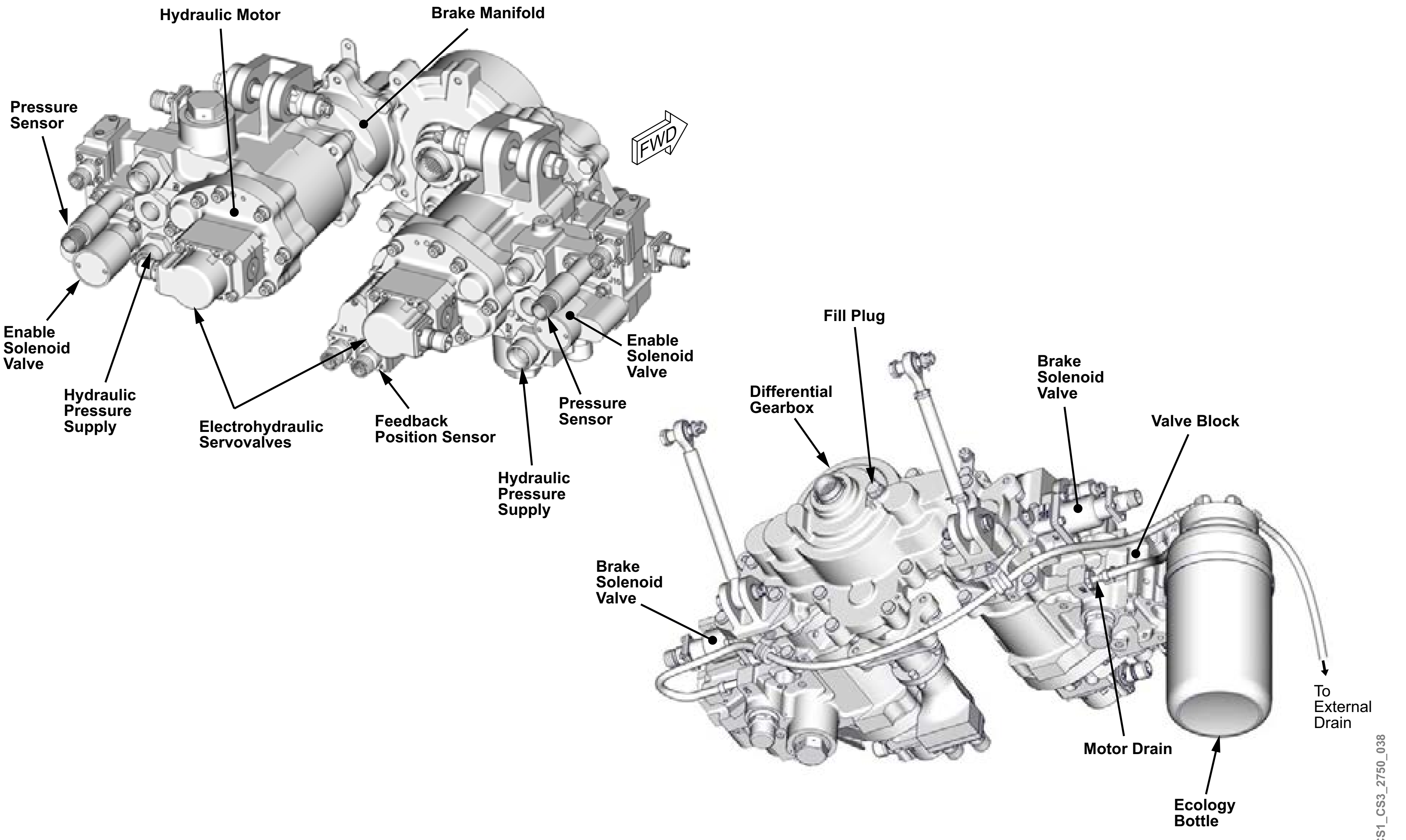


Figure 113: Slat Power Drive Unit and Ecology Bottle

SLAT T GEARBOX

Torque from the slat PDU is transmitted to a single T gearbox through a torque shaft. The T gearbox redirects input to 90° dual-outputs to drive each wing slat circuit. Torque shafts direct the output to two angle gearboxes. The T gearbox has a lubrication port for servicing.

ANGLE GEARBOXES

Two angle gearboxes transmit slat transmission torque through a 25.5° change in direction, to the left and right slat systems through torque shafts. The angle gearboxes have a lubrication port for servicing.

NOTE

Both types of gearboxes are currently lubricated with semifluid lubricant. If the alternative grease filled gearboxes are adopted, servicing will not be required.

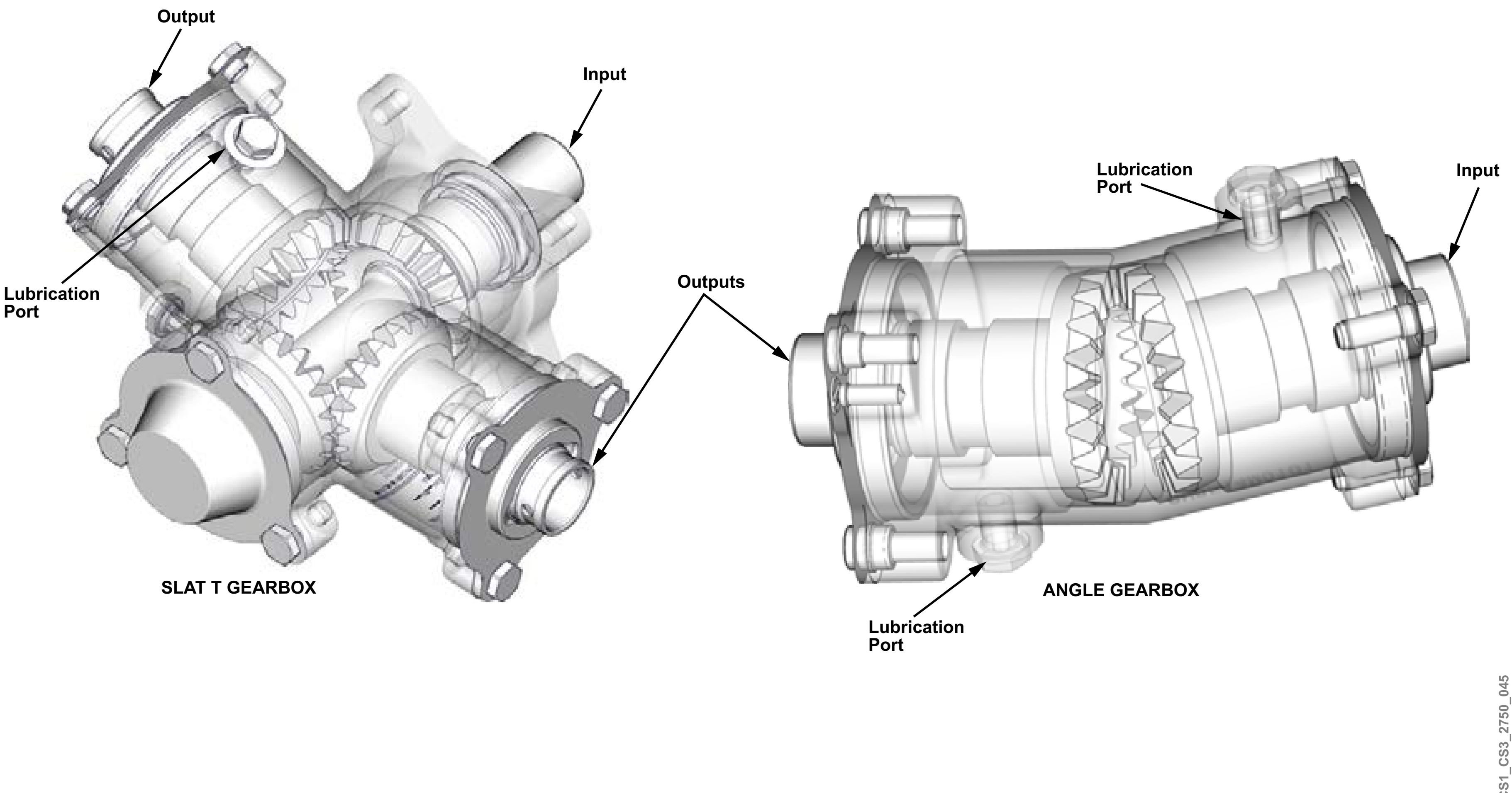


Figure 114: Slat T Gearbox and Angle Gearbox

TORQUE SHAFTS

The actuation system uses 32 torque shafts to transmit torque to the slat actuators. The torque shafts vary in length depending on location. The shafts are constructed from carbon fiber reinforced polymer (CFRP) for strength, corrosion resistance and weight savings. Some of the torque shafts have universal joints installed to add up to 5° of flexibility, and to cope with wing bending. Splined shafts are connected through a series of pinned and unpinned joints. The unpinned splines have sliding splined collars to further allow for wing bending.The pins locate the shaft at one end, preventing axial float. This prevents fretting of the shaft end splines.

TORQUE DISCS

Torque discs are used where a small change in direction is required. Torque discs provide sufficient flexibility to cope with wing bending. The nominal running angle is 0°, but they can accommodate up to ±1° of misalignment.

STEADY BEARINGS

Steady bearing assemblies are utilized to support the torque shafts. The transmission steady bearings are greased for life and do not require further servicing.

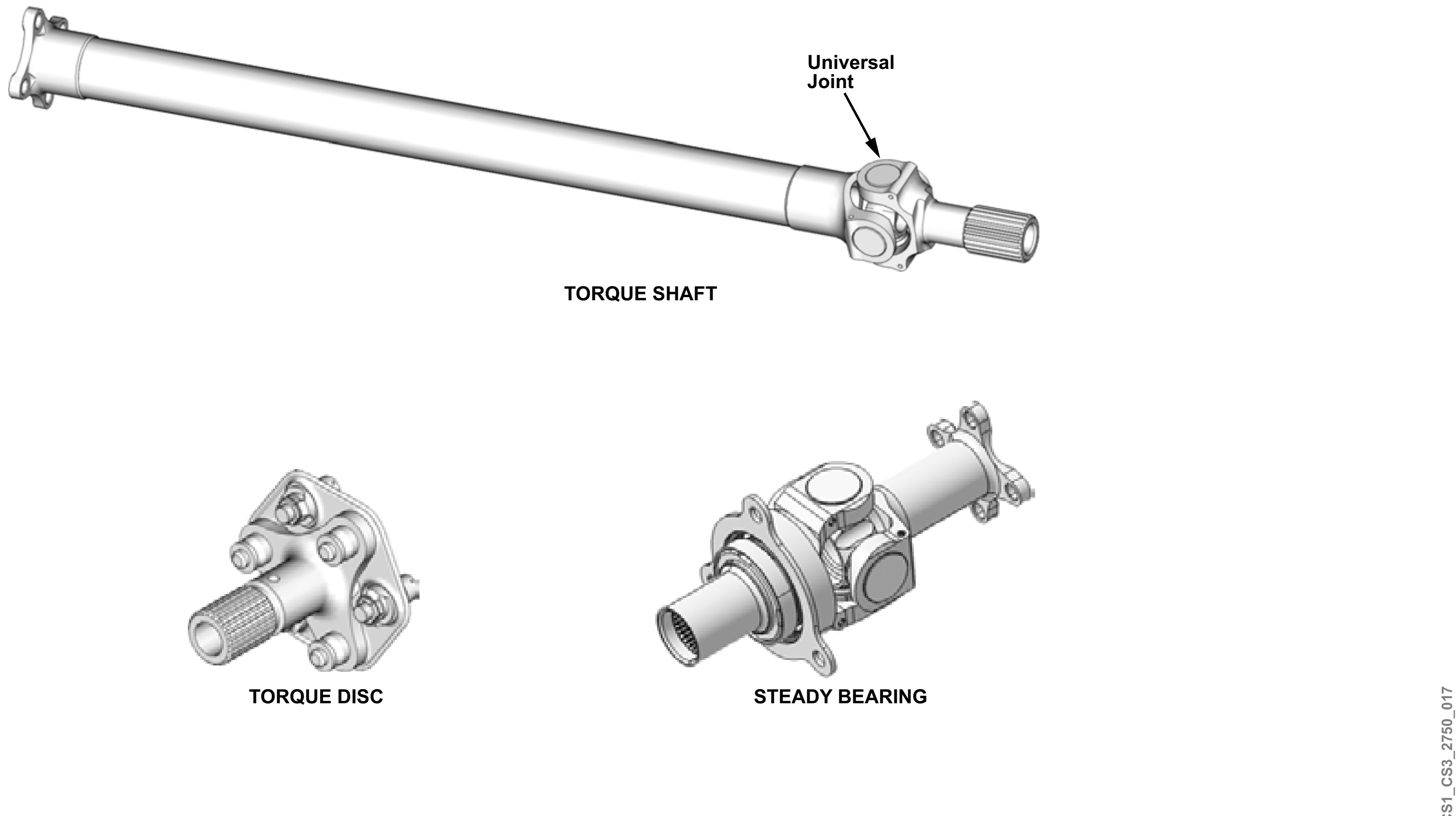


Figure 115: Torque Shaft, Torque Disc, and Steady Bearing

SLAT ACTUATORS

There are 16 slat actuators, that are also known as rotary geared actuators (RGAs). The RGAs convert high-speed and low torque from the PDU into slow speed and high torque to move the slats. The input shaft carries through the center to continue the drive train to the next component downstream. The output shaft has a drive gear, which engages the slat drive pinion to drive the slat rack assembly.

The actuators have lubrication fill and drain ports for servicing.

Each RGA has a torque limiter to prevent overtorque. There is an external indicator to show torque limiter activation.

Three types of actuators are used in the slat drive system. The operation is the same for all three types, only the physical configuration is different. Due to the size and torque of the three types, with Type I providing the most torque, the actuators are distributed to provide the greatest amount of torque inboard, decreasing in torque toward the wing tips. The actuators are arranged in the following sequence:

- Left and right no. 1 slats use Type I RGAs
- Left and right no. 2 slats use Type II RGAs
- Left and right no. 3 and no. 4 slats use Type III RGAs

SLAT OUTBOARD BRAKES

The slat outboard brakes (OBBs) are used to hold the transmission system between selections, as well as stopping and holding the system during abnormal conditions.

The brakes are normally applied (pressure off). The brakes are hydraulically released, using hydraulic pressure, when either brake solenoid is energized.

The OBB has a proximity switch to signal brake engaged/disengaged status to the slat/flap electronic control unit (SFECU).

A manual release shaft can be operated to release the OBB for maintenance purposes.

The OBB solenoid valves can be replaced separately.

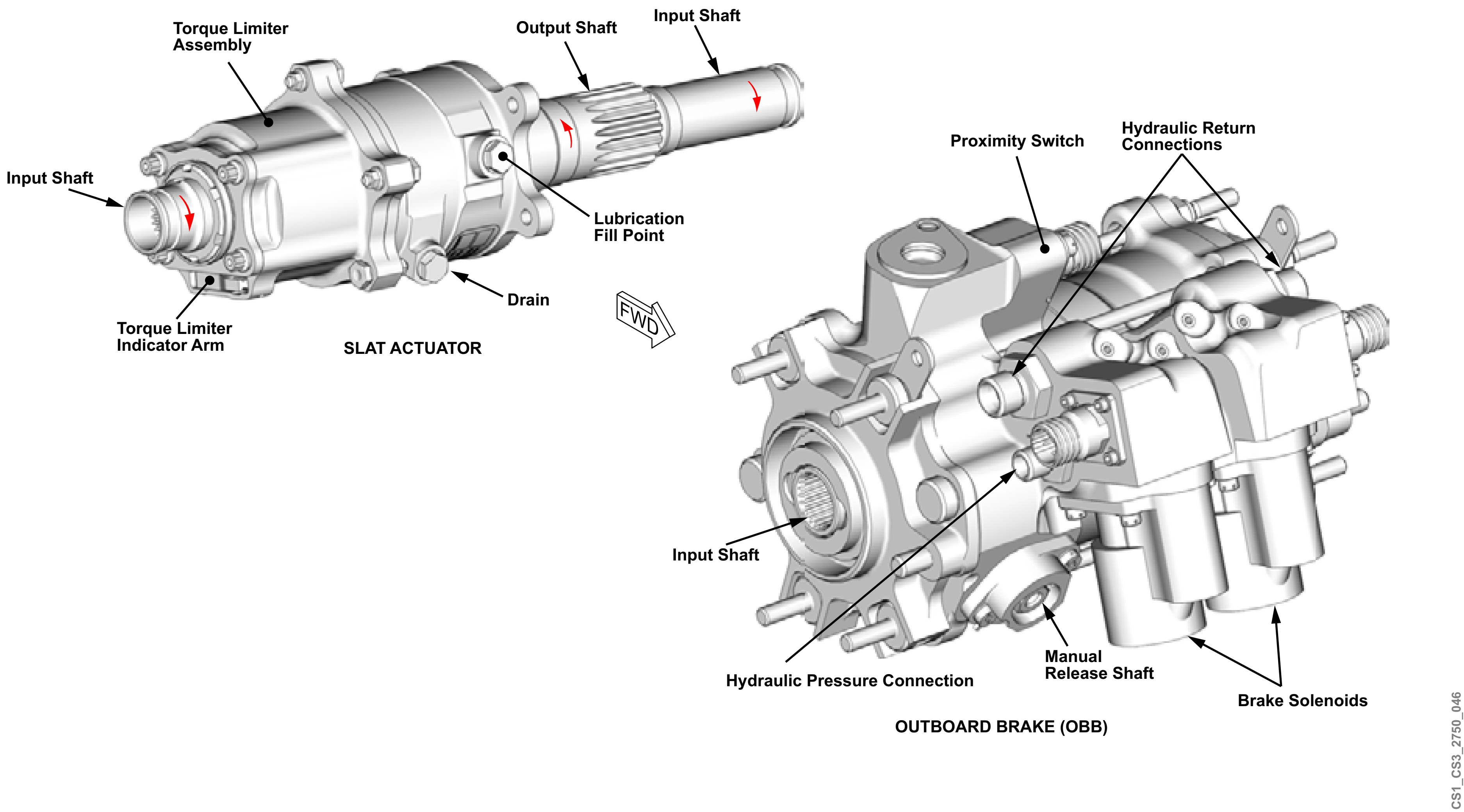


Figure 116: Slat Actuator and Outboard Brake

SLAT OUTBOARD POSITION ASYMMETRY SENSORS

There is a slat outboard position asymmetry sensor (OPAS) mounted at the outermost actuator on both ends of the slat actuation system.

The OPAS have two internal sensors. The sensors are continuously monitored by the SFECU through dual-channel input/output. The SFECU determines the difference between the left wing OPAS signal and the right wing OPAS signal which indicates the difference in position of the slat on one wing to the slat on the other wing. If the difference in position exceeds a defined limit, then wing to wing asymmetry of the slats is confirmed by the SFECU, and the OBBs are engaged to stop and hold the slats.

The OPAS is a line replaceable item.

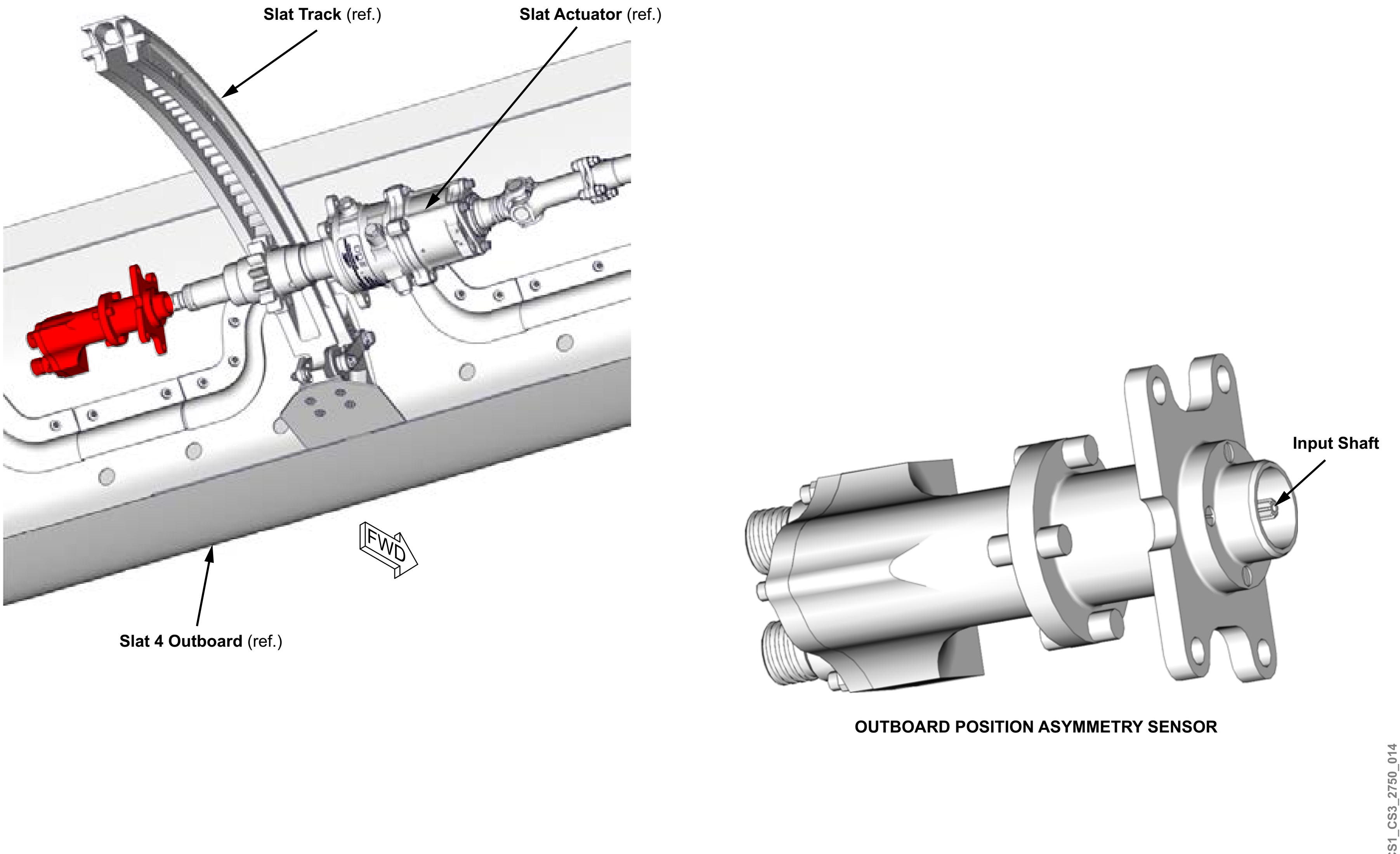


Figure 117: Slat Outboard Position Asymmetry Sensor

SLAT SKEW SENSOR ASSEMBLIES

The slat system has 16 slat skew assemblies. Each skew sensor assembly consists of two proximity sensors with their associated target strips.

A proximity sensor and its target strip is mounted on either side of the slat track. The proximity sensors are offset-mounted, relative to each other, to give 90° phase difference. The proximity sensors monitor the ferrous plates on the target sensor strips. Slat skew signals are sent to the landing gear and steering control unit (LGSCU). If a change in phase difference is detected, the LGSCU will command the slat/flap electronic control unit (SFECU) to command the OBBs to engage to stop and hold the slats.

The skew proximity sensors are line replaceable items.

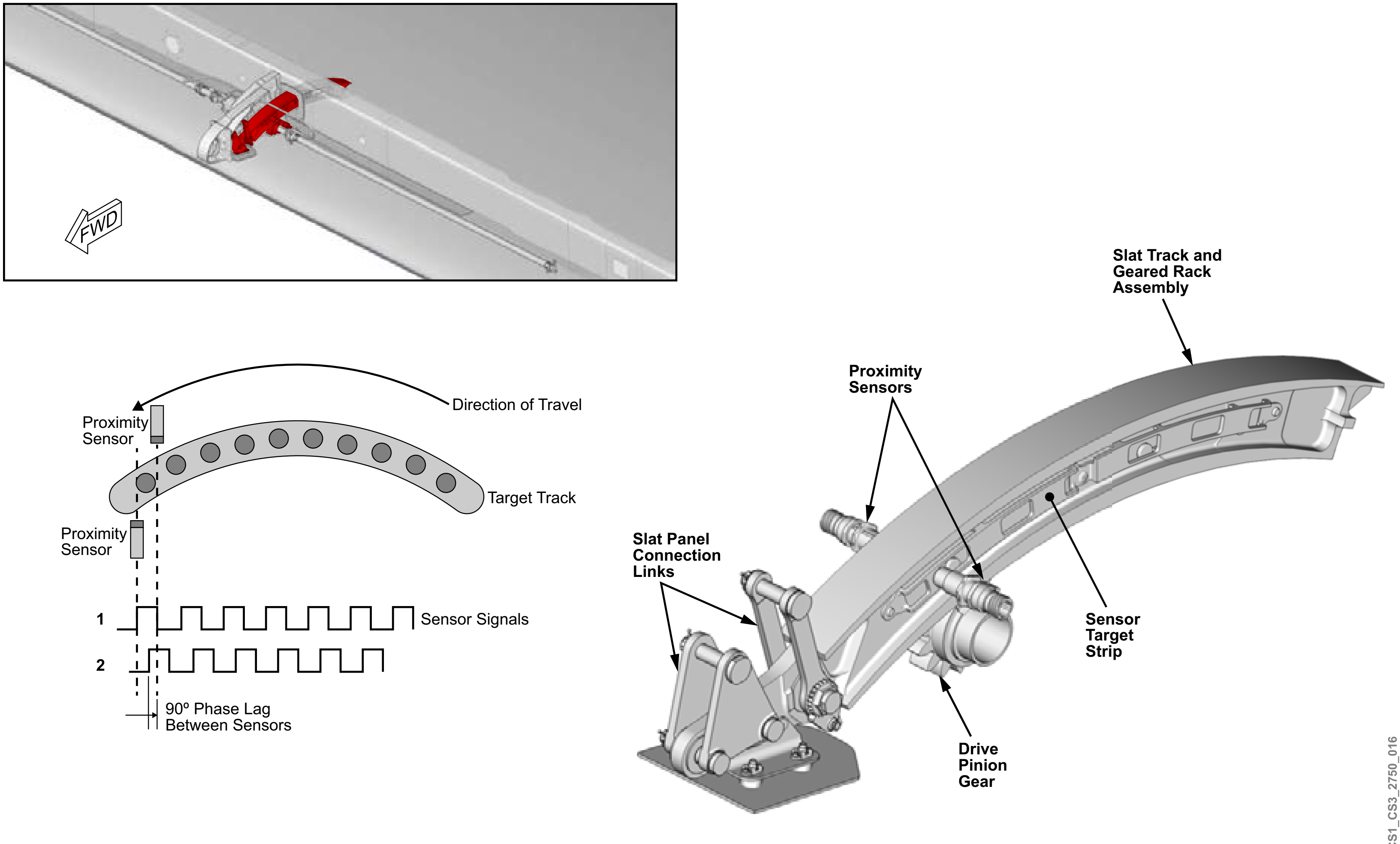


Figure 118: Slat Skew Sensor Assembly

SLAT TRACK CANS

The slat track cans are attached to the front spar inside the fuel tank and provide the space for the slat tracks to retract into it. Slat track cans installed in slat panel 1 do not have a drain port because they are designed to drain forward. Slat track cans for slat panel 2 and 3 are drained from the lower point via combination of a metallic tube permanently attached to the can, a flexible hose, and another metallic tube connected to a drain outlet in the wing bottom skin. Any accumulated water is drained overboard.

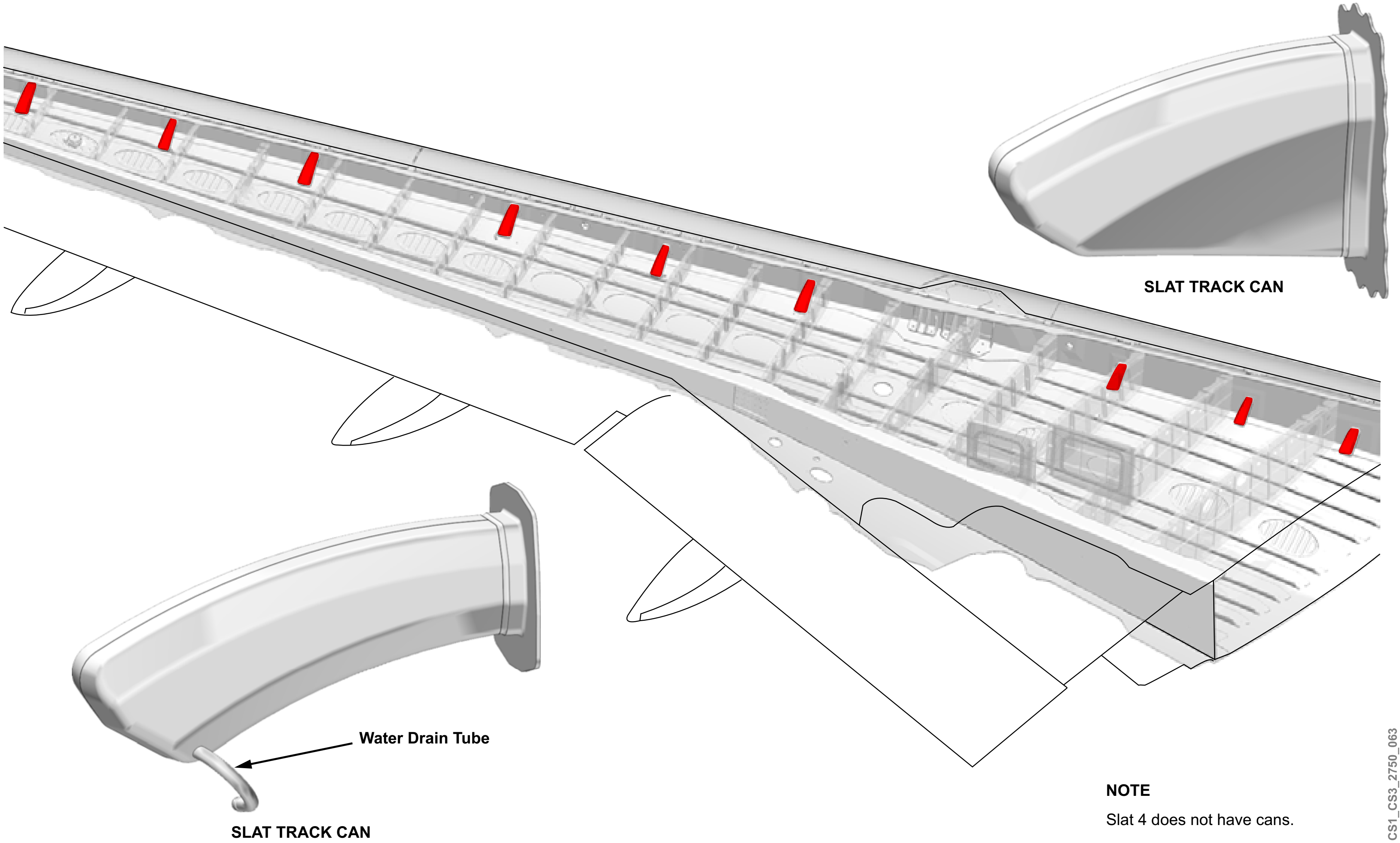


Figure 119: Slat Track Cans

DETAILED COMPONENT INFORMATION

POWER DRIVE UNIT

The power drive units (PDUs) provide the slat and flap actuation systems with mechanical power through an output shaft, as commanded by the slat/flap electronic control units (SFECUs), to drive the slat and flap surfaces against air loads and system friction.

The slat and flap PDUs are identical. The differences between the two units are the hydraulic power sources and that the flap PDU is mounted transversely in relation to the slat PDU.

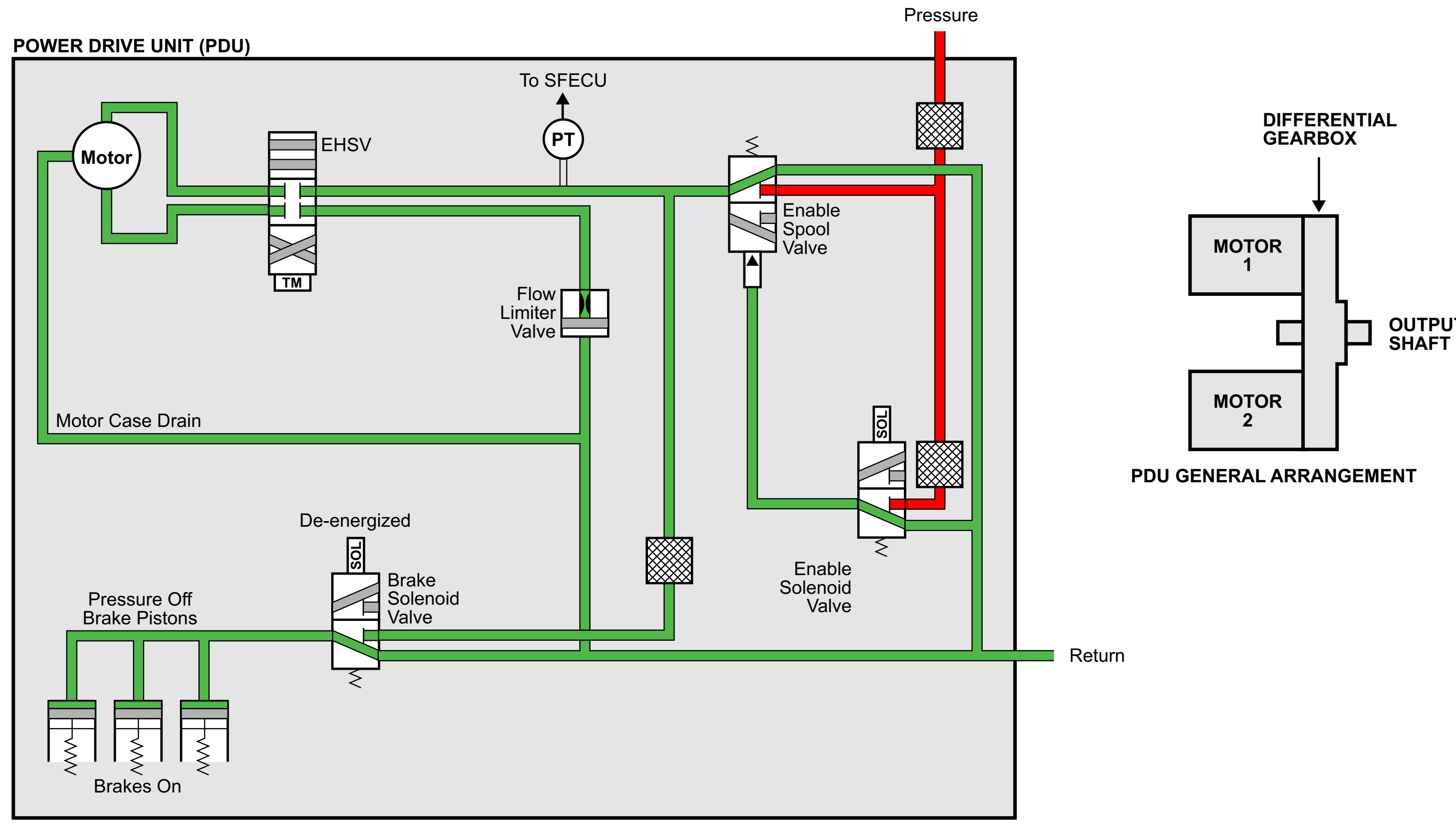
Strainers are located within the PDU to stop coarse debris from entering the control valves.

An enable valve is used to control hydraulic pressure to the torque motor controlled, electrohydraulic servovalve (EHSV). When the SFECU energizes the enable valve solenoid, the enable spool valve opens, supplying hydraulic pressure. The SFECU monitors the pressure supplied through a downstream pressure transducer.

The hydraulic motor direction and speed is controlled by the EHSV, which is controlled by the SFECU. Motor top speed is limited by a flow limiter valve in the return side of the EHSV.

Each PDU motor has an associated mechanical, pressure off brake (POB), to hold the actuation system when required. The brake is applied under all conditions except when being commanded to release by the SFECUs. The POBs are automatically applied in the event that either hydraulic or electrical power supply to the PDU is lost.

The PDUs have a speed summing differential gearbox, which combines the output from the dual-drive channels into a single PDU output drive. This is to enable the system to operate (at half speed and full torque) when one channel of the PDU has failed and hence meet the PDU availability requirements.



LEGEND

- Return Pressure
- System Pressure
- EHSV Electrohydraulic Servovalve
- PT Pressure Transducer
- SFECU Slat/Flap Electronic Control Unit
- SOL Solenoid
- TM Torque Motor
- In-Line Strainer

NOTE

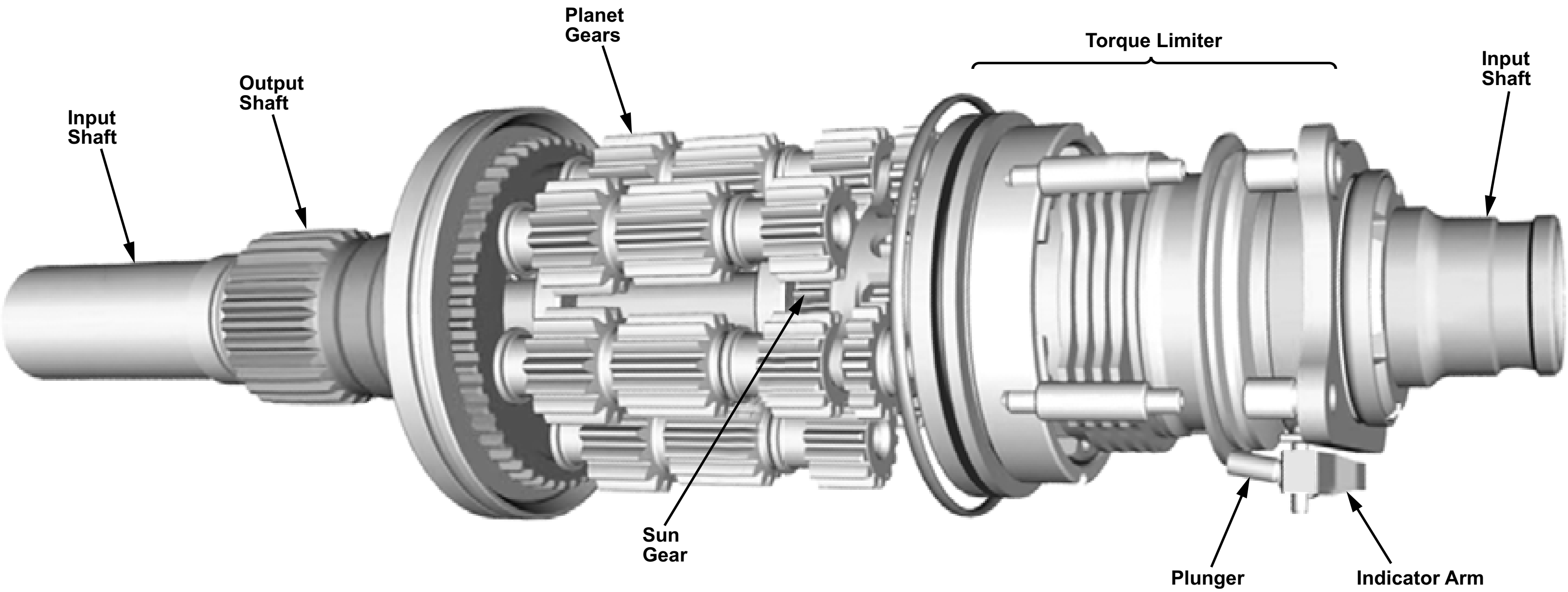
One PDU motor schematic is shown.
PDU is de-energized.

CS1_CS3_2750_057

Figure 120: Power Drive Unit Schematic

SLAT ACTUATORS

The slat actuators are of co-axial design with a drive shaft through the center, which drives epicyclic (sun and planet) gears to reduce speed and increase torque. Each actuator has a torque limiter to prevent overtorque. There is an external indicator to show torque limiter activation. The output shafts of the actuators have a drive gear, which engages the slat drive pinion to drive the slat rack assembly.



NOTE

Cover removed for clarity.

Figure 121: Slat Actuator Internal Arrangement

SLAT ACTUATOR TORQUE LIMITER

The slat and flap actuators contain a torque limiter. The torque limiter acts as a mechanical fuse to protect the aircraft structure and the slat drive mechanism from excessive loading.

Increased loads, generated by one or more actuation system components, progressively applies a braking force on the input shaft of the actuator. Should the torque limit setting be exceeded, the torque limiter locks out and prevents further rotation of the input shaft. If a lockout condition occurs an indicator arm protrudes from the actuator. This provides a visual indication of lockout. Once the cause of the lock out has been rectified the indicator arm can be reset. Reset involves pushing the indicator arm back into its housing, after unlocking the affected system.

The torque limiter consists of six balls contained with a ball-ramp mechanism, a brake assembly and several spring packs.

During normal operation all parts of the torque limiter rotate together. The balls within the ballramp mechanism remain in the center of the ramps and there is full clearance for the brake mechanism (rotor discs and stator discs).

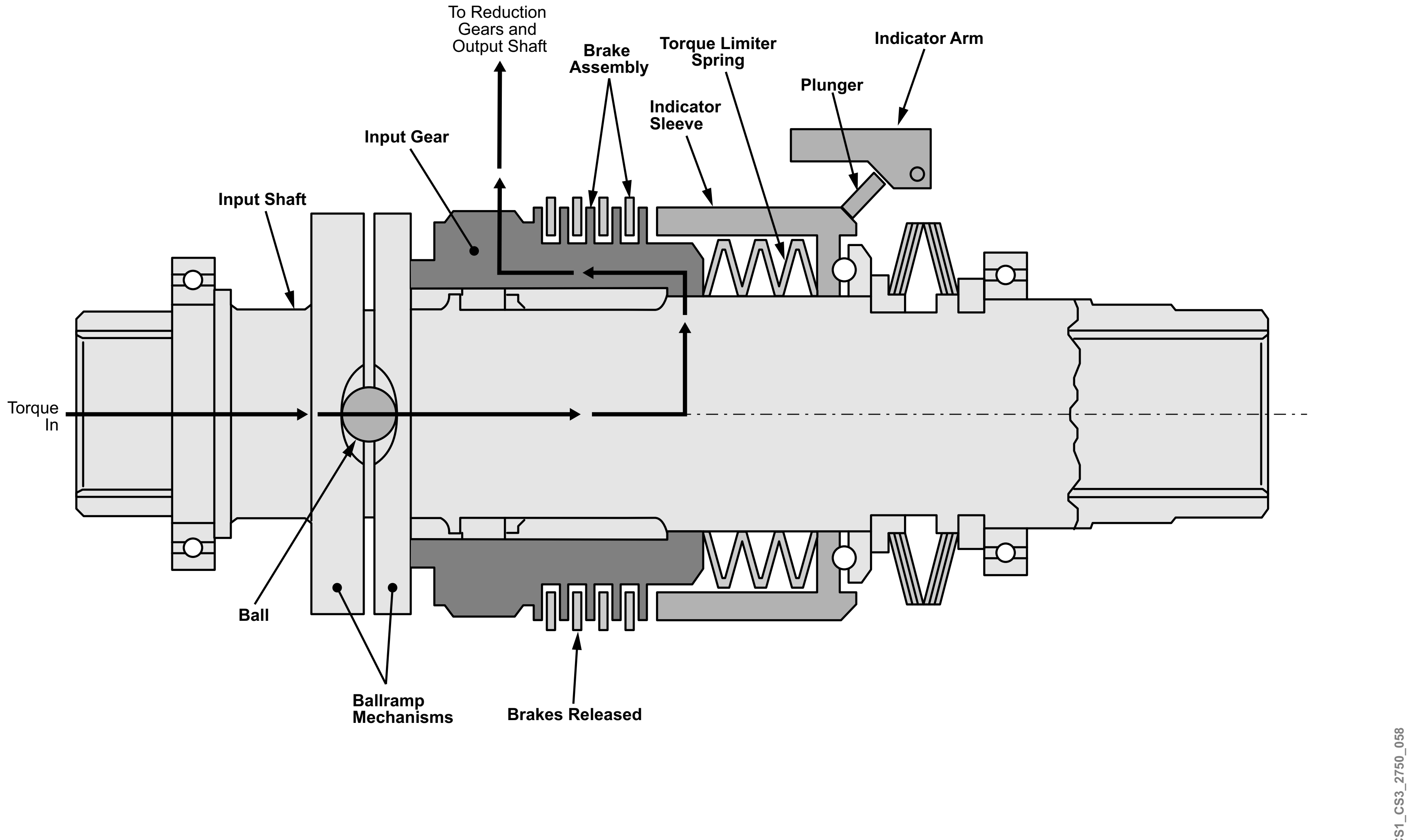


Figure 122: Slat Actuator Torque Limiter Operation - Normal

TORQUE LIMITER OPERATION - BRAKING AND LOCKOUT

During the start of a lockout, increased loading on the output shaft causes the balls to move up the ramps, compressing the torque limiter spring. Clearance in the brake assembly (rotor and stator discs) is reduced and the brake assembly starts to generate braking torque. Braking torque adds to the output torque sensed by the torque limiter, resulting in a self-servo effect. As the input shaft continues to rotate, the brakes apply, increasing pressure until the input shaft is locked.

Simultaneously, as the balls move further up the ramps, the assembly moves against the indicator spring. The indicator sleeve pushes the plunger, deploying the indicator arm. The plunger locks the indicator arm in the extended position.

The locked out actuator prevents the actuation system from moving the slats to the selected position. The failure to reach the selected position is detected by the slat/flap electronic control units (SFECUs), which shut down the actuation system and generate the appropriate EICAS messages and synoptic page indications.

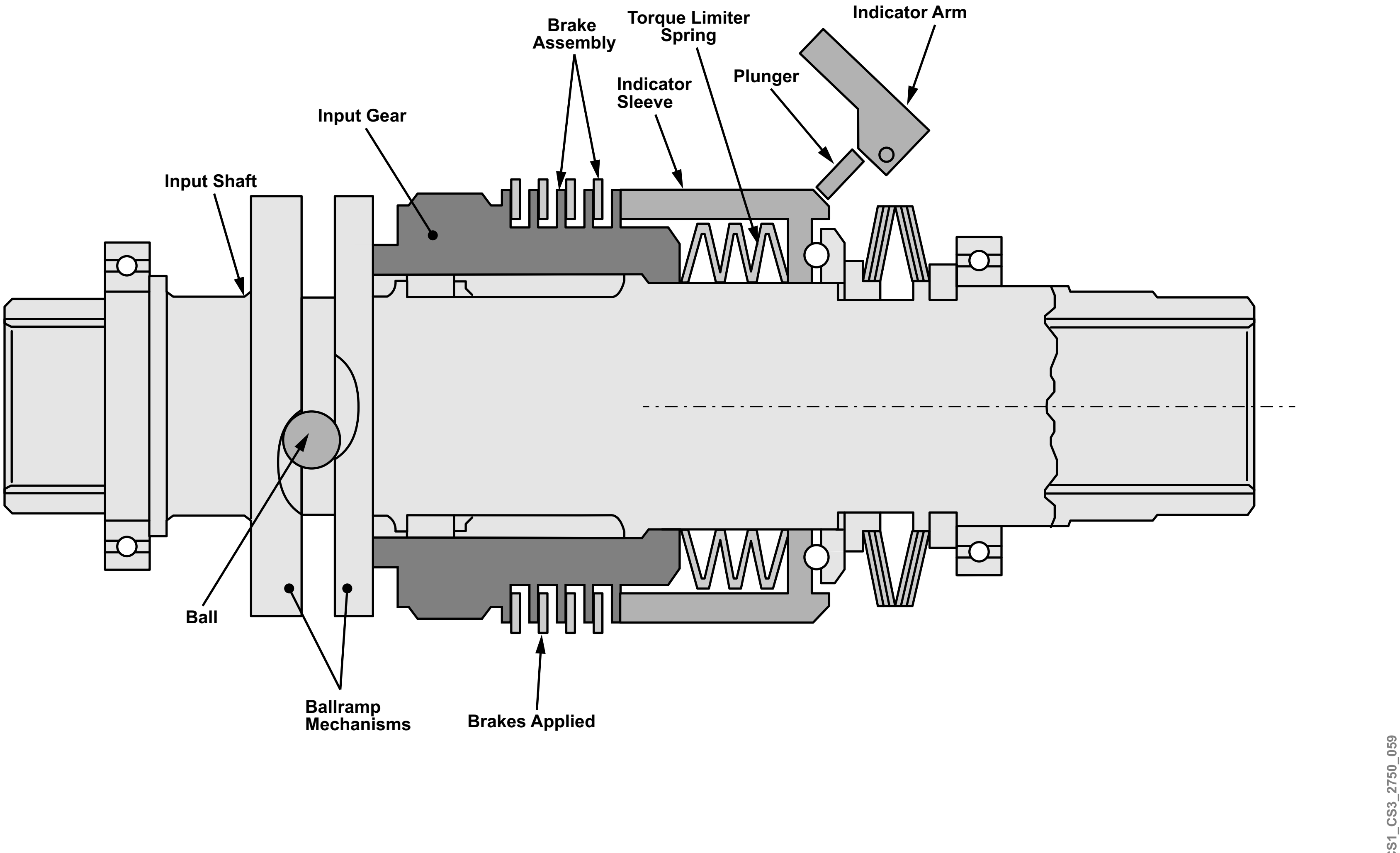


Figure 123: Slat Actuator Torque Limiter Operation - Braking and Lockout

OUTBOARD BRAKE

The outboard brakes (OBBs) are pressure off brakes (POBs). The brakes are normally applied, requiring hydraulic pressure to release.

The OBB consists of a drive shaft which drives a number of rotor discs housed in the brake disc pack. The stator plates of the brake disc pack are connected to the OBB housing. With no hydraulic pressure applied, the rotor discs are pressed against the stator discs by the disc spring pack, preventing rotation of the drive shaft.

A proximity sensor reports the position of the brakes to the slat/flap electronic control units (SFECUs).

The OBBs have a manual release, which allows the OBBs to be released without commands from the SFECUs. This enables the brakes to be released during maintenance operations. A maintenance mode indicator provides external indication of manual release. When rotated to the manual release position, the manual release wrench locks in place, and prevents closure of the access panel.

Two integral solenoid valves control the hydraulic supply to the piston(s) of the OBB. If one of the solenoid valves is energized, then the pressure acts on the corresponding piston. When hydraulic pressure is applied, the dual-pistons displace a pressure plate, which compresses the disc spring pack. This deflects the proximity target lever, moving the target away from the proximity sensor. The proximity sensor allows the system to compare the position of the brake release pistons to the commanded position and hence monitor for failures of the brake function. The proximity sensor signals are sent to the landing gear and steering control units (LGSCUs) and the SFECUs.

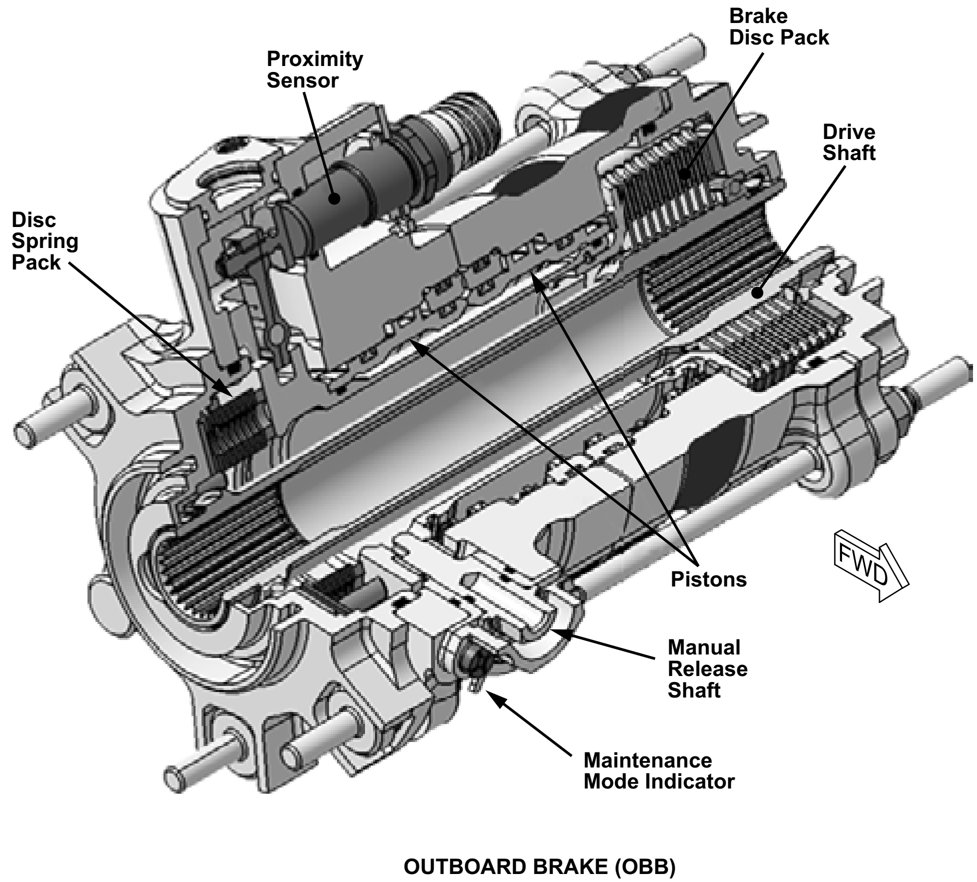


Figure 124: Outboard Brake

OUTBOARD BRAKE - BRAKES ON

Two integral brake solenoid valves control the hydraulic supply to the pistons of the brake disc pack. The solenoid valves are normally de-energized preventing hydraulic pressure from entering the pistons.

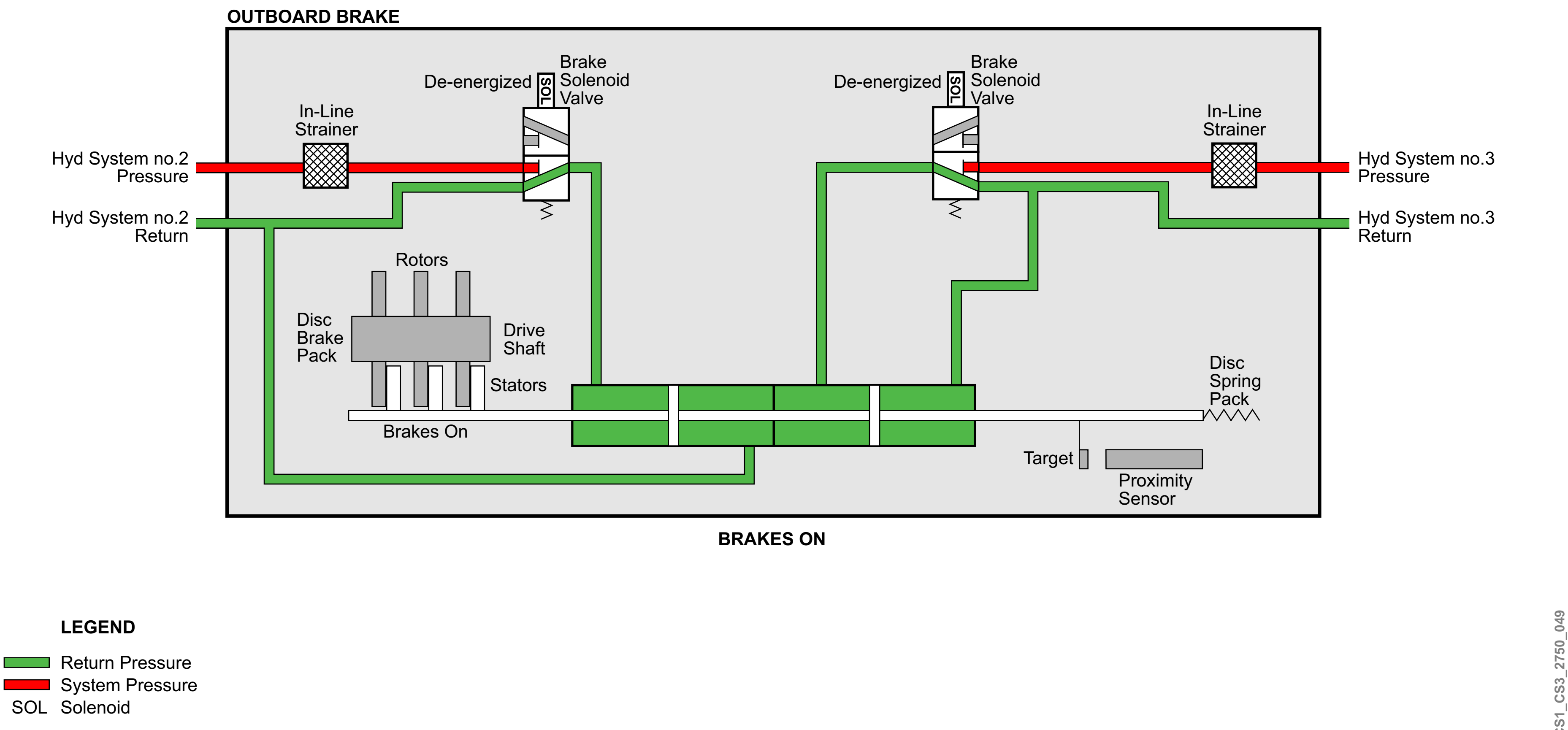


Figure 125: Outboard Brake - Brakes On

OUTBOARD BRAKE - BRAKES OFF

When any one of the solenoid valves is energized, pressure acts on the corresponding piston. The piston compresses the disc spring pack releasing the brakes.

The proximity sensor monitors the movement of the pistons and reports its position to the landing gear and steering control units (LGSCUs), which forward this information to the slat/flap electronic control units (SFECUs).

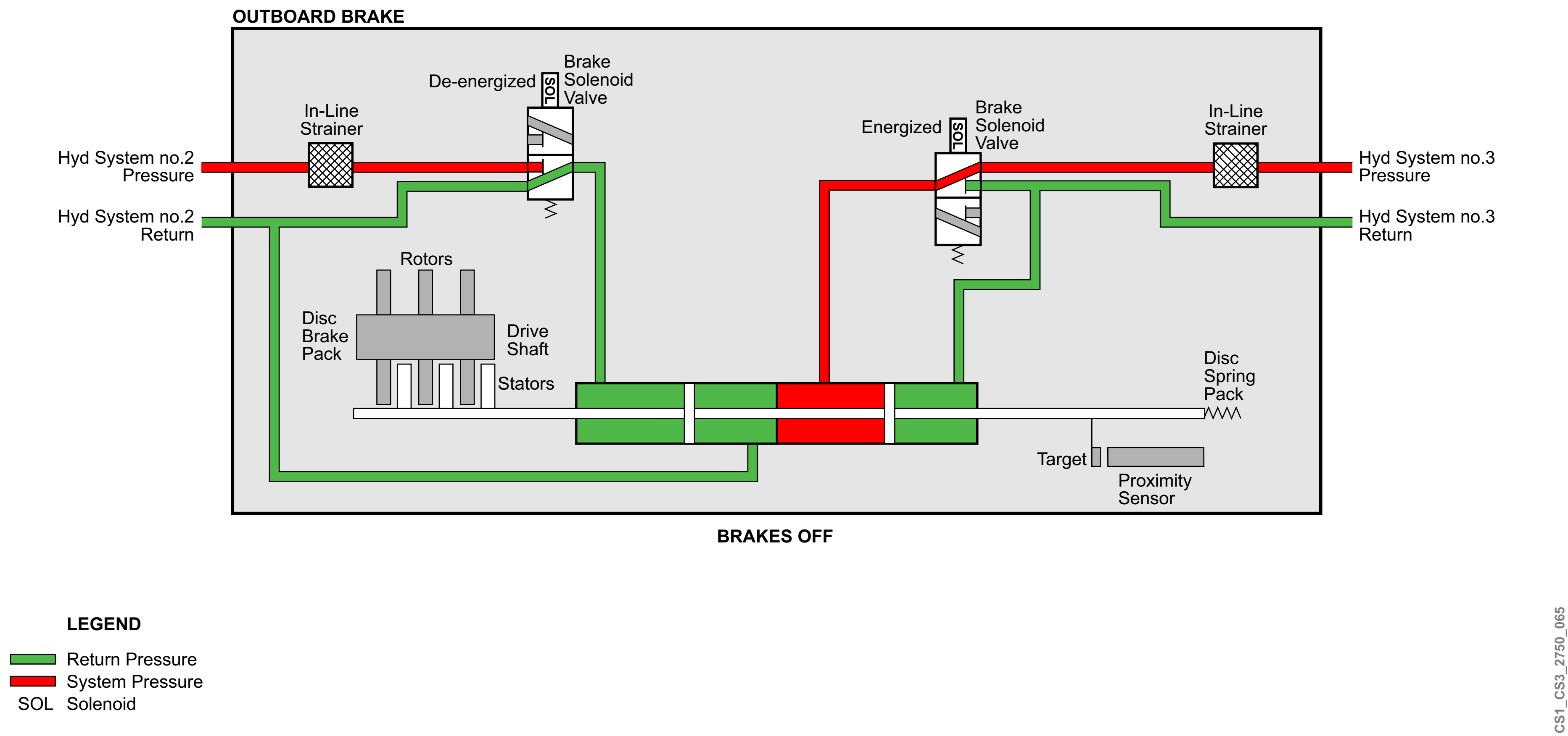


Figure 126: Outboard Brake - Brakes Off

SLAT TRACK MECHANISM

The slats are extended and retracted through the movement of slat tracks. The slat tracks have racks which engage with pinion gears. The pinion gears are rotated by the slat actuators.

Various rollers allow the tracks to extend and retract smoothly.

The main rollers restrain the motion of the slat tracks in the up-down direction. There are four rollers per track, two above the track and two below it. Any up-down loads on the slats will be transmitted to the wing through the vertical rollers. The forward bottom rollers also act as stops to prevent the track from overretracting and damaging the track cans.

The side rollers have single row needle bearings. They restrain the motion of the slat tracks in the inboard-outboard direction. The side rollers are mounted on the leading edge ribs. There are four rollers per track, two on the inboard rib and two on the outboard rib. Any inboard-outboard loads on the slats will be transmitted to the wing through the side rollers.

The pinion gears transmit the rotational movement from the slat actuator output shafts to the slat driving tracks. The pinion gears also act as down stops in case of forward bottom roller failure. The gears are mounted to the wing leading edge structure.

The pinion bearings are greased ball bearings. They support the pinion gears and permit them to rotate smoothly. The pinion bearings are mounted on the leading edge ribs. There are two pinion bearings per driving track, one on the inboard rib and one on the outboard rib.

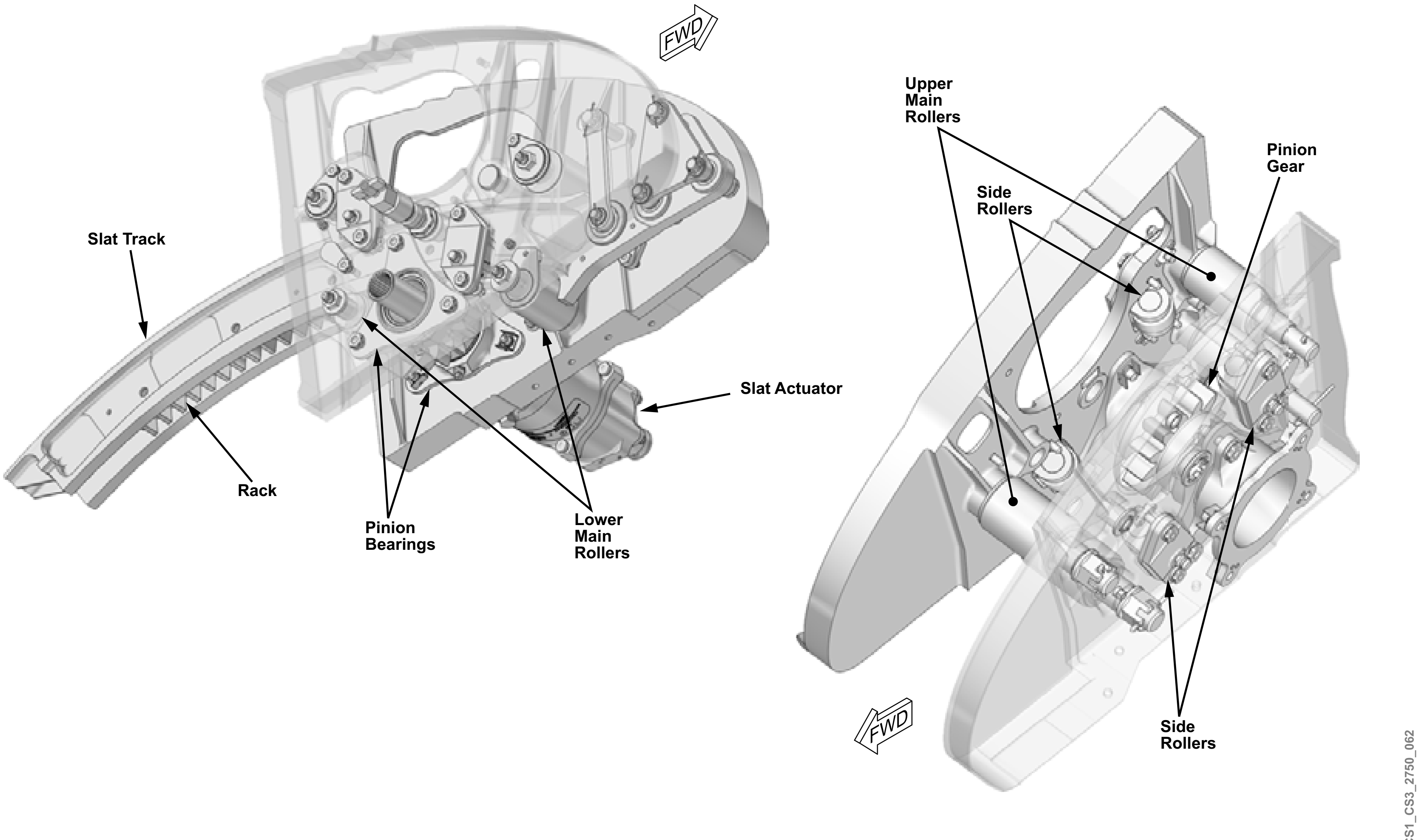


Figure 127: Slat Track Mechanism

27-50 FLAP ACTUATION SYSTEM

GENERAL DESCRIPTION

The mechanical flap actuation system begins at the power drive unit (PDU). Torque is transmitted to the gearboxes by torque shafts. The gearboxes transfer the torque through a series of torque shafts, universal joints, and steady bearings to the actuators, the outboard brakes (OBBs), and the outboard position asymmetry sensors (OPAS).

COMPONENT LOCATION

The flap actuation system consists of the following principal components:

- Power drive unit (PDU) and ecology bottle
- Gearboxes (Type 1, 2, and 3)
- Torque shafts and universal joints
- Torque discs
- Steady bearings
- Actuators
- Outboard brakes (OBBs)
- Outboard position asymmetry sensors (OPASs)

POWER DRIVE UNIT AND ECOLOGY BOTTLE

The flap PDU is located aft of the right main wheel well, in the wing-to-body (WTBF) fairing. An ecology bottle is located adjacent to the PDU.

GEARBOXES

There are five gearboxes. Three are located in the belly fairing, and one on each wing transmission circuit near WS250.

TORQUE SHAFTS AND UNIVERSAL JOINTS

The 27 torque shafts and 12 universal joints are located in the belly fairing, and along the left and right wing rear spars.

TORQUE DISCS

Five torque discs are located adjacent to specific flap actuators.

STEADY BEARINGS

Ten steady bearings located at various locations along the left and right wing rear spars.

ACTUATORS

There are eight flap actuators located along the left and right wing rear and auxiliary spars.

OUTBOARD BRAKES

The two OBBs are located between actuator 3 and 4 on each wing.

SKEW SENSORS

Two skew sensors are located at the forward end of left and right flap track no. 1, with six skew sensors located at the aft end of the remaining six flap tracks.

OUTBOARD POSITION ASYMMETRY SENSORS

An OPAS is located at the ends of the left and right flap actuation system.

NOTE

The PDU, OBBs, and OPASs are identical to those used in the slat system. Torque shafts, universal joints, torque discs, and steady bearings are similar to those on the slat system.

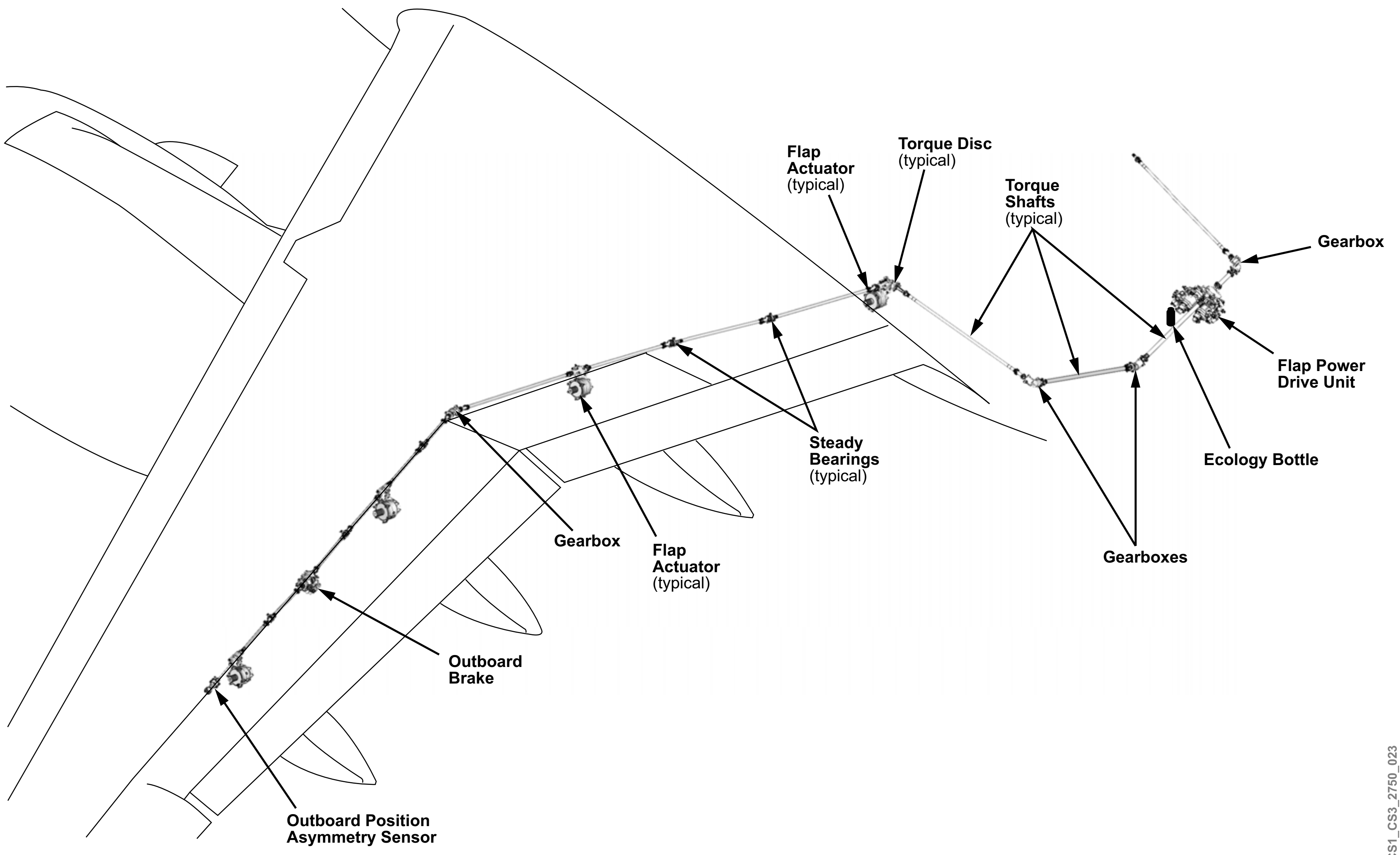


Figure 128: Flap Actuation System Components

COMPONENT INFORMATION

FLAP POWER DRIVE UNIT - OUTBOARD BRAKES AND OUTBOARD POSITION ASYMMETRY SENSORS

For information on the flap power drive unit (PDU), outboard brakes (OBBs) and outboard position asymmetry sensors (OPASs), refer to the slat system.

FLAP ACTUATORS

The flap actuators, also known as rotary geared actuators (RGAs), are internally similar in design to the slat actuators, with the exception of the inclusion of bevel gears and a down drive gear train (offset gearbox), dependent on the location of the actuator within the flap actuation system.

The flap system uses four different actuator types, which differ slightly in configuration for each installed position.

There are two flap panels per wing, with two flap actuators per panel. The actuators are designed such that they take the relatively high-speed/low torque revolution of the transmission and convert it to high torque low-speed to drive the flap panel.

The offset gearbox provides a through-drive to allow torque to be transmitted to the downstream actuators. It contains a takeoff gear train to drive the flap surface loads, and provides a torque limiting function which protects the aircraft structure in the case of a jam. It also provides an indication in the event of a torque limiter trip.

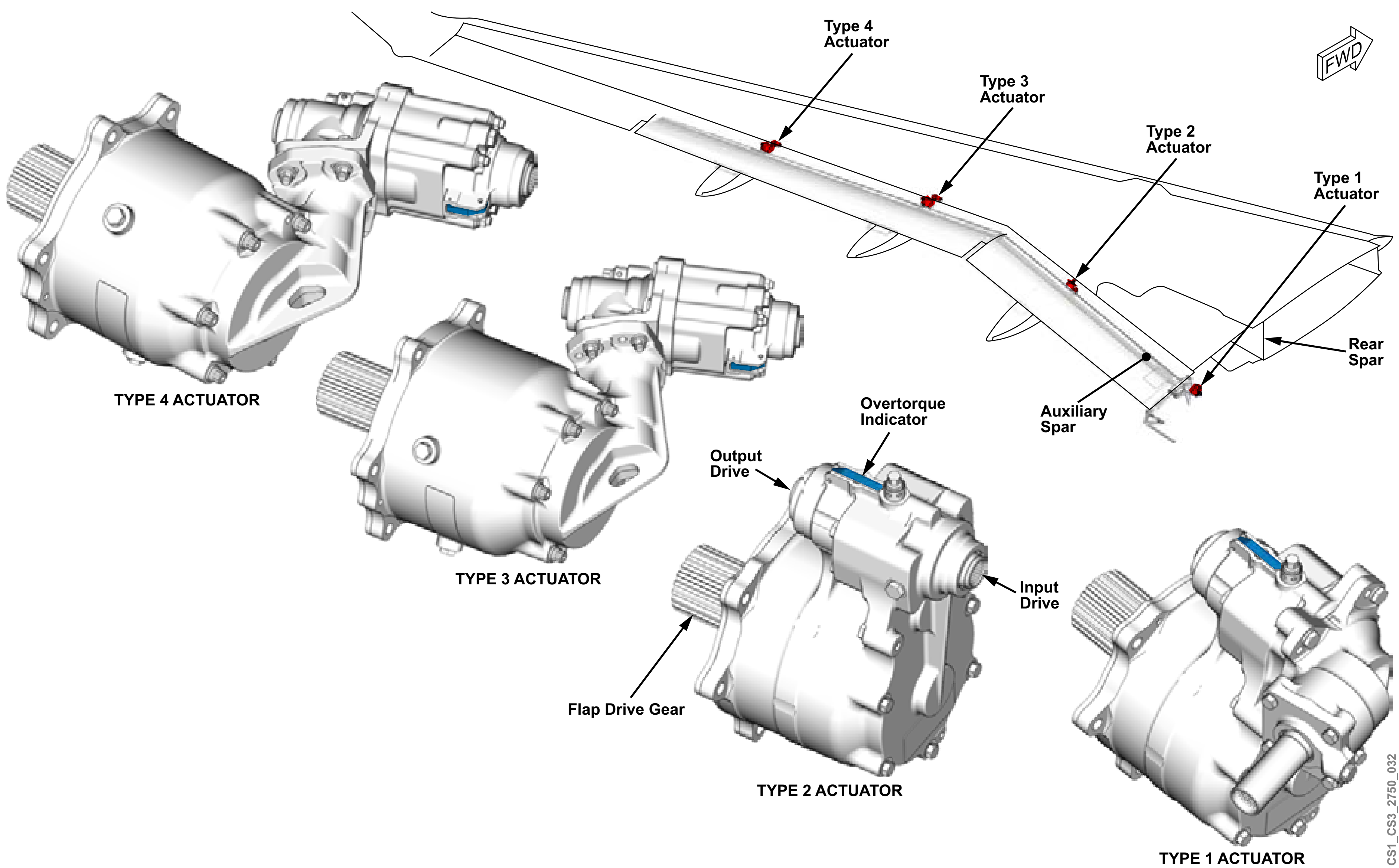
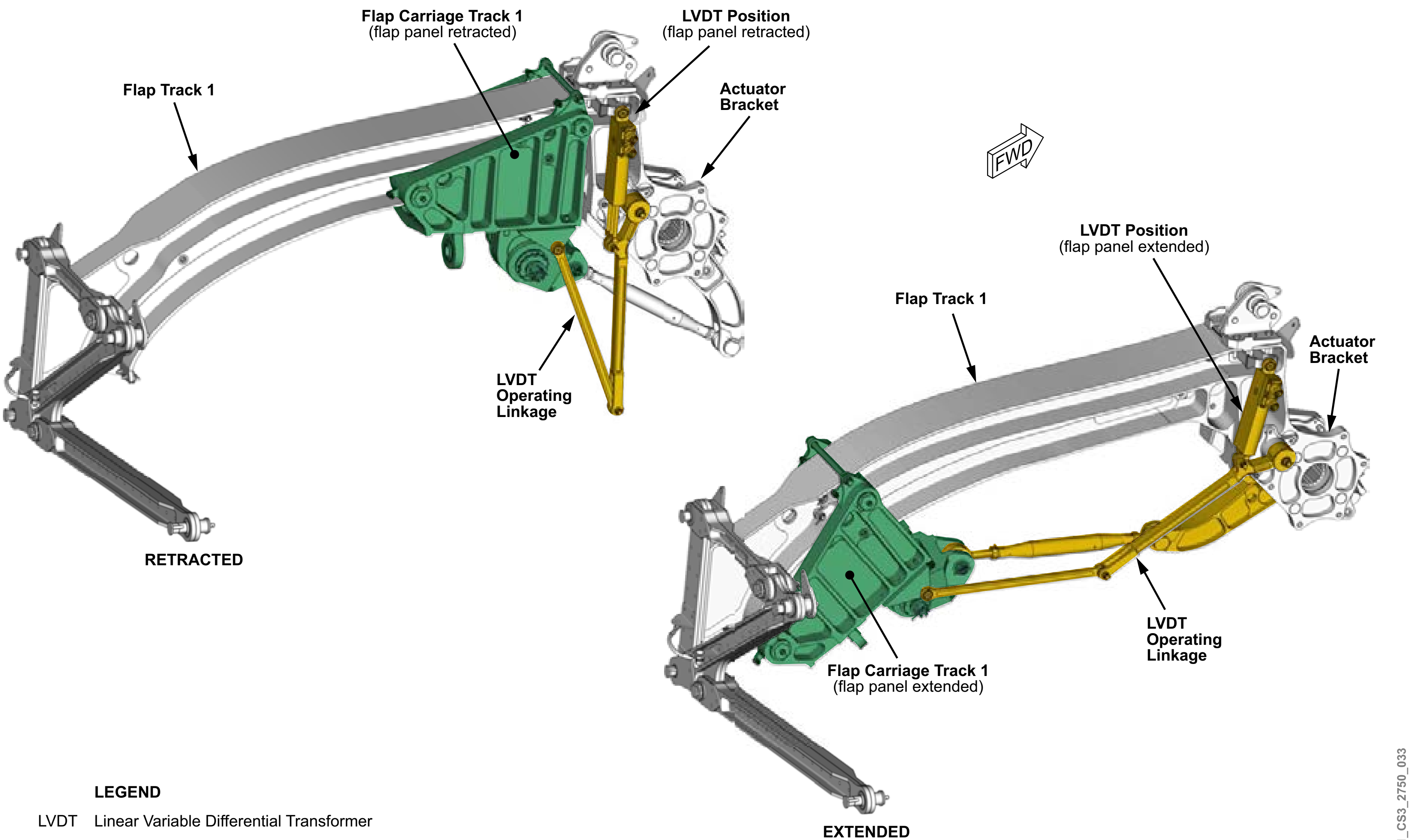


Figure 129: Flap Actuators

FLAP SKEW SENSOR SYSTEM

The flap skew sensor system uses linear variable differential transformers (LVDTs) and rotary variable differential transformers (RVDTs).

The left and right innermost flap tracks (no. 1) skew detection system uses an LVDT operated by a linkage. The flap skew linkage articulates with the flap carriage as it moves during panel extension and retraction resulting in linear displacement of the LVDT. The LVDT provides position data to the slat/flap electronic control units (SFECUs).



LEGEND

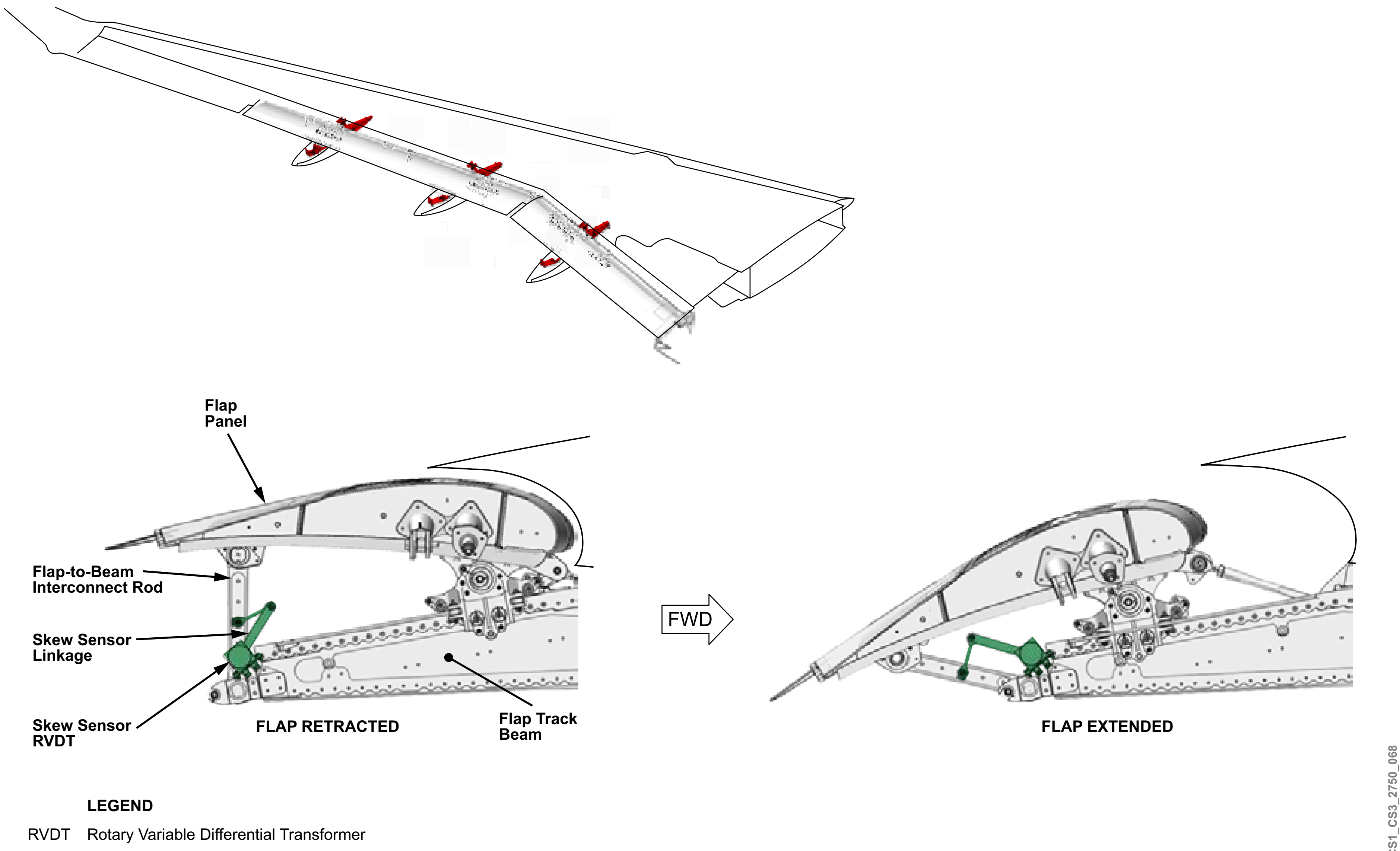
LVDT Linear Variable Differential Transformer

Figure 130: Flap Track No. 1 Skew Sensors

FLAP TRACKS 2 TO 4 SKEW SENSOR SYSTEM

Flap tracks 2, 3, and 4 use a rotary variable differential transformer (RVDT) for skew detection. Each RVDT is driven by a skew sensor linkage, which is attached to the flap-to-beam interconnect rod. As the flap panel moves, deflection of the interconnect rod drives the sensor linkage to rotate the RVDT.

The RVDT provides position data to the slat/flap electronic control units (SFECUs).



LEGEND

RVDT Rotary Variable Differential Transformer

Figure 131: Flap Tracks 2, 3, and 4 Skew Sensors

DETAILED COMPONENT INFORMATION

FLAP TRACK 1 ATTACHMENT AND DRIVE LINKAGES

The rotary actuator is attached to the track via two brackets secured to the track. The flap is connected at two points to the bottom of the carriage. The aft attachment is composed of a dog bone link allowing rotation around the inboard and outboard axis only. The forward attachment is composed of a swing link, which reacts to loading in the up, down, forward, and aft directions.

The actuator bracket is attached to the flap track. The rotary actuator moves the drive arm, which is connected to the flap through a drive rod.

When the rotary actuator moves the drive arm, the drive rod moves the flap panel rearward or forward, depending on the direction of rotation (extend/retract) of the actuator.

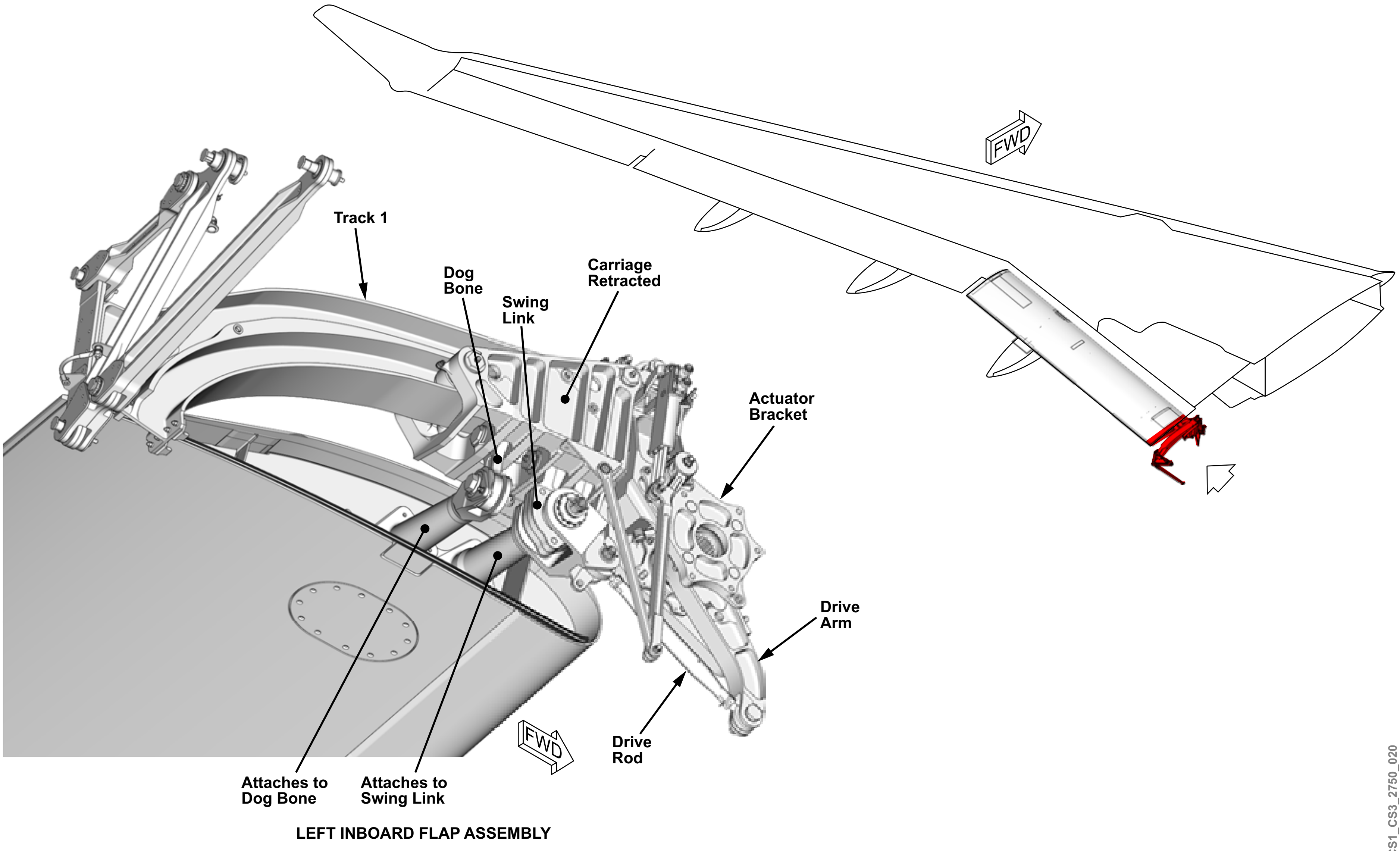


Figure 132: Flap Track 1 Attachments and Linkages

FLAP TRACK 1 CARRIAGE ASSEMBLY

The flap track 1 carriage assembly uses four main rollers to allow smooth translation of the flap carriage along the track. The two aft rollers ride in the lower groove of the flap track, while the forward rollers ride in the upper groove.

The side rollers are double row needle roller bearings and are provided with grease fittings. They keep the carriages aligned on the flap track. Any side loads on the flaps will be transmitted to the wing structure through the side rollers.

Eight guide rollers are used in the flap track carriage1 assemblies. They guide the side roller mounting brackets in the slot of track 1. These rollers are not loaded all the time and the load is minimal.

The box seal is installed in the wing-to-body fairing (WTBF) and provides a seal between flap track 1 and the WTBF. It consists of two halves, one covering the upper side and the other half covering the lower side. It provides an aerodynamic seal and prevents dirt accumulation. It has no impact on flap system operation.

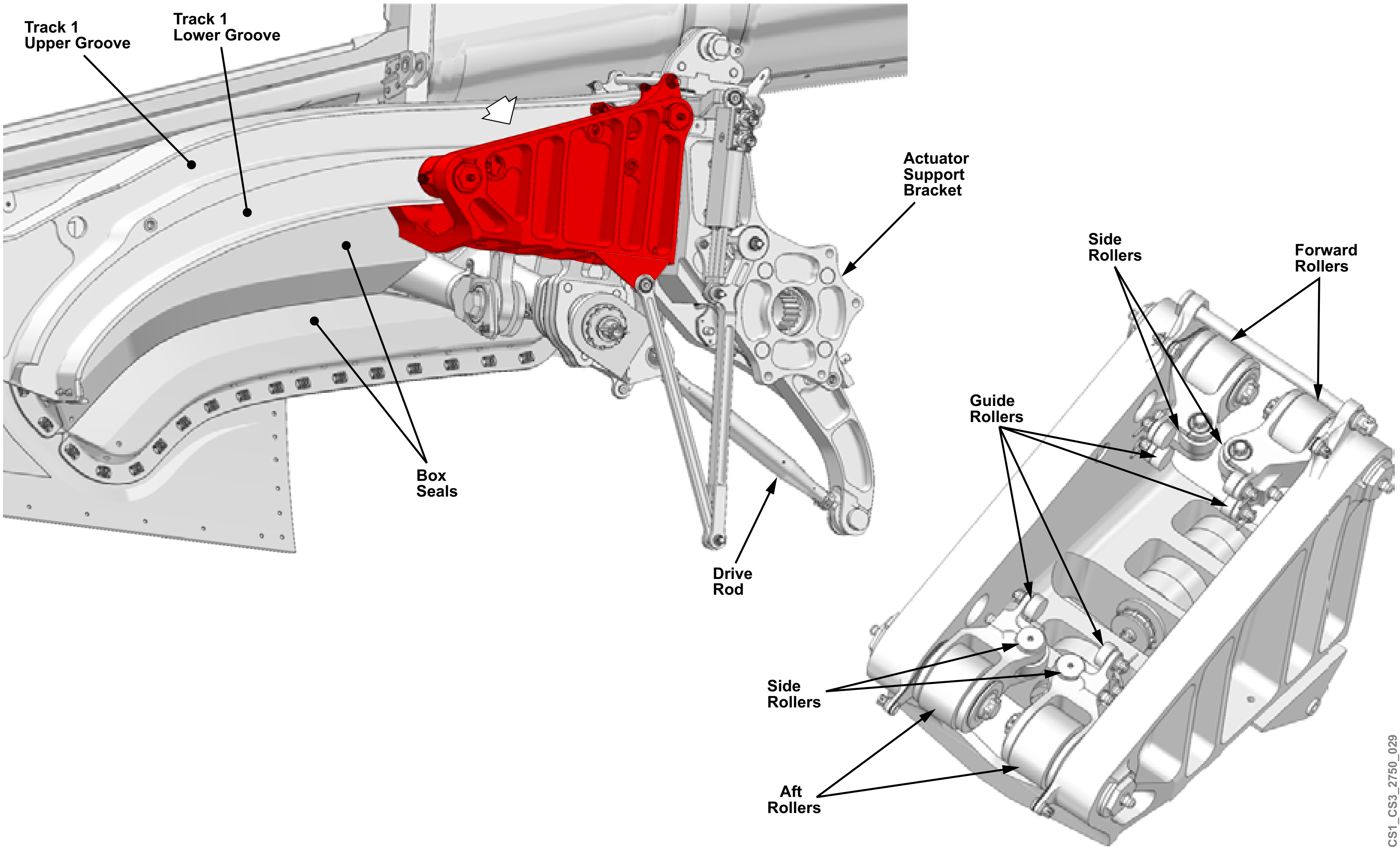


Figure 133: Flap Track 1 Carriage

FLAP TRACKS 2, 3, AND 4

The rotary geared actuators (RGAs) are attached to the wing structure, at the mid point of the flap track beam. The flap panel is connected at two points to the top of the carriage assembly. The aft attachment is composed of fitting, which connects to a flap-to-beam rod, attached to the aft end of the flap track beam. The forward attachment is composed of a single attach fitting and lug assembly, which connects directly to the carriage assembly.

The rotary actuator moves a drive arm, which drives the flap panel through a drive rod. The actuator-to-flap drive mechanism configuration is similar to that of flap track 1.

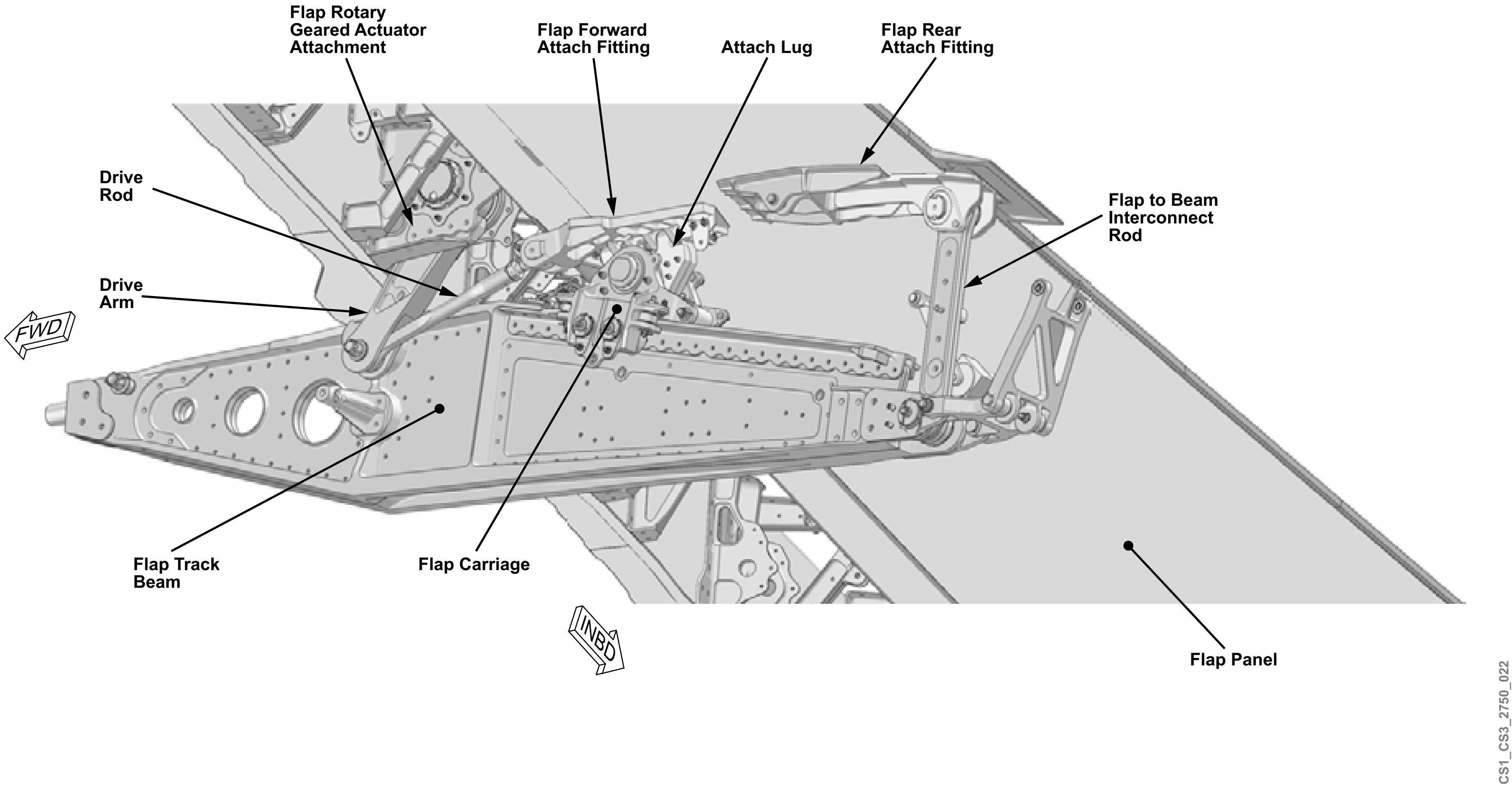


Figure 134: Flap Tracks 2, 3, and 4

FLAP TRACKS 2, 3, AND 4 CARRIAGE ASSEMBLIES

The flap tracks 2, 3, and 4 carriage assemblies use four flight rollers. The flight rollers are double row needle roller bearings, and are provided with grease fittings. They restrain the motion of the flap tracks in the up-down direction. Any up loads on the flaps will be transmitted to the wing structure through the flight rollers.

There are two ground rollers in each carriage assembly. The ground rollers are double row, needle roller bearings and are provided with grease fittings. They restrain the motion of the flap tracks in the down direction. Any down loads on the flaps will be transmitted to the wing structure through the ground rollers.

The four side rollers used in the carriages assemblies are double row, needle roller bearings and are provided with grease fittings. They keep the carriages aligned on the flap track. Any side loads on the flaps will be transmitted to the wing structure through the side rollers.

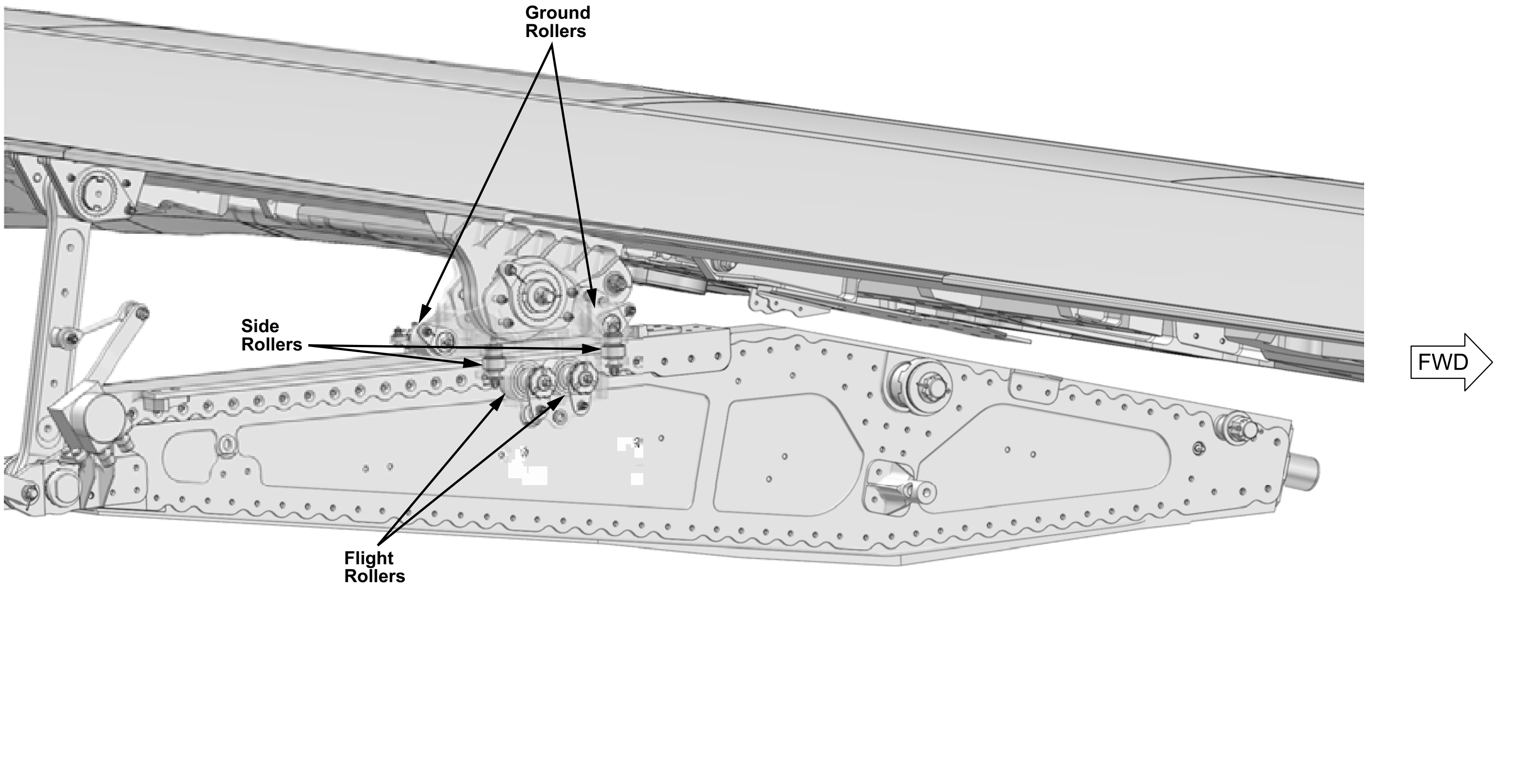


Figure 135: Flap Track 2, 3, and 4 Carriage Assemblies

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ATA 29 - Hydraulics

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

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HYDRAULICS - CHAPTER BREAKDOWN

29-10 HYDRAULIC SYSTEMS NO. 1 AND NO. 2

HYDRAULIC SYSTEM NO. 1

GENERAL DESCRIPTION

Hydraulic system no. 1 contains a reservoir which provides positive pressure supply to the left engine-driven pump (EDP) 1A and the power transfer unit (PTU) during all flight phases. A temperature switch, visual quantity indicator, temperature transducer, and fluid quantity transducer are provided for the reservoir.

Pressure switches, downstream of EDP 1A and the PTU, monitor pump output pressure. Pressurized fluid from EDP 1A and the PTU is routed through a pressure filter module, which contains a pressure-relief valve (PRV) to prevent overpressure. A pressure transducer on the pressure filter module transmits system hydraulic pressure signals. If the system hydraulic pressure decreases, a priority valve shuts off fluid flow to the landing gear system.

All users return fluid passes through the return filter module before re-entering the reservoir. Case drain fluid from EDP 1A and the PTU pump passes through its respective case drain filter before joining the return flow.

Hydraulic fluid temperature extremes in the system are avoided by routing the pump case drain flow to the hydraulic oil/fuel heat exchangers, located in the left wing fuel tank. At low temperatures, a thermally-actuated heat exchanger bypass valve diverts fluid away from the heat exchanger and back to the pump case drain flow line.

An electrically commanded firewall shutoff valve (FWSOV) stops the flow of fluid to EDP 1A during a system fluid overheat condition, or an engine fire.

Thermal fuses protect the system from excessive hydraulic temperatures. These fuses, located on the reservoir and in the return line of the pressure module, drain fluid overboard during an overheat.

The hydraulic system no. 1 service panel has pressure, suction, and fill ports. Fluid entering the hydraulic system from the service panel pressure port is filtered by the pressure filter module. Fill port fluid is filtered by the return filter module before returning to the reservoir.

Hydraulic system no. 1 can be depressurized from the manual depressurization valve that is mounted on the reservoir. A bleed relief valve on the reservoir is used for bleeding the hydraulic system.

A maintenance-free accumulator is used to assist in maintaining the hydraulic pump inlet pressure.

Hydraulic system no. 1 is assigned to the following aircraft hydraulic system users:

- Upper rudder power control unit (PCU)
- Left elevator outboard PCU
- Left thrust reverser
- Left multifunction spoiler 1 (L MFS 1)
- Left multifunction spoiler 3 (L MFS 3)
- Right multifunction spoiler 1 (R MFS 1)
- Right multifunction spoiler 3 (R MFS 3)
- Flap power drive unit (PDU)
- Left flap brake
- Right flap brake
- Left and right ground spoilers
- Landing gear

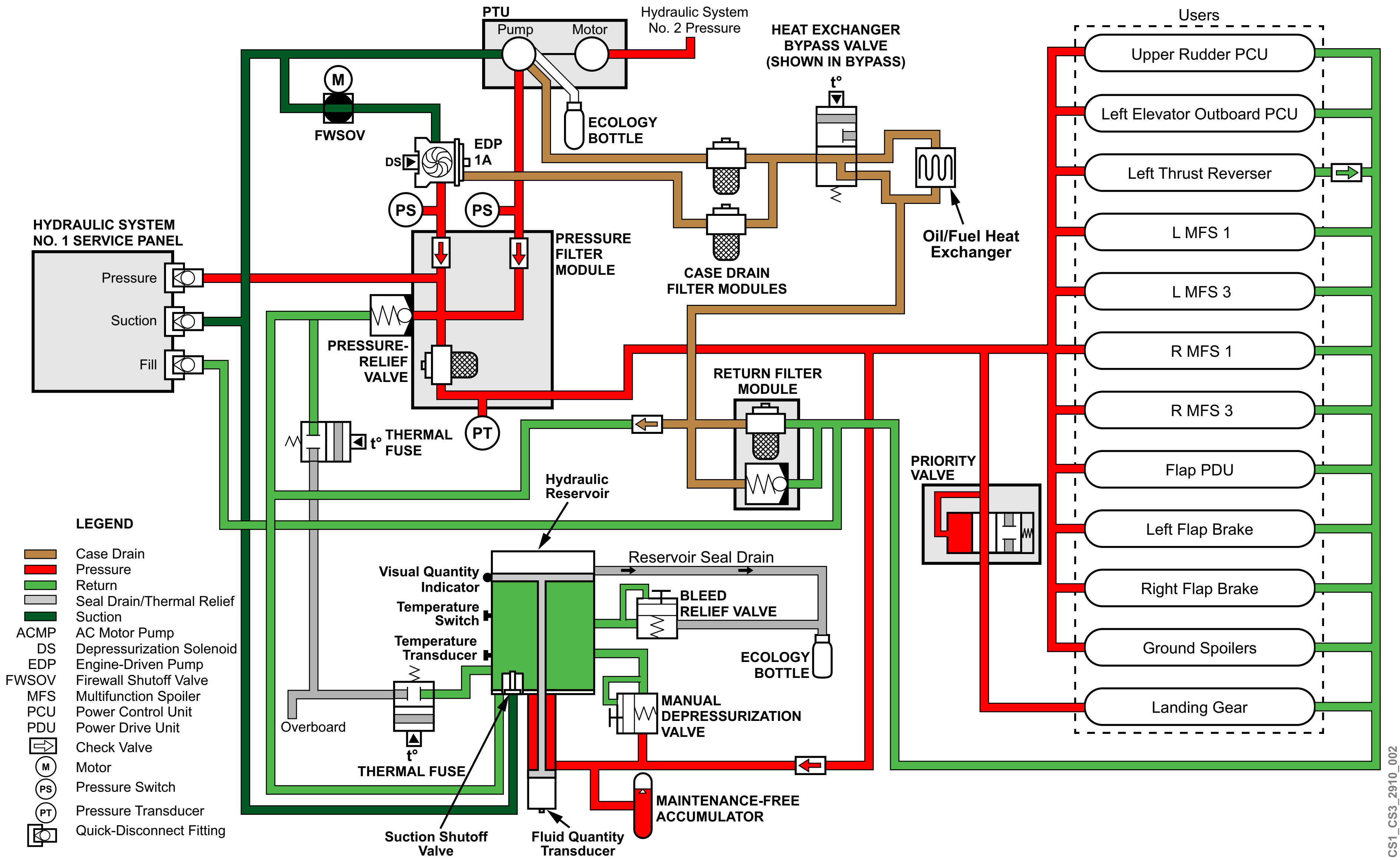


Figure 1: Hydraulic System No. 1 Schematic

COMPONENT LOCATION

Hydraulic system no. 1 consists of the following components:

- Reservoir
- Maintenance-free accumulator
- Pressure filter module
- Return filter module
- Priority valve
- Power transfer unit (PTU)
- PTU case drain filter module
- Ecology bottles
- Service panel
- Firewall shutoff valve (FWSOV) (Refer to figure 3)
- Engine-driven pump (EDP 1A) (Refer to figure 3)
- EDP case drain module (Refer to figure 3)
- Heat exchanger (Refer to figure 3)
- Heat exchanger bypass valve (Refer to figure 3)

RESERVOIR

The reservoir is located between fuselage frame FR 54 and FR 55, in the left aft portion of the wing-to-body-fairing (WTBF), behind door 195 DB.

MAINTENANCE-FREE ACCUMULATOR

The maintenance-free accumulator is mounted adjacent to the reservoir, between FR 55 and FR 56.

PRESSURE FILTER MODULE

The pressure filter module is located at FR 55, below the reservoir, behind door 195 CB.

RETURN FILTER MODULE

The return filter module is mounted below the reservoir, between FR 54 and FR 55.

PRIORITY VALVE

The priority valve is located below and inboard of the reservoir, between FR 54 and FR 55.

POWER TRANSFER UNIT

The power transfer unit (PTU) is located at FR 57, aft of the reservoir, inside the left aft portion of the WTBF.

POWER TRANSFER UNIT CASE DRAIN MODULE

The power transfer unit (PTU) case drain module is located inboard of, and below the PTU, between FR 56 and FR 57.

ECOLOGY BOTTLES

The ecology bottles are located below the reservoir, between FR 54 and FR 55.

SERVICE PANEL

The service panel is located between FR 54 and FR 55, behind door 195 CB.

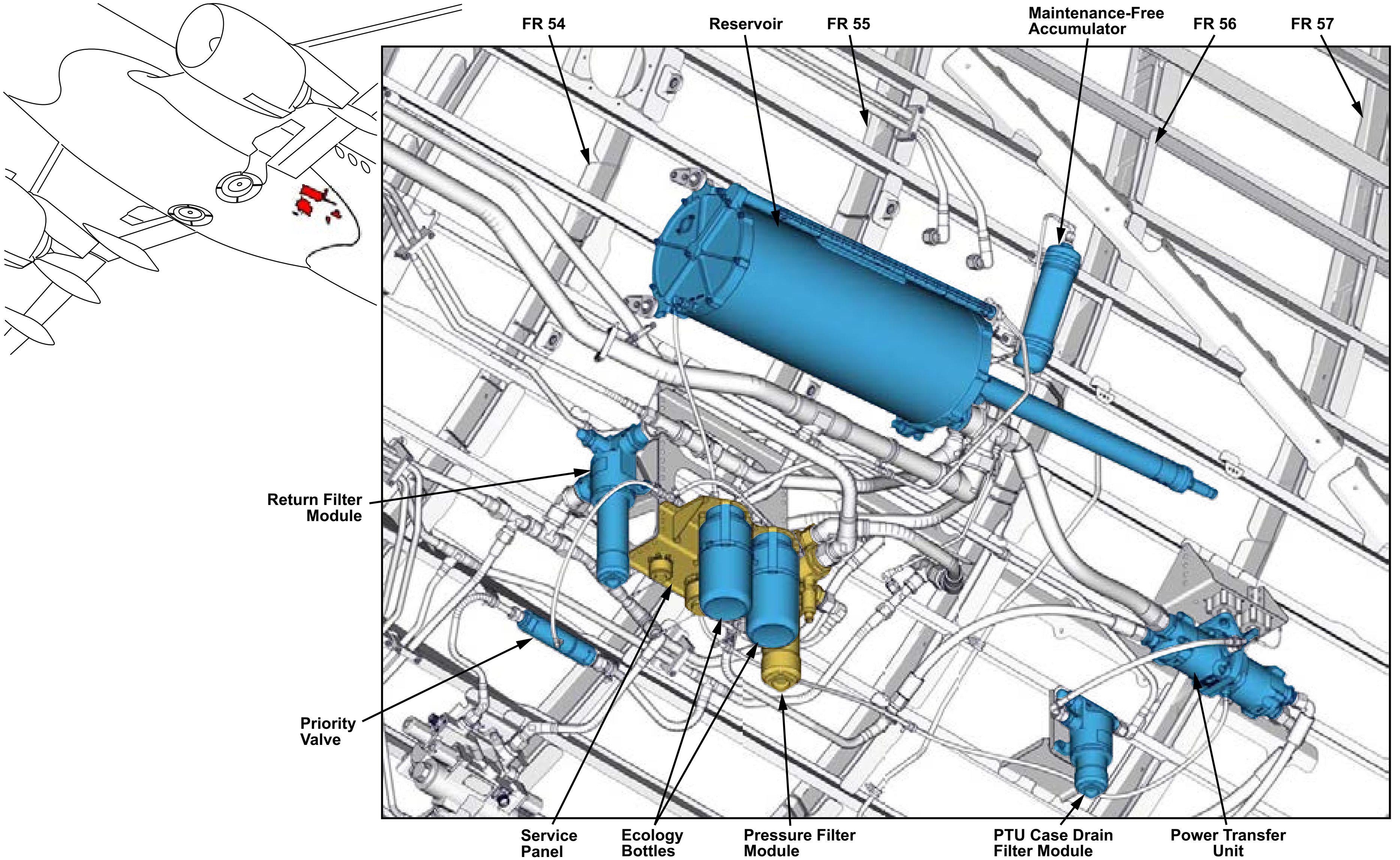


Figure 2: Hydraulic System No. 1 Hydraulic Bay Components

FIREWALL SHUTOFF VALVE

The firewall shutoff valve (FWSOV) is located at FR 48 inside the left mid portion of the WTBF, behind door 193 CB.

ENGINE-DRIVEN PUMP

The engine-driven pump (EDP 1A) is located on the left engine main gearbox assembly.

ENGINE-DRIVEN PUMP CASE DRAIN FILTER MODULE

The EDP 1A case drain filter module is located at FR 48, behind door 193 CB.

HEAT EXCHANGER

The heat exchanger is located in the left wing fuel collector tank.

HEAT EXCHANGER BYPASS VALVE

The heat exchanger bypass valve is located at FR 48, behind door 193 CB.

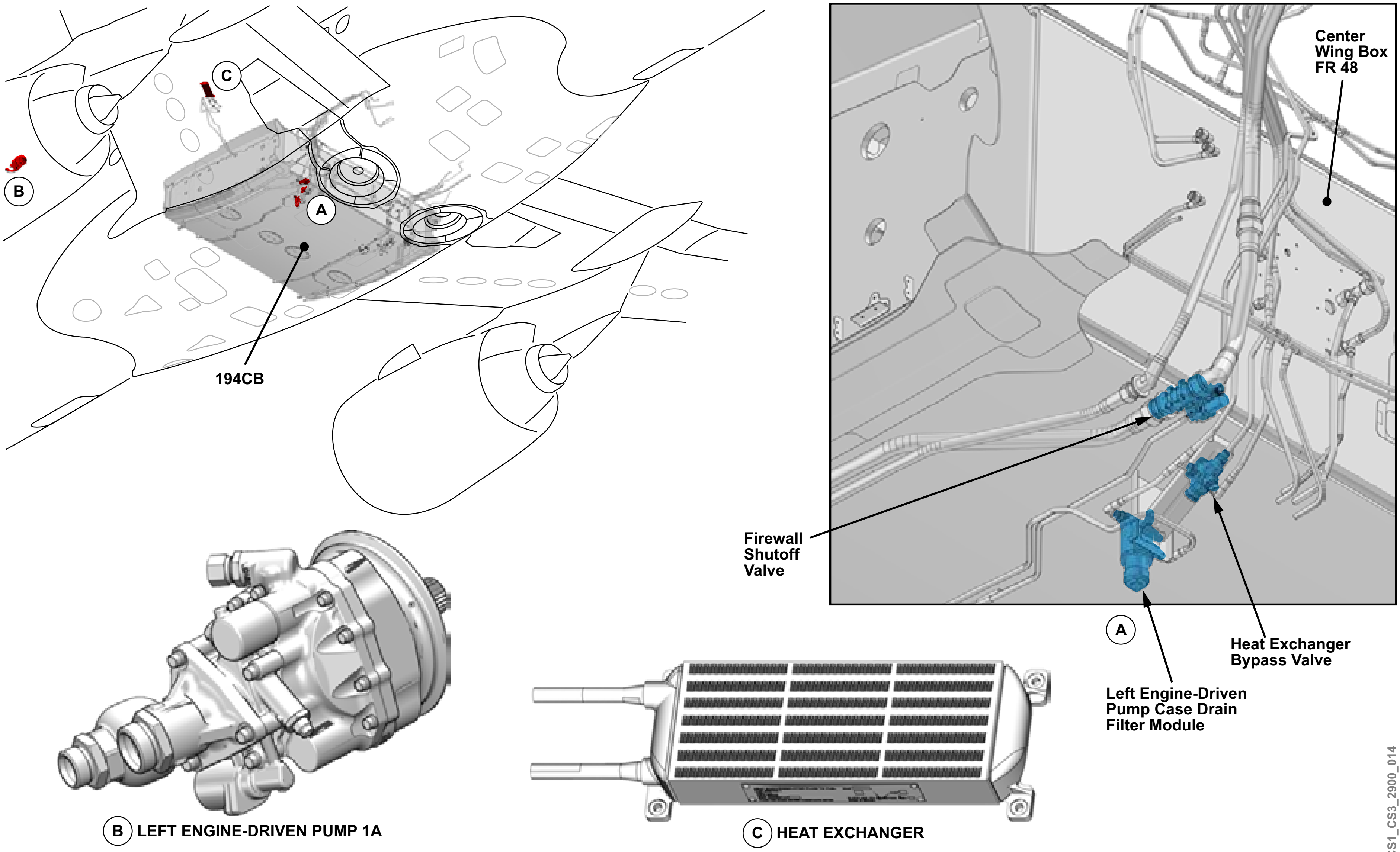


Figure 3: Hydraulic System No. 1 Components

HYDRAULIC SYSTEM NO. 2

GENERAL DESCRIPTION

Hydraulic system no. 2 contains a reservoir which provides positive pressure supply to the engine-driven pump 2A (EDP 2A), and AC motor pump 2B (ACMP 2B) during all flight phases. A temperature switch, visual fluid quantity indicator, temperature transducer, and fluid quantity transducer are provided for the reservoir.

Pressure switches, located downstream of each hydraulic pump, monitor pump output pressure. Pressurized fluid from hydraulic system no. 2 is routed through a pressure filter module, which contains a pressure-relief valve (PRV) to prevent overpressure. Two pressure switches, and a pressure transducer on the pressure filter module, transmits system hydraulic pressure signals. If system hydraulic pressure decreases, a priority valve shuts off fluid flow to the power transfer unit (PTU).

All users return fluid in the system passes through the return filter module before re-entering the reservoir. Case drain fluid from EDP 2A and ACMP 2B passes through their respective case drain filters before joining the system return flow.

Temperature extremes in the system are avoided by routing the pump case drain flow through hydraulic oil to the fuel heat exchangers, located in the right wing fuel tank. At low temperatures, a thermally-actuated heat exchanger bypass valve diverts fluid away from the heat exchanger and back to the pump case drain flow.

An electrically-commanded firewall shutoff valve (FWSOV) stops the flow of fluid to EDP 2A during a system fluid overheat condition, or an engine fire.

Thermal fuses protect the system from excessive hydraulic temperatures. These fuses, located on the reservoir and in the return line of the pressure module, drain fluid overboard during an overheat.

The for hydraulic system no. 2 service panel has pressure, suction and fill ports. Fluid entering the hydraulic system from the service panel pressure port is filtered by the pressure filter. Fill port fluid is filtered by the return filter module before returning to the reservoir.

Hydraulic system no. 2 can be depressurized from the manual depressurization valve that is mounted on the bootstrap reservoir. A bleed relief valve on the bootstrap reservoir is used for bleeding the hydraulic system. Fluid pressure from hydraulic system no. 2 drives the (PTU) motor, which mechanically drives the PTU pump in hydraulic system no. 1.

ACMP 2B and the hydraulic reservoir are each equipped with an ecology bottle. A bleed relief valve on the reservoir is used for bleeding the hydraulic system.

A maintenance-free accumulator is used to assist in maintaining the hydraulic pump inlet pressure.

Hydraulic system no. 2 is assigned to the following aircraft hydraulic system users:

- Middle rudder power control unit (PCU)
- Right elevator outboard PCU
- Right thrust reverser
- Right multifunction spoiler 4 (R MFS 4)
- Left multifunction spoiler 4 (L MFS 4)
- Left aileron outboard PCU
- Right aileron outboard PCU
- Nosewheel steering
- Slat power drive unit (PDU)
- Left slat brake
- Right slat brake
- Power transfer unit (PTU) motor

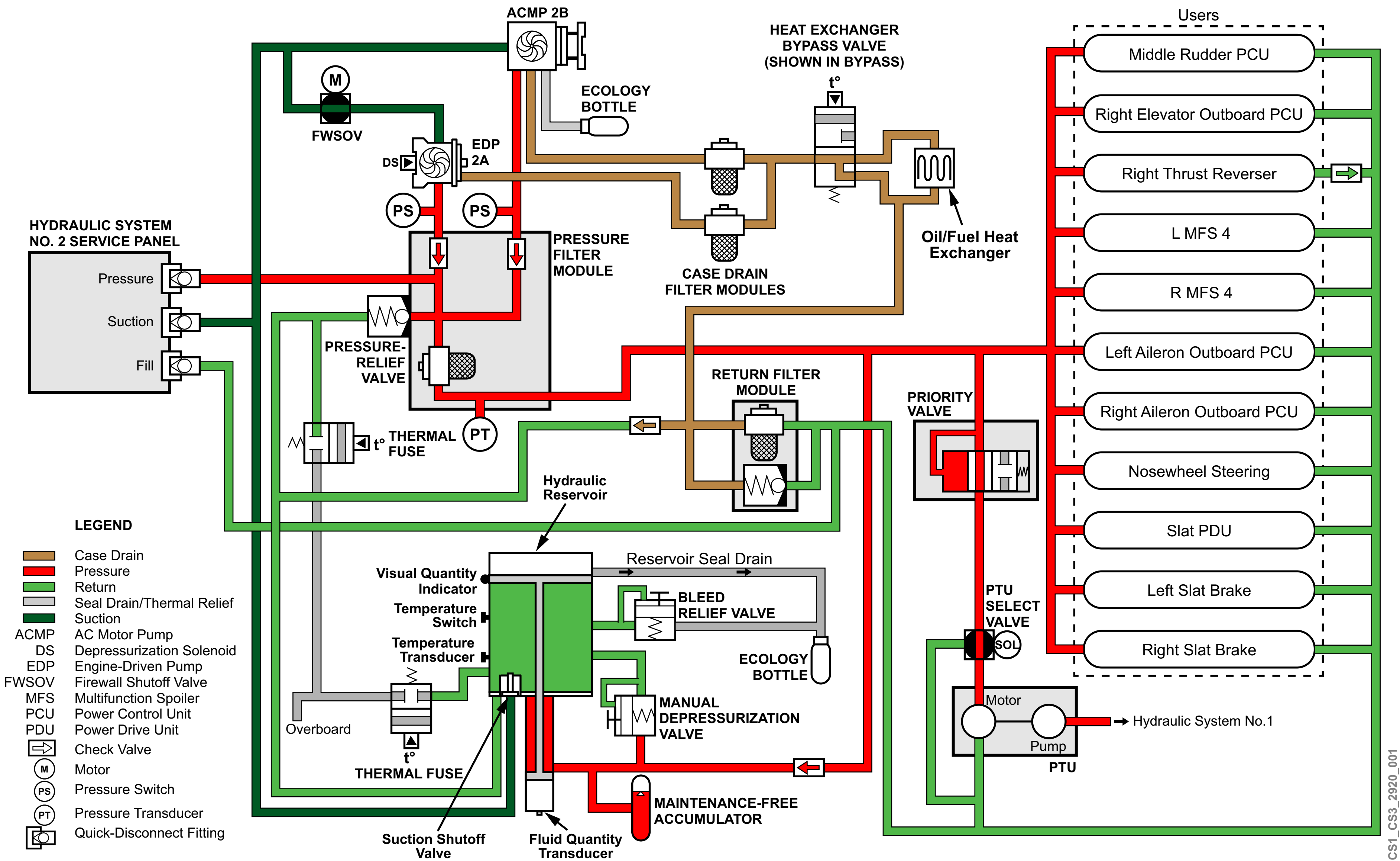


Figure 4: Hydraulic System No. 2 Schematic

COMPONENT LOCATION

Hydraulic system no. 2 consists of the following components:

- Reservoir
- Maintenance-free Accumulator
- Pressure filter module
- Return filter module
- Power transfer unit (PTU) priority valve
- AC motor pump (ACMP) 2B
- ACMP case drain filter module
- Cooling air duct
- Power transfer unit select valve
- Ecology bottles
- Service panel
- Engine-driven pump (EDP) 2A (Refer to figure 6)
- EDP case drain filter module (Refer to figure 6)
- Firewall shutoff valve (FWSOV) (Refer to figure 6)
- Heat exchanger (Refer to figure 6)
- Heat exchanger bypass valve (Refer to figure 6)

RESERVOIR

The reservoir is located between FR 54 and FR 55, behind door 196 FB, in the right aft portion of the WTBF.

MAINTENANCE-FREE ACCUMULATOR

The maintenance-free accumulator is located at FR 53, below the reservoir, inside the right aft portion of the wing-to-body fairing (WTBF).

PRESSURE FILTER MODULE

The pressure filter module is located at FR 53, below the reservoir, behind door 196 BB.

RETURN FILTER MODULE

The return filter module is located between FR 54 and FR 55, below the reservoir.

POWER TRANSFER UNIT PRIORITY VALVE

The power transfer unit (PTU) priority valve is located at FR 56, inboard of the reservoir.

AC MOTOR PUMP 2B

The ACMP 2B is located between FR 55 and FR 56, aft of the reservoir, behind door 196 GB.

AC MOTOR PUMP CASE DRAIN FILTER MODULE

The ACMP case drain filter module is located above the ACMP, inside the right aft portion of the WTBF.

COOLING AIR DUCT

The cooling air duct is attached to the motor portion of the ACMP.

POWER TRANSFER UNIT SELECT VALVE

The PTU select valve is located at FR 56, inboard of the reservoir.

ECOLOGY BOTTLES

The ecology bottles are located at FR 54, below the reservoir.

SERVICE PANEL

The service panel is located between FR 52 and FR 53, behind door 196 CB.

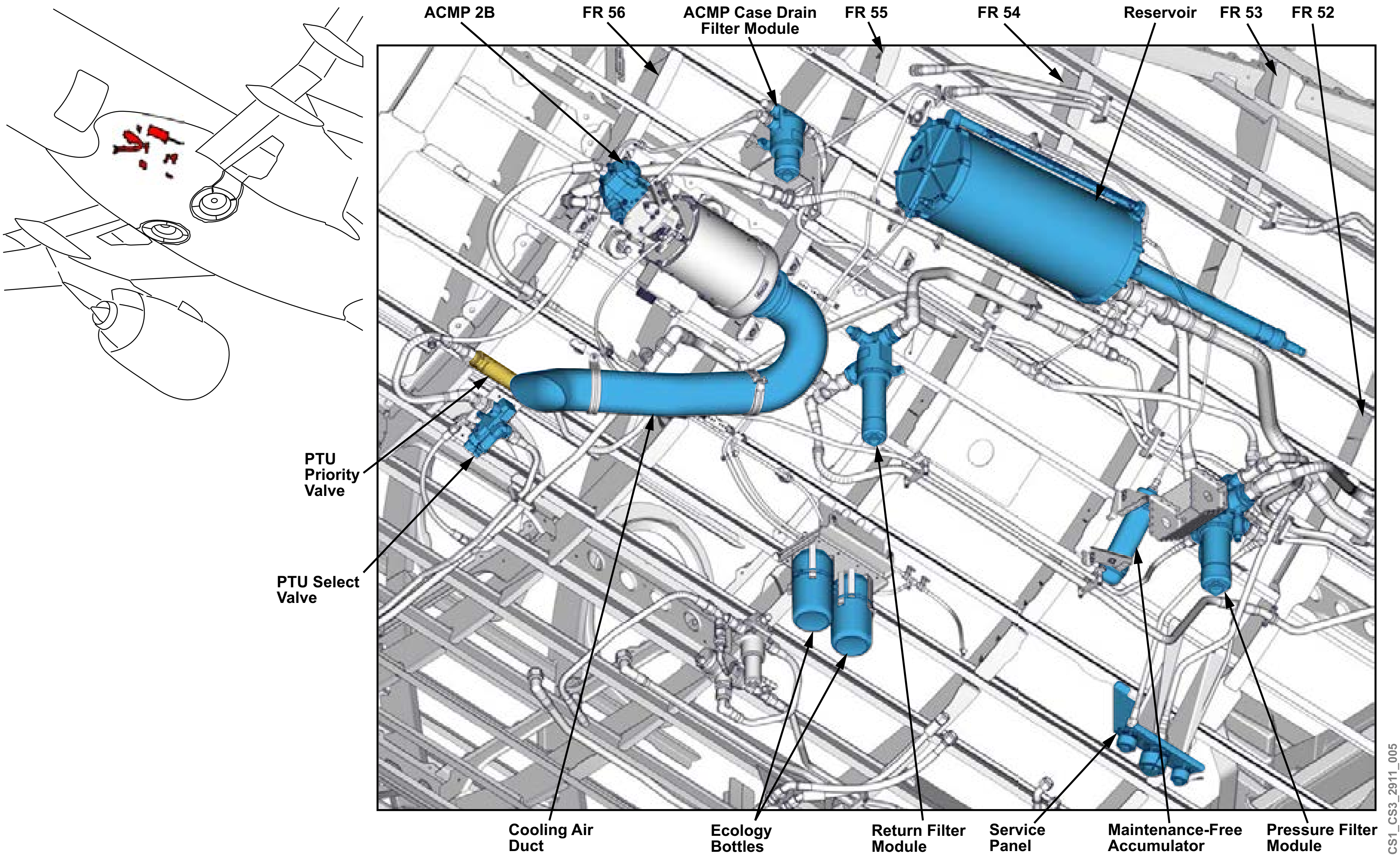


Figure 5: Hydraulic System No. 2 Hydraulic Bay Components

ENGINE-DRIVEN PUMP

The engine-driven pump (EDP) 2A is located on the right engine main gearbox assembly.

ENGINE-DRIVEN PUMP CASE DRAIN FILTER MODULE

The EDP 2A case drain module is located at FR 48, behind door 194 DB.

FIREWALL SHUTOFF VALVE

The firewall shutoff valve (FWSOV) is located at FR 48, behind door 194 DB.

HEAT EXCHANGER

The heat exchanger is located in the right fuel collector tank.

HEAT EXCHANGER BYPASS VALVE

The heat exchanger bypass valve is located at FR 48, behind door 194 DB.

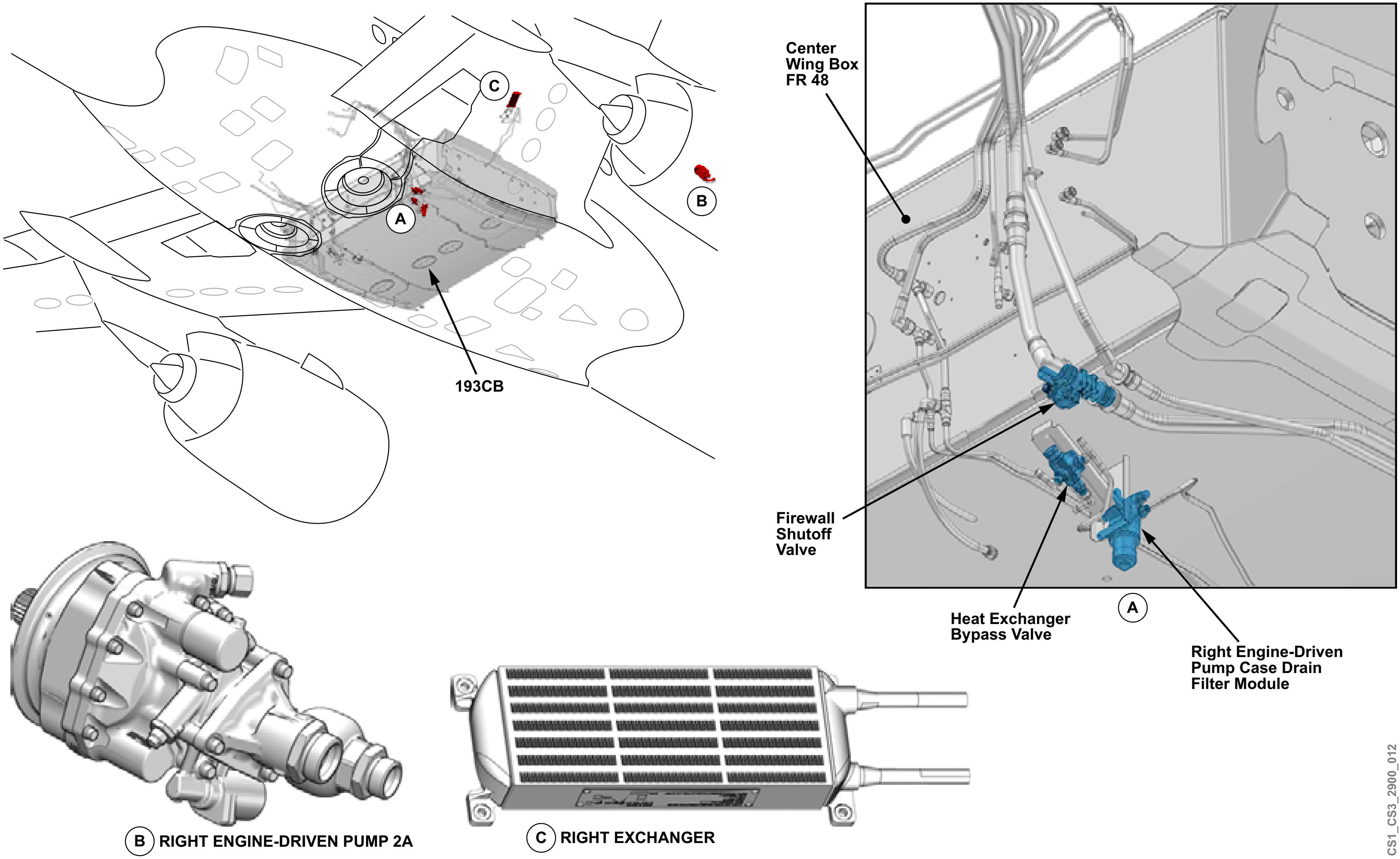


Figure 6: Hydraulic System No. 2 Components

HYDRAULIC SYSTEMS NO. 1 AND NO. 2

COMPONENT INFORMATION

The components of hydraulic system no. 1 and hydraulic system no. 2 are almost identical in appearance and operation.

RESERVOIR

The bootstrap type reservoir provides positive fluid feed to the hydraulic pumps.

- Hydraulic system no. 1 has a bootstrap reservoir with a capacity of 18,845.1 cm^3 (1150 in^3)
- Hydraulic system no. 2 has a bootstrap reservoir with a capacity of 16,387.1 cm^3 (1000 in^3)

The surface areas of the C Series bootstrap reservoir differential pistons convert 3000 psi to a pump inlet pressure of approximately 61 psi. The reservoirs have the following features:

- Visual quantity indicator
- Manual depressurization valve
- Low-pressure bleed/relief valve
- Fluid quantity transducer
- Temperature switch
- Temperature transducer
- Thermal fuse

MAINTENANCE-FREE ACCUMULATOR

The maintenance-free accumulator is installed on the high-pressure side of the reservoir in order to maintain reservoir bootstrap pressure after the hydraulic system is shut down. The accumulator requires no servicing.

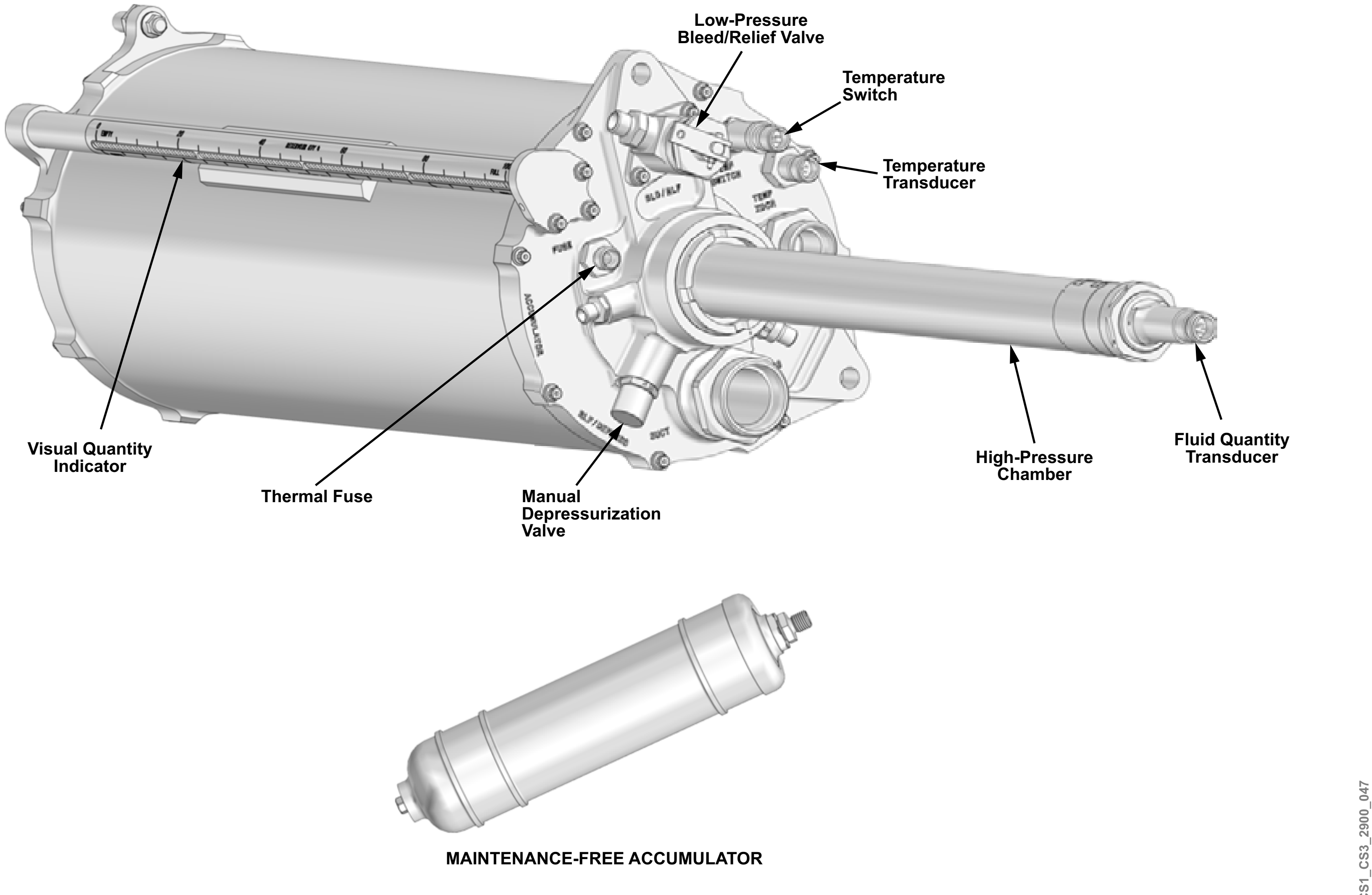


Figure 7: Bootstrap Reservoir and Maintenance-Free Accumulator

PRESSURE FILTER MODULE

Both hydraulic systems no. 1 and no. 2 have pressure filter modules. The module has a filter bowl and replaceable filter element. The filter bowl houses the element, which filters particles of 15 microns or larger. An automatic shutoff device prevents fluid loss when the filter element is removed, and limits air inclusion upon reassembly.

Fluid entering the system is filtered by the pressure filter module. A differential pressure indicator (DPI) provides visual indication to the maintenance crew that the filter element has a remaining dirt holding capacity of 30%. It must be manually reset with no flight deck indication provided.

To avoid nuisance indications, the indicator does not actuate when the hydraulic fluid is cold, or as a result of momentary flow surges. It does not actuate at fluid temperatures below 12°C (53.6°F).

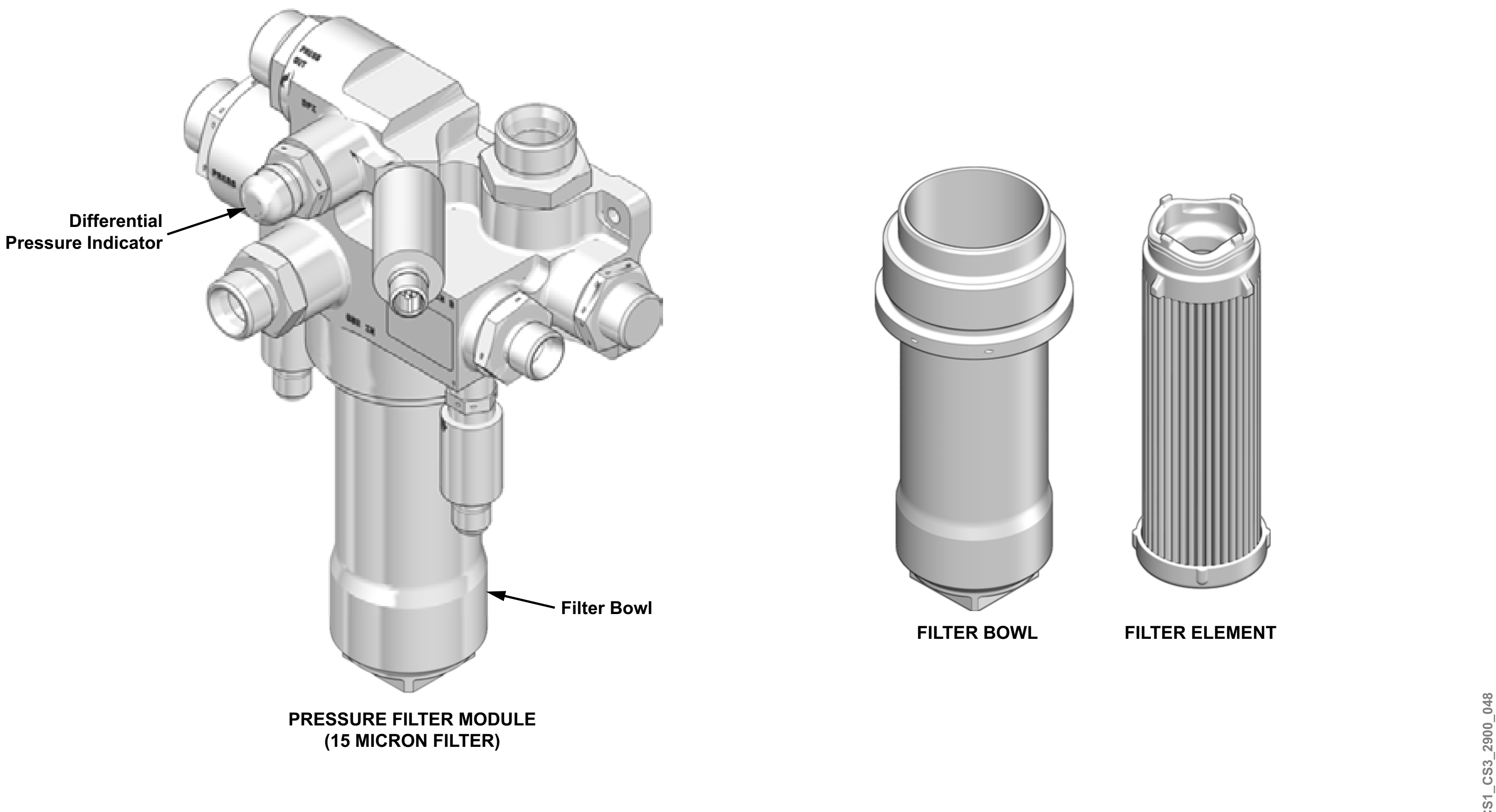


Figure 8: Pressure Filter Module Filter Element

CASE DRAIN FILTER MANIFOLD

Each hydraulic system contains a total of two case drain filter manifolds one for each system pump. These provide filtration of the pump case drain fluid before it joins the main return flow downstream of the return filter.

The module also houses a filter differential pressure indicator (DPI). The DPI is used to provide a visual indication of the filter element condition to the maintenance personnel. The DPI extends whenever the filter element has reached a level of approximately 70% blocked.

The filter element is contained in the bowl of the assembly and utilizes an automatic shutoff device to prevent leakage when the filter bowl and element are removed for maintenance. The filter contains a check valve in the outlet port to prevent backflow through the filter.

RETURN FILTER MODULE

The return filter module filters the return line fluid coming from the users before it re-enters the system reservoir. It also filters any incoming fluid from the ground service fill port. It consists of a 5 micron filter element, a filter element DPI, an automatic shutoff device for element maintenance, and a bypass relief valve to prevent restricted system return flow.

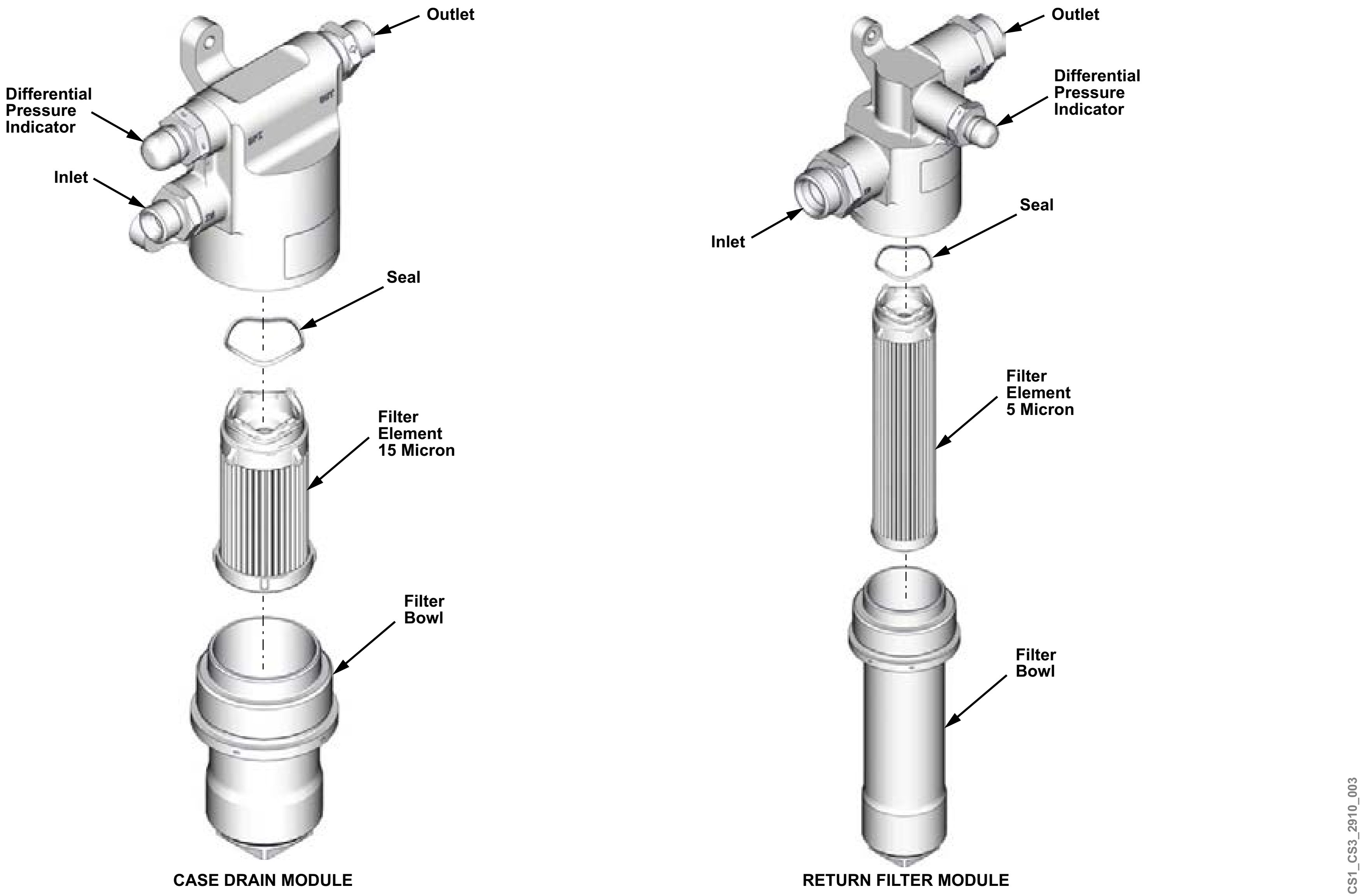


Figure 9: Case Drain Filter Manifold and Return Filter Module

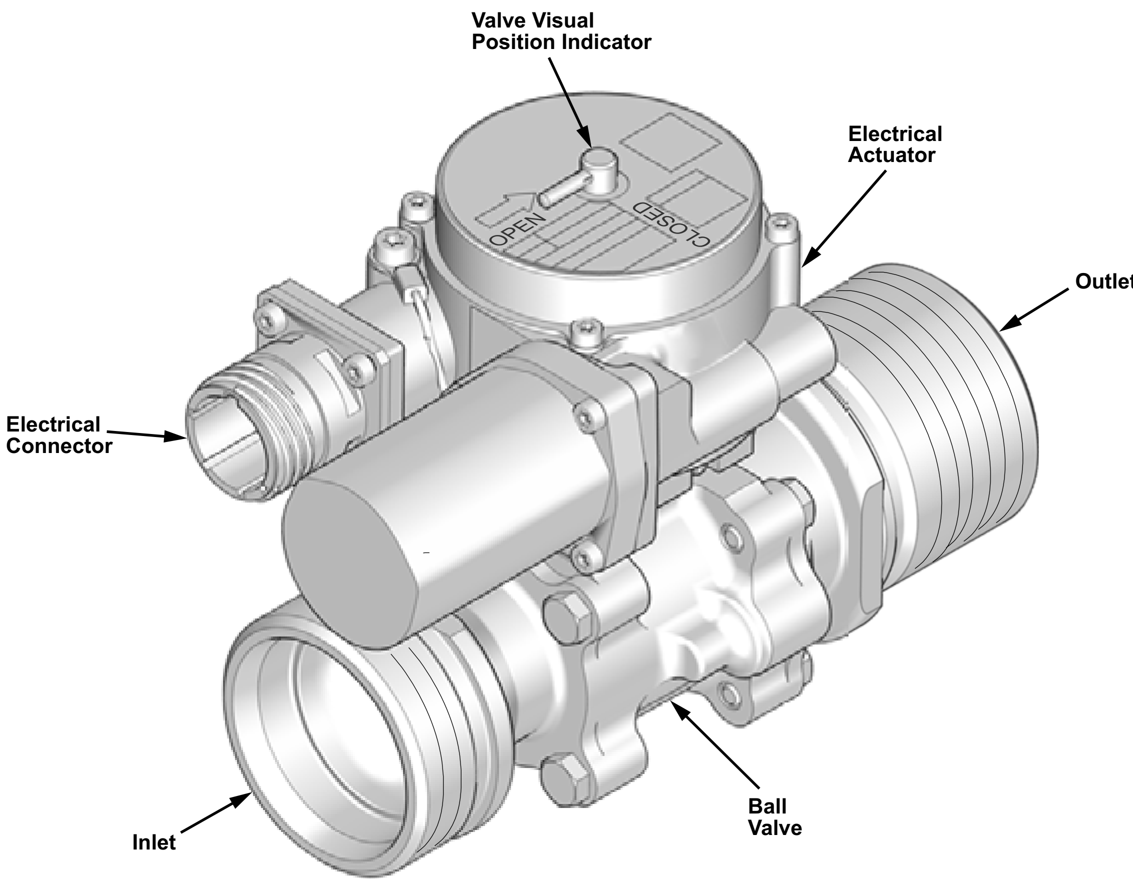
FIREWALL SHUTOFF VALVE

Both hydraulic systems no. 1 and no. 2 contain a 28 VDC operated firewall shutoff valve (FWSOV). The valves are located in the suction line of each EDP, and are housed at wing center box FR 48, behind door 193 CB (no. 1) and 194 DB (no. 2).

The FWSOV consists of an electrical actuator and a ball valve. The electrical actuator and ball valve can be replaced independently. A visual position indicator on the valve indicates valve position.

During an engine fire, this valve is closed by actuating the respective ENG FIRE switch on the flight deck overhead panel. The FWSOV can also be closed manually by actuating the respective HYD SOV switch on the flight deck overhead panel. In the event of hydraulic fluid overtemperature, the FWSOV closes automatically via the reservoir temperature switch.

Each FWSOV is equipped with an inlet and outlet port, and an electrical connector. When the valve is in the closed position, a built-in thermal relief valve provides protection from downstream thermal expansion overpressure. The valve position is transmitted to the hydraulic synoptic page through position switches within the actuator. A closed indication (CLSD) illuminates on the HYD SOV flight deck switch when the valve is closed.



NOTE

Left FWSOV shown.
Right FWSOV similar.

Figure 10: Firewall Shutoff Valve

DETAILED COMPONENT INFORMATION

RESERVOIR

Hydraulic systems no. 1 and no. 2 reservoirs are divided into high-pressure and low-pressure chambers. A seal drain at the low-pressure chamber routes piston seal leakage to an ecology bottle. The reservoir manual depressurization valve limits pressure within the reservoir high-pressure chamber. When manually actuated, it dumps system high-pressure into the reservoir low-pressure chamber.

The reservoir bleed/relief valve feature limits pressure within the low-pressure chamber. The relief valve has a cracking pressure of 145 psi, and resets at 90 psi. This valve, when manually actuated, bleeds fluid pressure from the low-pressure side of the reservoir and into the reservoir ecology bottle.

When low-pressure chamber piston (A1) contacts the suction shutoff valve seal area, it closes hydraulic fluid supply flow to the hydraulic pumps. If a case drain line breaks in the fuel tank, closing the suction shutoff valve prevents the hydraulic pump from running dry. Fuel ingest into the hydraulic system is a fire hazard.

A bootstrap reservoir uses a dual-piston where high-pressure hydraulic from the hydraulic pump outlet is applied to the small area of the piston (A2). This produces a low-pressure on the reservoir side of the piston. A major advantage of bootstrap reservoirs is that reservoir pressurization is maintained during aggressive flight maneuvers, including negative-G flight.

The high-pressure line to the reservoir includes a check valve. The check valve maintains reservoir pressure after the pump has been shut down, and hydraulic pump inlet pressure when the engine-driven pump (EDP) is not rotating.

The high-pressure chamber piston (A2) drives the low-pressure piston (A1) to contact the seal area and close the suction shutoff valve. The closed valve deprives the pump(s) of fluid, causing them to vapor lock and stop pumping.

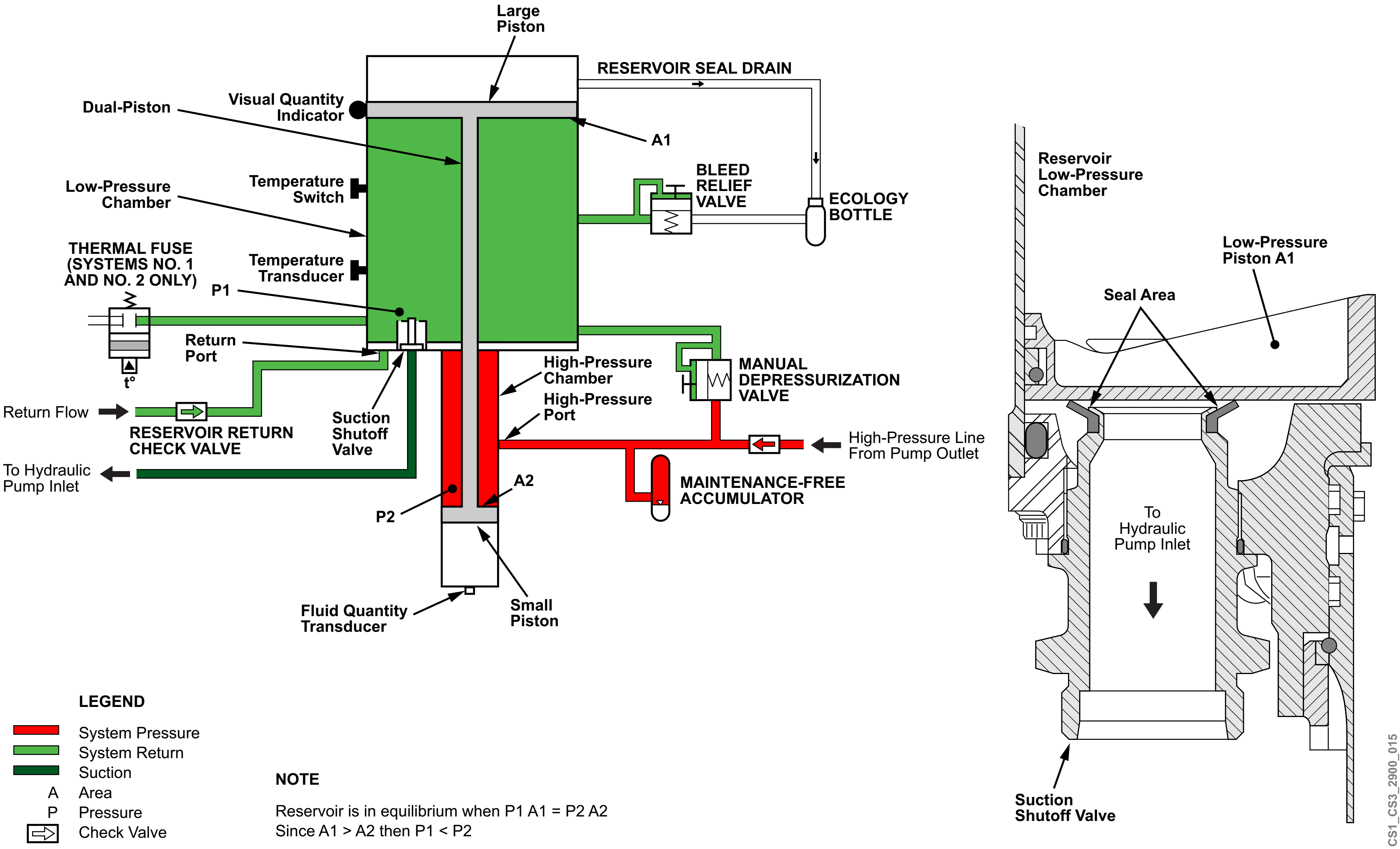
The reservoir is equipped with a return port and a high-pressure port. A fluid quantity transducer provides electronic fluid quantity indication on the hydraulic synoptic page in the flight deck. The reservoir also includes visual quantity indication for maintenance personnel. The visual indicator provides an indication of reservoir fluid level through a cable-driven slider linked to a piston position. A temperature transducer, mounted to the reservoir module, senses the temperature of the fluid in the low-pressure chamber.

A reservoir return check valve prevents fluid backflow to the return filter module when the hydraulic system is not operating.

A temperature transducer monitors the reservoir. If the temperature reaches 107°C (224.6°F) an the engine indication and crew alerting system (EICAS) HYD 1 (2) HI TEMP caution message is displayed. When the hydraulic fluid temperature exceeds 125°C (257°F), the temperature switch closes the hydraulic firewall shutoff valve (FWSOV). This prevents hydraulic fluid flow to the inlet of the engine-driven pump (EDP), disabling its operation without requiring the engine to be shut down.

A eutectic fuse, also referred to as a thermal fuse, is mounted on each system reservoir in the low-pressure chamber. The fuse is designed to melt at 144°C (291.2°F), and drains hydraulic fluid overboard.

A maintenance-free accumulator in hydraulic systems no. 1 and no. 2 is connected to its respective reservoir high-pressure chamber, and continues to pressurize the system after the hydraulic pumps have been shut down. The accumulator is precharged with helium and is maintenance free.



NOTE

Reservoir is in equilibrium when P1 A1 = P2 A2
Since A1 > A2 then P1 < P2

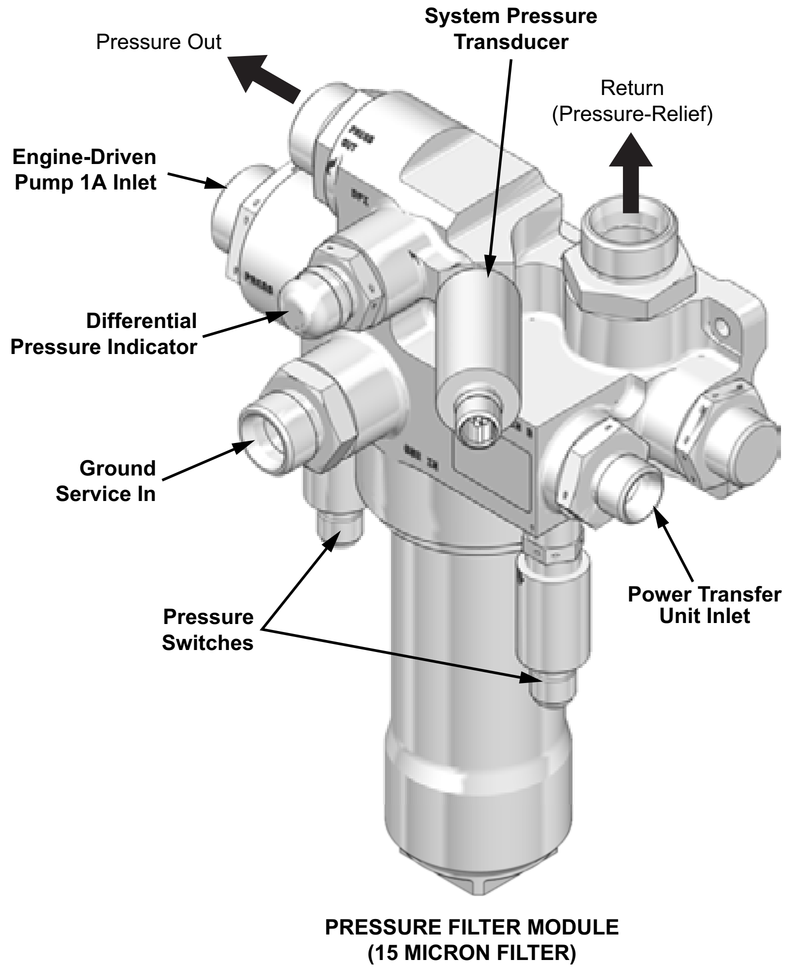
Figure 11: Hydraulic Systems No. 1 and No. 2 Reservoirs

PRESSURE FILTER MODULE

The pressure filter module contains two internal check valves, one in each pump inlet port. The check valves prevent the respective pumps from back driving the other. An internal pressure-relief valve in the filter module prevents system overpressure. The valve opens at 3600 psi and closes at 3250 psi.

The filter module also contains two pressure switches and a system pressure transducer. The pressure switches activate at 2400 psi and deactivate at 1800 psi.

One pressure switch on hydraulic system no. 1 pressure filter module monitors engine-driven pump (EDP) 1A, and the other pressure switch monitors the power transfer unit (PTU). One pressure switch on hydraulic system no. 2 pressure filter module monitors EDP 2A, and the other pressure switch monitors AC motor pump (ACMP) 2B.



NOTE

Hydraulic system no. 1 pressure filter module shown. Hydraulic systems no. 2 and no. 3 pressure filter modules similar.

Figure 12: Pressure Filter Module

ENGINE-DRIVEN PUMP

The engine-driven pump (EDP) is a variable displacement pressure regulating pump, which provides 3000 psi under varying loads. The EDP is connected by the drive shaft to the engine accessory gearbox, and operates when the engine is running. The drive shaft is designed to shear during pump seizure.

An attenuator at the pressure port reduces downstream fluid pressure pulsations that affect EDP performance. The EDP is provided with quick-disconnect fittings at the case drain port, suction port, and pressure port, and an electrical connector for the solenoid operated depressurization valve.

Due to restricted access, the EDP is equipped with a pre-attached hose at the case drain port. This hose is considered part of the pump. To avoid damage, it must not be used to support the pump during maintenance or handling.

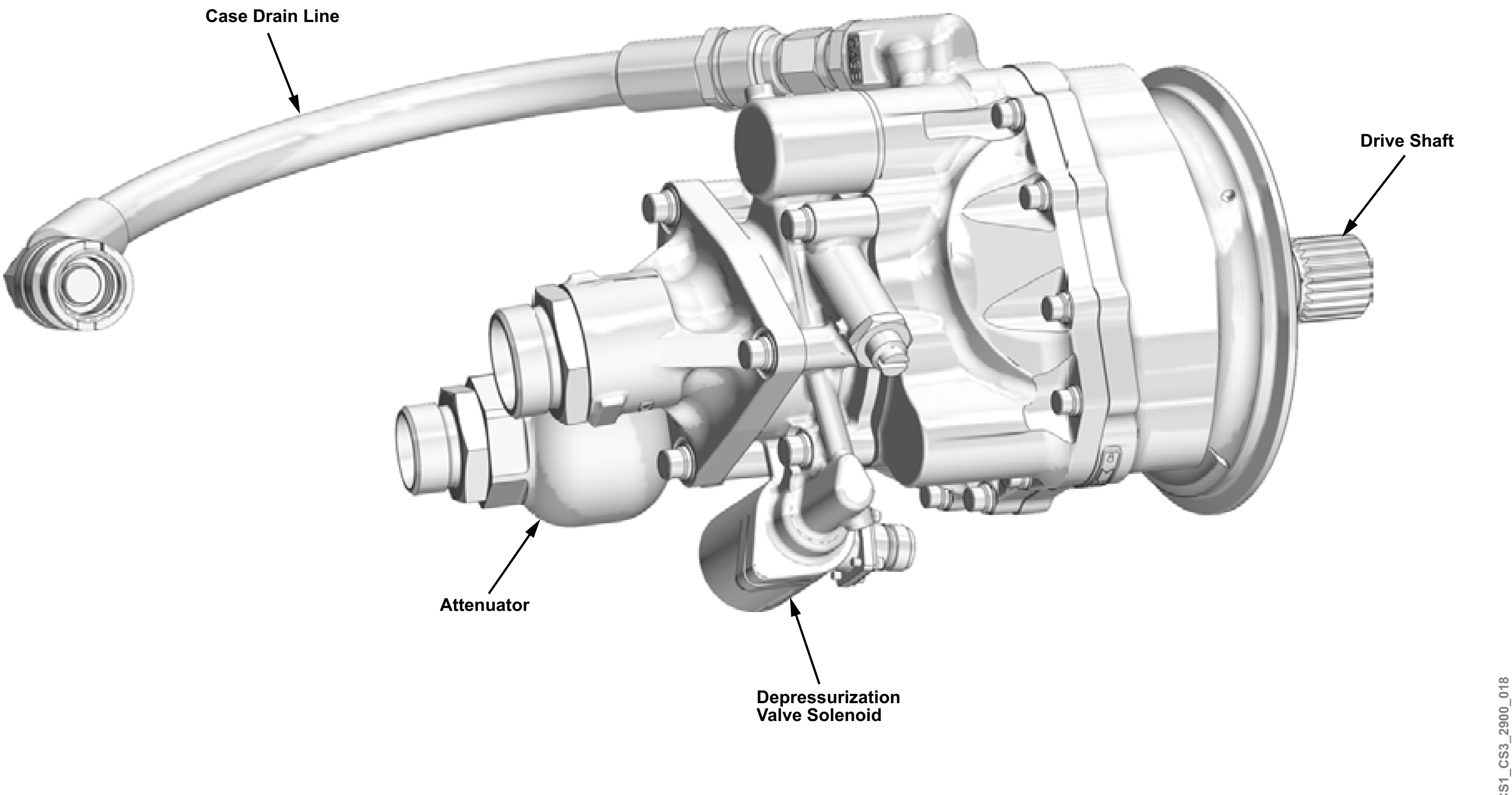


Figure 13: Engine-Driven Pump

POWER TRANSFER UNIT

The power transfer unit (PTU) is a fixed displacement, 37.8 L/m (10 gal/m), unidirectional pump.

The motor section of the PTU drives the pump section through a drive shaft. The motor section, pump section, and drive shaft are enclosed in one unit.

The pump section of the PTU is connected to hydraulic system no. 1 through the pump inlet port, pump discharge port, and pump case drain port. The motor section of the PTU is connected to hydraulic system no. 2 through the motor supply port and the motor return port.

When the motor section receives fluid flow, the speed of the unit varies in proportion to the demand on hydraulic system no. 1. The motor section is slightly larger in displacement than the pump side, which makes up for mechanical losses within the unit, while maintaining system pressure.

Fluid leakage through the seal drain port is directed by a hose to an ecology bottle.

The PTU is bolted on shock mounts to the fuselage attachment in order to reduce noise and vibration.

POWER TRANSFER UNIT SELECT VALVE

The power transfer unit (PTU) select valve controls the PTU motor operation. It is a 28 VDC solenoid operated three-way, two-position valve. The valve is normally commanded closed by the solenoid and pressure operated to the open position.

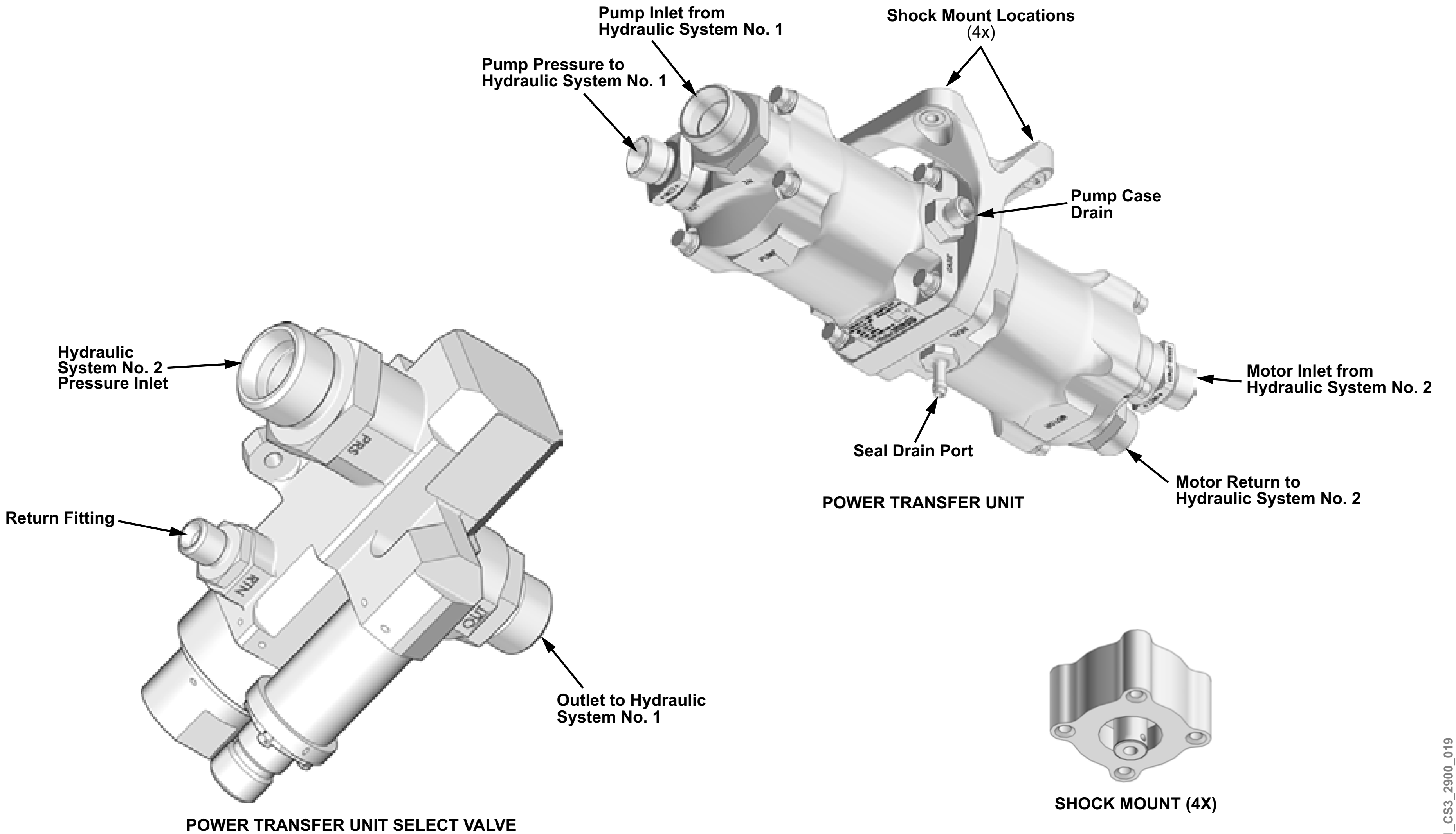


Figure 14: Power Transfer Unit and Power Transfer Unit Select Valve

AC MOTOR PUMPS

The AC motor pumps (ACMPs) are pressure-compensated type pumps, which vary the amount of fluid delivered to maintain the desired system pressure. The operation of the motor-driven hydraulic pump is similar to the EDPs. The ACMP is a variable displacement axial piston type unit. It includes a secondary fixed displacement pump to generate case drain flow for cooling the pump and motor. A pressure-relief valve protects the pump case in the event of line blockage. Pressure compensation varies displacement to maintain the desired discharge pressure and meet system flow demand.

The ACMP installation uses shock mounts to reduce vibration and noise within the passenger cabin. The ACMP contains a 115/200 VAC, 3-phase, 380-702 Hz variable frequency motor. The motor assembly incorporates a resistance temperature device (RTD). This provides data required for the control logic to protect the ACMP windings from overtemperature damage during normal aircraft operation. A cooling duct is attached to the ACMP, and vents hot air overboard through a duct flush to the fuselage.

The unit has two electrical connectors. Electrical power for the motor is transmitted through one electrical connector (J1), while a second connector is used for the winding temperature signal (J2). The outlet, suction, and case drain ports fittings are connected directly to the aircraft hydraulic system. A seal drain port fitting is connected to the aircraft ecology bottle through plastic tubing.

The ACMP in hydraulic system no. 2 is identical to the ACMPs in hydraulic system no. 3. Only the mounting configurations are different.

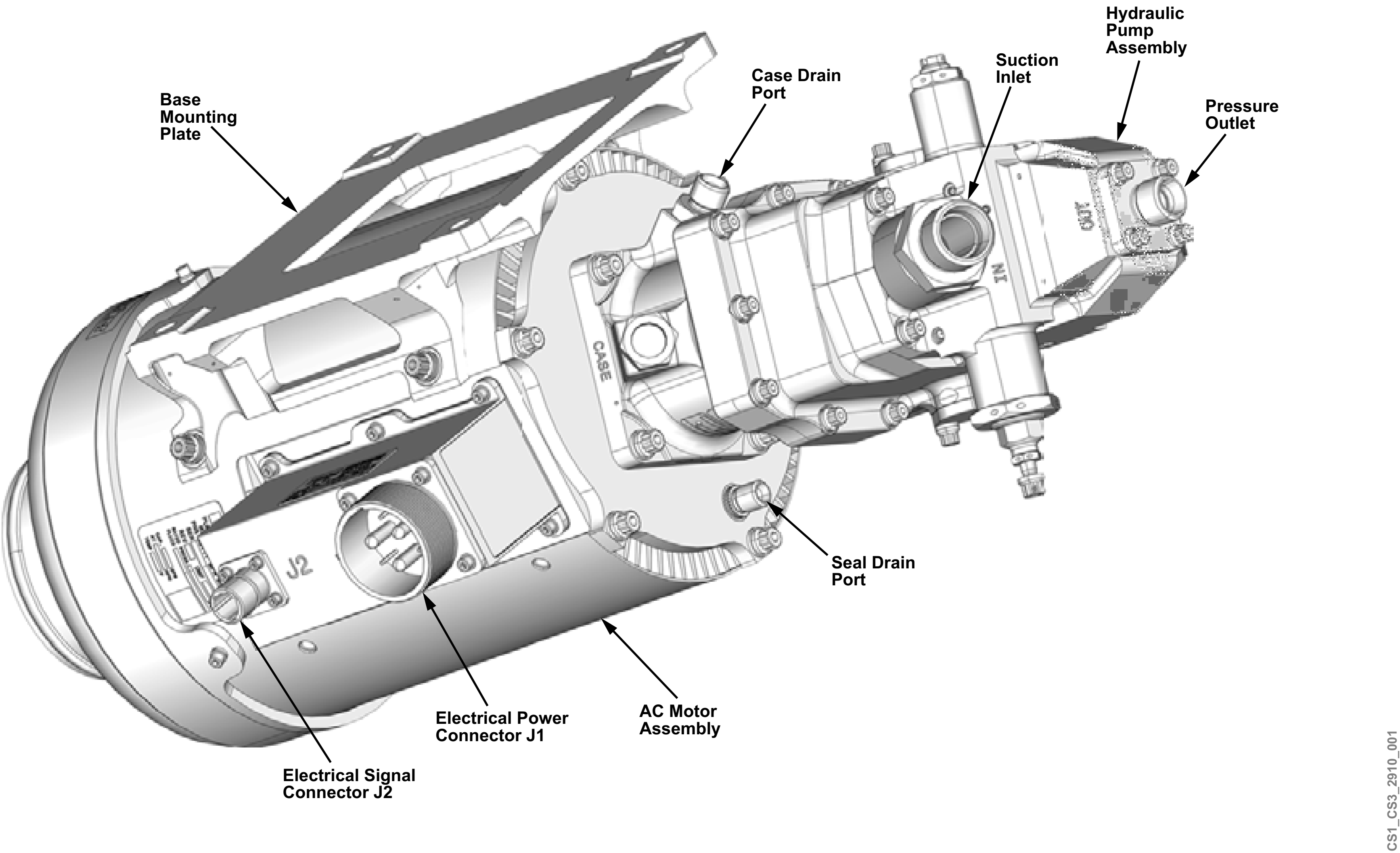


Figure 15: AC Motor Pumps

29-12 HYDRAULIC SYSTEM NO. 3

GENERAL DESCRIPTION

Hydraulic system no. 3 is pressurized by two AC motor pumps (ACMPs) or the ram air turbine (RAT). The ACMP 3A serves as the primary pump and is backed up by ACMP 3B.

Except for the horizontal stabilizer trim assembly and ground spoilers, the remaining flight controls receive power from hydraulic system no. 3 to ensure control in all three flight axes.

A priority valve ensures that hydraulic pressure is always available to the primary flight controls. When the pressure drops during periods of high demand, the valve closes to stop hydraulic flow to the the flaps and the slat power drive unit.

The ram air turbine (RAT), when deployed, provides hydraulic pressure to aircraft systems. A damped check valve and warming circuit check valve are provided downstream of the RAT.

Two nitrogen-charged accumulators in the system are used following a dual-engine failure. They provide flow to hydraulic system no. 3 during the transition to RAT operation, and dampen pressure transients caused by rapid movement of hydraulic users. Each accumulator has a pressure gauge and a charging valve on the service panel. The accumulator restrictor check valve, located on top of each accumulator, allows unrestricted flow of hydraulic fluid pressure out of the accumulator to hydraulic system no. 3 users during transient demand. It restricts hydraulic pressure flow into the accumulator, allowing the hydraulic system to be pressurized rapidly without immediately charging the accumulators.

Pressure transducers (PTs) are used to monitor and indicate the pressure in hydraulic system no. 3. They are installed in the nitrogen precharge circuit of the accumulators.

The reservoir is similar to the reservoirs of hydraulic systems no. 1 and no. 2, however it has a capacity of 14,748.36 cm3 (900 in.3), and provides positive inlet pressure to the ACMPs and the RAT. The reservoir is equipped with a temperature switch, visual quantity indicator, temperature transducer, bleed relief valve, manual depressurization valve, fluid quantity transducer, ecology bottle, and a maintenance-free accumulator.

Hydraulic system no. 3 is equipped with a RAT pressure filter module, RAT ecology bottle, a case drain filter module for each ACMP, a pressure filter module, pressure-relief valve, pressure transducer, and a return filter module. Check valves are located in various lines in the system to direct flow and prevent reverse flow. A pressure switch is located downstream of each ACMP. One ecology bottle serves both ACMPs.

Fluid in hydraulic system no. 3 is cooled by convection and radiation. The hydraulic system no. 3 service panel has pressure, suction, fill ports, accumulator charging valves, and pressure gauges.

Hydraulic system no. 3 is assigned to the following aircraft system users:

- Lower rudder power control unit (PCU)
- Left and right elevator inboard PCUs
- Left and right multifunction spoilers 2 (L MFS 2/R MFS 2)
- Left and right aileron inboard PCUs
- Left and right slat brakes
- Left and right flap brakes
- Ram air turbine (RAT) stow actuator
- Slat power drive unit (PDU)
- Flap PDU

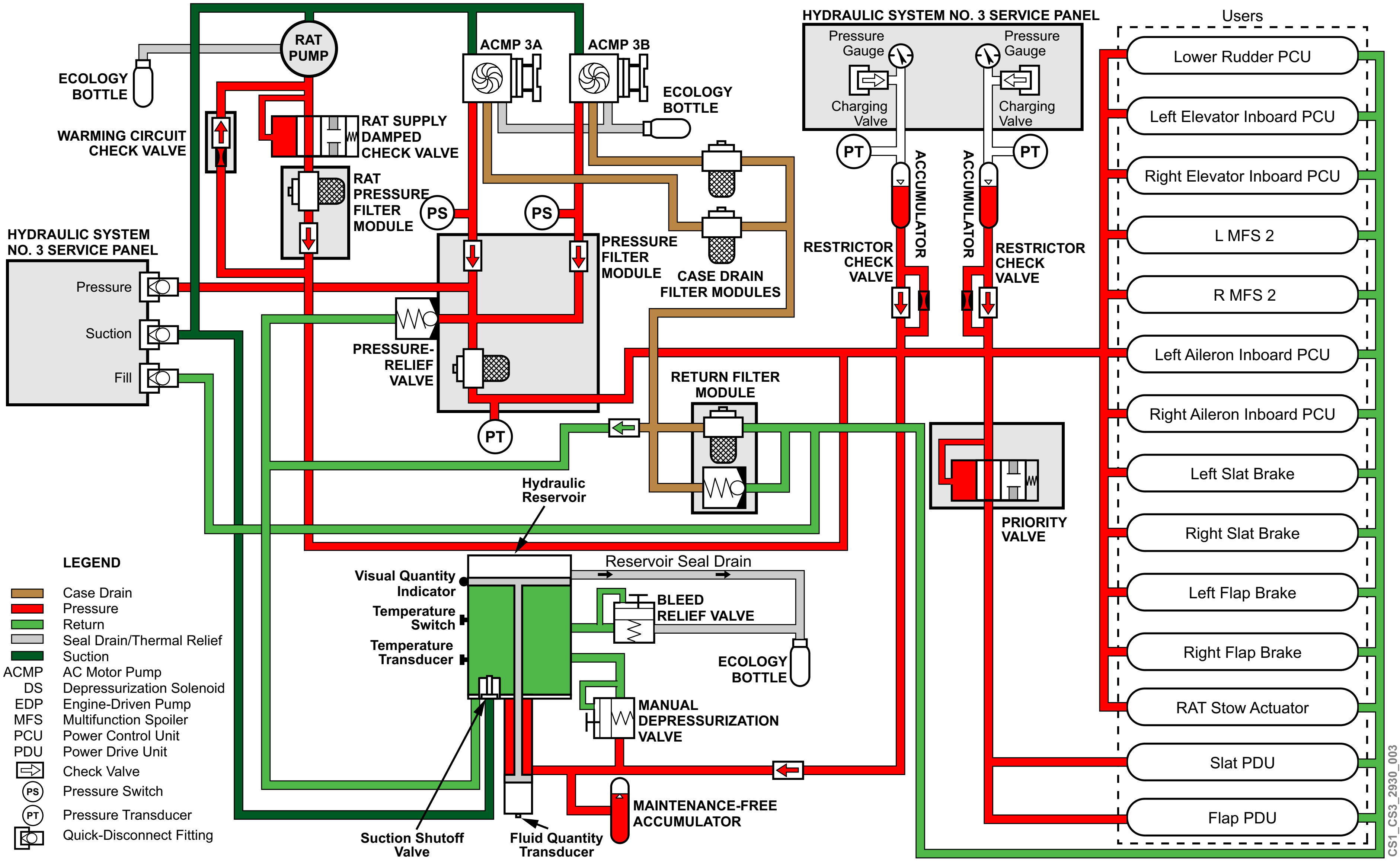


Figure 16: Hydraulic System No. 3 Schematic

COMPONENT LOCATION

The following hydraulic system no. 3 components are located on a rack in the aft equipment bay and accessed through the aft equipment bay door.

- Reservoir
- Maintenance-free Accumulator
- Pressure filter module
- Return filter module
- Case drain filter module
- Ecology bottles
- AC motor pumps (ACMPs)
- Nitrogen accumulators
- Service panel

The following components are located in the wing-to-body fairing:

- Ram air turbine (RAT) (Refer to figure 18)
- RAT hydraulic pump (Refer to figure 18)
- RAT pressure filter (Refer to figure 18)
- RAT priority valve (Refer to figure 18)
- RAT ecology bottle (Refer to figure 18)

RESERVOIR

The reservoir is located between FR 81 and FR 82, on the right side of the aft equipment bay.

MAINTENANCE-FREE ACCUMULATOR

The maintenance-free accumulator is located at FR 82, inboard and aft of the reservoir.

PRESSURE FILTER MODULE

The pressure filter module is located at FR 81, below the reservoir.

RETURN FILTER MODULE

The return filter module is located at FR 81, inboard of and below the reservoir.

CASE DRAIN FILTER MODULE

One of the case drain filter modules is located at FR 81 below the pressure filter module, while the other one is located at FR82 just forward of the maintenance-free accumulator.

ECOLOGY BOTTLES

The ecology bottles are located between FR 82 and FR 83, below and aft of the reservoir.

AC MOTOR PUMPS

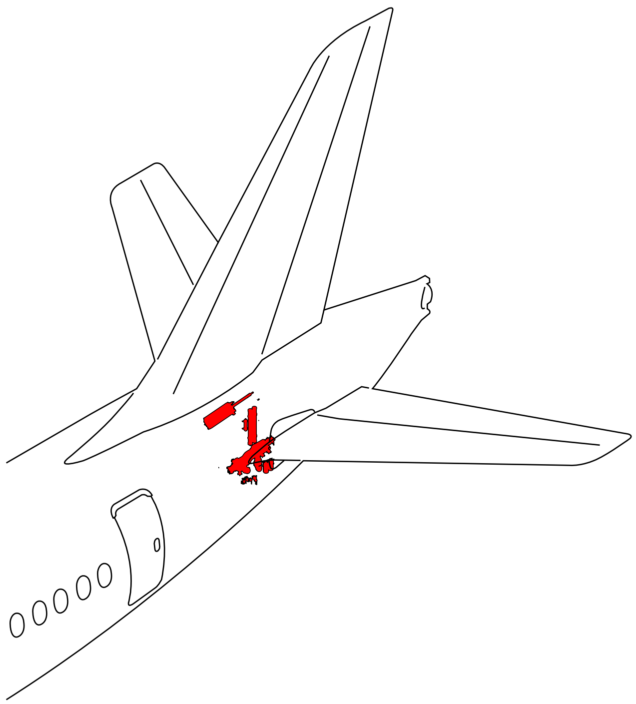
The ACMPs are located between FR 81 and FR 83, on the right side of the equipment bay.

NITROGEN ACCUMULATORS

The nitrogen accumulators are mounted vertically between FR 82 and FR 83, on the right side of the equipment bay.

SERVICE PANEL

The service panel is located between FR 81 and FR 82, in the inboard vertical panel of the hydraulic component rack, on the right side of the equipment bay.



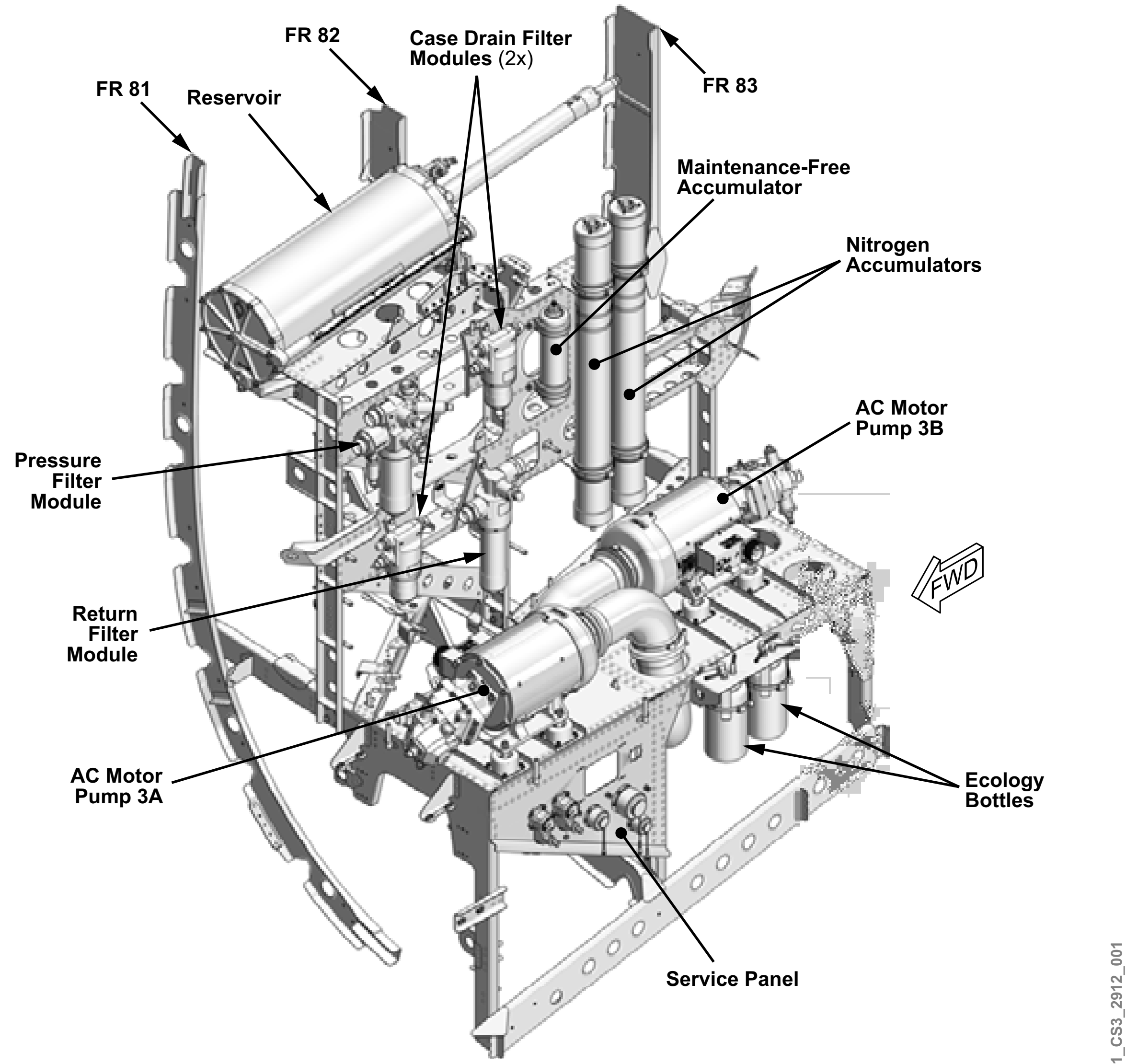


Figure 17: Hydraulic System No. 3 Components

RAM AIR TURBINE

The ram air turbine (RAT) is attached to pivot points at FR 52. When stowed, it folds forward and lies horizontally between FR 54 and FR 51, in the lower portion of the wing-to-body fairing (WTBF).

RAM AIR TURBINE HYDRAULIC PUMP

The RAT hydraulic pump is mounted at the upper (aft) end of the RAT. When the RAT is stowed, the pump is located at FR 53.

RAM AIR TURBINE PRESSURE FILTER

The RAT pressure filter is located between FR 54 and FR 55, in the WTBF.

RAM AIR TURBINE PRIORITY VALVE

The RAT priority valve is located between FR 53 and FR 54, mounted parallel to the stowed RAT.

RAM AIR TURBINE ECOLOGY BOTTLE

The RAT ecology bottle is located directly below the RAT hydraulic pump (with RAT stowed).

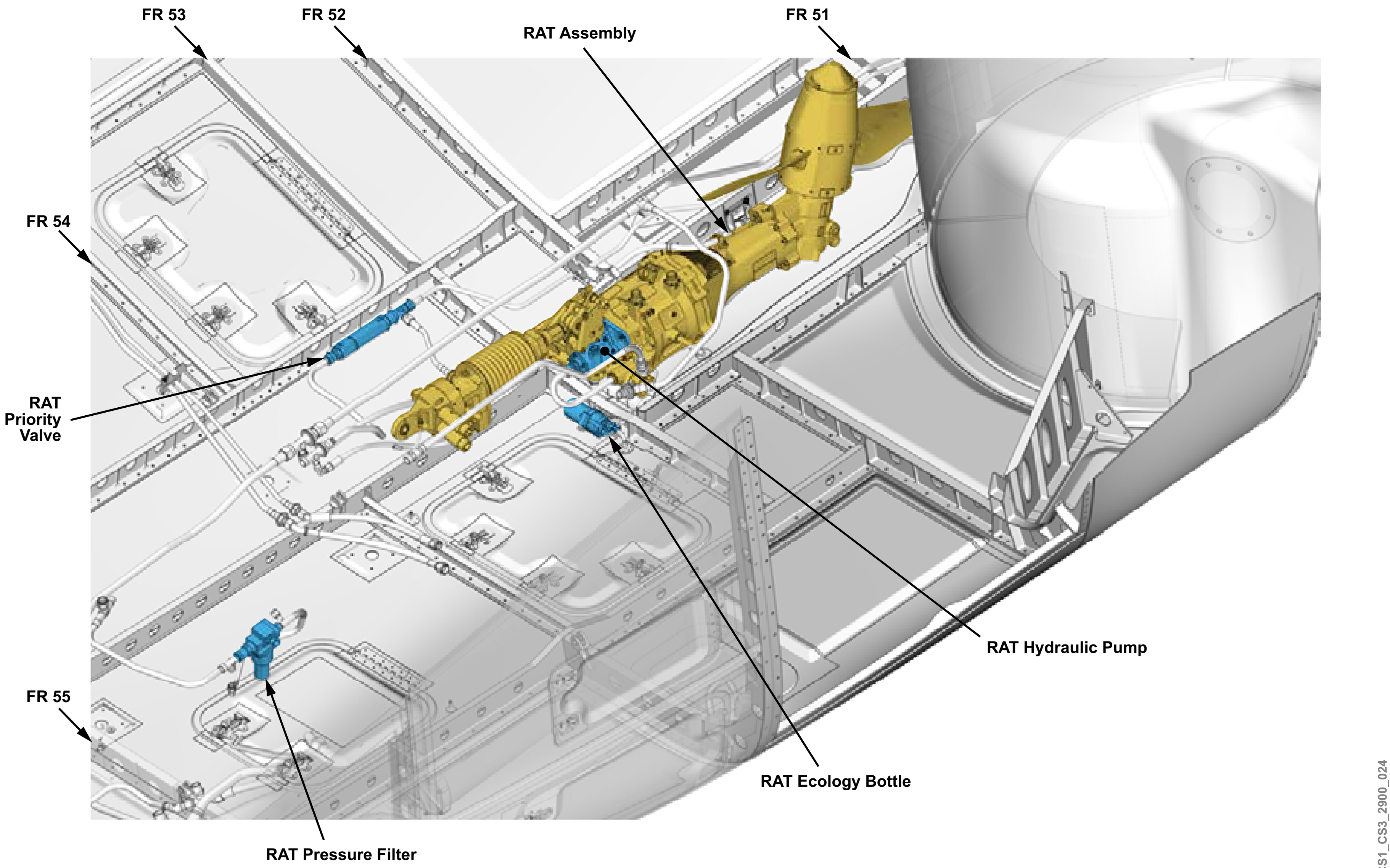


Figure 18: Ram Air Turbine Hydraulic Components

COMPONENT INFORMATION

RESERVOIR

The reservoir in hydraulic system no. 3 is similar in design and operation to the reservoirs in hydraulic system no. 1 and hydraulic system no. 2, but does not have a suction shutoff valve and a thermal fuse. The reservoir capacity is 13,109.7 cm^3 (800 $in.^3$). The position of components on the face of the hydraulic system no. 3 reservoir differs slightly from hydraulic systems no. 2 and no. 3.

The reservoir has the following mounted components:

- Visual quantity indicator
- Manual depressurization valve
- Low-pressure bleed/relief valve
- Fluid quantity transducer
- Temperature switch
- Temperature transducer

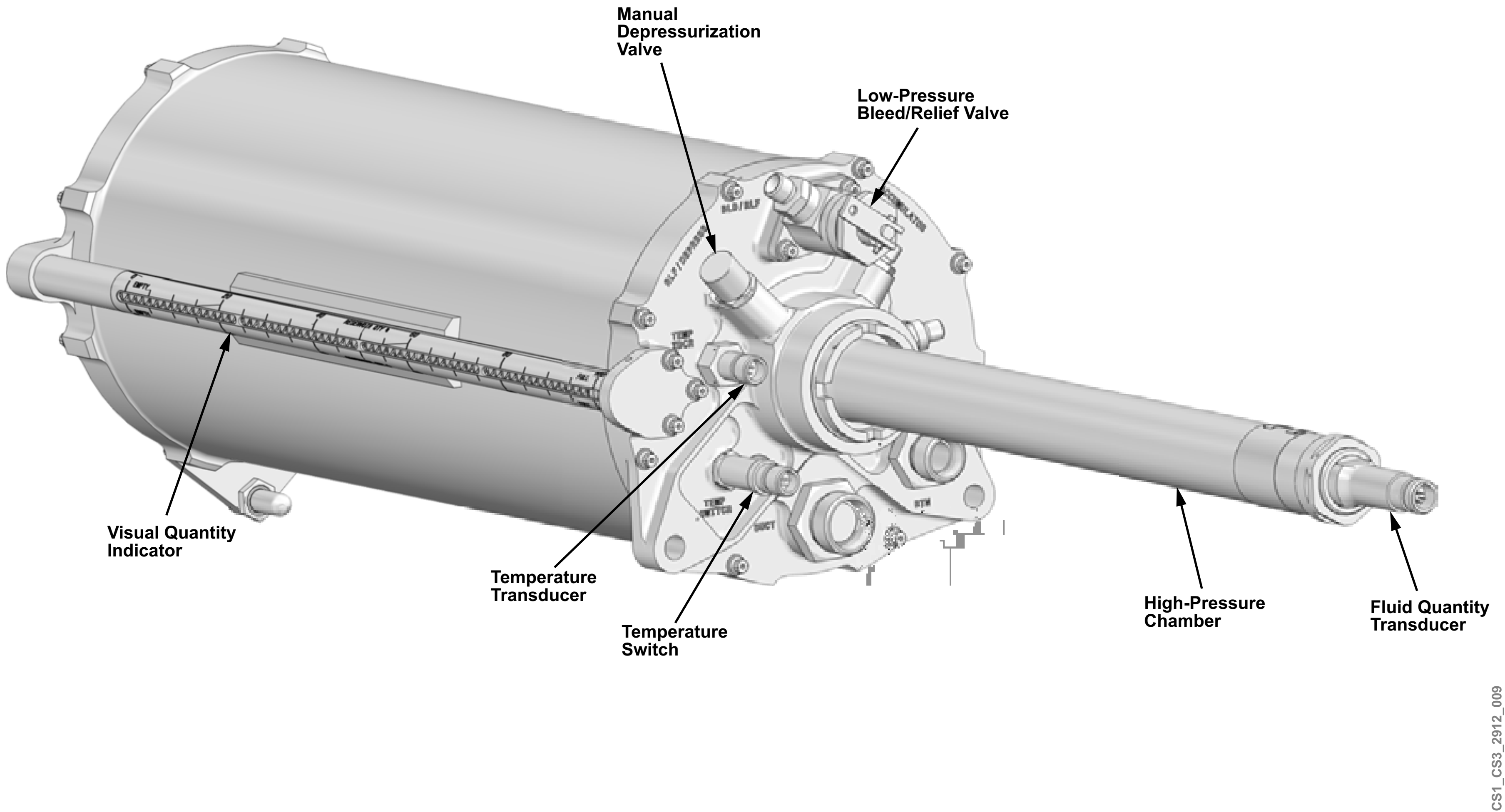


Figure 19: Hydraulic System No. 3 Reservoir

DETAILED COMPONENT INFORMATION

RAT HYDRAULIC PUMP

The RAT deploys either manually or automatically. It is deployed automatically when in the air and both AC BUS 1 and AC BUS 2 voltages are low.

There is no specific hydraulic trigger for deployment, because the loss of multiple hydraulic systems and loss of electrical capability (which would deploy the RAT) are so closely coupled that the electrical command is sufficient.

The RAT hydraulic pump incorporates a rate fuse. During pump startup, the rate fuse opens and connects the pump discharge to the pump suction inlet. At a certain point, the rate fuse valve closes and the pump operates as a pressure compensated pump, discharging fluid through the pressure port.

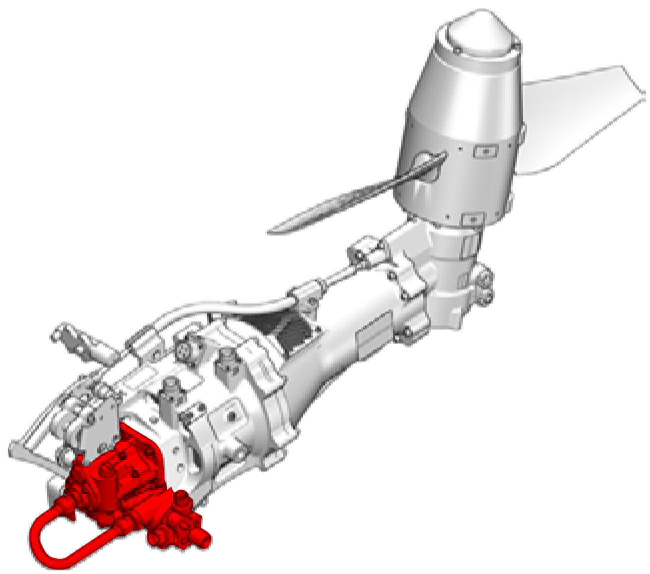
The rate fuse activates during RAT hydraulic pump startup to help minimize torque load on the pump and to prevent turbine stall.

A warming circuit check valve is installed in the hydraulic system no. 3 pressure circuit. When the RAT is stowed, the valve routes fluid through the RAT pump case to warm it. The fluid enters the discharge line of the pump and exits through the rate fuse valve installed in the pump inlet. Warming of the RAT ensures a smooth RAT start sequence when deployed, following extended periods of cold temperatures at high altitudes.

A damped check valve is installed in the pressure line downstream of the RAT. It prevents reverse flow, and also provides gradual fluid flow and pressure supply during RAT startup.

NOTE

RAT functional test is found in Aircraft Maintenance Publication (AMP) 24-23-01.



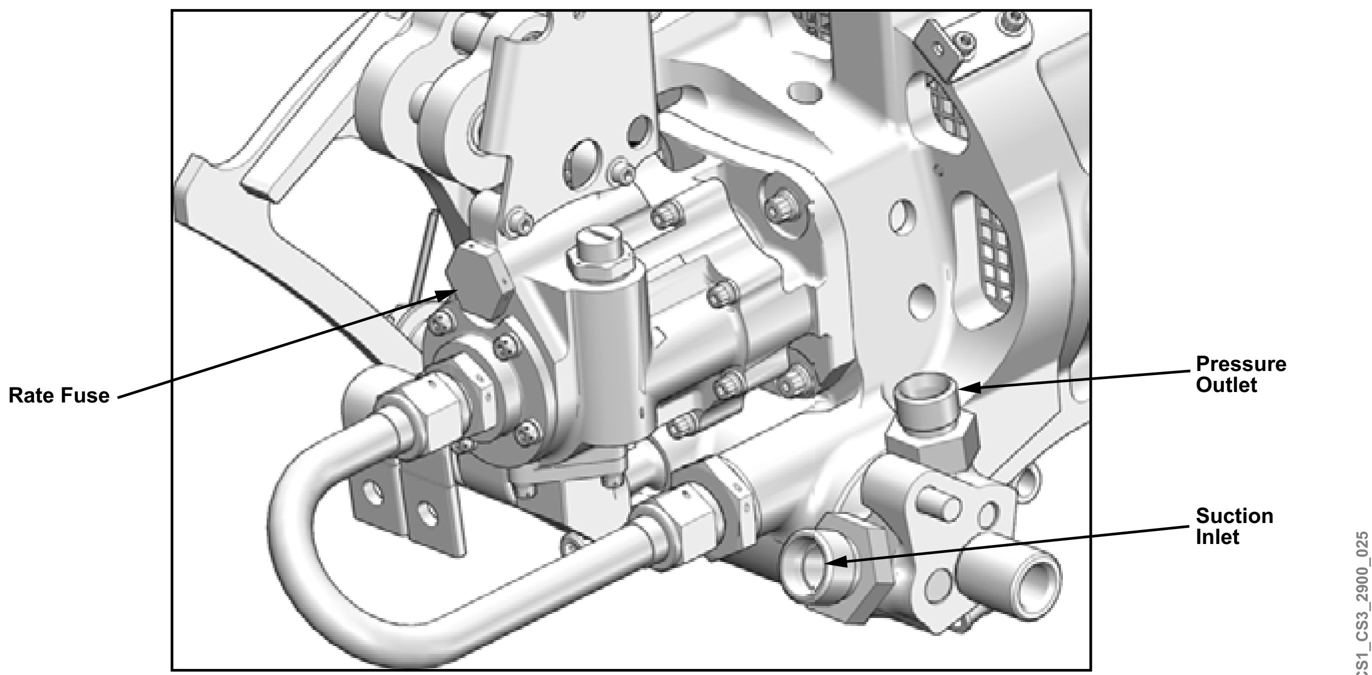


Figure 20: RAT Hydraulic Pump

29-30 HYDRAULIC INDICATING SYSTEM

GENERAL DESCRIPTION

Each of the hydraulic systems has transducers and switches that monitor and indicate the temperature, pressure, and fluid quantity of the system or hydraulic pump. The information is used for:

- Display on the HYDRAULIC synoptic page
- Hydraulic system servicing
- Hydraulic system status for other aircraft systems

The hydraulic indicating system provides the following information:

- Reservoir temperature
- Hydraulic quantity
- Hydraulic system pressure
- Pump or PTU pressure
- ACMP temperature
- Accumulator charge pressure

The hydraulic system fluid temperature is monitored at the reservoir by a temperature transducer.

A quantity transducer integral to the reservoir provides quantity information for display on the HYDRAULIC synoptic page. The fluid quantity transducer has excitation provided by the DMC. The quantity transducer is not a line replaceable unit (LRU). A visual quantity indicator is used for monitoring the reservoir during servicing.

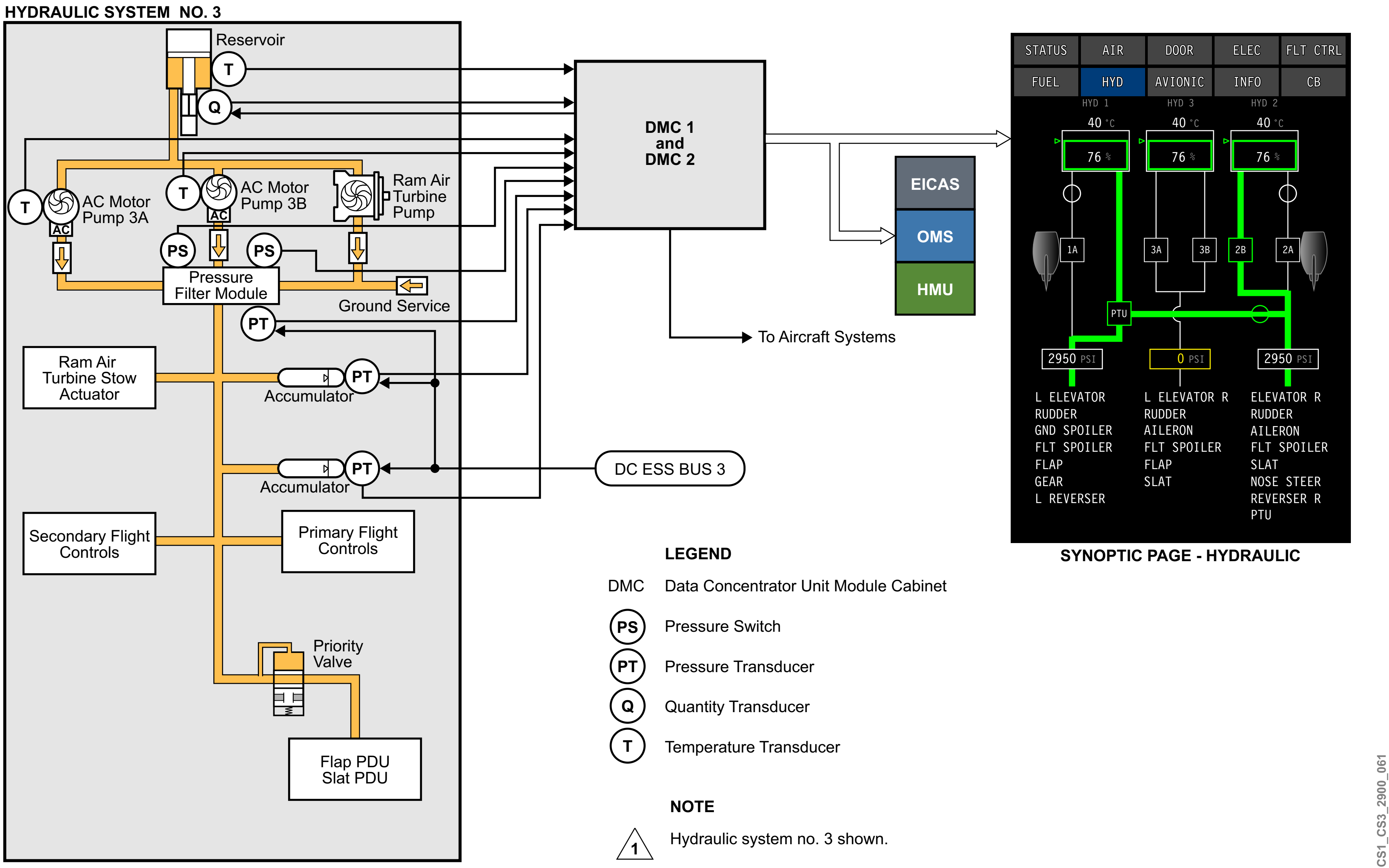
The pressure filter module has pressure switches to monitor the individual pump and PTU pressures. A pressure transducer at the outlet of the pressure filter module monitors the hydraulic system pressure.

Each ACMP has an internal temperature transducer that is used for monitoring and control of the ACMP.

In hydraulic system no. 3 only, the charge pressure for each accumulator is monitored by an accumulator charging gauge and a pressure gauge direct reading gauge on the hydraulic system no. 3 service panel. A pressure transducer provides accumulator charge pressure information to the DMCs.

Each hydraulic system pressure transducer is powered by 28 VDC supplied by the following BUSES:

- Hydraulic system no. 1 powered by DC BUS 1
- Hydraulic system no. 2 powered by DC BUS 2
- Hydraulic system no. 3 powered by DC ESS BUS 3



LEGEND

DMC Data Concentrator Unit Module Cabinet

PS Pressure Switch

PT Pressure Transducer

Q Quantity Transducer

T Temperature Transducer

NOTE

1 Hydraulic system no. 3 shown.

Figure 21: Hydraulic Indication Schematic

COMPONENT LOCATION

The following components of the hydraulic indicating system are installed on the hydraulic reservoir:

- Temperature transducer
- Fluid quantity transducer
- Visual quantity indicator

The following components of the hydraulic indicating system are installed on the hydraulic system pressure filter module:

- Hydraulic pump pressure switches
- Hydraulic system pressure transducer

The following components of the hydraulic indicating system are located in the aft compartment

- Accumulator charging gauge/pressure gauge (Refer to figure 23)
- Accumulator charge pressure transducers (Refer to figure 23)

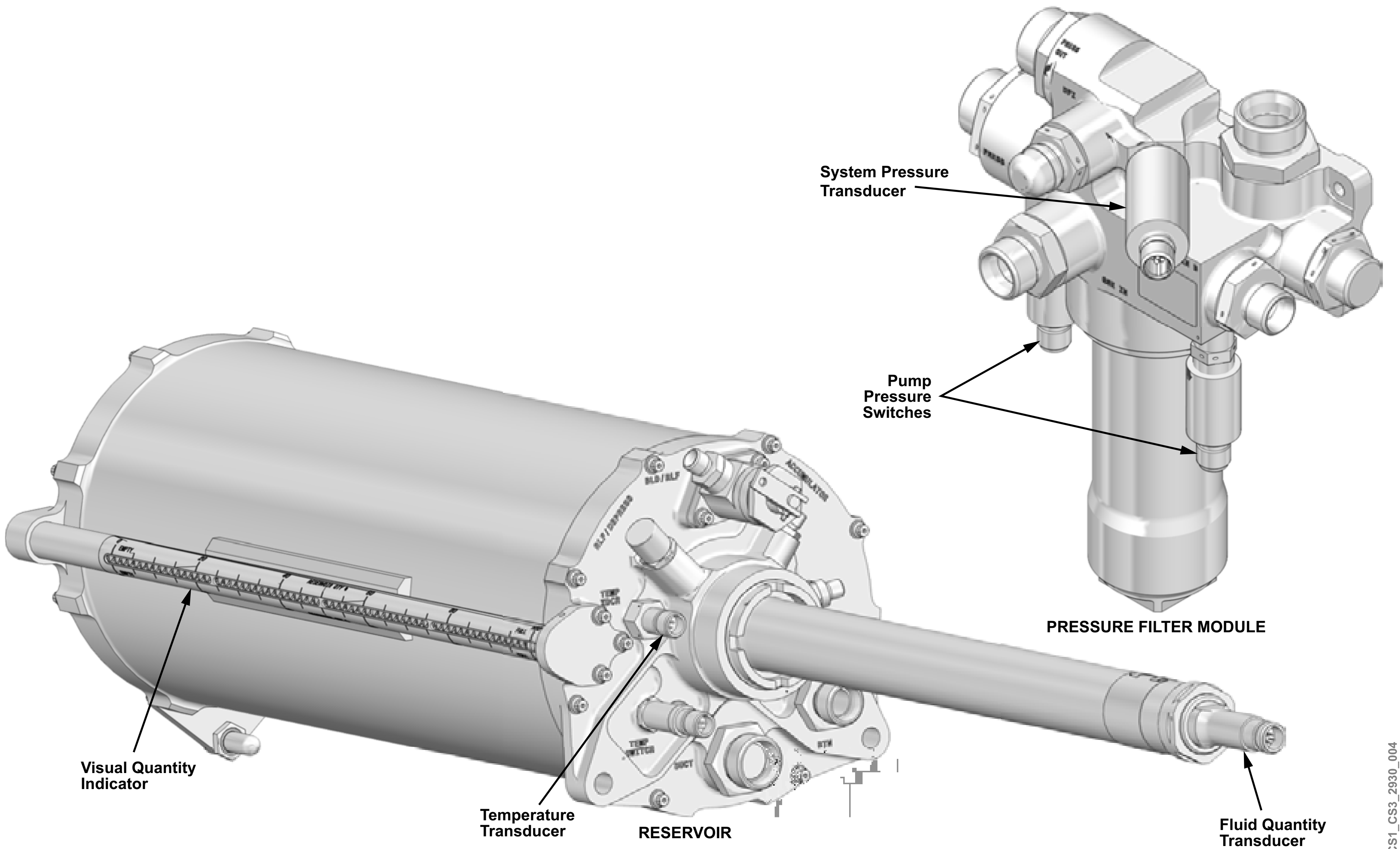


Figure 22: Hydraulic Indication System Component Location

ACCUMULATOR CHARGING GAUGE/PRESSURE GAUGE AND ACCUMULATOR CHARGE PRESSURE TRANSDUCERS

The accumulator charging gauge/pressure gauge and accumulator charge pressure transducers are located in the aft compartment.

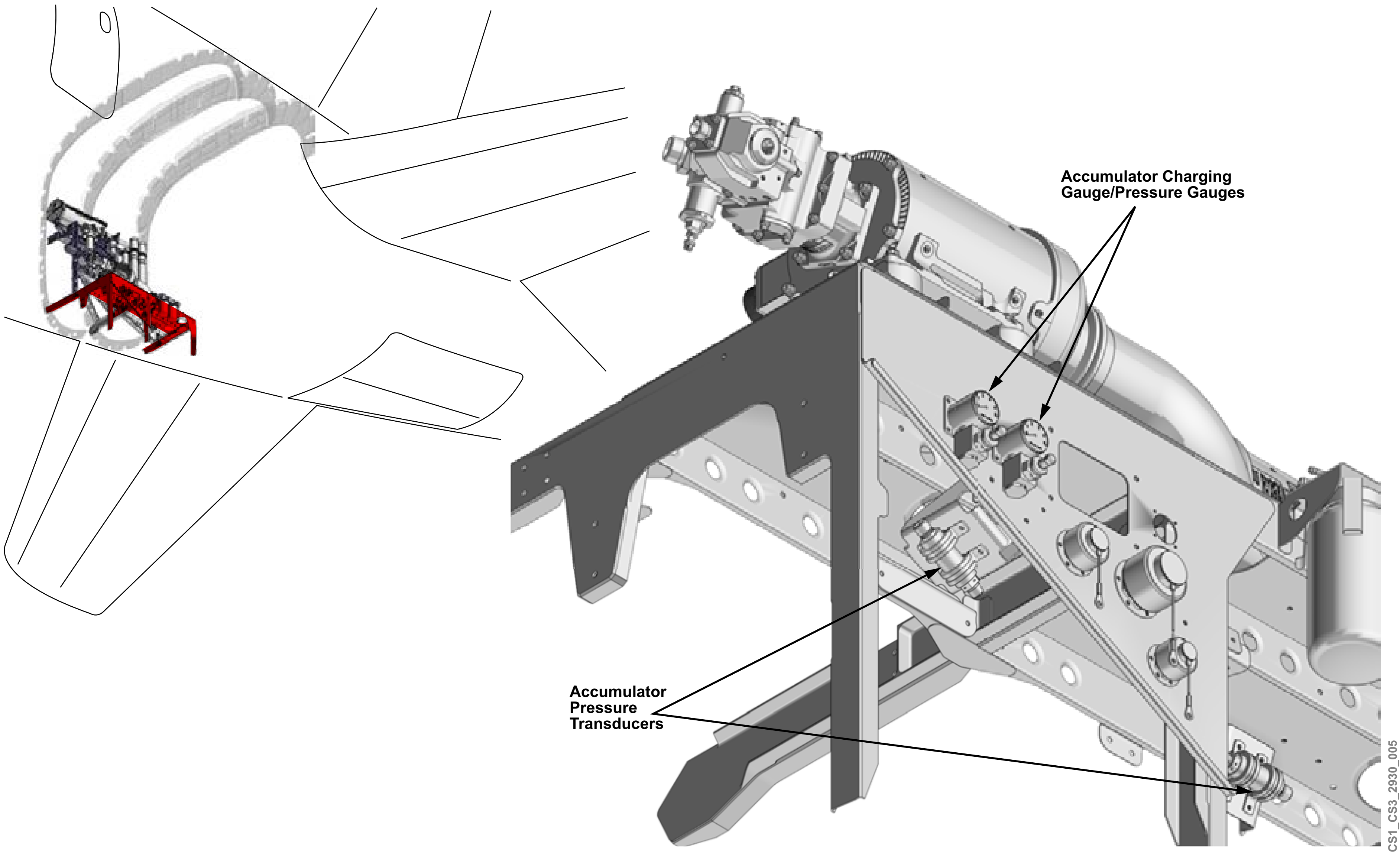


Figure 23: Accumulator Charging Gauge/Pressure Gauge and Accumulator Charge Pressure Transducer Location

CONTROLS AND INDICATIONS

HYDRAULIC SYNOPTIC PAGE

The following items are monitored in the three hydraulic systems, and are displayed on the HYDRAULIC synoptic page:

- System pressure in psi
- System reservoir fluid quantity from 0% to 100% range
- Fluid temperature in Celsius
- Pump pressure status as low, normal, or invalid
- Firewall shutoff valve (FWSOV) open, closed, in-transition, or invalid positions
- Power transfer unit (PTU) select valve open or closed position
- Power transfer unit (PTU) operational status

Valve position in normal and failed operations is represented by circular symbols. Names of hydraulic system users are displayed in color when they are pressurized or not pressurized.

Hydraulic fluid quantities are displayed in the reservoir. Hydraulic flow lines are displayed in green when the system is pressurized.

The RAT hydraulic pump and associated flow lines are only shown when the RAT is deployed, or control logic is invalid. When deployed, the RAT hydraulic pump pressure is displayed by a colored box.

Hydraulic pump pressures are displayed by colored boxes. Fluid temperature of each hydraulic system is displayed above the reservoir.

VALVE POSITION NORMAL OPERATION	
Symbol	Condition
	Close
	Open
	In Transition
	Invalid

VALVE POSITION FAIL OPERATION	
Symbol	Condition
	Close
	Open
	In Transition

PRESSURE	
Symbol	Condition
0 PSI	Low
2100 PSI	Normal
-- PSI	Invalid

FLOW LINES	
Symbol	Condition
	Normal Flow
	Low or No Flow
	Invalid

PUMPS	
Symbol	Condition
1A	Fail
3B	On
3A	Off
2B	Invalid

QUANTITY	
Symbol	Condition
76 %	Normal
6 %	Low
-- %	Invalid

TEMPERATURE	
Symbol	Condition
107 °C	Hi Temperature
40 °C	Normal
-- °C	Invalid

USERS	
Symbol	Condition
ELEVATOR	Pressure
GND SPOILER	No Pressure
X R REVERSER X	Invalid

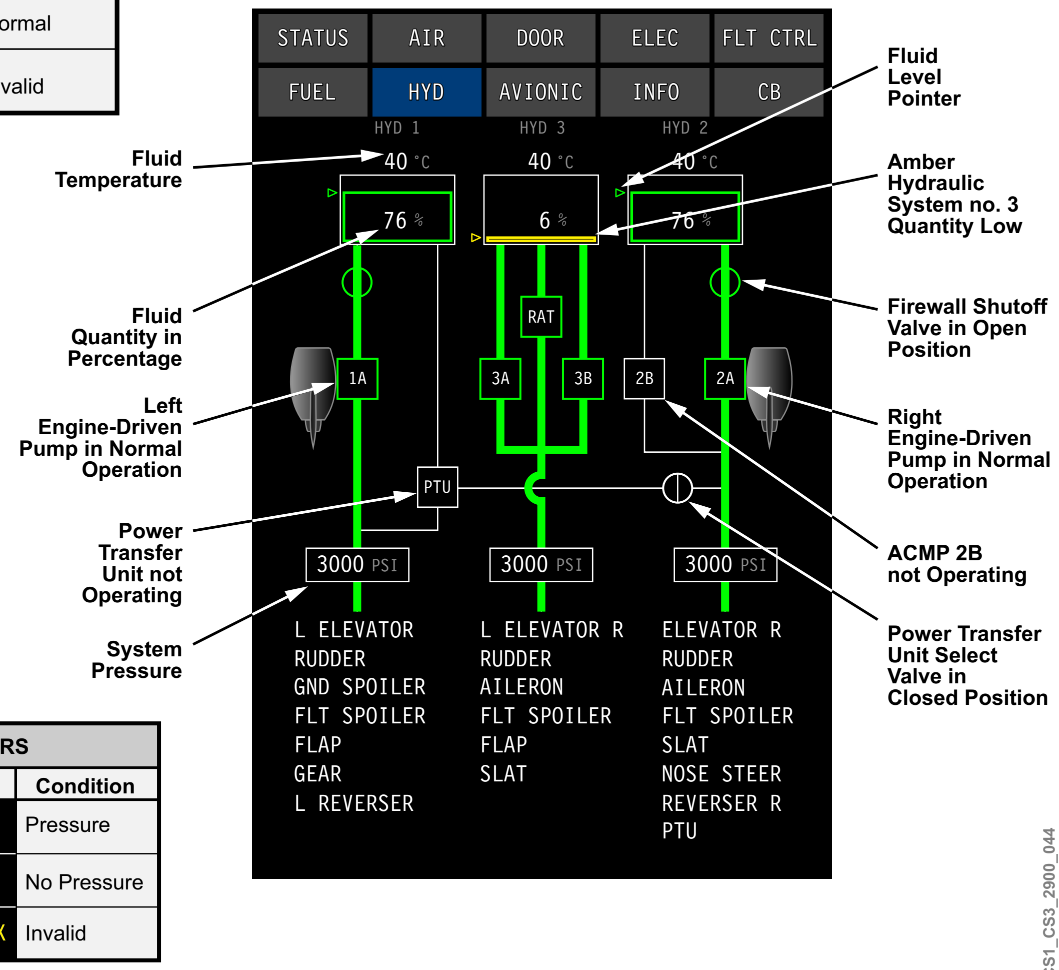


Figure 24: Hydraulic Indications

DETAILED DESCRIPTION

The reservoir temperature transducer supplies an analog temperature signal to DMC 1 and DMC 2.

The quantity indication is provided by a linear differential variable transducer (LVDT). The LVDT monitors the piston position to determine the quantity. The LVDT has an excitation signal supplied by the DMC. The quantity indication provides indication for each system is supplied to one DMC only. Hydraulic system no. 1 indication is supplied to DMC 1. Hydraulic system no. 2 and hydraulic system no. 3 provide quantity indication to DMC 2.

The pressure filter module pressure switches provide pressure-warning indication when the hydraulic pressure is above 2400 psi or below 1800 psi. Each pressure switch has two sets of contacts. One set is dedicated to DMC 1 and DMC 2 for low-pressure and high-pressure warning. The second set is used to supply a low-pressure warning to the electrical system for ACMP and PTU control logic.

The pressure filter module pressure transducer provides the hydraulic system pressure to DMC 1 and DMC 2

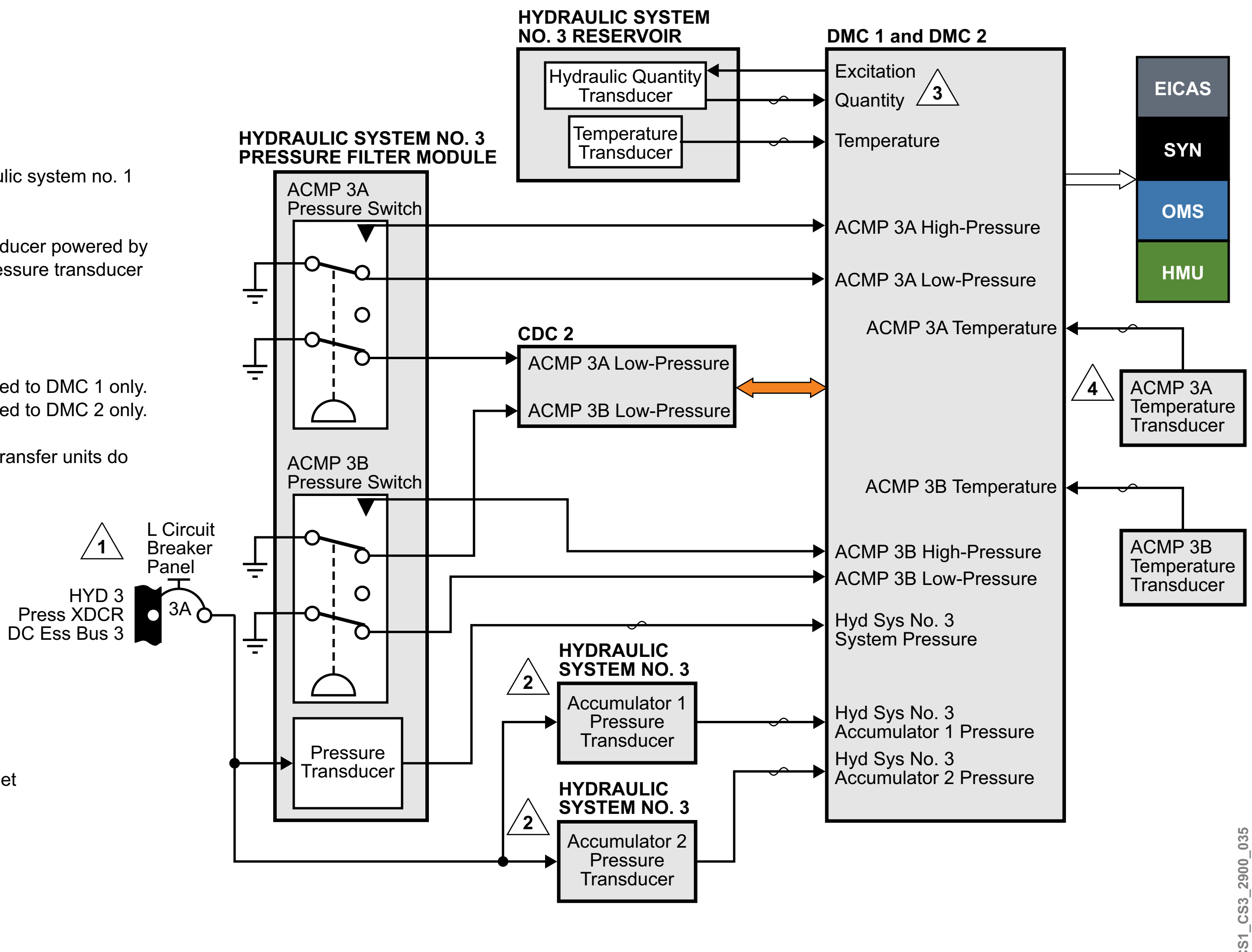
Each hydraulic system no. 3 accumulator has a pressure transducer installed in the charge line. The pressure transducer provides the accumulator charge pressure to DMC 1 and DMC 2 for use by the onboard maintenance system.

The ACMP temperature transducer protects the ACMP windings from over-temperature. The temperature transducer is used in the control logic to shut the pump off at 190° C (374°F).

NOTE

Hydraulic system no. 3 shown. Hydraulic system no. 1 and Hydraulic system no. 2 similar.

1. Hydraulic system no. 1 pressure transducer powered by DC BUS 1. Hydraulic system no. 2 pressure transducer powered by DC BUS 2.
2. Hydraulic system no. 3 only.
3. Hydraulic system no. 1 quantity supplied to DMC 1 only. Hydraulic system no. 2 quantity supplied to DMC 2 only.
4. The engine driven pumps and power transfer units do not have temperature monitoring.



LEGEND

ACMP	AC Motor Pump
CDC	Control and Distribution Cabinet
DMC	Data Concentrator Unit Module Cabinet
	AFDX A664
	Analog
	Discrete

Figure 25: Hydraulic Pressure, Temperature, and Quantity Schematic

MONITORING AND TESTS

The following pages provide the crew alerting system (CAS) and INFO messages for the hydraulic indicating system.

CAS MESSAGES

Table 1: CAUTION Messages

MESSAGE	LOGIC
HYD 1 HI TEMP	HYD 1 reservoir temperature > 107°C (225°F) or is active 30 seconds after reservoir temperature switch indicates high temperature.
HYD 2 HI TEMP	HYD 2 reservoir temperature > 107°C (225°F) or is active 30 seconds after reservoir temperature switch indicates high temperature.
HYD 3 HI TEMP	HYD 3 reservoir temperature > 107°C (225°F) or is active 30 seconds after reservoir temperature switch indicates high temperature.
HYD 1 LO PRESS	HYD 1 low-pressure detected by either EDP and PTU pressure switches or HYD 1 pressure sensor.
HYD 2 LO PRESS	HYD 2 low-pressure detected by either EDP and ACMP pressure switches or HYD 2 pressure sensor.
HYD 3 LO PRESS	HYD 3 low-pressure detected by either 3A and 3B pumps pressure switches or system HYD 3 pressure sensor.
HYD 1-2 LO PRESS	Low-pressure detected on HYD 1 and HYD 2.
HYD 2-3 LO PRESS	Low-pressure detected on HYD 2 and HYD 3.
HYD 1-3 LO PRESS	Low-pressure detected on HYD 1 and HYD 3.

Table 2: ADVISORY Messages

MESSAGE	LOGIC
HYD 1 LO QTY	This message appears in two scenarios: 1 - Aircraft is on the ground with reservoir quantity less than 18% 2 - Aircraft is in the air with reservoir quantity less than 13%
HYD 2 LO QTY	This message appears in two scenarios: 1 - Aircraft is on the ground with reservoir quantity less than 16% 2 - Aircraft is in the air with reservoir quantity less than 13%
HYD 3 LO QTY	This message appears in two scenarios: 1 - Aircraft is on the ground with reservoir quantity less than 14% 2 - Aircraft is in the air with reservoir quantity less than 11%
HYDRAULIC FAULT	Loss of non-critical functions or loss of redundancy in the hydraulic systems (refer to INFO messages).

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Table 3: INFO Messages

MESSAGE	LOGIC
29 HYDRAULIC FAULT - HYD 1 PRESS SNSR INOP	The hydraulic system no.1 pressure sensor output is invalid or sensor is indicating low-pressure when either the EDP 1A or the PTU pressure switches are indicating high-pressure.
29 HYDRAULIC FAULT - HYD 2 PRESS SNSR INOP	The hydraulic system no. 2 pressure sensor output is invalid or sensor is indicating low-pressure when either the EDP 2A, or ACMP 2B pressure switches are indicating high-pressure.
29 HYDRAULIC FAULT - HYD 3 PRESS SNSR INOP	The hydraulic system no. 3 pressure sensor output is invalid or sensor is indicating low-pressure when either the ACMP 3A or the ACMP 3B pressure switches are indicating high-pressure.
29 HYDRAULIC FAULT - HYD 3 ACCUM 1 SNSR INOP	The hydraulic system no. 3 accumulator no. 1 pressure sensor is invalid.
29 HYDRAULIC FAULT - HYD 3 ACCUM 2 SNSR INOP	The hydraulic system no. 3 accumulator no. 2 pressure sensor is invalid.
29 HYDRAULIC FAULT - HYD 1 RESERV TEMP SNSR INOP	The hydraulic system no. 1 temperature sensor is invalid.
29 HYDRAULIC FAULT - HYD 2 RESERV TEMP SNSR INOP	The hydraulic system no. 2 temperature sensor is invalid.
29 HYDRAULIC FAULT - HYD 3 RESERV TEMP SNSR INOP	The hydraulic system no. 3 temperature sensor is invalid.
29 HYDRAULIC FAULT - HYD PUMP 2B TEMP SNSR INOP	The hydraulic system no. 2 ACMP 2B temperature sensor is invalid.

Table 3: INFO Messages (Continued)

MESSAGE	LOGIC
29 HYDRAULIC FAULT - HYD PUMP 3A TEMP SNSR INOP	The hydraulic system no. 3 ACMP 3A temperature sensor is invalid.
29 HYDRAULIC FAULT - HYD PUMP 3B TEMP SNSR INOP	The hydraulic system no. 3 ACMP 3B temperature sensor is invalid.
29 HYDRAULIC FAULT - HYD EDP 1A PRESS SW INOP	The EDP 1A pressure switch is indicating both high and low pressure at the same time, or not indicating high or low pressure.
29 HYDRAULIC FAULT - HYD PTU PRESS SW INOP	The PTU pressure switch is indicating both high and low pressure at the same time, or not indicating high or low pressure.
29 HYDRAULIC FAULT - HYD EDP 2A PRESS SW INOP	The EDP 2A pressure switch is indicating both high and low pressure at the same time, or not indicating high or low pressure.
29 HYDRAULIC FAULT - HYD PUMP 2B PRESS SW INOP	The ACMP 2B pressure switch is indicating both high and low pressure at the same time, or not indicating high or low pressure.
29 HYDRAULIC FAULT - HYD PUMP 3A SW INOP	The ACMP 3A pressure switch is indicating both high and low pressure at the same time, or not indicating high or low pressure.
29 HYDRAULIC FAULT - HYD PUMP 3B PRESS SW INOP	The ACMP 3B pressure switch is indicating both high and low pressure at the same time, or not indicating high or low pressure.
29 HYDRAULIC FAULT - HYD CDC EDP 1A PRESS SW INOP	The EDP 1A LO pressure switch from the electrical system is not tracking the pressure switch in the avionics system.

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Table 3: INFO Messages (Continued)

MESSAGE	LOGIC
29 HYDRAULIC FAULT - HYD CDC EDP 2A PRESS SW INOP	The EDP 2A LO pressure switch from the electrical system is not tracking the pressure switch in the avionics system.
29 HYDRAULIC FAULT - HYD CDC ACMP 3A PRESS SW INOP	The ACMP 3A LO pressure switch from the electrical system is not tracking the pressure switch in the avionics system.
29 HYDRAULIC FAULT - HYD CDC ACMP 3B PRESS SW INOP	The ACMP 3B LO pressure switch from the electrical system is not tracking the pressure switch in the avionics system.
29 HYDRAULIC FAULT - HYD 1 QTY SNSR INOP	The quantity sensor signal is not valid, the monitor signal is not valid, or the quantity sensor monitor signal is not within the valid tolerance.
29 HYDRAULIC FAULT - HYD 2 QTY SNSR INOP	The quantity sensor signal is not valid, the monitor signal is not valid, or the quantity sensor monitor signal is not within the valid tolerance.
29 HYDRAULIC FAULT - HYD 3 QTY SNSR INOP	The quantity sensor signal is not valid, the monitor signal is not valid, or the quantity sensor monitor signal is not within the valid tolerance.

29-00 CONTROL AND OPERATION

GENERAL DESCRIPTION

CONTROLS

The hydraulic system is operated from the HYDRAULIC panel located on the flight deck overhead panel.

The AC motor pumps (ACMPs), firewall shutoff valves (FWSOVs), and the power transfer unit (PTU) are controlled from this panel.

HYDRAULIC Panel

The HYDRAULIC panel switches provide the following functions for the hydraulic system:

- Pushbutton annunciators (PBAs) for the left and right hydraulic firewall shutoff valves (FWSOVs)
- Three-position (OFF, AUTO, ON) rotary switches to operate the PTU, ACMP 2B, ACMP 3A, and ACMP 3B

The normal position for the four rotary switches is the AUTO selection. For maintenance purposes, selecting switch 2B to the ON position pressurizes hydraulic system no. 2, and selecting switches 3A or 3B to the ON position pressurizes hydraulic system no. 3. Selecting switch 2B to the ON position and then selecting the PTU switch to the ON position pressurizes hydraulic system no. 1.

System pressure is displayed on the hydraulic synoptic page when the hydraulic systems are pressurized.

A CLSD indication is illuminated on the HYD 1 SOV, and HYD 2 SOV PBAs when the hydraulic FWSOVs are closed. The position of these valves is displayed on the hydraulic synoptic page.

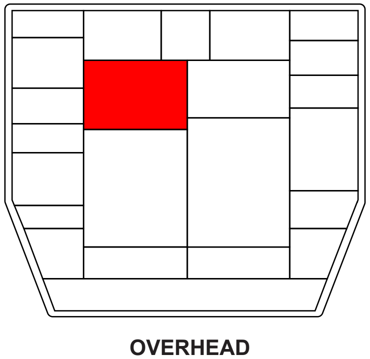


Figure 26: HYDRAULIC Panel

OPERATION

For maintenance purposes and without the use of a ground hydraulic cart, hydraulic system no. 2 is pressurized with ACMP 2B, prior to turning ON the PTU to pressurize hydraulic system no. 1.

HYDRAULIC SYSTEM NO. 1

The power transfer unit (PTU) is operated by the PTU switch on the hydraulic panel. This switch has an OFF, AUTO, and ON position. The normal selection for the PTU switch is the AUTO position.

After pressurizing hydraulic system no. 2 with ACMP 2B, selecting PTU switch to the ON position pressurizes hydraulic system no. 1.

HYDRAULIC SYSTEM NO. 2

The AC motor pump (ACMP) 2B is operated by the 2B rotary switch on the hydraulic panel. This switch has an OFF, AUTO, and ON position. The normal selection for the 2B switch is the AUTO position.

Pressurize Hydraulic System No. 2 for Maintenance with ACMP 2B

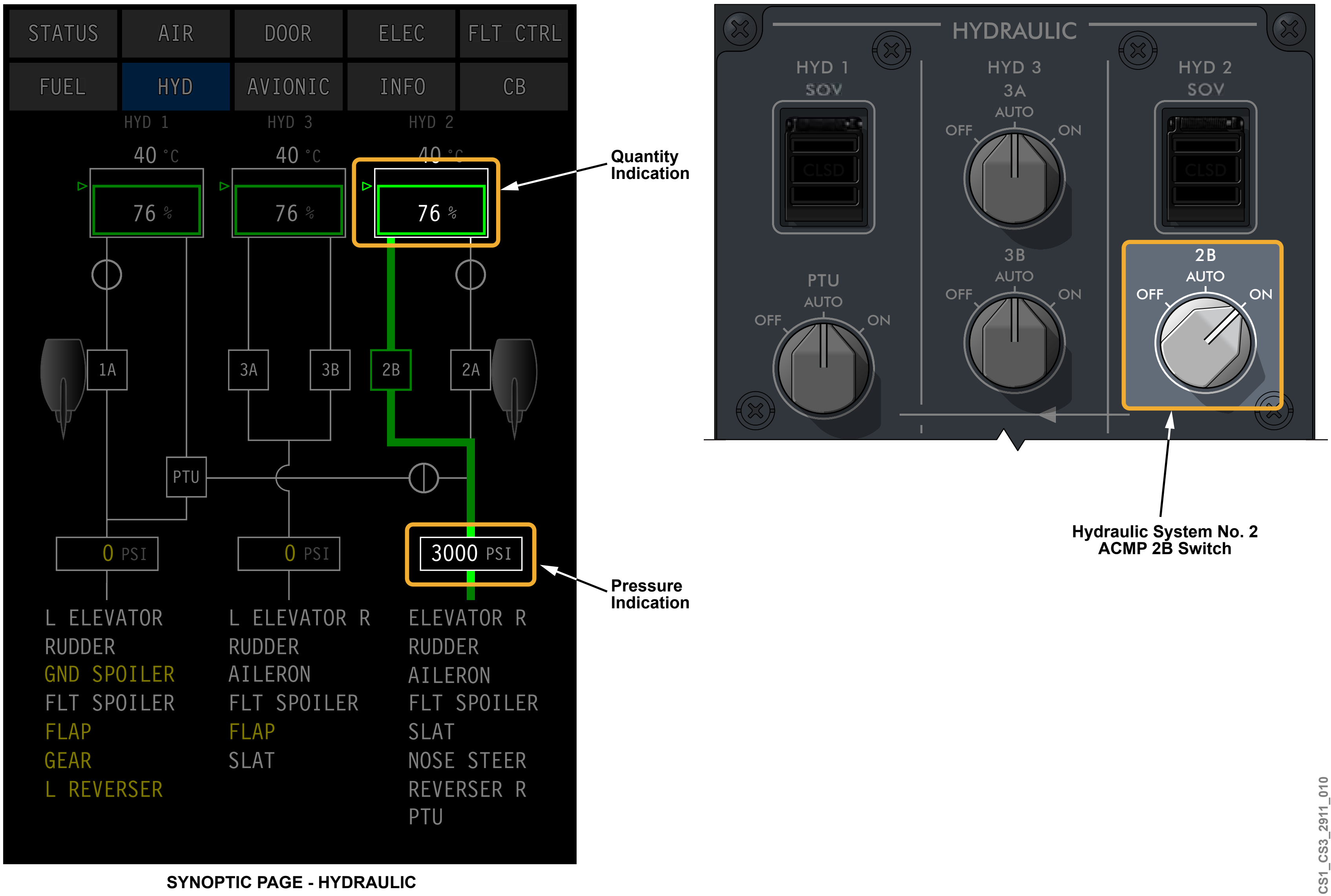
- On the HYDRAULIC panel, select the 2B switch to the OFF position
- On the HYDRAULIC synoptic page, check that hydraulic system no. 2 quantity is normal
- Pressurize hydraulic system no. 2 by selecting the 2B switch to the ON position
- On the HYDRAULIC synoptic page, verify that hydraulic system no. 2 is displaying pressure

Depressurizing Hydraulic System No. 2 After Being Pressurized for Maintenance with ACMP 2B

- On the HYDRAULIC panel, select 2B switch to the OFF position
- On the HYDRAULIC synoptic page, verify that ACMP 2B has shutdown and hydraulic system no. 2 pressure has dropped to 0 psi
- Return 2B to the AUTO position and verify that the pressure remains at 0 psi

CAUTION

Before pressurizing the aircraft hydraulic system, ensure all safety conditions have been met and are in accordance with the Aircraft Maintenance Publication (AMP).



SYNOPTIC PAGE - HYDRAULIC

Figure 27: Pressurize Hydraulic System No. 2 with AC Motor Pump 2B

HYDRAULIC SYSTEM NO. 1

Pressurize Hydraulic System No. 1 with the PTU for Maintenance

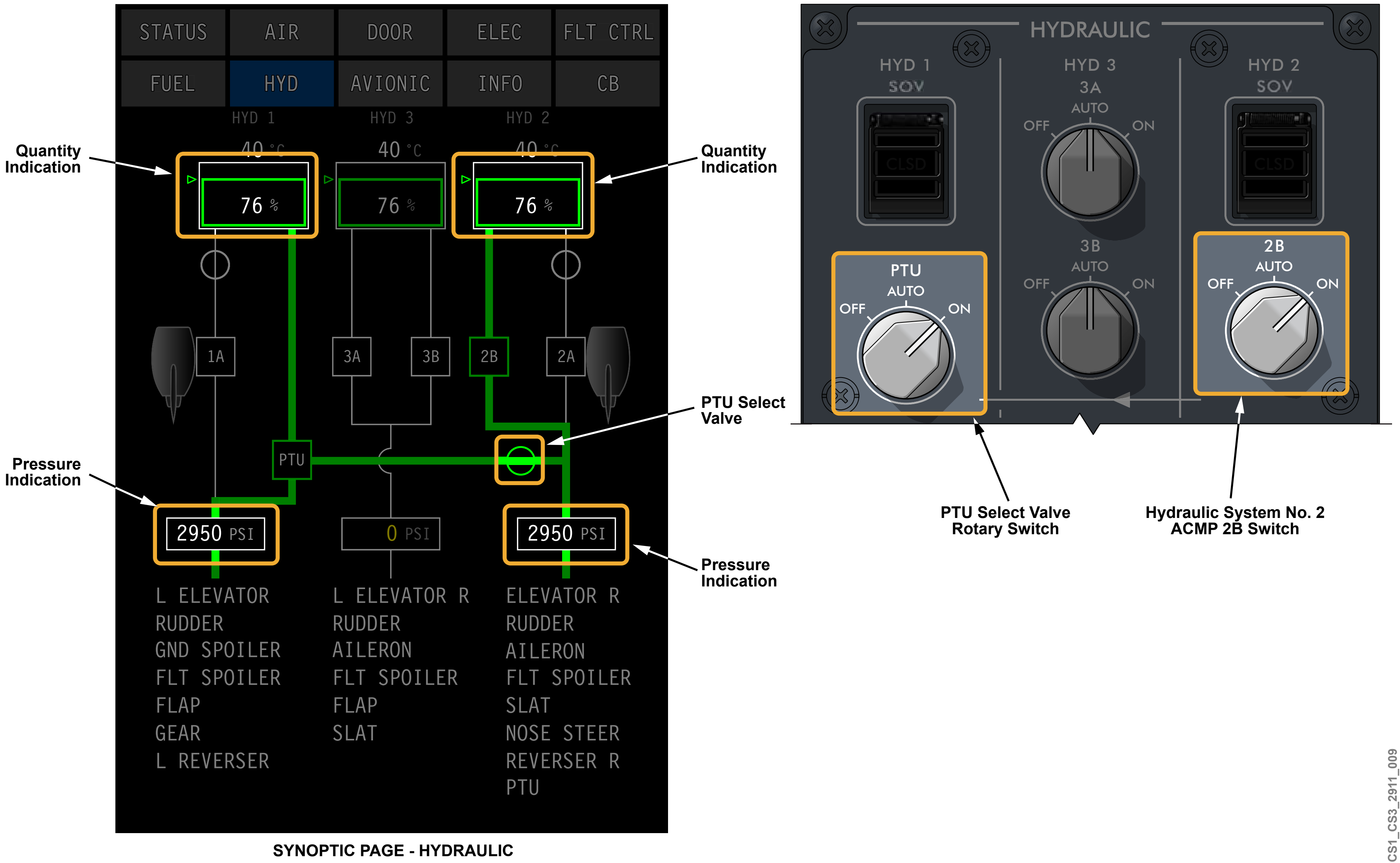
- Select 2B switch to the OFF position
- On the HYDRAULIC synoptic page, check that hydraulic system no. 1 and hydraulic system no. 2 quantities are normal
- On the HYDRAULIC panel, select 2B switch to the ON position
- On the HYDRAULIC synoptic page, verify that hydraulic system no. 2 is pressurized and that ACMP 2B is displayed as ON
- On the HYDRAULIC panel, select the PTU switch to the ON position
- On the HYDRAULIC synoptic page, verify that hydraulic system no. 1 is displaying pressure, and that the PTU is displayed ON, and PTU select valve is displayed as OPEN

Depressurizing Hydraulic System No.1 After Being Pressurized by the PTU for Maintenance

- On the HYDRAULIC panel, select the PTU switch to the OFF position
- On the HYDRAULIC synoptic page, verify that the PTU select valve is CLOSED, and that the PTU is displayed as OFF
- On the HYDRAULIC synoptic page, verify that hydraulic system no. 1 pressure has dropped to 0 psi
- On the HYDRAULIC panel, select 2B switch to the OFF position
- On the HYDRAULIC synoptic page, verify that ACMP 2B is displayed as OFF
- On the HYDRAULIC synoptic page, verify that the pressure in hydraulic system no. 2 drops to 0 psi
- Select the PTU switch and 2B switches to the AUTO position. The pressure in hydraulic system no. 1 and hydraulic system no. 2 should stay at 0 psi

CAUTION

Before pressurizing the aircraft hydraulic system, ensure all safety conditions have been met and are in accordance with the Aircraft Maintenance Publication (AMP).



SYNOPTIC PAGE - HYDRAULIC

Figure 28: Pressurize Hydraulic System No. 1 with Power Transfer Unit

HYDRAULIC SYSTEM NO. 3

Pressurizing Hydraulic System No. 3 with ACMP 3A and 3B for Maintenance

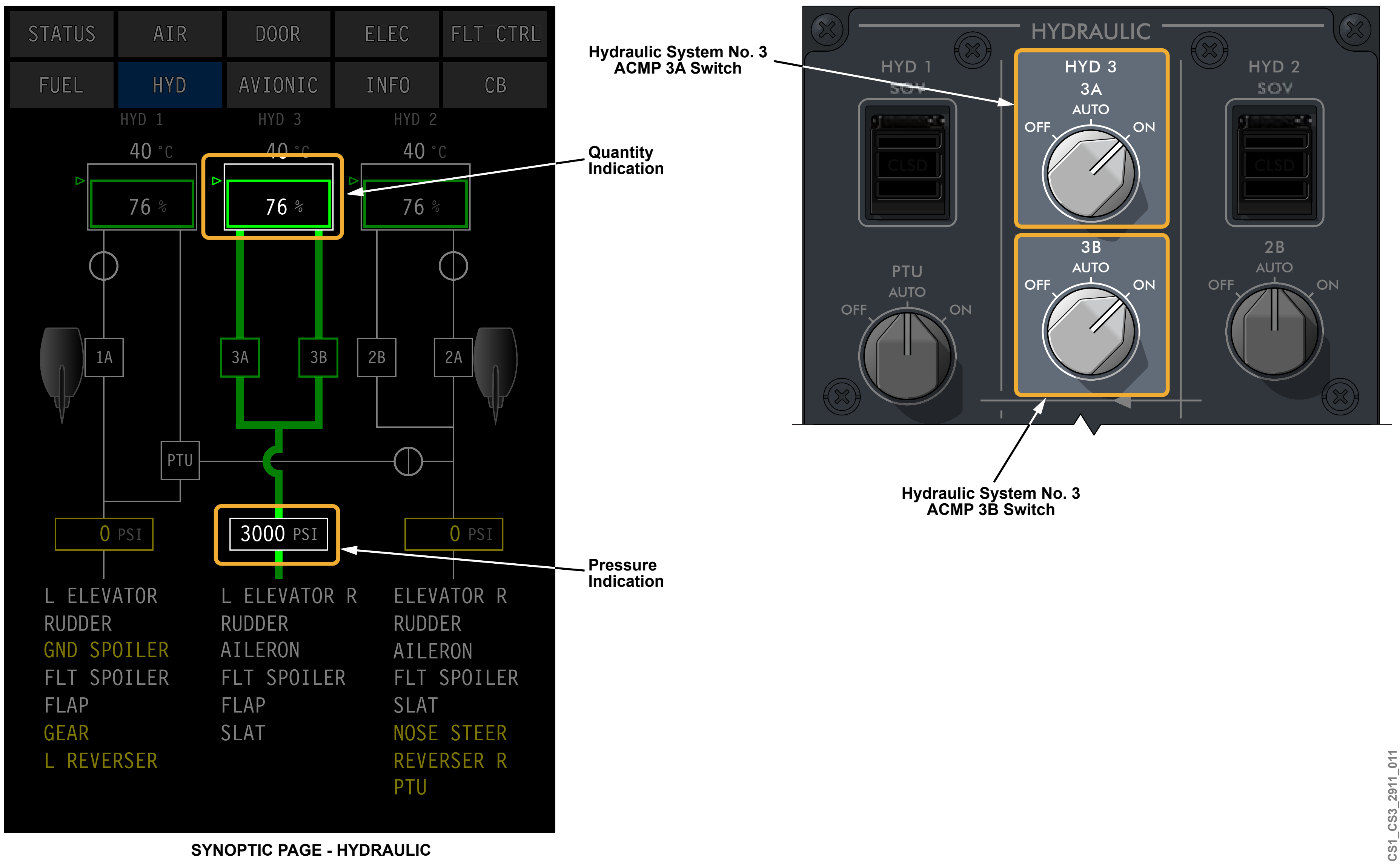
- Select 3A and 3B switches to the OFF positions
- On the HYDRAULIC synoptic page, check that the hydraulic system no. 3 quantity is normal
- On the HYDRAULIC panel, select the 3A (or 3B) switch to the ON position
- On the HYDRAULIC synoptic page, verify that ACMP 3A is ON and that hydraulic system no. 3 is pressurized

Depressurizing Hydraulic System No. 3

- On the HYDRAULIC panel, select 3A and 3B switches to the OFF position
- On the HYDRAULIC synoptic page, verify that ACMP 3A and ACMP 3B have shutdown, and hydraulic system no. 3 is displaying 0 psi
- Return 3A and 3B switches to the AUTO positions, and verify that the pressure in hydraulic system no. 3 stays at 0 psi

CAUTION

Before pressurizing the aircraft hydraulic system, ensure all safety conditions have been met and are in accordance with the Aircraft Maintenance Publication (AMP).



SYNOPTIC PAGE - HYDRAULIC

Figure 29: Pressurize Hydraulic System No. 3 with ACMP 3A and ACMP 3B

DETAILED DESCRIPTION

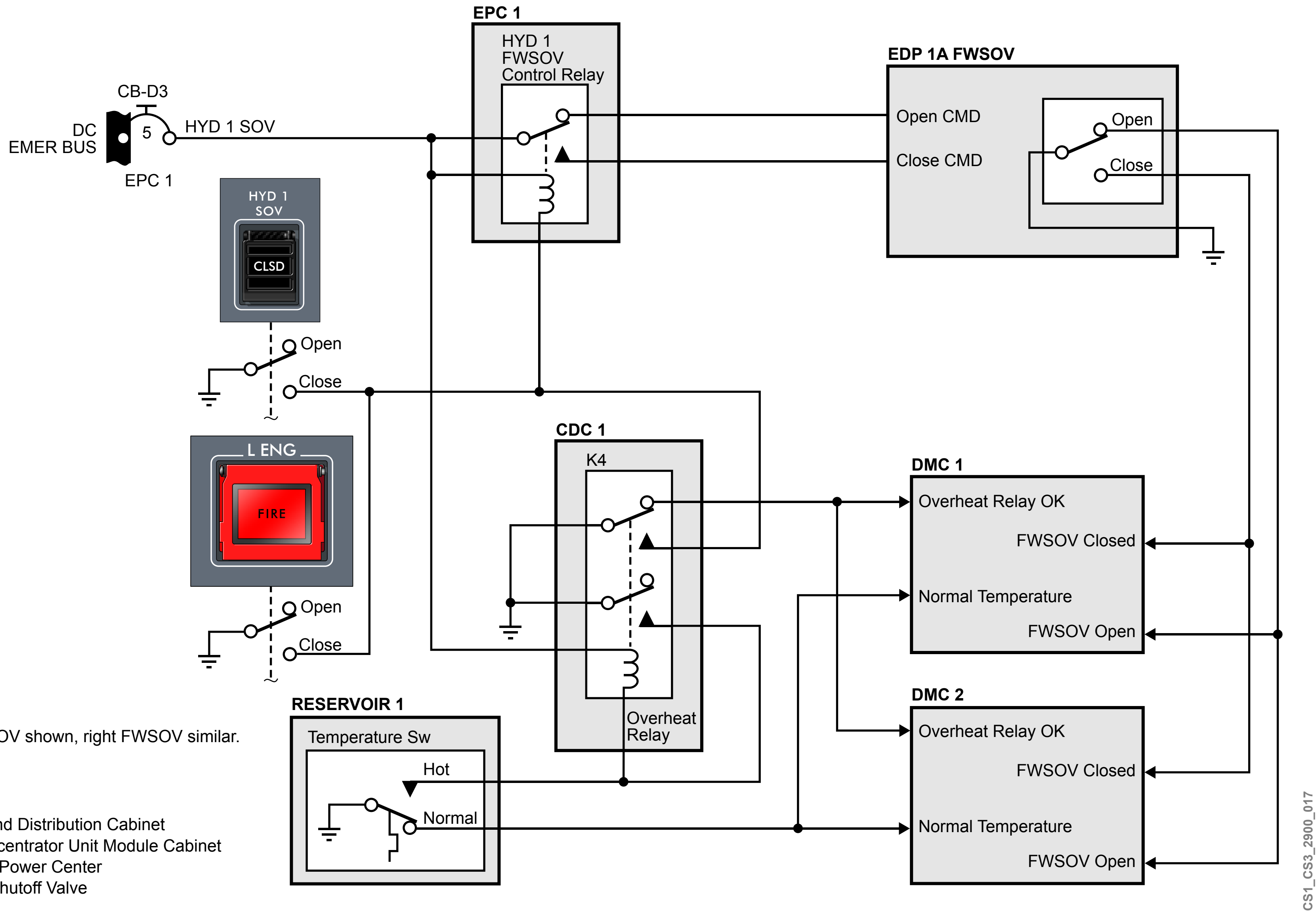
FIREWALL SHUTOFF VALVE CONTROL

Each FWSOV is controlled through an overheat relay and a control relay. The FWSOVs can be closed from the following three sources:

- HYDRAULIC panel PBA
- ENGINE FIRE panel PBA
- Temperature switch

Hydraulic system no. 1 and no. 2 FWSOV control relays are energized to close their respective FWSOV through their overheat relay. The overheat relay receives a signal from hydraulic system no. 1 (2) temperature switch when the reservoir fluid temperature is above 125°C (257°F). The overheat relay remains latched upon actuation, and can only be reset by removing electrical power from the relay.

- Hydraulic system no. 1 FWSOV control relay is energized to close the left FWSOV when the HYD 1 SOV PBA is pressed
- Hydraulic system no. 2 FWSOV control relay is energized to close the right FWSOV when the HYD 2 SOV PBA is pressed
- Hydraulic system no. 1 FWSOV control relay is energized to close the left FWSOV when the L ENG FIRE PBA is pressed
- Hydraulic system no. 2 FWSOV control relay is energized to close the right FWSOV when the R ENG FIRE PBA is pressed



NOTE

Left FWSOV shown, right FWSOV similar.

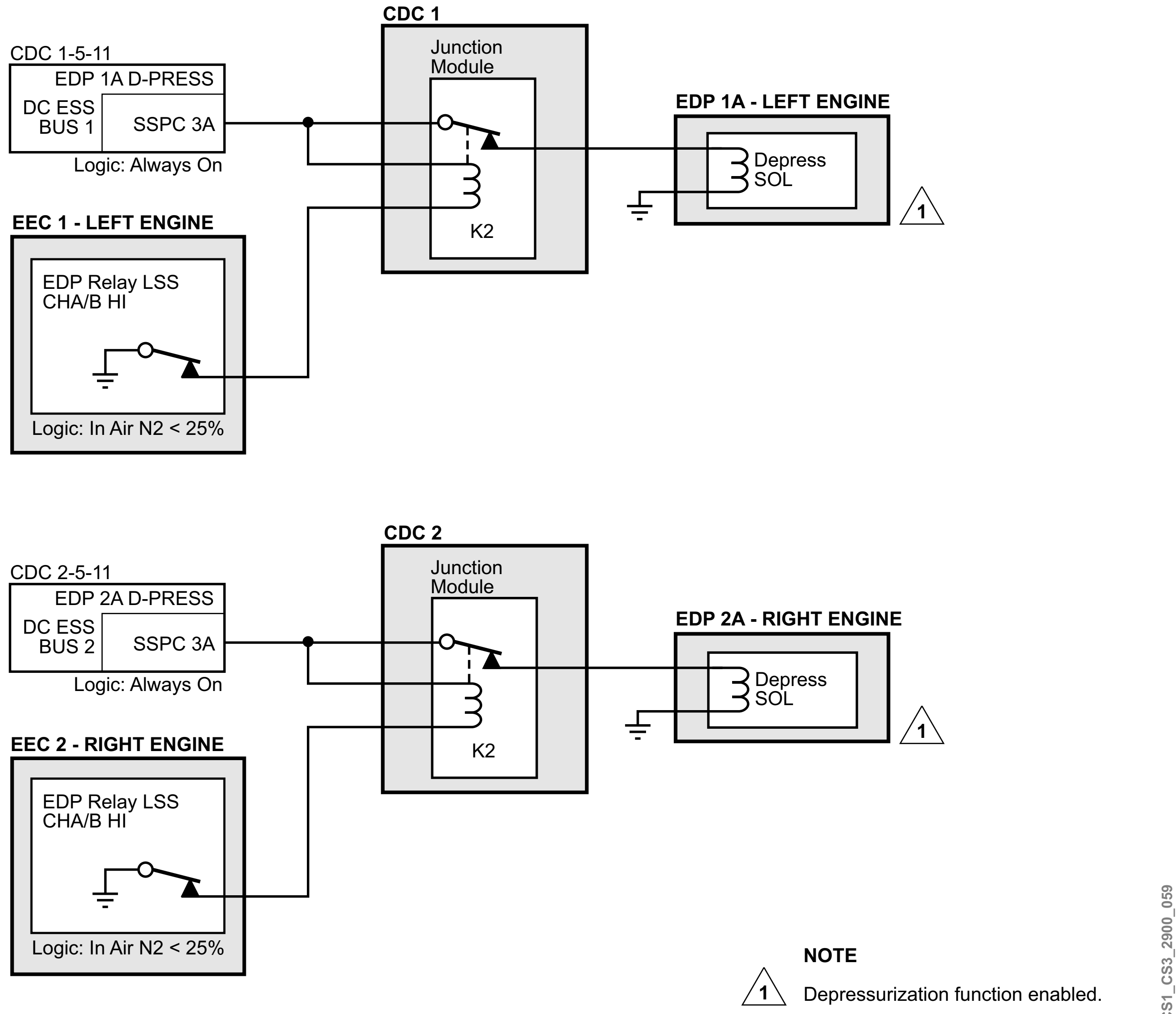
LEGEND

CDC	Control and Distribution Cabinet
DMC	Data Concentrator Unit Module Cabinet
EPC	Electrical Power Center
FWSOV	Firewall Shutoff Valve

Figure 30: Firewall Shutoff Valve Control

ENGINE-DRIVEN PUMP DEPRESSURIZATION

The pump incorporates a depressurization function, which is provided by an internal solenoid operated depressurization valve. The electronic engine control (EEC) relays a command to the control and distribution cabinet (CDC), to electrically energize the valve when the aircraft is in the air, and the engine N2 speed is less than 25%. When the depressurization valve is energized, it closes and blocks EDP discharge pressure, and relieves the engine from pump torque loads during in-flight engine starts.



LEGEND

CDC	Control and Distribution Cabinet
EDP	Engine-Driven Pump
EEC	Electronic Engine Control
SOV	Shutoff Valve

NOTE

1 Depressurization function enabled.

Figure 31: Engine Driven Pump Depressurization

AC MOTOR PUMP ELECTRICAL POWER CONTROL

The ACMP contactors are controlled by the bus power control units (BPCUs).

The BPCUs use ACMP ON enabling command signals to energize and close contactors located in the EPCs.

A 3-phase, 115 VAC is then directed by a dedicated AC BUS through a current transformer, located in the EPC, to its respective ACMP. The ACMP turns on when electrically powered. The BPCUs monitor for overcurrent protection through the current transformer.

To prevent overload, the BPCUs interface with the control and distribution cabinet (CDC) to stagger the startup of the ACMPs during simultaneous starts. The startup time delay is between 2.0 to 2.5 seconds.

The order of startup priority in AUTO selection is:

1. ACMP 3A
2. ACMP 3B
3. ACMP 2B

Load shedding priority provided by the CDCs is:

1. ACMP 3A
2. ACMP 2B
3. ACMP 3B

NOTE

There are no circuit breakers or solid-state power controllers (SSPCs) for the ACMPs. If maintenance is to be performed on the ACMPs, the ACMP contactors can be controlled from the eCB page in a similar manner to an SSPC (see ATA 24). The contactor can be locked out when the aircraft is in the MAINTENANCE MODE. If a trip occurs, the contactor cannot be reset from the eCB page. The BPCU power must be cycled to restore power to the contactor. If a new BPCU is installed, the contactor is set to the OUT position and the contactor must be set to IN for ACMP operation.

AC Motor Pump Control

The AC motor pumps (ACMPs) rotary switch positions are sent to the control and distribution cabinets (CDCs) through the data concentrator unit module cabinets (DMCs).

The CDCs command the bus power control units (BPCUs) to energize contactors in their respective electrical power centers (EPCs). The contactors power the ACMPs by their respective AC BUSES.

As an example, when the 2B switch is selected to AUTO, and CDC 1 has commanded BPCU 1 to energize and close the motor pump contactor (MPC 2B), ACMP 2B is powered by AC BUS 1.

Hydraulic system pressure, quantity, temperature, and pump status is processed by the data concentrator unit module cabinets (DMCs) for display on the HYD synoptic page.

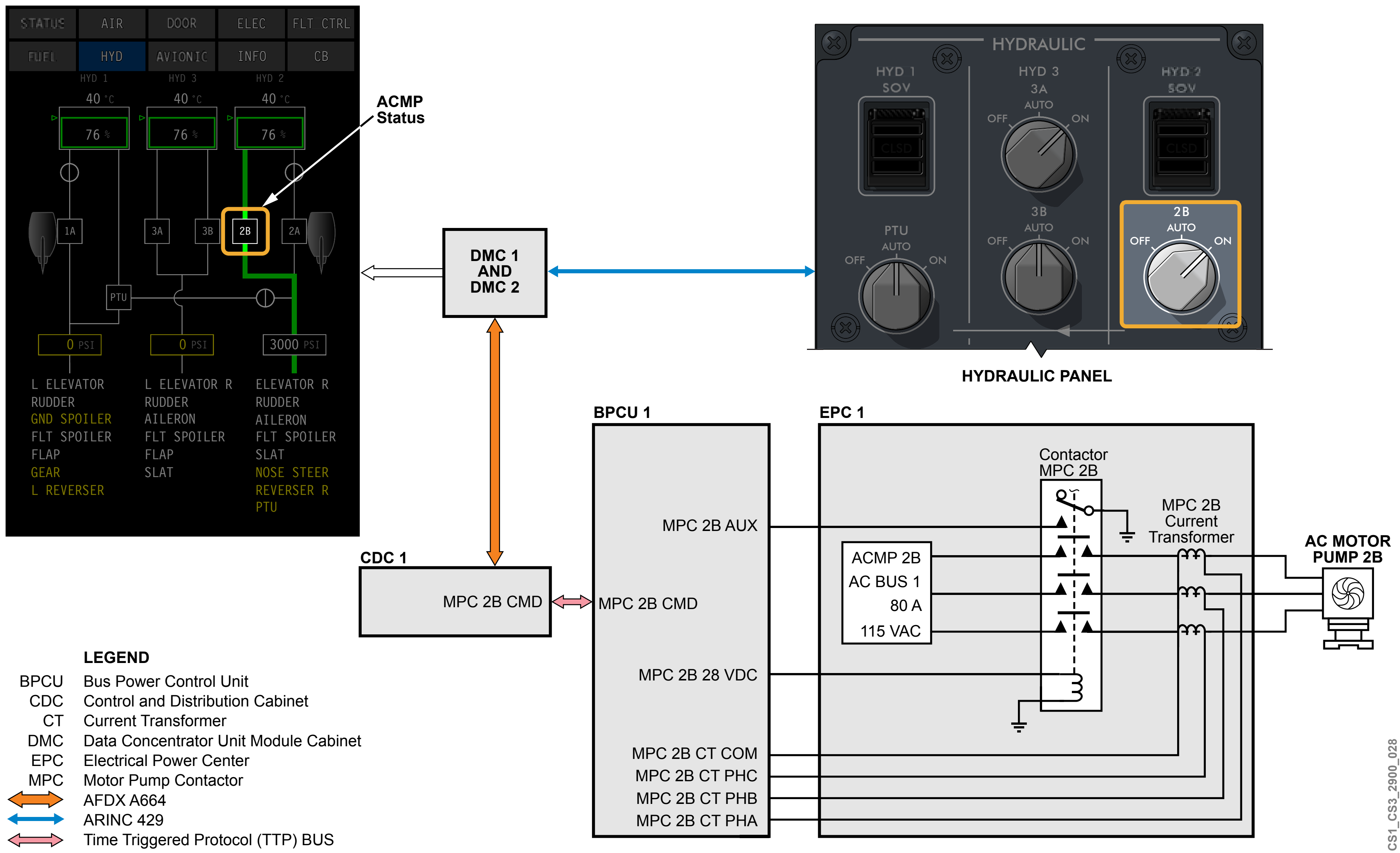


Figure 32: AC Motor Pump Electrical Power Control

ACMP LOAD SHEDDING

All three ACMPs can be operated when the aircraft is powered by two generators.

When only one generator is operational, load shedding is required to limit the electrical bus loading. The load shedding logic depends on:

- Electrical power source availability
- Hydraulic pump control switch position

External Power or APU Generator Only

Only two ACMPs can be operated at the same time when external power is used to power the aircraft. A third hydraulic pump cannot run, even if it is selected on.

Priority is always given to ACMP 2B since it can power the PTU. One hydraulic system no. 3 pump can also be selected on in addition to ACMP 2B. If ACMP 3A is selected ON, ACMP 3B is not available.

If ACMP 2B is not running, both ACMP 3A and ACMP 3B can be operated.

One Engine Generator

If the left or right generator has failed, but both engines are running and there is hydraulic pressure available from both engine-driven pumps, ACMP 2B is load shed. If the ACMP 2B switch is in AUTO or ON at the time of a load shed and the pump is running, ACMP 2B continues to be powered.

ACMP 3A and ACMP 3B are enabled to ensure all three hydraulic systems have a source of power. If ACMP 3A is operating at the time of a load shed, it remains operating and ACMP 3B is load shed. (Refer to Table 5).

Table 4: AC Motor Pump Load Shed - APU or External Power Only

ACMP 2B	ACMP 3A	ACMP 3B
ON	ON	OFF
ON	OFF	ON
OFF	ON	ON

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Table 5: AC Motor Pump Load Shed - One Engine Generator Only, Two Engines, and Two EDPs Running

ACMP 2B SWITCH IN AUTO OR ON AND PUMP OPERATING	ACMP 3A SWITCH IN AUTO OR ON AND PUMP OPERATING	ACMP 2B LOAD SHED	ACMP 3A LOAD SHED	ACMP 3B LOAD SHED
NO	NO	YES	NO	NO
NO	YES	YES	NO	NO
YES	NO	NO	YES	NO
YES	YES	NO	NO	YES

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POWER TRANSFER UNIT AND AC MOTOR PUMP CONTROL LOGIC

Power Transfer Unit Control

Hydraulic system no. 1 is normally powered by the engine-driven pump (EDP). The EDP operates when the engine is running. A power transfer unit (PTU) provides an additional source of hydraulic power to the system during certain flight phases, or when the EDP fails.

When the PTU rotary switch is in the ON position, the PTU runs continuously. When the PTU rotary switch is in AUTO position, the PTU runs during certain flight phases or EDP failure.

PTU ON Logic

The PTU ON logic requires:

- PTU rotary switch in the ON position
- Hydraulic system no. 1 quantity greater than 5%

PTU AUTO Logic

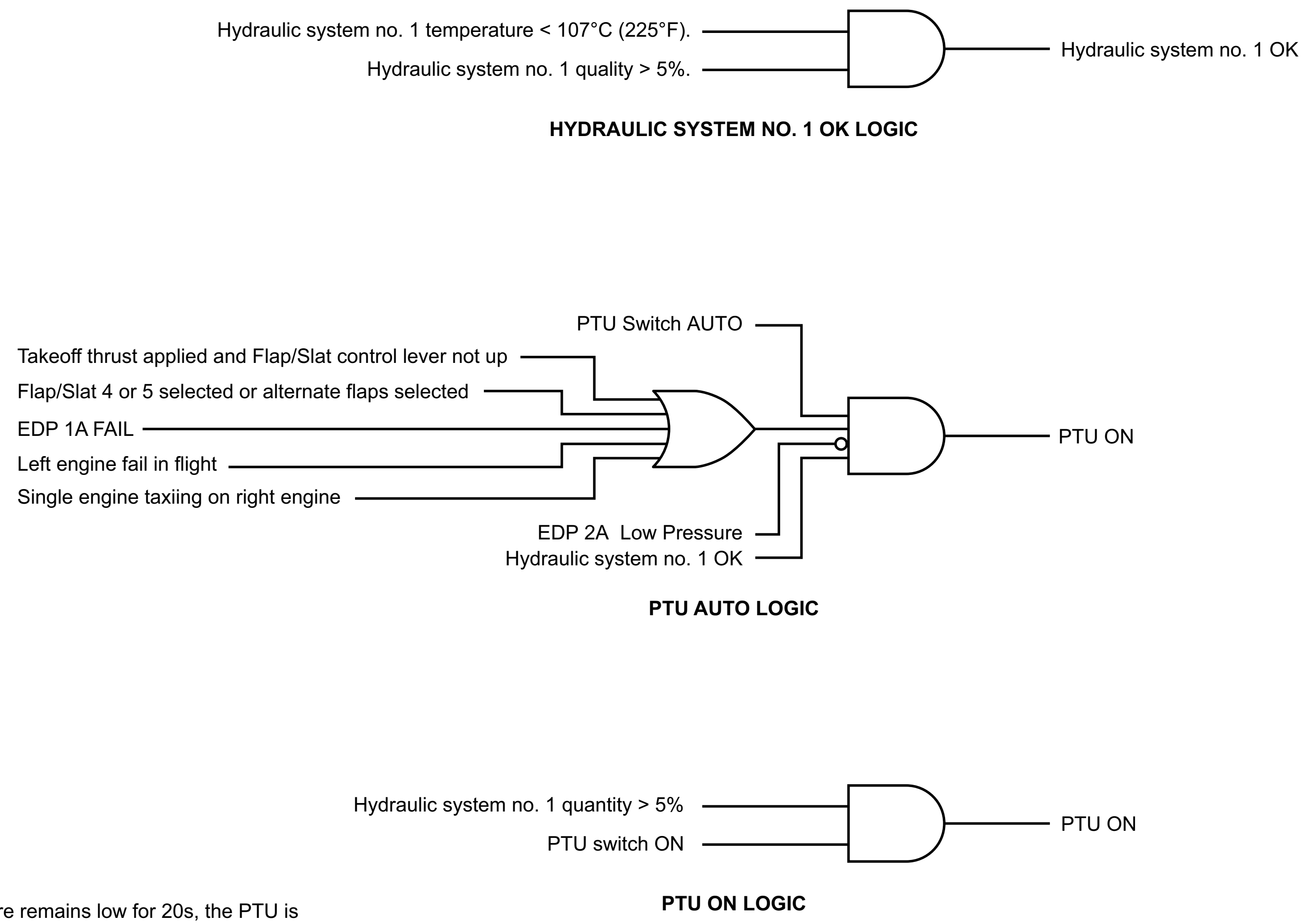
For AUTO operation, the following conditions are monitored:

- PTU rotary switch in AUTO
- Hydraulic system no. 1 OK which requires:
  - Hydraulic system no. 1 fluid temperature less than 107°C (225°F)
  - Hydraulic system no. 1 fluid quantity greater than 5%
- EDP 2A pump pressure

If the PTU ON logic and PTU AUTO logic conditions are met, the PTU turns on when:

- Takeoff
  - Takeoff thrust is applied. The throttle levers move past 20° throttle lever angle. The PTU runs until the FLAP/SLAT CONTROL LEVER (FSCL) is at 0 and the flap/slat position agrees with the FSCL
- Approach
  - The FSCL is moved to FLAPS 4, or FLAPS 5 or ALTN FLAP is selected in flight. The PTU turns off when the FSCL is at 0 and the flap/slat position agrees with the FSCL
- EDP 1A low-pressure with the left engine running
  - The PTU turns on and continues operating as long as the EDP 1A pressure is low. If the pressure increases, the PTU runs for an additional 2 minutes before turning off
- Left engine failure in flight
- Single engine taxi using the right engine. The PTU turns on when:
  - EDP 1A pressure is low
  - Right engine is running
  - Parking brake is selected OFF. If the parking brake is required during taxi, the PTU single engine taxi logic keeps the PTU running for 6 minutes after the parking brake is selected ON to prevent the cycling of the PTU.

If the PTU does not produce pressure within 20 seconds in the AUTO mode, the PTU is latched off. To reset the PTU operation, the PTU rotary switch must be selected to the OFF position.



NOTE

If PTU pressure remains low for 20s, the PTU is latched off. The PTU resets when the PTU switch is selected OFF.

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Figure 33: Power Transfer Unit Control

AC Motor Pump 2B Control

Hydraulic system no. 2 is normally powered by the engine-driven pump (EDP). The EDP operates when the engine is running. An ACMP provides an additional source of hydraulic power during certain flight phases to meet high demand or when the EDP fails.

When the PTU rotary switch is in the ON position, ACMP 2B runs at all times. When the PTU rotary switch is in AUTO position, ACMP 2B runs during certain flight phases or failure conditions.

ACMP 2B ON Logic

ACMP 2B ON logic requires:

- ACMP 2B switch selected to ON
- ACMP 2B temperature is less than 190°C (374°F)

If these conditions are met and the ACMP 2B pump is not load shed, the pump is commanded on.

ACMP 2B AUTO Logic

For AUTO operation, the following conditions are monitored:

- ACMP 2B switch in AUTO position
- External power not in use
- ACMP 2B is not load shed
- Hydraulic system no. 2 OK:
  - Hydraulic system no. 2 temperature less than 107°C (225°F)
  - Hydraulic system no. 2 quantity greater than 5%
  - ACMP 2B temperature less than 190°C (374°F)

If hydraulic systems no. 1 and no. 3 indicate low pressure and hydraulic system no. 2 is not OK, ACMP 2B is not disabled.

The system 107°C (225°F) temperature logic can be reset by moving the ACMP 2B rotary switch out of and back into AUTO while the right engine is running.

If the ACMP 2B ON logic or ACMP 2B AUTO logic conditions are met, ACMP 2B turns on when:

- PTU is commanded on
- Takeoff
  - Takeoff thrust is applied. The throttle levers move past 20° throttle lever angle. ACMP 2B runs until the FLAP/SLAT CONTROL LEVER (FSCL) is at 0 and the flap/slat position agrees with the FSCL
- Approach
  - The FLAP/SLAT CONTROL LEVER (FSCL) is moved to FLAPS 4, or FLAPS 5, or alternate flaps are selected in flight. ACMP 2B turns off when the FSCL is at 0 and the flap/slat position agrees with the FSCL
- Right engine failure in flight
- EDP 2A low-pressure with the right engine running
  - ACMP 2B turns on and continues operating as long as the EDP 2A pressure is low and the right engine is running. If the pressure recovers or the right engine is not running, ACMP 2B runs for an additional 2 minutes before turning off.
- Single engine taxi using the left engine. ACMP 2B turns on to provide nosewheel steering when:
  - EDP 2A pressure is low
  - Left engine is running
  - Parking brake is selected OFF. If the parking brake is required during taxi, the ACMP 2B single engine taxi logic keeps the ACMP 2B running for 6 minutes after the parking brake is selected ON to prevent cycling of the ACMP 2B.

If an ACMP 2B overheat occurs in flight, the pump is latched off for the remainder of the flight. The pump can be reset on the ground when the power is cycled.

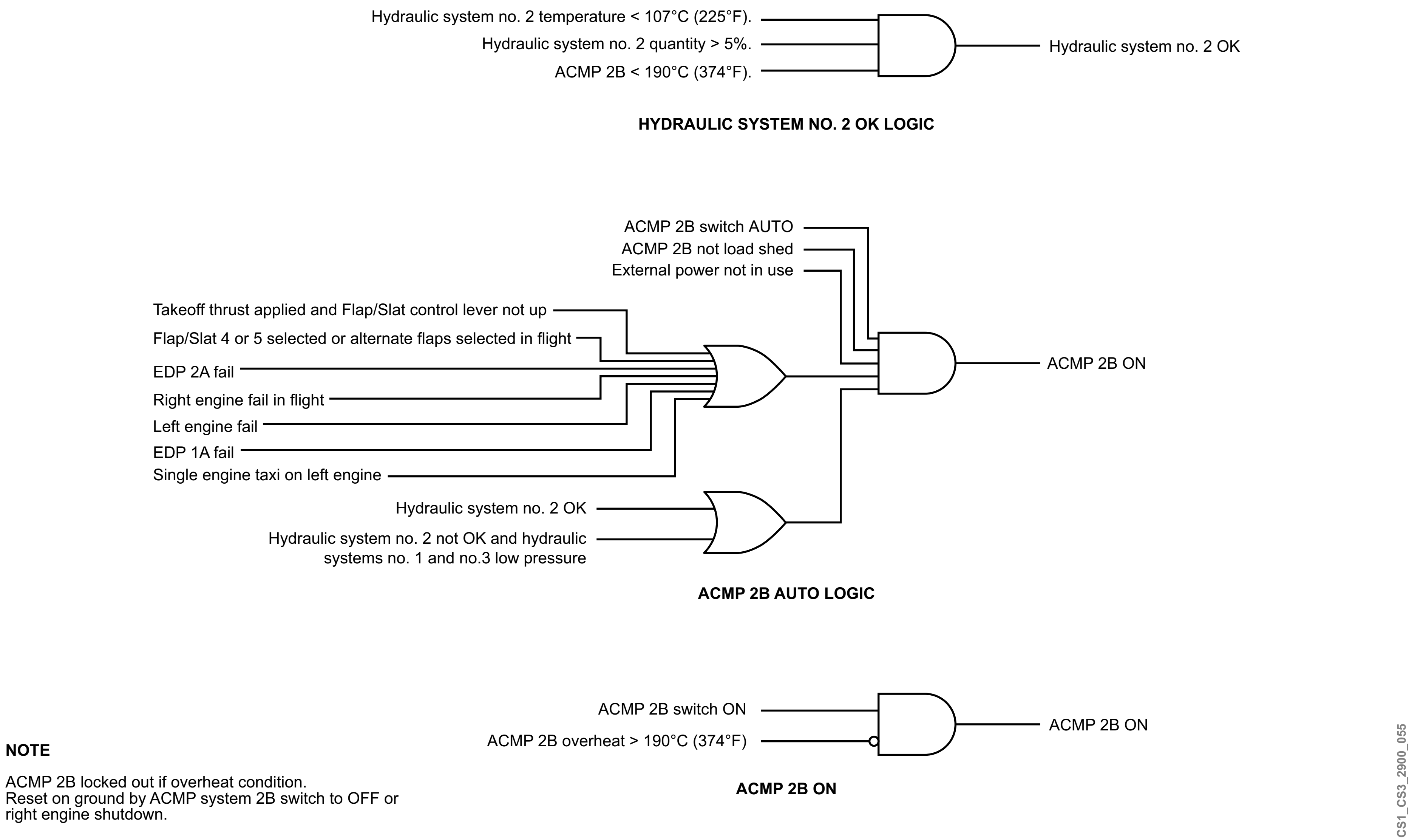


Figure 34: AC Motor Pump 2B Control

AC Motor Pump 3A Control

Hydraulic system no. 3 is normally powered by ACMP 3A. ACMP 3B provides an additional source of hydraulic power when the system demand is high during certain flight phases or when ACMP 3A fails.

ACMP 3A ON Logic

ACMP 3A ON logic requires:

- ACMP 3A switch selected to ON
- ACMP 3A temperature is less than 190°C (374°F)

If these conditions are met and the ACMP 3A pump is not load shed, the pump is commanded on.

ACMP 3A AUTO Logic

For AUTO operation, the following conditions are monitored:

- ACMP 3A switch in AUTO position
- External power not in use
- ACMP 3A pump is not load shed
- ACMP 3A temperature less than 190°C (374°F)
- Hydraulic system no. 3 temperature less than 107°C (225°F)
- Hydraulic system no. 3 less than 3350 psi
- ACMP 3B low-pressure
- Hydraulic system no. 3 OK:
    - Hydraulic system no. 3 quantity greater than 5%

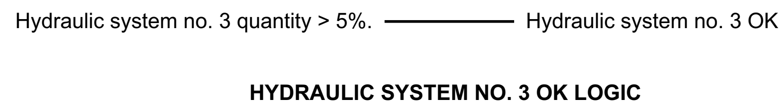
If hydraulic systems no. 1 and no. 3 indicate low pressure and hydraulic system no. 2 is not OK, ACMP 2B is not disabled.

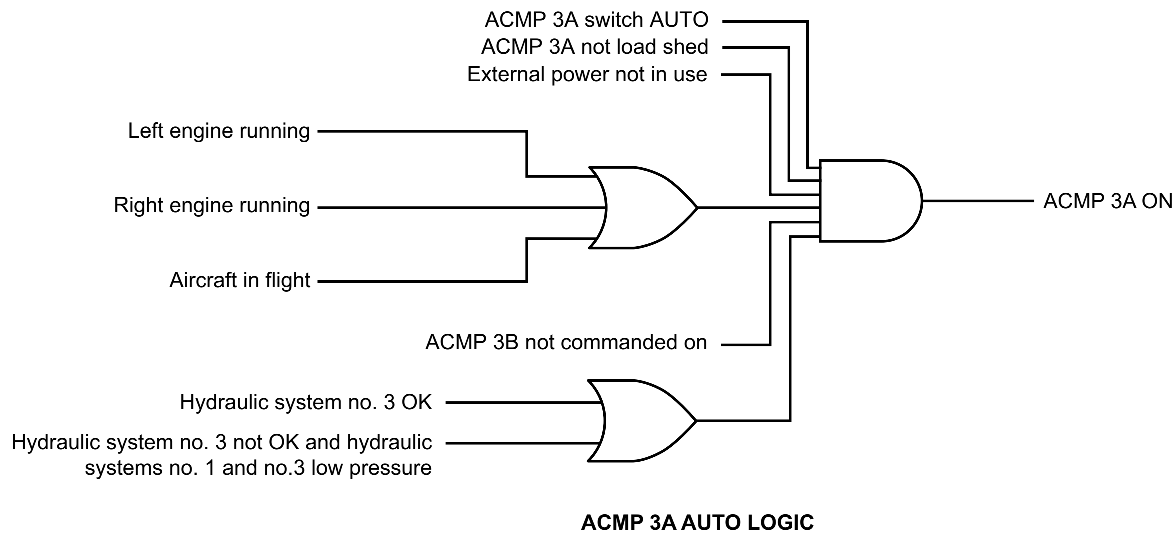
- Hydraulic system no. 3 temperature less than 125°C (257°F)
- ACMP 3B is not on. If ACMP 3B is turned on first and is producing pressure, ACMP 3A does not run

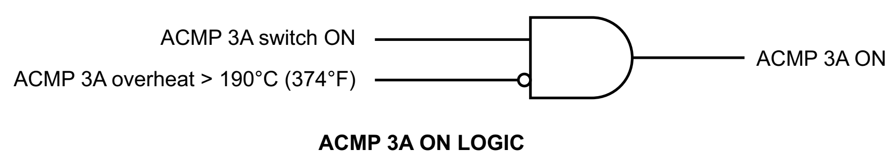
High system temperatures in hydraulic system no. 1 and hydraulic system no. 2 are prevented by the logic that closes the firewall shutoff valves. There is no firewall shutoff valve in hydraulic system no. 3, therefore additional logic is provided to turn off ACMP 3A when the system temperature reaches 125°C (257°F)

If the above conditions are met, ACMP 3A turns on for any of the following conditions:

- Left engine is running
- Right engine is running
- Aircraft is in flight







NOTE

An ACMP 3A overheat latches the pump off and it cannot be reset in flight.

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Figure 35: AC Motor Pump 3A Control

AC Motor Pump 3B Control

Hydraulic system no. 3 is normally powered by ACMP 3A. ACMP 3B provides an additional source of hydraulic power to the system during certain flight phases or when ACMP 3A fails.

ACMP 3B ON Logic

ACMP 3B ON logic requires:

- ACMP 3B switch selected to ON
- ACMP 3B temperature is less than 190°C (374°F)

If these conditions are met and the ACMP 3B pump is not load shed, the pump is commanded on.

ACMP 3B AUTO Logic

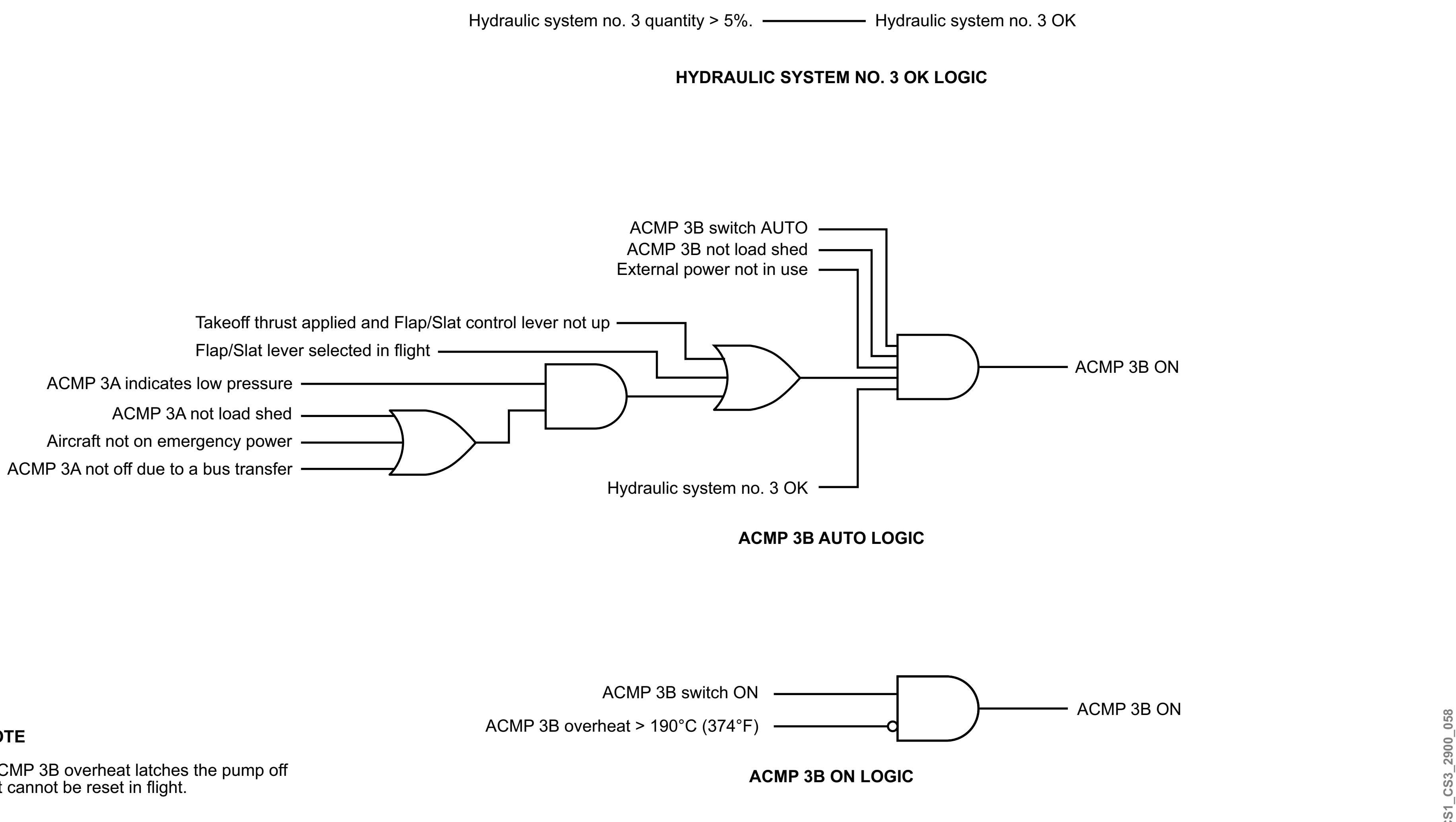
For AUTO operation, the following conditions are monitored:

- ACMP 3B rotary switch in AUTO position
- External power not in use
- ACMP 3B pump is not load shed
- Hydraulic system no. 3 OK:
  - Hydraulic system no. 3 quantity greater than 5%
- ACMP 3B temperature less than 190°C (374°F)
- Hydraulic system no. 3 temperature less than 107°C (225°F)
- Hydraulic system no. 3 less than 3350 psi
- ACMP 3A indicates low-pressure
- Hydraulic system no. 3 temperature less than 125°C (257°F)

High system temperatures in hydraulic system no. 1 and hydraulic system no. 2 are prevented by the logic that closes the firewall shutoff valves. In hydraulic system no. 3, there is no firewall shutoff valve, therefore additional logic is provided to turn off ACMP 3B when the system temperature reaches 125°C (257°F)

If the above conditions are met, ACMP 3B turns on to provide additional power during periods of high demand when:

- On the ground with the FSCL not at 0 and the flaps or slats in motion
- Takeoff
  - Takeoff thrust is applied. The throttle levers move past 20° throttle lever angle. ACMP 3B runs until the FLAP/SLAT CONTROL LEVER (FSCL) is at 0 and the flap/slat position agrees with the FSCL
- Descent
  - FSCL not at 0 in flight
- Approach
  - The FSCL is moved to Flaps 4, or Flaps 5, or alternate flaps are selected in flight. ACMP 3B turns off when the FSCL is at 0 and the flap/slat position agrees with the FSCL
- ACMP 3A indicates low-pressure and the following conditions have been true for 28 seconds:
  - Aircraft not on emergency power
  - ACMP 3A not off due to a bus transfer
  - ACMP 3A not load shed



NOTE

An ACMP 3B overheat latches the pump off and it cannot be reset in flight.

Figure 36: AC Motor Pump 3B Control

MONITORING AND TESTS

The following pages provide the crew alerting system (CAS) and INFO messages for the hydraulic system.

CAS MESSAGES

Table 6: CAUTION Messages

MESSAGE	LOGIC
HYD 1 SOV FAIL	HYD 1 SOV failed to close when required or is open when supposed to be closed.
HYD 2 SOV FAIL	HYD 2 SOV failed to close when required or is open when supposed to be closed.
HYD EDP 1A FAIL	HYD 1 low-pressure switch activated when left SOV is not closed.
HYD EDP 2A FAIL	HYD 2 low-pressure switch activated when right SOV is not closed.
HYD PTU FAIL	PTU low-pressure detected when PTU is required to be online.
HYD PUMP 2B FAIL	ACMP 2B low-pressure when pump required to be online.
HYD PUMP 3A FAIL	ACMP 3A low-pressure when pump required to be online.
HYD PUMP 3B FAIL	ACMP 3B low-pressure when pump required to be online.
HYD RAT PUMP FAIL	RAT deployed and hydraulic system no. 3 pressure is less than 1800 psi.

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Table 7: ADVISORY Messages

MESSAGE	LOGIC
HYD 1 SOV CLSD	HYD 1 SOV is closed automatically by the ENG FIRE PBA or the overheat relay.
HYD 2 SOV CLSD	HYD 2 SOV is closed automatically by the ENG FIRE PBA or the overheat relay.
HYDRAULIC FAULT	Loss of non-critical functions or loss of redundancy in the hydraulic systems (refer to INFO messages).

Table 8: STATUS Messages

MESSAGE	LOGIC
HYD 1 SOV CLSD	SOV 1 manually closed by the HYD 1 SOV PBA on the HYDRAULIC panel.
HYD 2 SOV CLSD	SOV 2 manually closed by the HYD 2 SOV PBA on the HYDRAULIC panel.
HYD PTU OFF	PTU switch at OFF position. No high-pressure indicated by the PTU pressure switch.
HYD PTU ON	PTU switch at ON position. High-pressure detected by the PTU pressure switch.
HYD PUMP 2B OFF	ACMP 2B switch is at the OFF position. No high-pressure is detected by the ACMP 2B pressure switch.
HYD PUMP 2B ON	ACMP 2B switch is in the ON position. High-pressure detected by the ACMP 2B pressure switch.
HYD PUMP 3A OFF	ACMP 3A switch is in the OFF position. No high-pressure detected by the ACMP 3A pressure switch.
HYD PUMP 3A ON	ACMP 3A switch is in the ON position. High-pressure detected by the ACMP 3A pressure switch.

Table 8: STATUS Messages (Continued)

MESSAGE	LOGIC
HYD PUMP 3B OFF	ACMP 3B switch is in the OFF position. No high-pressure detected by the ACMP 3B pressure switch.
HYD PUMP 3B ON	ACMP 3B switch is in the ON position. High-pressure detected by the ACMP 3B pressure switch.

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Table 9: INFO Messages

MESSAGE	LOGIC
29 HYDRAULIC FAULT - HYD 1 FWSOV SNSR INOP	The firewall shutoff valve is indicating both open and closed or the open or closed signal is not valid.
29 HYDRAULIC FAULT - HYD 2 FWSOV SNSR INOP	The firewall shutoff valve is indicating both open and closed or the open or closed signal is not valid.
29 HYDRAULIC FAULT - HYD EDP 1A DEPRESS INOP	EDP 1A is indicating high-pressure when the depress solenoid had been commanded ON and the engine speed is low and the hydraulic fluid is warmer than 10°C (50°F).
29 HYDRAULIC FAULT - HYD EDP 2A DEPRESS INOP	EDP 2A is indicating high-pressure when the depress solenoid has been commanded ON and engine speed is low, and the hydraulic fluid is warmer than 10°C (50°F).
29 HYDRAULIC FAULT - HYD 3 ACCUM 1 PRESS DEGRADED - HYD 3 ACCUM 2 PRESS DEGRADED	The HYD 3 ACCUM 1(2) PRESS FAIL message is active when the system no. 3 quantity okay and either ACMP 3A or 3B producing pressure and the difference between hydraulic system no. 3 pressure and the accumulator pressure is greater than 500 psi. The message is also active on the ground when this difference in pressure is less than 896 psi or greater than the 1496 psi with both pumps indicating low-pressure and hydraulic system no. 3 pressure is less than 300 psi.

Table 9: INFO Messages (Continued)

MESSAGE	LOGIC
29 HYDRAULIC FAULT - HYD PUMP 2B INOP	ACMP 2B overheat has occurred since last DMC power-up. The HYD PUMP 2B INOP INFO message and HYDRAULIC FAULT CAS appears and remains while aircraft is on ground or aircraft speed is below 50 kt.
29 HYDRAULIC FAULT - HYD PUMP 3A INOP	ACMP 3A overheat has occurred since last DMC power-up. The HYD PUMP 3A INOP INFO message and HYDRAULIC FAULT CAS appears and remains while aircraft is on ground, or aircraft speed is below 50 kt.
29 HYDRAULIC FAULT - HYD PUMP 3B INOP	ACMP 3B overheat has occurred since last DMC power-up. The HYD PUMP 3B INOP INFO message and HYDRAULIC FAULT CAS appears and remains while aircraft is on ground, or aircraft speed is below 50 kt.
29 HYDRAULIC FAULT - HYD PTU INOP	PTU FAIL CAS message has been set since last DMC power-up. The HYD PTU INOP INFO message and HYDRAULIC FAULT CAS appears and remains while aircraft is on ground, or aircraft speed is below 50 kt.

PRACTICAL ASPECTS

RESERVOIR BLEEDING

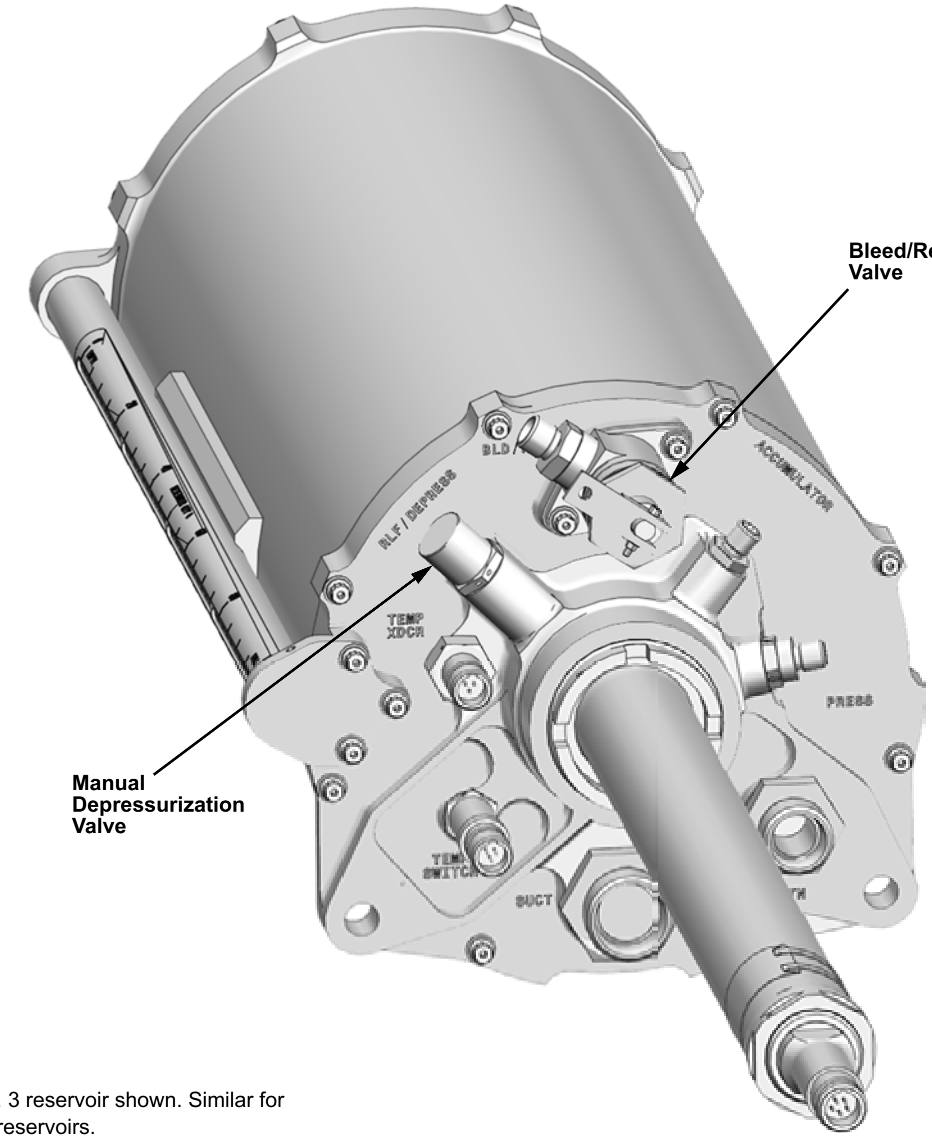
1. Pressurize the hydraulic system by operating the respective AC motor pump (ACMP) or power transfer unit (PTU).
2. Lift the lever of the reservoir bleed/relief valve to the open position for 3 seconds. Pressurized fluid and trapped air in the reservoir, drains into the ecology bottle.
3. Return the bleed/relief valve lever to its closed seated position.
4. Record the hydraulic visual quantity and compare with the quantity displayed on the synoptic page. Visual quantity should be within 5% of the synoptic page value.
5. Turn OFF the respective hydraulic pump (ACMP/PTU).
6. This procedure can be repeated to ensure that air has been removed from the system. However the reservoir must be refilled and the level should not be allowed to drop below 20% when bleeding the system.
7. Empty the ecology bottle.

NOTE

Refer to Aircraft Maintenance Publication (AMP) ATA 12 for proper procedure.

HYDRAULIC SYSTEM DEPRESSURIZATION

1. Ensure that the hydraulic system pumps (ACMP/PTU) are turned OFF.
2. Press and hold the manual depressurization valve for at least 20 seconds.
3. Release the manual depressurization valve.
4. Ensure that the manual depressurization valve has returned to the normal closed position.
5. Lift the reservoir bleed/relief valve lever for a couple of seconds to dump residual bootstrap reservoir pressure.
6. Return the bleed/relief valve lever to the closed seated position.
7. Empty the ecology bottle.



NOTE

Hydraulic system no. 3 reservoir shown. Similar for all hydraulic system reservoirs.

Figure 37: Reservoir Bleeding and Hydraulic System Pressure Depletion

HYDRAULIC SYSTEM SERVICING

Reservoir Fluid Level Check

1. Ensure that the landing gear safety pins are installed, thrust reversers are stowed, and that all spoilers are stowed.
2. Gain access to the hydraulic system reservoir and service panel.
3. Deplete hydraulic system pressure by pushing and holding the high-pressure relief and manual depressurization valve knob for at least 20 seconds.
4. Record hydraulic system fluid level indicated on the hydraulic synoptic page.
5. Turn ON the respective hydraulic pump (ACMP/PTU) to pressurize the system, and record the fluid level indicated on the hydraulic synoptic page.
6. For hydraulic system no. 1, a difference greater than 10% between the two recordings indicates the presence of air in the system. For hydraulic system no. 2, 12% is the maximum limit. A difference greater than 30% between the two recordings applies to hydraulic system no. 3.
7. If needed, carry out bleeding of the hydraulic system, by pressurizing the system with the hydraulic pump (ACMP/PTU) and manually actuating the bleed/relief valve for approximately 3 seconds.
8. Record the hydraulic system fluid level indicated on the hydraulic synoptic page after bleeding has been completed.
9. Turn OFF the hydraulic pump (ACMP/PTU).
10. Deplete the hydraulic system pressure by manually actuating the high-pressure relief and manual depressurization valve for at least 20 seconds.
11. Record hydraulic system reservoir fluid level indicated on the hydraulic synoptic page.
12. Calculate the difference between this reading and the last recorded reading after bleeding is performed.
13. Refer to the fluid level chart to verify that the change in level and level limits are met.
14. Service the hydraulic system with fluid if the fluid level is not in accordance with the fluid level chart.

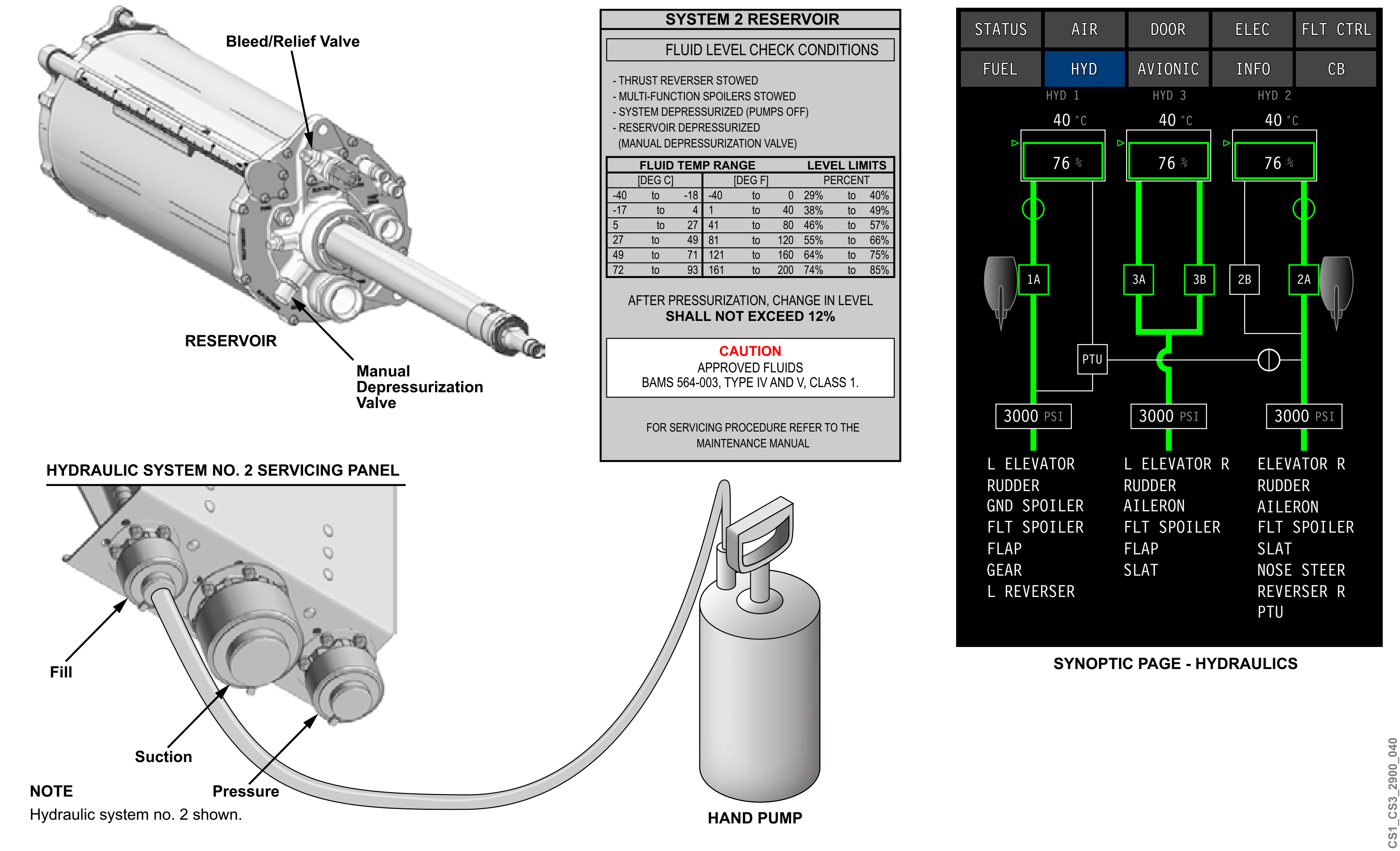


Figure 38: Hydraulic Reservoir Fluid Level Check

HYDRAULIC SYSTEM NO. 3 NITROGEN ACCUMULATORS PRESSURE CHECK AND SERVICING

1. Depressurize hydraulic system no. 3
2. With the system depressurized, press and hold high-pressure relief and manual depressurization valve for at least 20 seconds
3. Allow pressure in nitrogen accumulators to stabilize for 10 minutes
4. Check that the nitrogen accumulator gauges display pressure in accordance with the gas pressure chart
5. If the pressure is low, service the nitrogen accumulators with a nitrogen cart
6. Remove the dust cap from the charging valve. Connect the nitrogen cart hose to the nitrogen accumulator charging valve
7. Slowly adjust the nitrogen cart regulator pressure to chart value corresponding to present temperature
8. Slowly open the charging valve to fill the nitrogen accumulator to the pressure indicated on the chart
9. Close the charging valve when the nitrogen accumulator gauge pressure meets chart tolerances
10. Let the nitrogen accumulator pressure stabilize for a few minutes. Verify that the pressure indicated on the accumulator gauge is within chart tolerances
11. Close and disconnect the nitrogen cart from the nitrogen accumulator charging valve. Reinstall the dust cap on the charging valve

NOTE

Refer to Aircraft Maintenance Publication (AMP) ATA 12 for proper procedure.

SYSTEM 3 ACCUMULATOR

PRE-CHARGE CHECK CONDITIONS

- SYSTEM DEPRESSURIZED (PUMPS OFF)
- RESERVOIR DEPRESSURIZED (MANUAL DEPRESSURIZATION VALVE)
- LET THE NITROGEN PRESSURE BECOME STABLE
- CHECK PRE-CHARGE ON THE PRESSURE GAUGE

AMBIENT TEMPERATURE RANGE		GAS PRESSURE
[DEG C]	[DEG F]	[PSIG]
-40 to -29	-40 to -20	1162 to 1426
-29 to -18	-20 to 0	1215 to 1494
-18 to -7	0 to 20	1268 to 1562
-7 to -4	20 to 40	1321 to 1630
4 to 16	40 to 60	1374 to 1698
16 to 27	60 to 80	1426 to 1766
27 to 38	80 to 100	1479 to 1834
38 to 49	100 to 120	1532 to 1902
49 to 60	120 to 140	1585 to 1970

CAUTION

USE NITROGEN ONLY

FOR SERVICING PROCEDURE REFER TO THE MAINTENANCE MANUAL

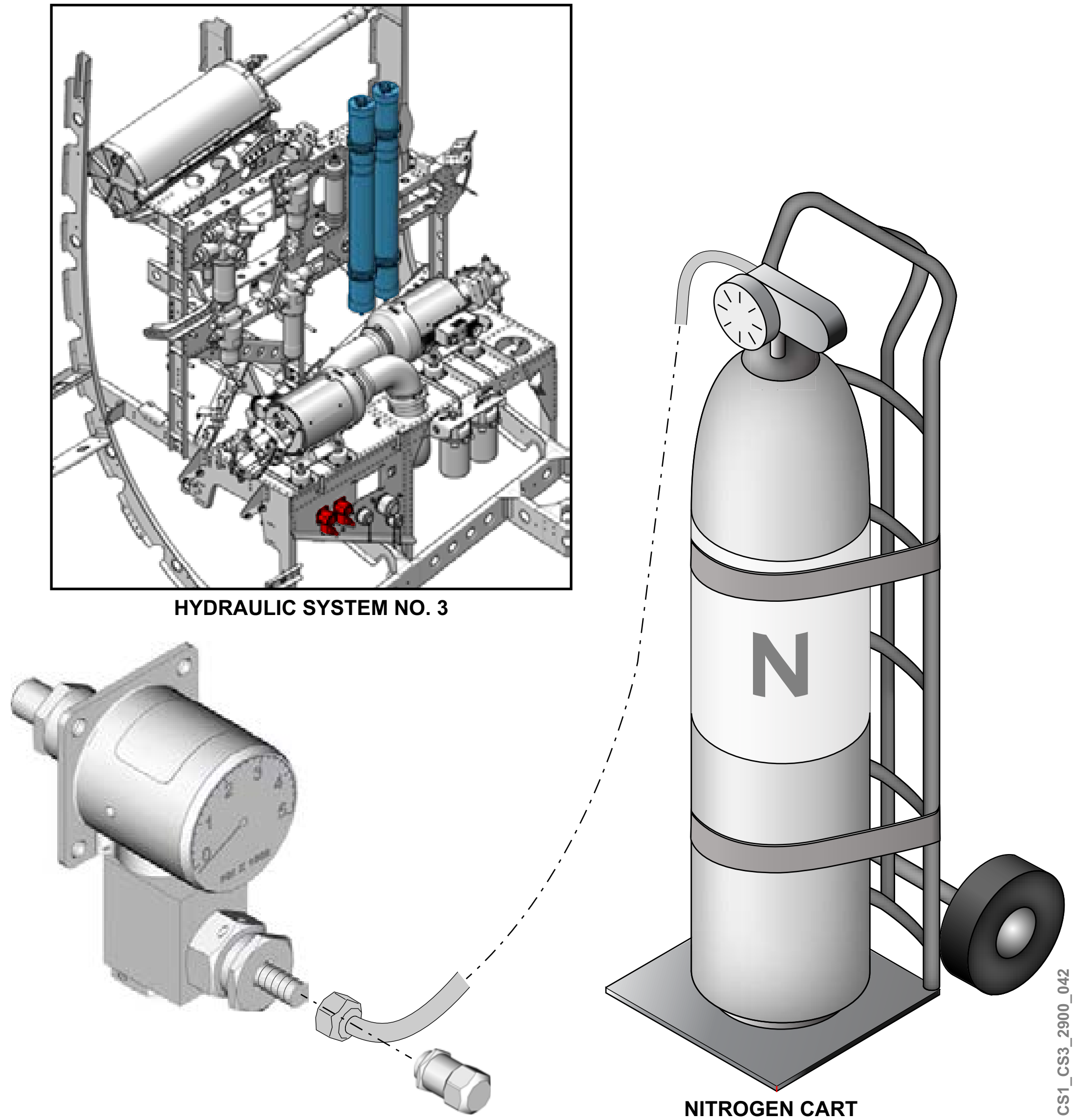


Figure 39: Hydraulic System No. 3 Nitrogen Accumulator Servicing

HYDRAULIC PUMP DEPRESSURIZATION TEST

The hydraulic pump depressurization (HPD) Test checks the ability of the EEC to command the depressurization of the engine hydraulic pump.

The EEC commands:

- A dry motor
- Confirms hydraulic pump is pressurized
- Commands depressurization
- Confirms the hydraulic pump is depressurized
- Resets the hydraulic pump
- Discontinues the dry motor

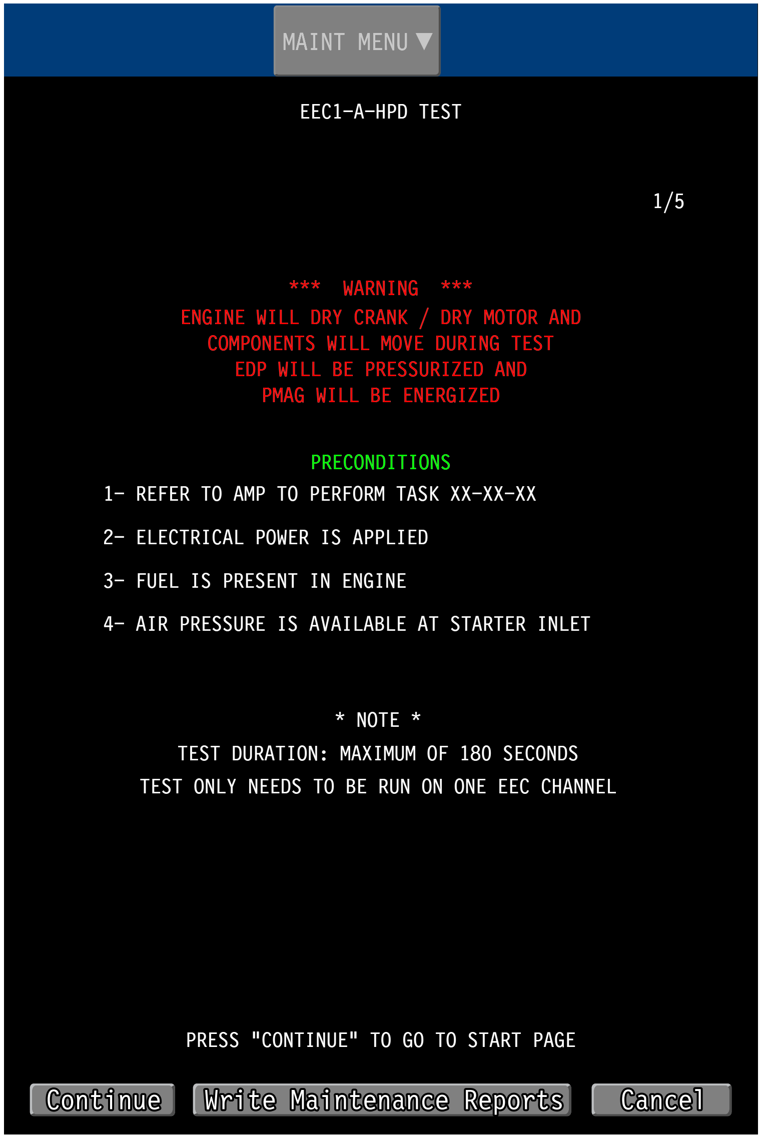


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Figure 40: Hydraulic Pump Depressurization Test

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ATA 32 - Landing Gear

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

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LANDING GEAR - CHAPTER BREAKDOWN

32-10 MAIN LANDING GEAR AND DOORS

GENERAL DESCRIPTION

The principal components of each main landing gear (MLG) assembly consist of a single-stage oleopneumatic-type shock strut with a folding side brace and associated downlock mechanism.

The shock strut consists of an upper main fitting and an inner sliding piston axle assembly.

The shock strut main fitting has an uplock roller, as well as fittings for the doors, pintle pins, and the side brace. The piston axle assembly has the axle at its lower end to support the wheel and brake assemblies. Torque links are installed between the strut main fitting and the piston axle assembly. A shimmy damper is mounted at the pivot point of the upper and lower torque links. Each MLG shock strut is attached to the wing structure by fore and aft pintle pins.

The folding side brace assembly is attached to the shock strut at its lower end, and the wing fitting at its upper end. The side brace supports and locks the MLG in the extended position.

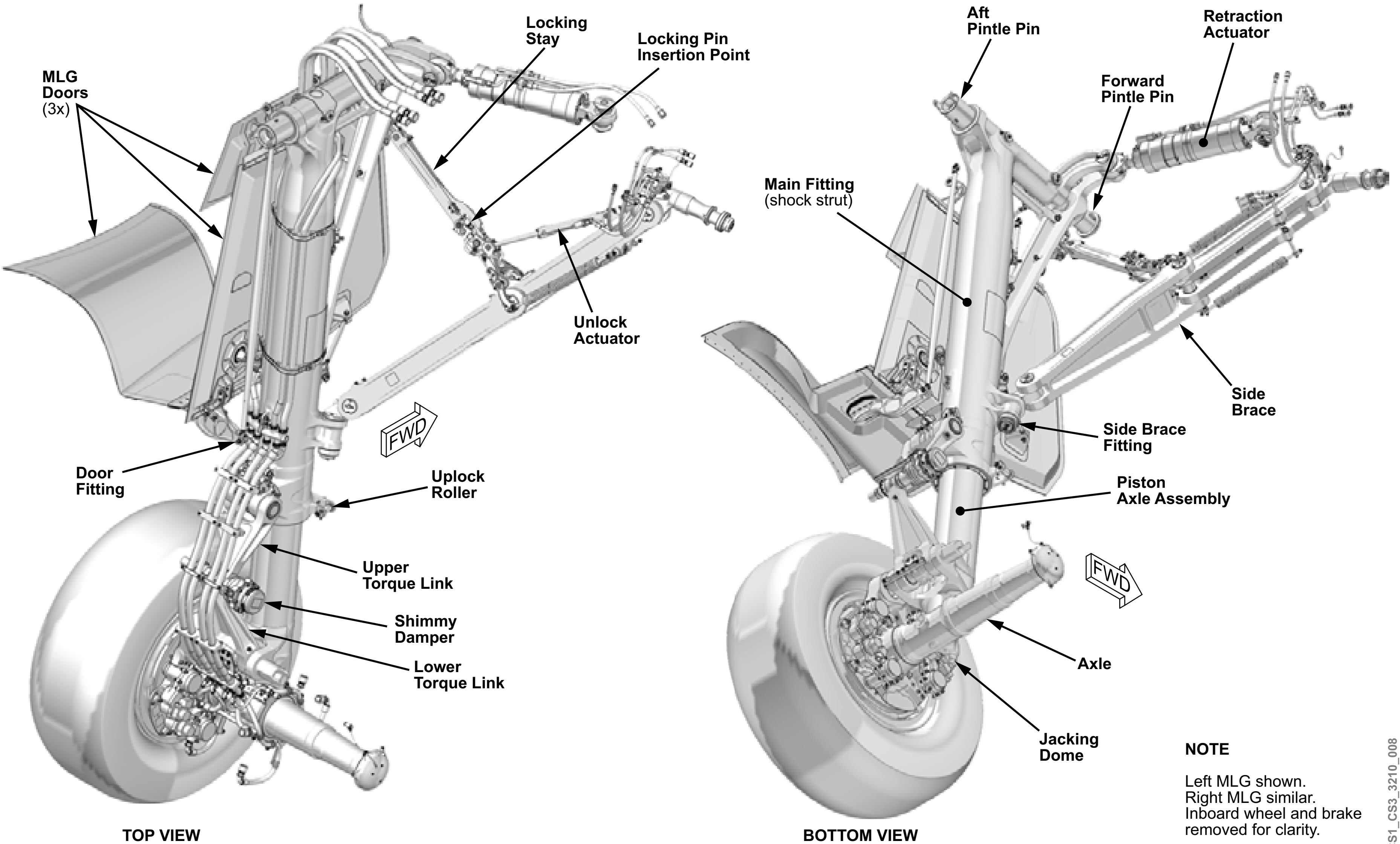
A hinged locking stay mechanism maintains the side brace overcenter lock when the MLG is extended. A hydraulic unlock actuator breaks the locking stay overcenter lock during MLG retraction. This breaks the side brace overcenter lock, allowing the side brace to fold and the MLG to retract.

The locking stay has an insertion point for a locking pin. The locking pin is used to lock the MLG in the down and locked position.

A hydraulic retraction actuator is attached between the shock strut upper pivot point and the wing mounting point. During retraction, the MLG moves inboard and into the wheel well. The MLG bay is enclosed by three mechanically-linked doors when the gear is retracted.

The axle has a jacking dome at its base for individual strut jacking to facilitate wheel and brake changes.

The MLG on the CS300 has been strengthened to support the heavier aircraft. The MLG on the CS300 is approximately 68 kg (150 lb) heavier.



NOTE

Left MLG shown.
Right MLG similar.
Inboard wheel and brake removed for clarity.

Figure 1: Main Landing Gear

DETAILED COMPONENT INFORMATION

SHOCK STRUT

The main fitting inner bore forms the upper part of the shock absorber upper cylinder. The piston axle assembly forms the lower part of the shock absorber sliding tube.

The shock strut is filled with nitrogen and hydraulic oil. The damping performance is optimized by means of a metering pin that restricts the oil passage as it passes through an orifice in the shock strut.

A nitrogen-charging valve is located on the upper part of the main fitting. Nitrogen is stored under pressure in the cylinder tube. A shock strut hydraulic oil servicing valve is located above the changeover valve on the main fitting.

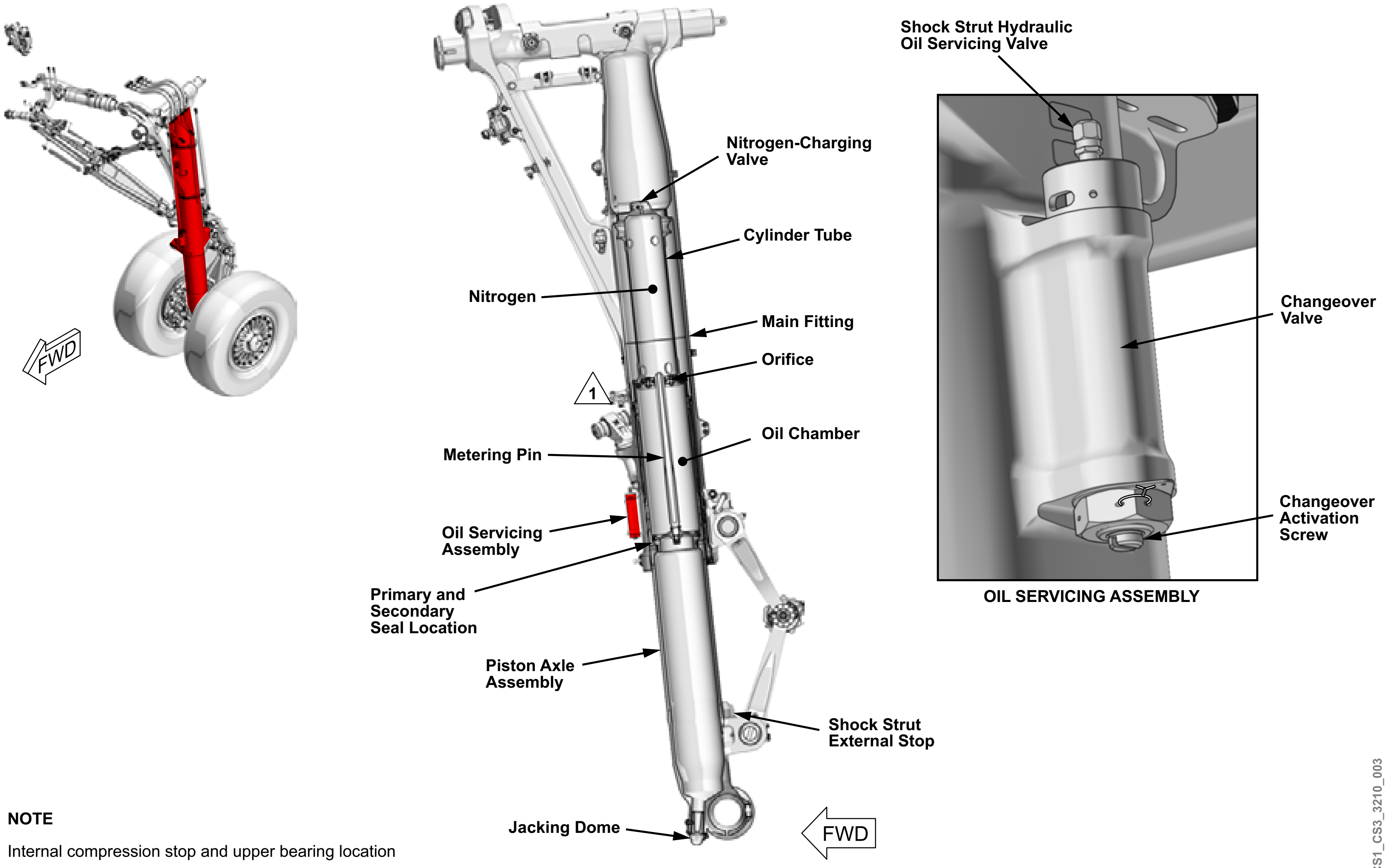
Two sets of primary and secondary seals are located between the main fitting lower bearing assembly and the sliding tube. These seals prevent oil leakage from the shock strut oil chamber.

Only one set of seals is active at a time. The seals are activated by high-pressure oil that passes through a changeover valve. In the event of primary seal leakage, the changeover valve can be manually activated. This action isolates the primary seal and energizes the secondary seal. Activation of the secondary seals is accomplished by rotating the changeover activation screw with a suitable tool until it bottoms out. Activation of the changeover valve to the secondary seals is considered a temporary shock strut leak rectification.

The piston axle sliding tube bearing surface and the wheel axle bearing surfaces have a special coating for heavy wear and high impact. Damage or wear on any of these surfaces must be evaluated in accordance with the Aircraft Maintenance Publication (AMP).

The shock strut is equipped with internal compression and extended stops. The shock strut is also equipped with a replaceable external stop located on the piston axle assembly above the lower torque link attachment. The external stop is used as a reference point when compressing the shock strut for fluid servicing.

A jacking dome is located at the bottom of the piston axle assembly to facilitate brake and wheel changes.



NOTE

1 Internal compression stop and upper bearing location

Figure 2: Shock Strut

TORQUE LINKS

The upper and lower torque links connect and keep the piston axle assembly centered with the main landing gear (MLG) main fitting.

The upper torque links are attached to the MLG main fitting with a pin that is locked in place by crossbolts. The shaft of the shimmy damper interconnects the two torque links. The upper torque link is secured to the shimmy damper housing with bolts. The lower torque link is secured to the shaft of the shimmy damper with a locking nut.

Steel targets are attached to the upper torque link. These targets are used by weight-off-wheels (WOFFW) proximity sensors to indicate the aircraft on ground, and aircraft in air conditions.

SHIMMY DAMPER

The shimmy damper dampens wheel and axle shimmy forces acting on the MLG main fitting.

The shimmy damper oil reservoir is located in the housing, and continues internally in the shaft. Damping orifices and damping chambers are located in the housing. Damping is achieved by forcing oil through a damping orifice from one damping chamber into another. The shimmy damper is equipped with a filling valve and a bleed point.

A spring-loaded oil reservoir cover avoids cavitation due to oil volume, oil temperature, and oil consumption changes.

A sight glass is located on the housing, along with a minimum and maximum level marking. A black indicator is displayed in the sight glass when the oil level is within limits. A red indicator is displayed in the sight glass when the oil level is too low. In this case, the shimmy damper must be replenished with oil.

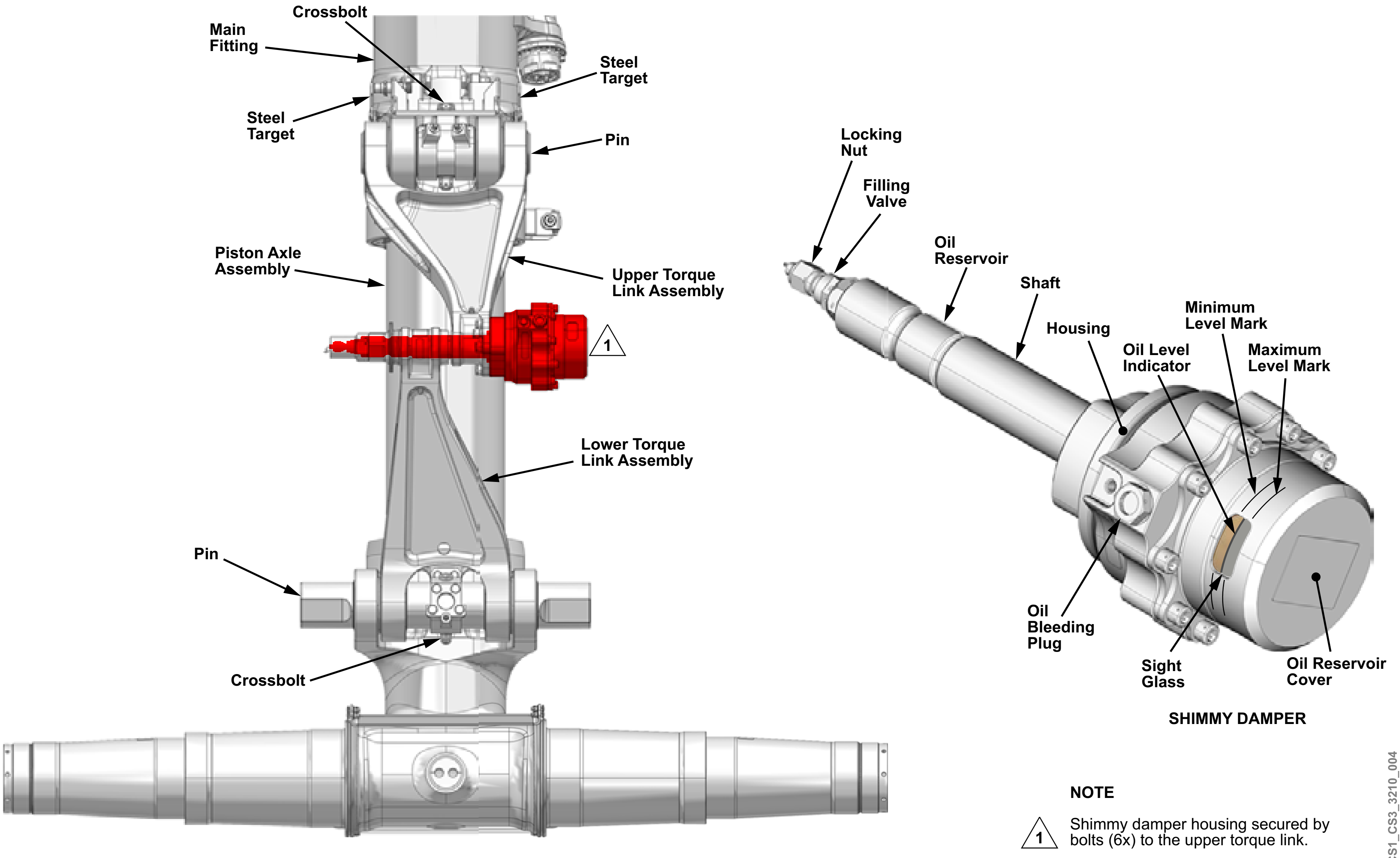


Figure 3: Torque Links and Shimmy Damper

SIDE BRACE

The MLG side brace provides the means of supporting and locking the MLG in the extended position. The upper side brace member is attached to the wing structure by a spindle. The lower side brace member is also attached to the MLG main fitting by a spindle. Each spindle serves as a rotatable crank bolt that accommodates the turning movement of the side brace.

The side brace is extended and locked by a locking stay when the MLG is extended. The side brace is folded on three axes and unlocks when the MLG is retracted. The upper side brace member has attachment points for the locking stay, unlock actuator, and downlock springs.

DOWNLOCK SPRINGS

There are two downlock springs on each MLG. The downlock springs are manufactured from titanium and are connected to the locking stay, and the upper side brace member.

A stainless steel spring retaining collar, provided with a bronze bushing in its bore, is located on each end of the downlock springs. A lubrication fitting is also provided at each collar.

When the MLG extends, the downlock springs pull the locking stay into the overcenter and locked position. The downlock springs keep the MLG in the down and locked position.

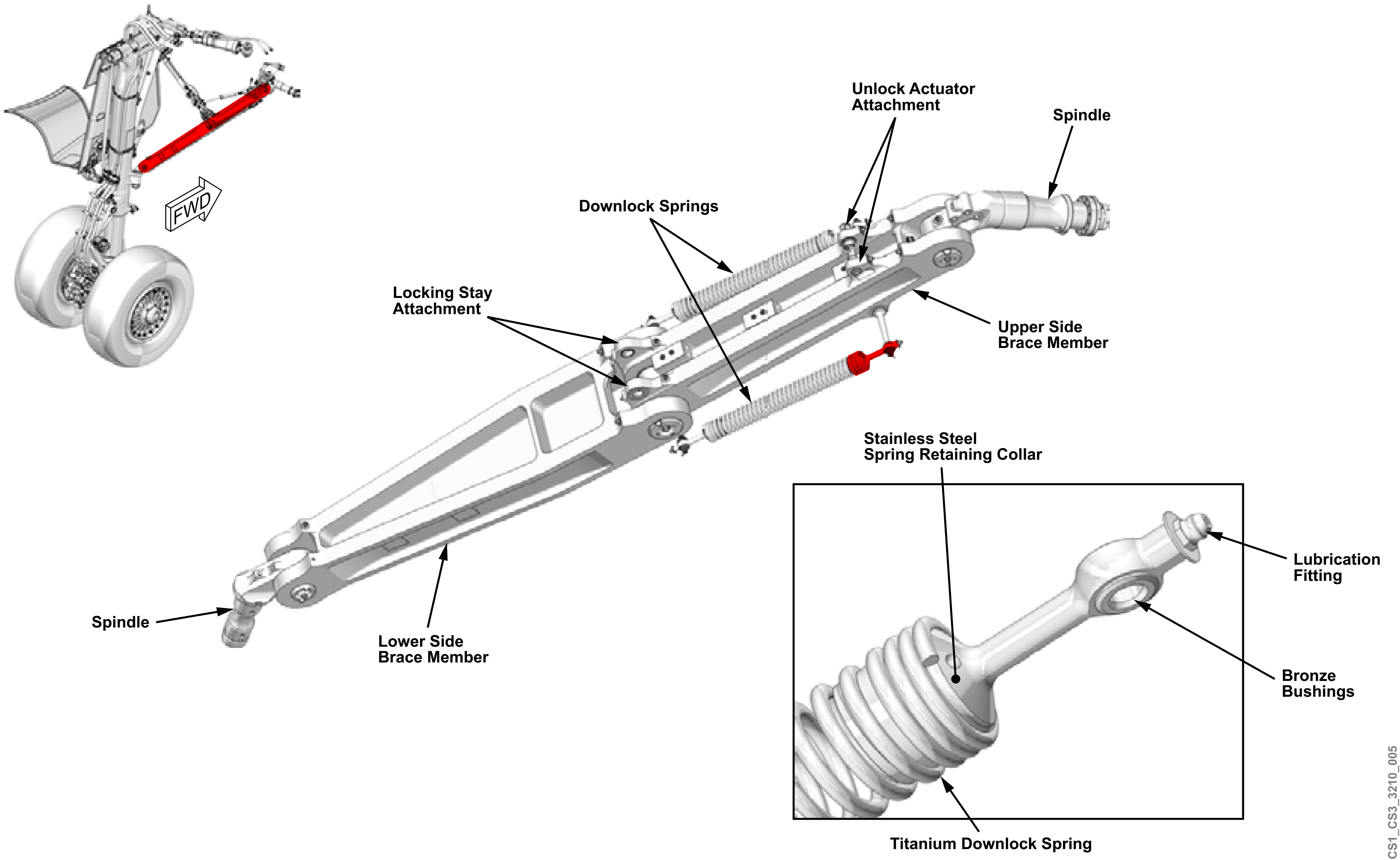


Figure 4: Side Brace and Downlock Springs

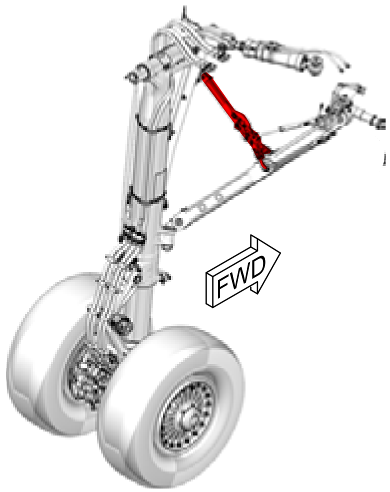
LOCKING STAY

The locking stay is a two-piece strut which overcenters at 5°, and locks the side brace in the extended position.

During retraction, the locking stay folds on all three axis points. The locking stay upper member is attached by a universal joint assembly to the MLG main fitting, and has the proximity sensor steel targets.

The 5° overcenter stop, which can be adjusted with shims, is located on the locking stay upper member. The locking stay lower member is hinged to the MLG upper side brace with a stainless steel attachment pin. The locking stay lower member has attachments for the two downlock spring attachment levers, two proximity sensors, adjustable overcenter stop bolt, and the unlock actuator.

The center joint of the locking stay has an insertion point for the gear locking pin to prevent unlocking of the locking stay.



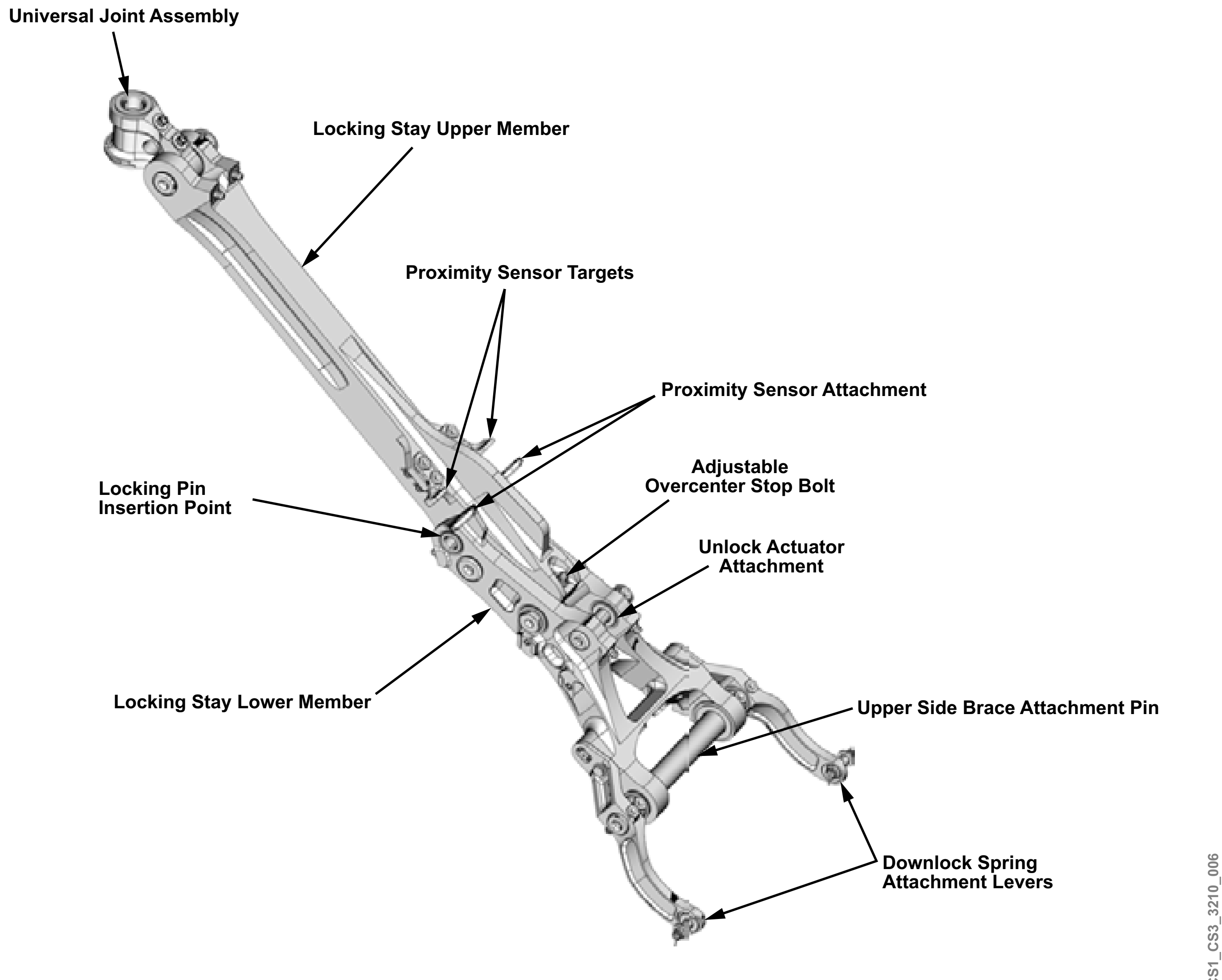


Figure 5: Locking Stay

GEAR DOORS

Each main landing gear (MLG) is provided with an inboard door, a mid door, and an outboard door. The doors are made from carbon fiber reinforced polymer (CFRP). The doors are connected to the MLG and follow its movement.

Following retraction, the doors completely close the MLG bay except at the tire circumference.

The mid door is a fixed door which is attached to the MLG main fitting at five points by adjustable rods.

The outboard door is attached to the wing structure by two hinge fittings. Two adjustable rods, attached to the MLG, move the outboard door. The inboard door is attached to the mid door by two hinge fittings. Two adjustable rods attached to the wing structure operate the inboard door.

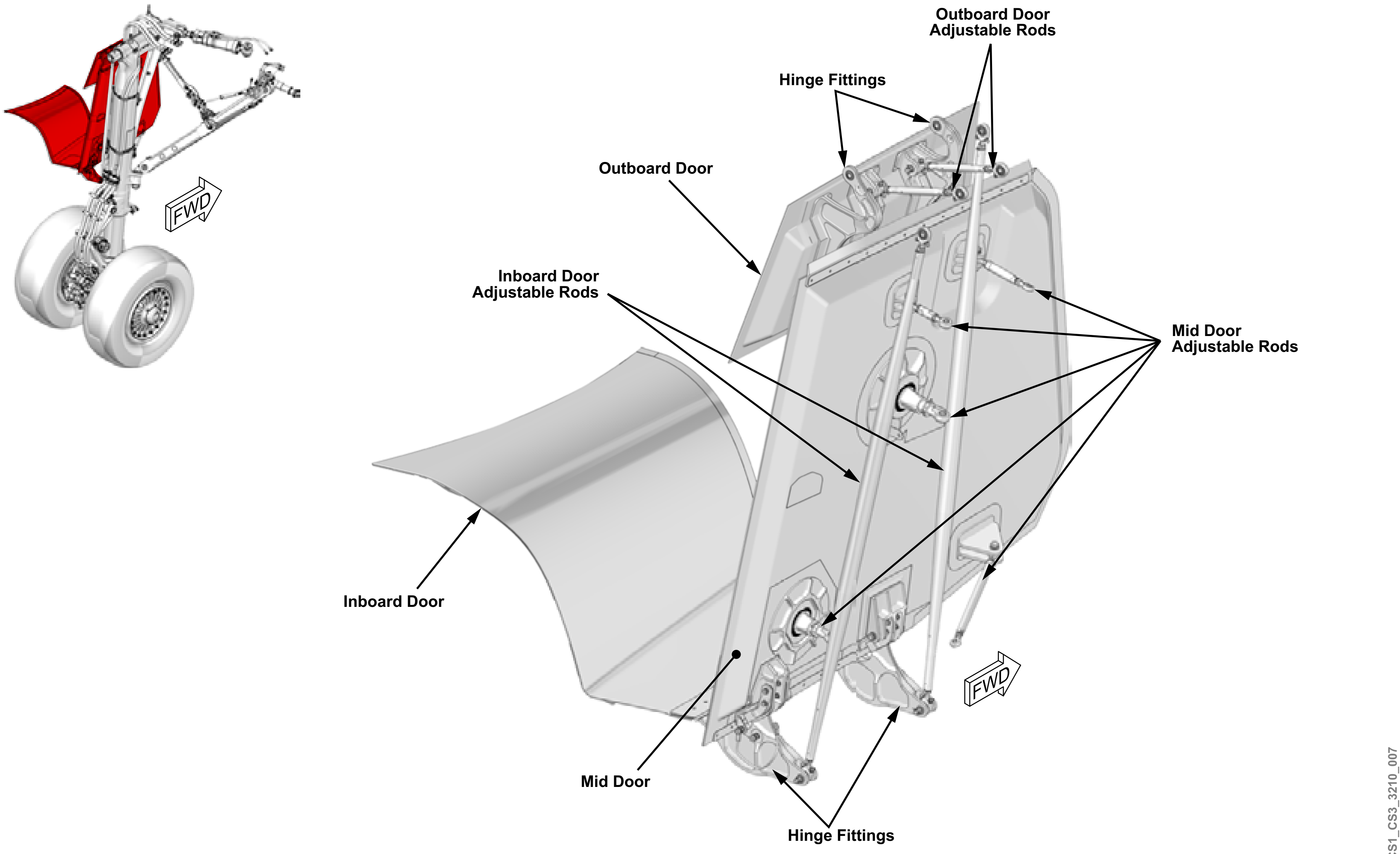


Figure 6: Gear Doors

PRACTICAL ASPECTS

MAIN LANDING GEAR SHOCK STRUT CHANGEOVER VALVE ACTIVATION

Shock strut oil leaks are temporarily fixed by manually activating the secondary seals through a changeover valve, in accordance with the AMP. Depressurization of the shock strut is not required when performing this task.

The changeover valve has no stripes. To activate the changeover valve the check nut is loosened, and the changeover activation screw is rotated until it is flush with the check nut. Tighten and lockwire the check nut after the task is completed. A shock strut oil leak check must be carried out after the secondary seals have been activated.

MAIN LANDING GEAR SHOCK STRUT SERVICING PLACARD

A placard for servicing the MLG shock strut with nitrogen is installed on the MLG main fitting. Warning messages posted on the placards and in the AMP must be observed.

Hydraulic oil servicing is performed in accordance with the AMP.

The servicing of the CS300 is different than on the CS100. Nitrogen-charging pressure is higher on the CS300 to support a heavier aircraft.

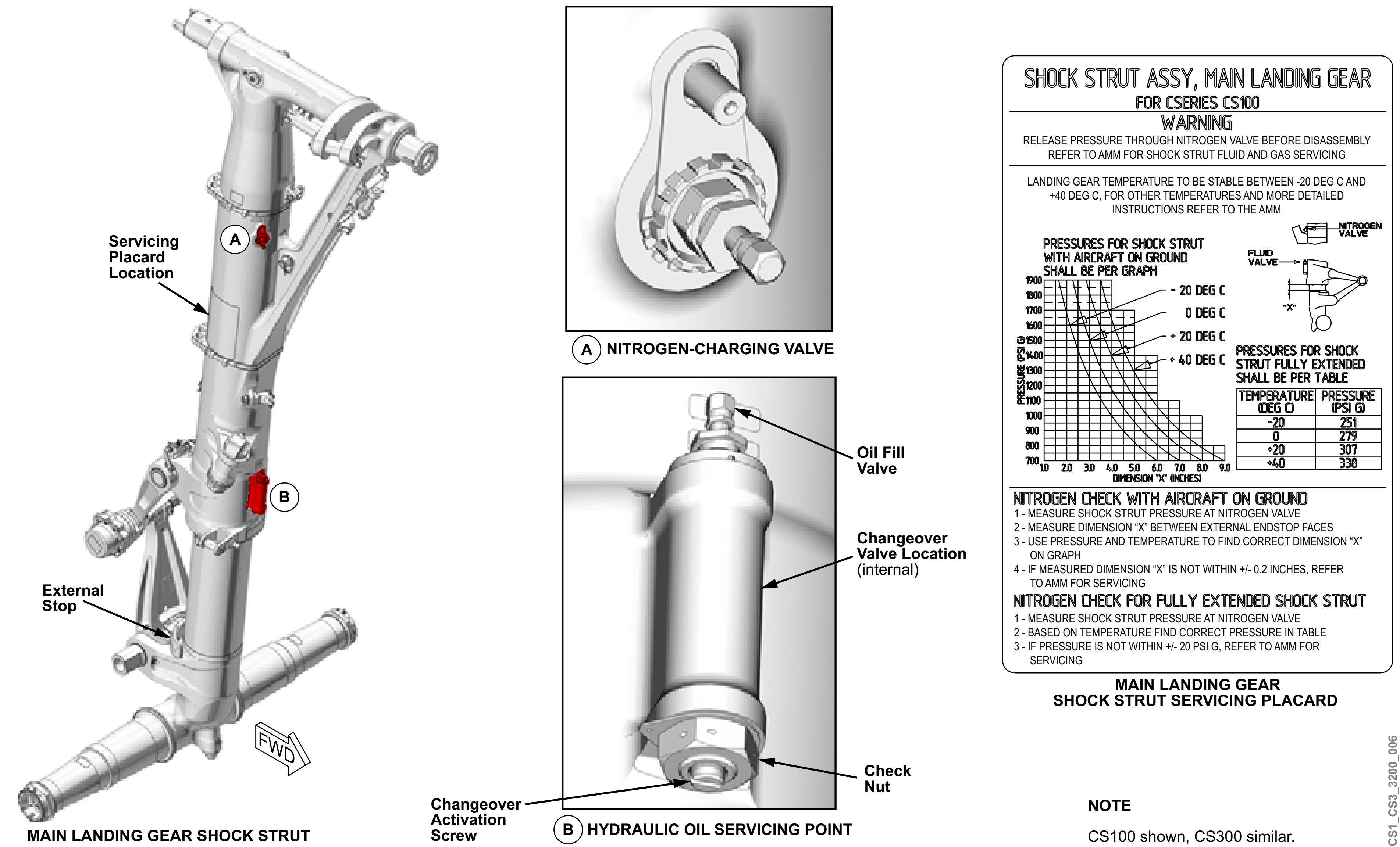


Figure 7: Main Landing Gear Shock Strut Servicing

SHIMMY DAMPER SERVICING

When oil in the shimmy damper is at the correct level, a black line is seen in the sight glass between the minimum and maximum lines.

A red indication is shown when the oil level drops below the minimum level. This indicates servicing is required.

Servicing is accomplished through the oil fill valve, using an oil servicing pump. During servicing, the oil bleeding plug on the shimmy damper allows the release of trapped air.

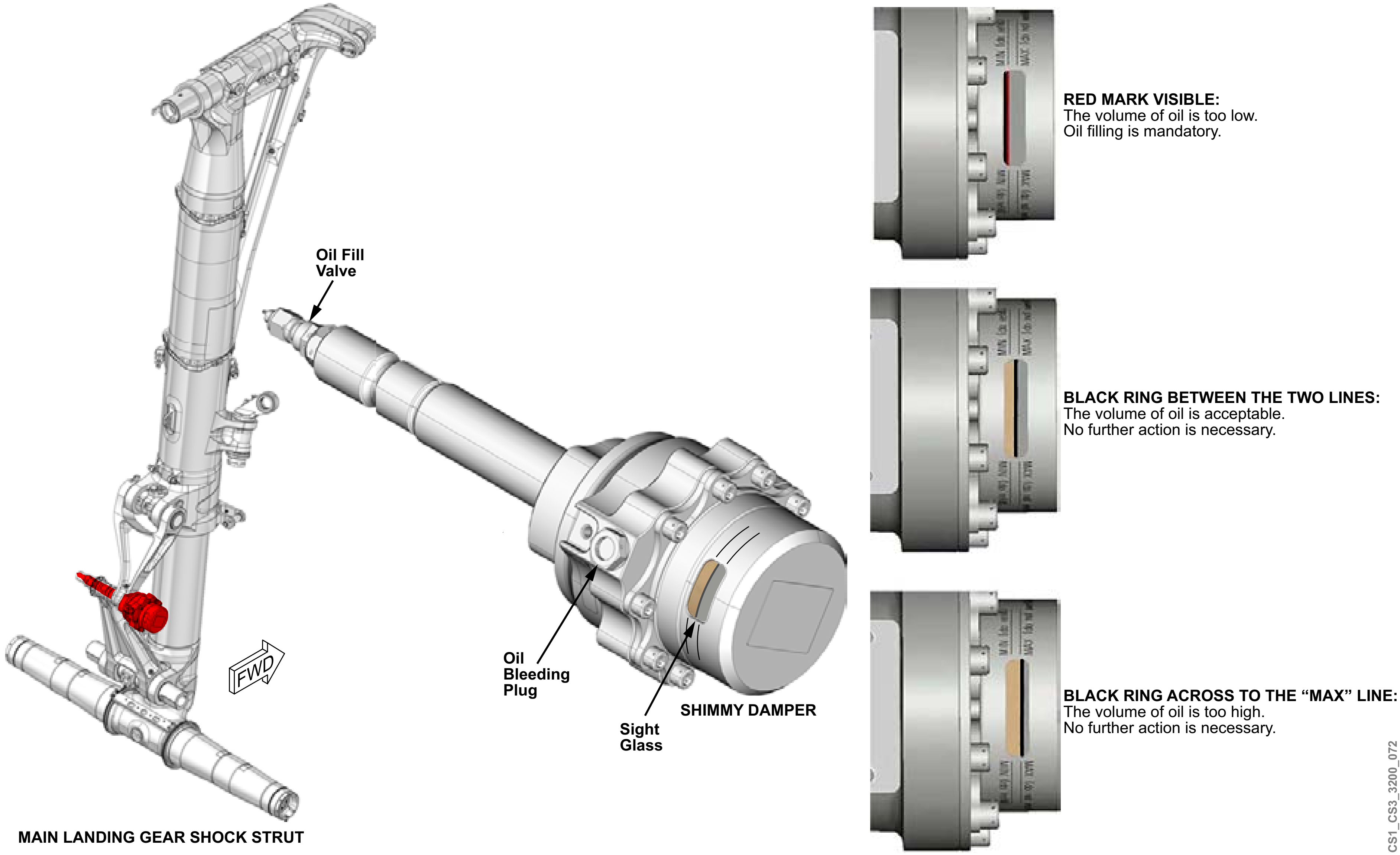


Figure 8: Shimmy Damper Servicing

32-20 NOSE LANDING GEAR AND DOORS

GENERAL DESCRIPTION

The principal components of the nose landing gear (NLG) consist of a single-stage, oleopneumatic-type shock strut with a folding drag brace, and associated downlock mechanism.

The shock strut consists of the Y-shaped main fitting, and an inner sliding piston axle assembly.

The piston axle assembly has an axle at its lower end to support the wheel assemblies. It also has a tow bar fitting on the front of the axle assembly. Torque links are installed between the main fitting and the piston axle assembly. An uplock roller is located on the front of the main fitting.

The main fitting is attached to the fuselage structure by left and right pintle pins.

The drag brace provides the means of supporting and locking the NLG in the extended position. The drag brace is attached to the main fitting at its lower end. It is attached using pintle pins to the fuselage fittings at its upper end. The drag brace is extended and locked when the gear is in extended position. The drag brace is unlocked and folded when the gear is in retracted position.

A locking stay assembly is attached to the drag brace at its forward end and to the shock strut main fitting at its aft end. It maintains the drag brace overcenter lock when extended. The locking stay assembly has an insertion point for a locking pin.

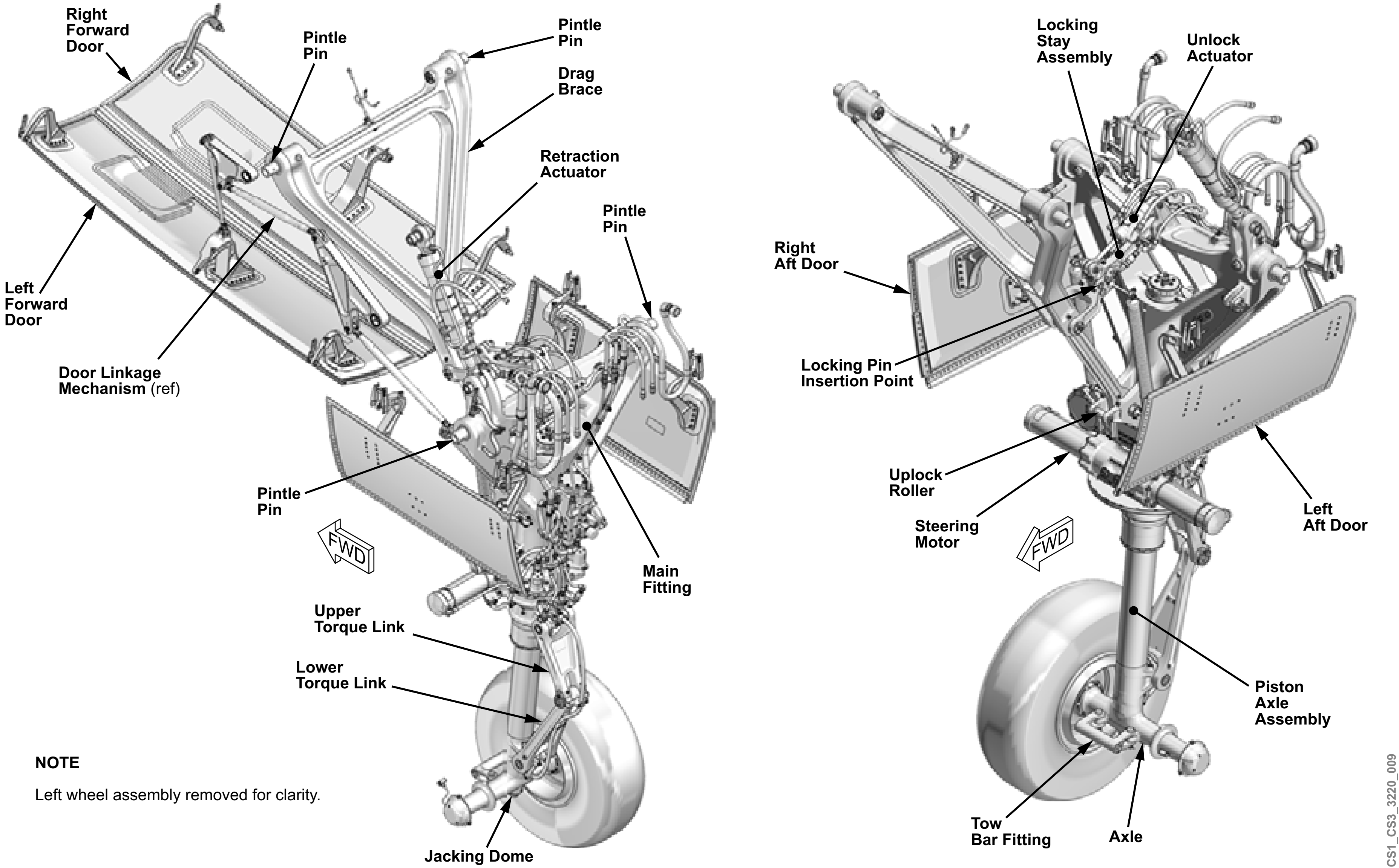
A hydraulic unlock actuator is attached between the locking stay and the main fitting. It unlocks the locking stay on retraction.

A hydraulic retraction actuator is attached between the main fitting upper pivot point, and the fuselage mounting point. During retraction the NLG moves forward into the wheel well.

The NLG bay is enclosed by two forward and two aft mechanically-linked doors when the gear is retracted. When extended, the two forward doors close and the two aft doors remain open.

Various steering components are mounted on the shock strut main fitting, including a steering motor uses to turn the nosewheels through the torque links.

The piston axle assembly has a jacking dome at its base to facilitate wheel changes.



NOTE

Left wheel assembly removed for clarity.

Figure 9: Nose Landing Gear

DETAILED COMPONENT INFORMATION

SHOCK STRUT

The NLG shock strut includes a single-stage oleopneumatic shock strut. The upper cylinder within the main fitting forms the upper part of the shock strut. The piston axle assembly serves as the sliding tube of the shock strut.

The shock strut is filled with nitrogen and oil. The damping performance is optimized by means of a metering pin that restricts the oil passage as it passes through an orifice in the shock strut.

Centering cams on the sliding tube and the main fitting hold the piston axle assembly centered whenever the NLG wheels are not on the ground. Internal bottoming stops are located at the top section of the main fitting. An external replaceable stop is located on the piston axle.

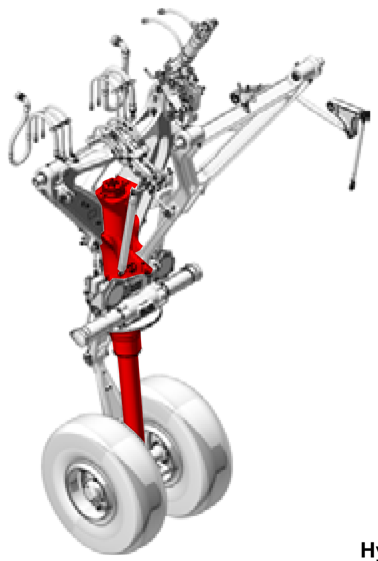
Primary and secondary seals are located between the main fitting lower bearing assembly and the sliding tube. The seals prevent oil leakage from the shock strut oil chamber. Only one set of seals is activated at a time.

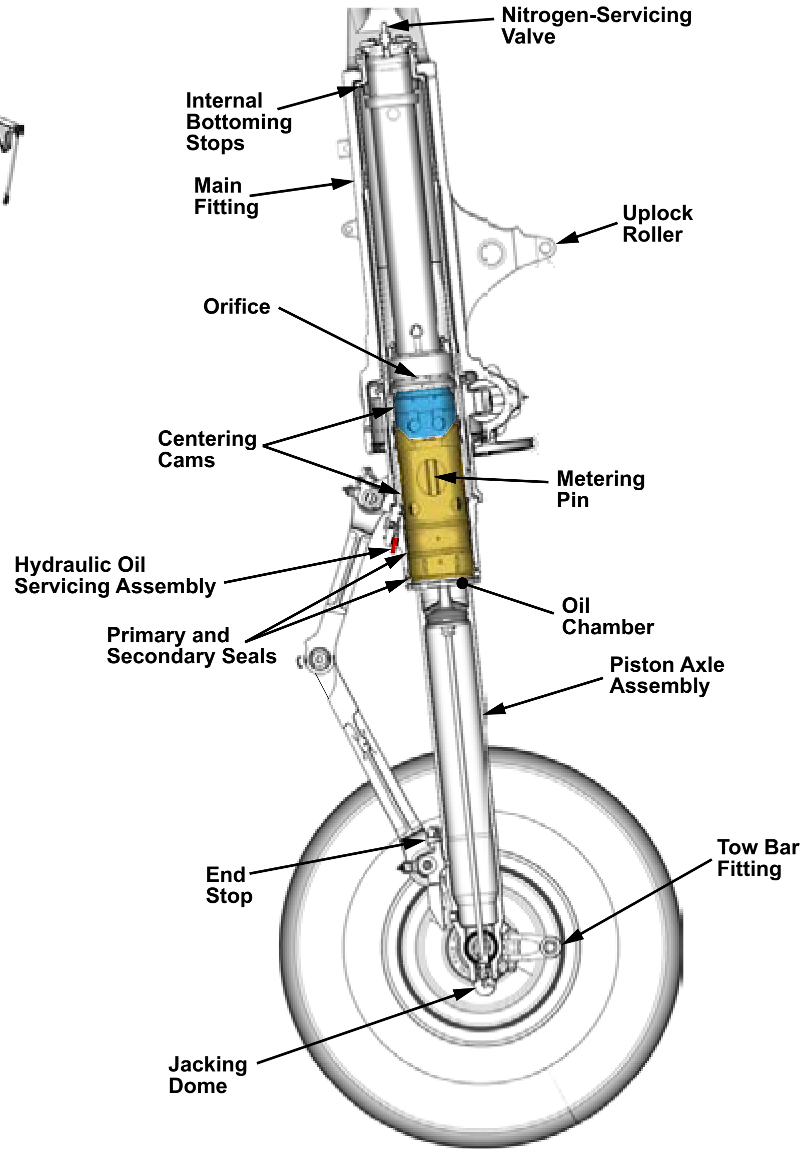
High-pressure oil passes through a changeover valve and activates the primary seal. In the event of primary seal leakage, the shock strut changeover valve can be activated manually by using a suitable wrench and turning the valve clockwise. This action isolates the primary seals and energizes the secondary seals. The red and green indication on the valve is visible when the primary seals are active. Only the red indication on the valve is visible when the secondary seals are activated. Activation of the changeover valve to the secondary seals is considered a temporary shock strut leak rectification.

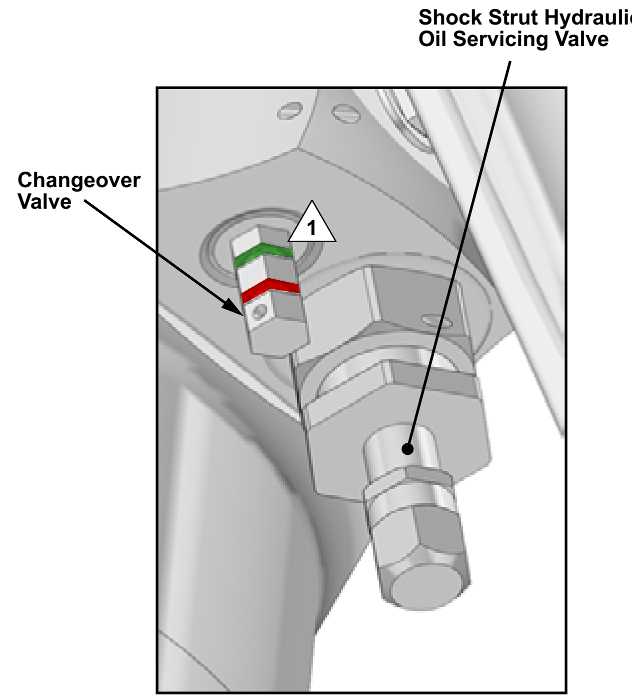
The piston axle assembly bearing surface and the wheel axle bearing surfaces have a special coating for heavy wear and high-impact. Damage or wear on any of these surfaces must be evaluated in accordance with the AMP.

A nitrogen-servicing valve for the shock strut is located at the top of the main fitting. A shock strut hydraulic oil servicing valve is located on the lower section of the NLG main fitting.

A tow bar fitting is located on the forward lower section of the piston axle assembly. An uplock roller is located on the front side of the main fitting. A jacking dome, to facilitate nosewheel changes, is provided at the bottom of the piston axle assembly.







HYDRAULIC OIL SERVICING POINT

NOTE

1 Green: Secondary seal not activated.
Red: Secondary seal activated.

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Figure 10: Shock Strut

TORQUE LINKS

Torque links are attached between the lower part of the piston axle assembly and the steering cuff, and are interconnected by a torque link bolt.

The upper torque link is attached to the steering cuff by the upper torque link pin and crossbolts.

The lower torque link is attached to the NLG piston axle by the lower torque link pin and crossbolts.

The torque links transfer the steering torque from the steering cuff to the nosewheels. They also prevent the piston axle from rotating freely within the NLG main fitting.

The upper torque link is provided with attachment points for aircraft weight-off-wheels (WOFFW) proximity sensors steel targets.

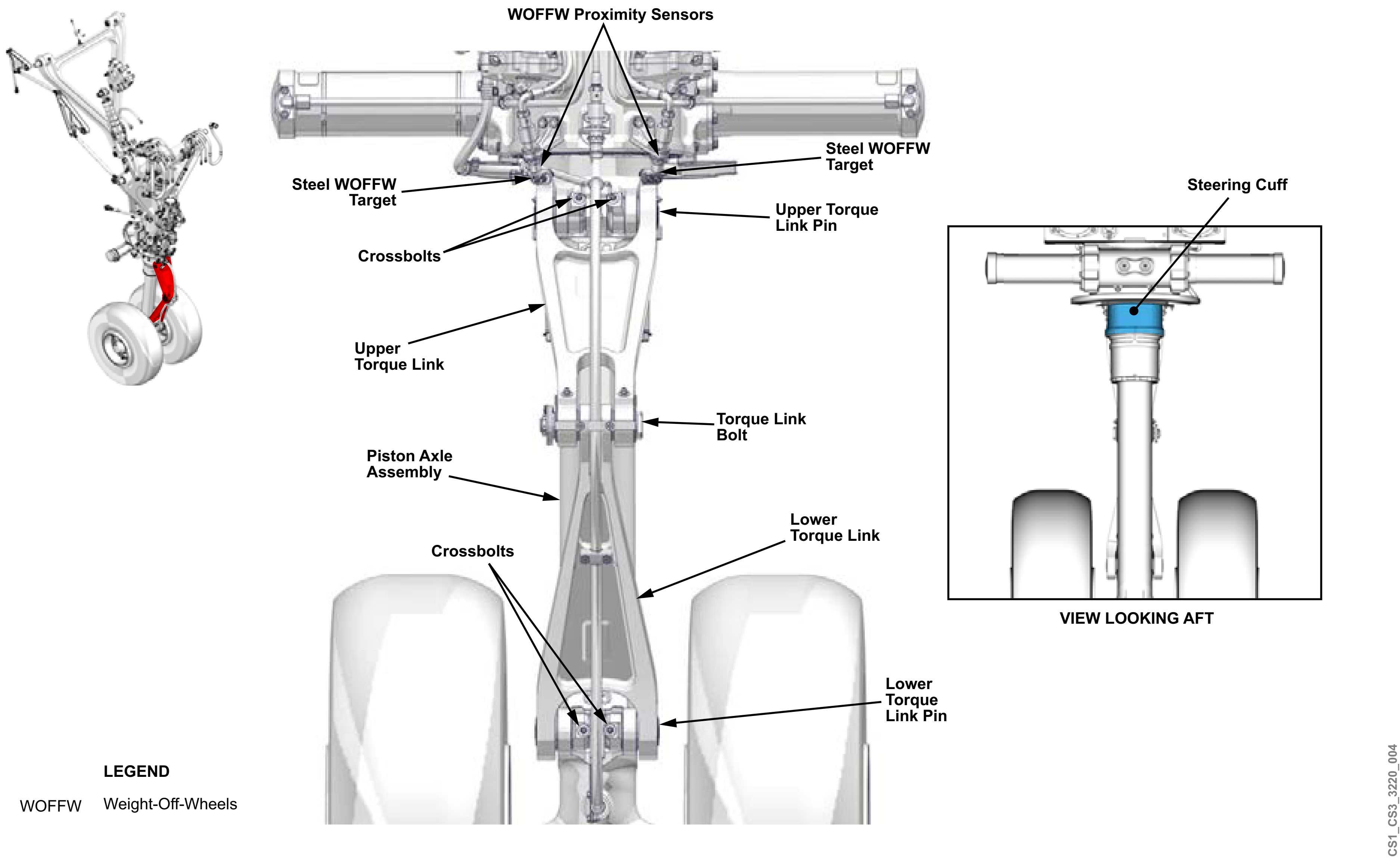


Figure 11: Torque Links

DRAG BRACE

The NLG drag brace consists of an upper and lower member.

The drag brace upper member is attached to the NLG bay structure, and the drag brace lower member is attached to the NLG main fitting. The drag brace upper member is connected to the aircraft structure with pintle pins. Each pintle pin is secured in place with a crossbolt.

When the NLG is in the extended position, the drag brace is extended, and locked by the locking stay. The drag brace is unlocked by the pulling force of the unlock actuator acting on the locking stay. The drag brace is folded when the NLG is in the retracted position.

DOWNLOCK SPRINGS

The two titanium downlock springs have retaining collars at each end. The springs are connected at the locking stay and the lower drag brace main fitting joint. The main purpose of the downlock springs is to overcenter the locking stay with spring force and hold it in position.

A stainless steel spring retaining collar, equipped with a bronze bushing in its bore, is located on each end of the downlock springs. A lubrication fitting is also provided at each collar.

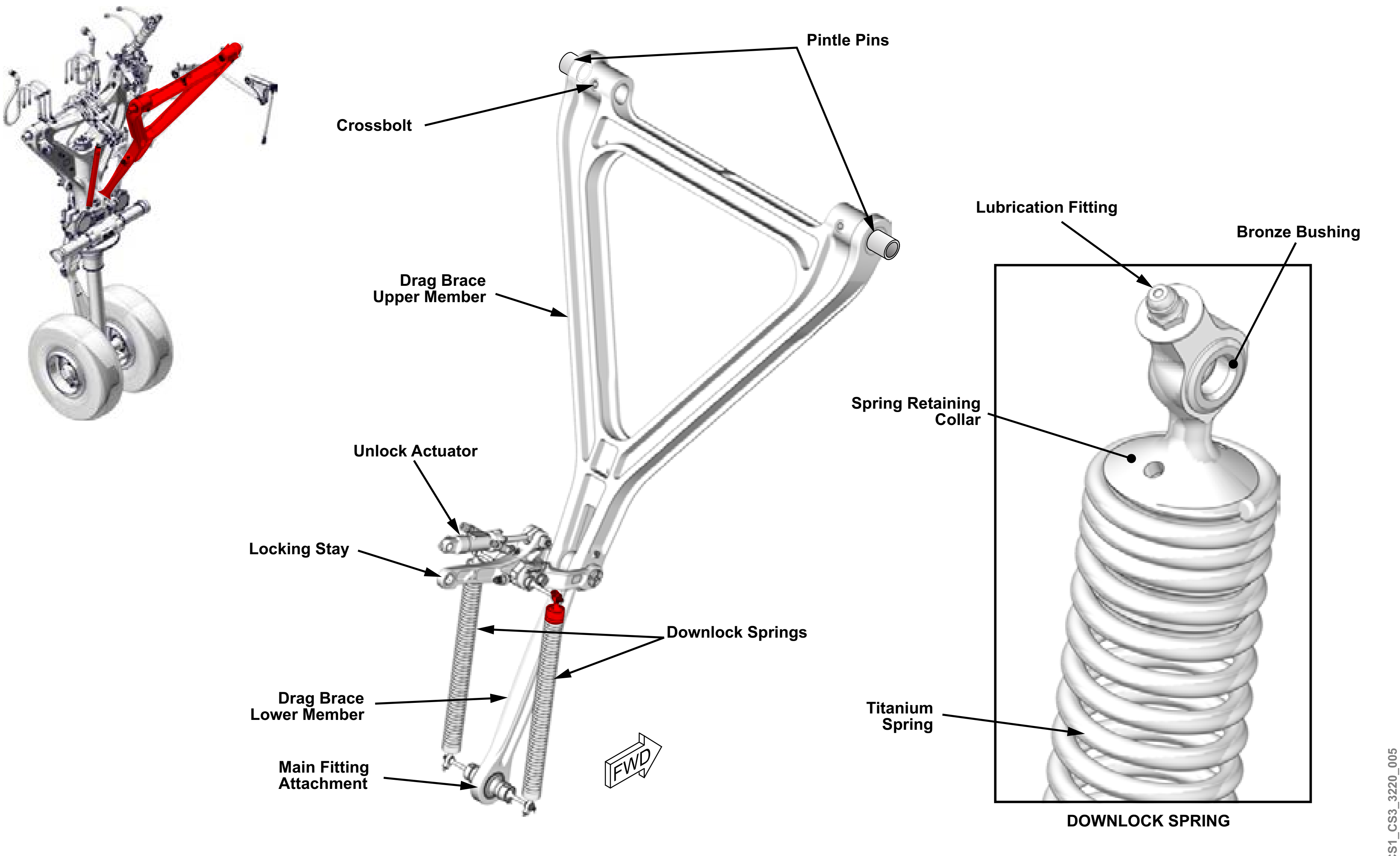


Figure 12: Drag Brace and Downlock Springs

LOCKING STAY

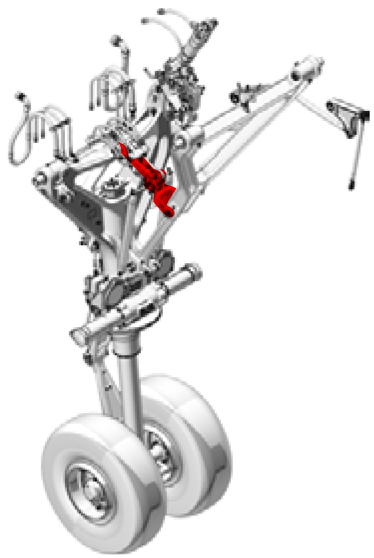
The locking stay is a two-piece hinged strut that locks the drag brace in the extended position and folds it during retraction.

Upper and lower locking stay members are designed to have a 5° overcentered stop position. An overcenter adjustment bolt is used to rig the overcenter position. The locking stay members are interconnected by a hinge pin. The upper member is attached to the NLG main fitting, and the lower member is attached to the drag brace attachment.

Attachment points are provided on the locking stay members for the proximity sensors, targets, downlock springs, and the unlock actuator.

Downlock springs pull the locking stay into the overcentered locked position when the NLG is extended.

The locking stay has a provision to insert the NLG locking pin.



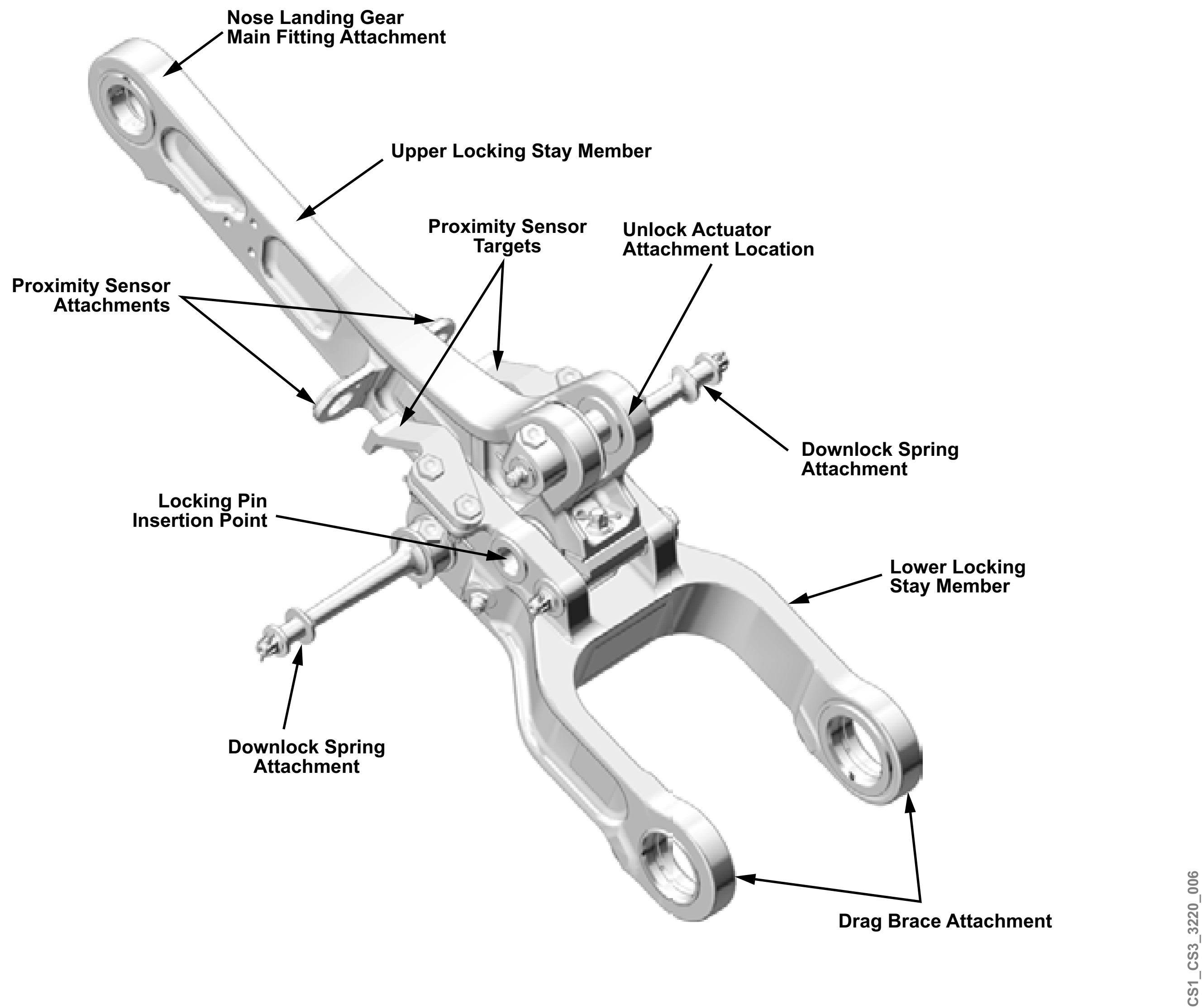


Figure 13: Locking Stay

CROSSBEAM

The crossbeam ensures that the nosewheels stay centered in the NLG box when the NLG is in the UP (retracted) position. The crossbeam is mounted at both ends to the upper side structure of the NLG box.

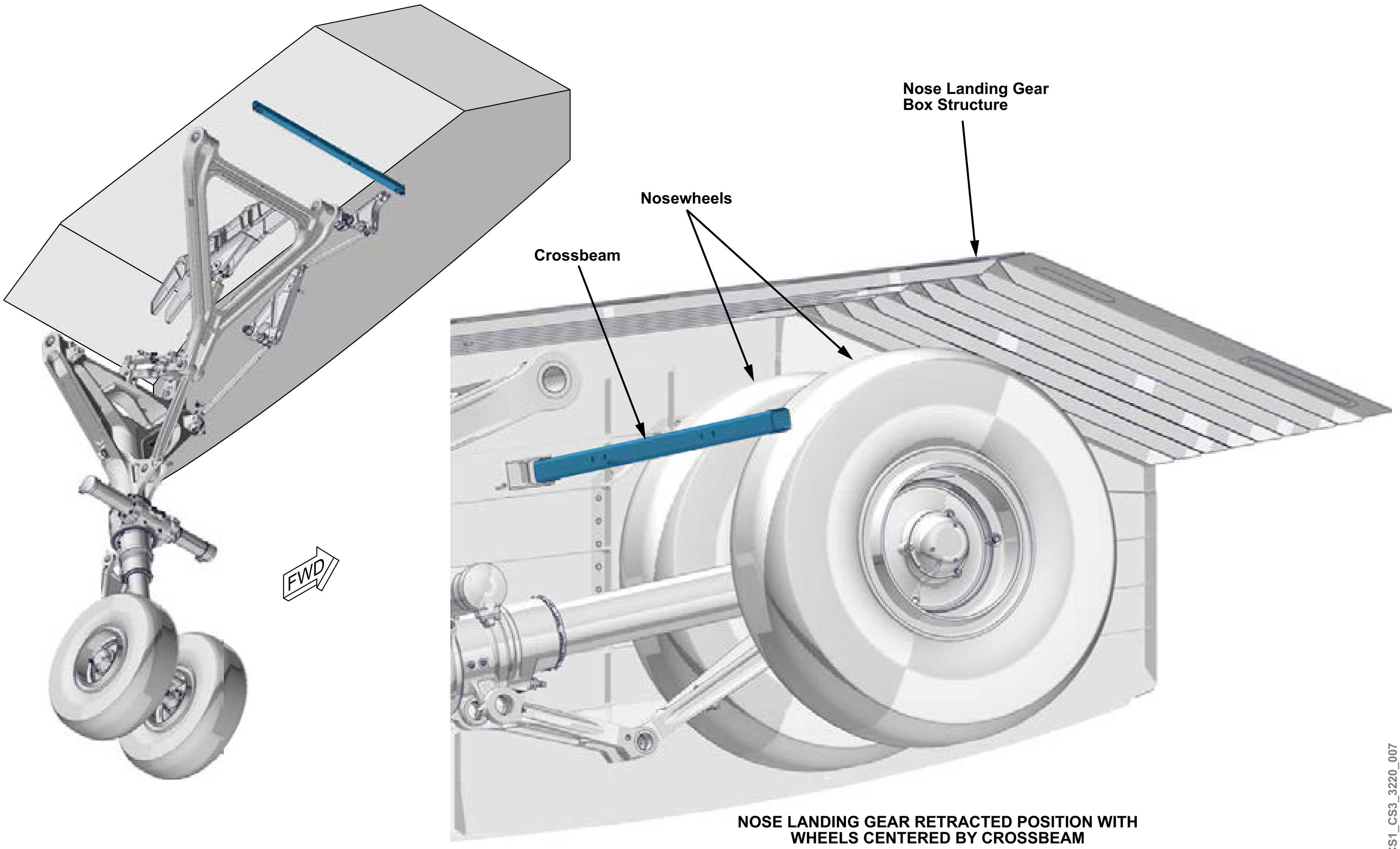


Figure 14: Crossbeam

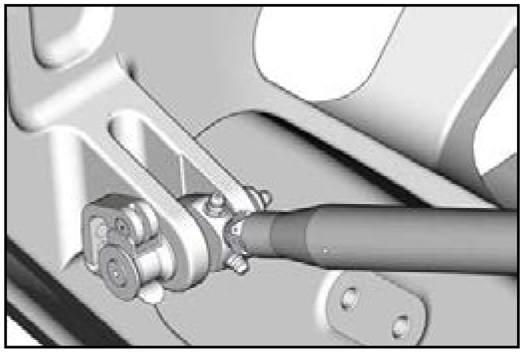
GEAR DOORS

The four NLG doors are hinged to the nose landing gear box structure, and are made from carbon fiber reinforced polymer (CFRP).

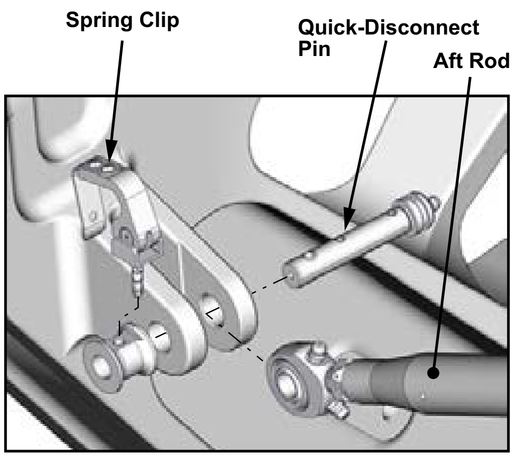
Each aft door is connected by an aft door rod to the NLG. The aft doors follow the NLG movement and stay open after NLG extension. The aft doors remain closed after NLG retraction.

Each forward door is connected to the NLG by a forward rod, forward bellcrank, mid rod, mid bellcrank, and an aft rod. The forward doors are driven open by the linkage during NLG extension and retraction. After the NLG has been retracted or extended, the forward doors are driven closed.

For maintenance purposes, each forward door can be opened by removing a quick-disconnect pin at the forward door disconnect mechanism. This mechanism is located at the aft rod attachment. A spring clip locks the pin in position. Releasing the spring clip allows removal of the pin. The aft rod must be reconnected to the door with the pin, and locked with the spring clip prior to aircraft dispatch.



FORWARD DOOR
DISCONNECT MECHANISM INSTALLED



FORWARD DOOR
DISCONNECT MECHANISM REMOVED

NOTE

Right side shown, left side similar.

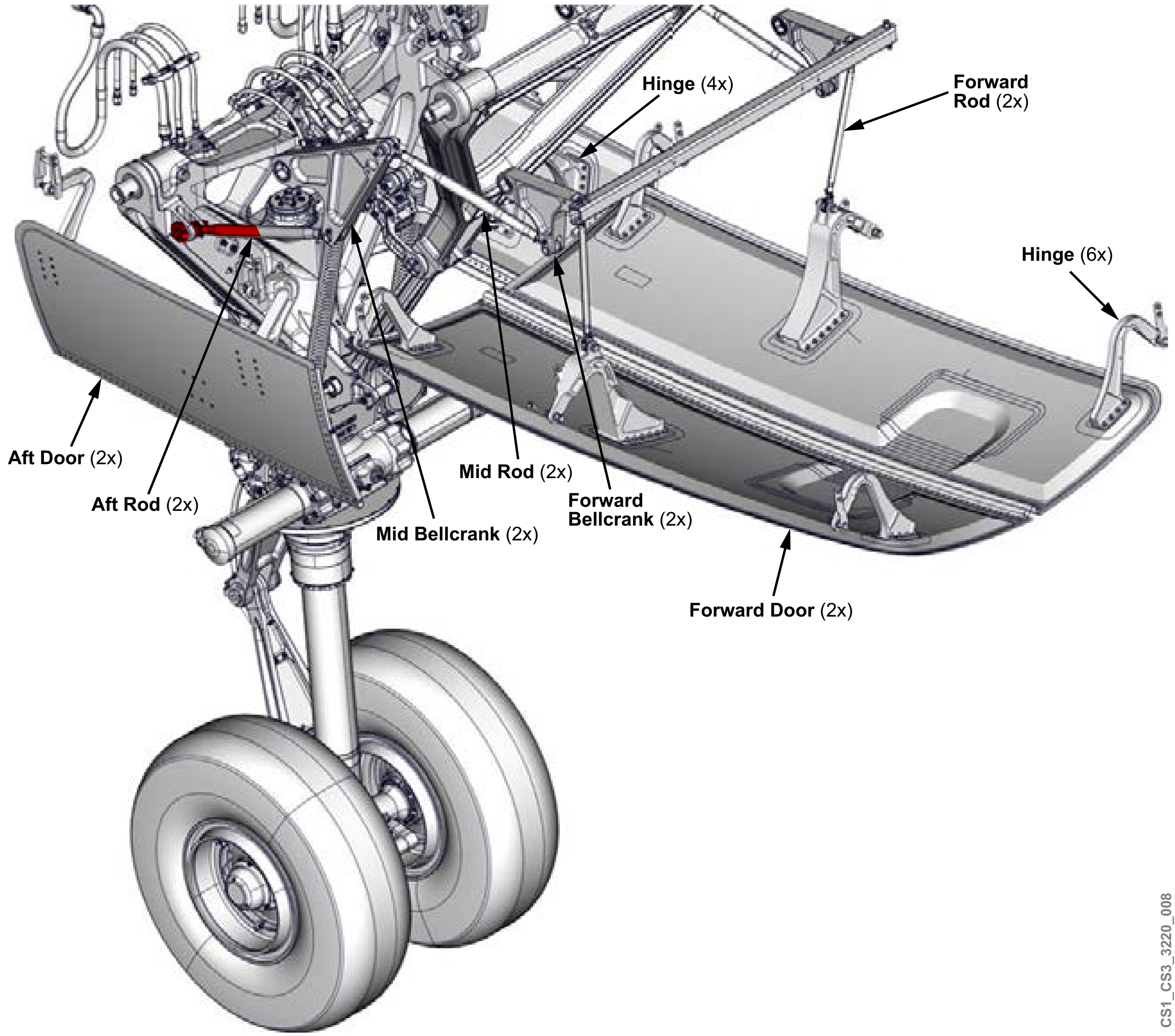


Figure 15: Gear Doors

PRACTICAL ASPECTS

NOSE LANDING GEAR SHOCK STRUT CHANGEOVER VALVE ACTIVATION

Internal shock strut oil leaks are temporarily fixed by manually activating the secondary seals through a changeover valve, in accordance with the AMP. Depressurization of the shock strut is not required when performing this task. A shock strut oil leak check must be carried out after the secondary seals have been activated.

Visible red and green stripes on the seal changeover valve indicate that the primary seals are active. The secondary seals are activated by completely tightening the changeover valve with a suitable tool, until only the red stripe is visible.

NOTE

If a leak is noticed on the NLG, even if very small, it is recommended to switch to the secondary seal immediately using the changeover valve to minimize the possibility of complete deflation. There are no limits to the number of activation and deactivation of the NLG seals.

NOSE LANDING GEAR SHOCK STRUT SERVICING

A servicing placard for servicing the NLG with nitrogen is attached to the steering motor right cylinder. Warning messages posted on the placards and AMP must be observed.

Hydraulic oil servicing is performed through a hydraulic oil servicing valve. Nitrogen charging is performed through a nitrogen-charging valve.

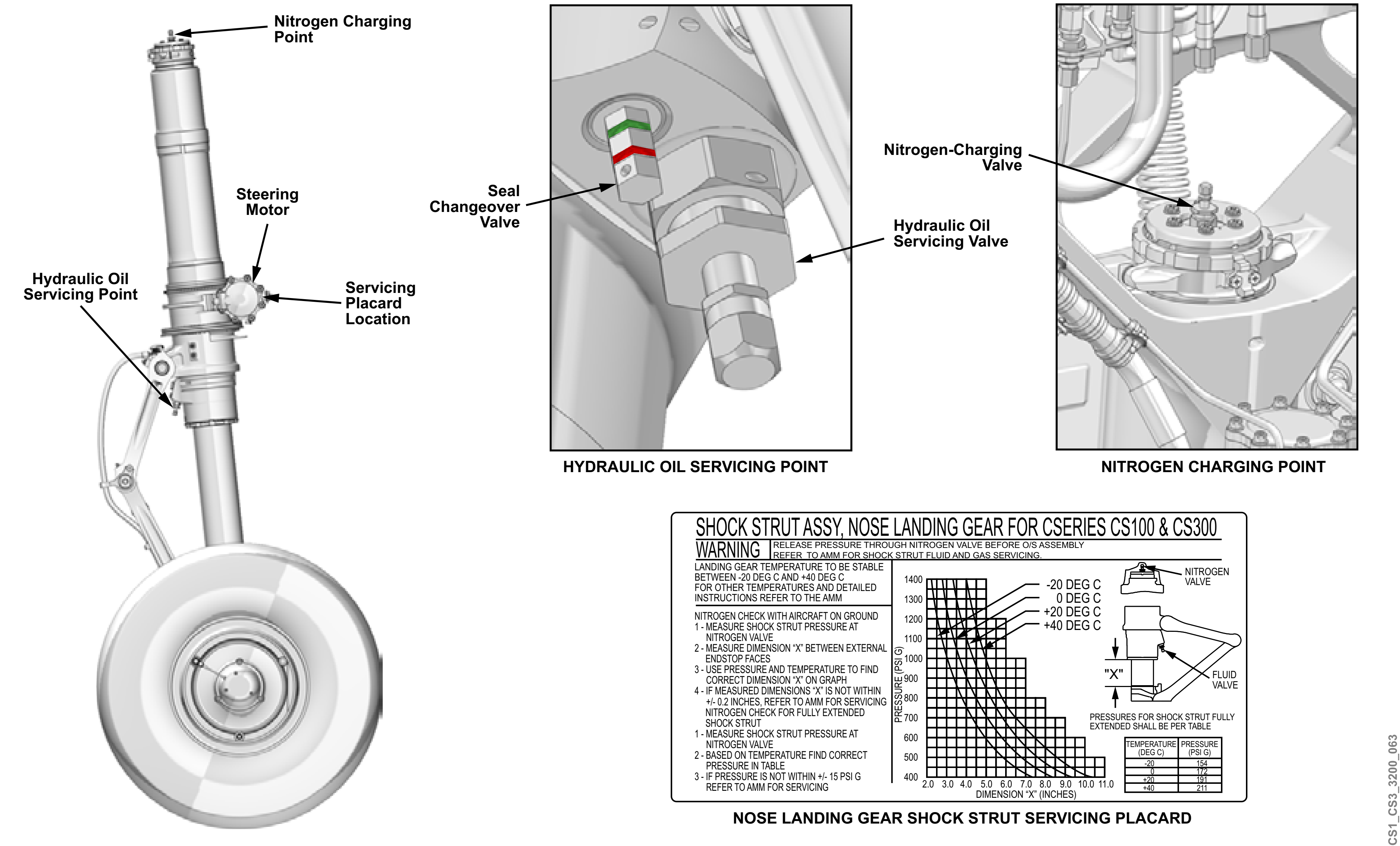


Figure 16: Nose Landing Gear Changeover Valve Activation and Shock Strut Servicing

32-30 EXTENSION AND RETRACTION

GENERAL DESCRIPTION

The landing gear system has a retraction function, a normal extension function, and an alternate extension system.

Two redundant landing gear and steering control units (LGSCUs) monitor signals from the landing gear control lever, uplock proximity sensors, downlock proximity sensors, and weight-off-wheels (WOFFW) proximity sensors. The LGSCUs are loadable components.

Retraction and normal extension of the landing gear is commanded by the landing gear control lever (LGCL) and controlled by the LGSCUs.

Depending on the LGCL UP or DN selection, the landing gear control valve (LGCV) hydraulically pressurizes a retraction actuator, an unlock actuator, and an uplock on each landing gear to retract or extend the gear. Hydraulic pressure comes from hydraulic system no. 1.

Alternate extension of the landing gear is controlled by the ALTN GEAR switch. This switch relays a signal to the LGSCU to release hydraulic pressure in the LGCV, and electrically release the uplocks. This action enables the landing gears to free fall to the extended position.

Landing gear status and position is displayed on the engine indication and crew alerting system (EICAS).

Each LGSCU is powered by two 28 VDC electrical power sources as follows:

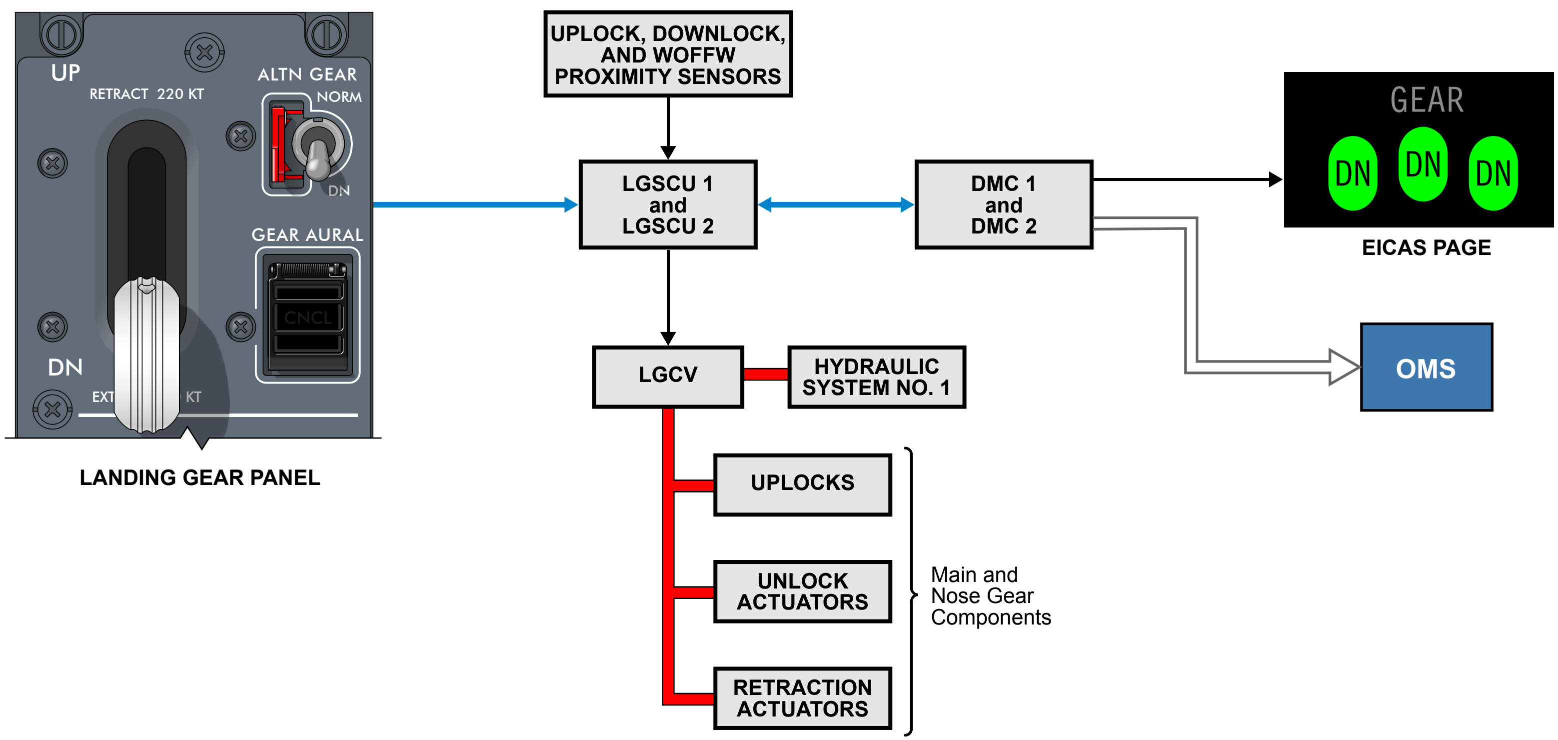
- LGSCU 1 is powered by DC BUS 2 and DC ESS BUS 1
- LGSCU 2 is powered by DC BUS 1 and DC ESS BUS 2

The alternate extension system is powered from DC ESS BUS 3.

RELAY DORMANCY CHECK

The LGSCUs automatically perform actuation of each relay within the landing gear system on the ground. The LGSCUs detect the switching of the relay output by monitoring the output lines from the relays.

This is done every 50 hours of aircraft powered on.



LEGEND

- Pressure
- LGCV Landing Gear Control Valve
- LGSCU Landing Gear and Steering Control Unit
- ARINC 429
- Discrete

NOTE

Power Sources:
- LGSCU 1 - DC BUS 2 and DC ESS BUS 1
- LGSCU 2 - DC BUS 1 and DC ESS BUS 2
- Alternate Extension System - DC ESS BUS 3

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Figure 17: Extension and Retraction System

COMPONENT LOCATION

UPLOCKS

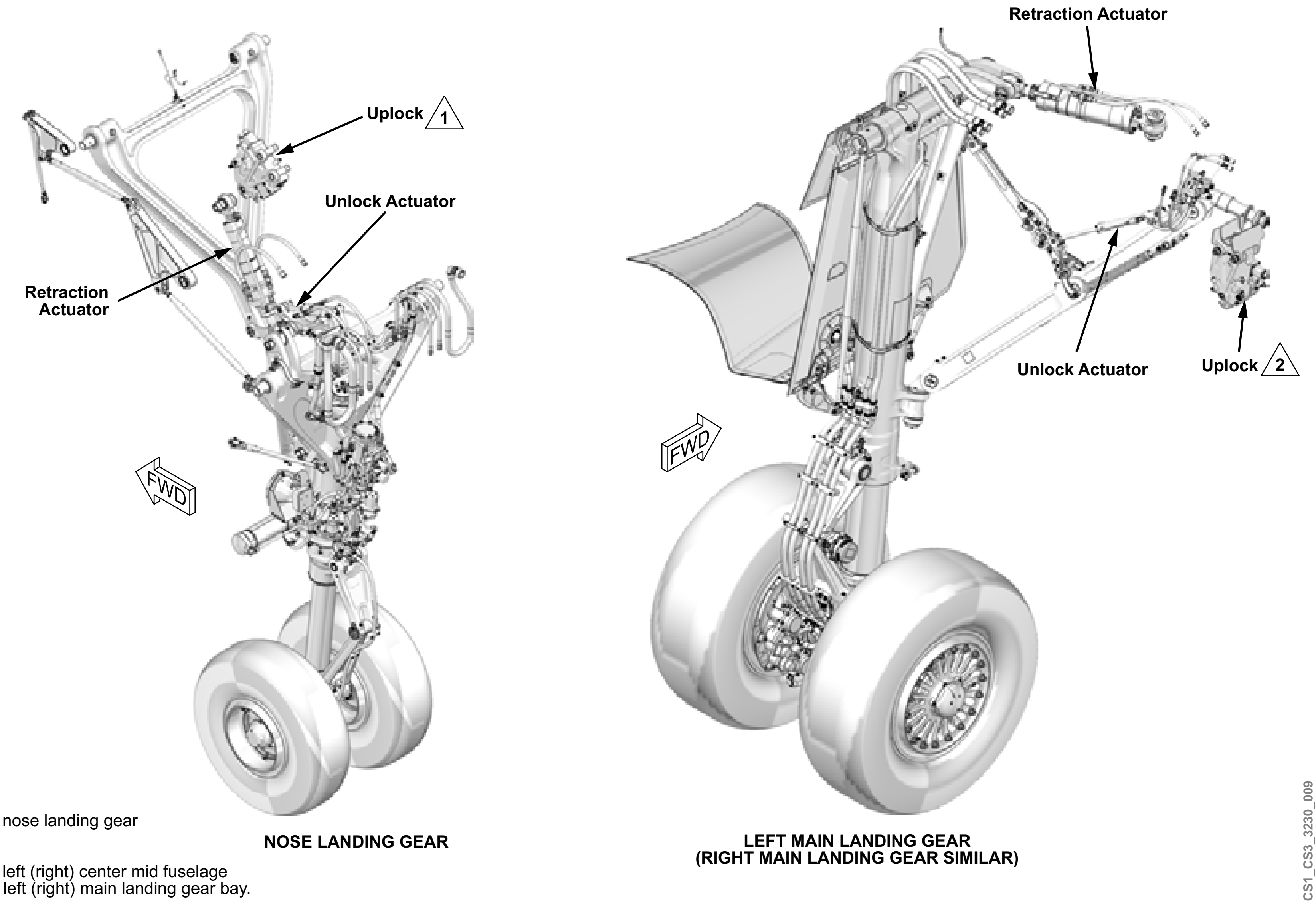
An uplock for each landing gear is located on the upper structure of its respective landing gear bay.

RETRACTION ACTUATORS

A retraction actuator for each landing gear is located in its respective landing gear bay.

UNLOCK ACTUATORS

An unlock actuator for each landing gear is located on its respective landing gear.



NOTE

1. Located on the nose landing gear box structure.
2. Located on the left (right) center mid fuselage structure of the left (right) main landing gear bay.

Figure 18: Extension and Retraction Component Location

LANDING GEAR AND STEERING CONTROL UNIT

Landing gear and steering control unit 1 (LGSCU 1) is located in the forward equipment bay. LGSCU 2 is located in the mid equipment bay.

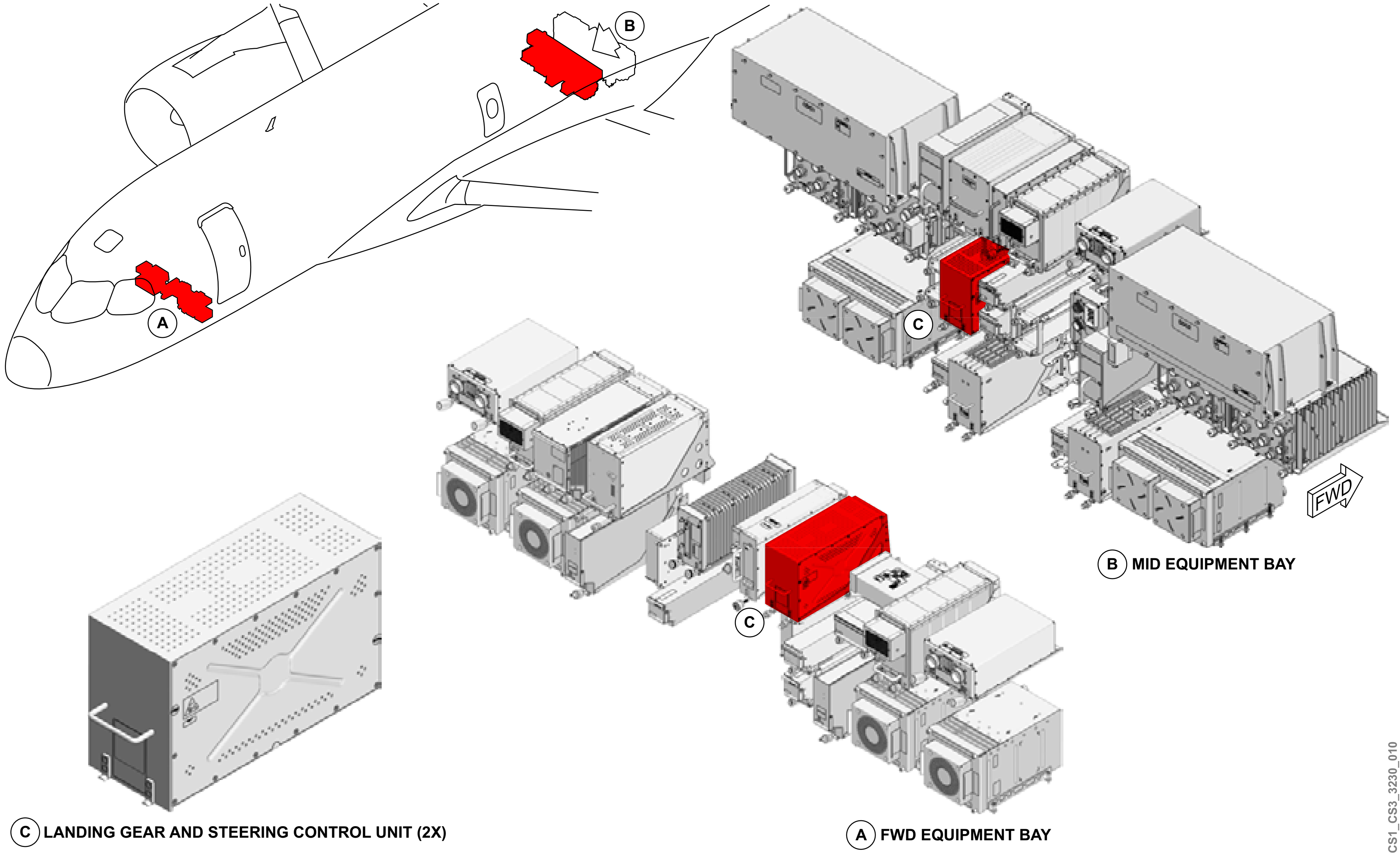


Figure 19: Landing Gear and Steering Control Unit

LANDING GEAR CONTROL VALVE

The LGCV is located at the lower center fuselage area.

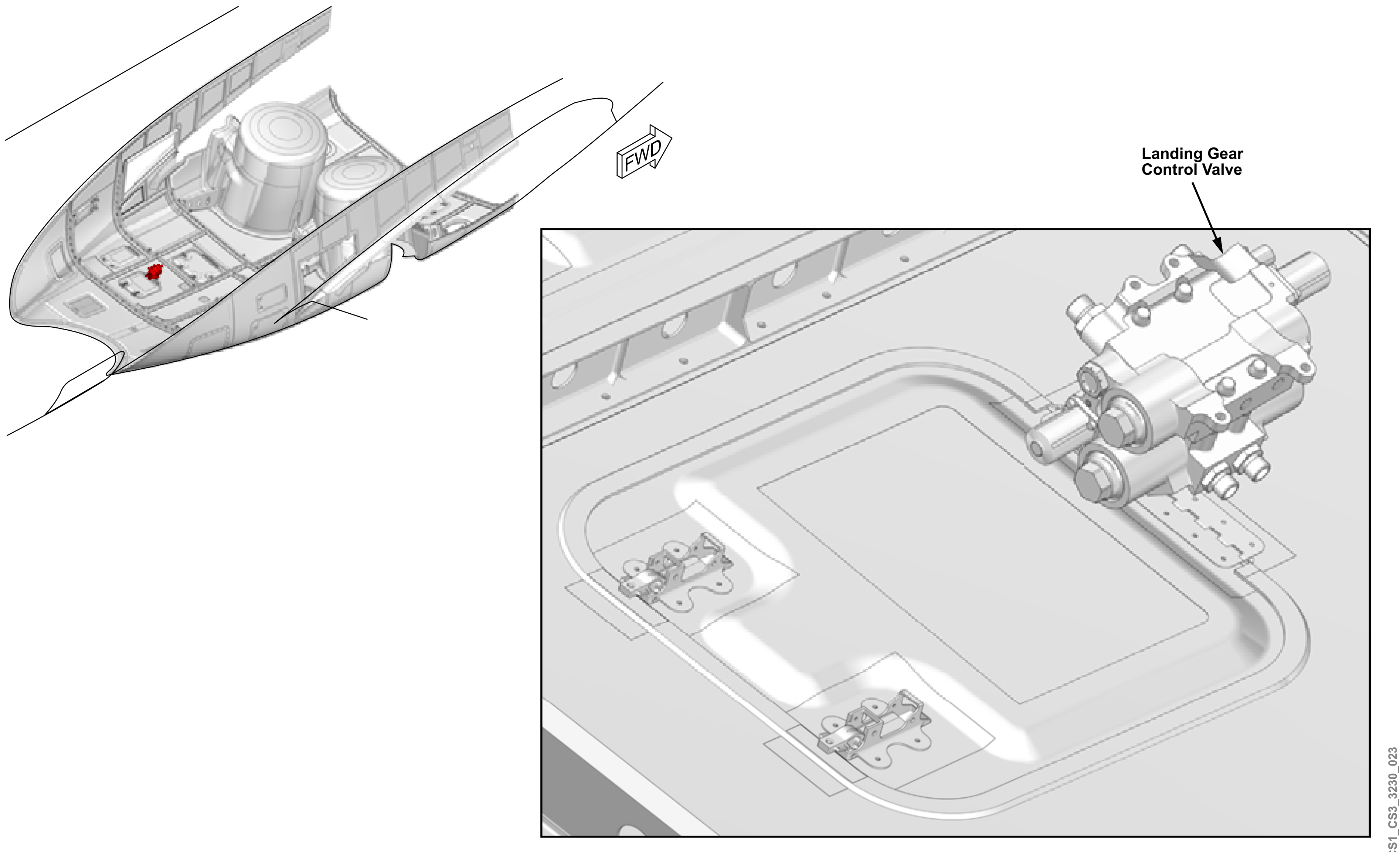


Figure 20: Landing Gear Control Valve Location

DETAILED COMPONENT INFORMATION

LANDING GEAR CONTROL VALVE

The landing gear control valve (LGCV) consists of a landing gear selector valve (LGSV), and a free fall selector valve (FFSV) attached together.

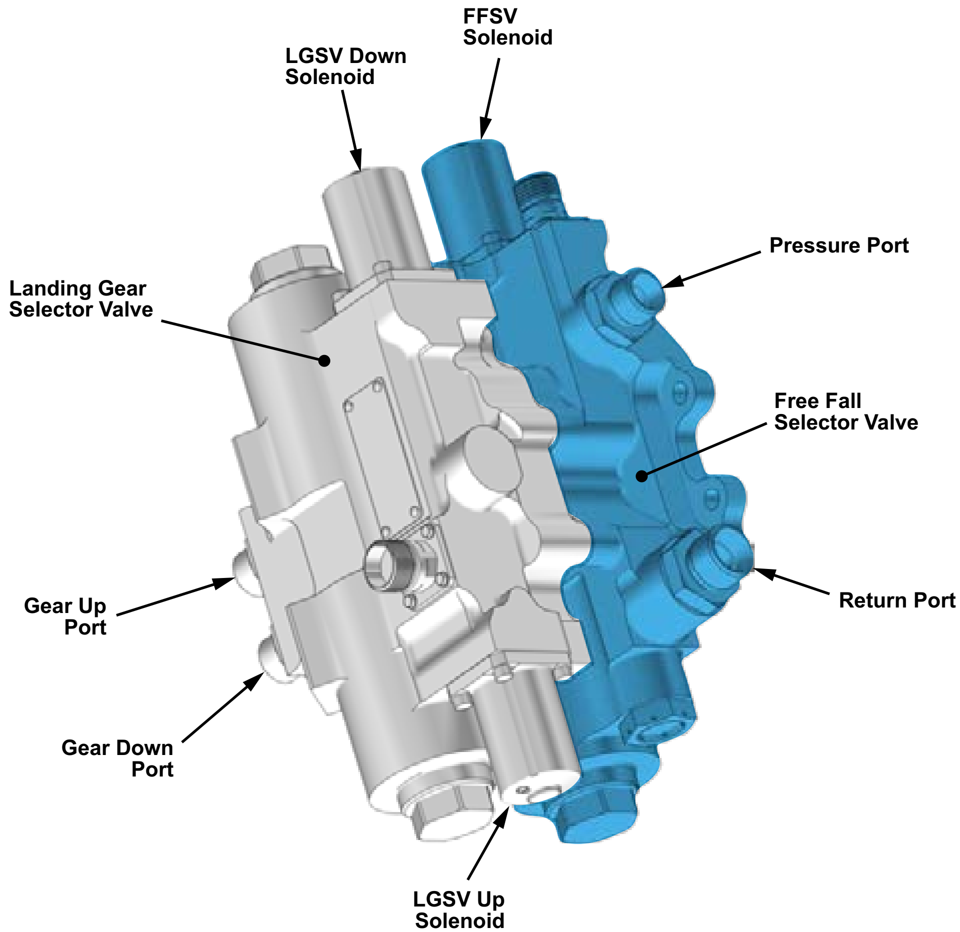
The LGCV has four hydraulic ports as follows:

- Pressure port
- Return port
- Gear up port
- Gear down port

The LGSV has an up and a down solenoid. The FFSV has a single solenoid. The FFSV solenoid is normally energized and routes hydraulic pressure to the LGCV solenoids.

During landing gear retraction or extension, the appropriate LGCV up or down solenoid is electrically energized, and directs fluid pressure to the respective gear up port or gear down port.

During alternate extension the FFSV and LGSV solenoids are electrically de-energized. When de-energized the FFSV solenoid prevents hydraulic fluid from entering the LGSV. At the same time, the gear up port and the gear down port are connected to the return port, dumping hydraulic pressure, enabling the landing gear to free fall.



LEGEND

FFSV Free Fall Selector Valve
LGSV Landing Gear Selector Valve

Figure 21: Landing Gear Control Valve

MAIN LANDING GEAR RETRACTION ACTUATOR

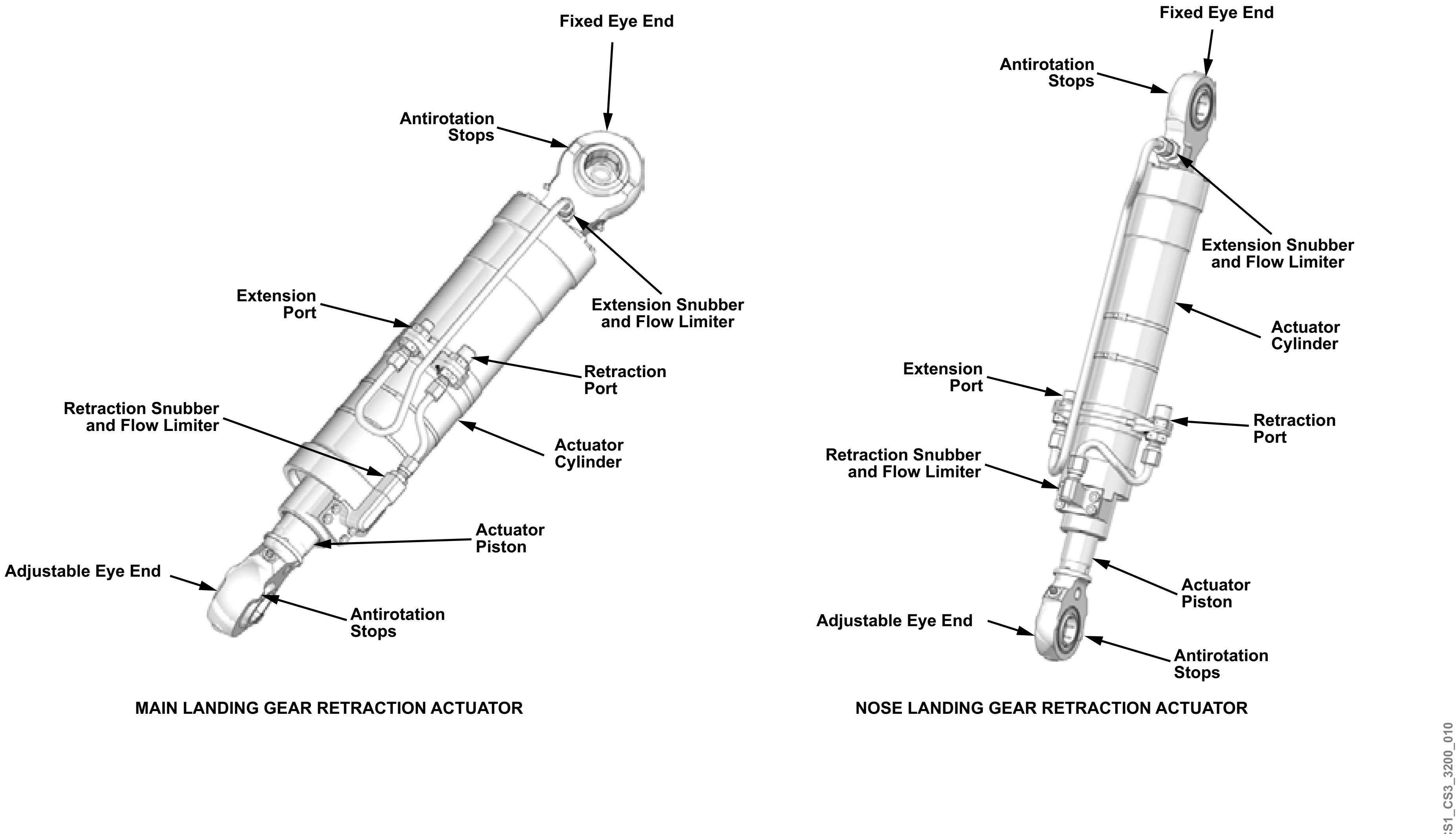
The main landing gear (MLG) retraction actuator is a double-acting hydraulic actuator. It consists of an actuator cylinder with an actuator piston. The actuator extension and retraction hydraulic lines are connected to externally-mounted extension and retraction ports. Internal flow limiters limit the speed of the piston movement. Internal snubbers provide a damping affect.

The fixed eye end is attached to the wing structure, and the adjustable eye end is attached to the a lug on the MLG main fitting.

Both eye ends are provided with spherical bearings and antirotation stops. These stops restrict actuator rotation.

NOSE LANDING GEAR RETRACTION ACTUATOR

Except for size, the nose landing gear (NLG) retraction actuator has the same design features as the MLG retraction actuator. The fixed eye end is mounted on the aircraft structure. The adjustable eye end is attached to the NLG main fitting.



MAIN LANDING GEAR RETRACTION ACTUATOR

NOSE LANDING GEAR RETRACTION ACTUATOR

Figure 22: Landing Gear Retraction Actuators

MAIN LANDING GEAR UNLOCK ACTUATOR

The main landing gear (MLG) unlock actuator is a double acting type hydraulic actuator. Its function is to unlock the locking stay overcenter position. It consists of an actuator cylinder with an actuator piston.

Flow control valves, installed at the extension and retraction ports, limit piston speed and provide damping. Damping reduces the impact on the locking stay at the end of the extension and retraction cycle. The unlock actuator cylinder is attached to the MLG upper side brace, and the actuator piston is attached to the locking stay upper member. The MLG unlock actuator unlocks the locking stay at a hydraulic supply pressure above 1200 psi.

NOSE LANDING GEAR UNLOCK ACTUATOR

Except for size, nose landing gear (NLG) unlock actuator has the same design characteristics as the MLG unlock actuator.

The NLG unlock actuator cylinder is attached to the NLG main fitting, and the actuator piston is attached to the NLG upper locking stay.

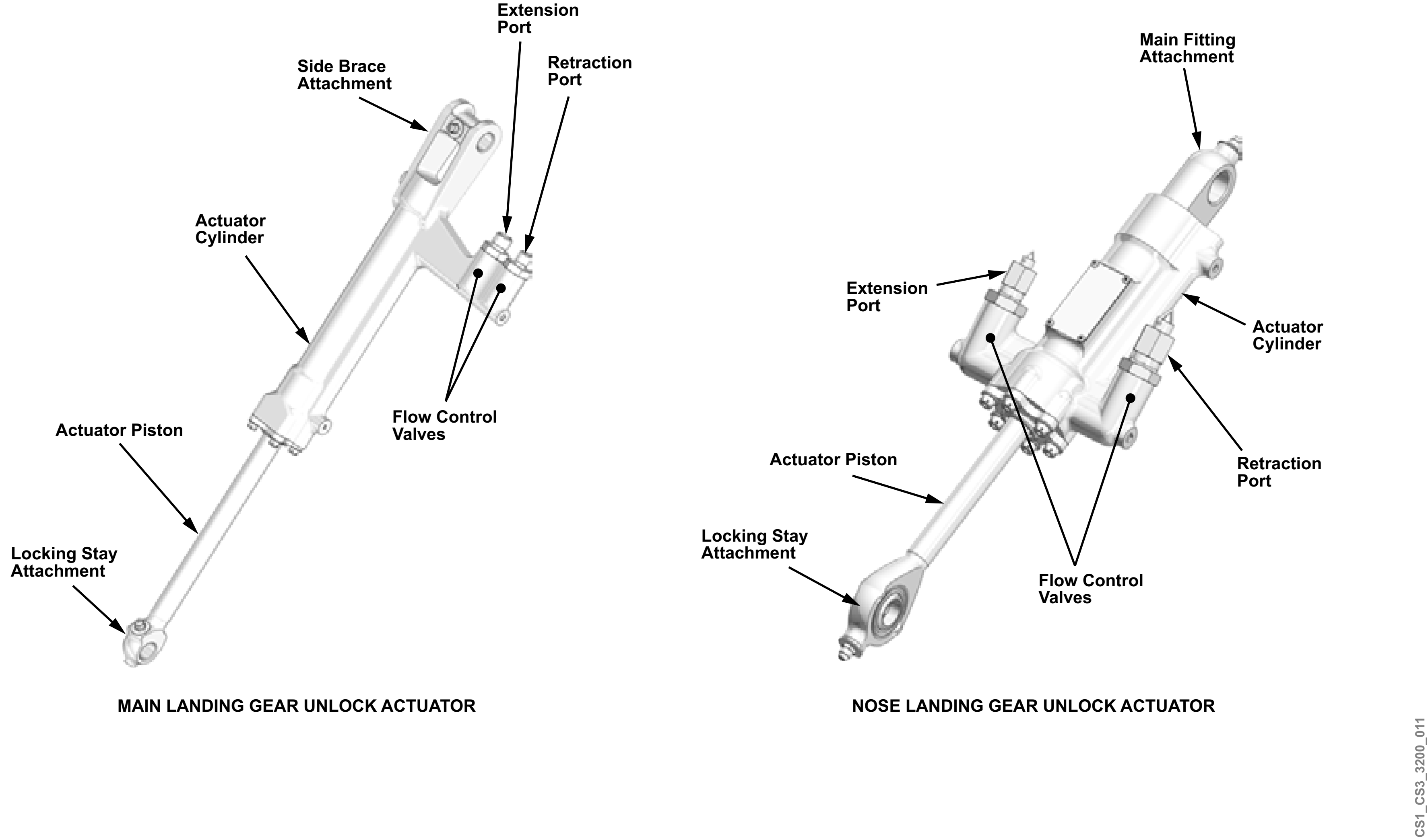


Figure 23: Landing Gear Unlock Actuators

UPLOCK

The uplock hook engages an uplock roller on the nose or main landing gear to hold it in the retracted position. During normal gear extension, hydraulic pressure is used to unlock the hook. During alternate extension, electrical power is used to unlock the hook.

During normal extension, hydraulic pressure from hydraulic system no. 1 applied at the pressure port, extends the plunger against its internal spring. The plunger pushes on the hydraulic activation arm, which releases the hook, assisted by the hook springs. In turn, the hook rotation releases the uplock roller. The two hook springs hold the hook in the unlock position, which must remain open to receive the gear uplock roller on gear retraction.

During retraction, the uplock roller of the retracting gear forces the hook into the locked position. The hook is mechanically held in the locked position by the activation arm and the hook springs. A stop pin limits hook movement in the locked and unlocked positions.

Two proximity sensors (on opposite sides of the uplock) sense the position of the activation arm and provide lock/unlock information to the LGSCUs.

During alternate extension, a DC motor drives a worm gear which rotates the cam disk. Rotation of the cam disk contacts a cam roller on the electrical activation arm, opening the hook. The torque applied by the cam disk is sufficient to overcome the friction caused by the weight of the uplock roller on the hook.

A microswitch, which follows the cam disk, is used to maintain electrical power to the DC motor until the worm gear has made a complete rotation. After 5 seconds electric power to the motor is automatically shutoff.

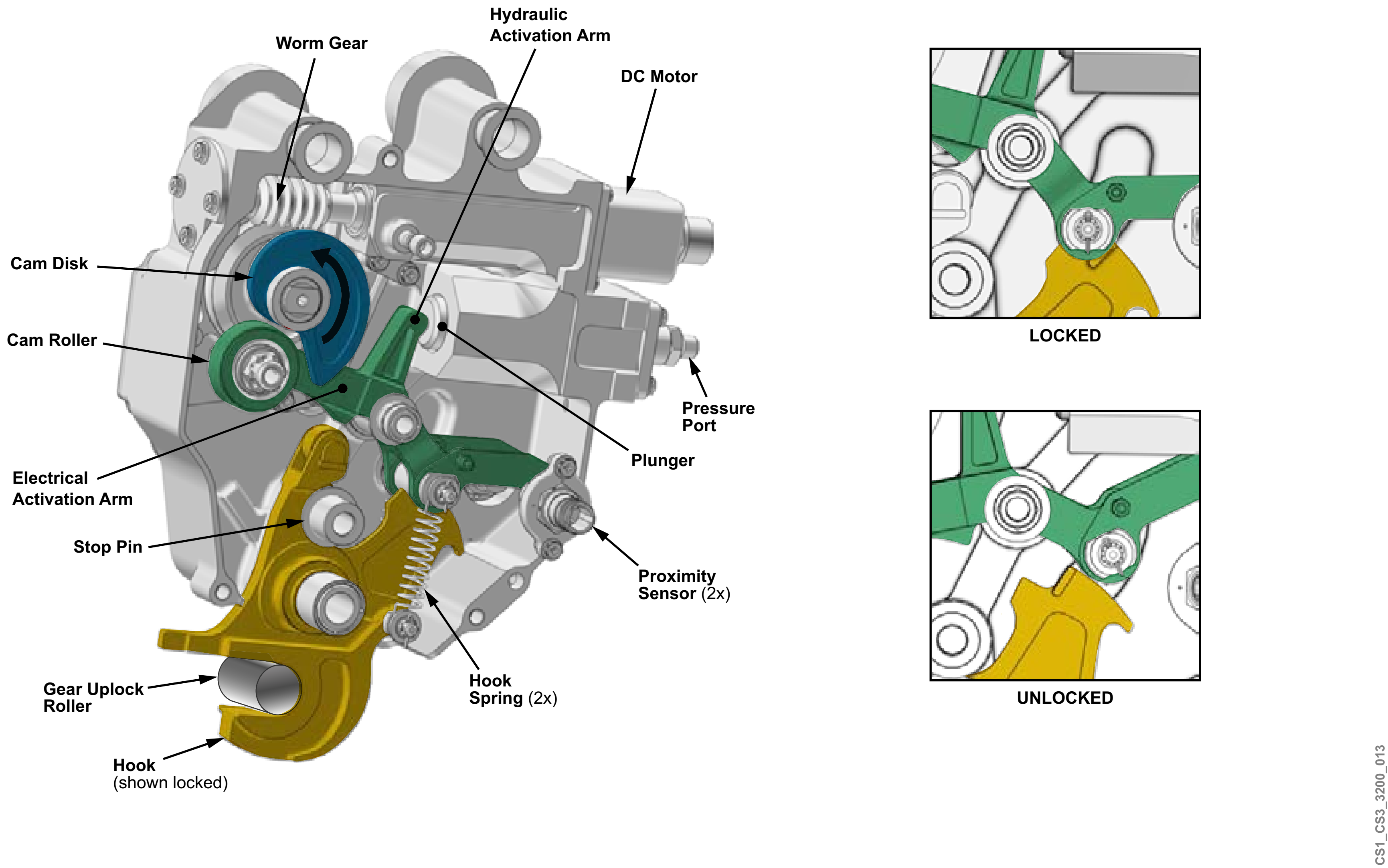


Figure 24: Landing Gear Uplock

CONTROLS AND INDICATIONS

EXTENSION AND RETRACTION CONTROLS

Landing Gear Control Lever

The landing gear control lever (LGCL) is located on the landing gear panel. Normal landing gear extension and retraction is carried out by operating the LGCL.

The lever has two positions:

- UP
- DN

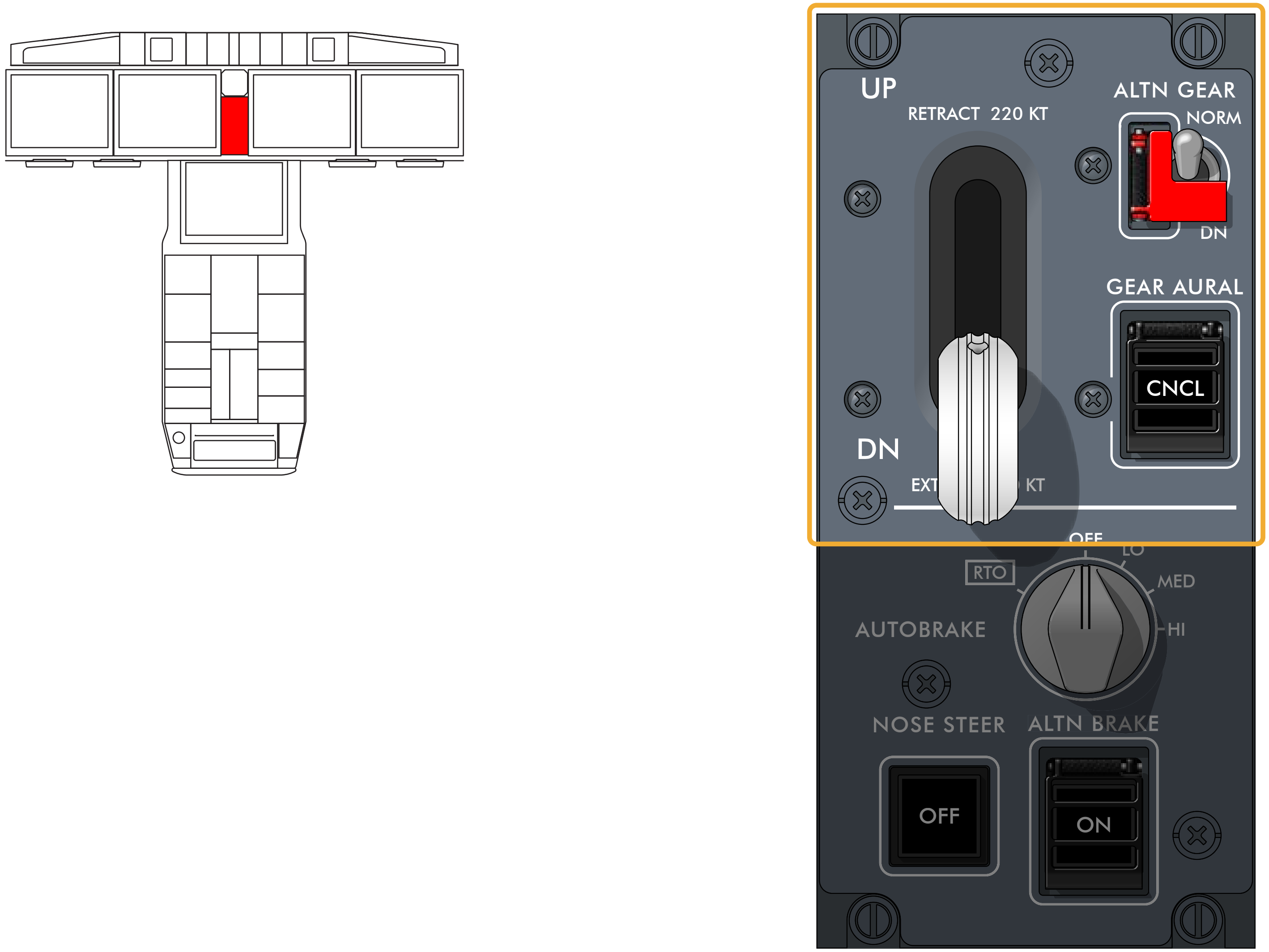
In order to make a selection on the lever, it must be pulled out of its detent and moved up or down. An internal cam in the LGCL ensures an UP or DN position. The LGCL does not have a neutral position selection.

ALT Gear Switch

The ALT GEAR switch is located on the landing gear panel. It is a guarded two-position toggle switch that has a NORM and DN selection. The switch is guarded to the normal position.

Gear Aural PBA

The GEAR AURAL pushbutton annunciator (PBA) is guarded and located on the landing gear panel. Selecting the PBA mutes the gear aural warning. The CNCL (cancel) legend illuminates to advise the crew that cancellation of the gear aural warning is active.



LANDING GEAR PANEL

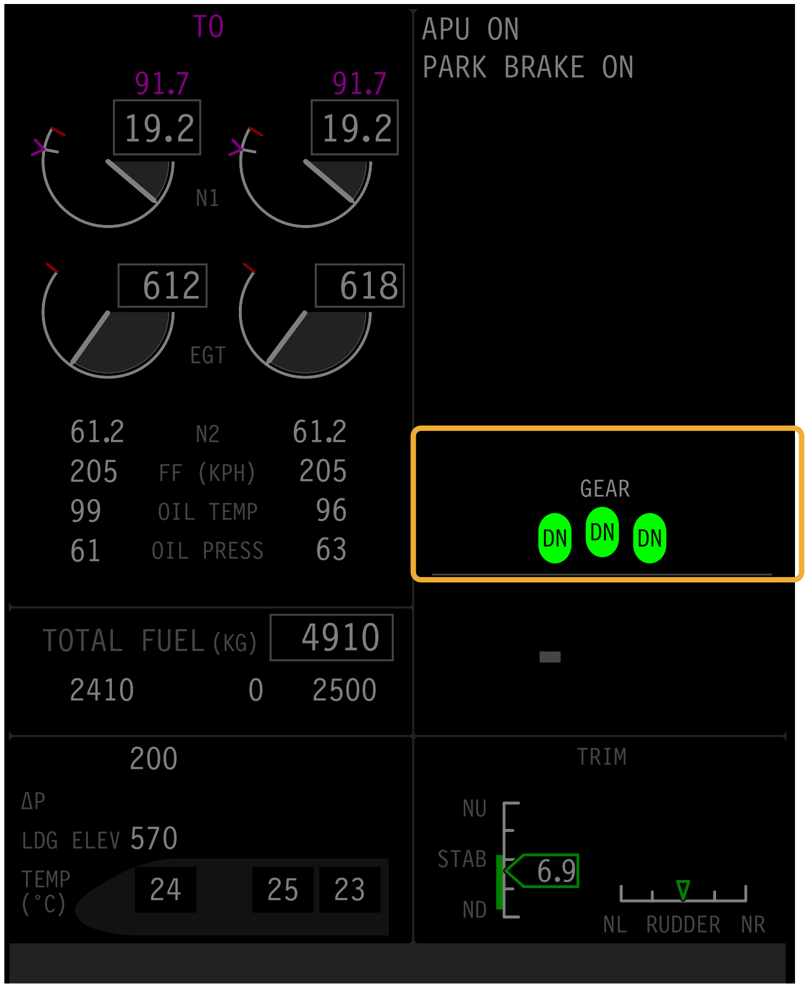
CS1_CS3_3200_016

Figure 25: Landing Gear Panel

EXTENSION AND RETRACTION INDICATIONS

Indication of the status and condition of the main and nose landing gear is displayed as color symbols on the EICAS page.

During approach and landing, if the gear is not down and locked the symbol displays a red box. This is accompanied by the red GEAR warning message on the EICAS and the gear aural warning. The aural can be muted by selection of the GEAR AURAL PBA on the landing gear panel.



EICAS PAGE

LANDING GEAR CONTROL PANEL

GEAR POSITION	
DN	Gear down and locked.
UP	Gear up and locked.
UP	After time delay, gear up and LGCL down.
UP	During approach or landing, after 30 seconds time delay, gear up and LGCL down. Accompanied by GEAR CAS warning message and aural warning.
	Gear in transit.
	After time delay, gear position not consistent with LGCL position.
	During approach or landing, after 30 seconds time delay, gear position not consistent with LGCL. Accompanied by GEAR CAS warning message and aural warning.
--	Unknown or invalid.
	(No indication) • Gear up and locked for more than 30 seconds.

LEGEND

LGCL Landing Gear Control Lever

CS1_CS3_3200_022

Figure 26: Extension and Retraction Indications

LANDING GEAR AURAL WARNING

If a landing is attempted, and the aircraft is not properly configured, a continuous GEAR aural warning is generated. Under certain conditions, the warning is mutable using the GEAR AURAL PBA. A white CNCL legend illuminates to indicate that the landing gear horn aural warning is muted.

MUTABLE LANDING GEAR AURAL WARNING

A mutable landing gear warning sounds when the landing gear is not down and locked, and the following conditions exist:

- Barometric altitude less than 15,000 ft
- All radio altimeters failed - no indication signal
- Both thrust levers at idle
- Flap/slat lever set to landing position

NON-MUTABLE LANDING GEAR AURAL WARNING

A non-mutable landing gear aural warning sounds when the landing gear is not down and locked, the radio altimeter is less than 1000 ft, and either of the following conditions exist:

- Flap/slat lever set to landing position
  AND
  Vertical descent rate is greater than 250 ft/min for 3 seconds
- Both thrust lever angles set less than 15°

LANDING GEAR AURAL WARNING RESET

The landing gear aural warning is automatically reset for any of the following conditions:

- Any thrust lever above idle
- Radio altimeter increases above 1100 ft
- Landing gear retraction

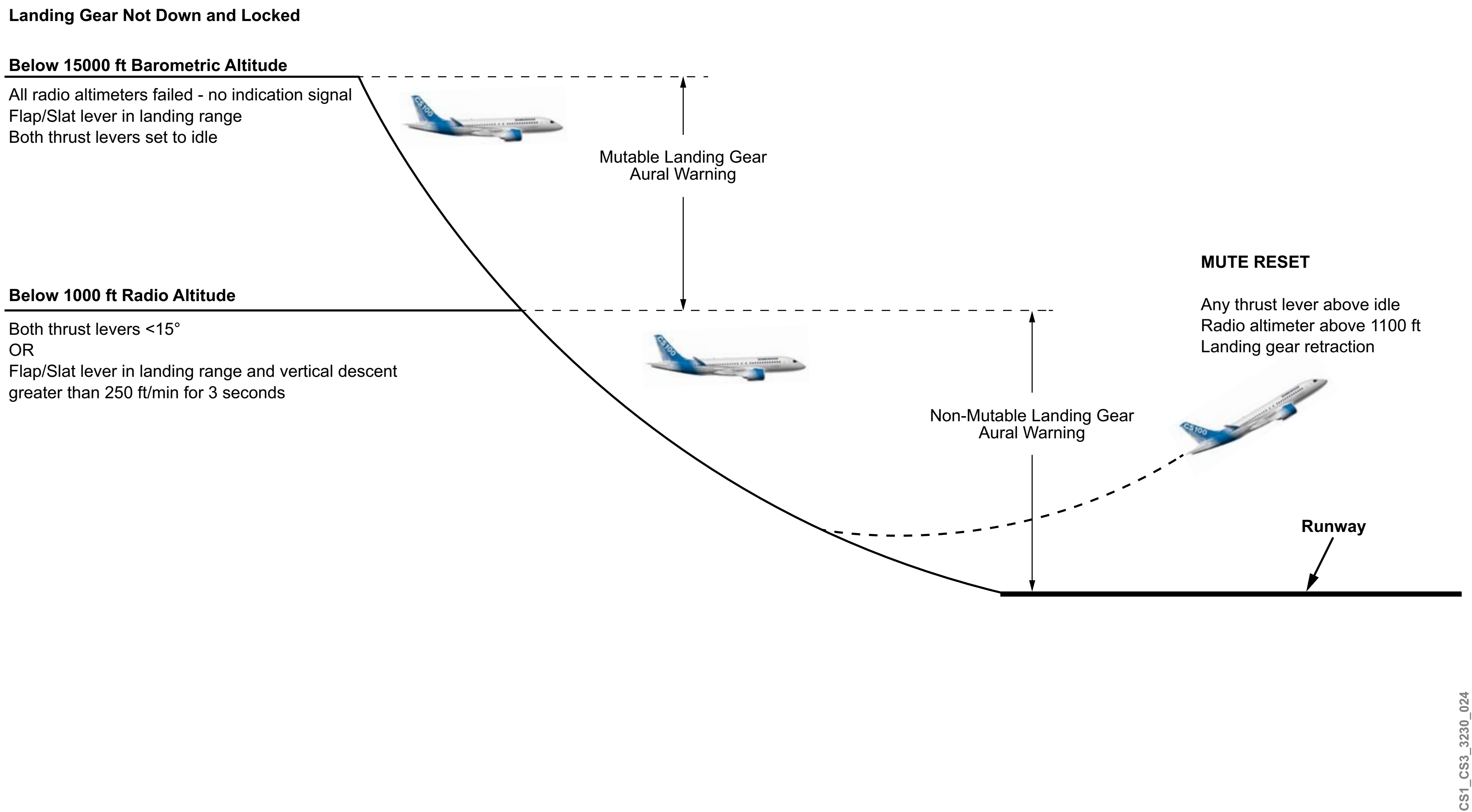


Figure 27: Landing Gear Aural Warnings

OPERATION

LANDING GEAR RETRACTION

Landing gear retraction is initiated by moving the landing gear control lever (LGCL) to the UP position.

The LGCL sends a gear up command to the LGSCUs. The aircraft must be in flight or on jacks (WOFFW) and the nose landing gear must be centered for the LGSCUs to accept the command. The LGSCUs send a command signal to energize the FFSV solenoid and LGSV up solenoid. The FFSV solenoid valve opens allowing hydraulic system no. 1 pressure to the LGSV up solenoid valve. This ports hydraulic pressure to the unlock actuators, which retract to overcome the downlock springs and break the overcenter lock on the locking stay.

Once the downlock proximity sensors detect that the gear is unlocked, the LGSCUs send position information to EICAS through the data concentrator unit module cabinet (DMC). The three GEAR symbols on the EICAS page change to white diagonal in-transit symbols.

In addition, the retraction actuators receive hydraulic pressure to extend, moving the gear to their up and locked positions. An uplock roller on each gear main fitting engages with the uplock hook. The hook is closed and mechanically locked. This action locks the gear in the up position.

Two proximity sensors in each uplock indicate that the gear is up and locked. The LGSCUs deactivate the FFSV and LGSV solenoid valves 2 seconds after the landing gear uplock status is confirmed.

When de-energized, the FFSV closes to isolate the LGSV from hydraulic system no. 1 pressure. The hydraulic pressure in the landing gear system depletes by porting fluid pressure to return.

When the uplock proximity sensors detect the hooks are engaged, the EICAS page GEAR symbols change to white UP symbols. Thirty seconds after the landing gear is up and locked, the indication is removed from EICAS.

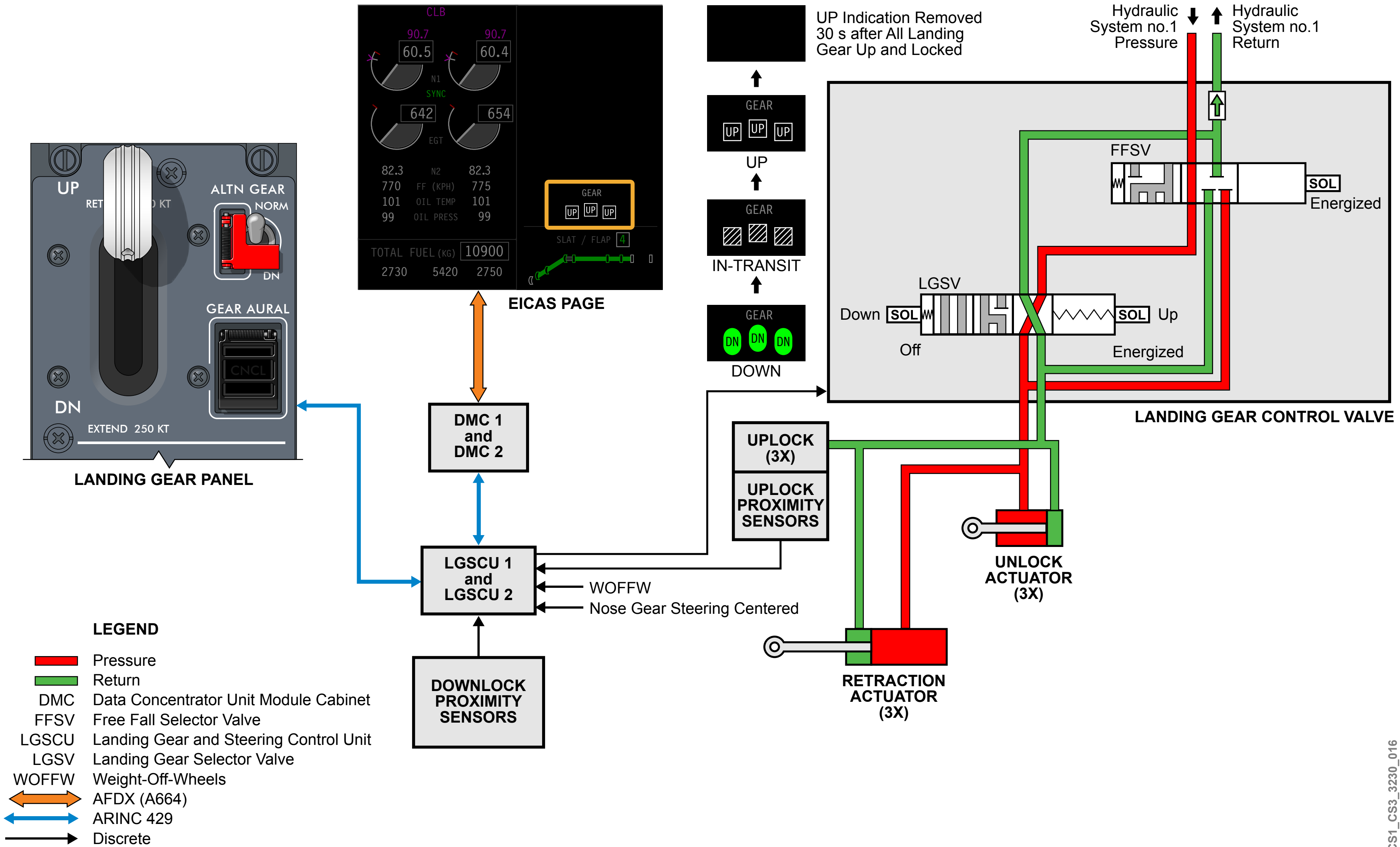


Figure 28: Landing Gear Retraction

NORMAL EXTENSION

Normal extension is initiated by moving the landing gear control lever (LGCL) to the DN position. Once activated the LGSCU performs a free fall selector valve (FFSV) jam open test. If the FFSV jam open test is passed, the normal extension sequence continues.

The LGCL sends a gear down command to the LGSCU. The LGSCU initially commands a gear retraction. This pressurizes the retraction actuators, raising the gear to reduce the weight on the uplock hooks while simultaneously releasing the hooks.

The uplock proximity sensors signal the unlocked state of the hooks. The LGSCU transmits this data to the EICAS, which changes the UP symbols to white diagonal in-transit symbols.

After 2 seconds, the LGSCU commands gear extension. The LGSCU energizes the FFSV solenoid and LGSV down solenoid. The LGSV ports hydraulic pressure to the retraction actuators, which retract to lower the landing gear to the extend position.

As the gears extend, hydraulic pressure extends the unlock actuators, which forces the locking stays to the overcenter position. Two downlock springs on each landing gear push and hold the locking stays at their over-center stop position. Two downlock proximity sensors on each locking stay indicate the down and locked position. Once the proximity sensors detect the locking stay is locked, the GEAR symbols change to green DN symbols.

As long as hydraulic pressure is available, the unlock actuators remain pressurized to the extended position, maintaining a positive overcenter on the locking stay. Once hydraulic pressure depletes, the locking stay overcenter is maintained by the downlock springs on each landing gear. A locking pin can be inserted to secure the locking stay in the overcenter position.

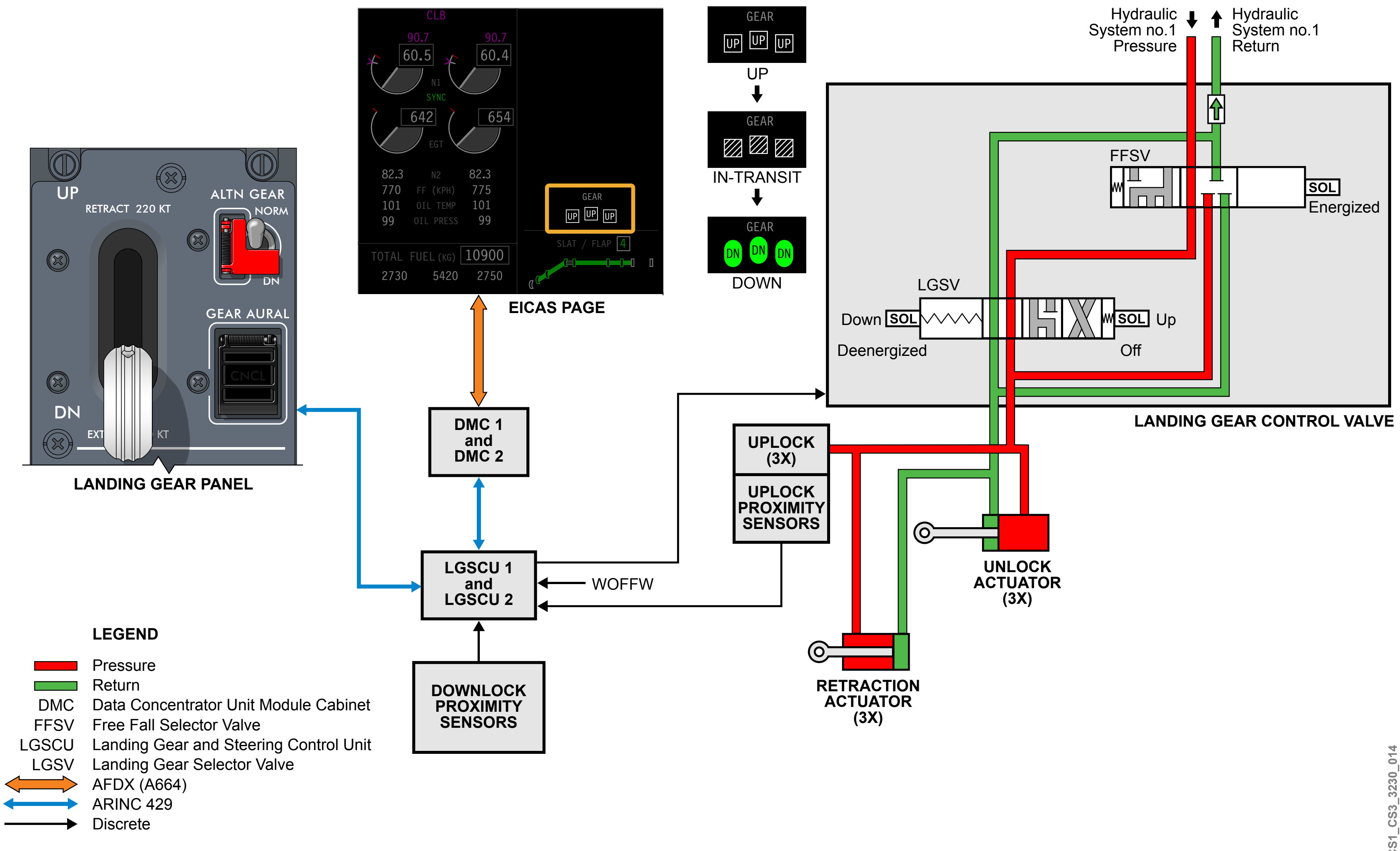


Figure 29: Normal Extension

ALTERNATE EXTENSION SYSTEM

The alternate extension system (AES) enables the free fall of the landing gear without hydraulic pressure. Power for the AES is provided from DC ESS BUS 3.

Landing gear alternate extension is initiated by moving the switch guard and selecting the ALTN GEAR switch to the DN position.

The DN selection routes 28 VDC to the DC motor in each uplock through an energized timer relay. The DC motor mechanically releases the uplock hook allowing the gear to free fall to the full extended position. After 5 seconds the timer relay de-activates, cutting power to the DC motors in the uplocks.

The DN selection also de-energizes all LGCV solenoids. The de-energized FFSV solenoid valve isolates the LGSV from hydraulic supply pressure. The de-energized LGSV and solenoid valves route all fluid to return. This removes all residual hydraulic pressure in the landing gear system.

When the uplocks are released, the uplock proximity sensors provide position information from the LGSCUs to EICAS. The three white UP symbols change to three white diagonal in-transit symbols on the EICAS page.

After a free fall extension, each landing gear is locked in the extended position by two downlock springs, which pull the locking stays into an overcenter configuration. The downlock proximity sensors on the locking stays signal the LGSCU, which relays the information to EICAS. The in-transit symbols change to green DN symbols.

Alternate Extension Dormancy Check

At every 550 weight-off-wheels (WOFFW) hours an automatic test is made of the alternate extension system (AES). Upon the next normal extension command (via the landing gear control lever), the LGSCUs activate landing gear relays within the electrical power center (EPC).

This commands an alternate extension instead of the normal extension. As a result, the electric motor of the uplock unlocks the hook, and the landing gear extend by free fall. After the landing gears are detected down and locked, the LGSCUs de-activate the relays, and the electric motor is powered off.

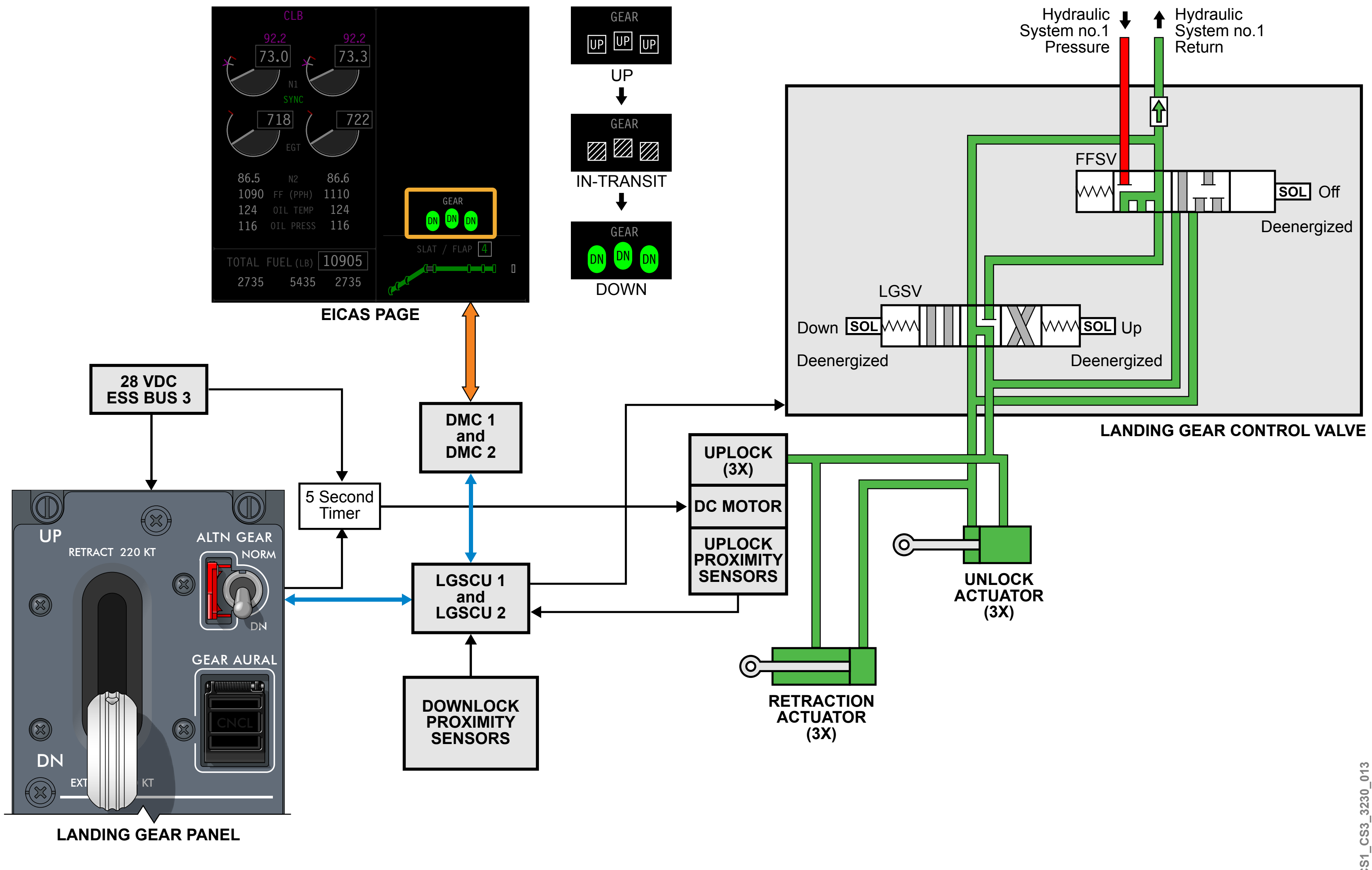


Figure 30: Alternate Extension

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the landing gear extension and retraction system.

CAS MESSAGES

Table 1: WARNING Message

MESSAGE	LOGIC
GEAR	Attempt to land is detected and any gear is not down and locked.

Table 2: CAUTION Messages

MESSAGE	LOGIC
GEAR DISAGREE	Discrepancy in landing gear position and commanded position of the LGCL handle after time delay.
GEAR FAIL	LG extend/retract system failed or no communication available.

Table 3: ADVISORY Messages

MESSAGE	LOGIC
GEAR FAULT	LG extended/retract system redundancy is lost or dormancy test detected failure of alternate extension. Refer to INFO messages.
ALTN GEAR DN	Alternate extension dormancy check is performed on the uplocks.

Table 4: STATUS Messages

MESSAGE	LOGIC
ALTN GEAR DN	Alternate extension switch is detected in the down position.
GEAR AURAL CNCL	Pilot silenced a mutable gear aural.

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Table 5: INFO Messages

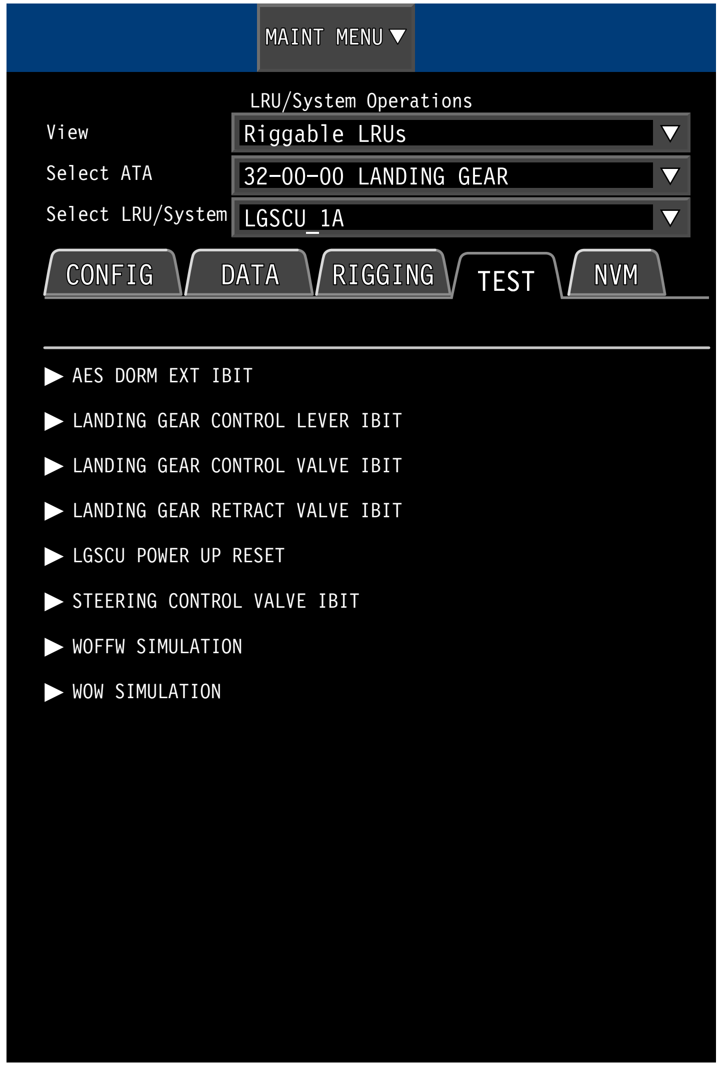
MESSAGE	LOGIC
32 GEAR FAULT - GEAR REDUND LOSS	One LGSCU detected an internal redundancy fault, or no/invalid date from one LGSCU to avionic.
32 GEAR FAULT - ALTN EXT INOP	Alternate extension dormancy test is reported as failed.
32 GEAR FAULT - LGCV REDUND LOSS	One LGSCU has detected a loss of electrical redundancy in the LG control valve.
32 GEAR FAULT - LGCL REDUND LOSS	One LGSCU has detected a loss of electrical redundancy in the LG control lever.
32 GEAR FAULT - LGCL INOP	Both LGSCUs detect failures on LGCL switches or LGCL is in UP position while ALTERNATE EXTENSION switch is activated.
32 GEAR FAULT - 28 V ESS REDUND LOSS	One LGSCU has detected a loss of electrical redundancy, 28 V ESS power is lost which results in all solenoids powered by this LGSCU being inoperative.
32 GEAR FAULT - 28 V NORM REDUND LOSS	One LGSCU has detected a loss of electrical redundancy, 28 V normal power is lost.
32 GEAR FAULT - LGSCU INTNL FAULT	One LGSCU detected a critical internal fault and requires complete isolation of the affected LGSCU (ie.CBs pulled, LGSCU removal).
32 GEAR FAULT - LGSCL SWITCH FAULT	One LGSCU detected a critical LGCL fault and requires complete isolation of the affected LGCL (ie.LGSCU CBs pulled, LGSCU removal).
32 GEAR FAULT - ALTN EXT RLY 1, 2 CONTACT FAULT	One LGSCU detected a critical AES RELAY 1, 2 fault and requires complete isolation of the affected relay 1, 2 circuit (ie.affected circuit CBs pulled, relay removal).
32 GEAR FAULT - L, R GEAR UPLK MOTOR FAULT	One LGSCU detected L, R MLG uplock motor not in neutral position. Can cause inability to uplock L, R gear.
32 GEAR FAULT - NOSE GEAR UPLK MOTOR FAULT	One LGSCU detected NLG uplock motor not in neutral position. Can cause inability to uplock NLG.

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM

The onboard maintenance system (OMS) can be used to perform tests of the extension and retraction system as follows:

- Landing gear control lever initiated built-in test (IBIT)
- Landing gear control valve IBIT
- Landing gear retract valve IBIT



CS1_CS3_3230_025

Figure 31: Extension and Retraction OMS Functions

32-61 LANDING GEAR INDICATING SYSTEM

GENERAL DESCRIPTION

The landing gear indicating system (LGIS) provides indication of the position and condition of the main landing gear and nose landing gear. It also provides indication of weight-off-wheels (WOFFW) status, used by many aircraft systems.

The landing gear and steering control units (LGSCUs) receive information provided by proximity sensors for gear retraction and extension and WOFFW sensing. Gear position is displayed on the engine indication and crew alerting system (EICAS), through the data concentrator unit module cabinets (DMCs). The system is monitored by the onboard maintenance system (OMS)

Each landing gear assembly has two sets of proximity sensors.

LGSCU 1 receives power from DC ESS BUS 1 and DC BUS 2. LGSCU 2 receives power from DC ESS BUS 2 and DC BUS 1.

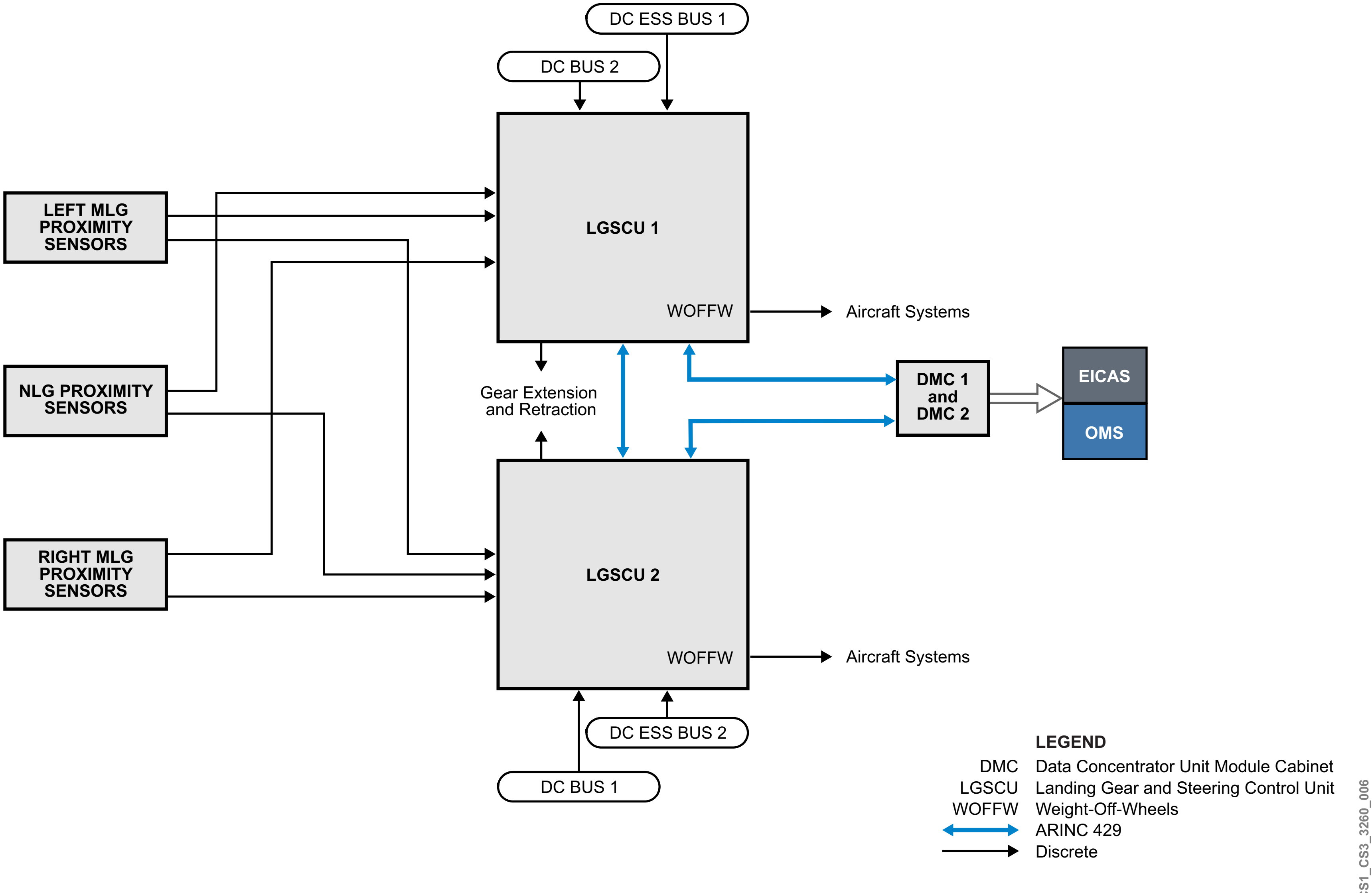


Figure 32: Landing Gear Indicating System - Simplified

COMPONENT LOCATION

UPLOCK PROXIMITY SENSORS

Two uplock proximity sensors are located in each of the main landing gear (MLG) and the nose landing gear (NLG) uplocks. The targets for the proximity sensors are internal to the uplocks.

DOWNLOCK PROXIMITY SENSORS

The downlock proximity sensors are located on the locking stay of the MLG and NLG. Each proximity sensor interfaces with a steel target.

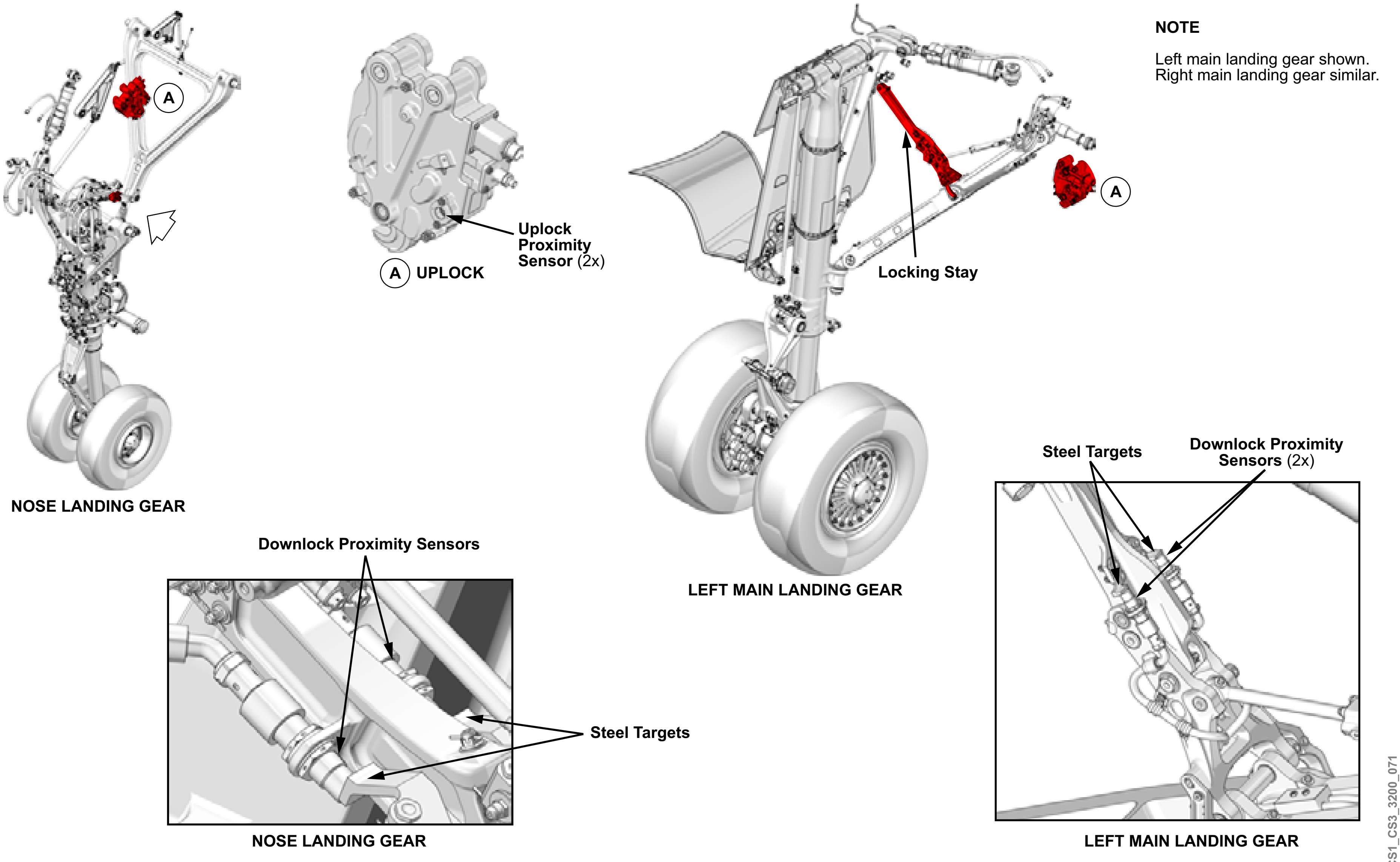


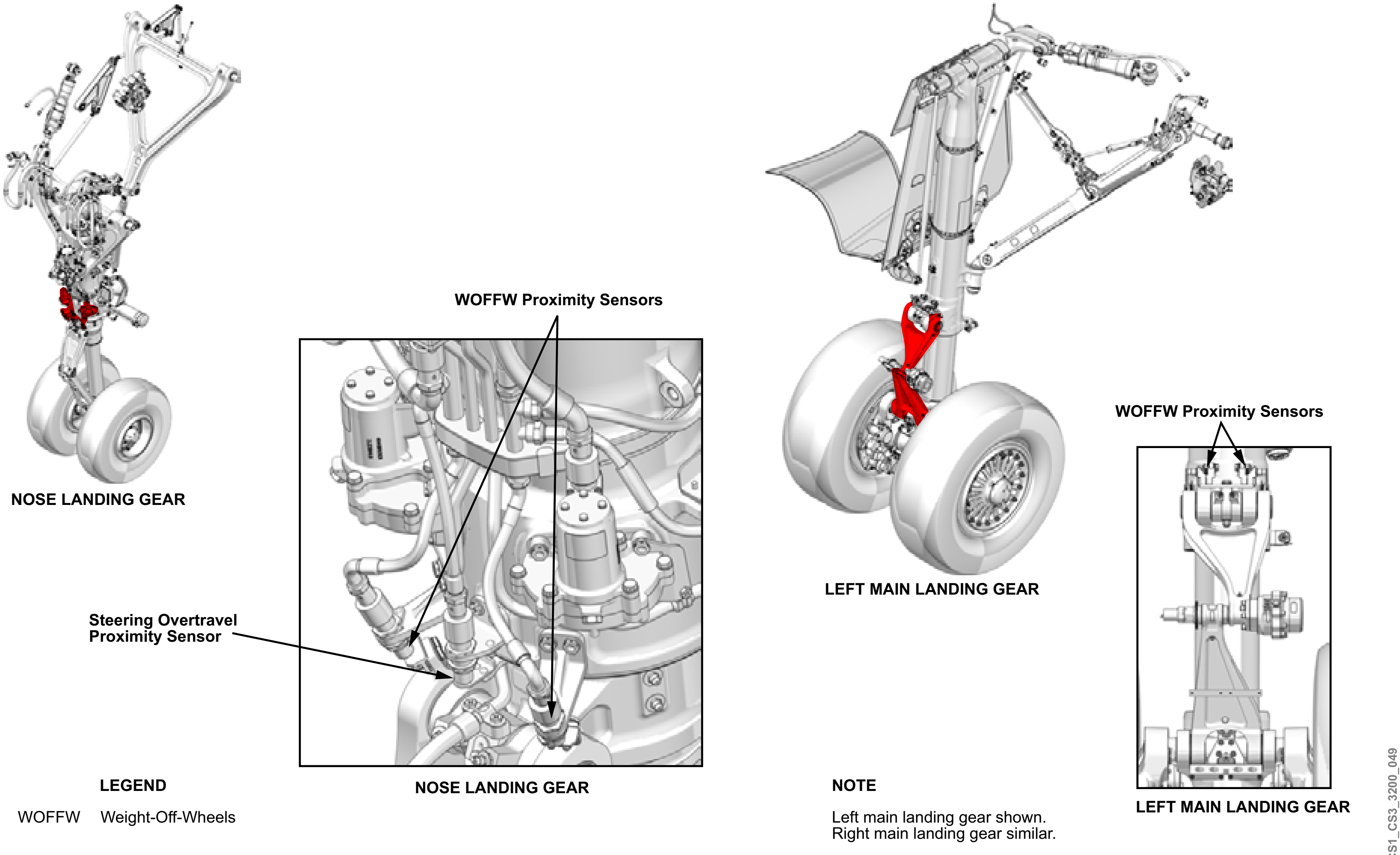
Figure 33: Landing Gear Indicating System Component Location

WEIGHT-OFF-WHEELS PROXIMITY SENSORS

The weight-off-wheels (WOFFW) proximity sensors are located on the main fitting of the MLG and NLG.

STEERING OVERTRAVEL PROXIMITY SENSOR

The steering overtravel proximity sensor is the main fitting of the nose landing gear.



LEGEND

WOFFW Weight-Off-Wheels

NOTE

Left main landing gear shown.
Right main landing gear similar.

Figure 34: Landing Gear Indicating System Component Location

DETAILED COMPONENT INFORMATION

LANDING GEAR AND STEERING CONTROL UNIT

The LGSCUs process and control the landing gear system operation.

The LGSCUs are each divided into a control path and a monitor path. The two paths crosstalk via ARINC 429. Each control path is further subdivided into a landing gear path and a steering path.

The LGSCUs operate in an active-active mode for landing gear control, and in an active-standby mode for steering control.

Each LGSCU is powered by two independent electrical power sources as follows:

- LGSCU 1 is powered by DC BUS 2 and DC ESS BUS 1
- LGSCU 2 is powered by DC BUS 1 and DC ESS BUS 2

The LGSCU processes inputs from the following components:

- Landing gear control lever (LGCL)
- Uplock proximity sensors
- Downlock proximity sensors
- WOFFW proximity sensors

Position signals from these components are received via discrete signals. The signals are communicated via ARINC 429 to the data concentrator unit module cabinets (DMCs) and the primary flight control computers (PFCCs), as well as being communicated between the two LGSCUs.

Control Path

The main function of the landing gear control path is to:

- Process inputs and transmit commands from the landing gear electromechanical and hydromechanical components
- Receive and transmit system signals from and to the DMCs
- Receive and transmit system signals from and to the PFCCs
- Communicate system input signals with the monitor path
- Communicate system input signals with the other LGSCU
- Perform an automatic alternate extension system (AES) relay dormancy check, every 50 hours, on cold power up
- Perform an automatic AES uplock motor dormancy check every 550 weight-off-wheels (WOFFW) hours.

Monitor Path

The main function of the monitor path is to:

- Process inputs from the landing gear position and sensing components
- Provide excitation voltage to the proximity sensors
- Receive and transmit system signals from and to the DMCs
- Receive and transmit system signals from and to the PFCCs
- Communicate system input signals with the control path
- Communicate system input signals with the other LGSCU

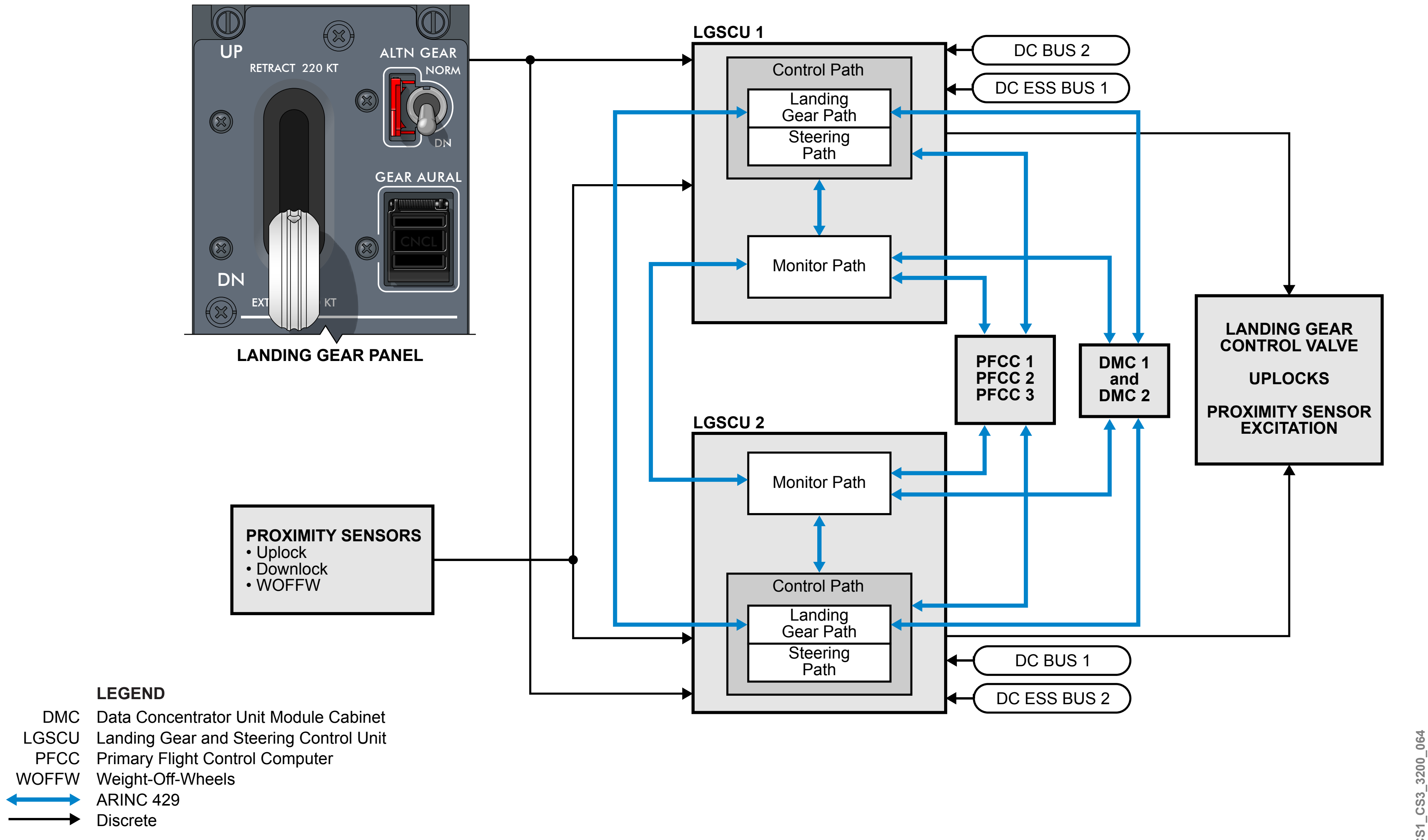


Figure 35: Landing Gear and Steering Control Unit Interface

DETAILED DESCRIPTION

The landing gear indicating system (LGIS) provides indication of the position and condition of the main landing gear and nose landing gear. It also provides indication of WOFFW status, used by many aircraft systems.

The LGSCUs receive information provided by proximity sensors for gear retraction and extension and WOFFW sensing. Gear position is displayed on the EICAS, through the DMCs. The system is monitored by the OMS.

Each of the main and nose landing gear have two sets of proximity sensors for monitoring the following:

- Weight-off-wheels (WOFFW)
- Downlock
- Uplock

In addition, the nose landing gear has an steering overtravel proximity sensor that is used in the nosewheel steering system.

The no. 1 proximity sensors are connected to LGSCU 1. The no. 2 sensors are connected to LGSCU 2. The LGSCUs provide excitation of the associated proximity sensors.

Failure of WOFFW sensors display as CAS INFO messages.

LGSCU 1 receives power from DC ESS BUS 1 and DC BUS 2. LGSCU 2 receives power from DC ESS BUS 2 and DC BUS 1.

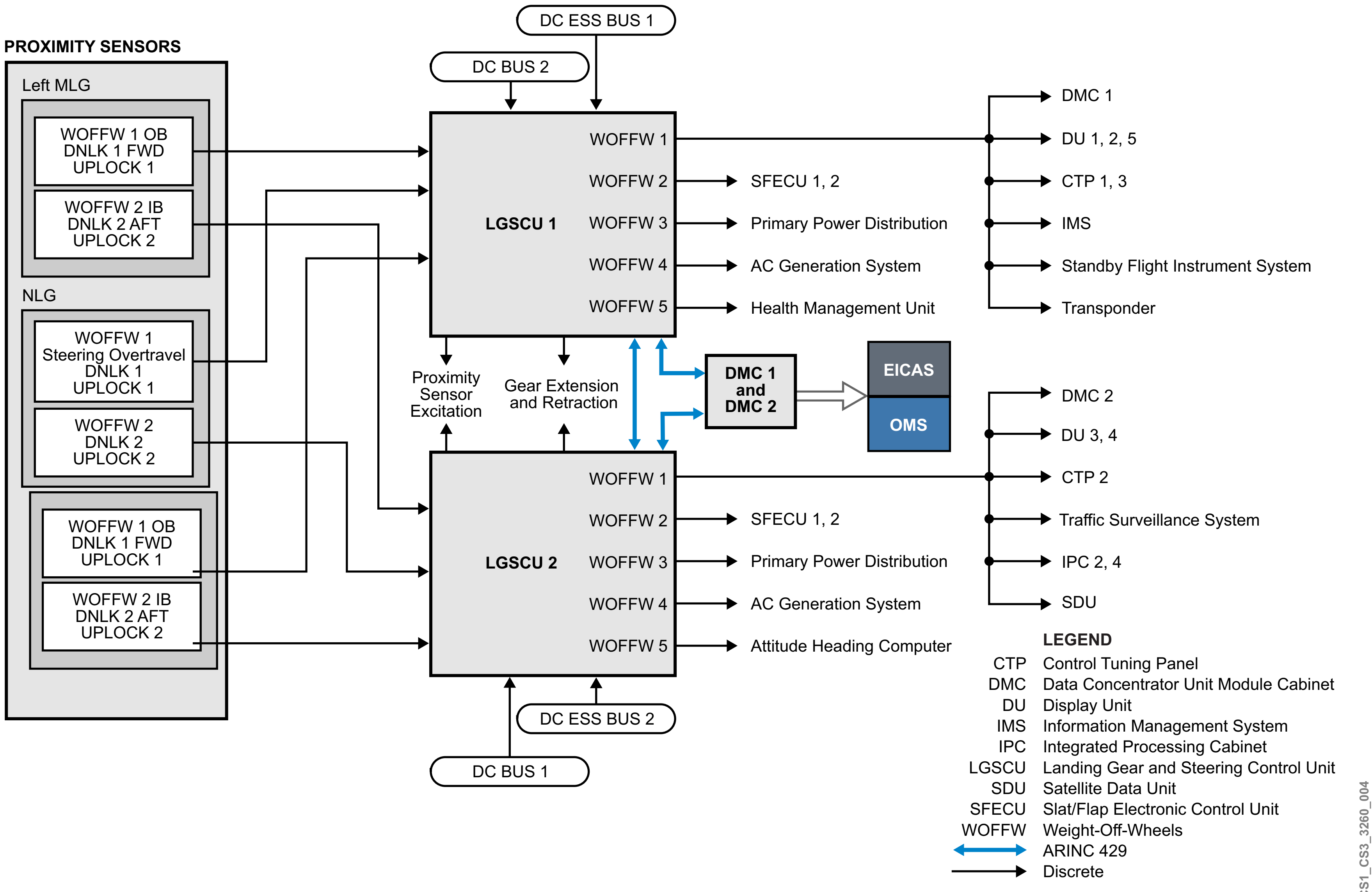


Figure 36: Landing Gear Indicating System

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the landing gear indicating system.

CAS MESSAGES

Table 6: CAUTION Message

MESSAGE	LOGIC
WOW FAIL	Weight-on-wheels (WOW) functionality lost.

Table 7: ADVISORY Message

MESSAGE	LOGIC
WOW FAULT	WOW functionality system redundancy is lost. Refer to INFO messages.

Table 8: INFO Messages

MESSAGE	LOGIC
32 GEAR FAULT - MULTIPLE GEAR REDUND LOSS	Both LGSCUs detected an internal redundancy fault, or no/invalid data from one LGSCU to the DMC.
32 GEAR FAULT - LGSCU 1 REDUNC LOSS	Both LGSCUs detected an internal redundancy fault, or no/invalid data from one LGSCU to the DMC.
32 GEAR FAULT - LGSCU 2 REDUNC LOSS	LGSCU 2 detected an internal redundancy fault, or no/invalid data from LGSCU 1 to the DMC.
32 GEAR FAULT - GEAR DNLK SENSORS INOP	All downlock sensors of one or more gears have an electrical fault or unreasonable state.
32 GEAR FAULT - GEAR UPLK SENSORS INOP	All uplock sensors of one or more gears have an electrical fault or unreasonable state.
32 GEAR FAULT - GEAR DNLK REDUND LOSS	One downlock sensor of at least one gear has an electrical fault or unreasonable state.

Table 8: INFO Messages

MESSAGE	LOGIC
32 GEAR FAULT - GEAR UPLK REDUND LOSS	One uplock sensor of at least one gear has an electrical fault or unreasonable state.
32 GEAR FAULT - LGCV REDUND LOSS	One LGSCU has detected a loss of electrical redundancy in the LG control valve.
32 GEAR FAULT - LGCL REDUND LOSS	One LGSCU has detected a loss of electrical redundancy in the LG control lever.
32 GEAR FAULT - LGCL INOP	Both LGSCUs detect failures on LGCL switches or LGCL is in UP position while ALTERNATE EXTENSION switch is activated.
32 GEAR FAULT - FFSV FAIL OPEN	Loss of capability to hydraulically disconnect extend and retract valve.
32 GEAR FAULT - 28 V ESS REDUND LOSS	One LGSCU has detected a loss of electrical redundancy, 28 V ESS power is lost which results in all solenoids powered by this LGSCU being inoperative.
32 GEAR FAULT - 28 V NORM REDUND LOSS	One LGSCU has detected a loss of electrical redundancy, 28 V normal power is lost.
32 WOW FAULT - R GEAR WOFFW REDUND LOSS	One right MLG WOFFW sensor has an electrical fault or an unreasonable state.
32 WOW FAULT - L GEAR WOFFW REDUND LOSS	One left MLG WOFFW sensor has an electrical fault or an unreasonable state.
32 WOW FAULT - NOSE GEAR WOFFW REDUND LOSS	One NLG WOFFW sensor has an electrical fault or an unreasonable state.

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Table 8: INFO Messages

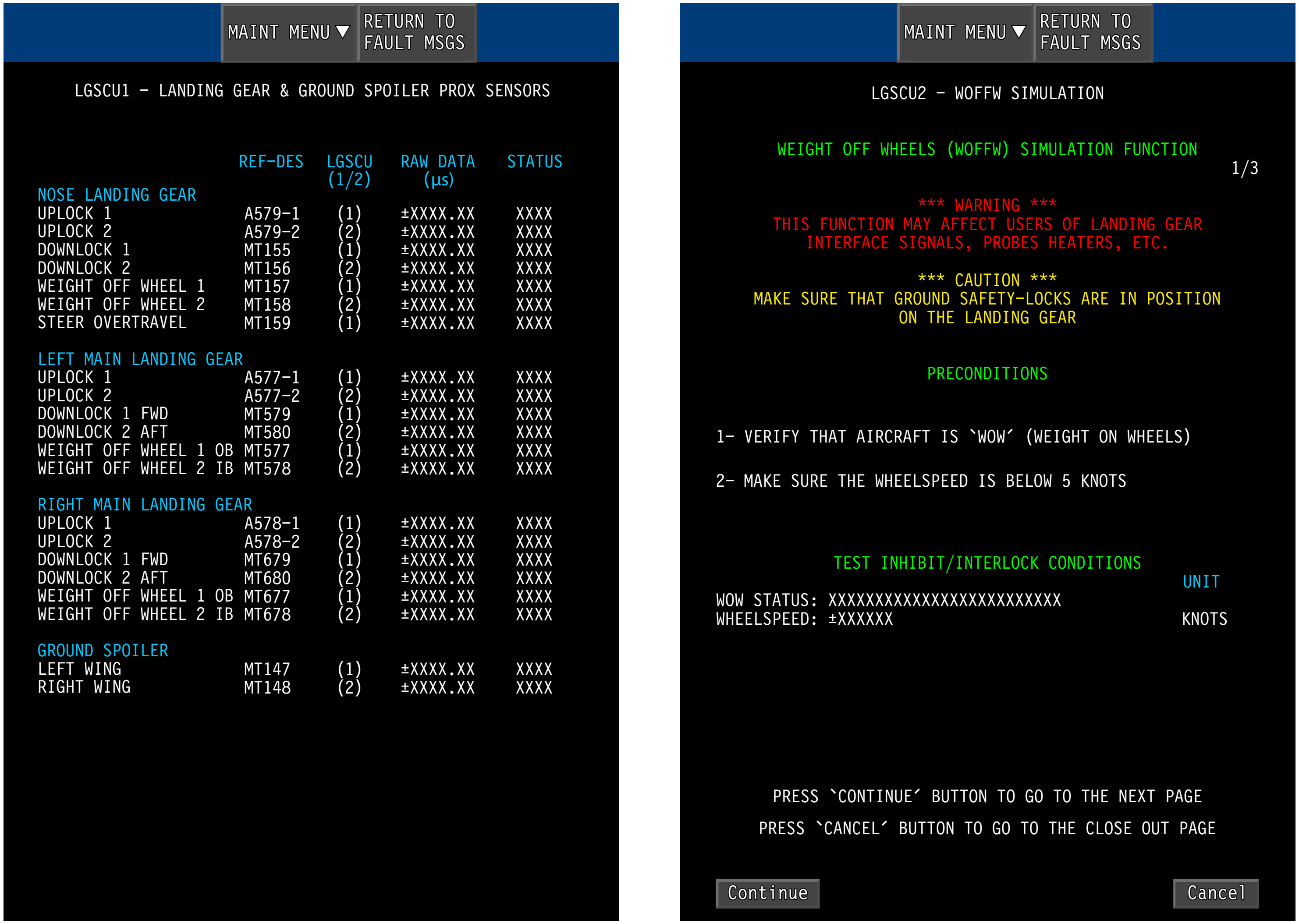
MESSAGE	LOGIC
32 GEAR FAULT - WOW OVERRIDE ACTIVE	OMS WOW override function is active.
32 GEAR FAULT - LGSCU INTNL FAULT	One LGSCU detected a critical internal fault and requires complete isolation of the affected LGSCU (ie.CBs pulled, LGSCU removal).
32 GEAR FAULT - LGSCL SWITCH FAULT	One LGSCU detected a critical LGCL fault and requires complete isolation of the affected LGCL (ie.LGSCU CBs pulled, LGSCU removal).
32 GEAR FAULT - ALTN EXT RLY 1, 2 CONTACT FAULT	One LGSCU detected a critical AES RELAY 1, 2 fault and requires complete isolation of the affected relay 1, 2 circuit (ie.affected circuit CBs pulled, relay removal).
32 GEAR FAULT - L, R GEAR UPLK MOTOR FAULT	One LGSCU detected L, R MLG uplock motor not in neutral position. Can cause inability to uplock L, R gear.
32 GEAR FAULT - NOSE GEAR UPLK MOTOR FAULT	One LGSCU detected NLG uplock motor not in neutral position. Can cause inability to uplock NLG.
32 GEAR FAULT - ALTN EXT INOP	Alternate extension dormancy test is reported as failed.

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM

The onboard maintenance system (OMS) monitors the proximity sensors of the landing gear indicating system (LGIS).

The OMS can also be used to simulate WOFFW and weight-on-wheels (WOW).



CS1_CS3_3260_005

Figure 37: Landing Gear Indicating System OMS Data

32-40 WHEELS AND BRAKES

GENERAL DESCRIPTION

Wheels and brakes includes the following:

- Landing gear wheels
- Tire pressure indicating system (TPIS)
- Brake control system (BCS)

LANDING GEAR WHEELS

MAIN LANDING GEAR WHEELS AND TIRES

Each main landing gear (MLG) wheel is equipped with a 42 x 15 R21 PR26 radial tire, mounted on two wheel halves. The wheel halves are bolted together. The wheel is fitted with two tapered roller bearings.

The wheel hubs on the CS300 are strengthened to support its greater aircraft weight. As a result the CS300 hub weighs approximately 62 kg (137 lb), whereas the CS100 hub weighs approximately 54.3 kg (120 lb).

Drive keys for engaging the brake rotors are mounted on the wheel inner hub surface. Heat shields, mounted on the inside of the wheel inboard hub, protect the wheel and tire from excessive brake temperatures.

The MLG wheel is secured on the axle with an axle nut. The axle nut is secured by two locking bolts.

Three thermal fuse plugs are installed 120° apart in the wheel hub, facing the brake heat stack to provide controlled pressure release on the brake to prevent a burst tire. The thermal fuse plugs melt at 199 ± 5.5°C (390 ± 10°F).

A tire pressure sensor holder and an inflation valve are mounted on the MLG wheel outboard hub face.

The MLG tire is fitted with nitrogen through the inflation valve. The MLG tire pressure for CS100 is 200 psi. The MLG tire pressure for CS300 is 224 psi. An overinflation protection plug releases the nitrogen gas in the tire between 365 psi to 435 psi.

The wheel hub cap, known as the tire pressure module (TPM), is mounted on the outboard hub face by three fasteners. The tire pressure sensor attaches to the tire pressure sensor holder. A metal tube interconnects the tire pressure sensor and TPM.

An axle thread protector must be used to prevent damage to the axle during wheel change. Ensure that the other landing gear wheels are chocked during wheel change. Release of the park brake during MLG wheel change is required in order to turn the wheels during installation.

NOSE LANDING GEAR WHEELS AND TIRES

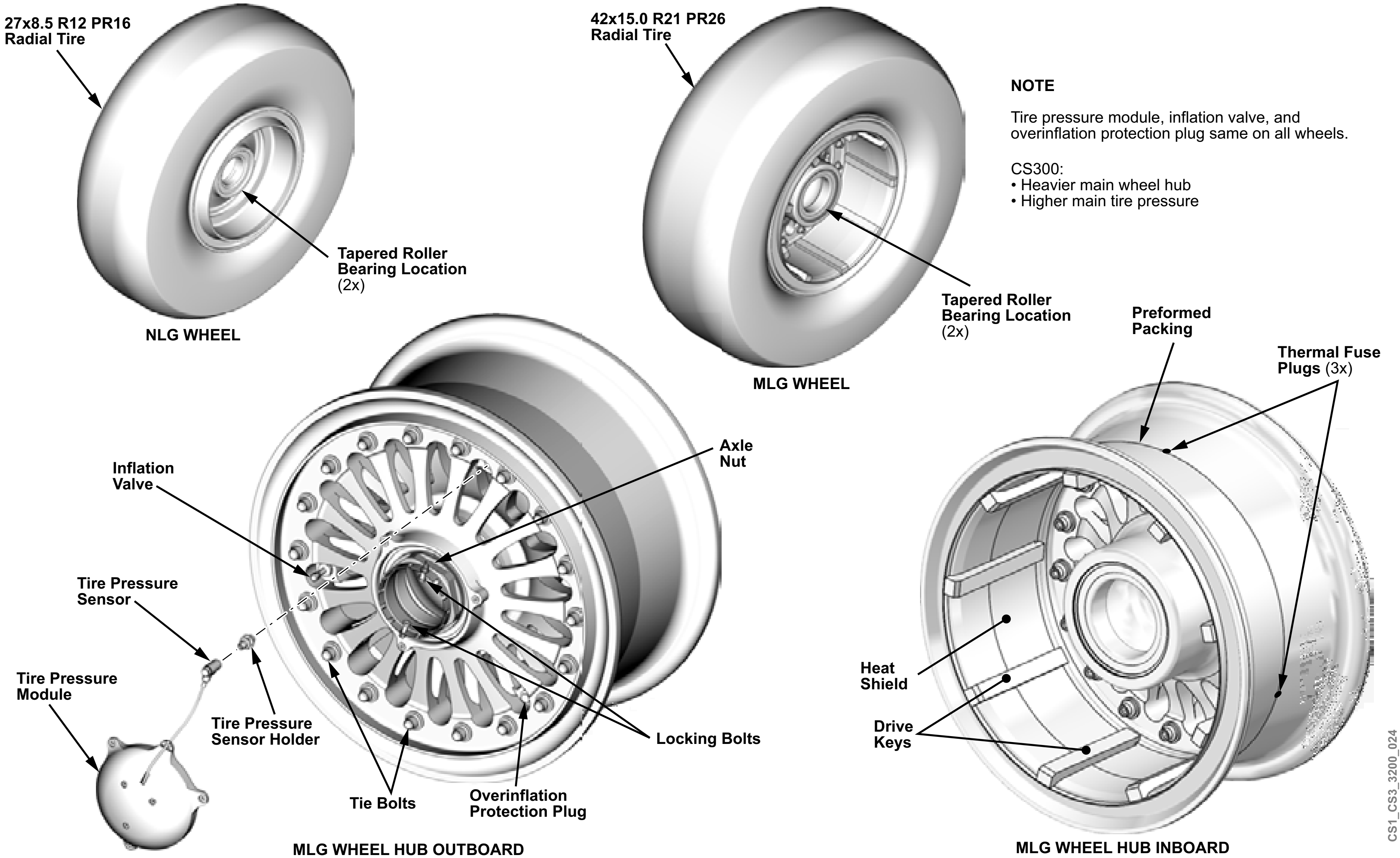
Each nose landing gear (NLG) wheel is equipped with a 27 x 8.5 R12 PR16 radial tire, mounted on two wheel halves. The wheel halves are bolted together.

The NLG wheel is secured on the NLG axle with an axle nut. The axle nut is secured by two locking bolts.

The wheel is fitted with two tapered roller bearings. The wheel outboard hub face contains an inflation valve, over inflation protection plug, and a tire pressure indicator sensor.

The NLG tire is filled with nitrogen through the inflation valve. Tire NLG pressure for CS100 and CS300 is 150 psi. The overinflation protection plug releases the nitrogen gas in the tire between 365 psi and 435 psi.

The tire pressure module (TPM) is mounted on the outboard hub face by three fasteners. A metal tube interconnects the tire pressure sensor and TPM.



NOTE

Tire pressure module, inflation valve, and overinflation protection plug same on all wheels.

CS300:
- Heavier main wheel hub
- Higher main tire pressure

Figure 38: Landing Gear Wheels and Tires

TIRE PRESSURE INDICATION SYSTEM

GENERAL DESCRIPTION

Each main wheel and nosewheel is equipped with a tire pressure module (TPM), and an axle interface module (AIM). The tire pressure in each wheel is monitored by a tire pressure sensor which is part of the TPM. The signal is then sent wirelessly to the AIM. The AIM is installed in the wheel axle, and is wired to a tire pressure monitoring unit (TPMU).

The tire pressure monitoring unit (TPMU) is the main processing unit of the tire pressure indication system, and powers the AIM. It is powered from DC BUS 1.

The TPMU processes information received from each AIM, and sends the signals to the DMCs for tire pressure indication on the STATUS synoptic page.

The tire pressure module (TPM):

- Senses tire pressure through a tire pressure sensor mounted on the outboard wheel hub tire pressure sensor holder
- Provides wireless signal to the AIM

Each nose landing gear wheel is provided with an axle interface module (AIM). The AIM is the communication interface between the TPM and the tire pressure monitoring unit (TPMU). The AIM is powered by the TPMU and energizes the TPM through a magnetic link.

Each main wheel AIM is also part of a wheel speed transducer (WST). The WST measures the respective main wheel rotational speed for the brake antiskid system. The rotational speed is achieved by coupling the wheel speed transducer shaft in the AIM to the rotating main wheel TPM.

The TPMU has field-loadable software.

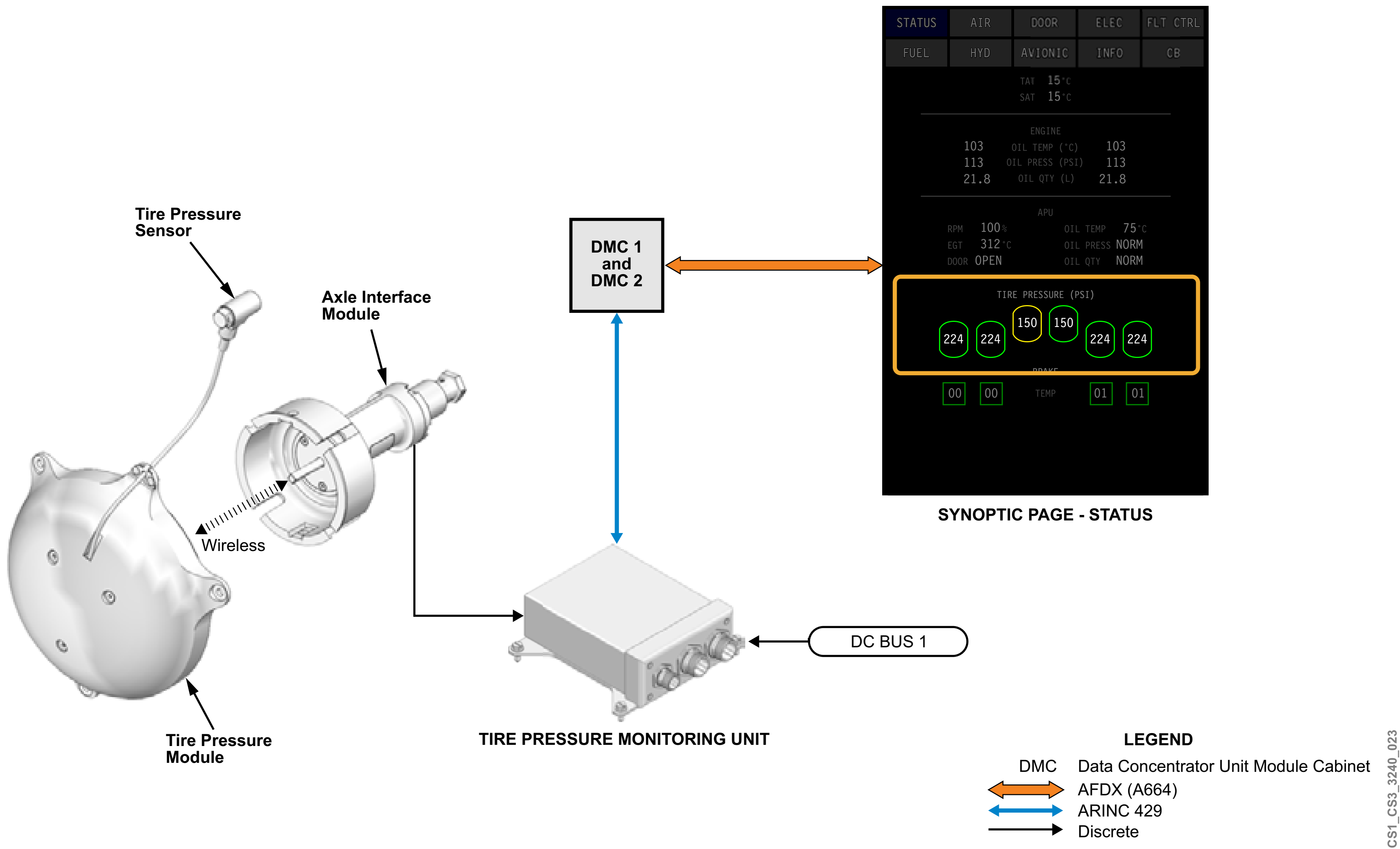


Figure 39: Tire Pressure Indicating System

COMPONENT LOCATION

Tire Pressure Monitoring Unit

The tire pressure monitoring unit (TPMU) is located in the forward equipment bay.

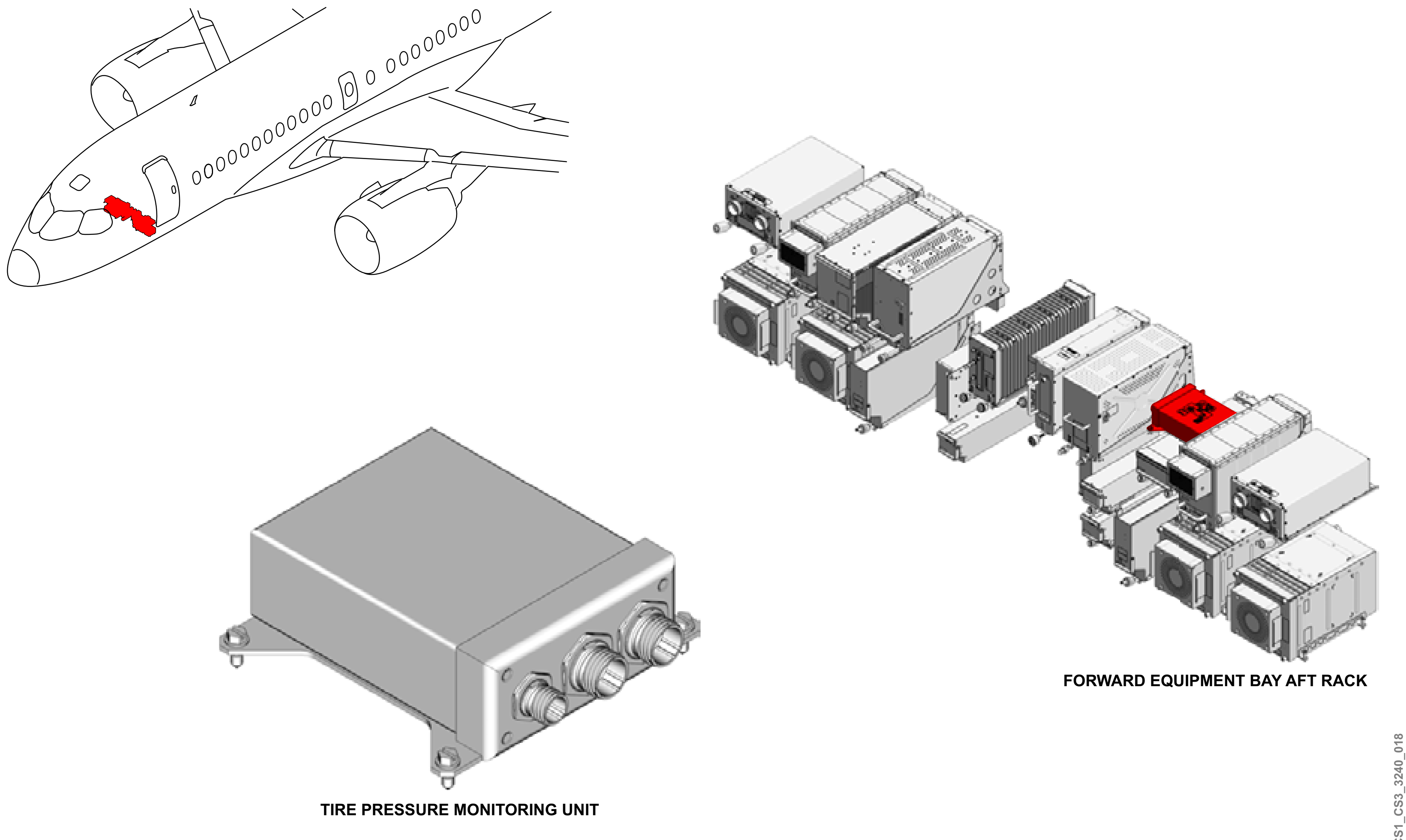


Figure 40: Tire Pressure Monitoring Unit

Tire Pressure Module

A tire pressure module is an integral part of each wheel hub cap.

Axle Interface Module

An axle interface module (AIM) is located in the axle of the nose wheel.

Axle Interface Module/Wheel Speed Transducer

An axle interface module (AIM)/wheel speed transducer (WST) is located in the axle of each main wheel.

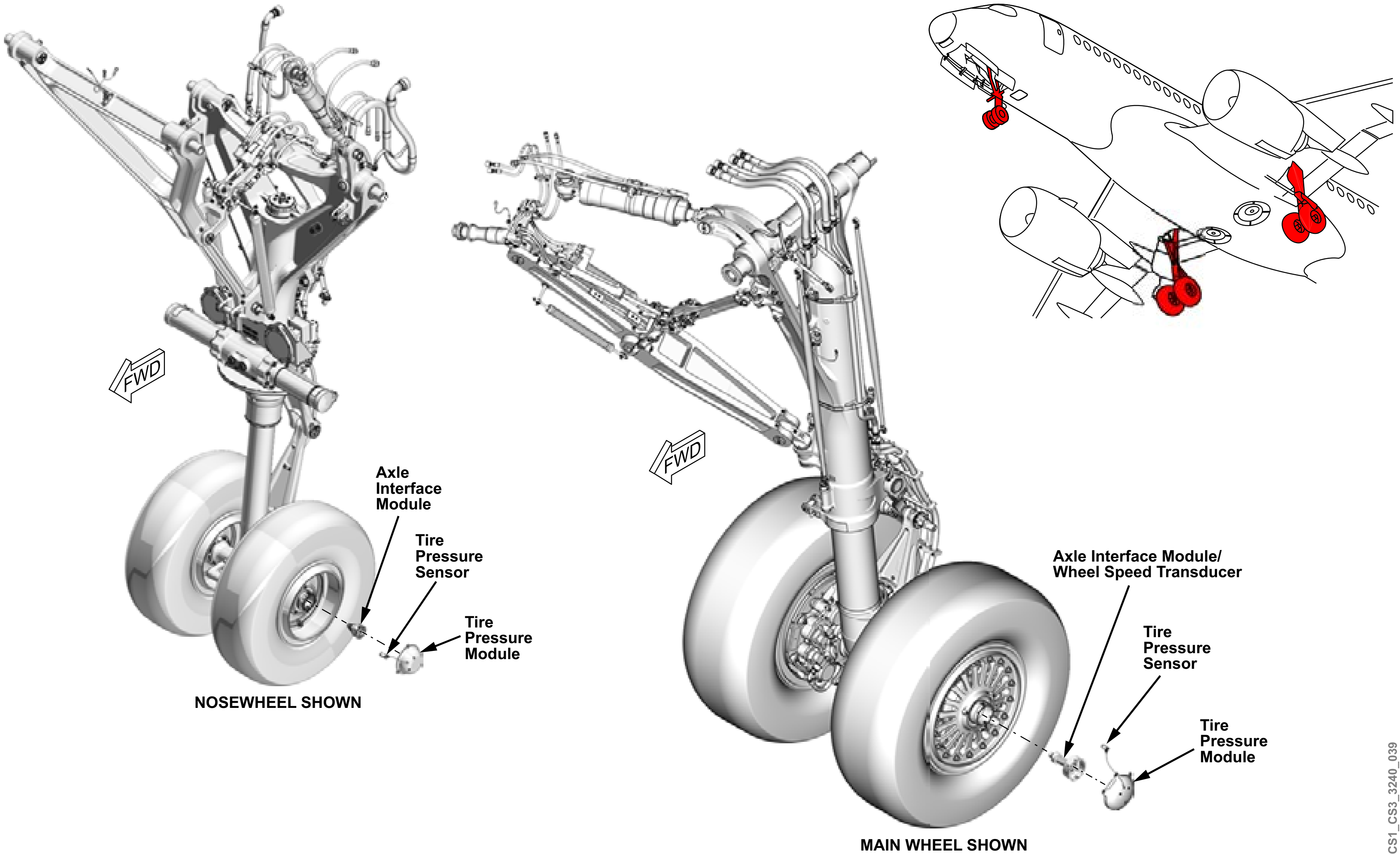
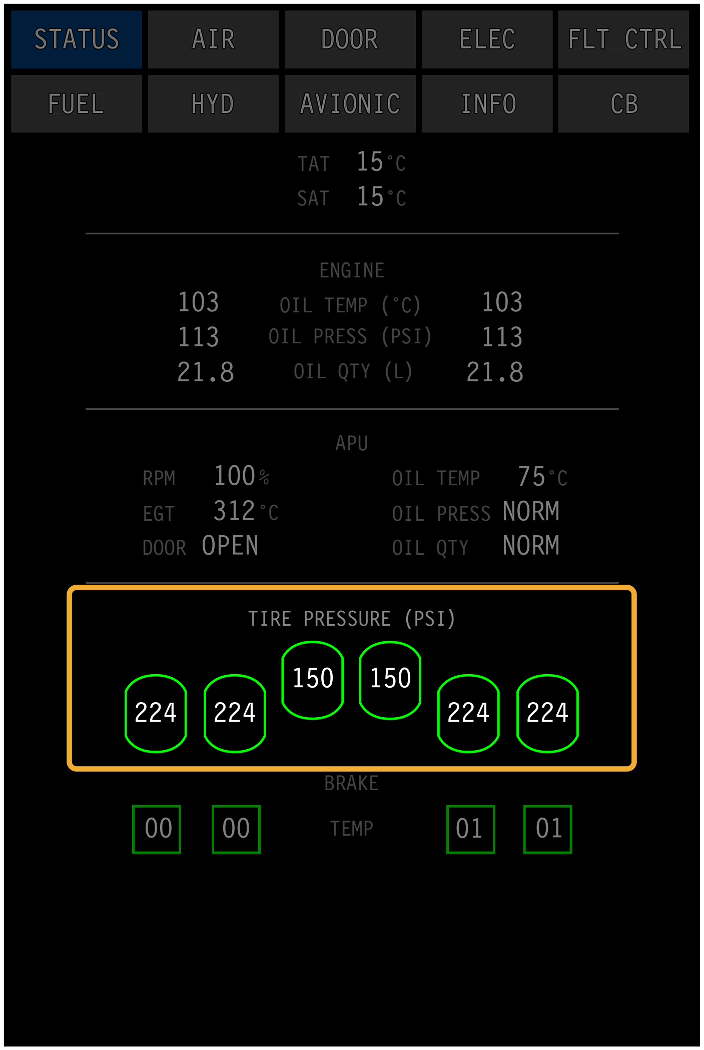


Figure 41: Tire Pressure Sensor, Tire Pressure Module, and Axle Interface Module

CONTROLS AND INDICATIONS

The tire pressure indication is displayed on the STATUS synoptic page. The pressure in each wheel is graphically displayed in color and psi.



SYNOPTIC PAGE - STATUS

TIRE PRESSURE INDICATION		
SYMBOL	**DESCRIPTION ON GROUND**	**DESCRIPTION IN FLIGHT**
155	Pressure ≥ nominal	Pressure ≥ 90% nominal
140	90% nominal ≤ Pressure < nominal	70% nominal ≤ Pressure < 90% nominal
129	Pressure < 90% nominal	Pressure < 70% nominal
---	Pressure invalid	Pressure invalid

NOTE

CS300 shown.

CS1_CS3_3200_029

Figure 42: Tire Pressure Indication

DETAILED DESCRIPTION

In flight only, when the tire pressure is too low in either nosewheel tire, a NOSE TIRE LO PRESS caution message is displayed on the EICAS. The respective nosewheel tire pressure indication is also displayed in amber on the STATUS synoptic page.

A TIRE LO PRESS advisory message is displayed on the EICAS on the ground if any tire pressure is less than 90% of normal pressure, and in flight when any main tire pressure is less than 70% of normal pressure.

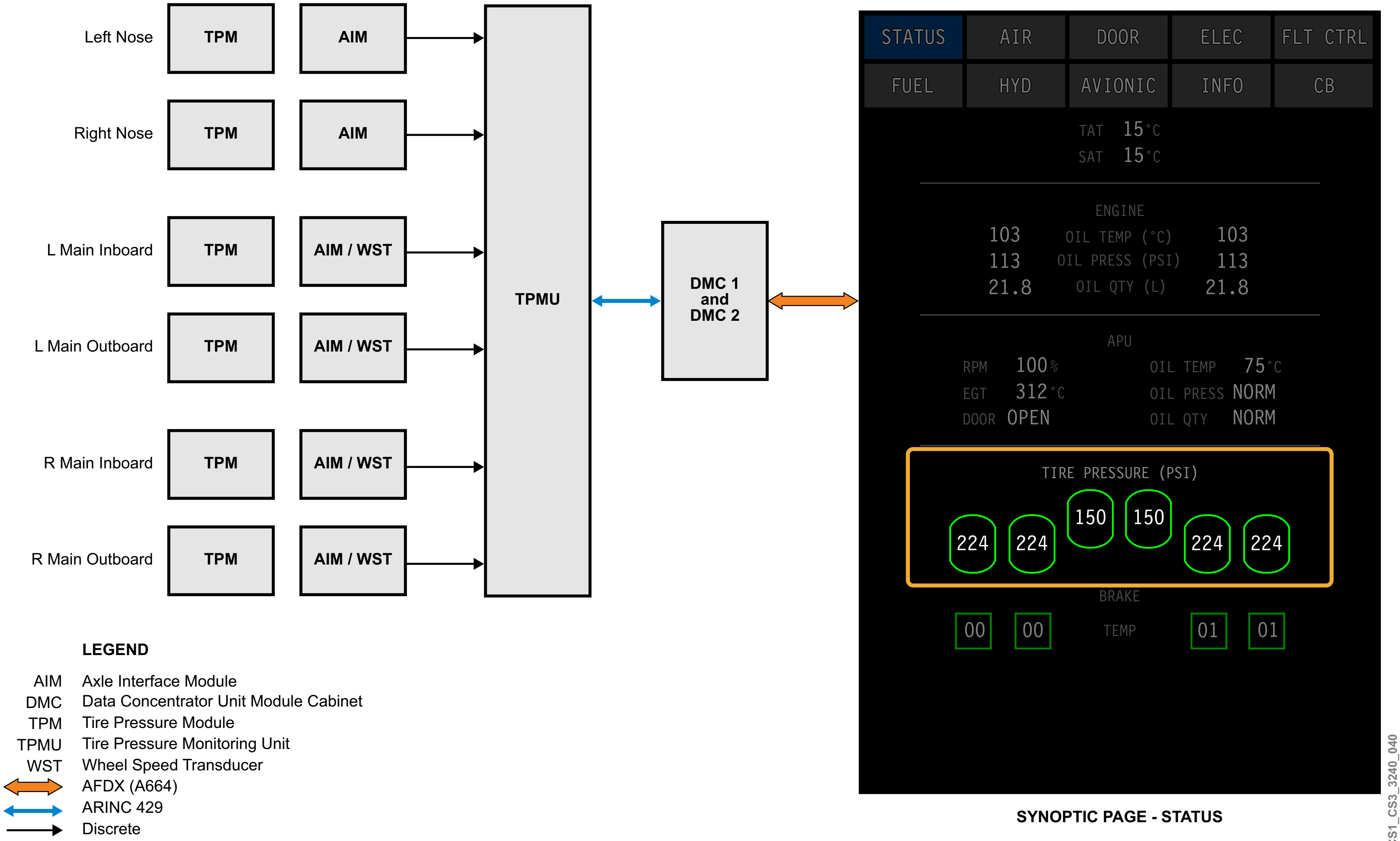


Figure 43: Tire Pressure Indication

BRAKE CONTROL SYSTEM

GENERAL DESCRIPTION

The brake control system (BCS) features electric brakes that provide deceleration capability in normal, alternate, autobrake, gear retraction braking, and park brake modes.

Each brake consists of a brake stack that is clamped by four electric motor actuators (EMAs) to provide braking force. The EMAs are commanded in pairs by electric motor actuator controllers (EMACs). Two EMACs as located within each of the four electric motor control units (EMCUs).

Two brake data concentrator units (BDCUs) operate in an active/active configuration, and provide braking commands to the EMCUs. Communication between the EMCUs is through ARINC 429, CAN BUSES and discrete signals.

Braking inputs are provided from the brake pedals, the AUTOBRAKE rotary switch, the ALTN BRAKE PBA, and the PARK BRAKE rotary switch. The parking brakes can also be applied from the PARK BRK switch on the electrical/towing service panel.

Wheel speed information is provided to the EMCUs by a wheel speed transducer, located with the axle of each main wheel.

Thrust lever position is provided to the BDCUs.

The BDCUs provide data to the EICAS and onboard maintenance system (OMS), through the data concentrator unit module cabinets (DMCs). The BDCUs also provide brake system information to the primary flight control computers (PFCCs).

The BDCUs provide the signal for the TOW/NO TOW lights on the electrical/towing service panel and the TOWING/NO TOWING lights on the TOWING CONTROL BOX.

The BDCUs and EMCUs are powered from DC ESS BUSES 1 and 2.

Normal Braking Mode

Normal braking mode provides a proportional individual aircraft brake clamping force based on brake pedal position. This mode includes software-based braking protection, which includes antiskid control, locked wheel protection, and touchdown protection. A derotation braking function is activated after main wheel touchdown with the nose gear in the air, limiting the braking force. This improves stopping distance performance, while preventing abrupt nosewheel impact.

Alternate Braking Mode

Selecting the ALTN BRAKE PBA provides proportional individual aircraft brake clamping force based on brake pedal position, but has no braking protection.

Autobrake Mode

The autobrake mode is commanded by the AUTOBRAKE rotary switch, without using the brake pedals on landing or rejected take off (RTO). The switch has five selections (RTO, OFF, LO, MED and HI). The software commands a proportional level of brake clamping force based on the selection. The clamping force is limited at the brake by braking protection.

Gear Retraction Braking Mode

Gear retraction braking is activated at landing gear retraction to stop the main wheel rotation after takeoff, prior to stowing in the landing gear bay.

Park Brake Mode

The park brake mode provides brake clamping forced, either from the flight deck switch, or an external PARK BRK switch. With the park brake selected, a mechanical linkage in each EMA gear train maintains clamping force when electrical power is removed. Electrical power is required to release the park brake.

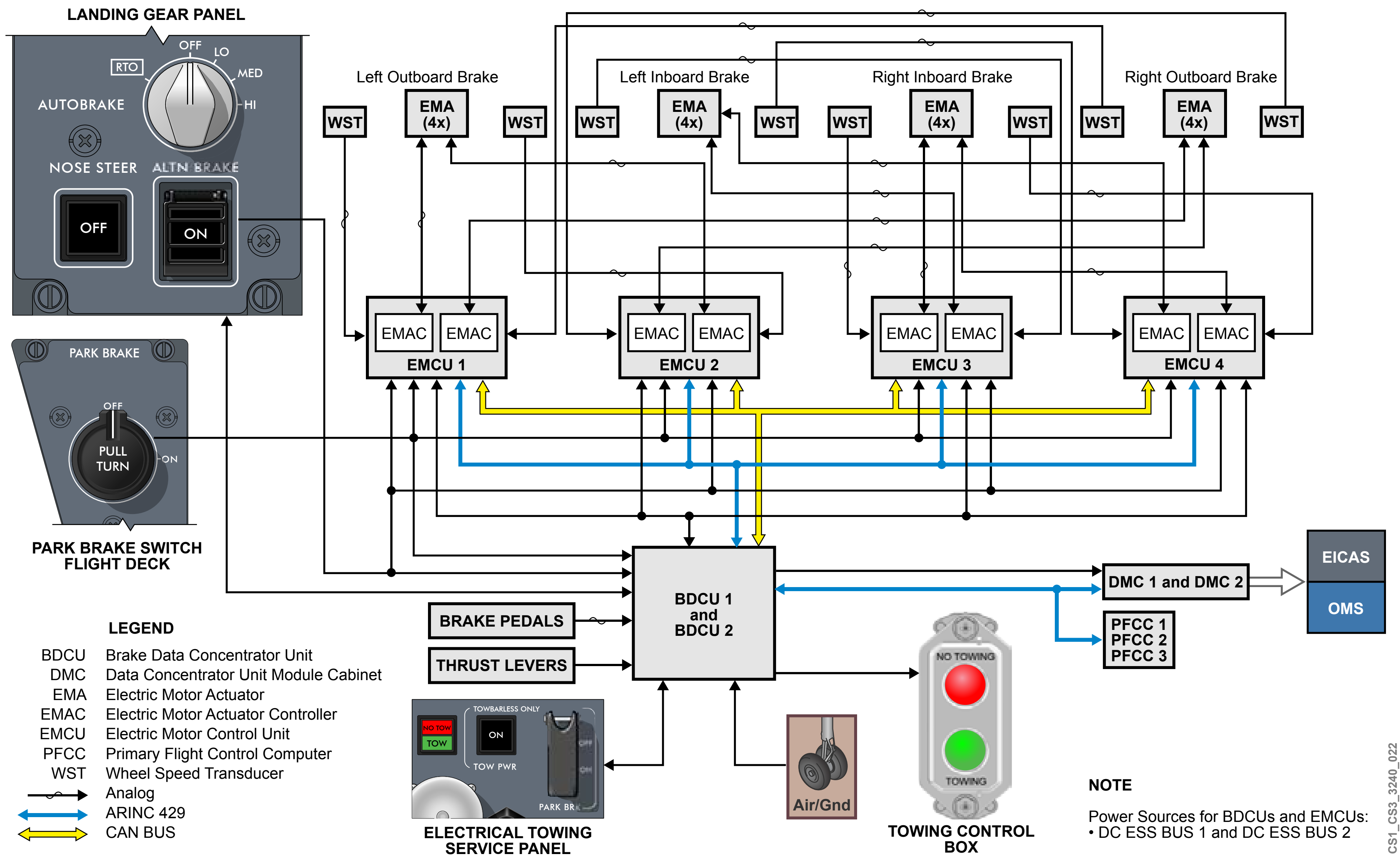


Figure 44: Brake Control System

COMPONENT LOCATION

Pedal Position Transducer

A left and a right pedal position transducer (PPT) is located on each rudder pedal assembly.

Brake

A brake is located on either side of each main landing gear wheel axle.

Electric Motor Actuator

Four electric motor actuators (EMAs) are mounted on each brake.

Wheel Speed Transducer

A wheel speed transducer (WST) is located in each main wheel axle.

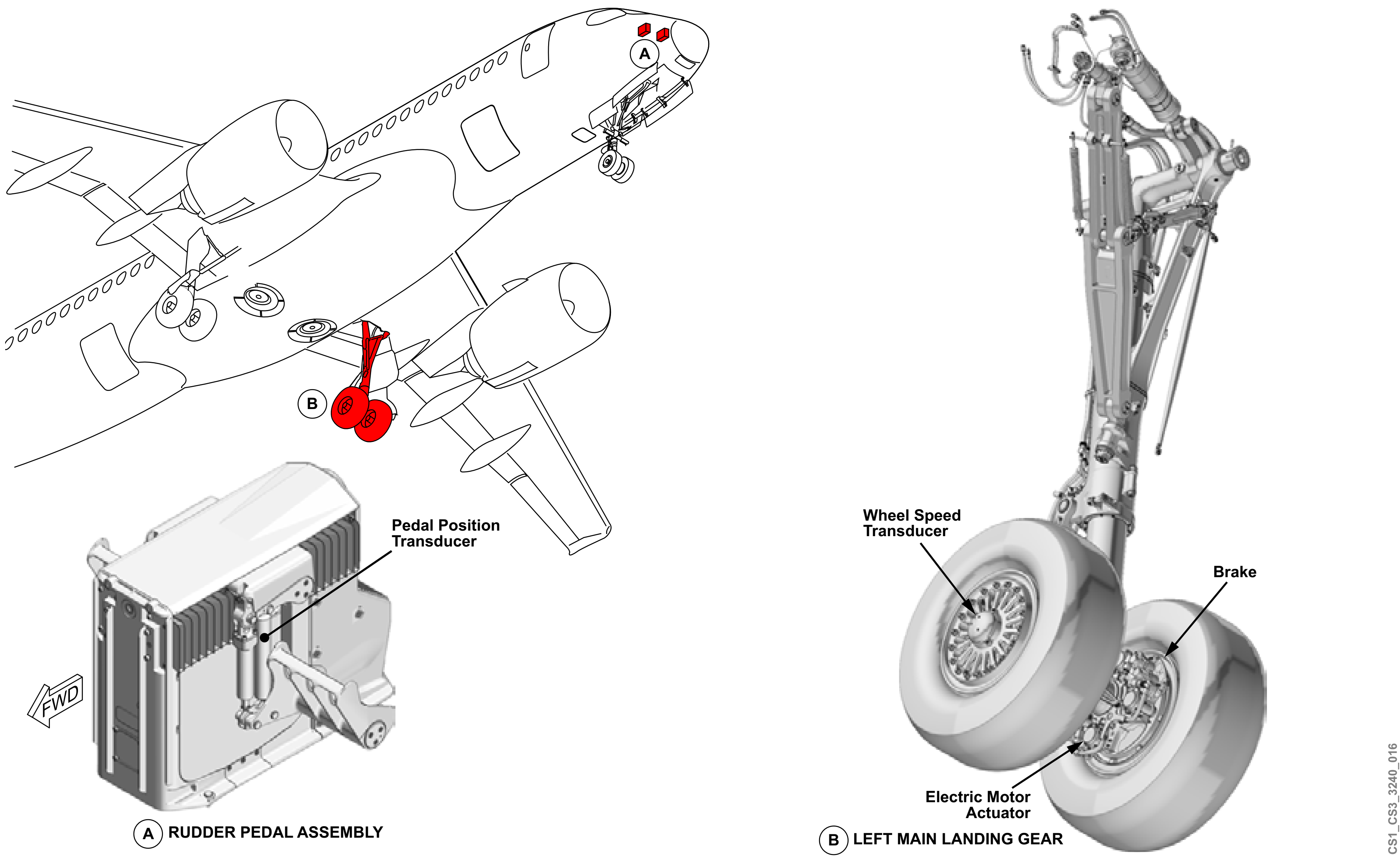


Figure 45: Brake Control System Component Location

Brake Data Concentrator Unit

Two brake data concentrator units (BDCUs) are located in the mid equipment bay.

Electric Motor Control Unit

Four electric motor control units (EMCUs) are located in the mid equipment bay.

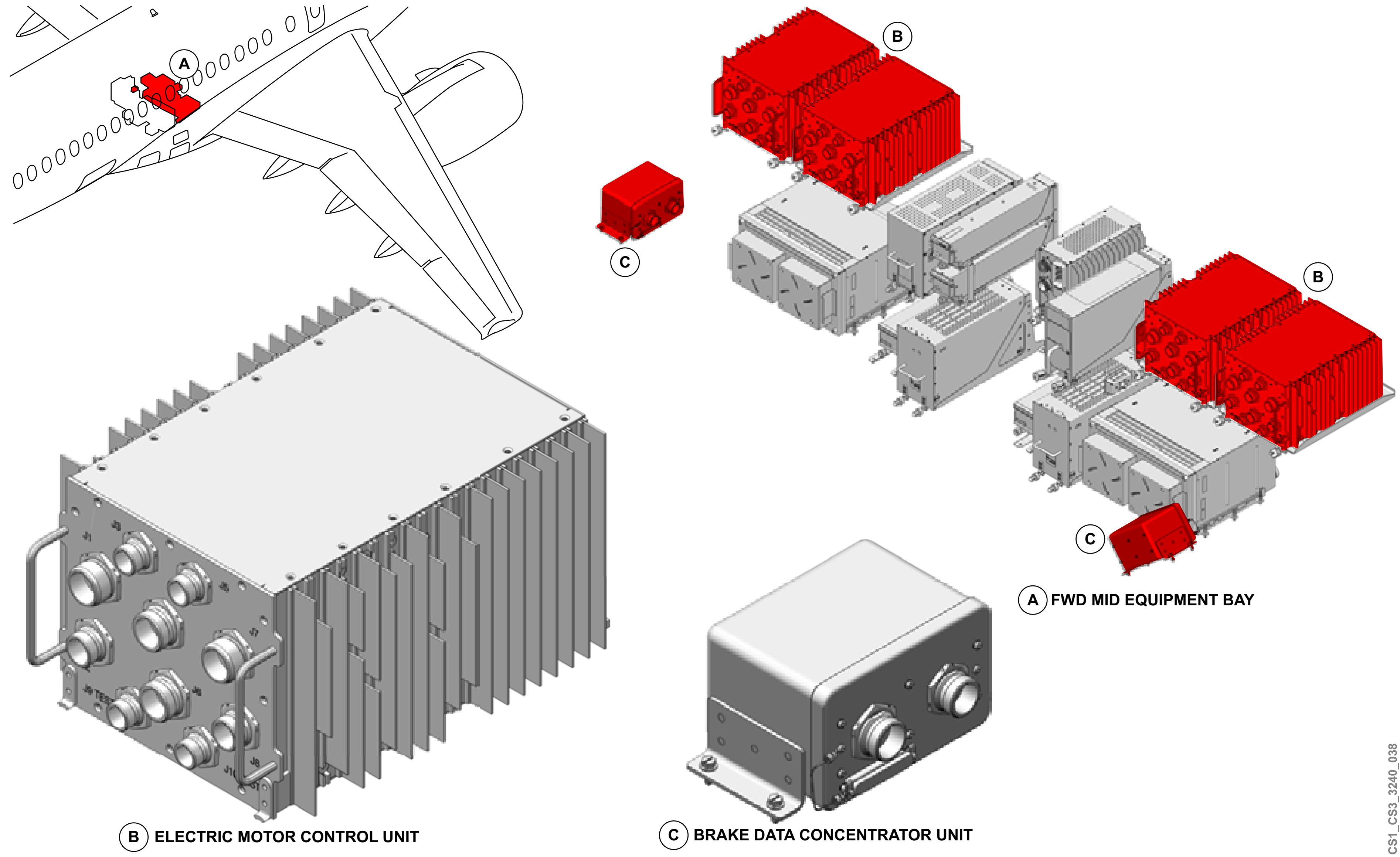


Figure 46: Brake Control System Component Location

COMPONENT INFORMATION

Brake

The principal components of the brake include the following:

- Carrier plate, torque tube, and back plate
- Brake stack consisting of the following:
  - Pressure plate
  - End plate
  - Stators
  - Rotors
- Electric motor actuators (EMAs)
- Torque takeout
- Brake wear pin
- Brake temperature sensor

The carrier plate with torque tube and back plate form the principal structure of the brake. Four EMAs are attached to the carrier plate and deliver brake stack clamping pressure.

The EMAs compress the brake stack by applying pressure to the rotors clamped between the pressure plate, stators, and end plate. This causes friction on the rotors and brake the aircraft. The brake rotor drive channels interface with the main wheel hub drive keys.

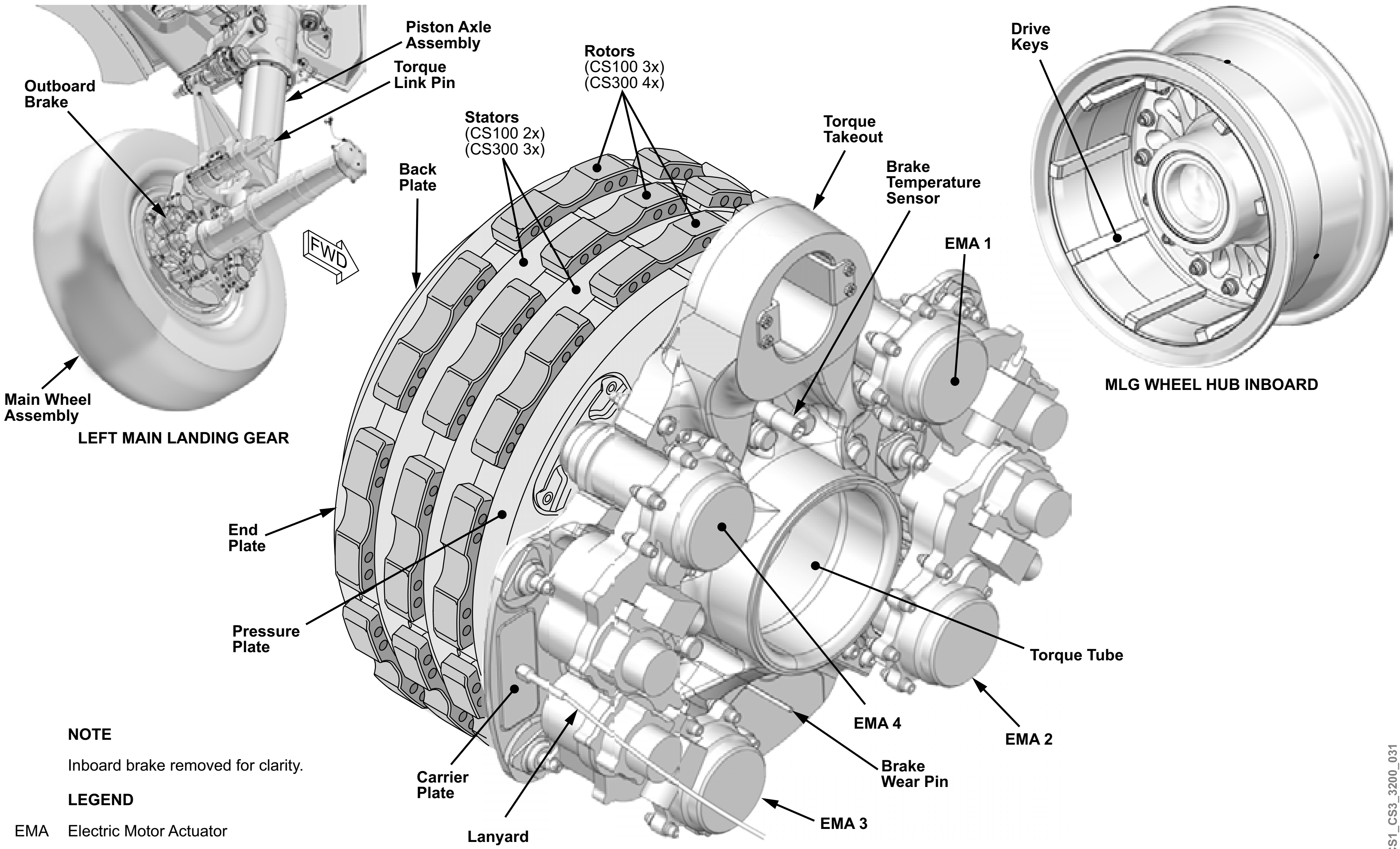
The CS100 carbon brake stack has three rotors and two stators. The CS300 carbon brake stack has four rotors and three stators.

The brake is mounted on the main wheel axle. The torque takeout is aligned with the torque link pin on the piston axle assembly, and is secured in place by the main wheel. The torque takeout transfers braking forces to the landing gear structure.

A brake wear pin protrudes through the carrier plate. The park brake must be applied when measuring brake wear.

The brake also includes a brake temperature sensor.

A lanyard connected between the inboard and outboard brakes hold the brake in position during wheel removal.



NOTE

Inboard brake removed for clarity.

LEGEND

EMA Electric Motor Actuator

Figure 47: Brake

Axle Interface Module/Wheel Speed Transducer

Each main landing gear axle contains an axle interface module (AIM)/wheel speed transducer (WST). Each coil of the dual-coil WST provides wheel speed data independently to an electric motor actuator controller (EMAC).

The WST derives its energy from an internal permanent magnet. A shaft in the wheel speed transducer interfaces with a coupler in the main wheel hub cap for wheel rotational motion input.

Pedal Position Transducers

There are four pedal position transducers (PPT). Two PPTs are installed at each rudder pedal mechanism. One end of the PPT is attached to the brake pedal lever, and the opposite end to the rudder pedal structure.

The PPT is a dual-channel linear variable differential transformer (LVDT), that interfaces directly with the two BDCUs. The PPT has two redundant return springs that return the brake pedal to the 0 pedal position to prevent uncommanded braking.

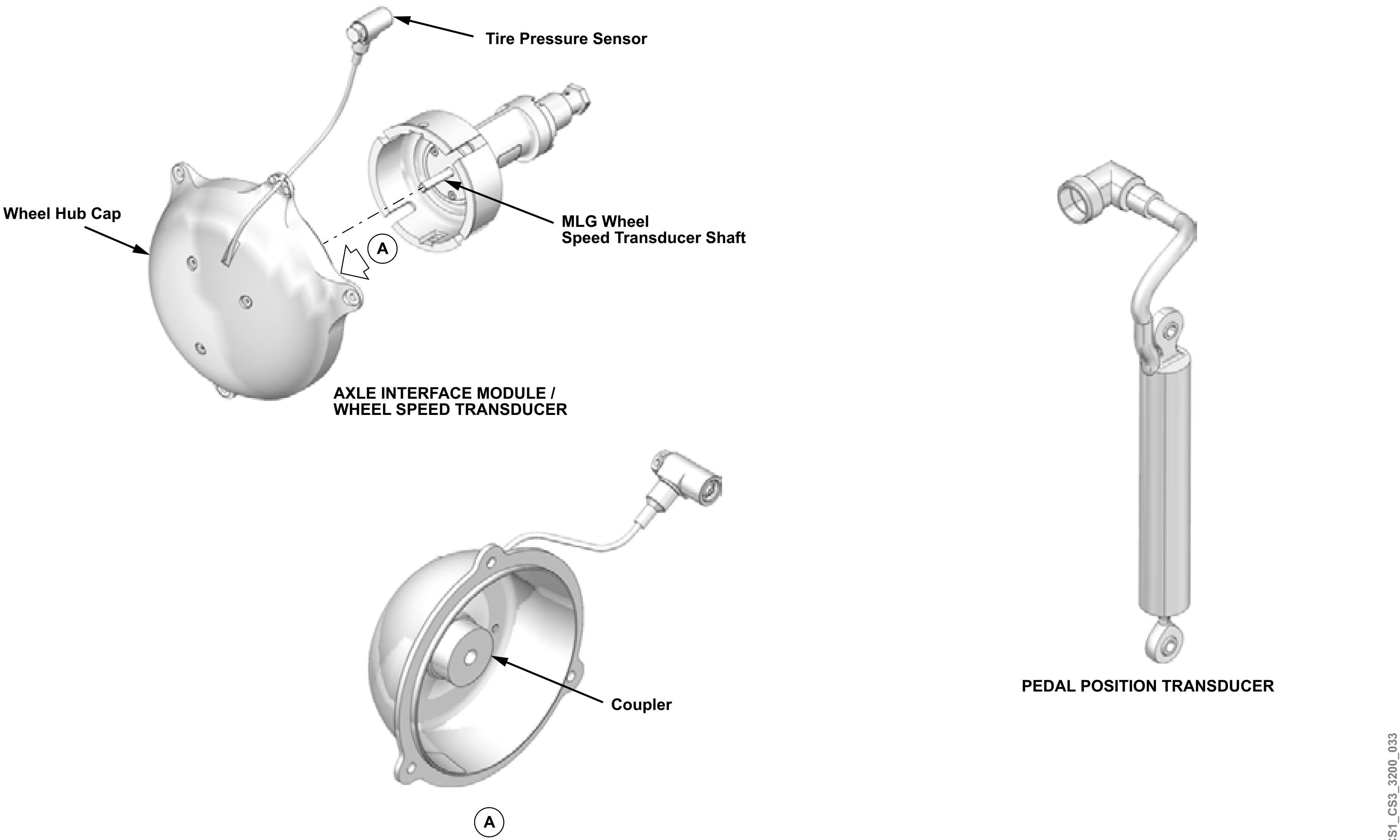


Figure 48: Wheel Speed Transducer and Brake Pedal Transducer

DETAILED COMPONENT INFORMATION

Electric Motor Actuator

There are four electric motor actuators (EMAs), also called electromechanical actuators, mounted on each brake. Each EMA is capable of providing a force up to 5216.3 kg (11,500 lb). Clamping force can be viewed on the OMS.

The EMA has a right and left unit. The electrical connectors of the EMAs are pinned differently in order to prevent cross-connection.

Each EMA contains a 120 VDC, 3-phase brushless motor. The motor engages a gear mechanism that drives a ballscrew nut to provide the brake clamping force.

Located within the EMA is a hall effect sensor for motor position monitoring and a force sensor. A resistive thermal device (RTD) is also located with the EMA, to monitor motor winding temperature.

When PARK BRAKE is selected, the park brake module is commanded to apply a force of 1360.8 kg +/-227.3 kg (3,000 +/-500 lb) for 1.5 seconds. An internal mechanical lock engages after electrical power is removed to maintain the clamping force. The mechanical lock is disengaged when electrical power is available and the park brake switch has been selected to OFF.

The EMAs are numbered 1 to 4 in a clockwise manner. The EMAs are controlled in diagonal pairs (1 and 3, 2 and 4) to ensure that no single failure results in the loss of more than one pair of EMAs.

CAUTION

To prevent EMA overheat, do not apply the brake continuously for more than 1 minute when the brake pedal is commanded more than 90%.

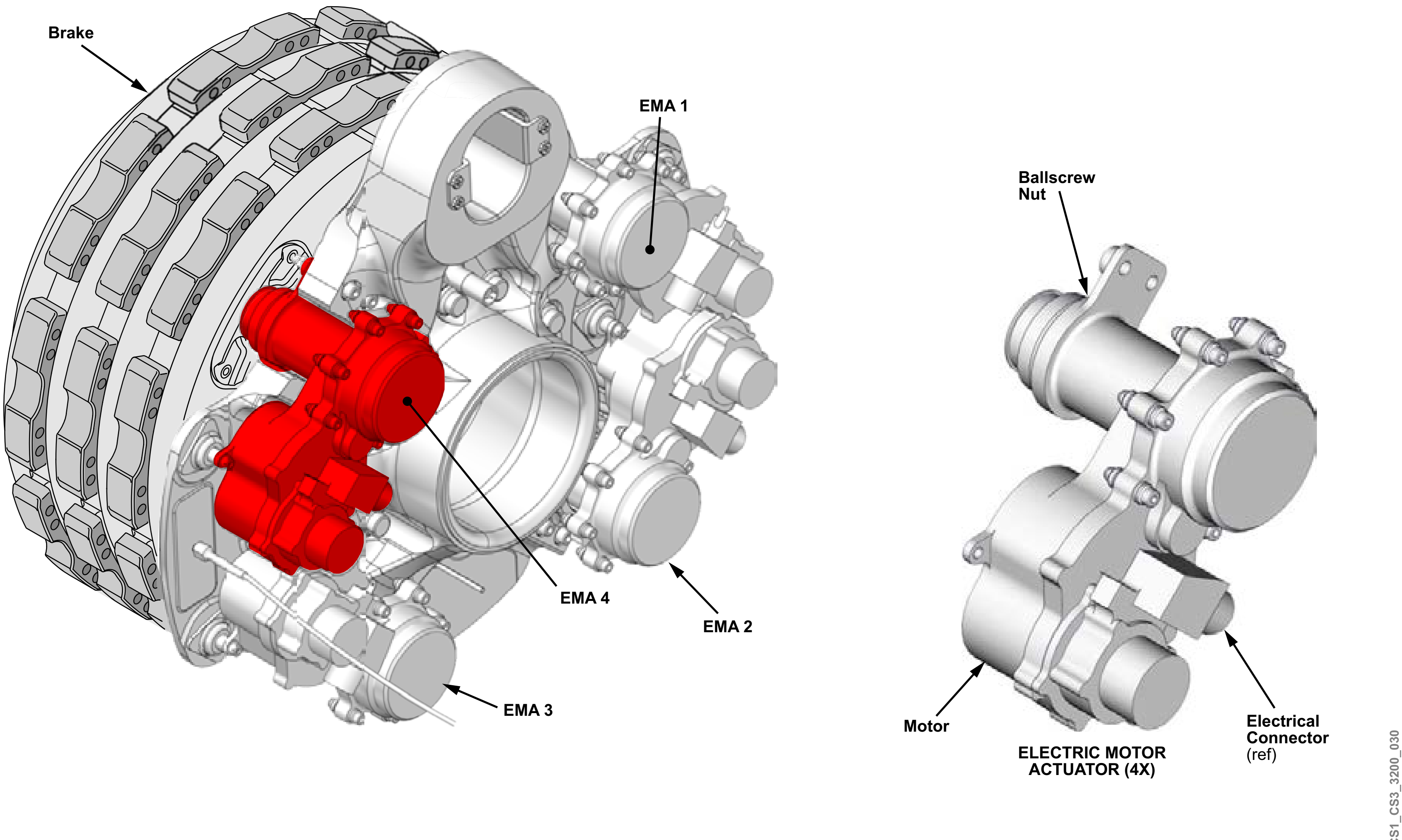


Figure 49: Electric Motor Actuator

Electric Motor Actuator Arrangement

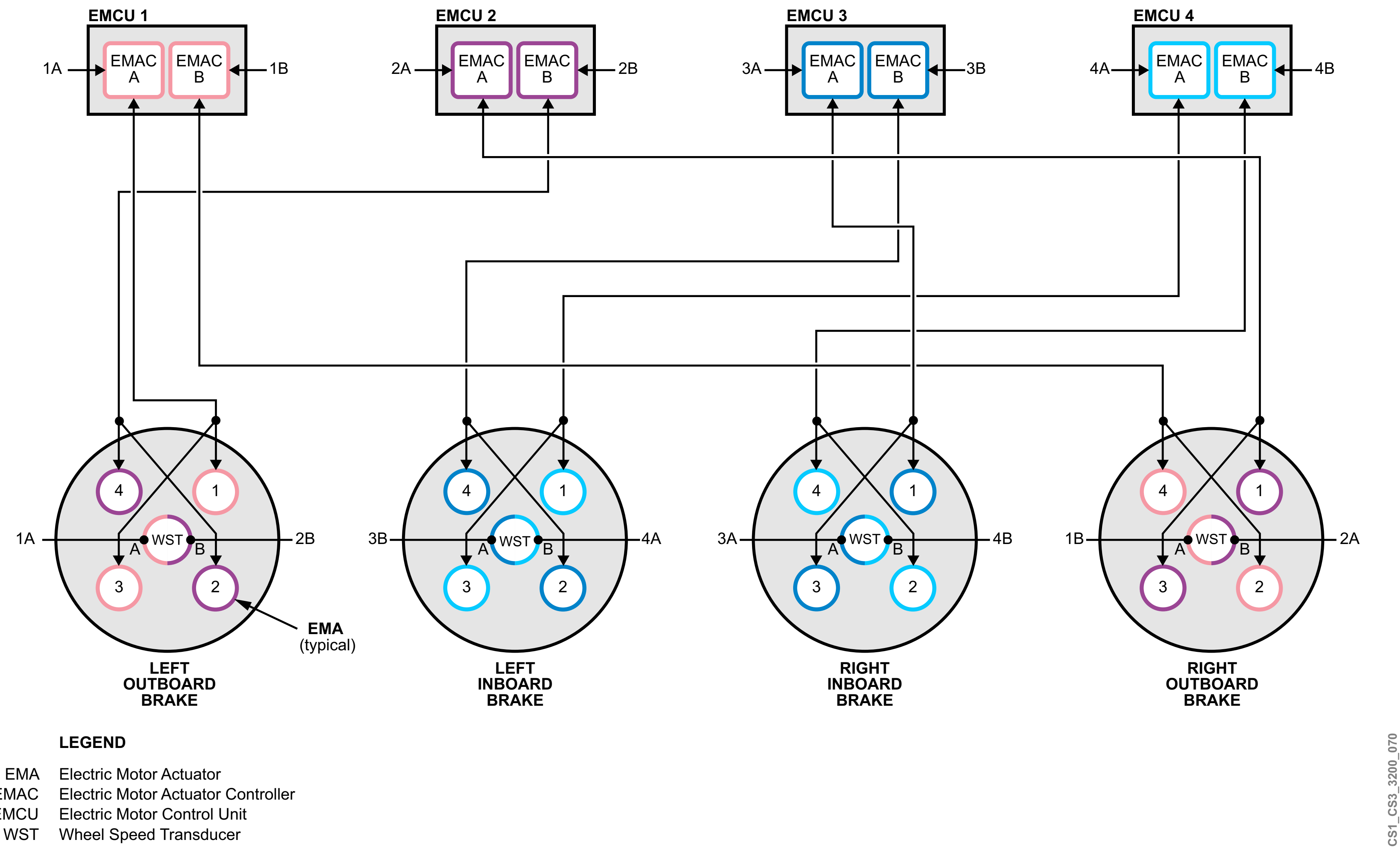
Each electric motor control units (EMCUs), provide braking control to opposite brakes. The arrangement is as follows:

- EMCU 1 - Left outboard and right outboard brakes
- EMCU 2 - Left outboard and right outboard brakes
- EMCU 3 - Left inboard and right inboard brakes
- EMCU 4 - Left inboard and right inboard brakes

This arrangement ensures that the loss of a single EMCU does not result in the loss of more than 25% braking capacity. In addition, the loss of two EMCUs does not result in the loss of more than 50% braking capacity.

Each electric motor actuator controller (EMAC) controls two electric motor actuators (EMAs). The EMAs are mounted opposite to each other on the brake. This arrangement ensures that the loss of single EMAC does not result in the loss of more than 50% braking capacity on the affected brake.

Each coil of the dual-coil wheel speed transducer (WST) is connected to the associated EMAC to provide braking protection.



LEGEND

EMA	Electric Motor Actuator
EMAC	Electric Motor Actuator Controller
EMCU	Electric Motor Control Unit
WST	Wheel Speed Transducer

Figure 50: Electric Motor Actuator Arrangement

Brake Data Concentrator Unit

The brake data concentrator unit (BDCU) consists of a normal mode and an alternate mode. Each section provides the braking commands to the EMCUs.

The BDCU normal mode section receives inputs from the following:

- PARK BRAKE switch, in the flight deck and on the external/towing service panel
- AUTOBRAKE switch
- Landing gear control lever
- Thrust levers
- Brake pedals

The normal mode has a software-based function for determining brake level and providing protection, and generates a brake applied signal based on brake pedal inputs. The BDCUs provide the normal mode braking commands, via ARINC 429, to each EMAC within the EMCUs.

The BDCU alternate mode section receives inputs from the following:

- Brake pedals
- ALTN BRAKE PBA

The alternate mode monitors brake pedal level for braking control. No protection is available for alternate braking. Alternate mode breaking commands are sent as analog signals to the EMCUs.

The BDCUs receive discrete information from the EMCUs, to indicate if a braking force greater than 90.7 kg (200 lb) is inadvertently applied by an EMA.

CAN BUS communication takes place between the EMCUs and the BDCUs. The information transmitted over the CAN BUS is primarily wheel speed, which is routed to other aircraft systems via the data concentrator unit module cabinets (DMCs). Wheel speed is also used by the other EMACs for locked wheel protection. Other information carried over the CAN BUS includes brake force, used by the flight data recorder, and failure reporting information from the EMACs, EMAs, and wheel speed transducers (WSTs).

The BDCUs communicate information for EICAS and the OMS, which is provided from the BDCUs to the DMCs via ARINC 429. The primary flight control computers (PFCCs) receive EICAS data and wheel speed information for spoiler deployment via ARINC 429.

The alternate mode provides discrete signals to the DMCs for park brake on, park brake fail, and alternate brake on. This provides backup to flight deck indication, in case normal communication between the BDCUs and DMCs is inoperative.

The BDCUs provide discrete signals to control the autobrake solenoid in the AUTOBRAKE rotary switch and to operate the tow lights.

Power supply for the normal mode section is as follows:

- DC ESS BUS 1 for BDCU 1
- DC ESS BUS 2 for BDCU 2
- When the TOW PWR switch on the electrical/towing service panel is selected ON, BDCU 1 and BDCU 2 receive power from the DC EMER BUS.

Power supply for the alternate mode section is as follows:

- DC ESS BUS 2 for BDCU 1
- DC ESS BUS 1 for BDCU 2

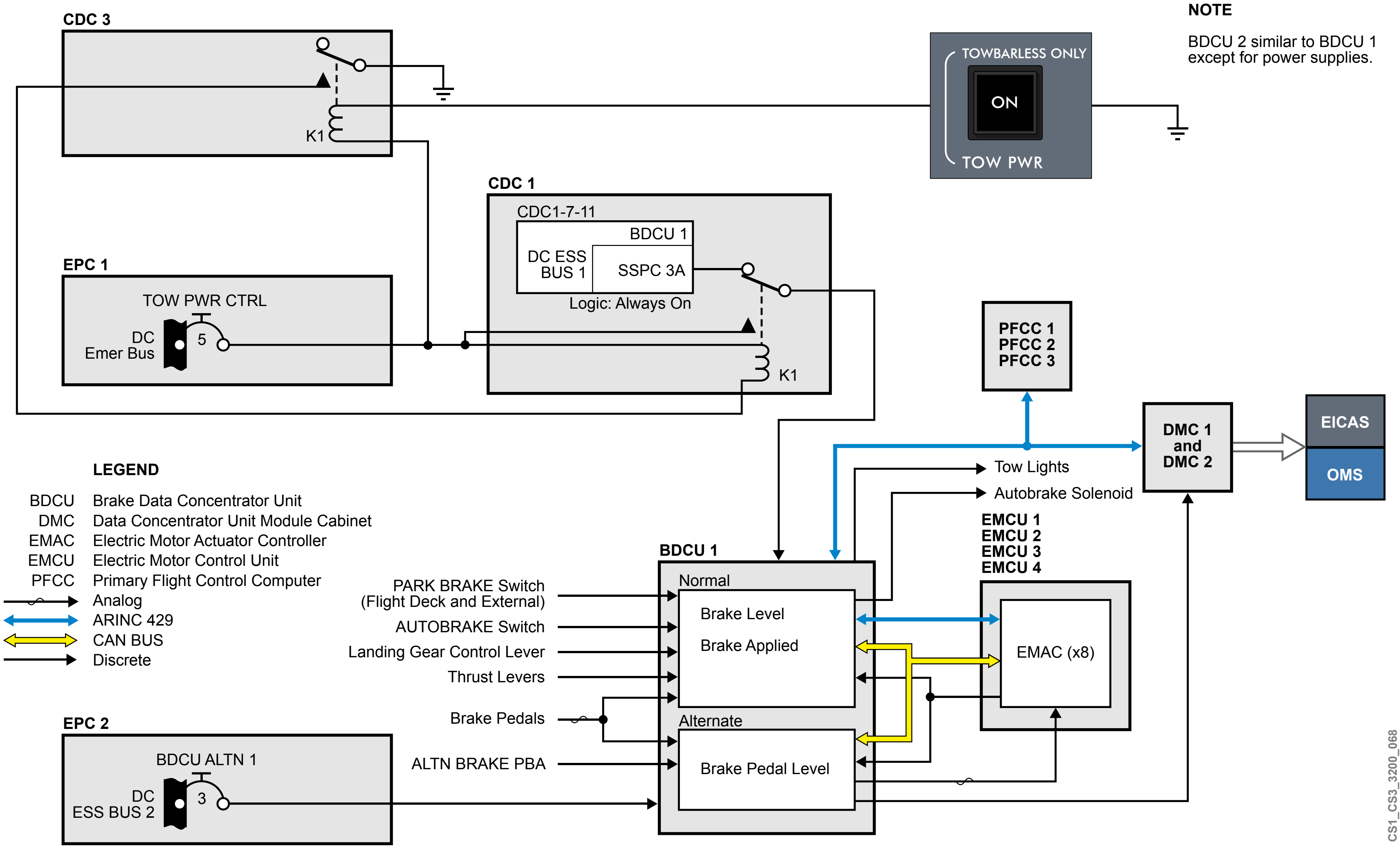


Figure 51: Brake Data Concentrator Unit

Electric Motor Control unit

Each electric motor control unit (EMCU) contains two EMACs. These are designated as EMAC A, and EMAC B.

Each EMAC has a brake command function that receives braking commands from the BDCU, and applies braking protection. The brake enable function ensures that the BDCU is generating a valid braking command. Brake control and park brake control functions are isolated to ensure park brake is available in the event of a failure.

The EMACs operate independently of each other, and provide 120 VDC to two EMAs for brake application.

The BDCU provides the normal mode braking command via ARINC 429.

When the PARK BRAKE switch is selected to ON, the EMAC provides a signal to the park brake module within the EMA for 1.5 seconds. The internal mechanical lock holds the clamping force until the PARK BRAKE switch is selected OFF. The BDCU sends alternate mode braking commands via analog signals.

A discrete enable signal is provided from the ALTN BRAKE PBA. The alternate brake mode overrides the normal mode braking command.

Each EMAC monitors one coil of the dual-coil wheel speed transducer, to independently control its EMA for braking protection.

Braking data and wheel speed are communicated over the CAN BUS to the BDCU. The EMACs provide discrete signals to the BDCU to indicate if a braking force greater than 90.7 kg (200 lb) is inadvertently applied by an EMA.

The EMACs receive analog signals from the EMAs. These include motor position monitoring, force applied and motor winding temperature.

Each EMAC receives power from DC ESS BUS 1 or DC ESS BUS 2. When the TOW PWR switch on the electrical/towing service panel is selected ON, the EMACs receive electrical power from the DC EMER BUS.

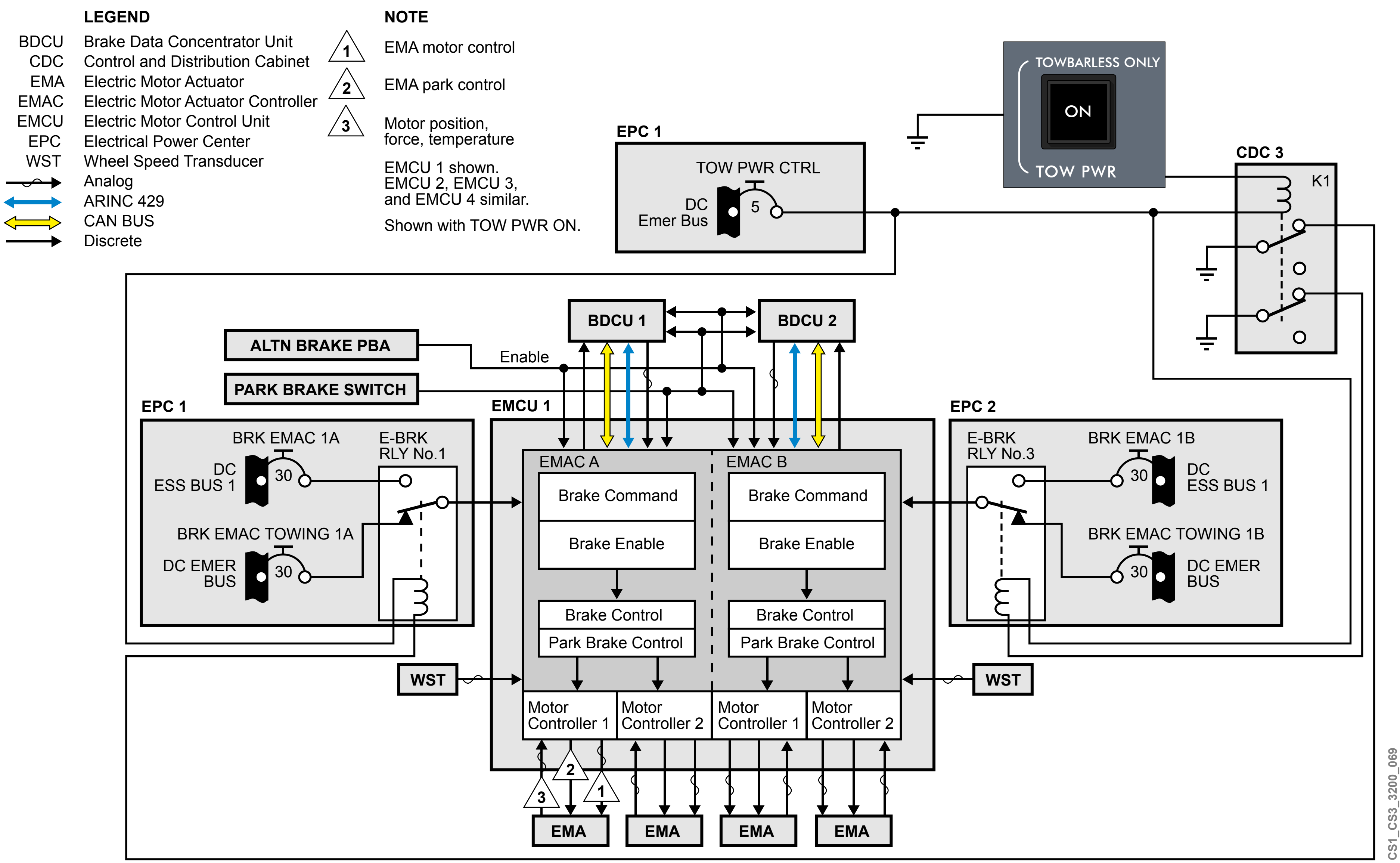


Figure 52: Electric Motor Control Unit

CONTROLS AND INDICATIONS

Flight Deck Controls

Brake Pedals

The rudder pedal assemblies have the functions of rudder control, nosewheel steering (NWS) control, and brake application. Applying pressure to the top portion of the pedal moves the pedal position transducer, which sends braking command to the brake data concentrator units (BDCUs).

Alternate Brake PBA

The guarded ALTN BRAKE PBA is located on the landing gear panel. When the PBA is selected ON, the alternate braking mode is commanded. The ON legend is displayed when the PBA is selected. Alternate braking selection overrides normal braking.

Autobrake Switch

The AUTOBRAKE rotary switch is located on the landing gear panel. This switch is a five position (RTO, OFF, LO, MED, and HI) rotary switch with dual-contacts controlled by the same action. Upon successful arming of the autobrake system, an internal solenoid holds the switch in the selected position. If a disarm condition of the autobrake system occurs, the spring-loaded switch returns to the OFF position.

Park Brake Switch

The flight deck park brake switch is a two-position switch, located on the center pedestal of the flight deck. A pull-to-turn action is required to prevent inadvertent deactivation of the switch.

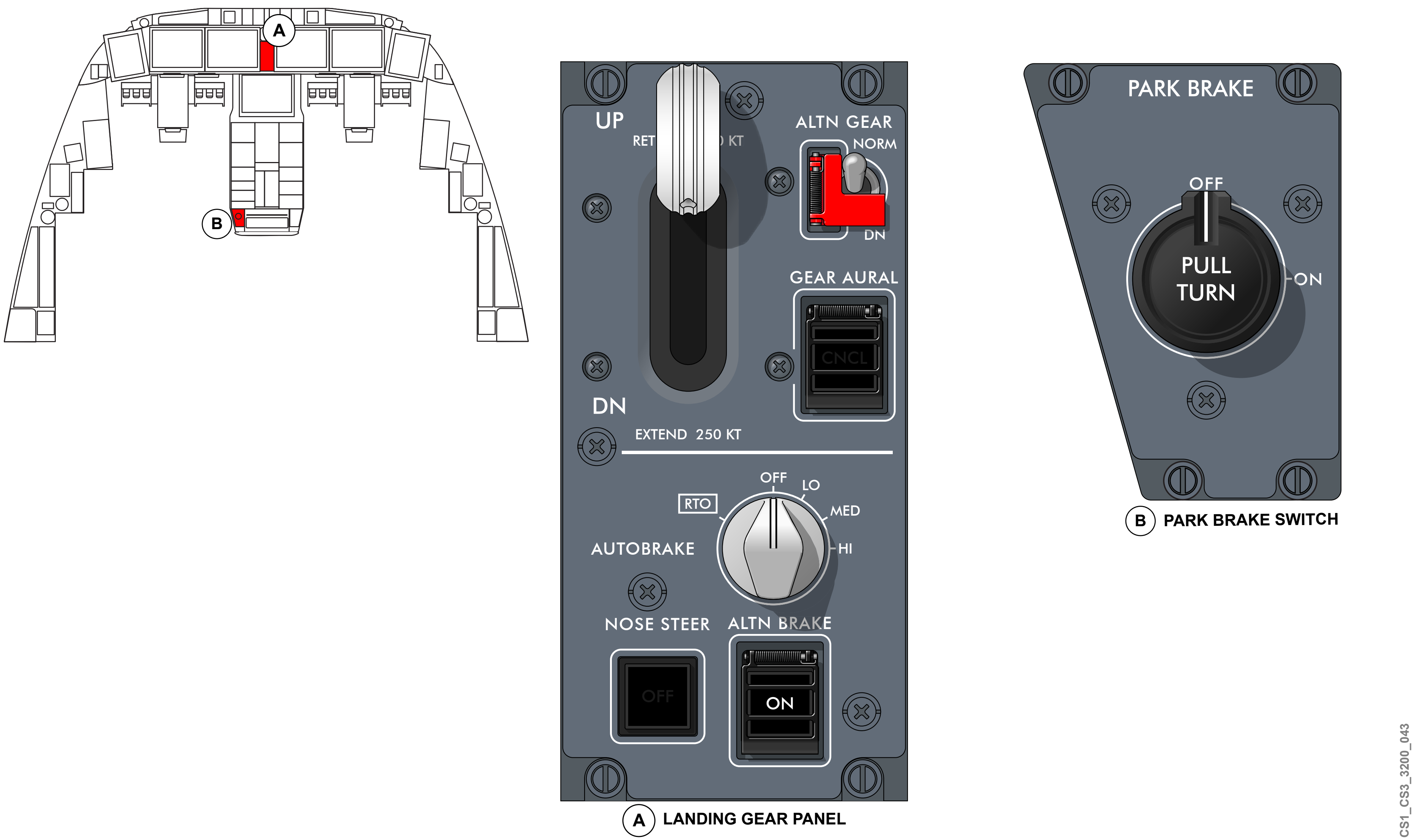


Figure 53: Flight Deck Brake System Controls

External Brake Controls and Indications

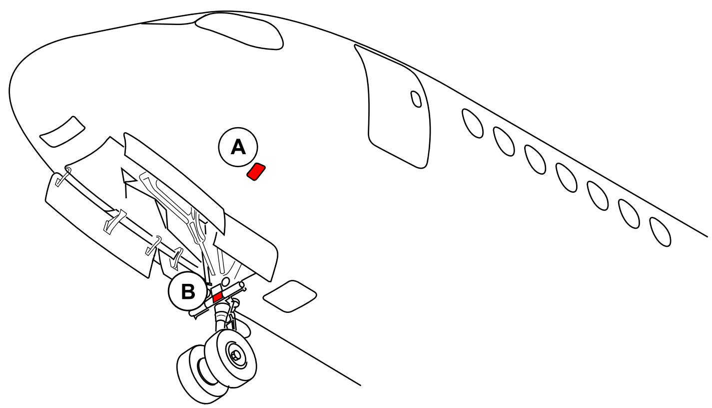
Park Brake Switch

The PARK BRK switch is located on the ELECTRICAL/TOWING SERVICE PANEL. The switch receives power from the DC EMER BUS through the TOW PWR switch. The BDCUs monitor both positions of the switch to prevent uncommanded braking.

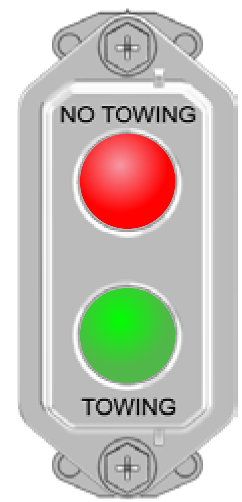
Towing Lights

A red NO TOW light and a green TOW light are located on the ELECTRICAL/TOWING SERVICE PANEL. Corresponding red NO TOWING and green TOWING lights are located on the towing indication box. The towing lights are powered from the DC EMER BUS, and controlled by both BDCUs.

The aircraft cannot be towed until the green TOW light is on. If the aircraft is powered, the flight deck PARK BRAKE switch and NOSE STEER PBA must be selected OFF. If the aircraft is unpowered, the ELECTRICAL/TOWING SERVICE PANEL PARK BRAKE switch must be selected OFF. This indicates that it is safe to tow the aircraft without damaging the brake or steering components.



A ELECTRICAL/TOWING SERVICE PANEL



B TOWING CONTROL BOX

CS1_CS3_3240_032

Figure 54: External Park Brake Controls and Indications

Brake Temperature Indication

The brake temperature monitoring system (BTMS) consists of one brake temperature sensor (BTS) in each electric brake. Colors and digits are used for pilot awareness and are displayed on the STATUS synoptic page as follows:

Green:

- If the brake temperature is between 00 and 06, no action is required

White:

- If the brake temperature is between 07 and 14, brake energy is limited to takeoff with the possibility of a blown wheel fuse plug in case of rejected takeoff. A brake HI TEMP advisory message is displayed on the EICAS.

Red:

- If the brake temperature is 15 and above (max 20), a BRAKE OVHT warning message is displayed on the EICAS

Amber:

- Dashes indicate invalid temperature indications

Brake Wear

Brake wear is computed by the electric motor actuator controllers (EMACs), and transmitted to the DMCs for display on EICAS. Brake wear indication is displayed below the brake temperature indication on the STATUS synoptic page. Each main wheel brake is displayed separately.

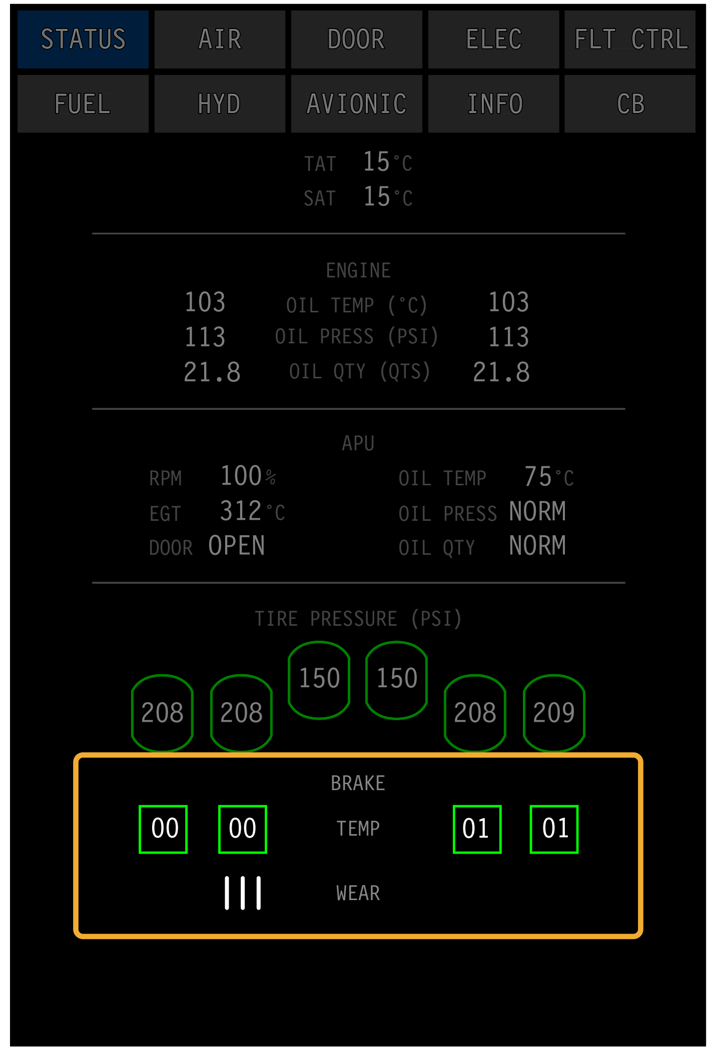
- No indication means that the brake is okay and not worn
- Three white bars indicates the brake is worn and needs to be replaced in less than 100 flight cycles
- Three amber bars indicate that the brake is worn and must be replaced immediately before the next flight

When brakes are 100% worn, in addition to three amber bars, a BRAKE FAIL caution message is displayed on the EICAS.

- An amber cross displays invalid brake wear indication

BRAKE TEMPERATURE INDICATIONS	
03	Temperature in green range (00 to 06).
09	Temperature in white range (07 to 14).
16	Temperature in red range (15 to 20).
- -	Temperature invalid.

BRAKE WEAR INDICATIONS	
	Brake OK (no indication).
	Brake to be replaced in less than 100 flight cycles.
	Brake to be replaced immediately.
	Brake wear indication invalid.



SYNOPTIC PAGE - STATUS

CS1_CS3_3200_045

Figure 55: Brake Temperature and Brake Wear Indications

DETAILED DESCRIPTION

Braking Protection

Braking protection is available to each individual brake in the normal and autobrake modes. Braking protection includes the following:

- Antiskid protection
- Locked wheel protection
- Touchdown protection

Antiskid Protection

Antiskid protection provides modulated brake control to prevent a deep skid at the individual brake to minimize the stopping distance. Antiskid is available for wheel speeds up to 204 kt. Wheel speeds less than 10 kt do not generate the required wheel rotation signal to provide antiskid protection.

Antiskid protection uses the wheel speed transducer signal to determine wheel speed. If one wheel speed transducer (WST) coil fails, the wheel speed signal from the other coil is used for wheel speed measurement.

Locked Wheel Protection

Locked wheel protection relieves brake clamping force in order to recover from a locked wheel condition.

Locked wheel protection pairs each main wheel WST, with another inboard or outboard wheel WST. This feature enables the second paired wheel speed to be used as reference.

The EMACs provide locked wheel protection to a given wheel by controlling the brake release. The brake release occurs when the wheel speed is less than 30% of the reference wheel speed.

Locked wheel protection is inhibited when the reference wheel speed is below 30 kt. This feature avoids braking command release during turning maneuvers on the ground.

If locked wheel protection is active for more than 5 seconds, locked wheel protection is inhibited (braking allowed) for that wheel. The locked wheel protection re-engages when the reference speed is above 30 kt.

Touchdown Protection

Touchdown braking protection prevents any brake force from being applied before weight-on-wheels (WOW) is confirmed. Touchdown braking protection prevents each main wheel tire from bursting on landing due to a locked wheel (brakes applied) condition.

Touchdown protection is set by the brake data concentrator units (BDCUs) when WOW status is false (in the air), and the main wheel speed is less than 30 kt. Touchdown protection is inhibited when the WOW status is true, and the main wheel speed is greater than 35 kt.

Landing Gear Retraction Braking

When the landing gear control lever (LGCL) is selected up, the BDCUs provide antiskid-controlled commands to all brakes for 5 seconds. This stops the main wheels from rotating during retraction.

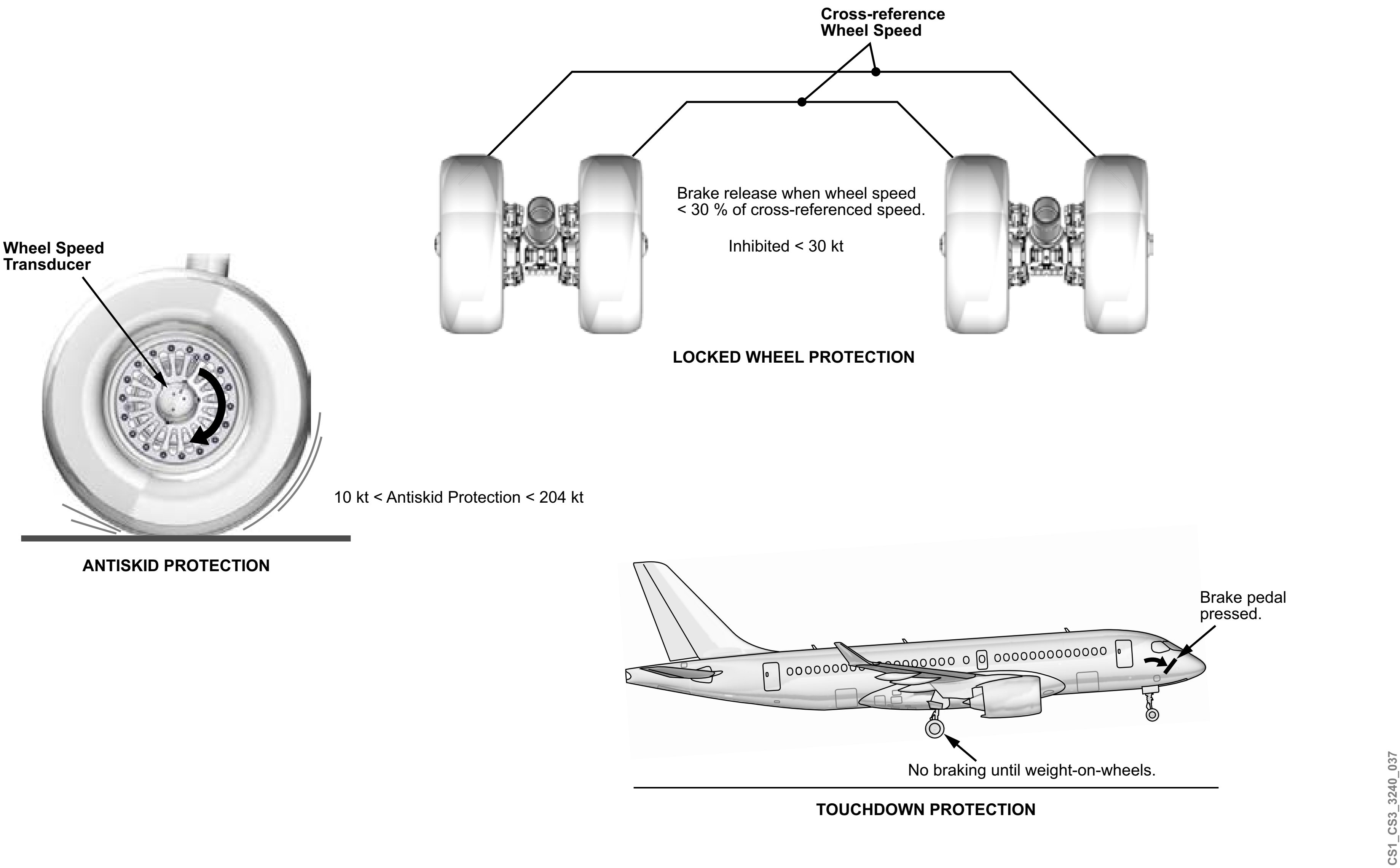


Figure 56: Braking Protection

Normal Braking Mode

Normal braking is available when the following conditions are met:

- Weight-on-wheels (WOW)
- ALTN BRAKE PBA OFF
- PARK BRAKE switches OFF (flight deck and external)
- AUTOBRAKE switch OFF

Since the BDCUs operate in an active-active mode, a failure of one BDCU automatically switches control to the other BDCU. The EMACs use the highest brake pedal input from either BDCU in determining braking force.

Normal braking is achieved by using the brake pedals. The input from the brake pedal position transducers is provided by two independent paths in the BDCUs, which generate two output commands. The two output command signals are the brake level signal, and the brake applied signal. The brake level signal is software-based, and the brake applied signal is based on brake pedal inputs.

These two independent signals are transmitted to the appropriate EMACs. The EMACs provide software-based braking protection signals to provide optimum braking performance. They also provide brake enable and brake actuator control signals.

The brake actuator control signals are transmitted to the electric motor actuators (EMAs) in the brake units. The strength of the signal determines the clamping forced applied by the EMAs. The commanded brake level uses EMA feedback information.

The EMAC is also supplied with wheel speed information by the wheel speed transducers for braking protection.

In the event that the nosewheels are not on the ground when the braking is commanded, the brake level in the BDCUs is limited by a derotation function, which sets a maximum braking level threshold until the disabling conditions are met.

Any one of the following conditions disables the derotation function and provide the normal pedal braking command.

- Nosewheels confirmed on ground (weight-on-nosewheels)
- Aircraft pitch angle is less than 1.5°
- Time delay of 5 seconds after main gear wheel spin-up

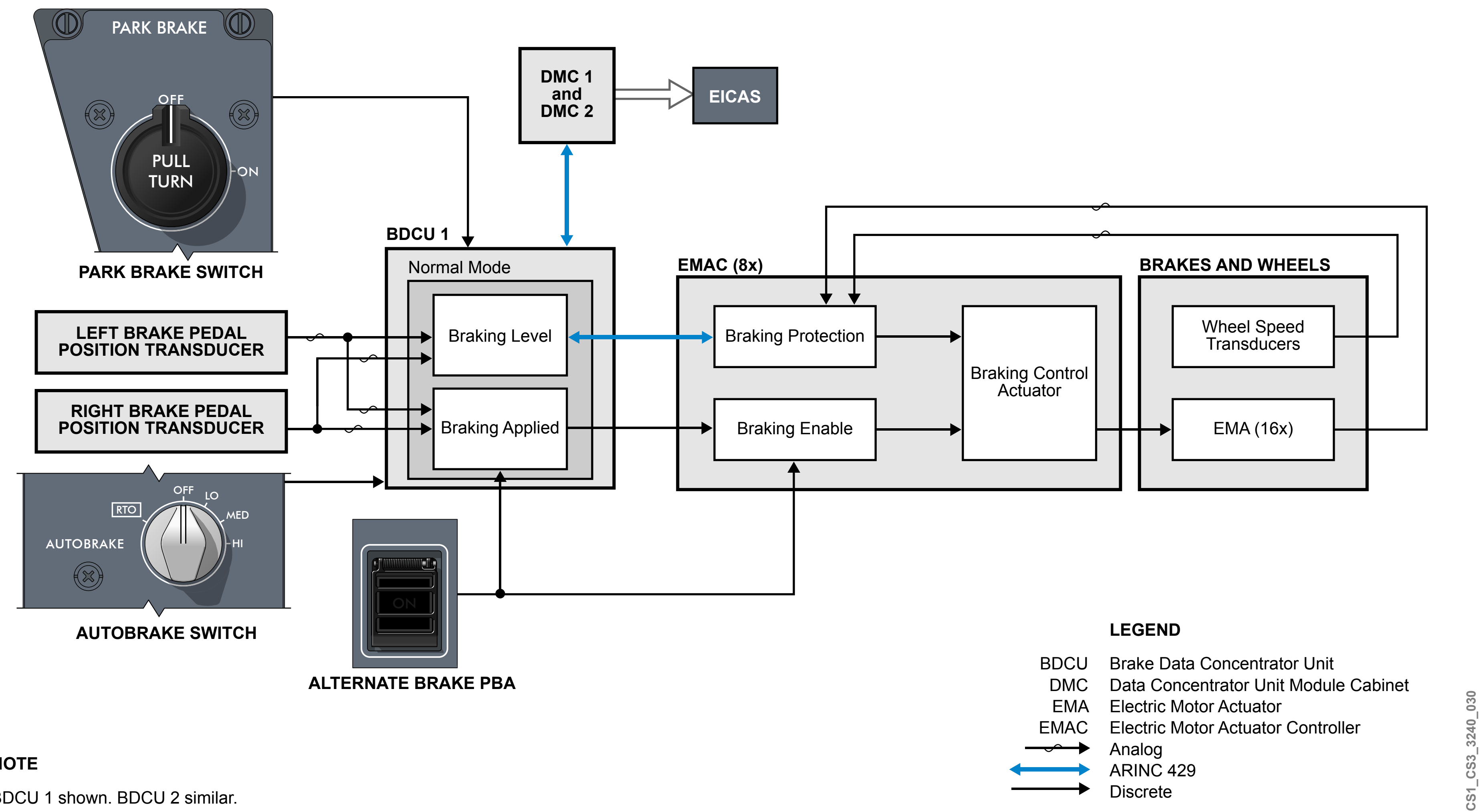
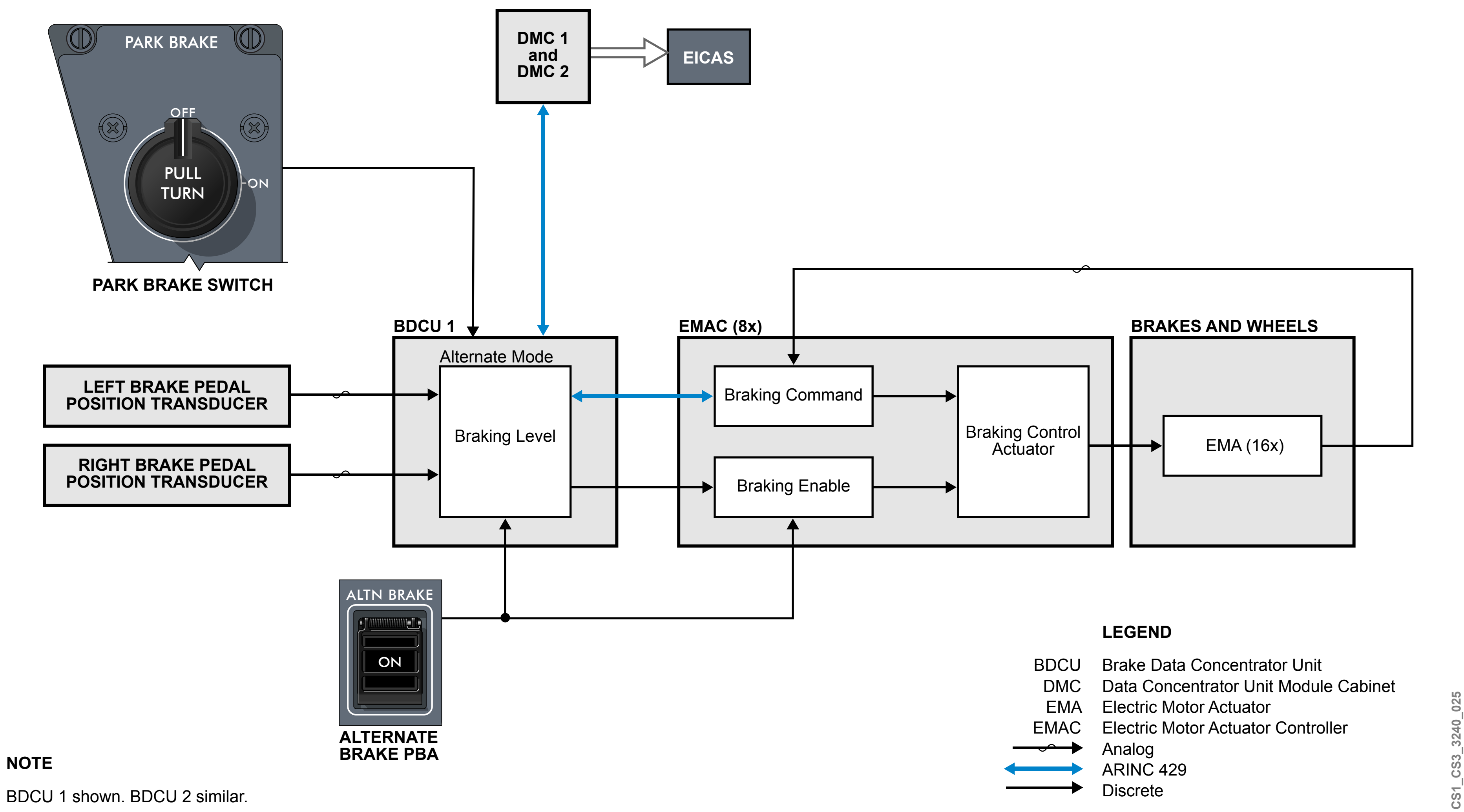


Figure 57: Normal Braking

Alternate Braking Mode

Alternate braking is enabled by selecting the ALTN BRAKE PBA ON. In addition, the PARK BRAKE switch must be OFF, and the aircraft must be weight-on-wheels (WOW).

The input from the brake pedal position transducers is provided to the BDCU alternate mode section, and switches the EMACs to alternate command. The BDCUs provide the braking level to the EMACs, based on the pedal position transducer inputs. Alternate mode is provided without any braking protection.



NOTE

BDCU 1 shown. BDCU 2 similar.

Figure 58: Alternate Braking

Autobrake Mode

The autobrake mode uses the AUTOBRAKE switch position signals, and interfaces with the normal mode sections of the BDCUs. Braking protection such as antiskid, locked wheel, and touchdown protection is active during autobraking.

In flight, the AUTOBRAKE switch solenoid must be energized to hold the switch in the desired position. The solenoid is armed by both BDCUs. The solenoid arms for the selected position, when the following conditions are true:

- Antiskid status is normal
- Airspeed greater than 60 kt
- ALTN BRAKE PBA OFF
- Weight-off-wheels

The braking level and brake applied signals are then provided to all EMACs by the BDCUs. Upon landing, autobraking activates when the thrust lever angles are below 5°.

The AUTOBRAKE deceleration levels for the LO, MED, and HI switch positions are as follows:

- LO - 4 ft/sec^2
- MED - 6 ft/sec^2
- HI - 14.5 ft/sec^2

During landing the AUTOBRAKE selection can be changed without disarming the system.

If the throttles are advanced above idle or if any brake pedal is pressed more than 20%, the autobrake mode is disabled.

Rejected Takeoff Condition

The AUTOBRAKE switch has a rejected takeoff (RTO) position that can be selected prior to takeoff. The RTO deceleration rate is a function of the initial speed when the RTO was initiated. This deceleration rate remains constant until the aircraft stops.

The RTO selection provides the following deceleration rates when activated:

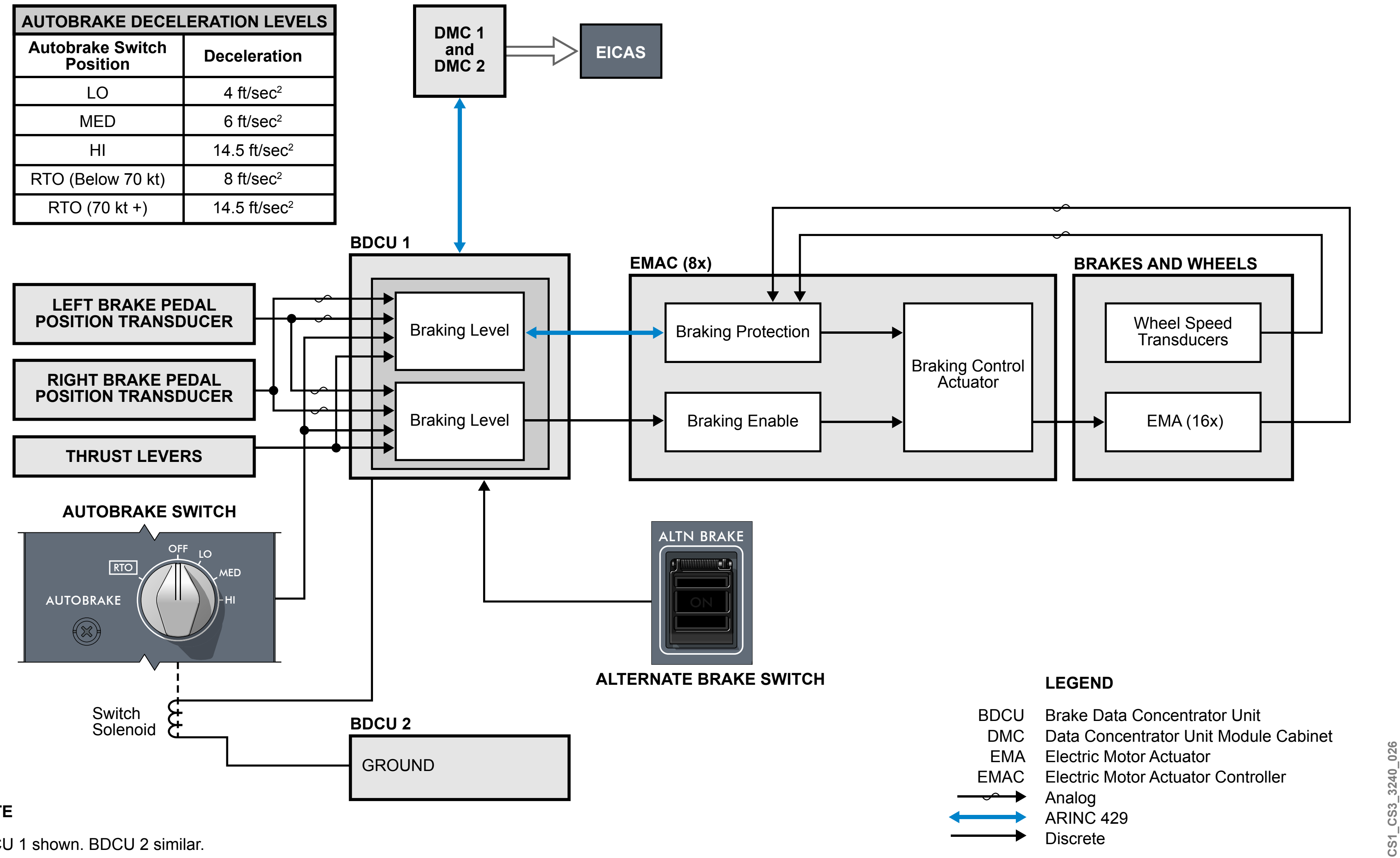
- Below 70 kt - 8 ft/sec^2
- 70 kt and over - maximum braking available

The RTO function arms when the AUTOBRAKE switch is in RTO position, and:

- Antiskid status is normal
- Weight-on-wheels (WOW)
- ALTN BRAKE PBA OFF
- Airspeed less than 55 kt

The RTO function is enabled when airspeed is greater than 60 kt with both thrust levers above idle. The RTO function is activated when both thrust levers are retarded to idle.

AUTOBRAKE DECELERATION LEVELS	
Autobrake Switch Position	Deceleration
LO	4 ft/sec^2
MED	6 ft/sec^2
HI	14.5 ft/sec^2
RTO (Below 70 kt)	8 ft/sec^2
RTO (70 kt +)	14.5 ft/sec^2



NOTE

BDCU 1 shown. BDCU 2 similar.

Figure 59: Autobrake Mode

Flight Deck Park Brake Control

When the flight deck PARK BRAKE switch is selected to the ON position, the BDCUs and EMACs receive discrete signals to apply the parking brake. The arrangement allows the EMACs to provide brake force without BDCU input, allowing the park brake to be applied in the alternate brake mode.

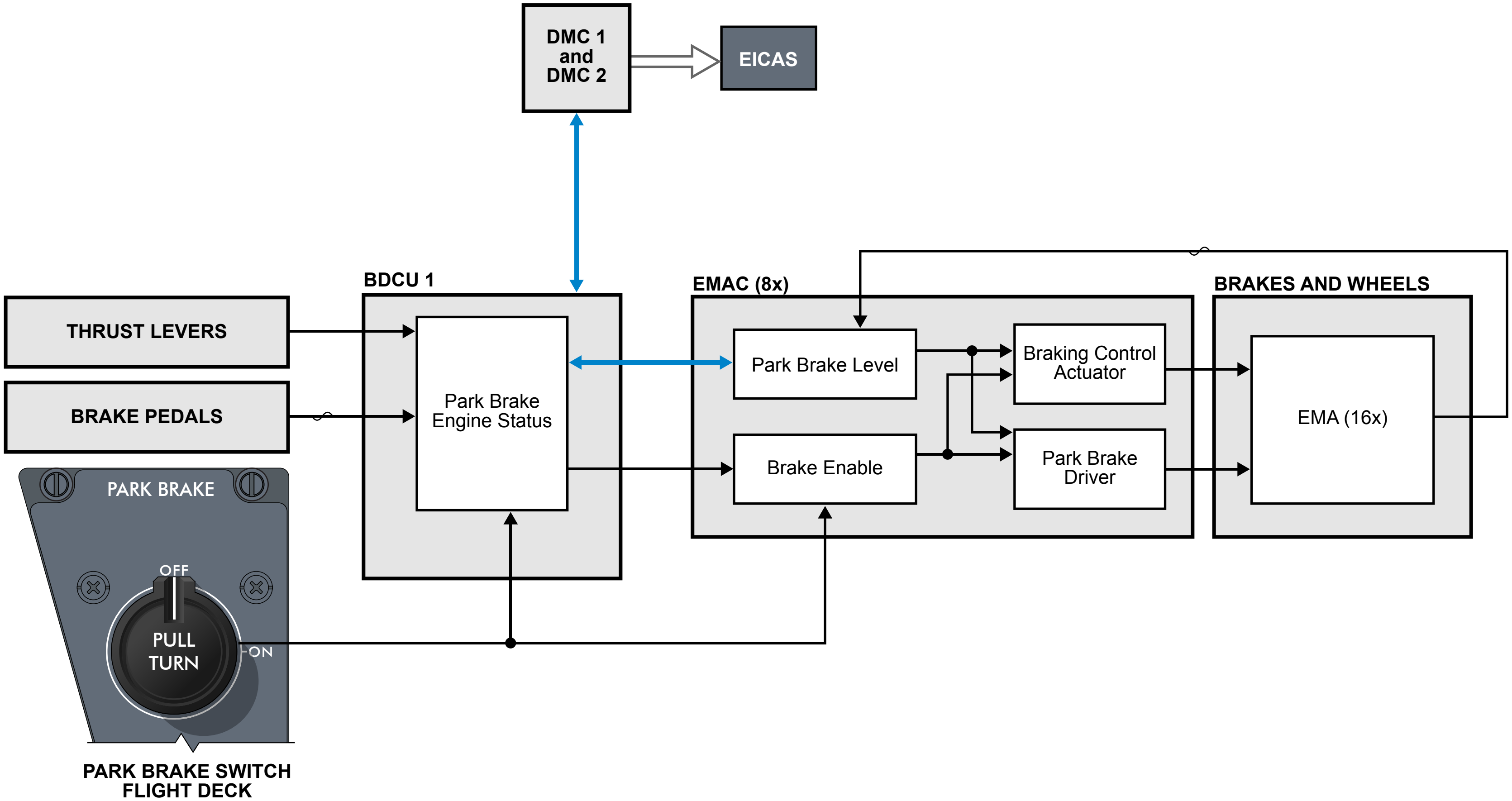
Park brake force command is set to 1360.8 kg (3000 lb) for 1.5 seconds after which the internal mechanical lock of the EMA engages. This maintains the clamping force.

The BDCUs monitor thrust lever position. If either engine is running and thrust lever position is advanced greater than 5°, a park brake force of 5216.3 kg (11,500 lb) is applied and the mechanical lock is engaged. This is designed to prevent undesired aircraft movement during engine run-up. To protect against mechanical lock failure, the EMA force is reduced to approximately 2268 kg (5000 lb) after 5 seconds.

When the thrust lever position is returned to idle, the EMACs reduce the brake force to 1360.8 kg (3000 lb), and re-engage the mechanical lock. If the thrust lever angle or engine running information is not available, the BDCU defaults the park brake force to 2540.1 kg (5600 lb).

To release the park brake lock, electrical power must be applied, and the flight deck park brake switch selected OFF.

If the park brake is ON, and the brake pedals are depressed more than 70%, the park brake releases and the brake pedal command is applied. The park brake is reapplied when the brake pedal position is less than 40%.



LEGEND

BDCU	Brake Data Concentrator Unit
DMC	Data Concentrator Unit Module Cabinet
EMA	Electric Motor Actuator
EMAC	Electric Motor Actuator Controller
	Analog
	ARINC 429
	Discrete

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NOTE

BDCU 1 shown. BDCU 2 similar.

Figure 60: Flight Deck Park Brake Control

External Park Brake Mode

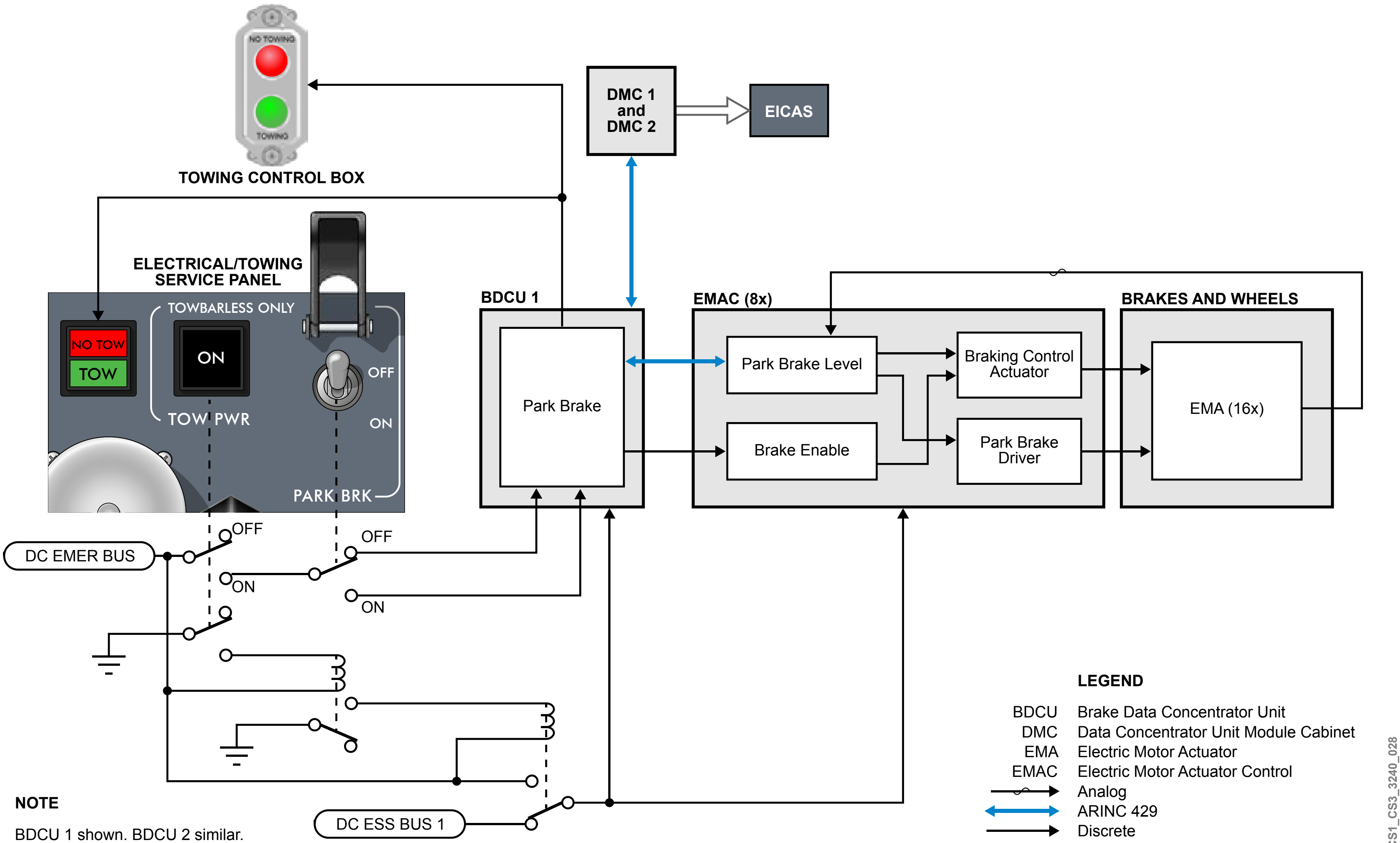
The external PARK BRK switch, located on the ELECRTRICAL/TOWING SERVICE panel, is used to set or release the parking brake during towing when no power is on the aircraft. The PARK BRK switch does not operate until the TOW PWR PBA is selected ON.

The PARK BRK switch is a guarded switch and is left in the ON position when the guard is closed and the ELECTRICAL/TOWING SERVICE PANEL door is closed.

A relay supplies 28 VDC from the DC EMER BUS to the BDCU, EMACs and the PARK BRK switch. When the aircraft is ready to be towed, the PARK BRK switch is selected OFF and the parking brake is released as indicated by the green TOW lights on the nose landing gear TOWING CONTROL panel and the ELECTRICAL/TOWING SERVICE PANEL. Once the aircraft is parked, the PARK BRK switch is selected ON and the parking brake is set. Once the parking brake is set, the TOW PWR PBA can be selected OFF.

When power is applied to the aircraft, the flight deck PARK BRAKE switch overrides the ELECTRICAL/TOWING SERVICE PANEL PARK BRK switch

As with flight deck parking brake control, a force of 1360.8 kg (3000 lb) is applied by the EMAs for 1.5 seconds, after which the EMA mechanical locks engage.



NOTE

BDCU 1 shown. BDCU 2 similar.

Figure 61: External Park Brake Control

MONITORING AND TESTS

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

CAS Messages

Table 9: WARNING Messages

MESSAGE	LOGIC
CONFIG BRAKE	Park brake is applied when in takeoff mode.
BRAKE OVHT	Any brake temperature that reaches an indication of 15 or more.

Table 10: CAUTION Messages

MESSAGE	LOGIC
NOSE TIRE LO PRESS	Nose tire pressure below 70% of normal pressure in flight.
BRAKE FAIL	Total loss of braking in the normal mode (NORM BRAKE FAIL) and alternate mode (ALTN BRAKE FAIL).
NORM BRAKE FAIL	Total loss of braking in the normal mode or communication failure from BDCU.
PLT BRAKE PEDAL FAIL	Normal and alternate modes pilot pedal braking not available on L and/or R pilot brake pedals.
CPLT BRAKE PEDAL FAIL	Normal and alternate modes copilot pedal braking not available on L and/or R copilot brake pedals.
L BRAKE FAIL	Commanded braking force not reachable or 2 EMAs failed or loss of antiskid protection on left MLG brake. Refer to INFO messages.
R BRAKE FAIL	Commanded braking force not reachable or 2 EMAs failed or loss of antiskid protection on right MLG brake. Refer to INFO messages.
BRAKE ON	Detection of uncommanded brake force greater than 90.7 kg (200 lb) or park brake is on in the air.
AUTOBRAKE FAIL	Failure of autobrake system after being armed or autobrake function cannot be armed.
PARK BRAKE FAIL	Failure to set parking brake or insufficient parking force present after commanding the function.

Table 11: ADVISORY Messages

MESSAGE	LOGIC
TIRE LO PRESS	On ground, any tire < 90% of nominal pressure. In flight, main tire < 70% of nominal pressure.
TIRE PRESS FAULT	One or more sensors failed on tire. Refer to INFO messages.
L BRAKE DEGRADED	Partial loss of functionality on the left brake. Refer to INFO messages.
R BRAKE DEGRADED	Partial loss of functionality on the right brake. Refer to INFO messages.
BRAKE FAULT	Loss of redundant or fault of non-critical function for the brake control system. Refer to INFO messages.
BRAKE HI TEMP	Any brake temperature reaches threshold 07 or greater.
ALTN BRAKE FAIL	Failure of the alternate brake function or loss of a pilot and copilot brake pedal transducer.

Table 12: STATUS Messages

MESSAGE	LOGIC
PARK BRAKE ON	Park brake switch has been selected and parking brake on.
ALTN BRAKE ON	Alternate brake switch has been selected and confirmed on.
AUTOBRAKE LO	Autobrake switch is in the LO position and arming conditions are acceptable.
AUTOBRAKE MED	Autobrake switch is in the MED position and arming conditions are acceptable.
AUTOBRAKE HI	Autobrake switch is in the HI position and arming conditions are acceptable.
AUTOBRAKE RTO	Autobrake switch is in the RTO position and arming conditions are acceptable.

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Table 13: INFO Messages

MESSAGE	LOGIC
32 TIRE PRESS FAULT - L MLG INBD TPIS INOP	The left main landing gear inboard TPM is inoperative or the left main landing gear inboard AIM is inoperative.
32 TIRE PRESS FAULT - L MLG OUTBD TPIS INOP	The left main landing gear outboard TPM or AIM is inoperative.
32 TIRE PRESS FAULT - R MLG INBD TPIS INOP	The right main landing gear inboard TPM or AIM is inoperative.
32 TIRE PRESS FAULT - R MLG OUTBD TPIS INOP	The right main landing gear outboard TPM or AIM is inoperative.
32 TIRE PRESS FAULT - L NOSE TPIS INOP	The left nose landing gear TPM or AIM is inoperative.
32 TIRE PRESS FAULT - R NOSE TPIS INOP	Right nose landing gear TPM or AIM is inoperative.
32 TIRE PRESS FAULT - TPMU INOP	The TPMU is inoperative. Possible failure of TPIS interfaces such as TPMU label or ARINC 429, or pin programing.
32 L BRAKE FAIL - L WHEEL SPEED INOP	Two channels on the same wheel speed transducer on the left landing gear wheels (ROB and/or RIB) have failed. Antiskid protection on at least one of the left landing gear wheels is not provided, except for the locked wheel function which inhibits braking for up to 5 seconds.
32 L BRAKE FAIL - L BRAKE CODE 1 INOP	A significant failure(s) has occurred that warrants cautious braking methods (ex. loss of multiple EMAs) on the left gear brakes.

Table 13: INFO Messages

MESSAGE	LOGIC
32 R BRAKE FAIL - R WHEEL SPEED INOP	Two channels on the same wheel speed transducer on the right landing gear wheels (ROB and/or RIB) have failed. Antiskid protection on at least one of the right landing gear wheels is not provided, except for the locked wheel function which inhibits braking for up to 5 seconds.
32 R BRAKE FAIL - R BRAKE CODE 1 INOP	A significant failure(s) has occurred that warrants cautious braking methods (ex. loss of multiple EMAs) on the right gear brakes.
32 L BRAKE DEGRADED - L GEAR EMAC INOP	A failed EMAC that control two EMAs on the left landing gear brakes. This implies that braking force from two EMAs may not be available (75% of the brake force is available on the left gear).
32 L BRAKE DEGRADED - L GEAR 2 EMA INOP	Two EMAs failure on the left landing gear brakes. This implies that commanded braking force for two EMAs may not be available (75% of the brake force is available on the left gear).
32 L BRAKE DEGRADED - L GEAR 1 EMA INOP	One EMA failure on the left landing gear brakes. This implies that commanded braking force from one EMA may not be available (88% of the brake force is available on the left gear).
32 L BRAKE DEGRADED - L WHEEL SPEED CHAN INOP	At least one wheel speed transducer channel (LIB and/or LOB) on the left landing gear wheels have failed. Antiskid protection is not provided to two EMAs (half a brake) on the left landing gear wheel(s).
32 R BRAKE DEGRADED - R GEAR EMAC INOP	A failed EMAC that controls two EMAs on the right landing gear brakes. This implies that the commanded braking force from two EMAs may not be available (75% of the brake force is available on the right gear).
32 R BRAKE DEGRADED - R GEAR 2 EMA INOP	Two EMAs failure on the right landing gear brakes. This implies that commanded braking force from two EMAs may not be available (75% of the brake force is available on the right gear).

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Table 13: INFO Messages

MESSAGE	LOGIC
32 R BRAKE DEGRADED - R GEAR 1 EMA INOP	One EMA failure on the right landing gear brakes. This implies that commanded braking force from one EMA may not be available (88% of the brake force is available on the right gear).
32 R BRAKE DEGRADED - R WHEEL SPEED CHAN INOP	At least one wheel speed transducer channel (RIB and/or ROB) on the right landing gear wheels failed. Antiskid protection is not provided to two EMAs (half a brake) on the right landing gear wheel(s).
32 BRAKE FAULT - BDCU 1 ALTN INOP	Alternate brake mode from BDCU 1 is not available.
32 BRAKE FAULT - BDCU 2 ALTN INOP	Alternate brake mode from BDCU 2 is not available.
32 BRAKE FAULT - BDCU 1 NORM INOP	Normal brake mode from BDCU 1 is not available.
32 BRAKE FAULT - BDCU 2 NORM INOP	Normal brake mode from BDCU 2 is not available.
32 BRAKE FAULT - BRAKE CODE 2 INOP	A loss of a non-critical redundant function in the brake control system.
32 BRAKE FAULT - GEAR RETRACT INOP‘	A failure of the gear retraction braking function. Main wheels spindown before entering the wheel bin area may not occur.
32 BRAKE FAULT - IFT INOP	Since the test was not performed, unknown braking issues may be present on landing.
32 BRAKE FAULT - THROTTLE RVDT INOP	Failure of the engine throttle RVDT. Autobrake function cannot be armed and is disabled/not operational.

Table 13: INFO Messages

MESSAGE	LOGIC
32 BRAKE FAULT - WOW DISAGREE	A discrepancy between MG_WOW and air speed < 60 kt and defaults the system to a ground condition.
32 BRAKE FAULT - L PILOT PEDAL SENSOR REDUND LOSS	Loss of a left pilot pedal transducer redundant channel. No loss of braking function.
32 BRAKE FAULT - R PILOT PEDAL SENSOR REDUND LOSS	Loss of a right pilot pedal transducer redundant channel. No loss of braking function.
32 BRAKE FAULT - L COPILOT PEDAL SENSOR REDUND LOSS	Loss of the left copilot pedal transducer redundant channel. No loss of braking function.
32 BRAKE FAULT - R COPILOT PEDAL SENSOR REDUND LOSS	Loss of the right copilot pedal transducer redundant channel. No loss of braking function.
32 BRAKE FAULT - BRAKE TEMP SENSOR INOP	A LOB, LIB, RIB, and/or ROB brake temperature sensor failed.

PRACTICAL ASPECTS

The OMS is used to perform retraction of the EMAs, and to provide a return to service (RTS) test.

EMA Retraction

The EMAs of a brake can be individually retracted.

RETURN TO SERVICE

During RTS testing the EMAs are completely retracted then advanced against the brake stack.

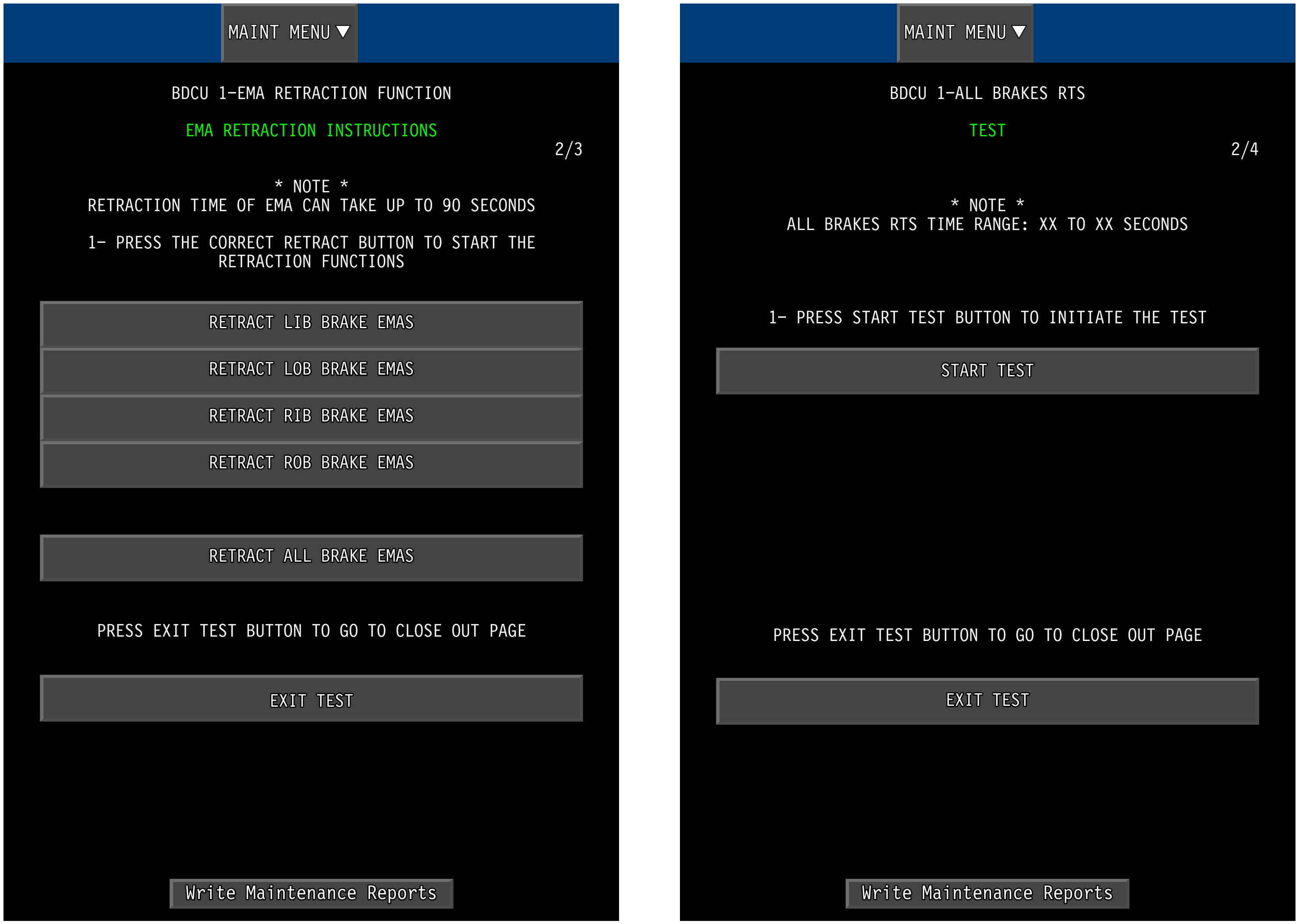

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Figure 62: Landing Gear OMS Functions

Brake System Data

The OMS is used to monitor the following data:

- Brake pedal position
- Brake wear data/prediction
- Electric motor actuator (EMA) brake force

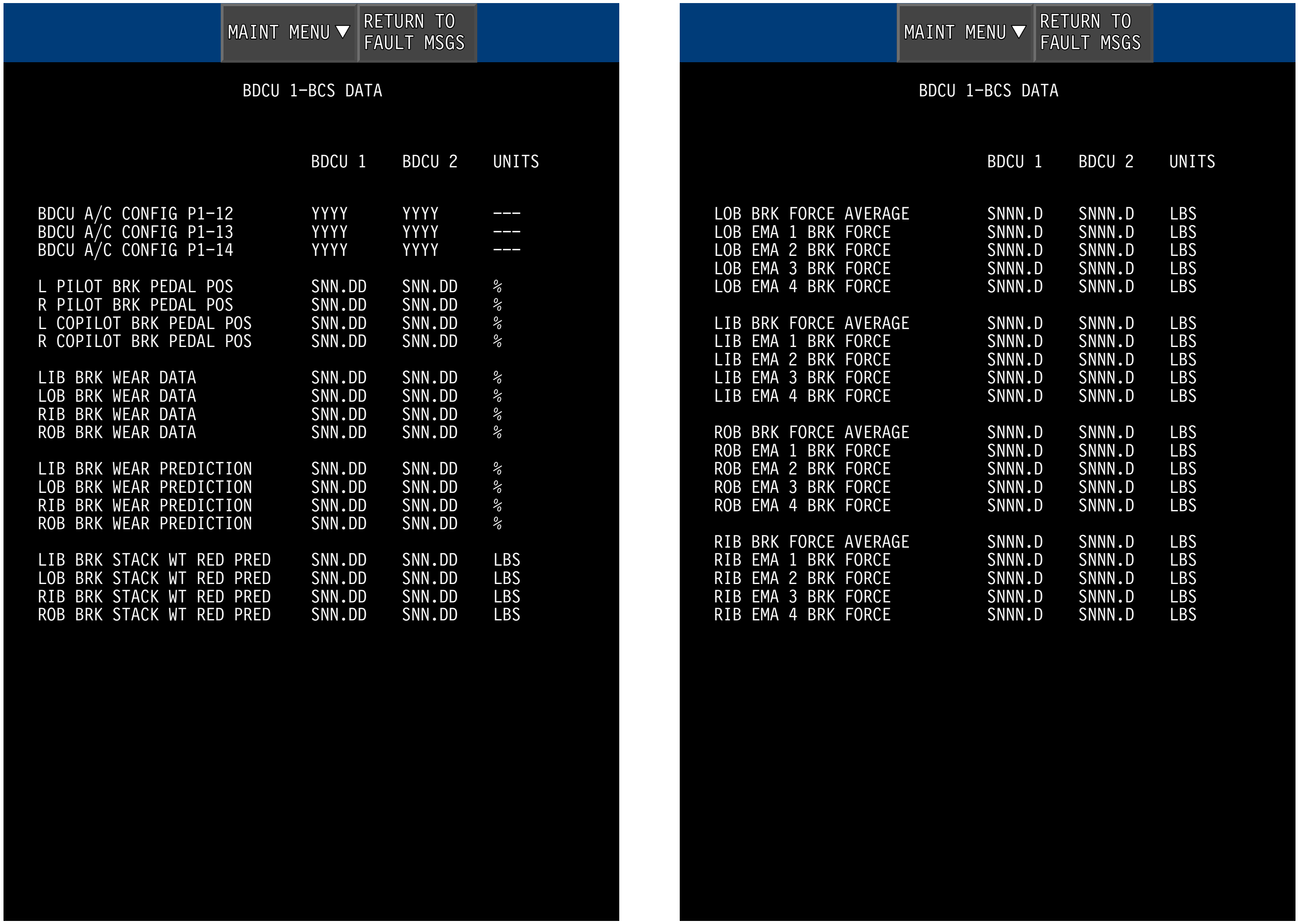

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Figure 63: Brake OMS Data

Brake System Monitoring

The OMS is used to monitor parameters of the brake system including the following:

- Tire pressure
- Brake temperature
- Axle interface module/wheel speed transducer temperature

In addition, the non-volatile memory (NVM) of the tire pressure monitoring unit (TPMU) can be erased.

CS1_CS3_3240_035

Figure 64: Brake System OMS Monitoring

32-51 NOSEWHEEL STEERING SYSTEM

GENERAL DESCRIPTION

The nosewheel steering (NWS) system is a steer-by-wire system. There are no mechanical connections between the flight deck controls and the steering components. The system is electronically controlled by two landing gear and steering control units (LGSCUs), which operate in an active/standby configuration.

The primary controls of the NWS system are the steering tiller, and the rudder pedals. The steering tiller is installed on the pilot side and a second optional steering tiller can be installed on the copilot side. If the copilot steering tiller is fitted to the aircraft, the steering commands from the two tillers are processed by the LGSCUs to a single steering command.

The rudder pedals are used during takeoff and landing. Rudder pedal inputs are routed through the primary flight control computers (PFCCs), which also provide steering inputs during rollout after an autoland.

A PEDAL DISC pushbutton, located on the steering tiller, disconnects the rudder pedal inputs which allows the rudder to operate without moving the nosewheels.

The NWS system has a powered steering range of ± 80° during tiller steering, and ± 9° with rudder pedal steering. The steering angle range automatically decreases with increased ground speed.

The steering control valve (SCV) is powered by hydraulic system no. 2 pressure. The SCV directs pressure to either side of the rack-and-pinion type steering motor on the NLG main fitting. Displacement of the motor piston rack rotates the nosewheels. The SCV maintains pressure to the steering motor to avoid NLG shimmy.

Two steering feedback unit sensors are mounted to the NLG main fitting to provide steering angle signals to the LGSCU.

A NOSE STEER PBA on the landing gear panel turns the nosewheel steering off for towing and maintenance. The NLG can be turned up to 130° from the aircraft centerline, and can be accomplished without disconnecting the torque links. The NOSE STEER PBA must be selected to the OFF position, and the park brake released before towing.

The steering overtravel proximity sensor monitors steering angles to alert the crew if the nosewheel steering angle has been exceeded. The proximity sensor target shears at an angle approximately ±135°, and provides visual indication of excessive steering angle.

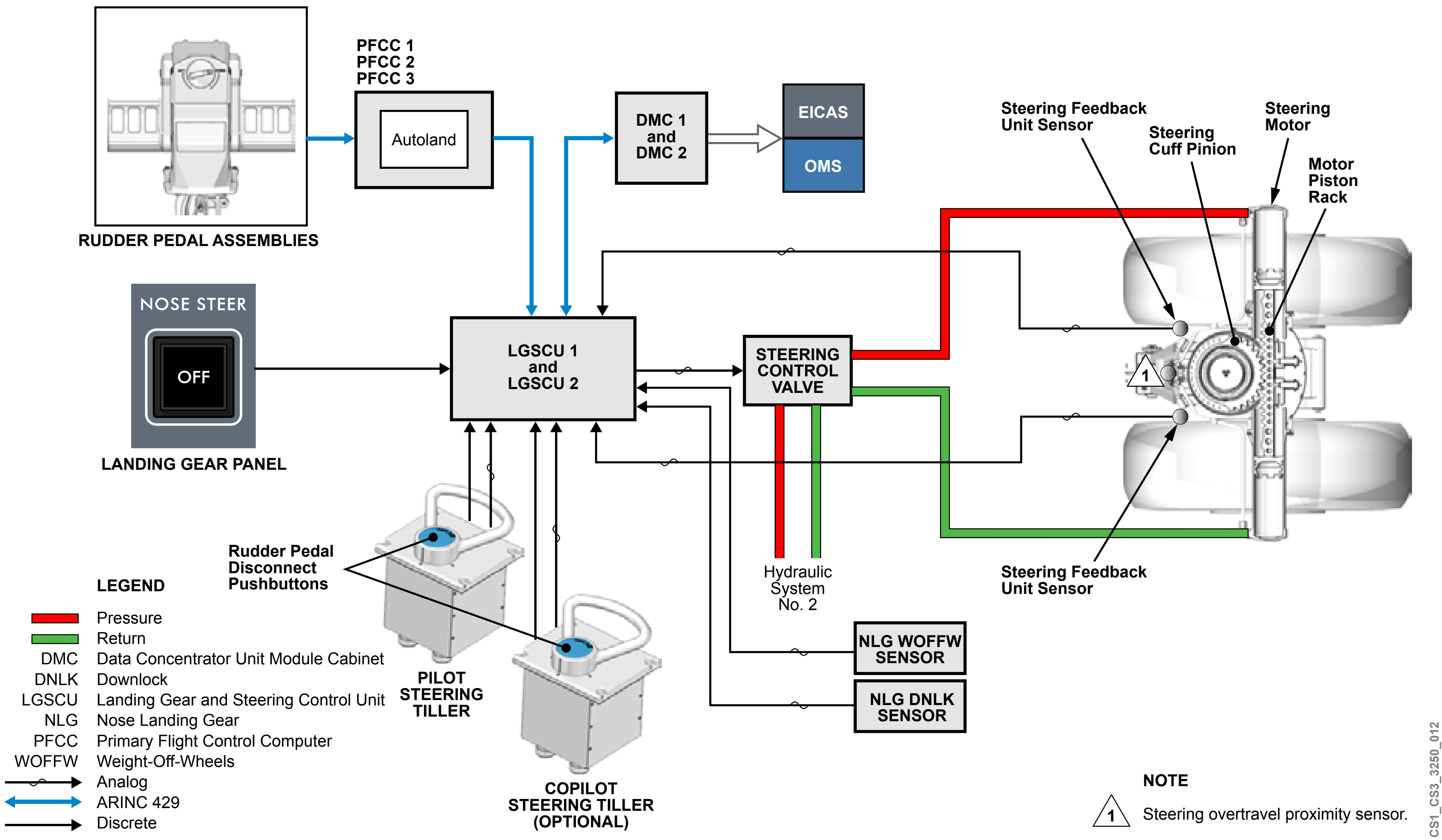


Figure 65: Nosewheel Steering System

COMPONENT LOCATION

STEERING CONTROL VALVE

The steering control valve is located on the aft side of the NLG main fitting.

STEERING MOTOR

The steering motor is located on the forward side of the NLG main fitting.

STEERING CUFF

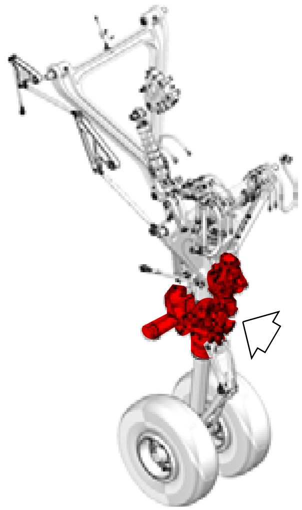
A steering cuff is located at the lower end of the NLG main fitting.

STEERING OVERTRAVEL PROXIMITY SENSOR

A steering overtravel proximity sensor is located on the aft side of the NLG main fitting, between the steering feedback sensors.

STEERING FEEDBACK UNIT SENSORS

Two steering feedback unit sensors are located on the aft side of the NLG main fitting.



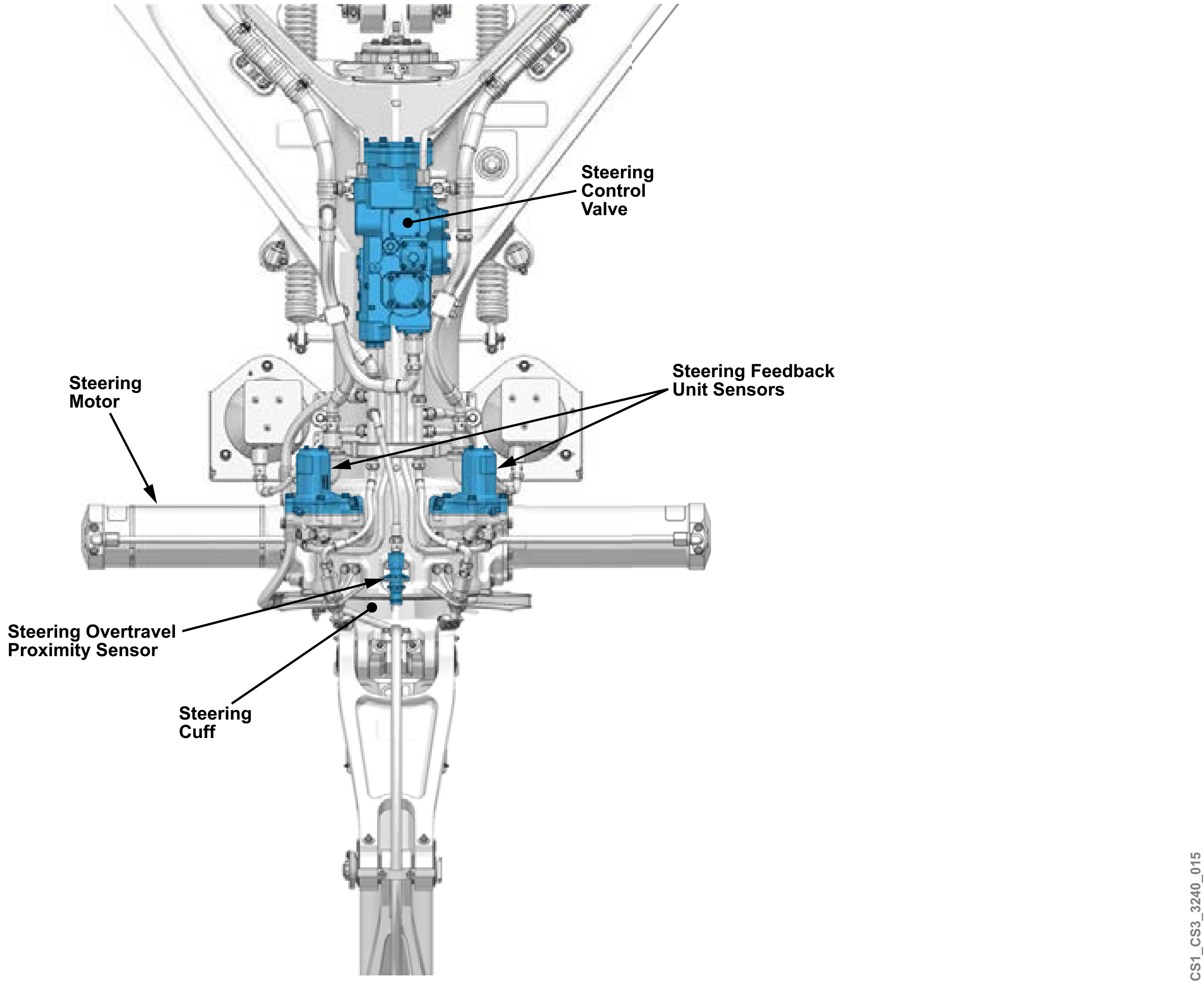


Figure 66: Nosewheel Steering System Component Location

STEERING TILLER

A steering tiller is located in the flight deck on the left console. An optional steering tiller is available for the right console.

LANDING GEAR AND STEERING CONTROL UNIT

Landing gear and steering control unit 1 (LGSCU 1) is located in the forward equipment bay. LGSCU 2 is located in the mid equipment bay.

RUDDER PEDAL ASSEMBLIES

A left rudder pedal assembly and a right rudder pedal assembly are located in the flight deck.

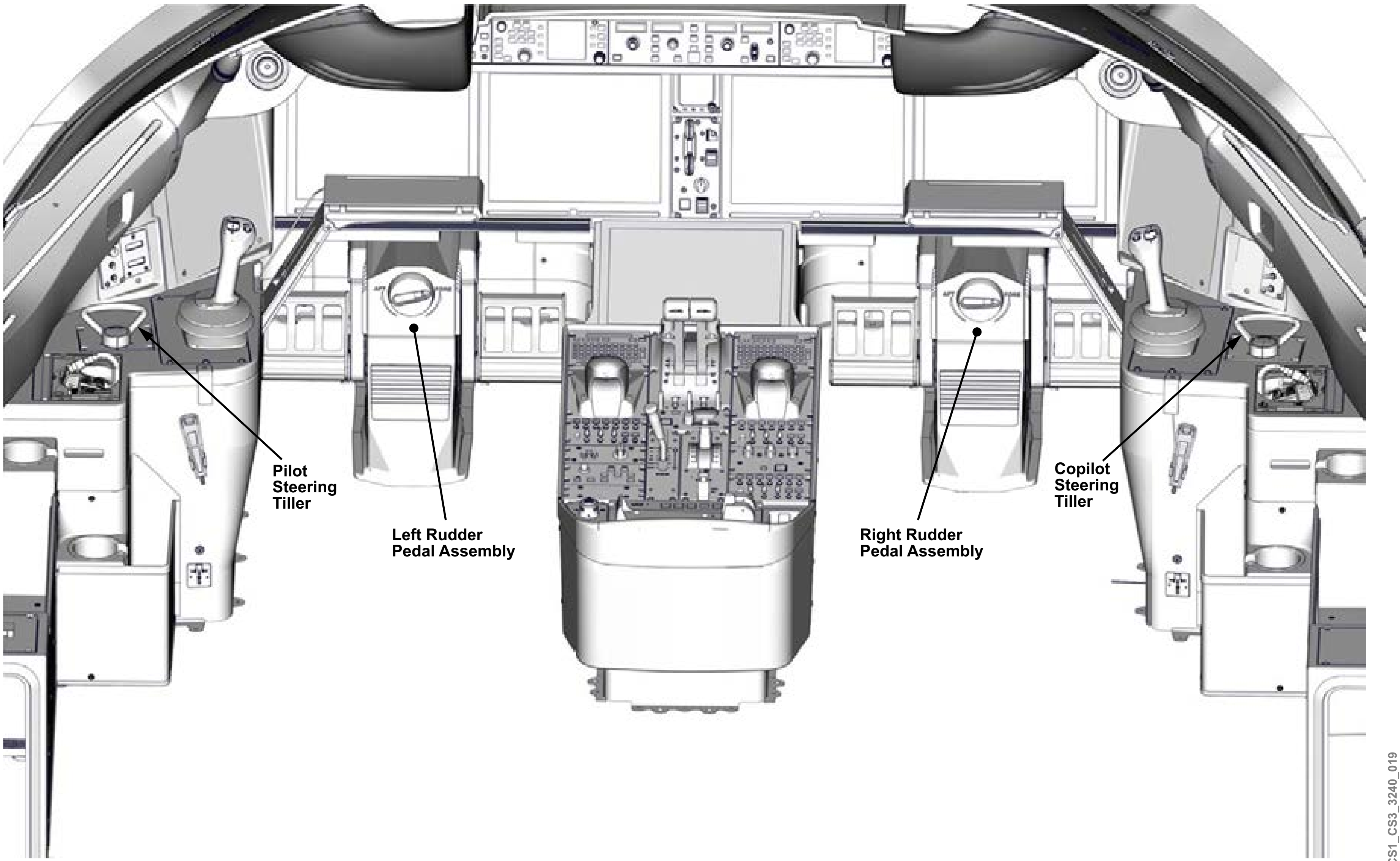


Figure 67: Steering Tiller and Rudder Pedal Assemblies Location

COMPONENT INFORMATION

STEERING CONTROL VALVE

The steering control valve (SCV) is controlled by electrical signals from the active LGSCU, and routes hydraulic pressure to the steering motor.

The SCV is provided with an inlet supply pressure port, two steering motor supply ports, and an electrical connector. An internal filter is located downstream of the supply pressure port. A compensator in the valve maintains hydraulic back pressure in the steering motor when the hydraulic system is not pressurized, to provide shimmy damping.

STEERING MOTOR

The steering motor consists of a cylinder and a rack. The cylinder is divided into two sections. Each section is equipped with a piston and a hydraulic supply port. Hydraulic pressure acting on a piston moves the rack, which engages a pinion on the steering cuff to rotate it.

STEERING CUFF

Rotation of the steering cuff rotates the NLG piston axle assembly through the NLG torque links. The steering cuff is equipped with gears for driving the steering feedback unit sensors, and a pinion that engages the steering motor rack. A baulk ring on the steering cuff supports the steering motor rack during the passive steering (towing) range.

STEERING FEEDBACK UNITS

Two steering feedback units engage a gear on the steering cuff. When the steering cuff moves, it rotates the feedback unit gear, which operates the feedback unit sensor (resolver). The resolver transforms the rotation to an electric analog signal, which is processed by the LGSCU as the nosewheel steering angle deflection. An electrical connector is located on the resolver.

STEERING OVERTRAVEL PROXIMITY SENSOR

The overtravel proximity sensor relays a message to the LGSCU when the position of the nosewheel steering angle has been exceeded. The NOSE STEER MISALIGN caution message is displayed on the EICAS to alert the crew that nosewheel steering angles (± 130°) are beyond the active steering range.

The steering overtravel bracket shears at an angle approximately ±135°. The shear section of the target is connected with a lanyard for retention.

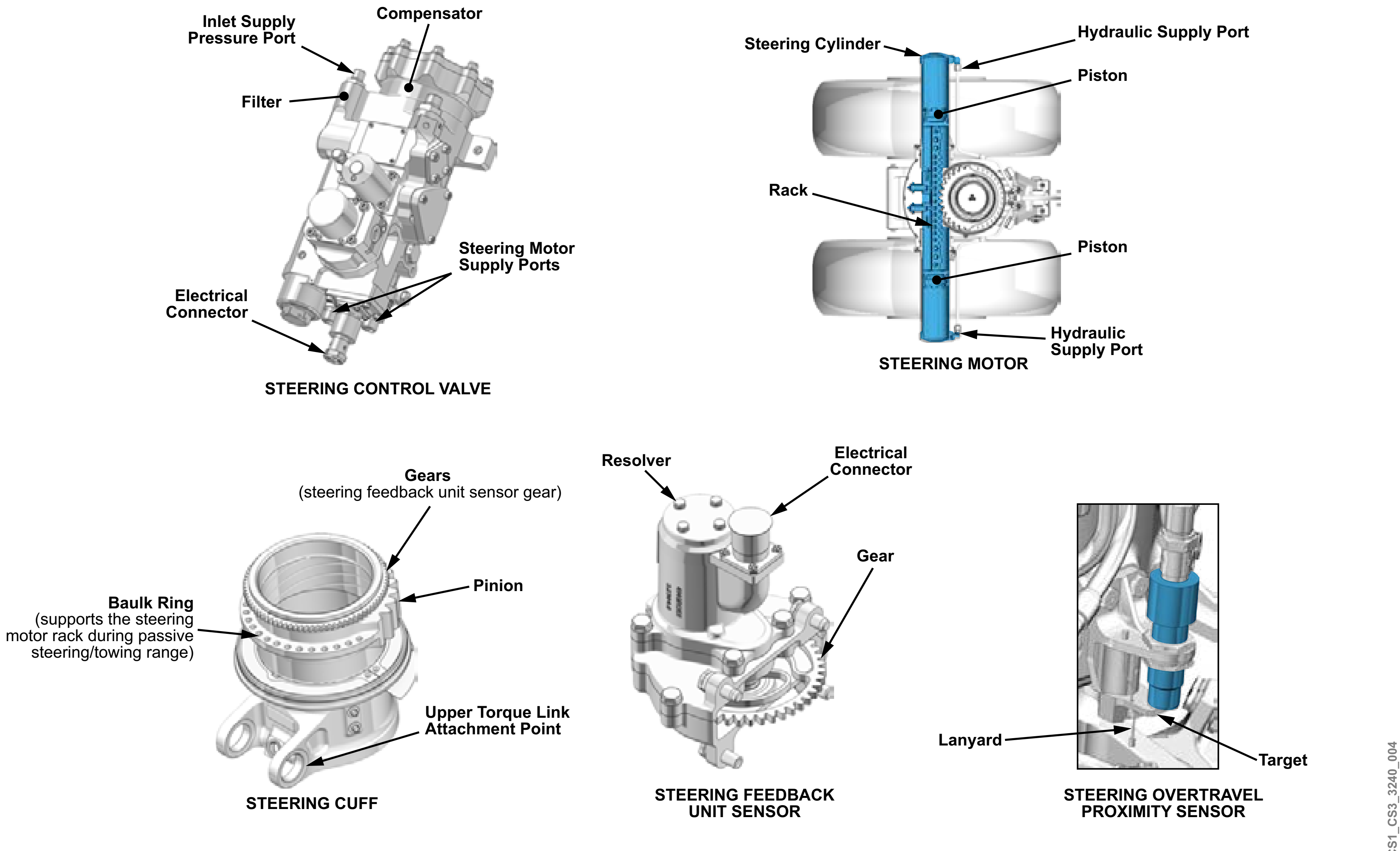


Figure 68: Nosewheel Steering System Components

DETAILED COMPONENT INFORMATION

STEERING MOTOR

The steering motor consists of two opposing steering cylinders and pistons, which drive a motor rack. The motor rack gear teeth mesh with the teeth of the steering cuff pinion gear. A left/right deflection of the motor rack results in rotation of the pinion gear. This rotates the steering cuff, which is connected to the upper torque link. The rotation is translated through the upper torque link to the lower torque link, turning the nosewheels in the respective direction.

The steering motor is mounted on the forward face of the NLG main fitting. Hydraulic lines mounted on the main fitting are routed to each end of the steering motor.

The motor rack has a gear disengagement feature which allows a towing angle that exceeds the steering limits. The disengagement occurs when the steering cuff has rotated past the last gear tooth on the pinion. A recess on the steering cuff allows the steering motor teeth to disengage, causing the rack to ride on the baulk fitting on the steering cuff. This allows a towing deflection angle of approximately ±130° without disconnecting the torque links. Two grease fittings are located on the steering motor.

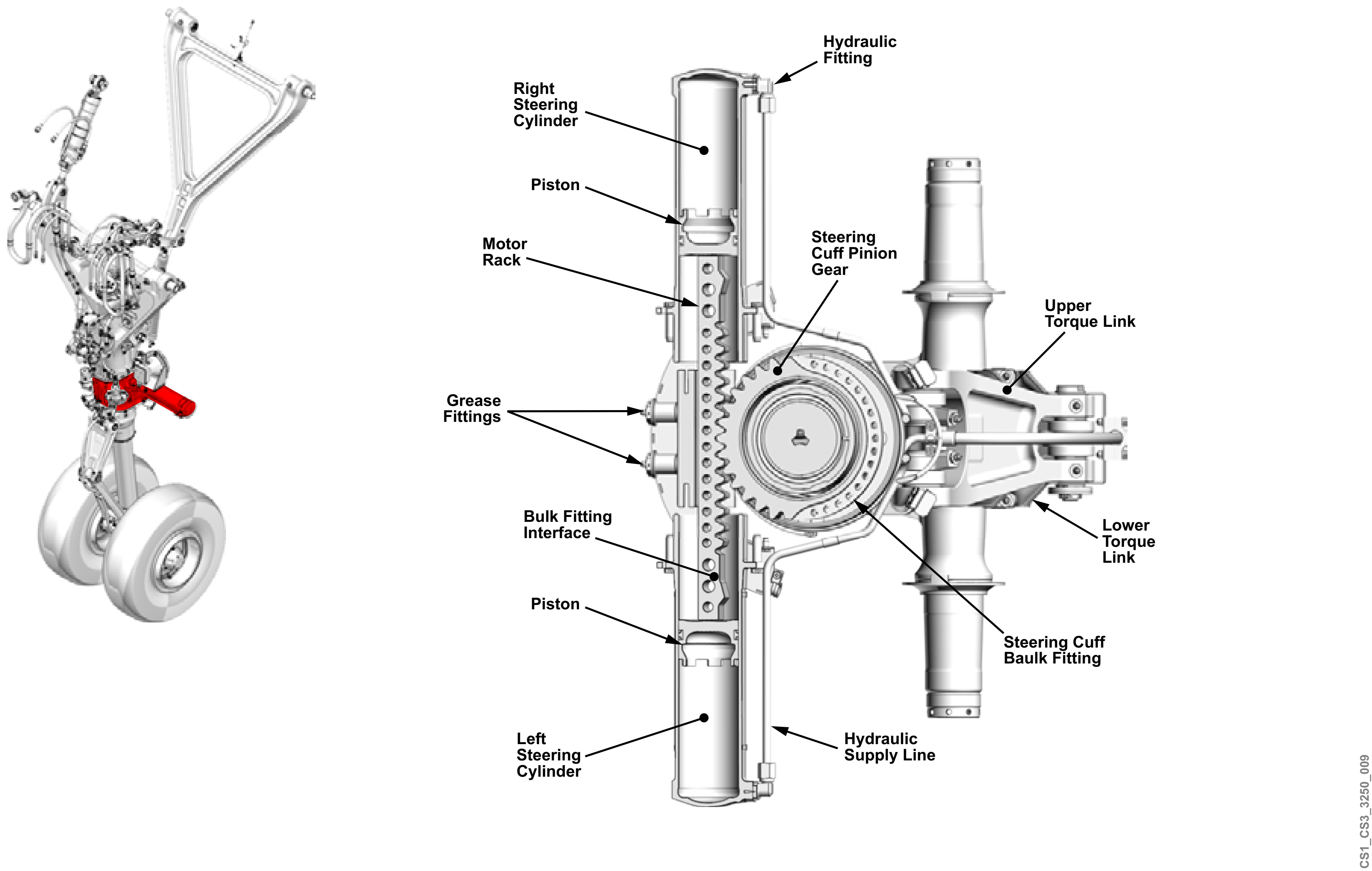


Figure 69: Steering Motor

STEERING CONTROL VALVE OPERATION - NOSE STEERING OFF

When nosewheel steering is commanded off, the EHSV is de-energized and returns to the neutral position. The enable valve solenoid is also de-energized, which closes the spring-loaded bypass valve.

With hydraulic pressure at the pressure port, the supply pressure check valve (SPCV) opens, opening the mechanically-linked compensator pressure check valve (CPCV). With the enable valve closed, the bypass valve also closes, assisted by a spring. This routes hydraulic pressure to both sides of the steering motor providing shimmy damping. Pressure in the SCV is regulated by the compensator regulating valve.

If no hydraulic pressure is available, the SPCV and the CPCV close. The compensator fluid inside the SCV and the steering motor are isolated from the aircraft hydraulic supply and return lines. The compensator maintains back pressure to provide damping. The anticavitation valves hold pressure within the actuator to prevent cavitation when no hydraulic pressure is available.

Each steering motor supply line has a pressure-relief valve (PRV) connected to the opposite pressure supply line of the steering motor. When the aircraft is towed, the rack and pinion gear moves the steering actuator. As the pressure builds in the actuator, the PRV opens, porting fluid to the opposite side of the actuator.

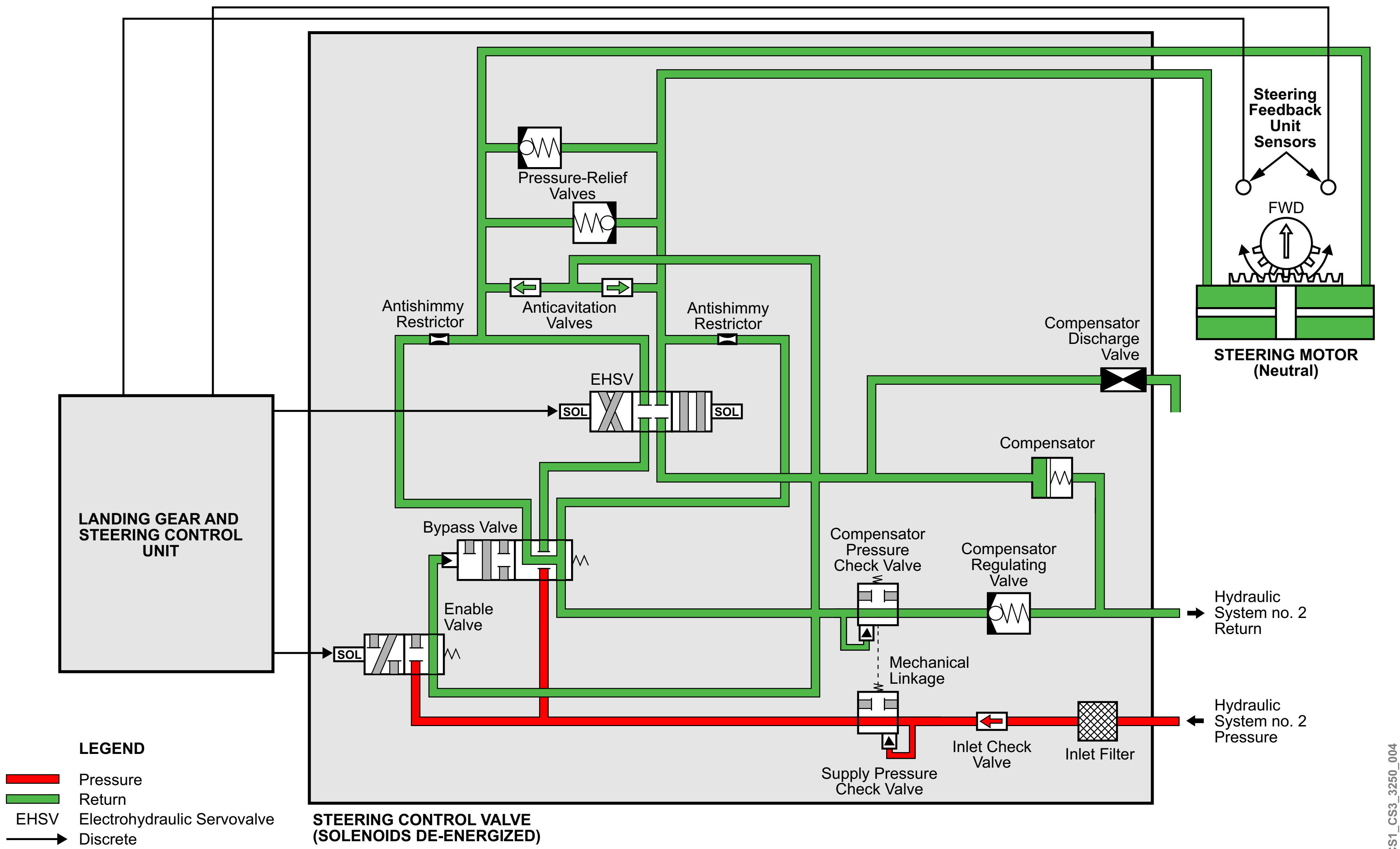


Figure 70: Steering Control Valve Operation - Nose Steering OFF

STEERING CONTROL VALVE OPERATION - NOSE STEERING ON

On power-up, the active LGSCU signals the SCV to apply hydraulic power to the steering motor. The SCV controls the steering motor position through the electrohydraulic servovalve (EHSV), and the steering feedback unit sensors.

When hydraulic pressure enters the valve, it passes through the inlet check valve and the supply pressure check valve. The supply pressure check valve is mechanically connected to the compensator pressure check valve. This ensures the two valves open and close together. When the NWS is ON, the LGSCU energizes the enable valve solenoid to open, sending pressure to the bypass valve. The bypass valve is hydraulically piloted to provide pressure to the EHSV.

Depending on the steering command, the EHSV directs hydraulic pressure through one of the two ports in the SCV to the steering motor.

The compensator maintains fluid supply within the SCV in the event of abrupt fluid transfer inside the valve. A compensator discharge valve is used to manually deplete fluid pressure in the SCV during maintenance.

Flow from the steering motor is routed back through the EHSV, through the compensator pressure check valve, compensator regulating valve, and then to the hydraulic return line.

If the hydraulic pressure is lost, the SPCV and CPCV close. The anticavitation valves maintain fluid in the actuators to prevent cavitation.

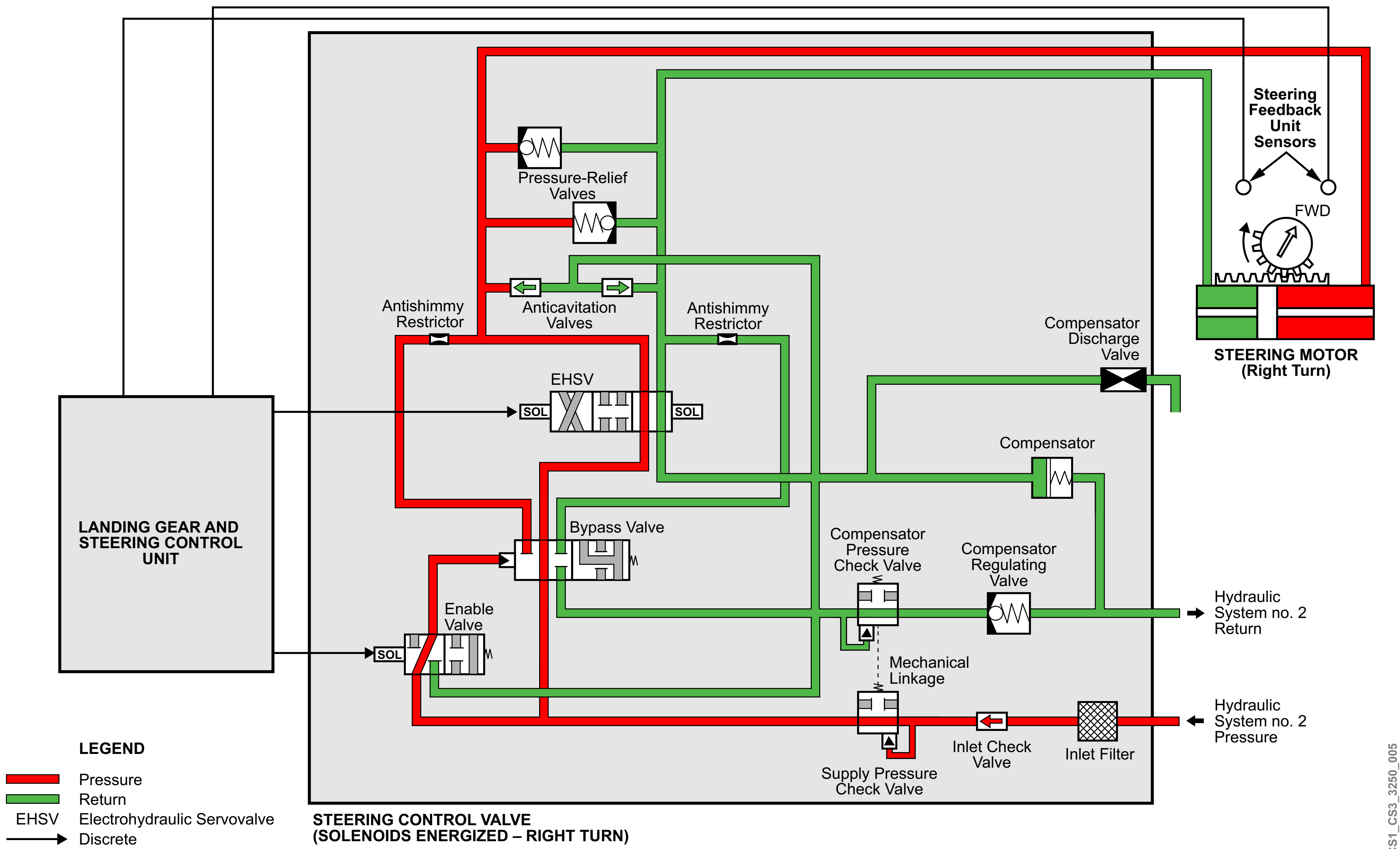


Figure 71: Steering Control Valve Operation - Nose Steering ON

CONTROLS AND INDICATIONS

STEERING TILLER

Rotation of the tiller steering affects four internal resolvers, which send electrical signals to the landing gear and steering control unit (LGSCU).

The steering tiller has ± 80° rotation with internal stops. The steering tiller centers automatically by a balanced spring and the movement is damped to avoid rapid movement. A PEDAL DISC pushbutton, located in the center of the handwheel, disables rudder pedal steering inputs when depressed and held.

RUDDER PEDAL ASSEMBLIES

Each rudder pedal assembly contains rudder pedal position rotary variable differential transformers (RVDTs). The RVDTs transmit rudder pedal position commands to the primary flight control computers (PFCCs). The PFCCs relay this information as steering commands to the LGSCU.

NOSE STEER PBA

The NOSE STEER PBA is located on the landing gear panel. In the default on position, the PBA sends an electrical signal to the LGSCU to activate the nosewheel steering system. Pressing the PBA turns the nosewheel steering system off, and illuminates the OFF light in the PBA.

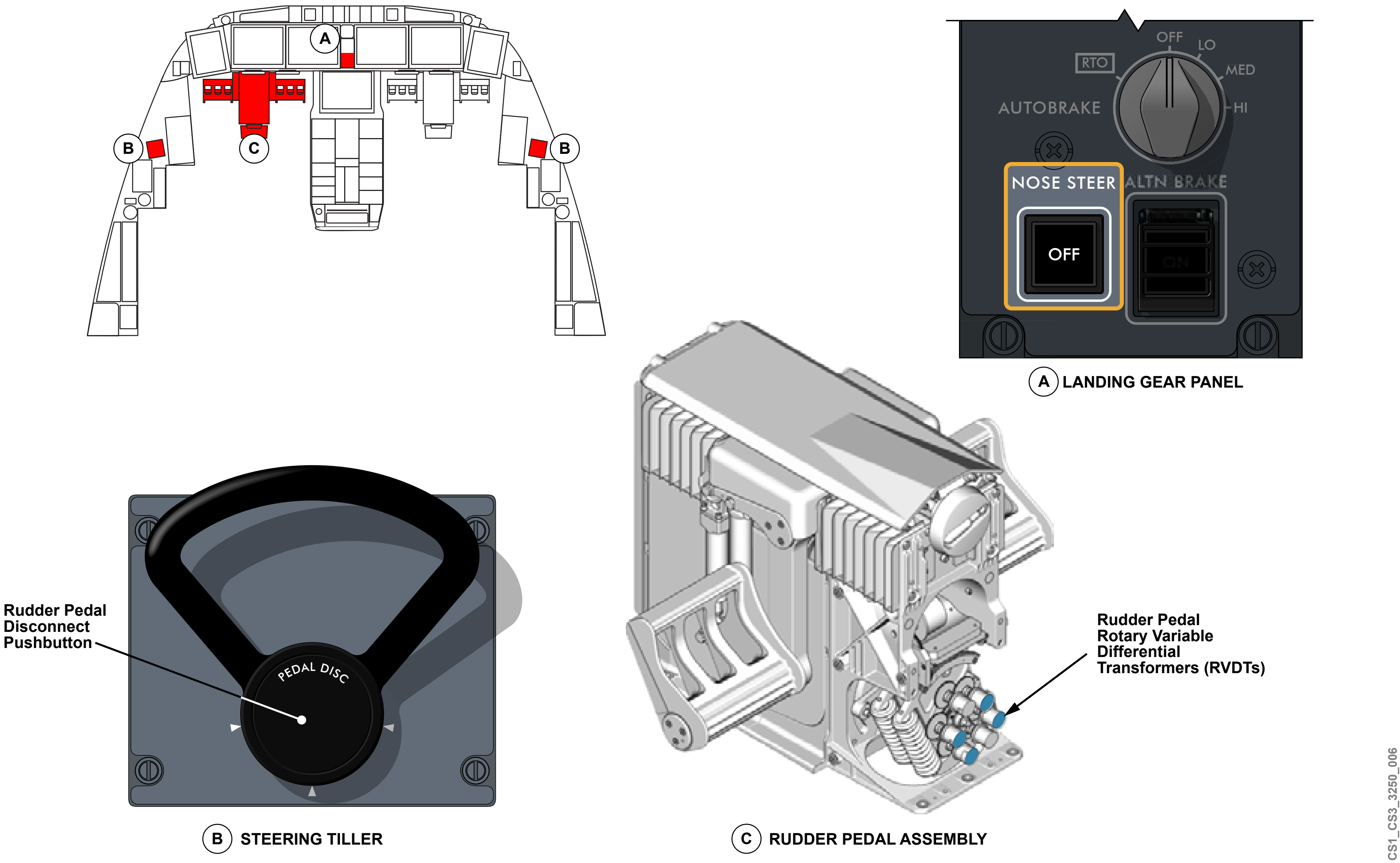


Figure 72: Nosewheel Steering System Controls and Indications

DETAILED DESCRIPTION

At electrical power-up of the aircraft, the two LGSCUs are powered. The hydraulic steering control valve (SCV) is pressurized as soon as hydraulic system no. 2 pressure is available.

The NWS system is enabled when the following conditions are met:

- NOSE STEER PBA selected ON
- NLG down and locked
- Weight-on-wheels (WOW)

Steering inputs are processed in the steering control path of each LGSCU. The steering control paths communicate with each other, and operate in active/standby configuration. Only one steering control path is active at a time. The paths switch between active and standby at each aircraft power-up.

Steering inputs are received from:

- Pilot steering tiller
- Copilot steering tiller (optional)
- Rudder pedals
- Active PFCC

The steering control path compares the steering tiller, rudder pedal, or active PFCC inputs with the steering feedback unit sensor position. If the steering position differs from the input, the SCV directs hydraulic pressure to the respective port of the steering motor.

If both steering tiller and rudder pedal inputs are in the same direction, the greater of the two inputs is used as the steering command.

When the steering tiller and rudder pedal inputs are in the opposite direction, the inputs are arithmetically summed to provide a steering command.

If the optional copilot steering tiller is installed and steering inputs are received from both tillers in the same direction, the greater of the two inputs is used as the steering command.

If the steering tillers are operated in the opposite direction, the inputs are arithmetically summed to provide a steering command.

During an autoland rollout, the steering control path receives the nosewheel steering (NWS) commands from the active PFCC to track the instrument landing system (ILS) localizer.

If a steering control path fails, the active LGSCU generates a NOSE STEER FAULT advisory message and de-energizes the SCV. The standby LGSCU becomes active and provides steering commands to the SCV.

If both LGSCU steering control paths fail, the SCV is de-energized, and a NOSE STEER FAIL caution message is displayed on EICAS.

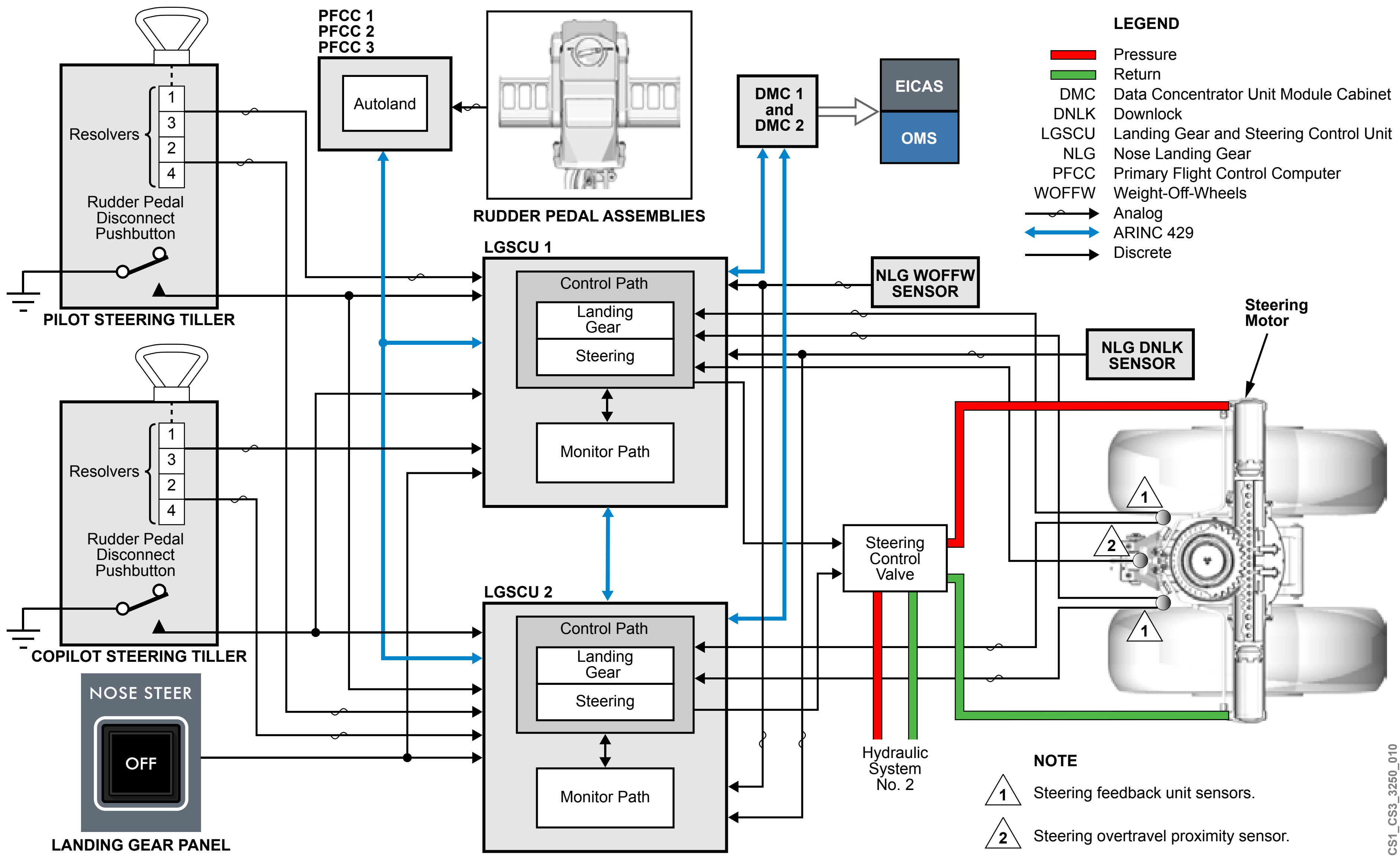


Figure 73: Nosewheel Steering Detailed Description

MONITORING AND TESTS

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

CAS MESSAGES

Table 14: CAUTION Messages

MESSAGE	LOGIC
NOSE STEER FAIL	Nosewheel steering system failed or no communication is available from both LGSCUs.
NOSE STEER MISALIGN	Nosewheel angle is beyond active steering range.

Table 15: ADVISORY Message

MESSAGE	LOGIC
NOSE STEER FAULT	Loss of redundancy of nosewheel steering system, or reduced functionality. Refer to INFO messages.

Table 16: STATUS Messages

MESSAGE	LOGIC
PEDAL STEER DISC	Rudder pedal disconnect is active.
NOSE STEER OFF	Steering disabled (castor mode) via action on the nose steer off switch.

Table 17: INFO Messages

MESSAGE	LOGIC
32 NOSE STEER FAULT - L TILLER INOP	Both LGSCUs have detected a left tiller fault.
32 NOSE STEER FAULT - R TILLER INOP	Both LGSCUs have detected a right tiller fault.
32 NOSE STEER FAULT - OVERTRAVEL DET INOP	Either LGSCU1 has detected its steering overtravel proximity sensor is not operational, or LGSCU1 is not operative.
32 NOSE STEER FAULT - TILLER DEGRADED	The active LGSCU does not receive wheel speed data:
32 NOSE STEER FAULT - STEER REDUND LOSS	One LGSCU detected an internal redundancy fault, or no/invalid data from one LGSCU to avionic.
32 NOSE STEER FAULT - LGSCU INTNL FAULT	One LGSCU detected a critical internal fault and requires complete isolation of the affected LGSCU (ie. CBs pulled, LGSCU removal).

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM

The onboard maintenance system (OMS) provides the capability to perform the following:

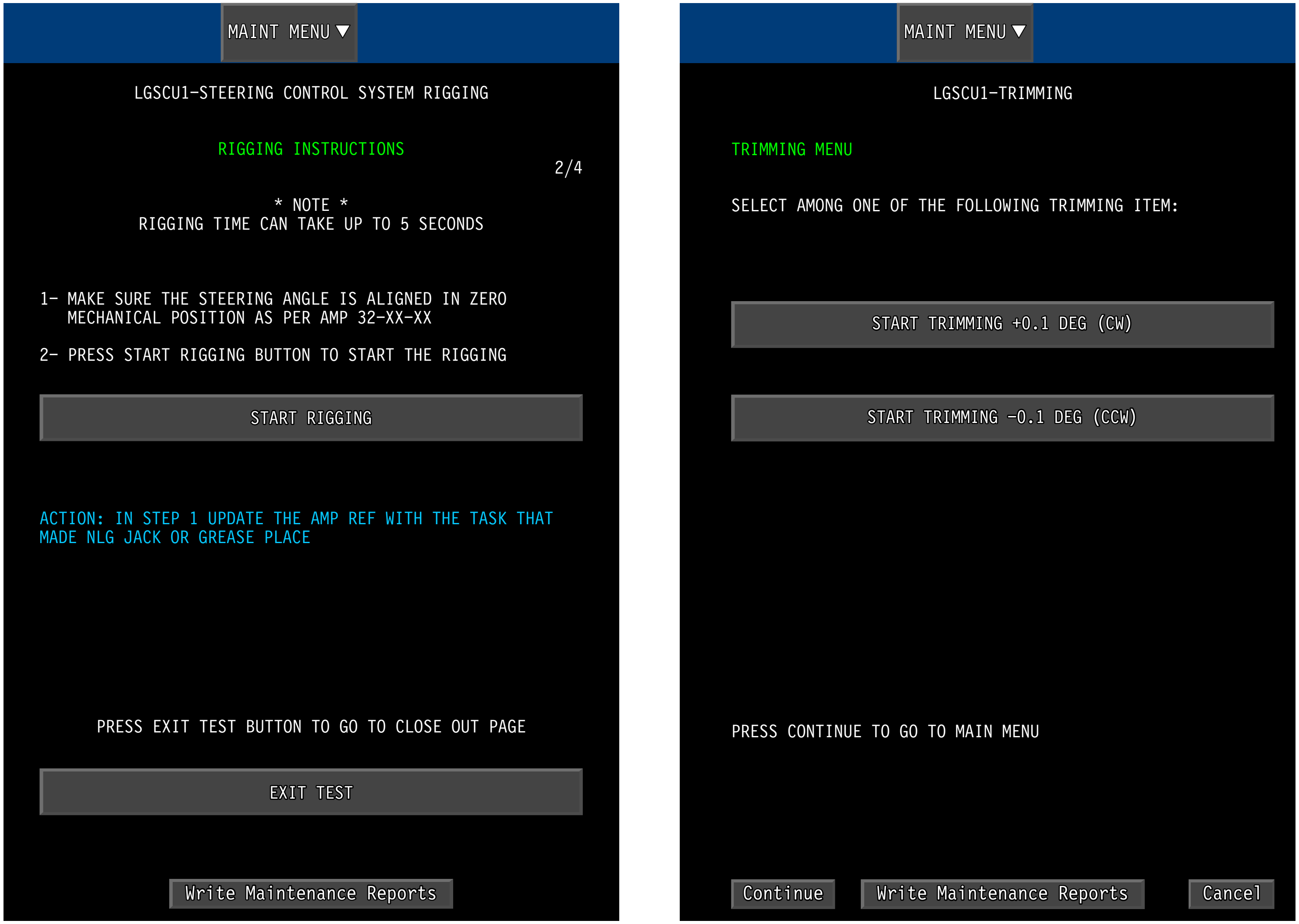
- Steering control system rigging
- Steering trimming

Steering Control System Rigging

The steering control system rigging sets the electrical zero for the steering feedback unit sensors when the nose gear is centered.

Steering Trimming

The steering trimming function provides the capability to trim the nosewheel steering if the aircraft has a tendency to pull to one side during taxi.



CS1_CS3_3251_001

Figure 74: Nosewheel Steering OMS

NOSEWHEEL STEERING OVERTRAVEL

The steering overtravel proximity target breaks off at the shear section when the sensor target impacts the nose gear structure, (approximately ±135° of center), indicating that the maximum allowable steering angle has been exceeded. The lanyard retains the broken piece. The target assembly must be replaced, and the nosewheel steering mechanism inspected in accordance with the Aircraft Maintenance Publication (AMP).

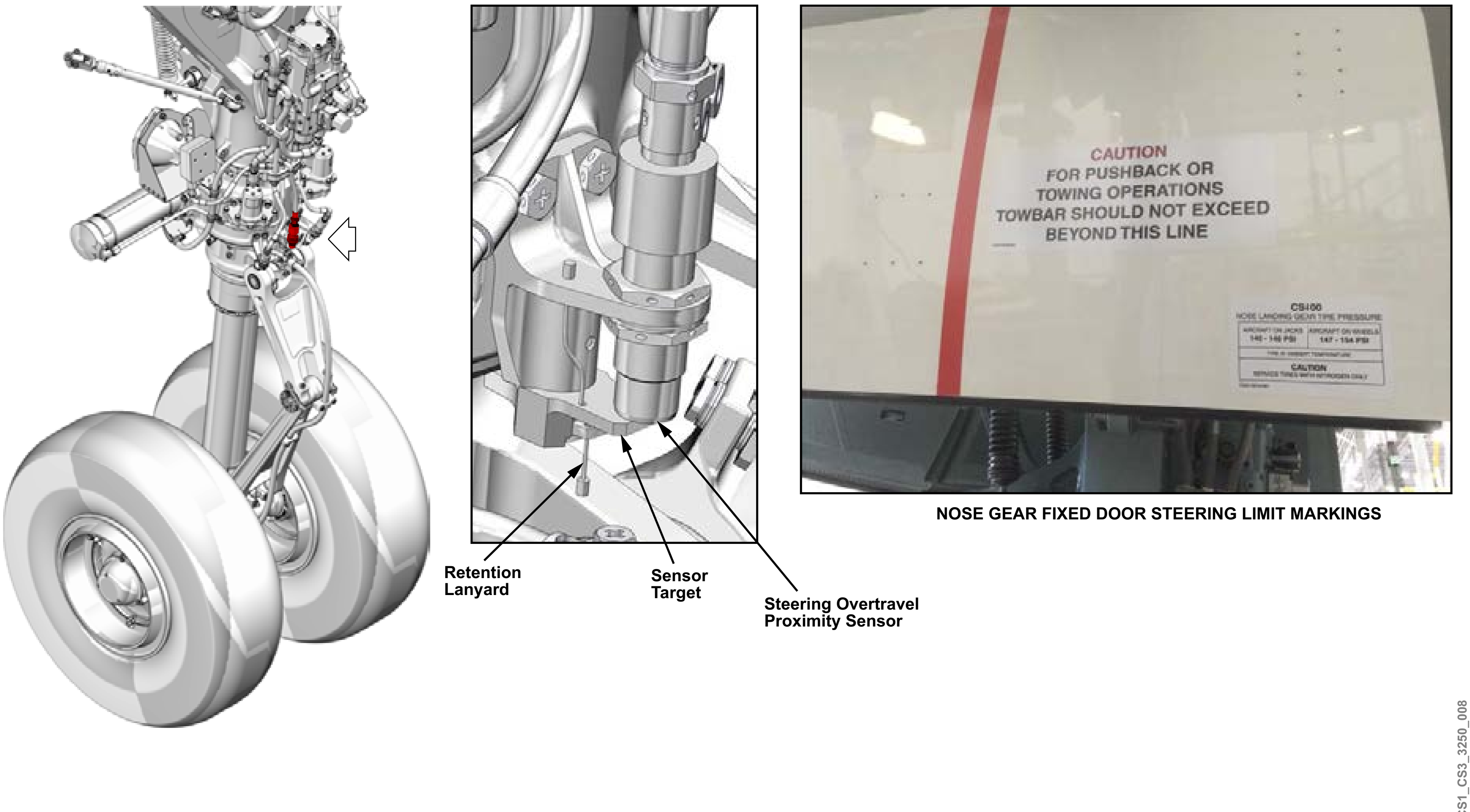


Figure 75: Nosewheel Steering Overtravel

COMPENSATOR DISCHARGE VALVE

A compensator discharge valve is used to manually deplete fluid pressure in the steering control valve (SCV) during maintenance.

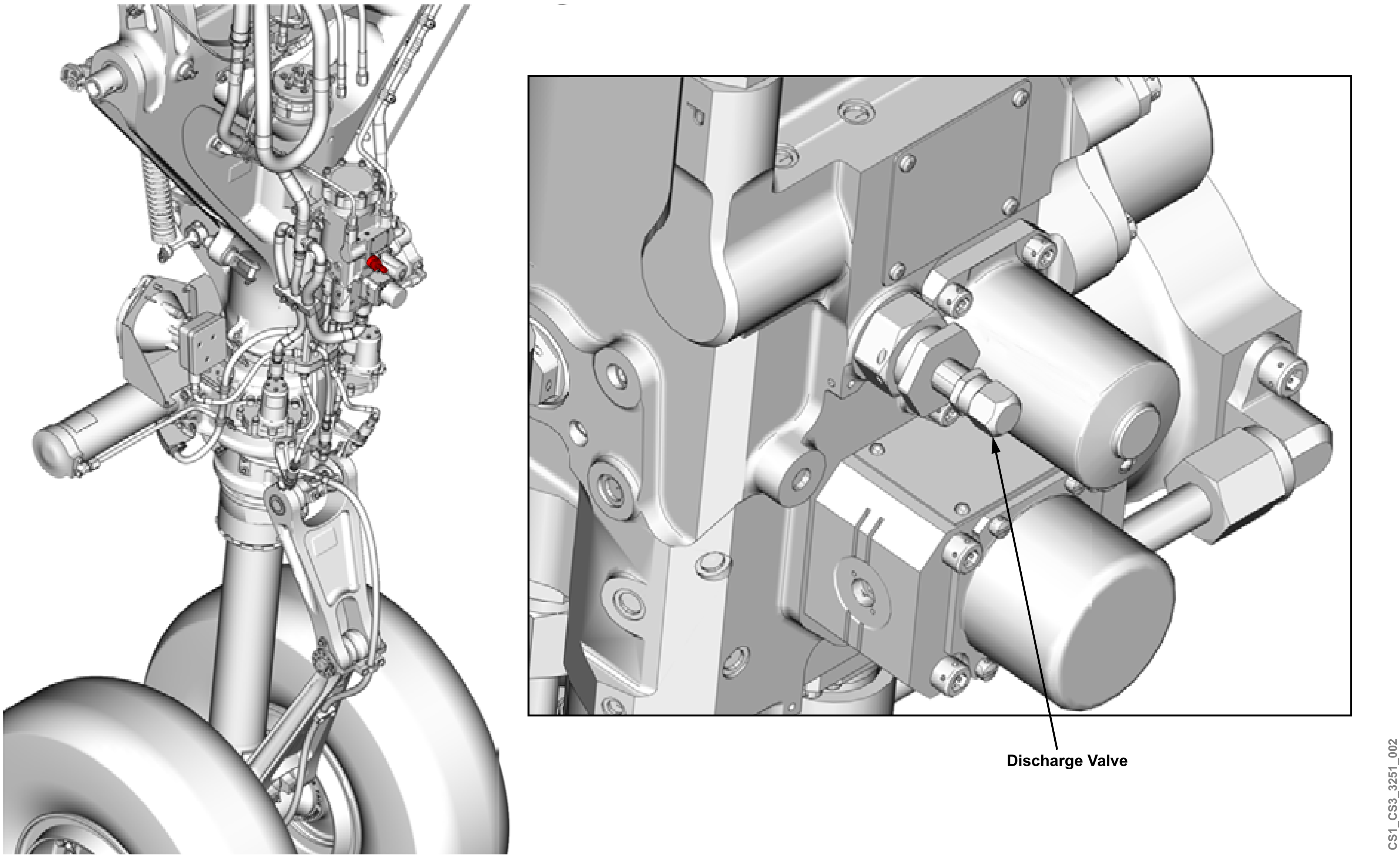


Figure 76: Compensator Discharge Valve

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ATA 35 - Oxygen

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

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OXYGEN - CHAPTER BREAKDOWN

35-10 CREW OXYGEN SYSTEM

GENERAL DESCRIPTION

Oxygen for the flight crew members is stored at a nominal pressure of 1850 psi in a 2.18 cu. m (77 cu. ft) cylinder. The regulator supplies oxygen at a reduced pressure to the crew mask stowage boxes. The regulator toggle lever is used to shut off the supply of oxygen to the crew masks. A low-pressure switch monitors the supply to the crew masks.

A pressure/temperature transducer monitors the cylinder pressure through a high-pressure sense line and provides temperature compensated pressure readings to the engine indication and crew alerting system (EICAS). The pressure/temperature transducer is powered by DC ESS BUS 1.

When an overpressure in the oxygen bottle or low-pressure oxygen line occurs, oxygen vents through the overboard discharge indicator (ODI).

The crew oxygen system is replenished through an oxygen ground service panel located on the forward left nose.

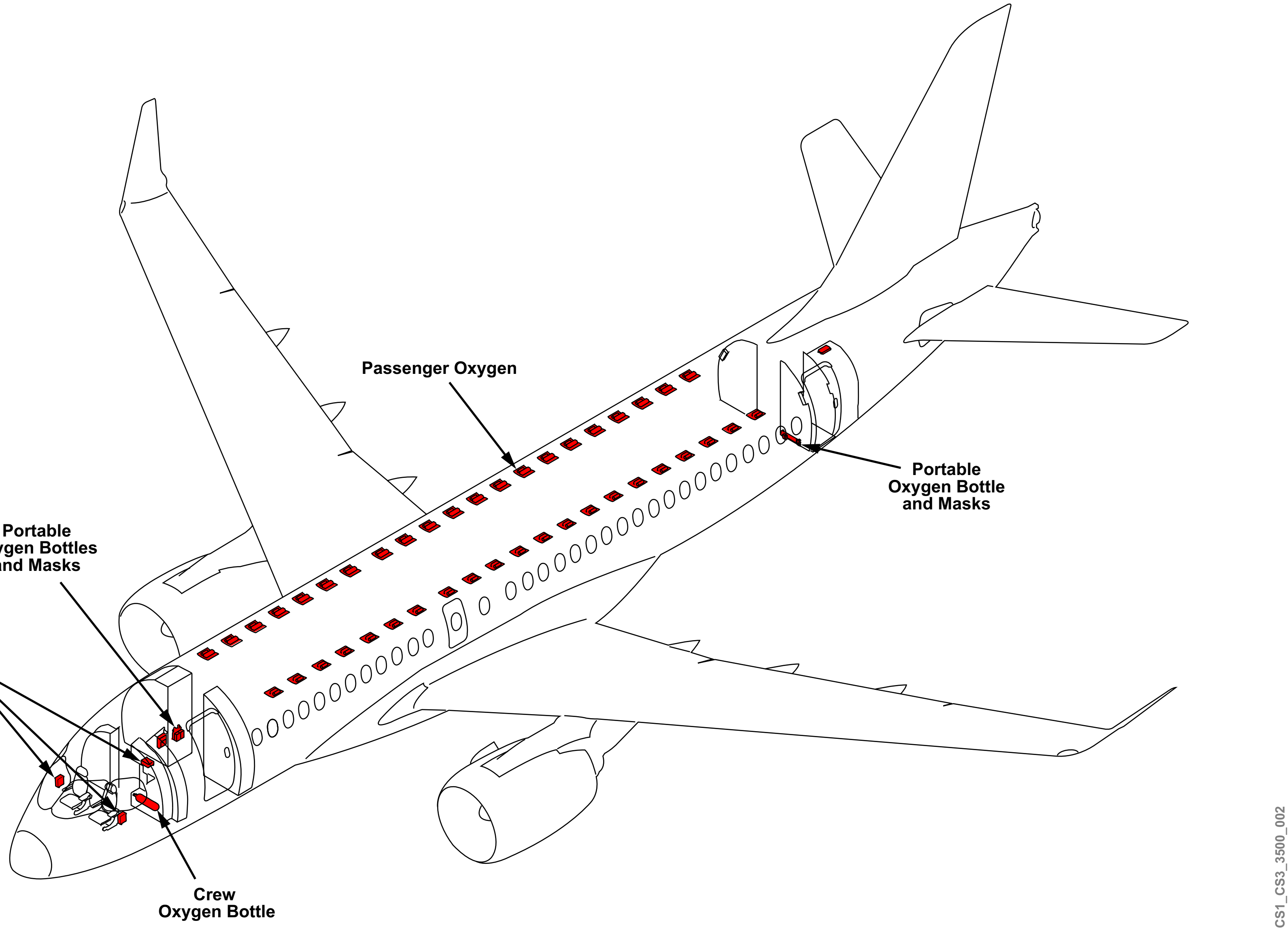


Figure 1: Crew Oxygen

COMPONENT LOCATION

The major components of the crew oxygen system include:

- Oxygen cylinder and regulator assembly
- Crew mask stowage boxes and crew masks
- Pressure/temperature transducer
- Low-pressure switch
- Overboard discharge indicator

OXYGEN CYLINDER

The oxygen cylinder is secured to the aircraft structure with clamps in a dedicated compartment below the 3rd crew member seat in the flight deck.

CREW MASK STOWAGE BOX AND CREW MASK

Three mask stowage boxes are installed in the flight deck, one each on the left and right side consoles, and a third on the aft left wall. Each box contains the crew full face mask.

PRESSURE/TEMPERATURE TRANSDUCER

The pressure/temperature transducer and pressure switch are located in the oxygen bottle compartment. The pressure and temperature transducer is connected to a high-pressure port on the regulator assembly.

LOW-PRESSURE SWITCH

The low-pressure switch is located in the mask supply line downstream of the bottle.

OVERBOARD DISCHARGE INDICATOR

The overboard discharge indicator (ODI) is installed on the lower left side of the fuselage.

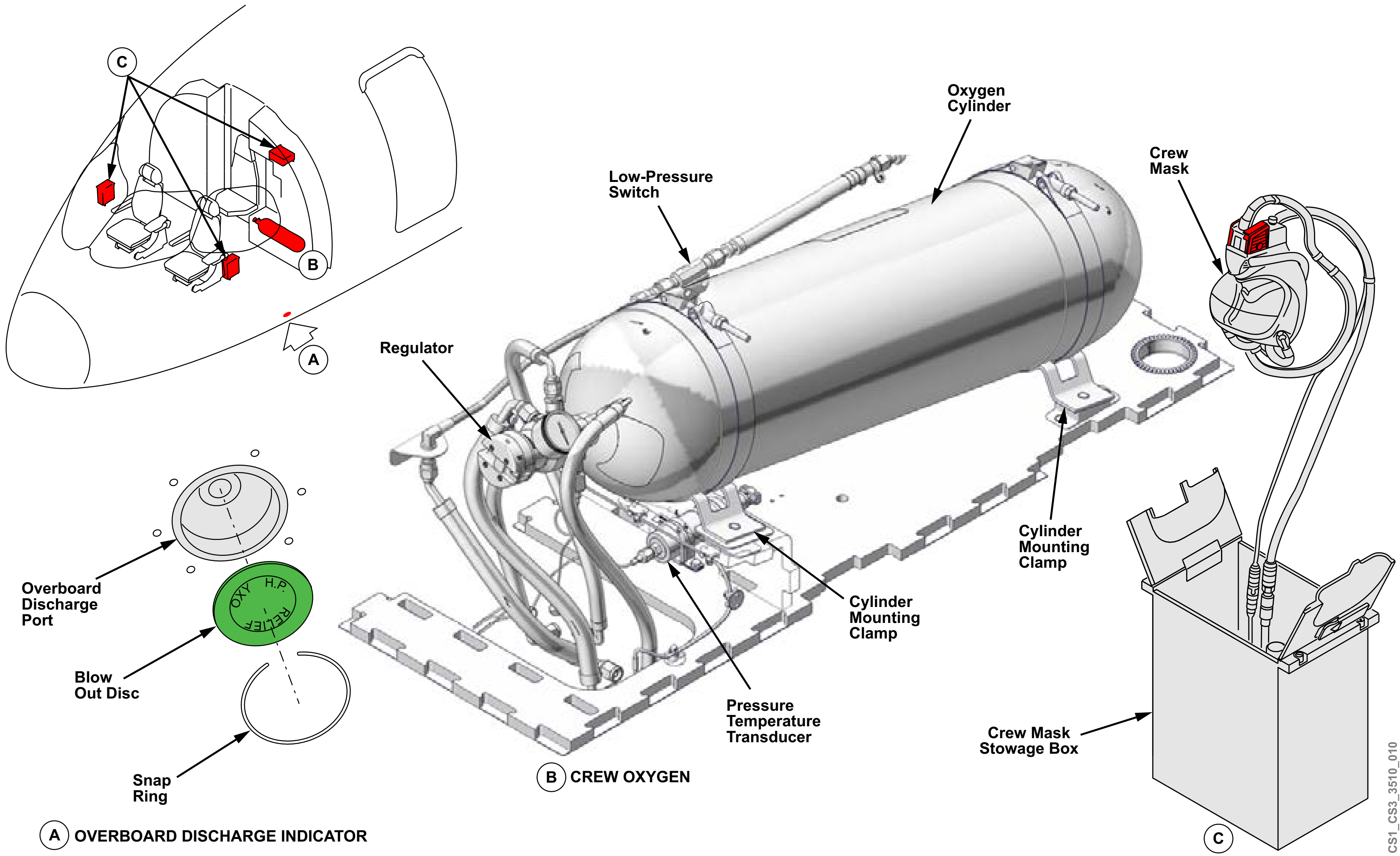


Figure 2: Crew Oxygen Component Location

DETAILED COMPONENT INFORMATION

CREW OXYGEN CYLINDER

The oxygen cylinder is constructed of a light-weight seamless aluminum sleeve that is overwrapped with Kevlar fiber. The cylinder is installed on an angle for easy removal. The compartment is ventilated to prevent the accumulation of high concentrations of oxygen from any minor leaks.

The regulator is attached to the cylinder. The regulator is a single inlet, self-venting device that has three high-pressure connections, three low-pressure connections, and a direct reading pressure gauge to indicate the bottle pressure. The regulator reduces high cylinder pressure to a lower distribution pressure of 78 psig for the crew oxygen masks.

The ON/OFF toggle lever controls the flow of oxygen into the distribution line for the crew masks. The lever must be secured to the ON position.

An integral low-pressure relief valve prevents pressure buildup in the low pressure tube by relieving excess pressure through the relief valve. In the event the pressure rises above 94 psig, the valve opens to vent the excess pressure through the overboard discharge indicator and then resets.

The high-pressure relief provides protection when an overpressure of 2500 psig occurs in the oxygen cylinder, due to an overfill or a temperature rise. When the pressure exceeds a safe limit, a frangible disk ruptures to vent the bottle contents through the overboard discharge indicator.

A refill line connects the bottle to the ground service panel.

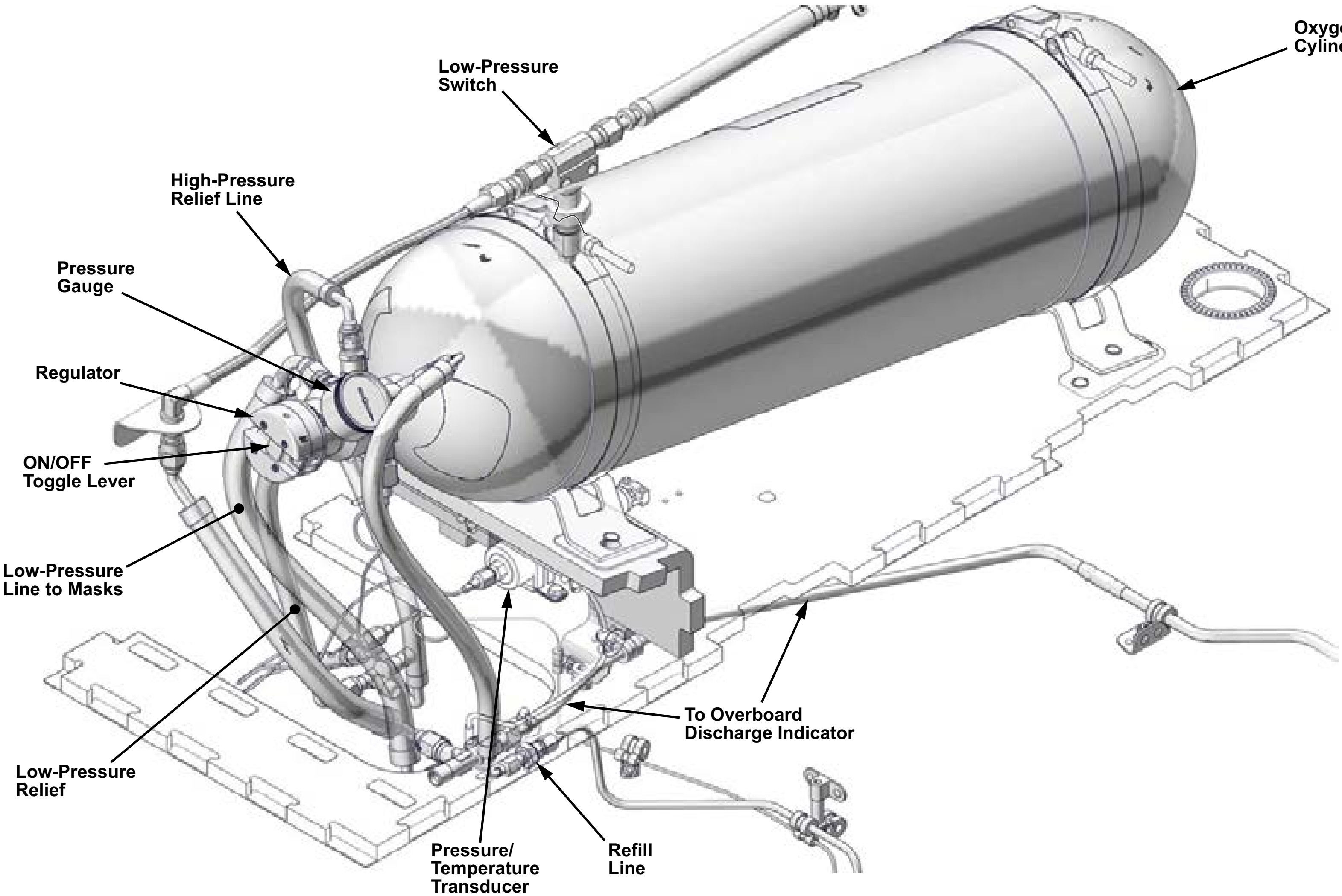


Figure 3: Crew Oxygen Cylinder

CREW MASK STOWAGE BOX

There are three mask stowage boxes. Each mask stowage box has oxygen and microphone connection, a pneumatic assembly, a flow indicator, and a PRESS TO TEST AND RESET lever.

The mask connects to the supply hose using a bayonet type coupling with a spring-loaded check valve. The valve is closed whenever the mask is disconnected from the supply hose.

CREW MASK

Each full face, quick-donning crew mask includes a regulator, a pneumatically controlled inflatable harness, oxygen hose, and a microphone for communication purposes.

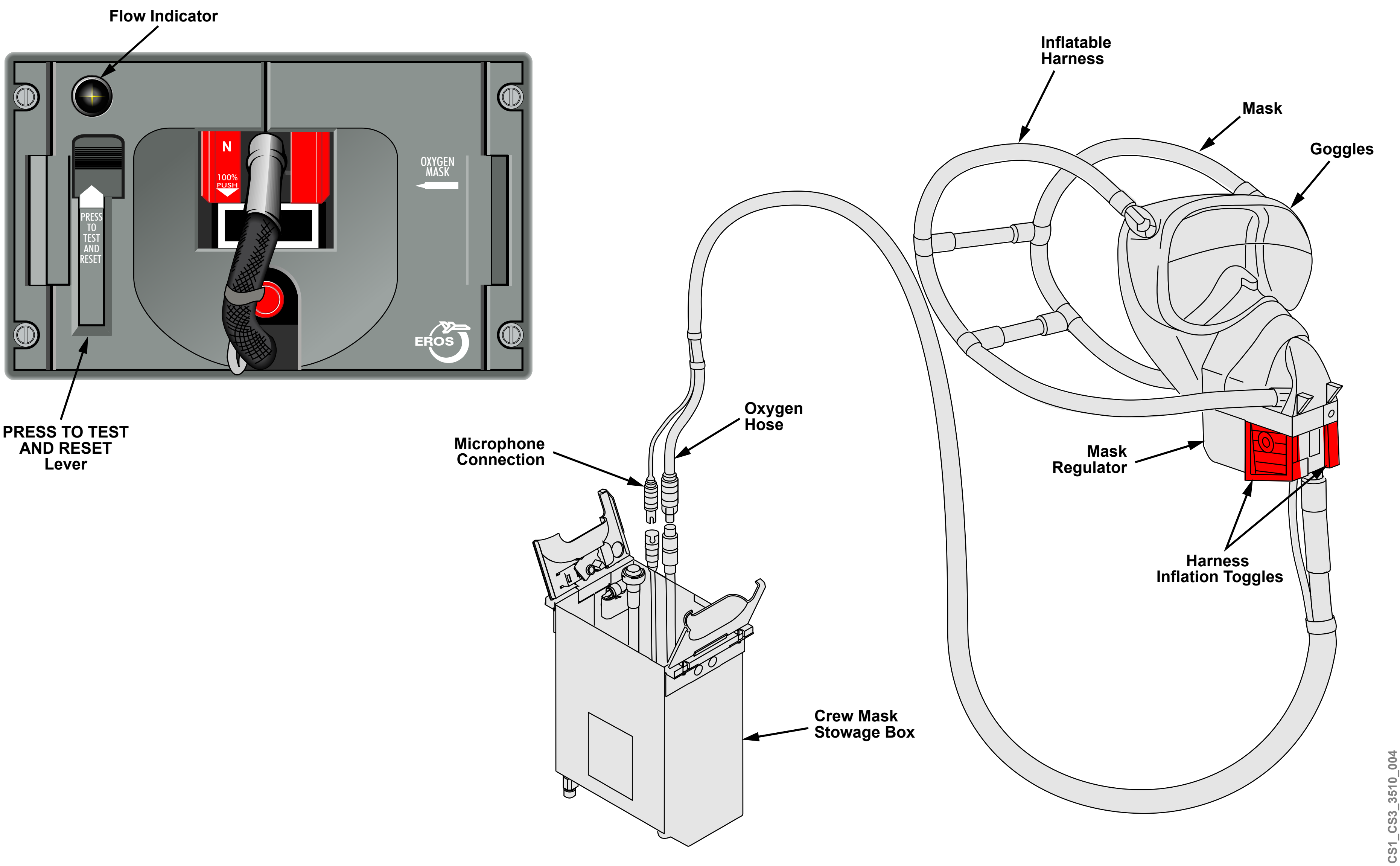


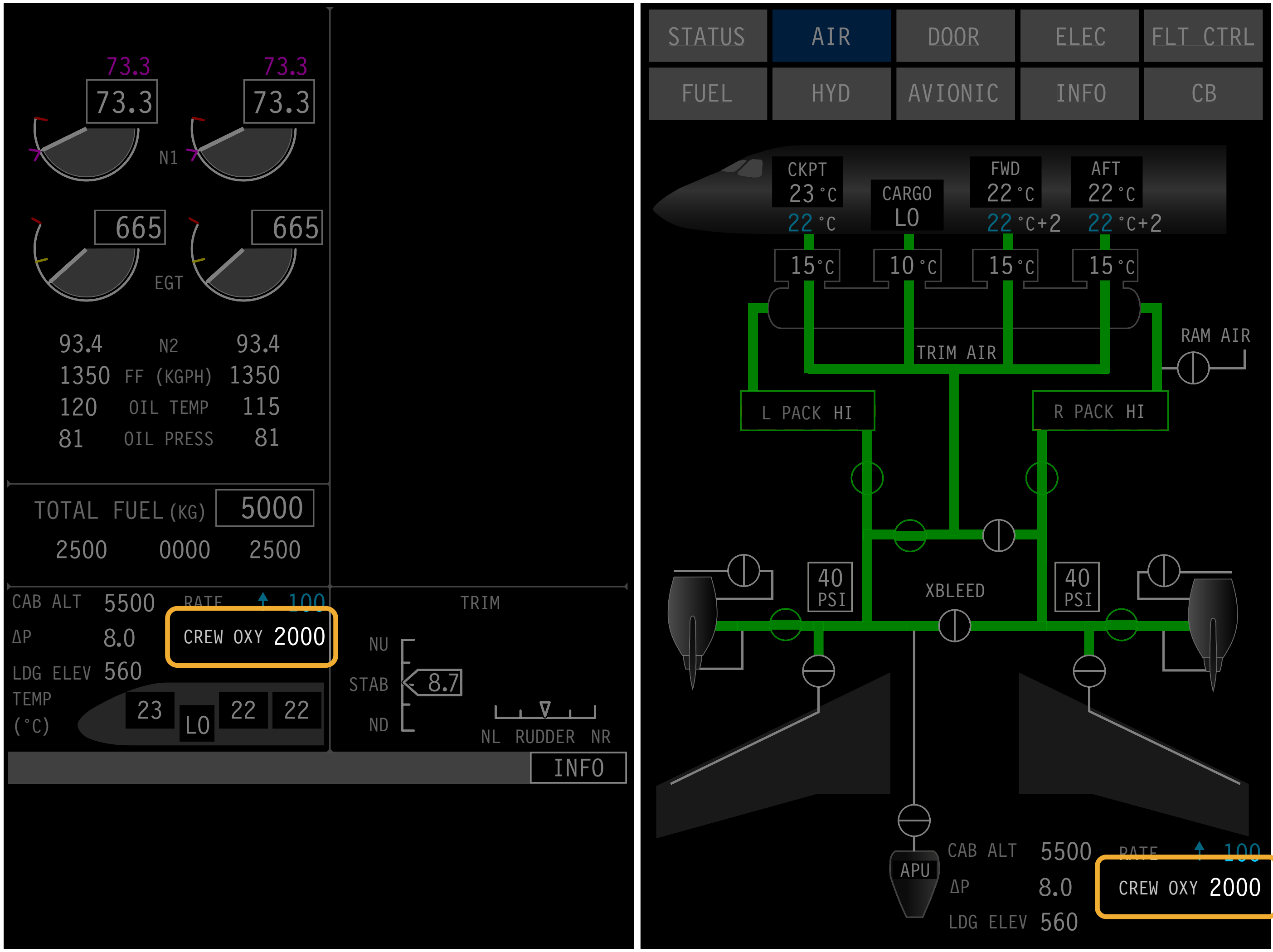
Figure 4: Crew Mask Stowage Box and Crew Mask

CONTROLS AND INDICATIONS

EICAS AND SYNOPTIC PAGE INDICATIONS

The EICAS page and AIR synoptic page provide a digital readout of the oxygen cylinder pressure. Amber dashes indicate a failure of the pressure and temperature transducer or the loss of a valid pressure signal.

When the system is within normal operating pressure, the digital readout is white, and changes to amber when the pressure drops below 1,050 psi.



EICAS

AIR SYNOPTIC PAGE – AIR

Value	Color	Description
900	Amber	Crew oxygen quantity pressure below caution threshold.
2000	White	Crew oxygen quantity pressure within normal range.
----	Amber	Invalid value.

CS1_CS3_3510_005

Figure 5: Crew Oxygen Indication

OPERATION

Oxygen flow to the mask is indicated by a blinker on the pneumatic assembly. When oxygen is flowing, the blinker indicates a yellow cross. When there is no flow, the blinker remains black.

When the mask is removed and oxygen is supplied to the mask, an OXY ON flag comes into view from behind the left door. When the mask is restowed, the OXY ON flag is reset by pressing the PRESS TO TEST AND RESET lever. The PRESS TO TEST AND RESET lever also verifies the operation of a stowed crew mask when pressed momentarily.

Oxygen flow to the mask is controlled by the N/100% selector and the emergency flow control knob. When the N/100% selector is set to N, a mixture of oxygen and ambient air is supplied to the mask. In the 100% position, only oxygen is supplied to the mask. If the cabin altitude is greater than 10,688 m (35,000 ft), the mask supplies 100% oxygen regardless of the N/100% switch position.

The smoke protection function is tested with the N/100% selector to set 100%. When the flow control knob is set to emergency, there should be a slight overpressure in the mask and goggles.

The mask microphone function can be tested when the mask is worn. Select the flight interphone system (refer to ATA 23) and speak into the microphone while listening on the flight deck speaker.

CREW MASK TEST WHEN STOWED

To test the emergency flow, push down and hold the PRESS TO TEST AND RESET lever. Push on the PRESS TO TEST knob. The flow indicator shows a continuous flow.

The mask can be tested without removing it from the stowage box. To test the mask, push down and hold the PRESS TO TEST AND RESET lever. The flow indicator shows a momentary flow.

FULL CREW MASK TEST

When the toggle levers are squeezed, the mask stowage box doors open and the mask can be removed from the box. The stowage box shutoff valve opens when the left door is open and oxygen flows through the regulator and inflate the harness. With the harness inflated, the crew member can quickly place the mask over their head. Releasing the toggle levers causes the harness to deflate allowing the mask to form a tight fit. When the mask is removed and oxygen is available, the OXY ON flag is visible on the left door of the stowage box.

When the PRESS TO TEST knob is in the normal position, oxygen is supplied on demand. If the PRESS TO TEST knob is turned to the EMERGENCY position, the mask is supplied a constant flow of 100% oxygen at a positive pressure.

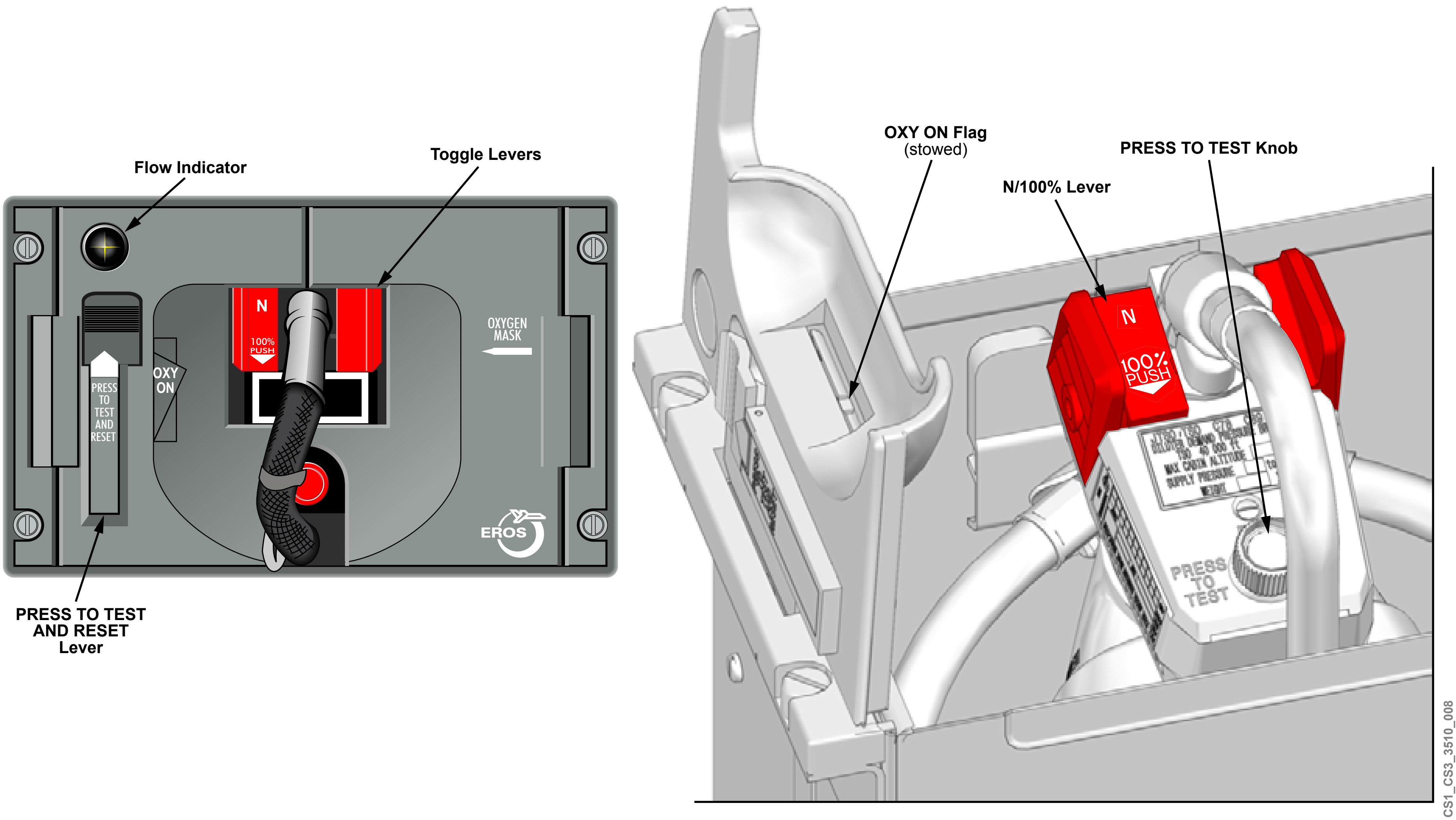
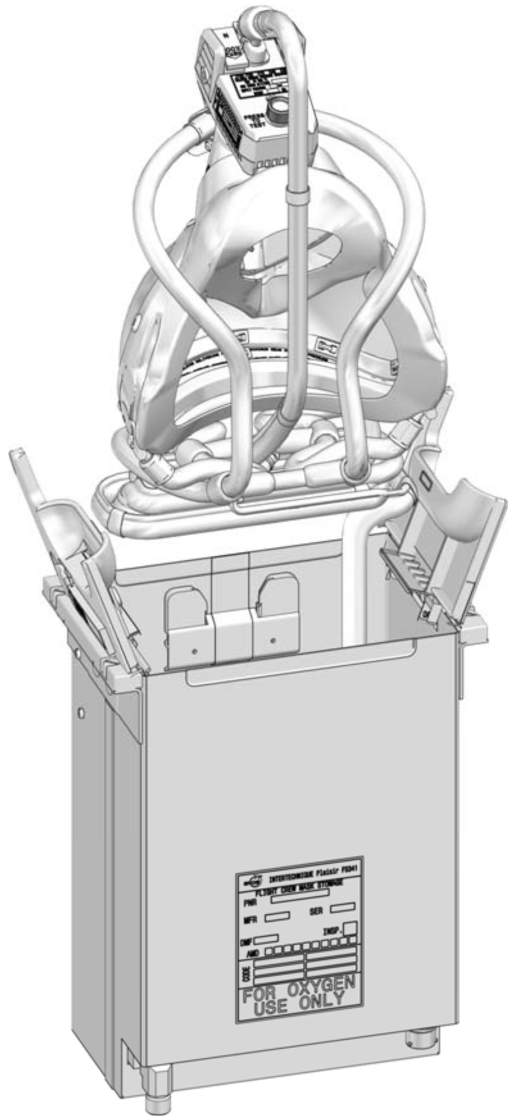


Figure 6: Crew Mask Testing

CREW MASK STOWAGE

To stow the mask, coil the oxygen hose into the bottom of the stowage box without turning the mask and fold the harness into the mask. The mask regulator sits against the STOP in the stowage box when correctly installed. Push the doors closed and engage the door pins in the mask toggle levers. Selecting the PRESS TO TEST AND RESET lever resets the OXY ON flag.



CS1_CS3_3510_011

Figure 7: Crew Mask Stowage

DETAILED DESCRIPTION

The oxygen bottle and supply lines are protected against high pressures. Both low-pressure (LP) and high-pressure (HP) safety relief outlets of the regulator are connected to the ODI through a common pipe. A green nylon disc labeled OXY. H.P. RELIEF is held in place over the discharge port by a snap ring. The disc ruptures at 80 ± 30 psig pressure, allowing oxygen to vent overboard. Small volumes of oxygen from the LP line can vent around the ODI disc without popping it. A steady flow will dislodge the disc, indicating a loss of oxygen.

The pressure and temperature transducer supplies pressure and temperature information to data concentrator unit module cabinet (DMC) 2. The pressure and temperature outputs are combined to present a temperature compensated pressure display in the flight deck. The transducer is also used to generate a CREW OXY LO PRESS caution message on EICAS if it detects low pressure in the high-pressure line.

The low-pressure switch sends a signal to DMC2 to generate a CREW OXY LO PRESS caution message if it detects low pressure in the low-pressure distribution line.

NOTE

> The CREW OXY LO PRESS caution message could be an indication that the regulator ON/OFF toggle switch has been left in the closed position.

A microswitch, in the crew mask stowage box, enables the crew mask microphone function in the audio control panel when oxygen is supplied to the mask.

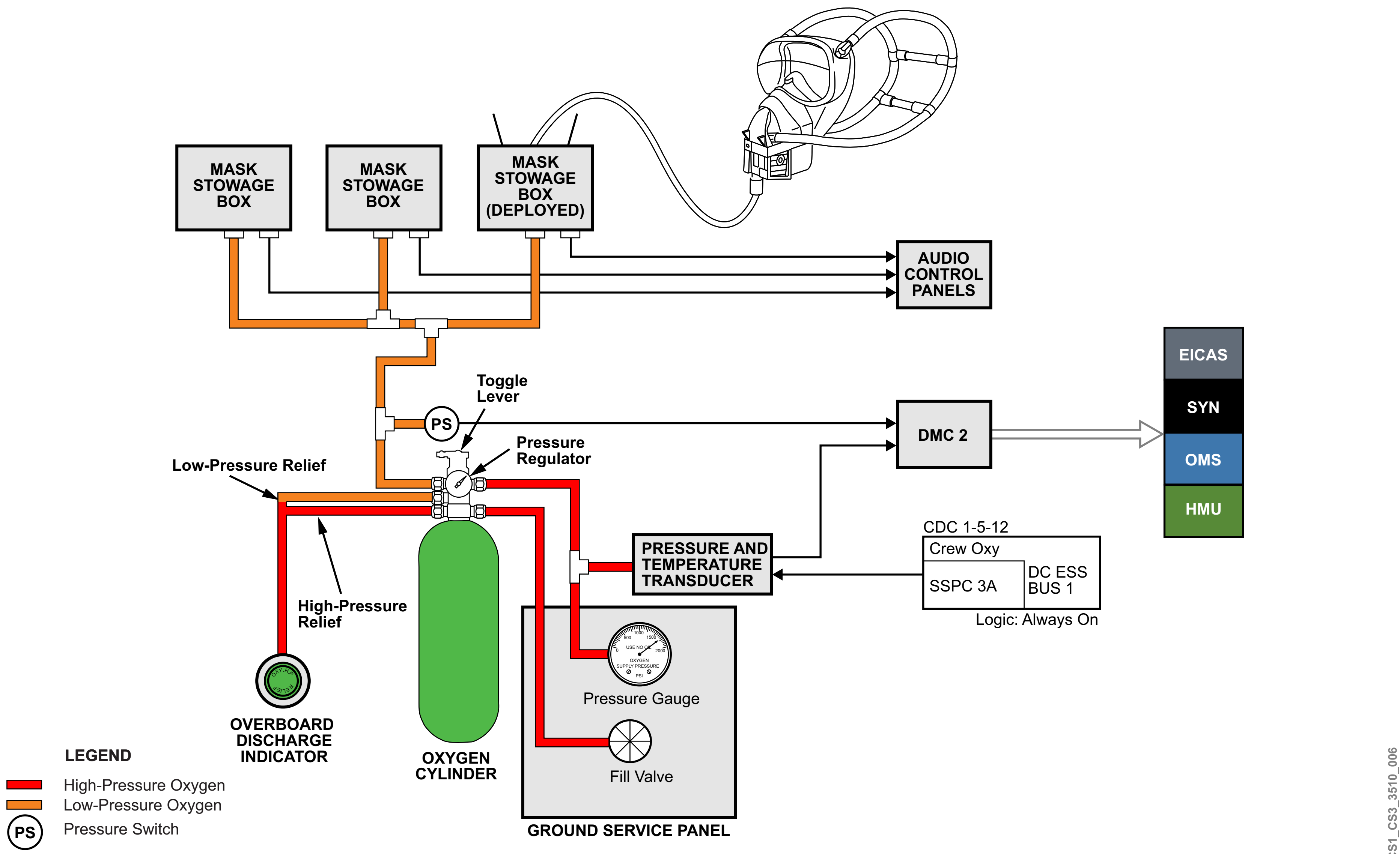


Figure 8: Crew Oxygen System Operation

MONITORING AND TESTS

The following page provides the CAS messages for the crew oxygen system.

CAS MESSAGES

Table 1: CAUTION Message

MESSAGE	LOGIC
CREW OXY LO PRESS	Message is displayed if low pressure in the supply distribution line is detected, or when the crew oxygen bottle pressure is low.

PRACTICAL ASPECTS

OXYGEN GROUND SERVICE PANEL

The crew oxygen system is serviced through an oxygen ground service panel located on the forward left nose. The ground service panel includes a fill valve, temperature correction table, and a pressure gauge.

The oxygen fill valve contains a core valve that is normally closed to prevent the crew cylinder oxygen contents from leaking out. The core valve opens when the oxygen servicing equipment is connected and supplying oxygen to the cylinder.

A direct reading pressure gauge provides an oxygen pressure indication during replenishment.

A temperature correction table is installed adjacent to the gauge which indicates maximum fill pressure versus ambient temperature. The maximum rate of charging should not exceed 200 psig per minute.

WARNING

OBEY ALL THE OXYGEN SERVICING SAFETY PRECAUTIONS WHEN WORKING ON THE OXYGEN SYSTEM AND ITS COMPONENTS. FAILURE TO OBEY THE SERVICING SAFETY PRECAUTIONS, CAN CAUSE A FIRE OR AN EXPLOSION.

CAUTION

Only personnel familiar with the servicing safety precautions are permitted to service of the crew oxygen system.

CAUTION

Do not pressurize the crew oxygen cylinder more than 200 psi each minute. The heat generated can cause a fire or an explosion.

Observe all safety precautions before starting work on the oxygen system. The crew oxygen cylinder is serviced with aviators breathing oxygen only as follows:

- Remove the dust cap and connect the oxygen servicing equipment to the fill valve
- Service the oxygen bottle to 1850 psi at 21°C (70°F) or the equivalent pressure based on the temperature compensation chart. If cabin ambient temperature is higher than outside ambient temperature, cabin ambient temperature will be used to determine full pressure charge. If not, outside ambient temperature will be used
- Close the supply valve of the oxygen servicing equipment
- Check the oxygen pressure at the servicing panel pressure gauge
- Disconnect the oxygen servicing equipment from the fill valve and install the dust cap
- In the flight deck, check the oxygen pressure display on the EICAS STATUS page

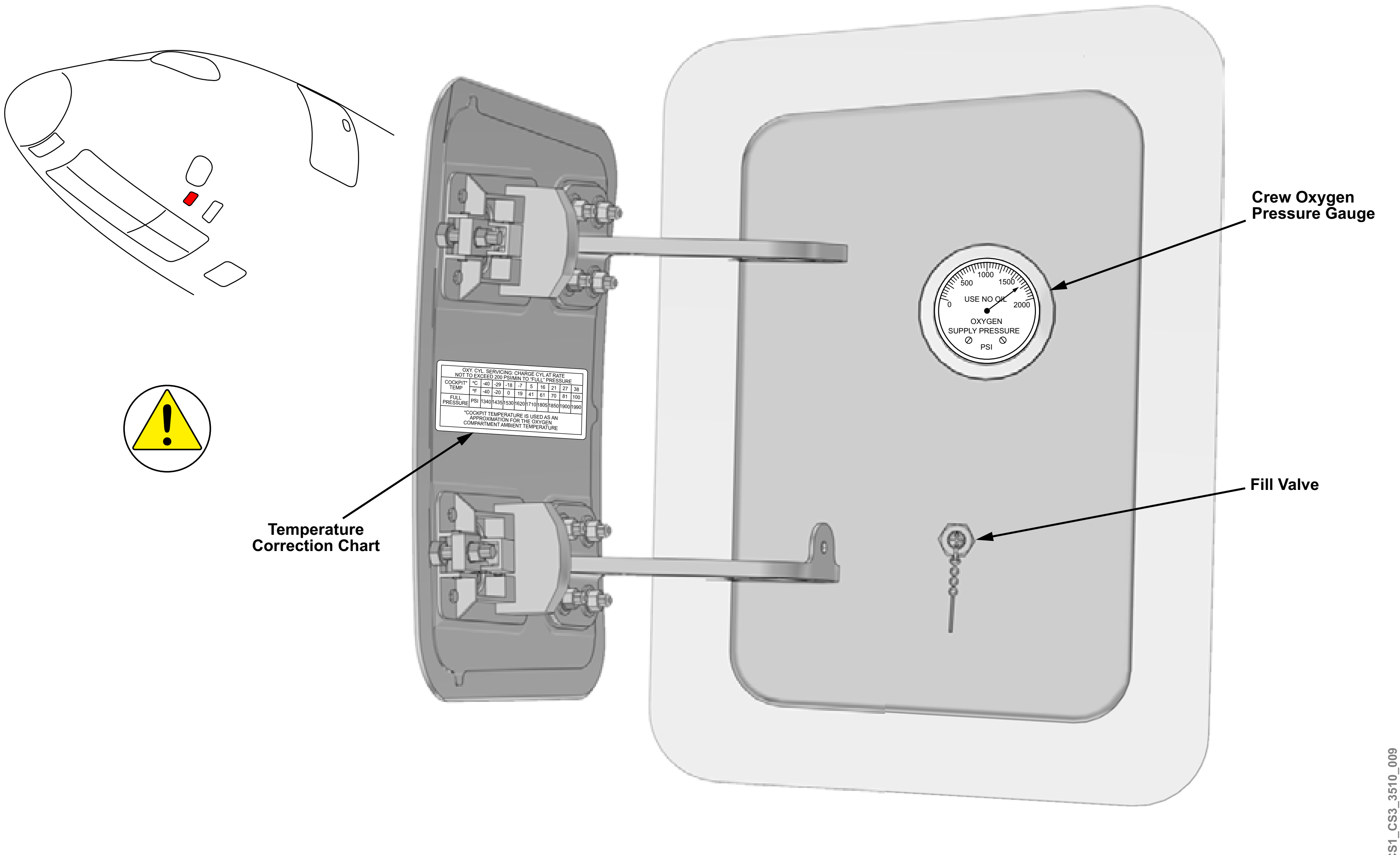


Figure 9: Crew Oxygen Servicing

35-20 PASSENGER OXYGEN SYSTEM

GENERAL DESCRIPTION

Individual passenger oxygen generators supply oxygen to masks located above the passenger and flight attendant seats. Individual oxygen cylinders supply oxygen to masks located in each lavatory. When the cabin altitude exceeds 14,000 ft, the passenger oxygen system deploys automatically. If required, the flight crew can manually deploy the passenger oxygen system from the flight deck using the PAX OXY DEPLOY PBA.

Data concentrator unit module cabinet (DMC) 1 and DMC 2 generate an EICAS message when control and distribution cabinets (CDC) 1 and CDC 2 provide 28 VDC to the oxygen dispensing unit latches.

The cabin management system (CMS) turns ON the passenger ordinance signs, cabin sidewall and lavatory ceiling lighting, increases the passenger address (PA) level, and pauses the in-flight entertainment, music, and prerecorded announcements. The CMS provides a chime alert to the crew indicating that the oxygen masks have deployed.

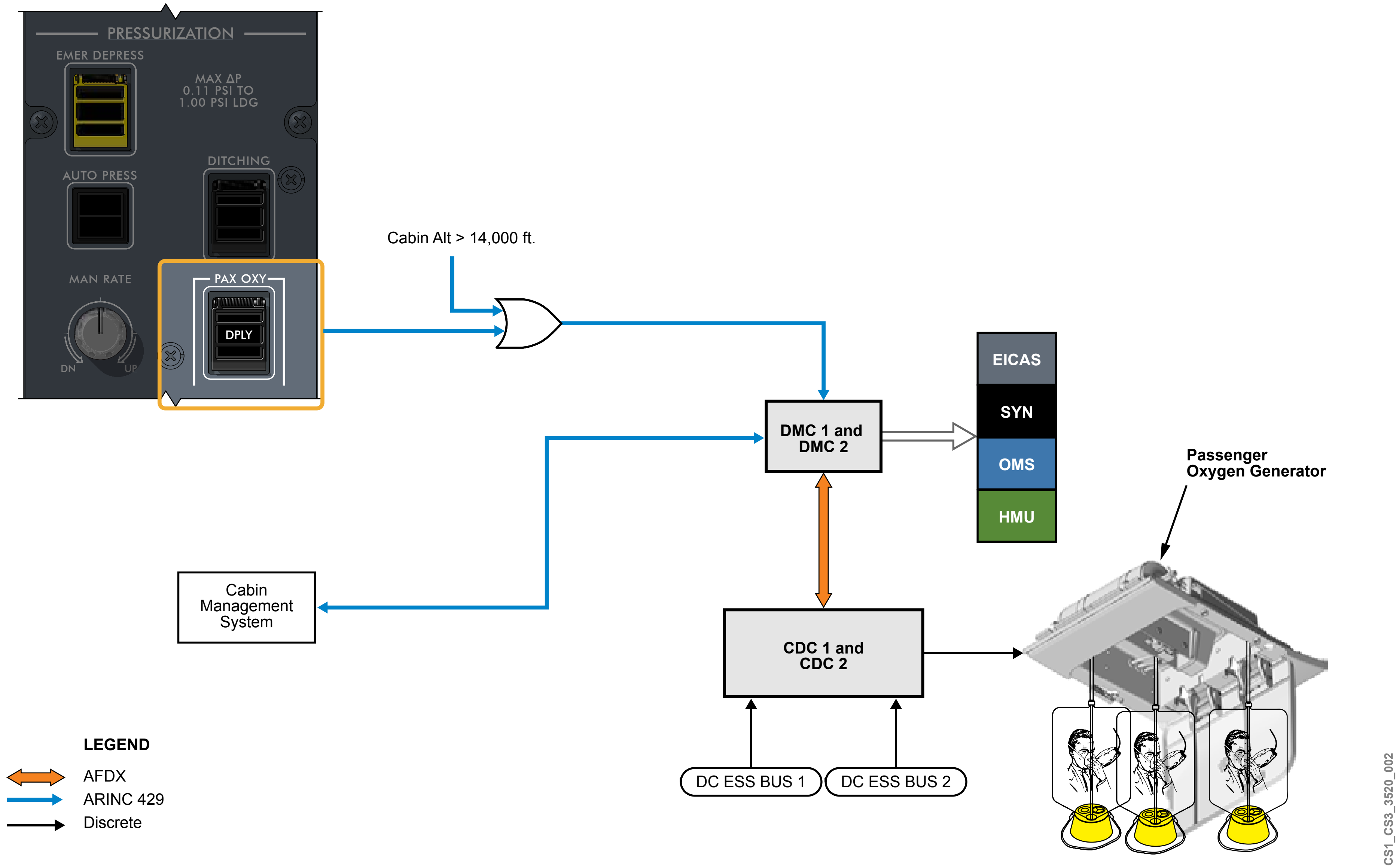


Figure 10: Passenger Oxygen Layout

COMPONENT LOCATION

The passenger oxygen dispensing units are located in the following areas:

- Passenger cabin
- Lavatories
- Flight attendant areas

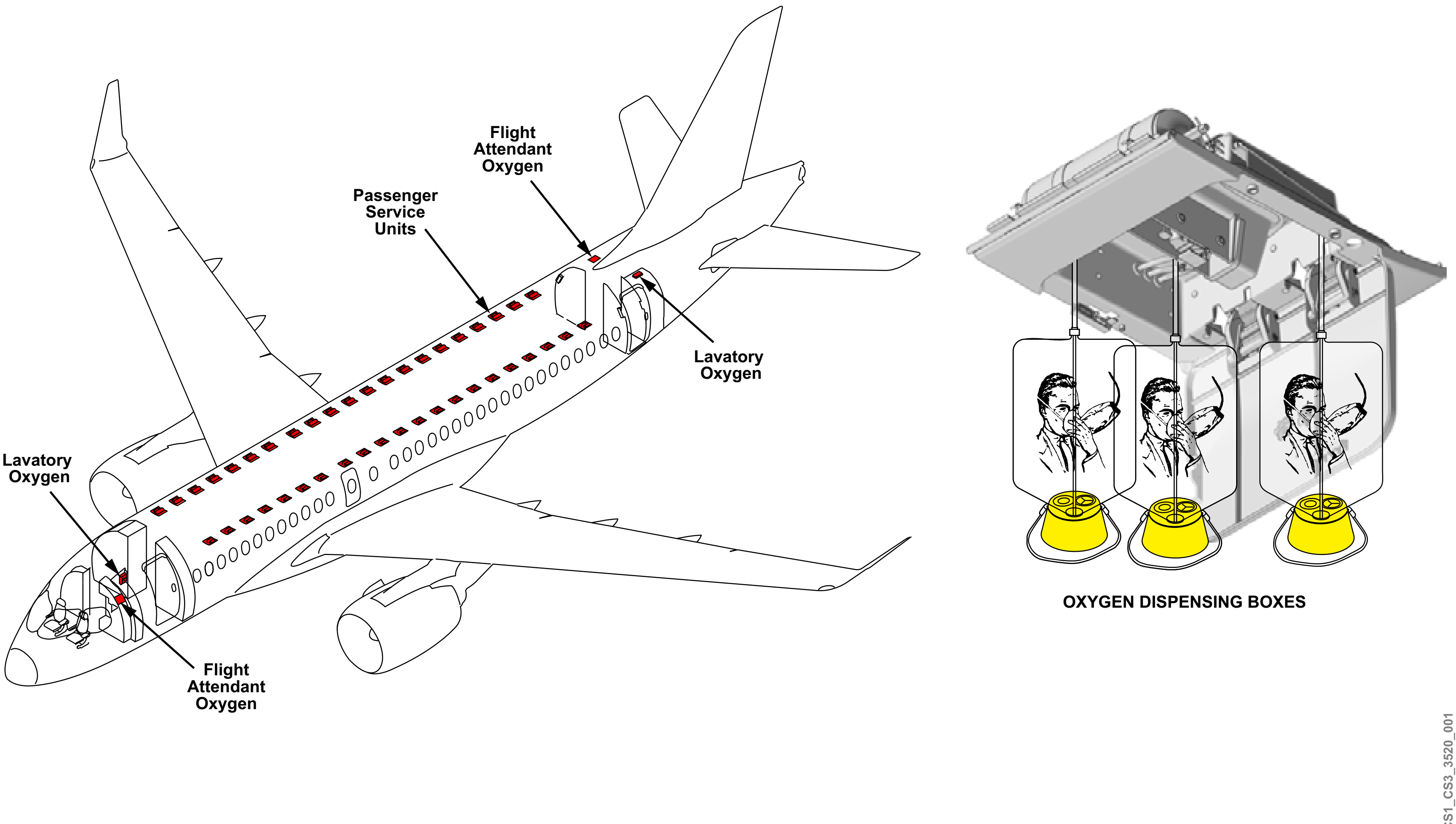


Figure 11: Passenger Oxygen Component Locations

COMPONENT INFORMATION

PASSENGER OXYGEN DISPENSING BOX

The oxygen dispensing boxes store from two to four passenger masks depending on location. The oxygen dispensing boxes are installed above each row of passenger seats in the passenger service unit (PSU). Additionally there are oxygen dispensing boxes located above the flight attendant seats, in the entry ceiling near the forward and aft passenger doors, and in each lavatory.

The oxygen dispensing box has a latch to open the door to release the oxygen masks. Each latch has a manual release in the event of a failure. There is also a test latch for testing mask deployment on the aircraft. The operation of all the dispensing units is similar.

PASSENGER OXYGEN GENERATORS

The passenger oxygen generators are self-contained, passenger activated units with a useful life of 15 years. These generators supply oxygen to the passenger masks during cabin depressurization and are used only once.

A visual indicator shows whether the generator has been used. Under normal conditions, the indication is blue and turns black when the oxygen generator has been activated.

PASSENGER OXYGEN MASKS

The passenger masks are continuous flow masks consisting of a molded silicone rubber face piece with an adjustable headband, and a vinyl reservoir bag. A vinyl tube from the reservoir connects the mask to the oxygen generator or lavatory oxygen cylinder. A lanyard attached to the mask pulls a release pin that activates the oxygen generator or lavatory oxygen cylinder when the mask is pulled.

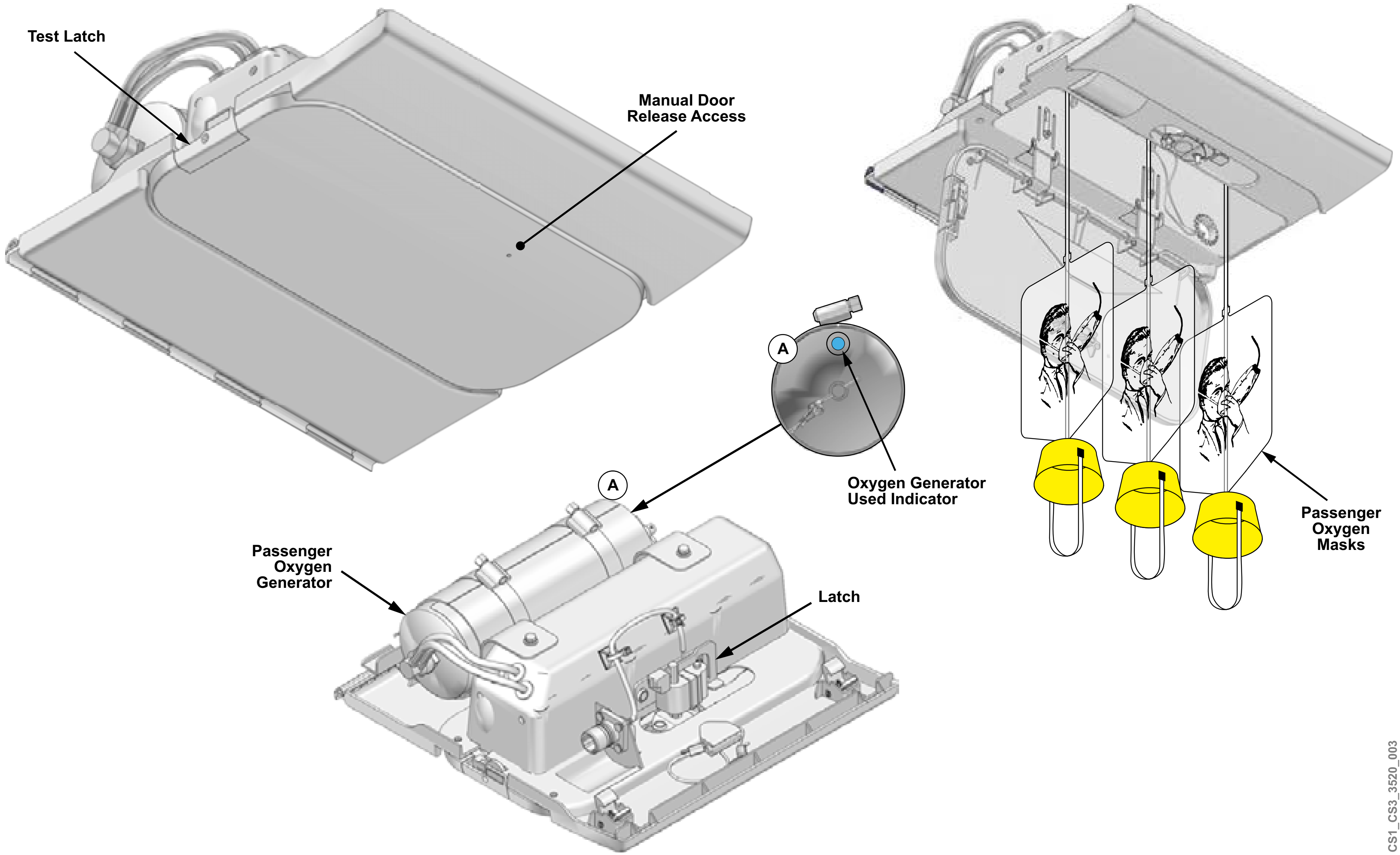


Figure 12: Passenger Oxygen Components

LAVATORY DISPENSING BOX

The lavatory oxygen dispensing boxes installed in the lavatories are required to be tamper proof and can not contain a chemical generator. The lavatory dispensing boxes use an oxygen bottle in place of the chemical generator.

The oxygen dispensing unit has a latch to open the door and release the oxygen masks. Each latch has a manual release in the event of a failure. There is also a test latch to limit door travel when doing mask deployment tests. The operation of the lavatory dispensing box is similar to the passenger oxygen dispensing boxes.

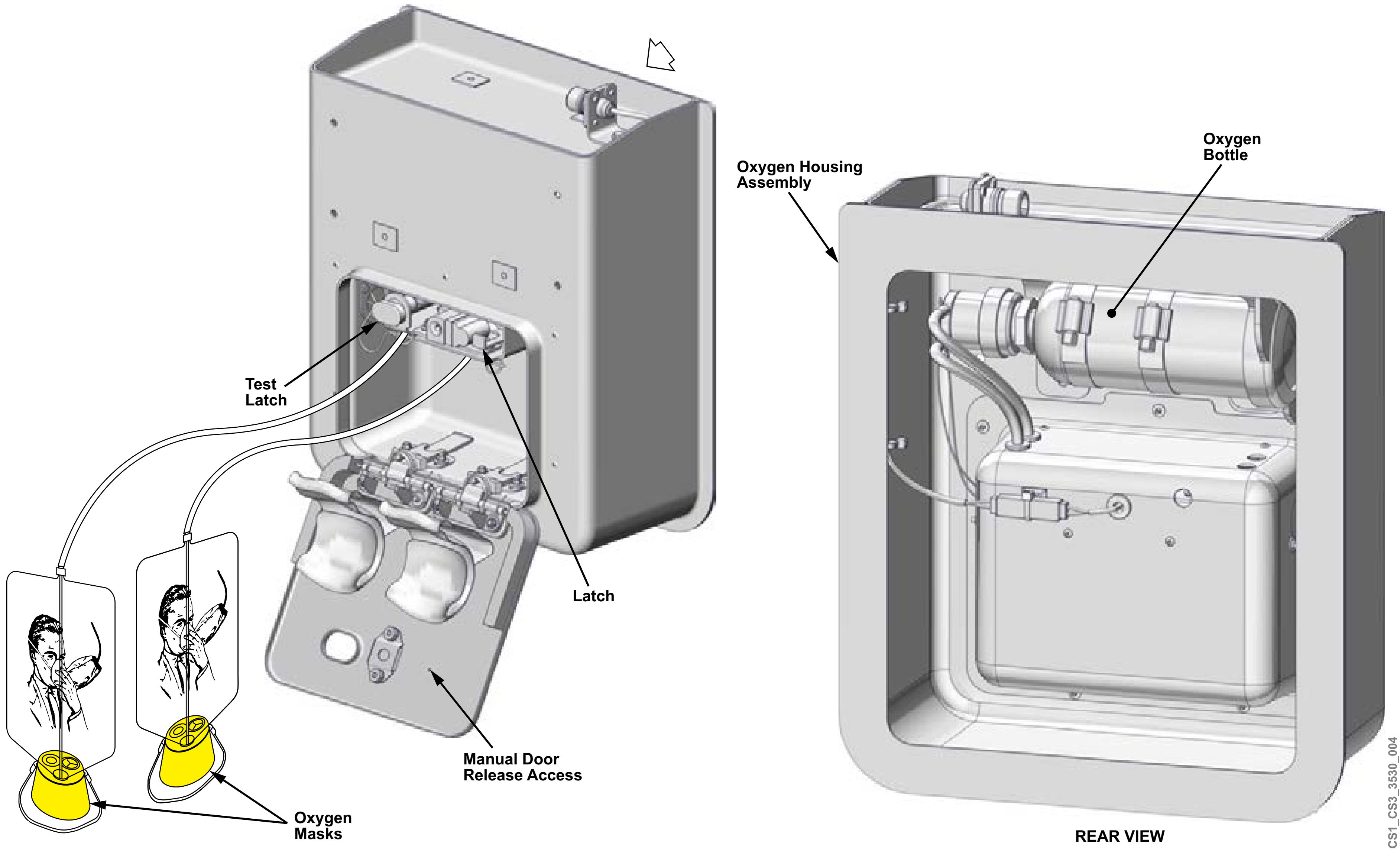


Figure 13: Lavatory Oxygen Dispensing Unit

DETAILED COMPONENT INFORMATION

PASSENGER OXYGEN GENERATOR

The generators installed on non extended-range operation (ETOPS) aircraft have a 13 minute oxygen supply. Those installed on ETOPS configured aircraft have a 22 minute oxygen supply.

A thermal insulation blanket surrounds the generator and a heat shield is installed on the cabin side of the generator to reduce the temperatures on the surrounding structure. The shield has a red warning placard indicating that the generator is hot when activated.

The passenger oxygen generator is a steel cylinder containing iron and sodium chlorate and a primer. It has a striker pin, a release pin, and an output port. The release pin holds the striker pin against the spring force.

The release pin is attached to a release cable that connects to a lanyard on the masks. The masks are connected to the outlet port through a manifold.

When a passenger or attendant pulls on a deployed oxygen mask, the lanyard and release cable pull the release pin out of the striker pin. The spring moves the striker pin into the primer causing a chemical reaction that creates oxygen. Once the generator is activated, it cannot be stopped.

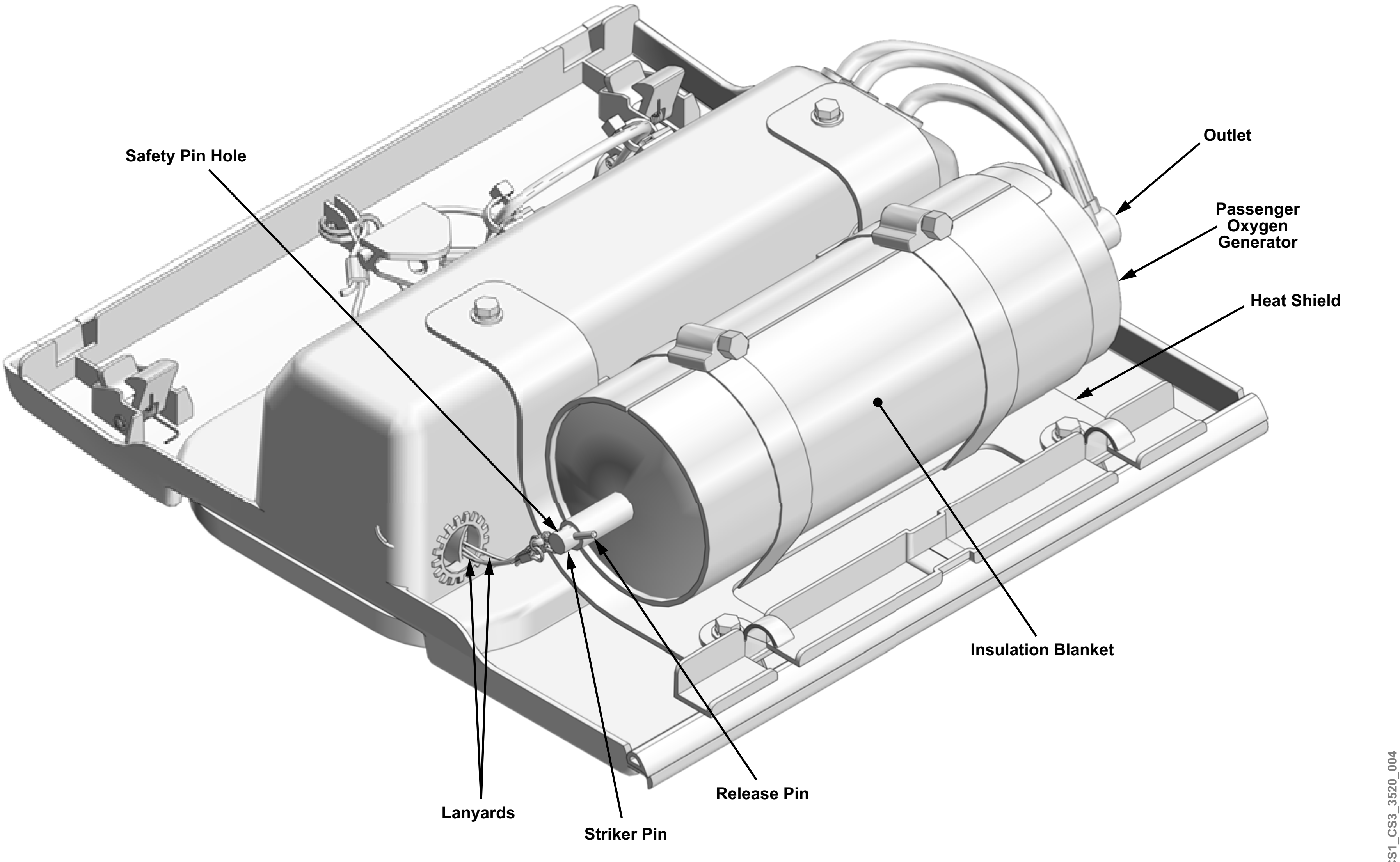


Figure 14: Passenger Oxygen Generator

PASSENGER OXYGEN LATCH

The passenger oxygen latch mechanism holds the oxygen dispensing unit door in the closed position. In the event of sudden cabin depressurization, electrically controlled mechanical latches will release the door and allow the masks to deploy. A solenoid mounted on the latch mechanism is used to release the latch. When oxygen is required, a 28 VDC signal is applied to the solenoid for 5 seconds, the latch releases allowing the door to open and the masks drop out. The oxygen dispensing unit door can be opened manually using a release tool if the latch fails to open the door.

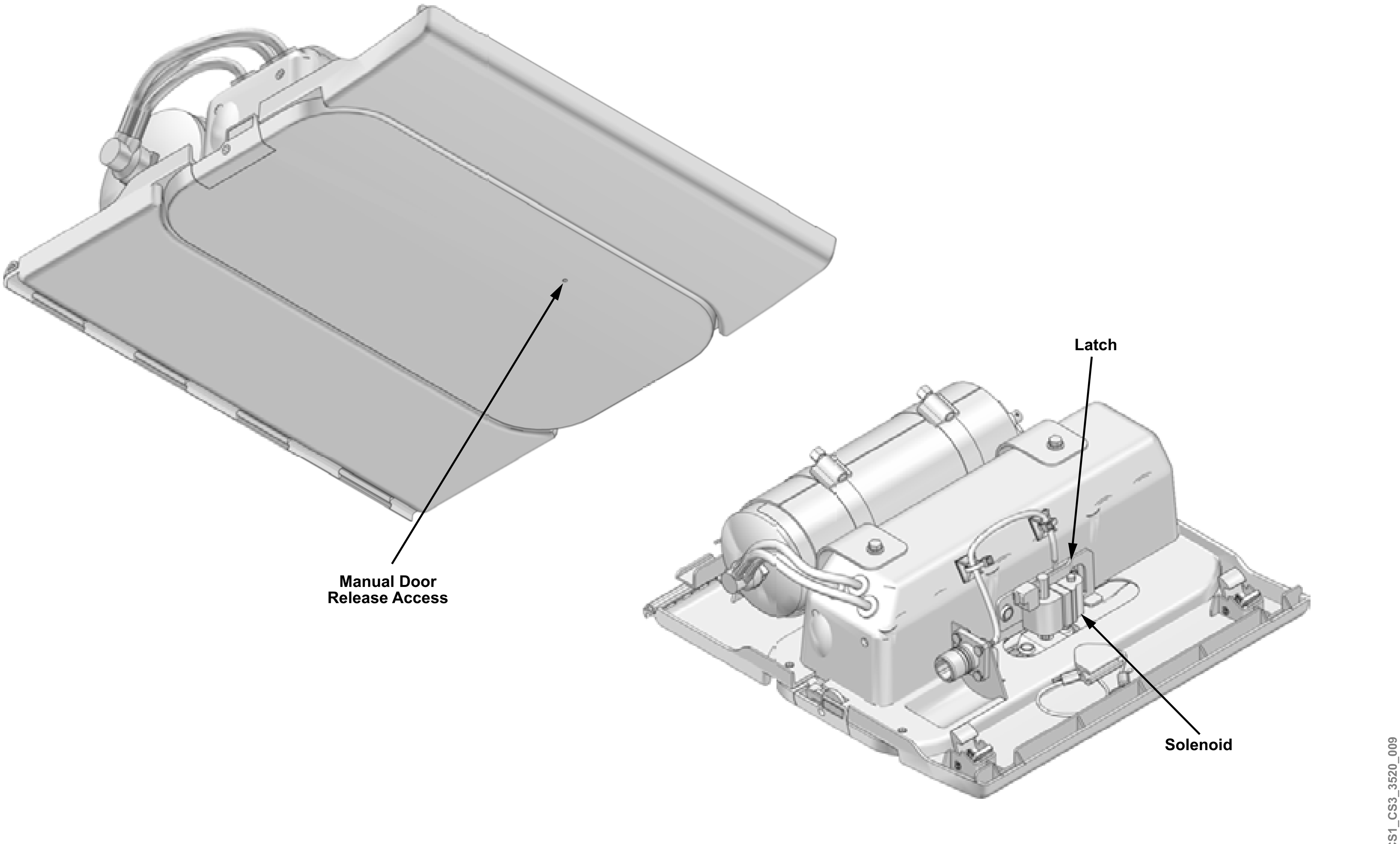
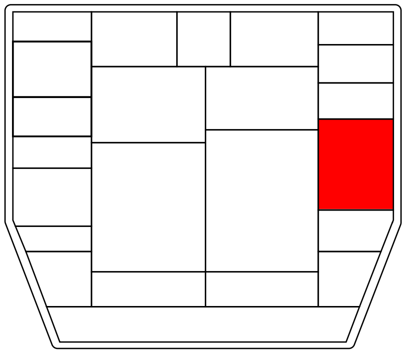


Figure 15: Passenger Oxygen Latch

CONTROLS AND INDICATIONS

PASSENGER OXYGEN PBA

The guarded PAX OXY PBA on the overhead PRESSURIZATION panel manually deploys the oxygen masks. The white DPLY legend illuminates when masks are automatically or manually deployed.



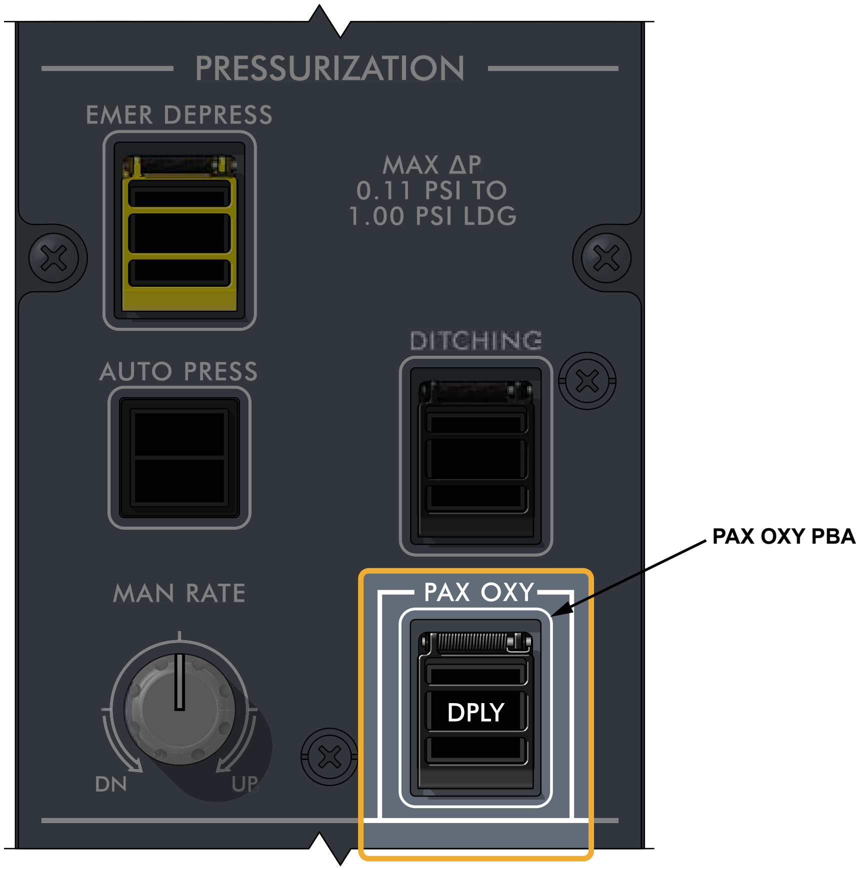


Figure 16: Passenger Oxygen Controls

DETAILED DESCRIPTION

The passenger oxygen system can be deployed automatically or manually at anytime should the cabin altitude become excessive.

AUTOMATIC DEPLOYMENT

The CDCs receive cabin altitude from the integrated air system controllers (IASCs), and FMS landing elevation information from the DMCs. If the cabin altitude is at 4519 m ± 52.7 m (14,826 ft ± 173 ft) or higher, the SSPCs will provide 28 VDC power to energize the oxygen dispensing unit latches on the passenger service units (PSUs) for 5 seconds.

MANUAL DEPLOYMENT

The DMCs provide the state of the PASS OXY switch to the CDCs. When the PAX OXY PBA is pressed, the associated CDCs provide 28 VDC power to energize the oxygen dispensing unit latches on the PSUs for 5 seconds.

OPERATION

During automatic or manual deployment, each CDC provides two 28 VDC signals. CDC 1 energizes the left hand odd and right hand even latches. CDC 2 energizes the left hand even and right hand odd latches.

The CDCs provide feedback to the DMCs that oxygen latches have been energized. This feedback generates an EICAS PASS OXY DEPLOYED advisory message, and a DPLY light on the PAX OXY PBA. The DPLY light remains ON until the passenger oxygen solid-state power controllers SSPCs are cycled.

The CDCs also provide information to the cabin management system (CMS) that an oxygen deployment occurred. The CMS does the following:

- Turns on the FASTEN SEAT BELT and RETURN TO SEAT passenger ordinance signs
- Increases PA audio level
- Turns on sidewall lights
- Turns on lavatory ceiling lights
- Pauses the audio, prerecorded announcements, and in-flight entertainment system
- Generates a triple high chime

When the cabin altitude is below 3657.6 m (12,000 ft):

- FASTEN SEAT BELT and RETURN TO SEAT passenger ordinance signs remain on
- PA audio returns to normal level
- Sidewall lights are controllable
- Lavatory ceiling lights are controllable
- Restores control to the audio, prerecorded announcements, and in-flight entertainment system

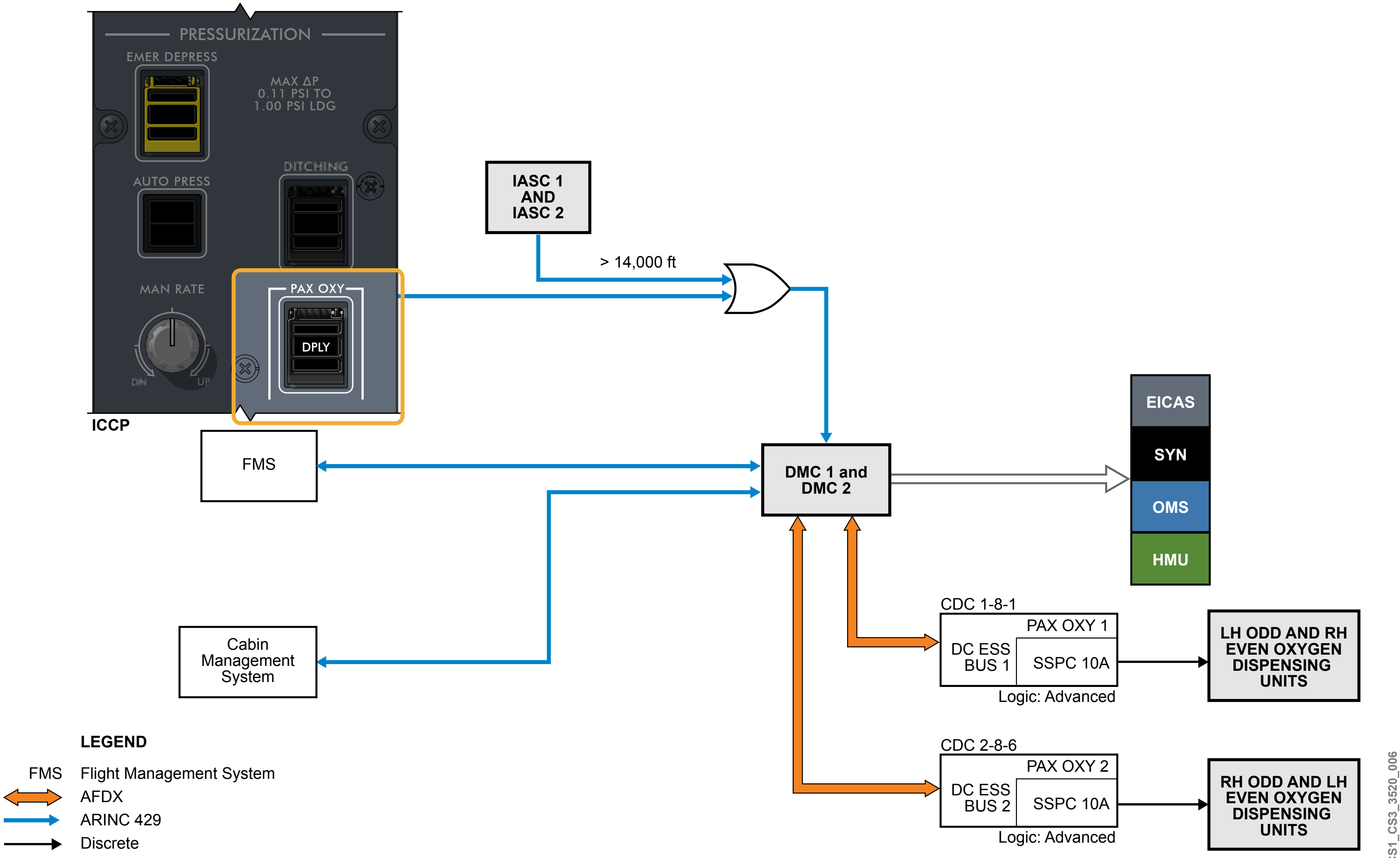


Figure 17: Passenger Oxygen Operation

MONITORING AND TESTS

The following page provides the CAS messages for the passenger oxygen system.

CAS MESSAGE

Table 2: STATUS Message

MESSAGE	LOGIC
PAX OXY DPLY	Passenger oxygen automatically or manually deployed.

PRACTICAL ASPECTS

PASSENGER OXYGEN GENERATOR

Before a serviceable oxygen generator is removed from the passenger service unit, a safety pin must be installed to prevent the generator from firing. The safety pin must be installed before the release pin is removed. Do not pull on the lanyards before the safety pin is installed.

To install the safety pin, pull the striker out until the safety pin hole is exposed.

WARNING

MAKE SURE THE SAFETY PIN IS INSTALLED IN THE OXYGEN GENERATOR WHEN PERFORMING MAINTENANCE ON THE MASK CONTAINER. THE OXYGEN GENERATOR IS A PYROTECHNIC-ACTIVATED DEVICE. IF THE RELEASE PIN CONNECTED TO OXYGEN GENERATOR IS REMOVED WHILE THE SAFETY PIN IS NOT INSTALLED, THE GENERATOR WILL OPERATE. THE OXYGEN GENERATOR CAN HAVE A SURFACE TEMPERATURE HIGHER THAN 232°C (450°F) AND CAN CAUSE BURNS.

Cap all open oxygen lines and fittings when removing the passenger oxygen generator.

CAUTION

Make sure that a protective cap or plug is installed in all of the open pipes and component fittings immediately after they are disconnected. Failure to do this, can contaminate the oxygen system.

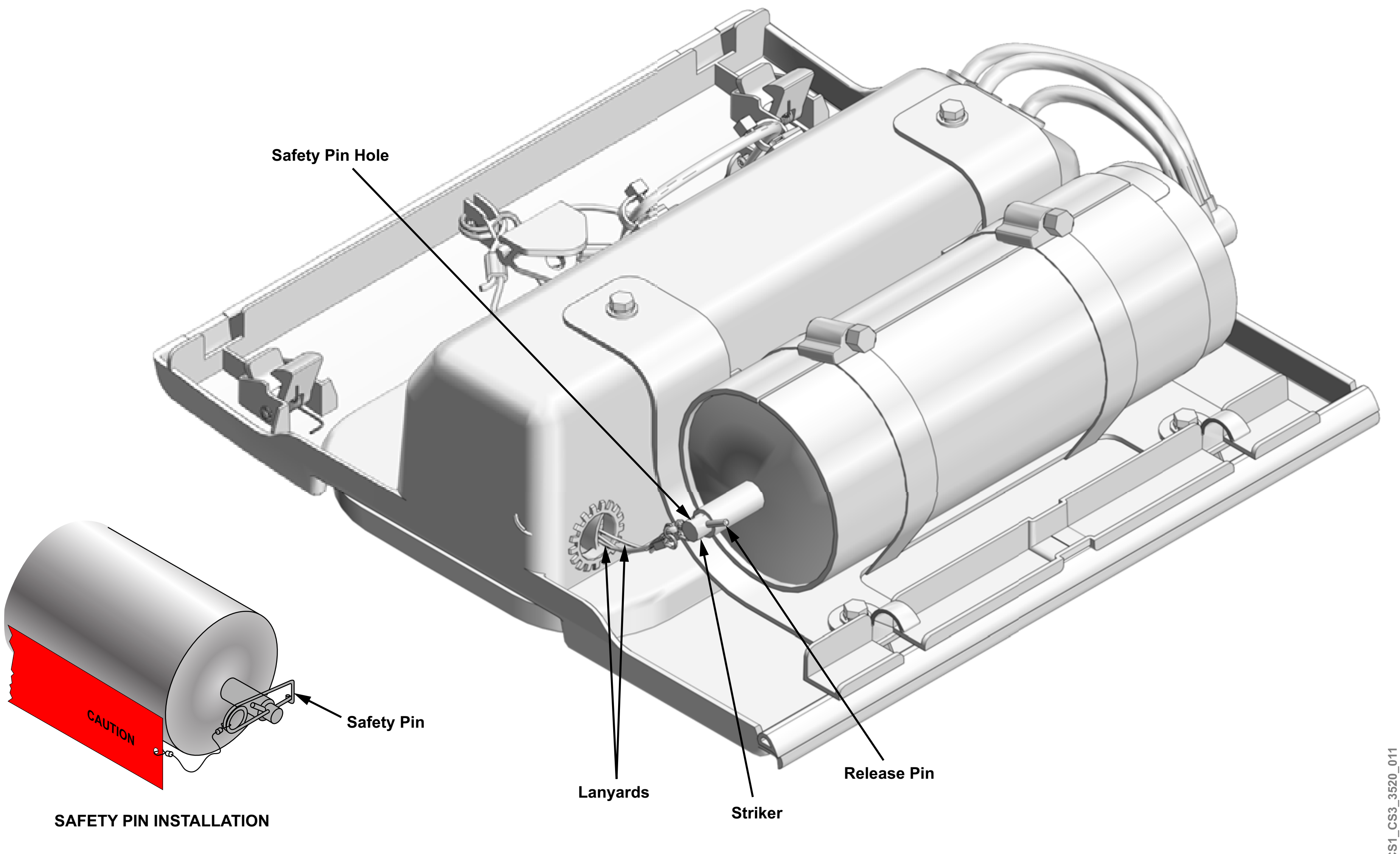


Figure 18: Passenger Oxygen Generator

PASSENGER OXYGEN TEST LATCH

In case of a latch failure, a manual release tool is used to open the door. The tool is inserted into a small diameter hole on the door and contacts a lever on the latch, forcing it to release the door.

The oxygen dispensing unit contains a mechanical test latch to allow for periodic checks of the door opening without deployment of the masks. The test latch, when activated, holds the door open approximately 5° from the normal closed position. This enables the testing of the electric door latch and the opening of the door without the need to re-package the oxygen masks. The latch is colored red on the edges to ensure that it is not accidentally left in the ON position during normal aircraft operation.

NOTE

Make sure the test latch is set to the test position on all mask containers before testing the passenger oxygen deployment system.

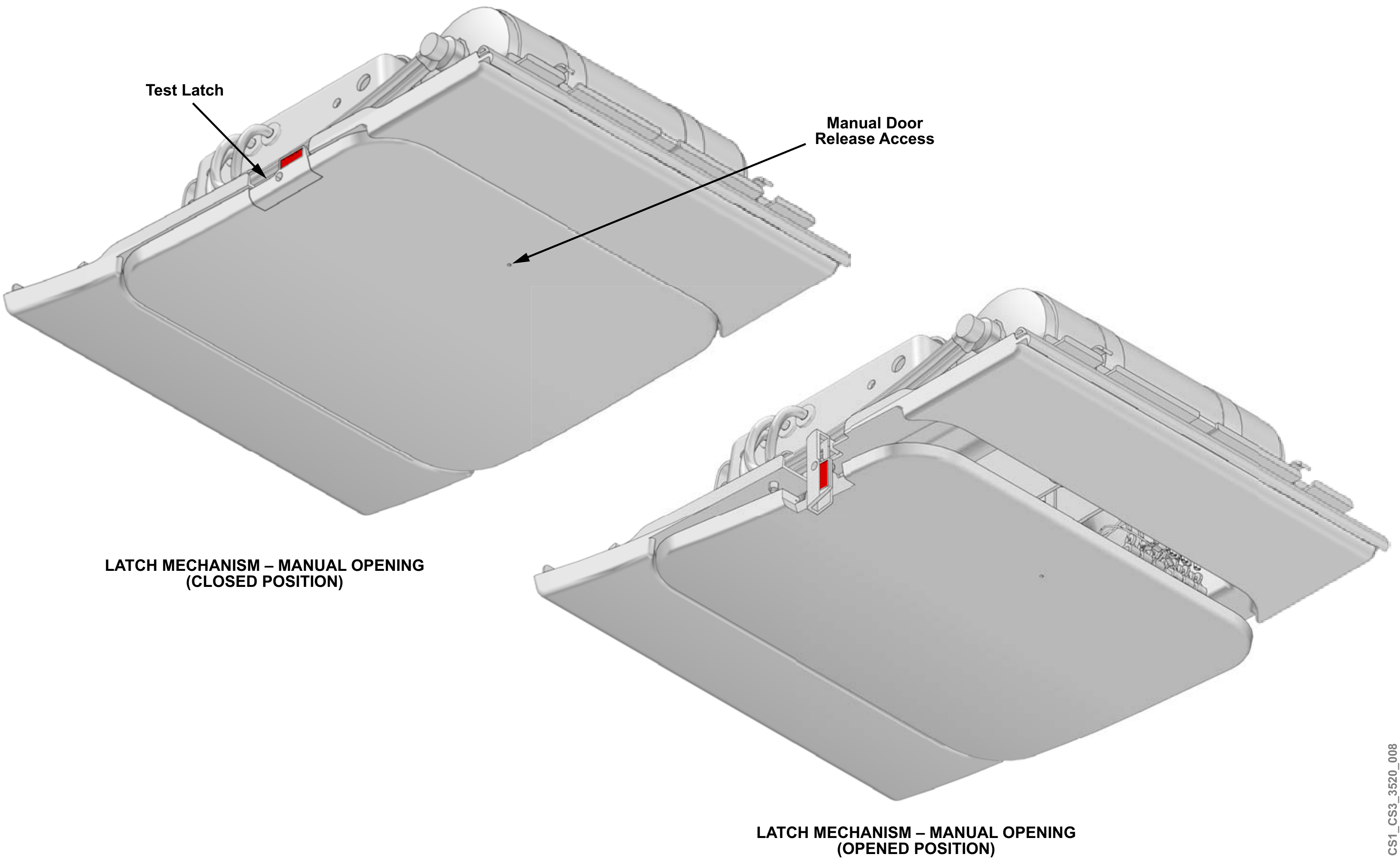


Figure 19: Passenger Oxygen Test Latch

35-30 PORTABLE OXYGEN

GENERAL DESCRIPTION

The portable oxygen system provides oxygen to the crew and passengers at any location for use as first aid or during emergencies.

COMPONENT LOCATION

The portable oxygen consists of the following components:

- Portable oxygen cylinder and mask

PORTABLE OXYGEN CYLINDER AND MASK

The portable oxygen cylinders are located in the forward closet and aft windscreen. The exact locations may vary depending on the aircraft configuration.

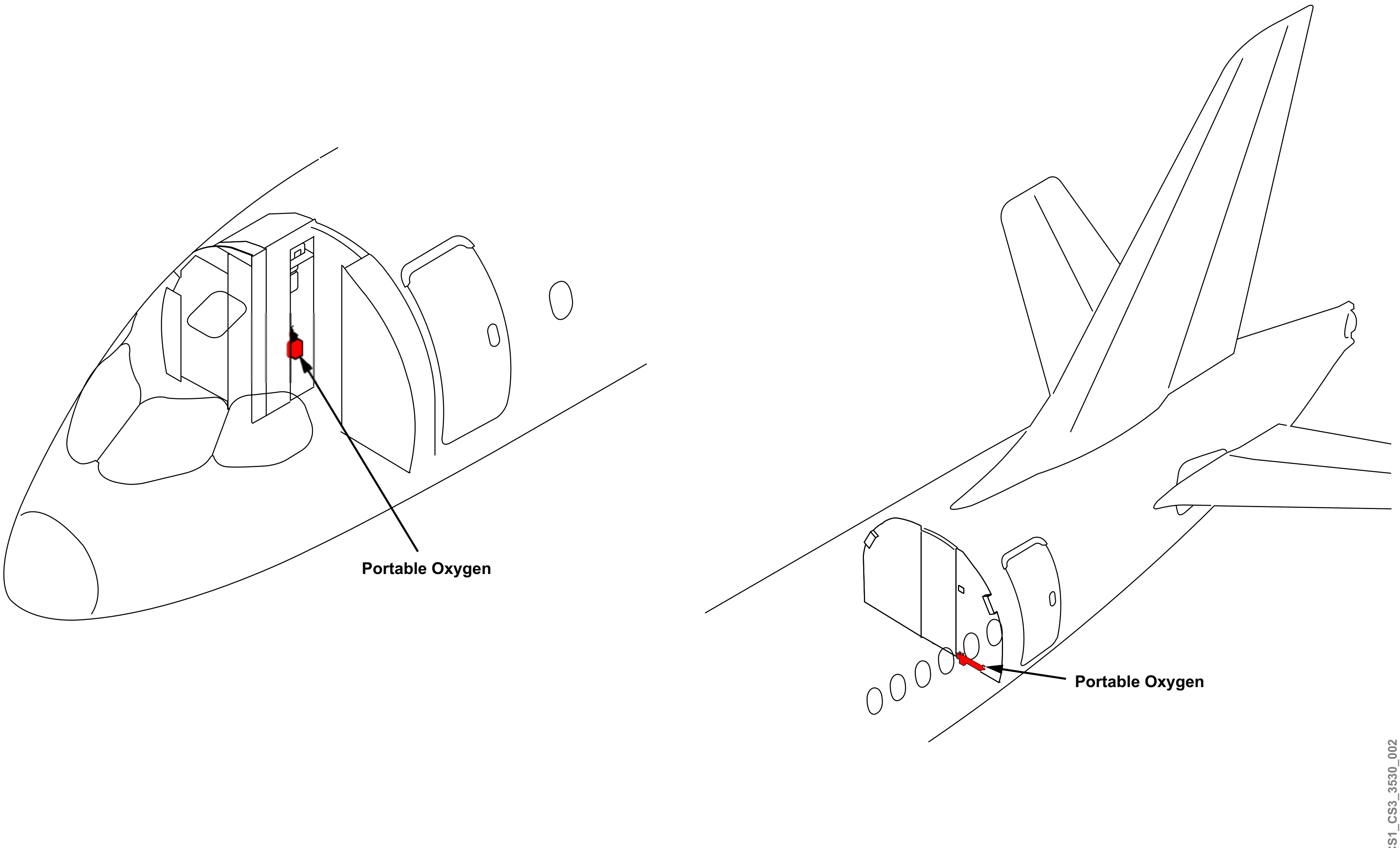


Figure 20: Portable Oxygen Location

COMPONENT INFORMATION

The portable oxygen consists of the following components:

- Portable oxygen cylinder and mask

PORTABLE OXYGEN CYLINDER AND MASK

The portable oxygen cylinders are pressurized gas cylinders, used to supply oxygen to the portable masks. Normal bottle pressure is 1800 psi.

The portable oxygen cylinders contain the following:

- ON/OFF valve
- 0 to 2000 psig pressure gauge
- High-flow and low-flow oxygen outlets
- Tote bag and continuous flow oxygen masks

The portable masks are disposable, continuous flow oxygen masks. Two portable masks are kept in tote bags beside each portable oxygen cylinder. Each portable mask has a feed tube, a flow indicator, a face mask, and an exhaust check valve. The flow indicator is installed in the feed tube and indicates green when oxygen is supplied to the portable mask. The mask connects to the bottle using a bayonet fitting.

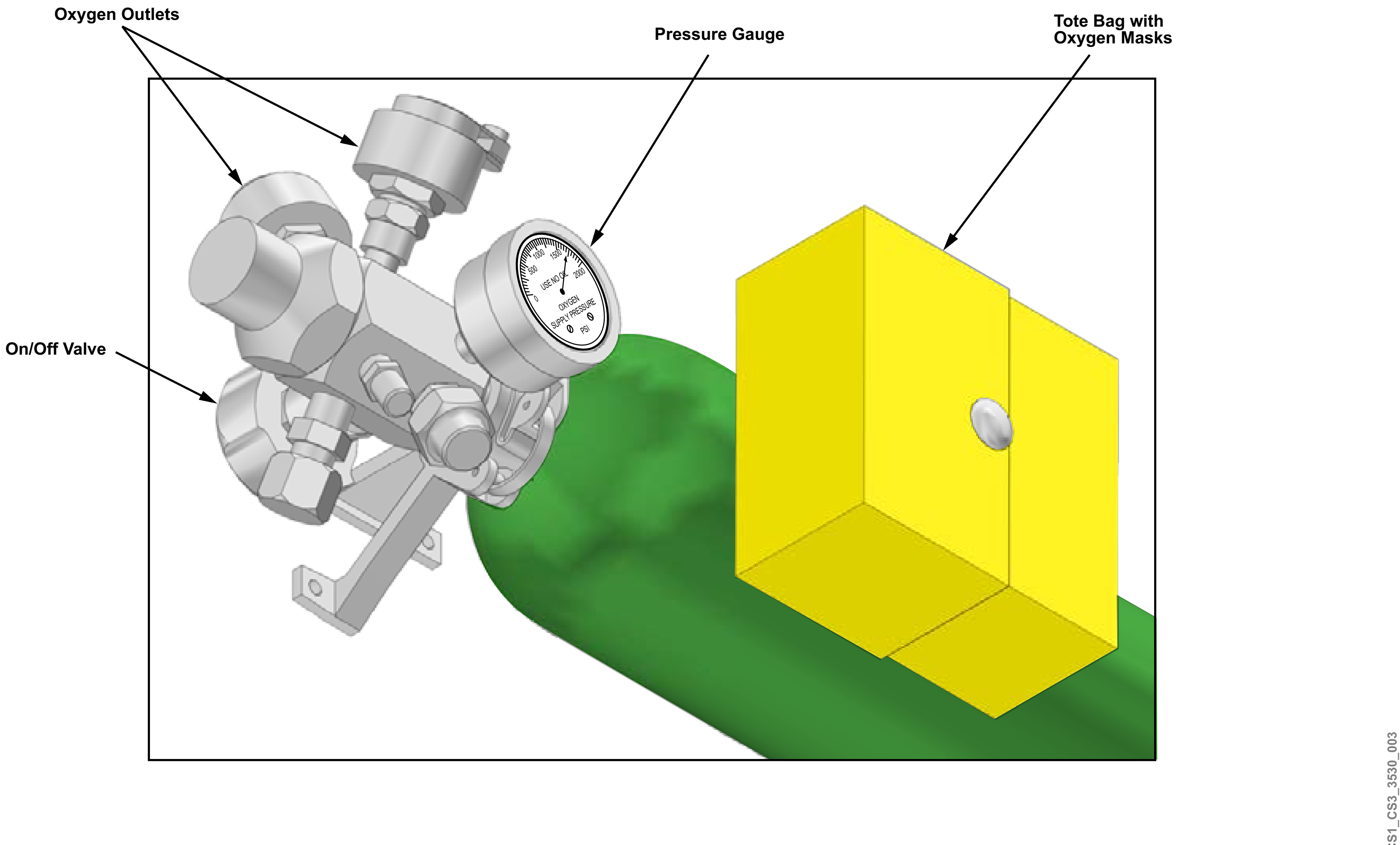


Figure 21: Portable Oxygen Components

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ATA 38 - Water and Waste

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

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List of Figures

WATER AND WASTE - CHAPTER BREAKDOWN

38-00 WATER AND WASTE SYSTEM CONTROLLER

GENERAL DESCRIPTION

The water and waste system controller (WWSC) controls and monitors the water and waste system components and the heating elements associated with the water and waste system. The WWSC is powered by DC BUS 1.

The WWSC configures the potable water system for filling, draining, or flight based on selections made at the water service panel. The two water pumps are also controlled based on water service panel selections. The WWSC monitors the pumps for failures, and selects the idle pump when a failure occurs.

The vacuum-operated waste system reports its status to the WWSC. The WWSC inhibits the system when the waste tank is full.

The heaters are powered by AC BUS 1. The WWSC monitors the temperature sensors for each heating element and provides a turn on signal to the heaters when required.

The WWSC, and the water and waste system components connected to the CAN BUS have field-loadable software.

The WWSC communicates with the data concentrator unit module cabinet (DMC) to provide fault messages to the onboard maintenance system (OMS) and the health management unit (HMU). Faults are also displayed on the cabin management system (CMS) crew terminal.

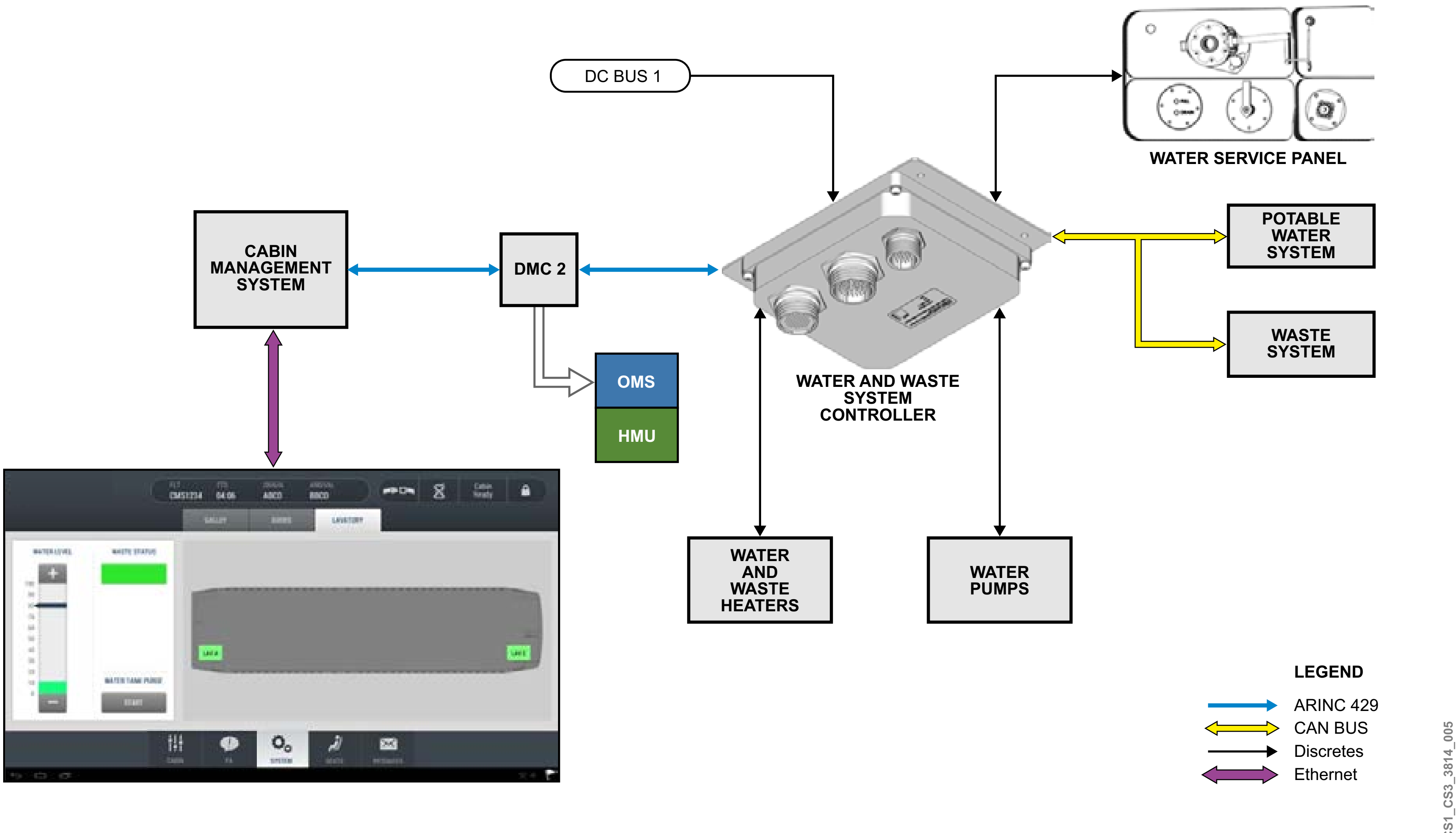


Figure 1: Water and Waste System Controller - General Description

COMPONENT LOCATION

The WWSC is a module installed on the left side of the rear fuselage on a rack just in front of the waste tank.

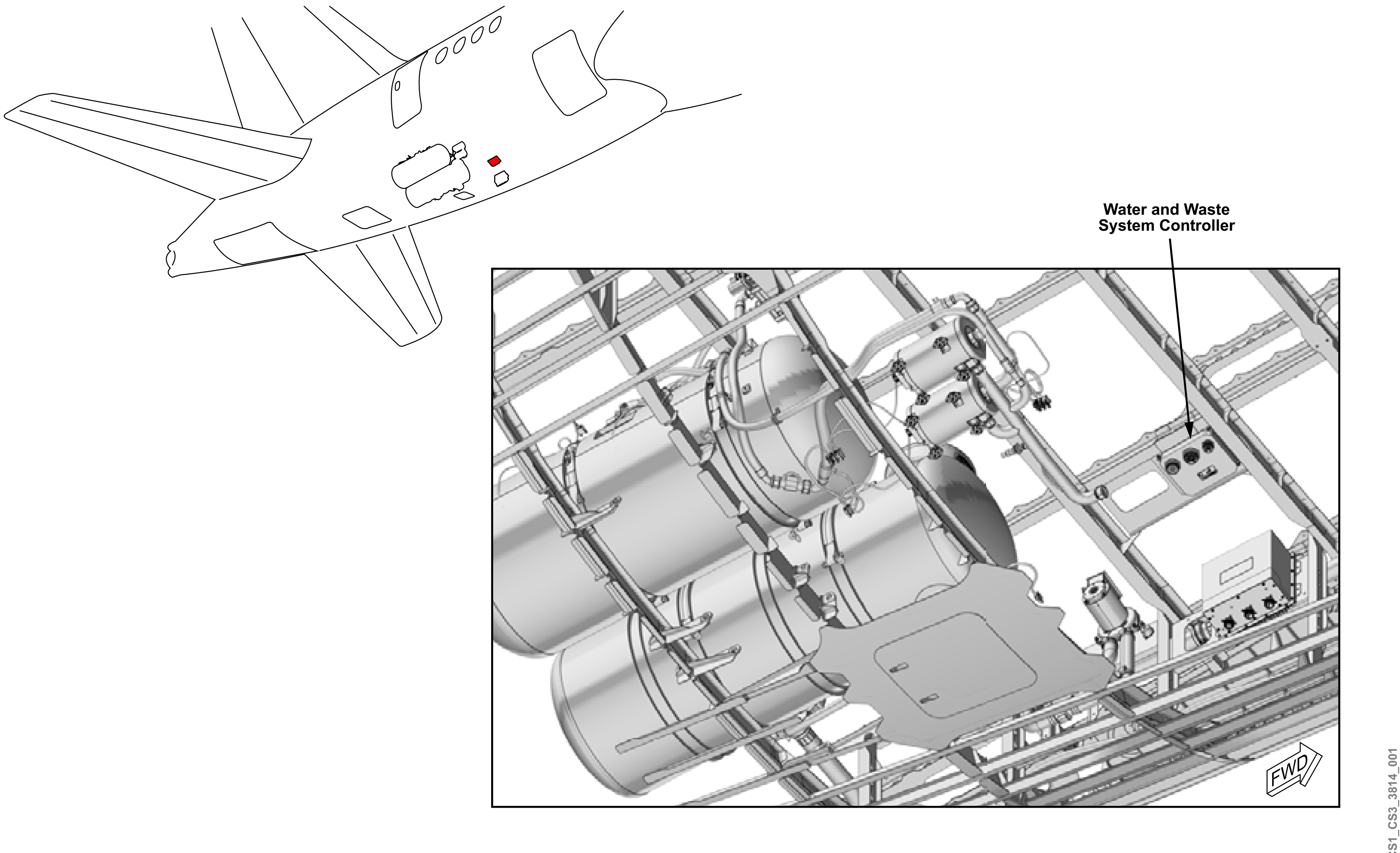


Figure 2: Water and Waste System Controller

38-11 POTABLE WATER STORAGE AND DISTRIBUTION

GENERAL DESCRIPTION

The potable water system supplies pressurized water to the faucets of the lavatories, water spigots, and the coffee makers in the galleys. Water is also supplied in the lavatories for flushing the vacuum toilets.

Potable water is stored in a single tank located below the floor, aft of the aft cargo compartment. The water tank has a usable volume of 159 L (42 gal). A capacitive level sensor is installed in the tank for continuous volume indication.

The potable water system is automatically operative when electrical power is available, water is in the tank, and the water system service panel door is closed. During normal operation, the distribution system is pressurized via a single pump. During system filling, the tank is vented overboard through the aft drain mast. A four-port motor-operated valve controls the filling and venting.

The water distribution lines consist of a main distribution line located on the left side of the aircraft, and separate branch lines to supply the individual lavatories and galleys. The water system plumbing is designed to withstand freezing without damage. The main line has a 2º slope to allow proper drainage overboard and back to the tank. The highest point of the main line is over the wing box. Branch lines are installed between the frames, and are sloped back to the main trunk line to permit drainage.

The water tank is filled from a service panel located in the aft fuselage on the right side of the aircraft. The water service panel is equipped with a fill fitting and a FILL/DRAIN switch. The fill fitting is heated to prevent freezing, and capped to prevent contamination and possible leakage into the service panel. The FILL/DRAIN switch controls the fill and drain functions. The DRAIN LED on the water indication panel illuminates when the FILL/DRAIN switch is in the DRAIN position.

The water system drains through two heated drain masts and through the fill fitting at the service panel. On the ground, the system drain function is activated by the FILL/DRAIN switch at the water service panel. The water system can also be drained in flight from the cabin management system (CMS) terminal.

The water system operation is controlled by the water and waste system controller (WWSC). The WWSC has field-loadable software.

Quantity indication is provided at the CMS terminal. A preselected quantity of water can be input from the CMS terminal. The water indication panel has a FULL LED that illuminates when the water tank is filled to the selected quantity.

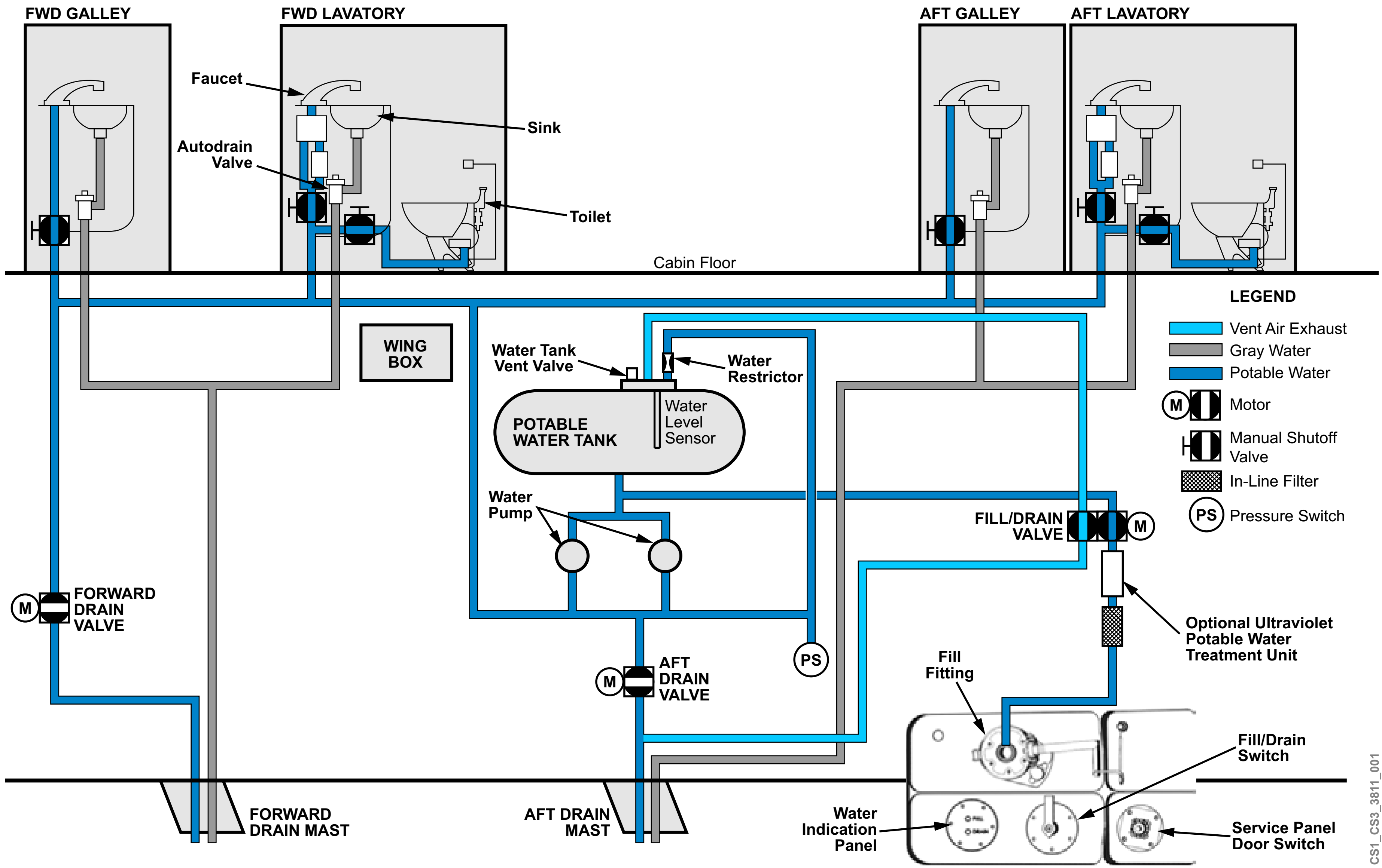


Figure 3: Potable Water System Storage and Distribution

COMPONENT LOCATION

The following components are installed in the potable water system:

- Potable water tank
- Potable water filter
- Fill/drain valve
- Drain valve
- Drain mast
- Water service panel
- Optional ultraviolet potable water treatment unit

POTABLE WATER TANK

The potable water tank is located behind the aft cargo compartment bulkhead adjacent to the vacuum waste tank, below the cabin floor.

POTABLE WATER FILTER

The potable water filter is located in the fill line above the water service panel.

FILL/DRAIN VALVE

The FILL/DRAIN valve is located above the potable water tank.

DRAIN VALVES

The forward and aft drain valves are in the lower fuselage with their respective drain mast.

DRAIN MAST

The two heated drain masts are located on the external underside of the aircraft at the longitudinal centerline.

OPTIONAL ULTRAVIOLET POTABLE WATER TREATMENT UNIT

The optional ultraviolet (UV) water treatment unit is located on the right side of the aircraft near the potable water tank.

WATER SERVICE PANEL

The potable water system service panel is located on the lower aft fuselage.

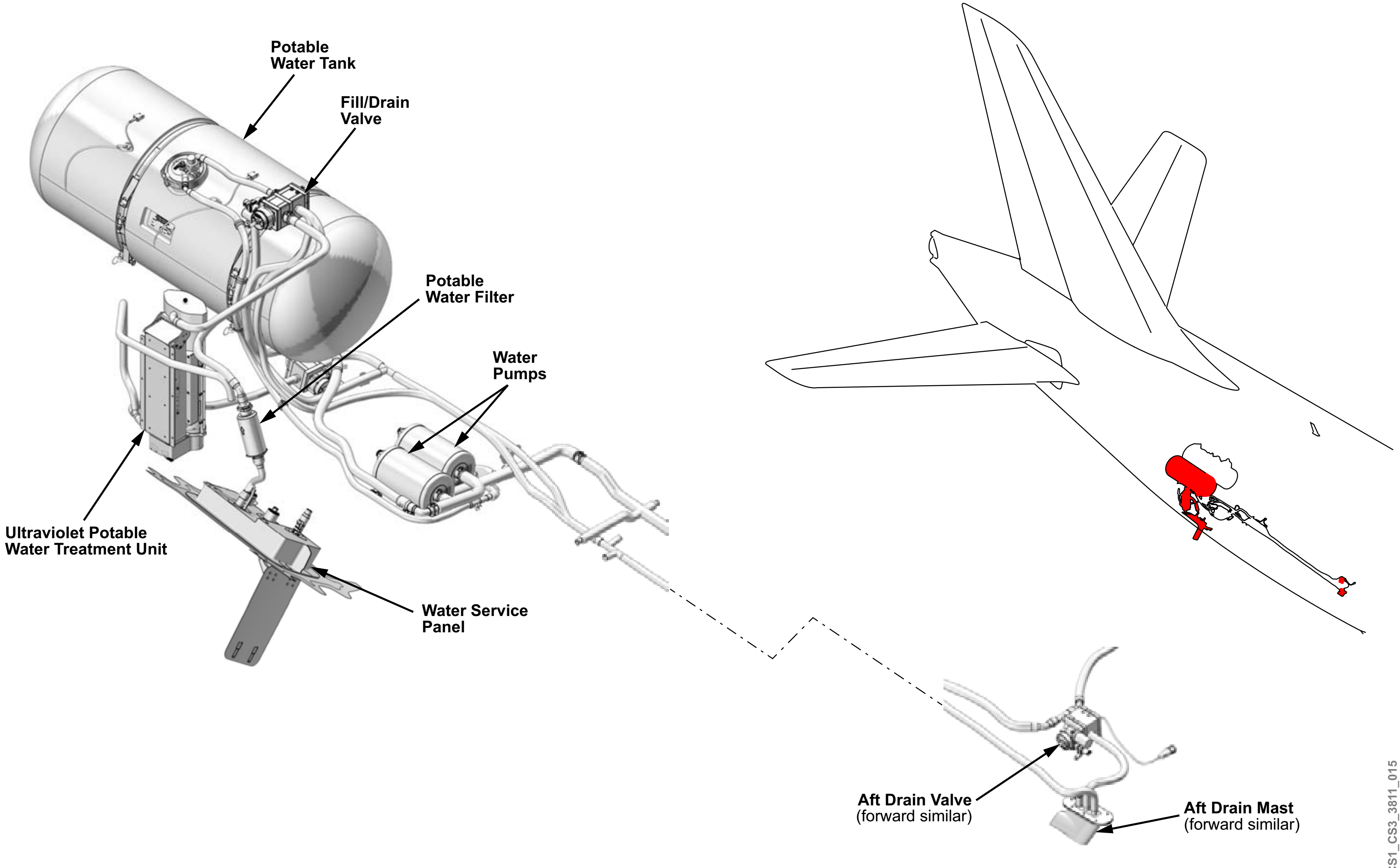


Figure 4: Potable Water Storage and Distribution - Component Location

COMPONENT INFORMATION

POTABLE WATER TANK

The potable water tank stores water to supply lavatories and galleys on ground and during flight.

The water tank is made of corrosion resistant stainless steel with a total capacity of approximately 174 L (46 gal) with a usable capacity of 159 L (42 gal), and a 15 L (4 gal) expansion space. A level sensor installed in the tank provides indication to the cabin management system (CMS) crew terminal.

The tank is located behind the aft cargo compartment bulkhead adjacent to the vacuum waste tank, below the cabin floor crossbeams. It is mounted in a horizontal position by two band clamps, and supported by a structural cradle attached to the aircraft structure.

The tank is vented to the cabin to allow air to enter the tank as the water is pumped out and consumed. The tank vent valve closes during the potable water tank filling operations, and opens at completion of the filling operation to allow the fill line to drain. A vent screen is installed in the valve to prevent foreign matter from entering the water system. An overflow tube, set at approximately 101% of tank level, is provided to limit tank volume in case of a malfunction of the water level control loop, and safety protection against overfilling.

The vent valve, overflow port, and level sensor are installed at the top of the tank. The fill, drain, and distribution lines are installed at the bottom of the tank.

The nominal system pressure inside the tank is at cabin ambient pressure.

FILL/DRAIN VALVE

The FILL/DRAIN valve allows filling, overflow, and ventilation of the water system on ground. The FILL/DRAIN valve is a DC motor operated four-port valve which controls the flow through two separate, parallel water and air lines. The balls are clocked so that both lines are open or closed. A mechanical valve position indicator, located above the tank, may be used to manually open or close the valve in case of a fault condition. A heater and insulation are installed on the valve to prevent freezing.

OPTIONAL ULTRAVIOLET POTABLE WATER TREATMENT UNIT

An optional ultraviolet (UV) water treatment unit is a supplemental device to treat the water as it enters the aircraft. The UV unit is positioned in the fill line, downstream of the filter. The unit is powered when the operator selects the FILL position on the service panel FILL/DRAIN switch. The UV unit operates on 115 VAC 400 Hz ground power. The fill valve is opened after a delay of 80 seconds to allow the lamps to warm up. The unit is equipped with the UV intensity sensor to monitor lamp output, and to indicate lamp change. The UV unit is powered and operated until filling is complete.

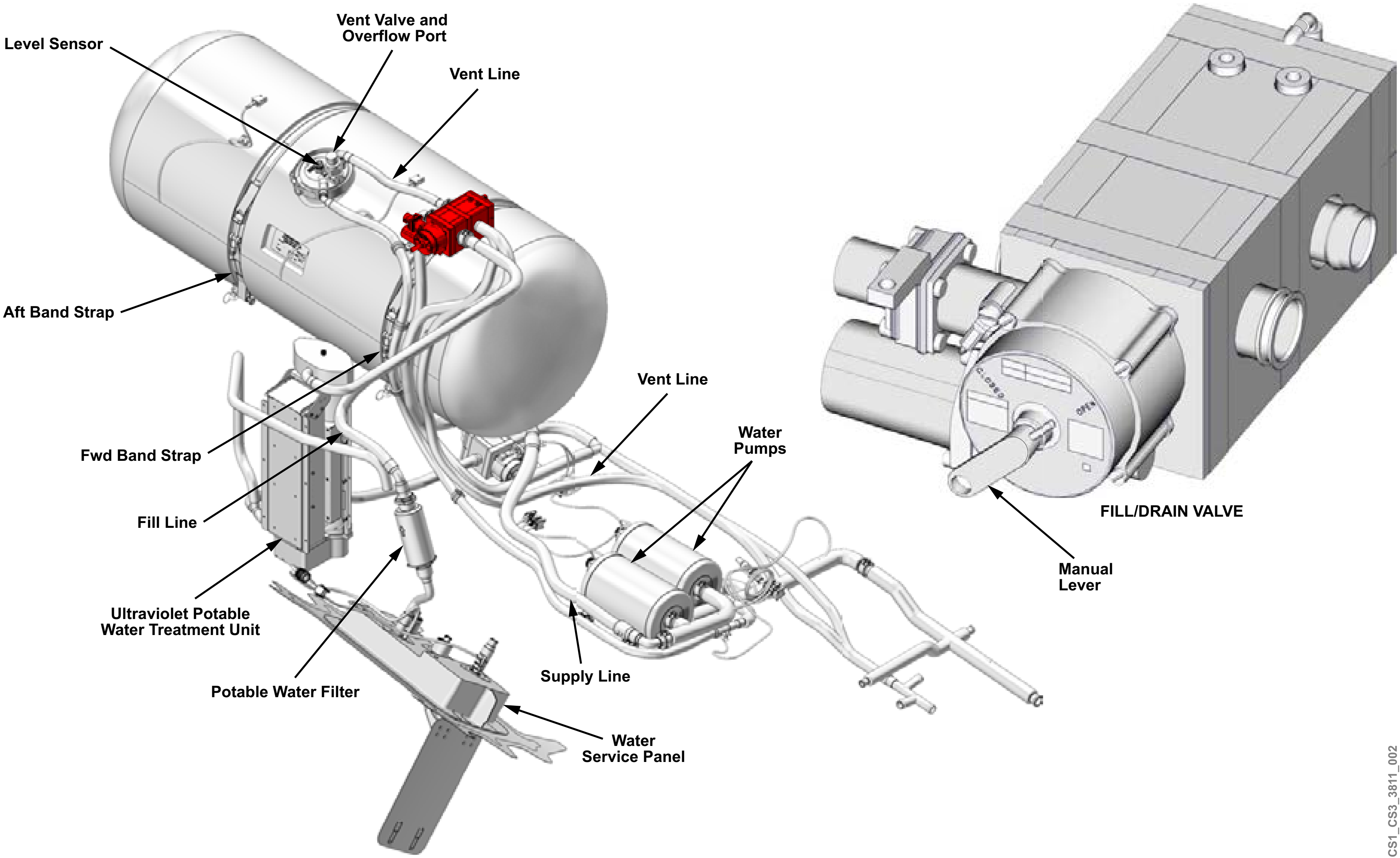


Figure 5: Potable Water System Components

WATER SERVICE PANEL

The FILL/DRAIN switch, fill fitting, and water system indication panel are installed on the service panel. These components are accessed by opening a service panel door.

FILL/DRAIN Switch

The FILL/DRAIN switch provides discrete inputs to the WWSC to control the servicing modes for the potable water system. The FILL/DRAIN switch has three servicing mode selections:

- **FILL** - The FILL position opens the FILL/DRAIN valve to the FILL position and turns off the pump. The tank is filled and air is vented through the valve
- **FLIGHT** - The FLIGHT mode configures the valves to the flight position and turns on the water pump
- **DRAIN** - The DRAIN mode configures the valves to the DRAIN position in order to drain the tank and plumbing

The switch is manually turned to the FLIGHT position before the service panel door can be closed. The service panel door can not be closed if the switch is not in the FLIGHT position.

Water Service Panel Door Switch

The water service panel door switch sends a discrete signal to the WWSC indicating the service panel door is OPEN.

The switch also sends a discrete signal to override the safety thermostat on the forward drain mast, and allows the forward water drain valve to open during cold weather conditions.

Fill Fitting

The fill fitting has a lever lock with sealing cap to protect the fitting from contamination, and to prevent leakage of any residual water that remains in the fill line after servicing.

Water System Indication Panel

The water system indication panel is installed in the water system service panel and provides indication of the potable water servicing modes. The indication panel consists of two light-emitting diodes (LEDs). These indicate either:

- **FULL** - When the potable water tank has achieved its preselected full level
- **DRAIN** - When the system valves are in the open position during system servicing

The WWSC controls the indication lights.

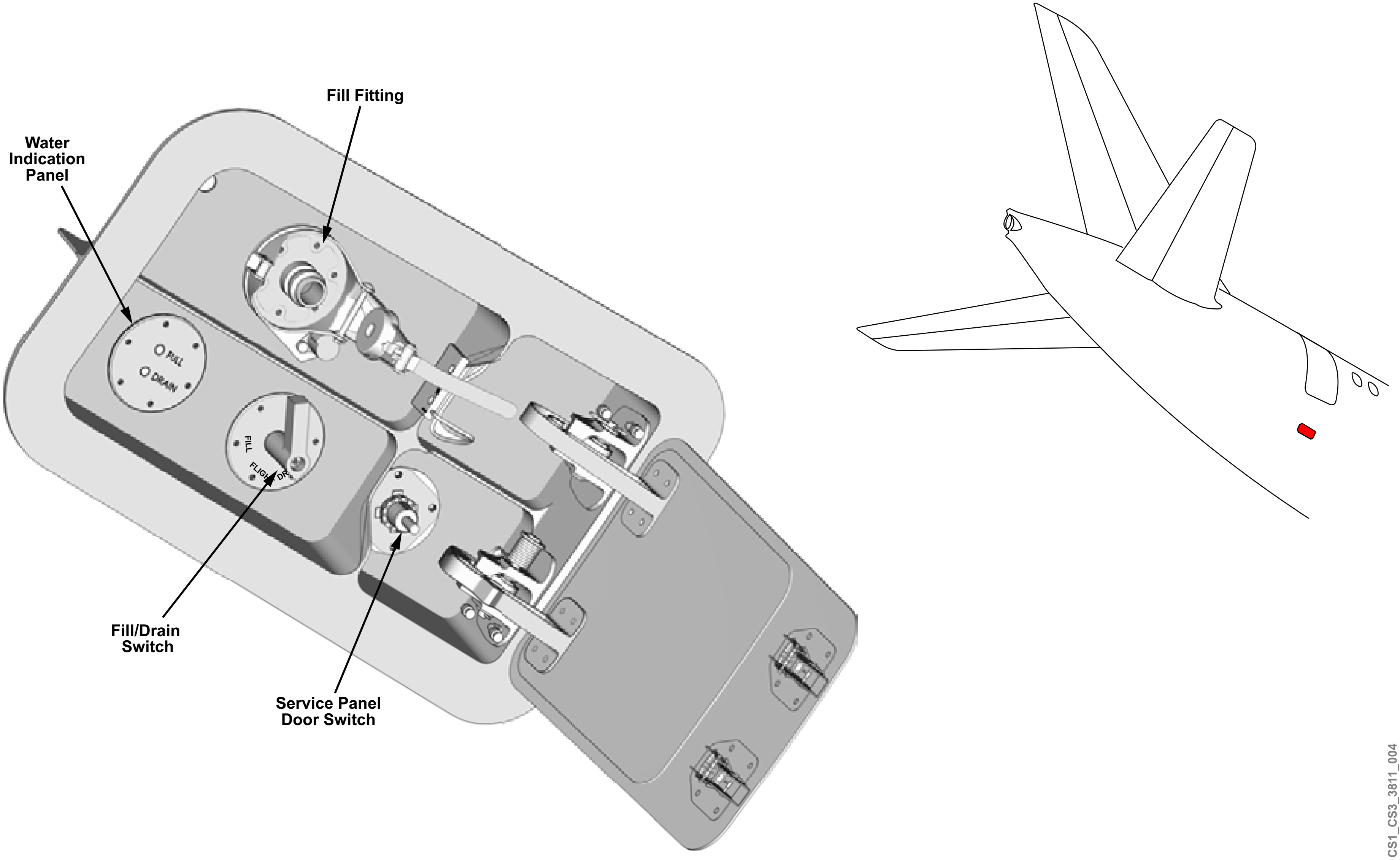
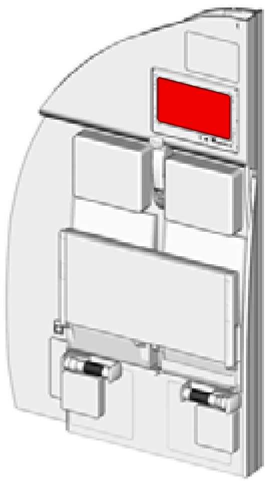


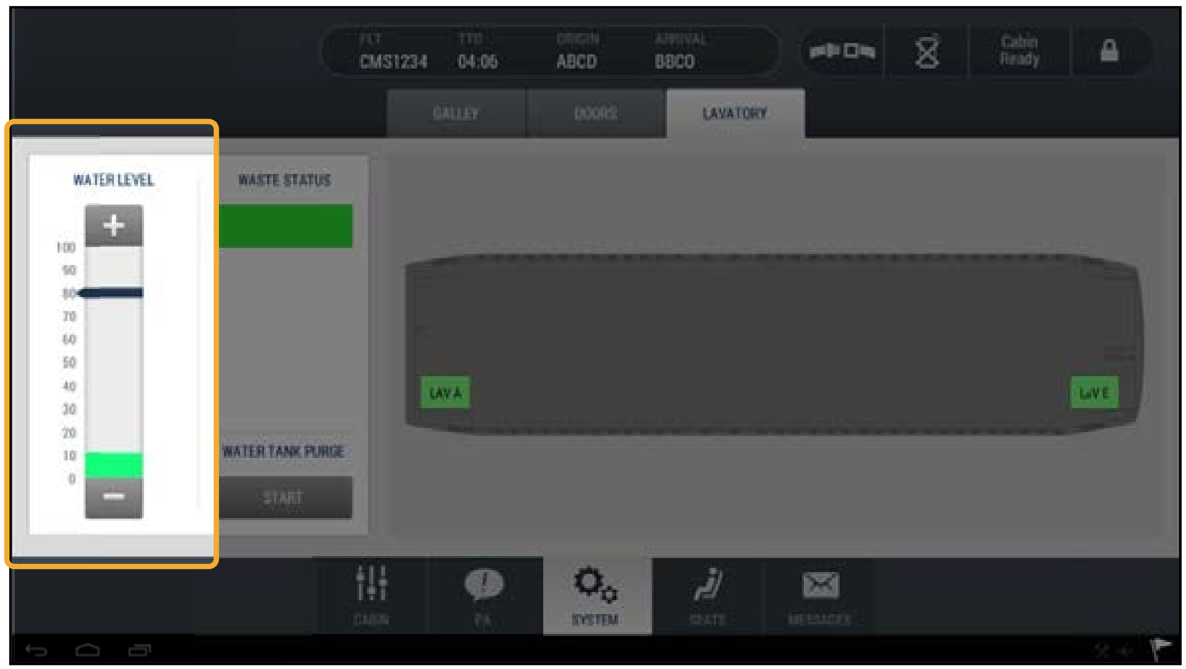
Figure 6: Water Service Panel

CONTROLS AND INDICATIONS

The water quantity is preselected on the CMS terminal water and waste display.



FORWARD FLIGHT ATTENDANT SEATS



CREW TERMINAL

CS1_CS3_3811_016

Figure 7: Water and Waste Cabin Management System Display

DETAILED DESCRIPTION

When power is applied, the WWSC performs a self-check of all connected components. Any existing faults are cleared unless they reoccur during the power-up test. After completing the power-up test, the system indicates the current system status faults, and commands the water valves to their FLIGHT positions.

The potable water system is automatically operative when power is available, water is in the tank, and the water system service panel door is closed. The potable water system is powered by DC BUS 1.

One pump starts when water is present, power is available, and the water system service panel door switch indicates closed. When the water tank is empty, pump operation is inhibited. The water and waste system controller (WWSC) monitors the water level sensor. If a valid empty signal is received for 1 minute, the pump operation is commanded OFF via a discrete message to control and distribution cabinet (CDC) 5.

A continuous level sensor provides water level information to the WWSC. The WWSC calculates quantity and sends this information to the CMS for display. Water fill preselection can be made at CMS in 10% increments.

The FILL/DRAIN valve is powered by 28 VDC supplied to the valve through the WWSC from DC BUS 1. The normal position of the valve in flight is CLOSED. A thermal overheat protector removes power from the motor in case of an overcurrent condition, and automatically resets to the closed position when the temperature falls below the tripping point.

The water system indicator at the water service panel indicates the water system status. The FULL indication illuminates when the water tank level matches the water quantity preselection made. The DRAIN light indicates the system is configured for draining the water system.

On the ground, the water system is drained using the FILL/DRAIN switch at the water service panel. When the water service panel door is open, the door switch overrides the drain mast low-temperature thermostat position to power the forward drain valve.

In flight, the water system is drained using the PURGE selection on the CMS terminal. The PURGE function is active when:

- Aircraft is below 10,000 ft
- Landing gear is not down
- Forward drain mast heater is operational

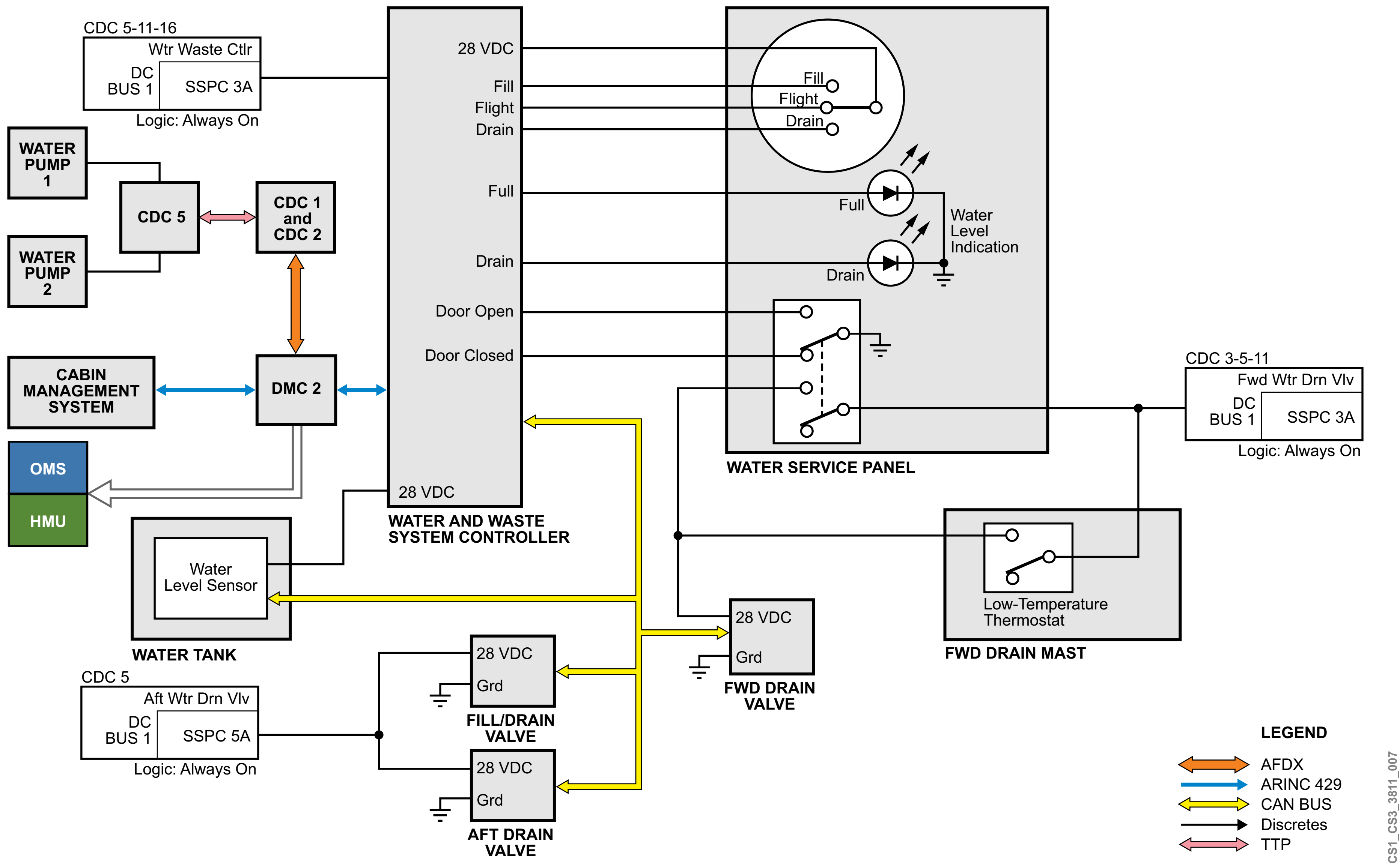


Figure 8: Potable Water System - Detailed Description

MONITORING AND TEST

The following page provides the cabin management system (CMS) water and waste displayed messages.

CABIN MANAGEMENT SYSTEM MESSAGES

Table 1: Cabin Management System Messages

MESSAGE	LOGIC
WATER SYSTEM INOP	The water tank is empty, both pumps are faulty, or FWD/AFT drain valves failed closed or FWD/AFT drain valves lost communication.
WATER SERVICE DOOR OPEN	The water service panel door is open or the door switch has failed.
IN-FLIGHT PURGE IN PROGRESS	The in-flight purge is in progress.
IN-FLIGHT PURGE INHIBIT	The in-flight purge has been inhibited for one faulty drain mast or altitude greater than 10,000 ft, or a drain valve is faulty or has lost communication.

PRACTICAL ASPECTS

POTABLE WATER TANK SERVICING

Potable Water System Filling

The potable water system is serviced from the water service panel, located on the right side of the lower AFT fuselage. When the water service panel door is open, the service panel switch signals the water and waste system controller (WWSC) that the aircraft is in service mode.

Setting the FILL/DRAIN switch to the FILL mode closes the FWD and AFT drain valves, and opens the FILL/DRAIN valve. A quick connect fitting with a lever lock cap is located in the service panel. A fill hose is connected to the fill nipple, and FILL is selected at the FILL/DRAIN switch on the service panel.

Water is pumped into the tank from the ground service equipment. An optional water level configuration allows the preselection of the water tank to a level less than 100%. The default value is 100%. The water level indicator has + or - buttons to increase or decrease the preselect water level. The preselected level is indicated by a blue pointer.

The water level sensor monitors the level of water during the filling process and automatically closes the FILL/DRAIN valve when the preselected quantity is achieved.

During the FILL mode, both water pumps are disabled to prevent back pressure. The FULL light illuminates when the full level is reached. FLIGHT is then selected at the FILL/DRAIN switch. The fill hose can be removed and the lever lock is closed on the fill nipple. The service panel door can be closed, which enables pump operation.

Potable Water Disinfection

The potable water system is serviced from the water service panel, located on the right side of the lower aft fuselage. Water system disinfection is performed periodically according to airline operating procedures. The aircraft maintenance publication (AMP) must be followed for system disinfection. Typically the system should be filled with 100 ppm chlorine solution and allowed to soak for 1 hour. The disinfecting solution should be run through the faucets and toilet rinse valves for complete system cleaning.

NOTE

Carbon filters must be removed prior to the disinfection procedure.

WARNING

DRAIN THE POTABLE WATER SYSTEM AT LEAST ONCE EVERY THREE DAYS. IF THE WATER SYSTEM IS NOT DRAINED FREQUENTLY, BACTERIA CAN GROW IN THE SYSTEM. CONSUMING CONTAMINATED WATER MAY CAUSE ILLNESS.

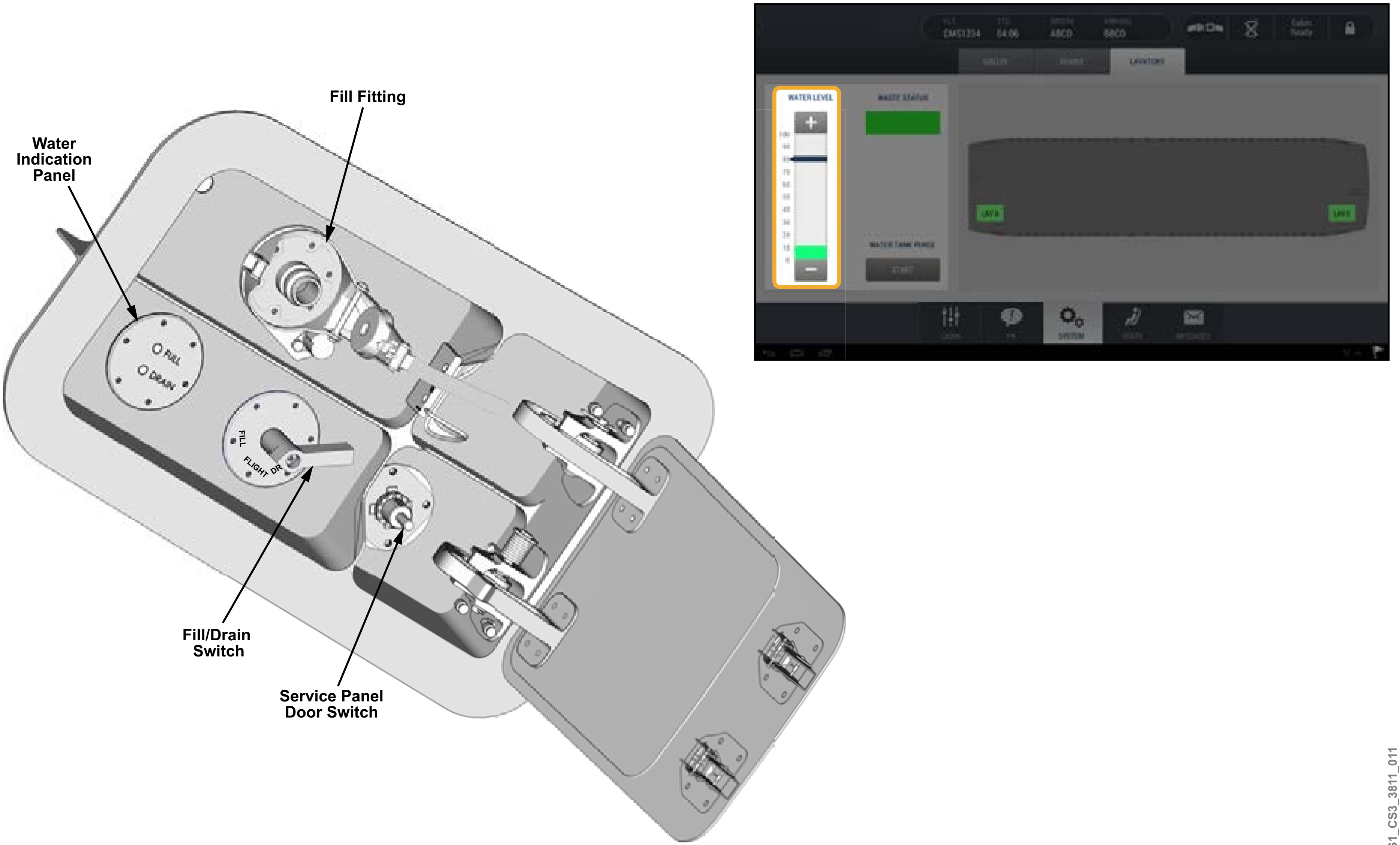


Figure 9: Potable Water Tank Servicing

Potable Water Draining

The potable water system is drained from the water service panel, located on the right side of the lower aft fuselage. If required, a suitable container should be placed under the forward and aft drain masts to catch the drained water.

Water system draining is started by selecting DRAIN on the service panel FILL/DRAIN switch. The WWSC commands the FILL/DRAIN valve, and the forward and aft drain valves, to move to their DRAIN positions. The WWSC also sends a discrete signal to the water system indication panel to illuminate the DRAIN light.

The water drains from the forward and aft drain masts are assisted by one water pump. Complete draining must be ensured by observing the drain masts. Water should stop dripping before FLIGHT is selected at the FILL/DRAIN switch.

After draining is completed, the FILL/DRAIN switch is moved to the FLIGHT position and the WWSC commands the valves to their FLIGHT position. The service panel door is closed to complete the operation.

Purging (Optional System)

In flight, the water system is purged using the WATER TANK PURGE selection on the CMS terminal. The purge function is active when:

- Aircraft is below 10,000 ft
- Landing gear is not down
- Forward drain mast heater is operational

NOTE

The forward drain valve power is routed through the forward drain mast thermostat. If the thermostat is open, the forward drain valve does not open. This prevents water from freezing on the drain mast. On the ground, the water service panel door switch overrides the forward drain mast thermostat and provides power to the forward drain valve.

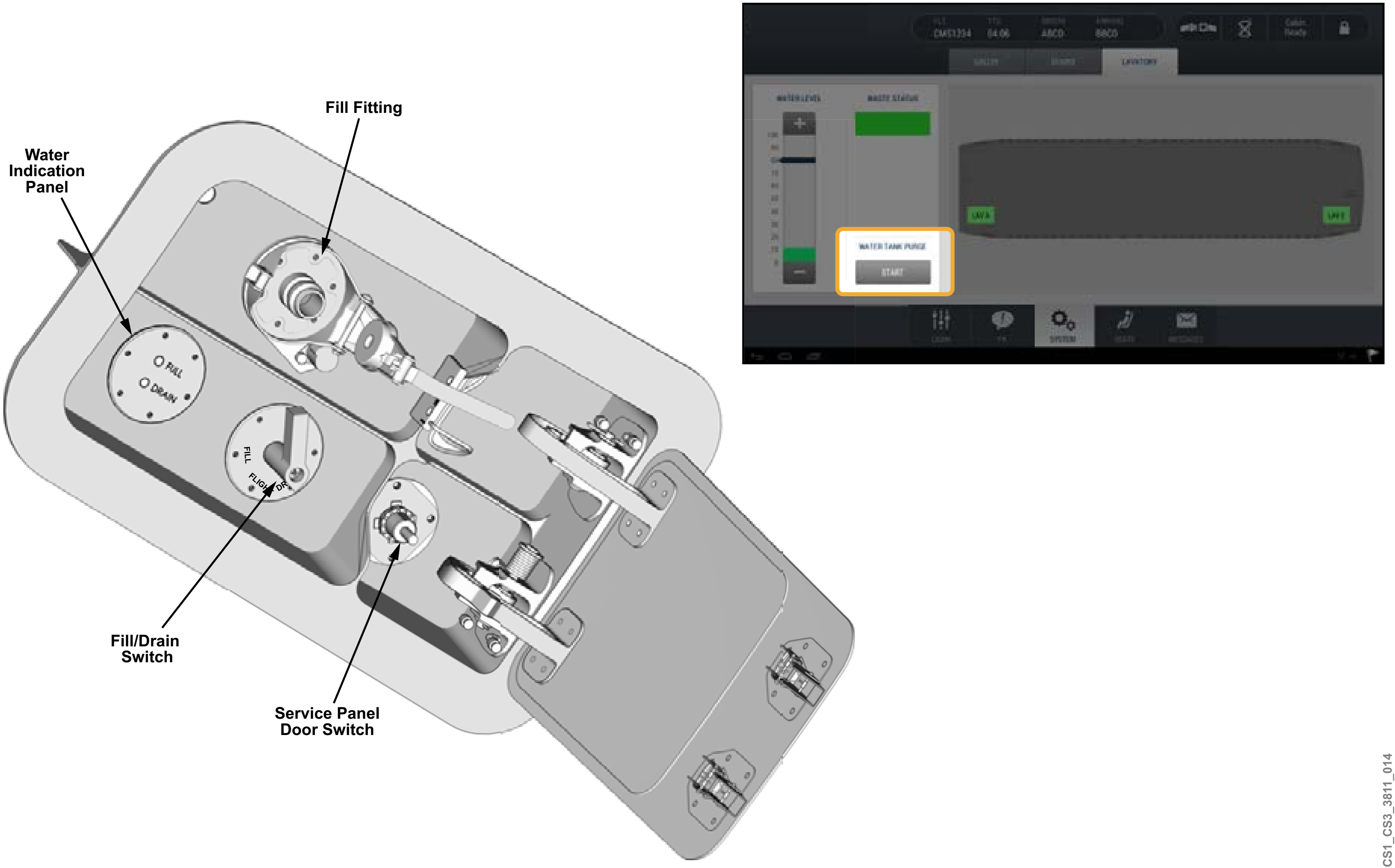


Figure 10: Potable Water Draining and Purging

38-15 WATER PUMPS

GENERAL DESCRIPTION

The potable water pumps provide pressure to the potable water distribution lines. The potable water system is pressurized by two water pumps, only one pump is required for normal operation. The pumps are installed in parallel and are located below control and distribution cabinet (CDC) 5, aft of the cargo liner. The pumps are orientated to ensure proper drainage. Each pump is heated and installed with an insulation blanket.

The pumps are gravity fed from the tank. A water pump manifold connects the two pump inlets to the supply line from the tank. The manifolds are heated and insulated to keep them from freezing.

Each water pump operates on 115 VAC power. The AC power is converted to 230 VDC, powering a DC motor that drives the centrifugal pump assembly. The maximum pressure generated at the no-flow condition is 42.6 psig.

A recirculation loop is installed, which directs some flow from the pumps back to the storage tank. The purpose of the recirculation loop is to keep a small amount of water moving through the pump during no-flow conditions on the aircraft and to keep the pumps cool. An orifice installed at the tank restricts the flow rate to approximately 1.0 gpm through the recirculation loop.

A water pressure switch monitors the water line pressure. In the case of low-pressure, a signal is sent to the WWSC which automatically shuts down the active pump and turns on the idle one.

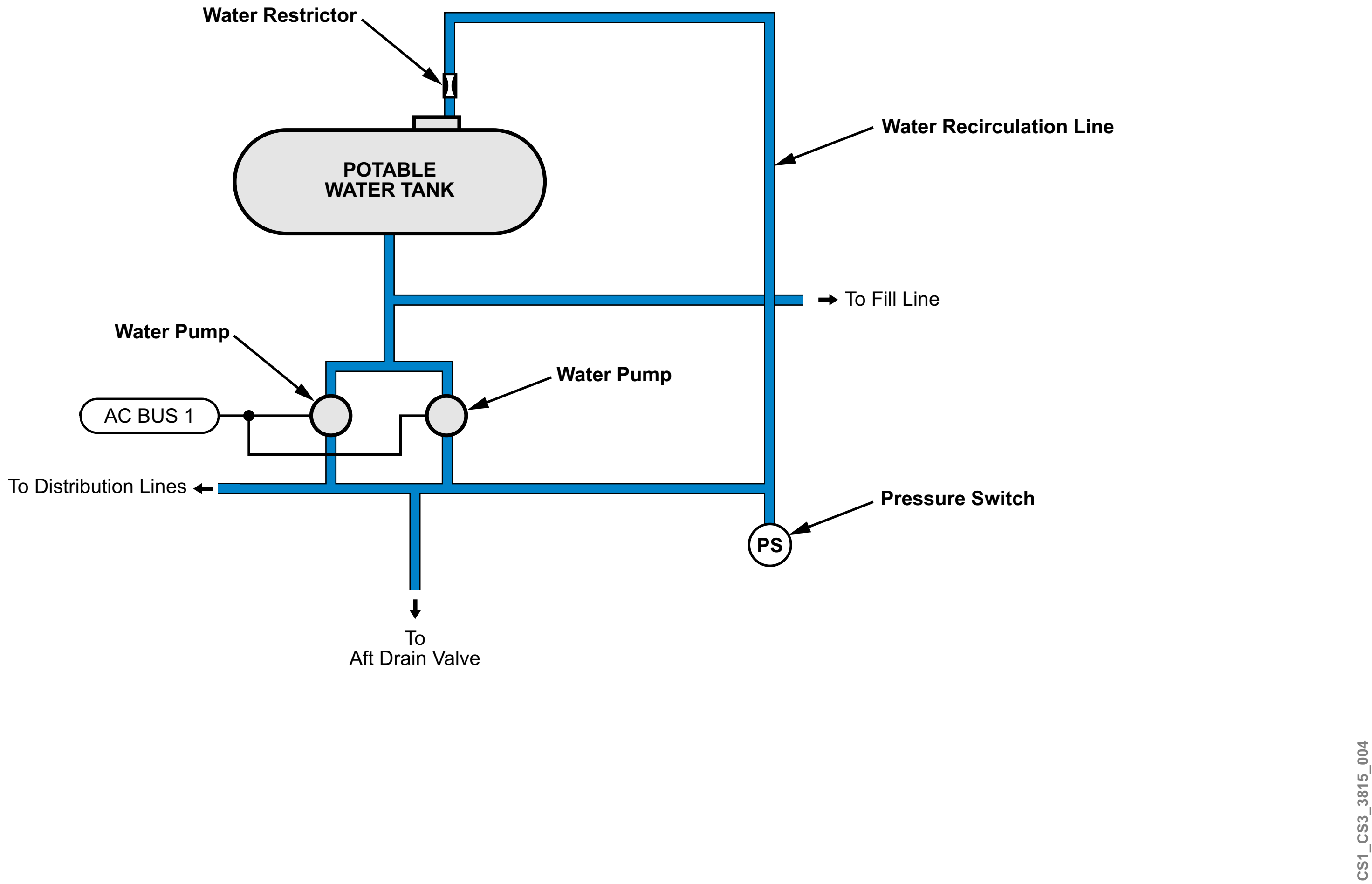


Figure 11: Water Pumps - Potable Water Tank Pressurization System

COMPONENT LOCATION

The following components are installed in the potable water pressurization system:

- Water pump

WATER PUMPS

The water pumps are installed forward of the potable water tank.

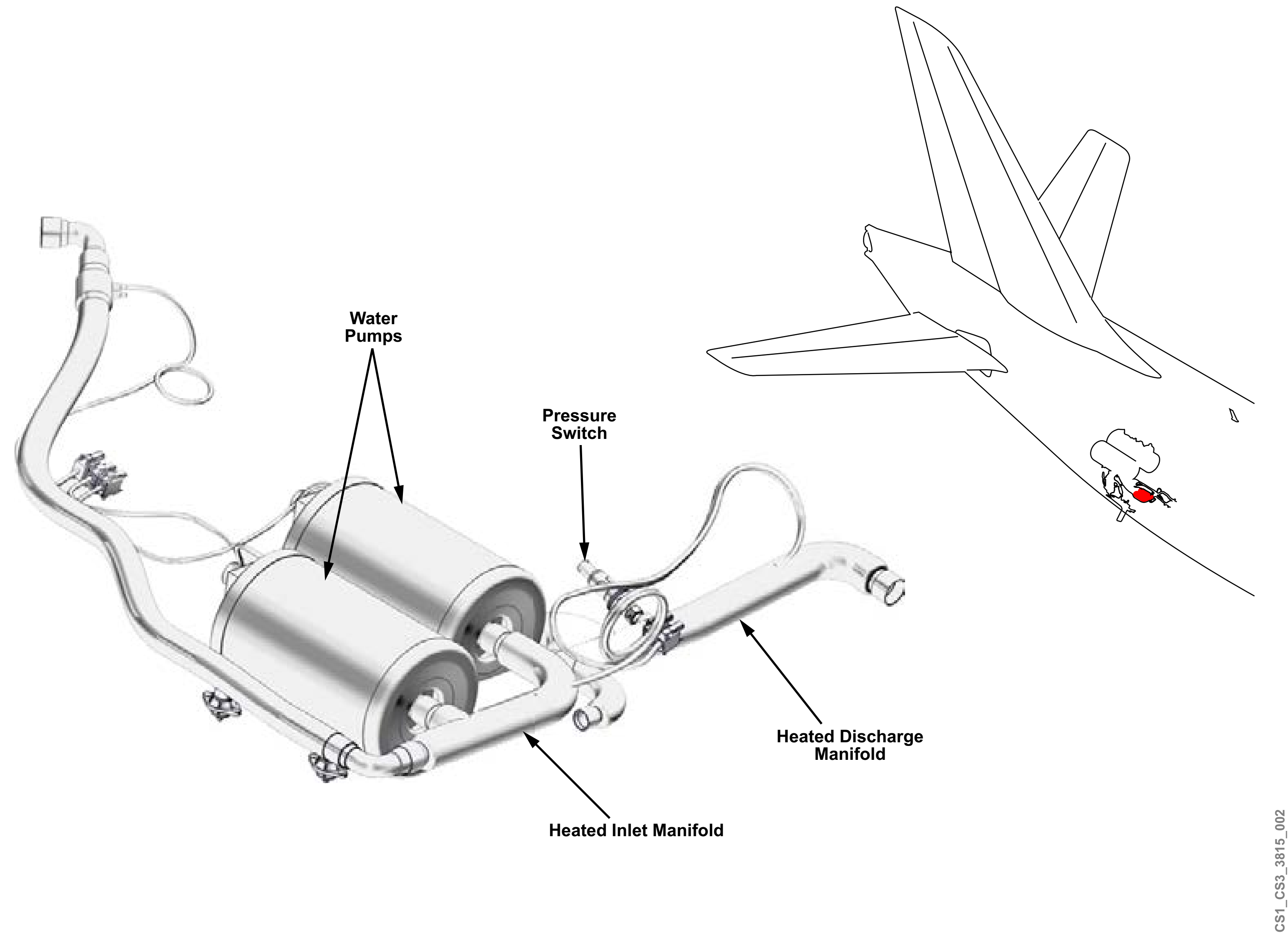


Figure 12: Water Pressurization - Component Location

DETAILED DESCRIPTION

The water pumps are controlled by the water and waste system controller (WWSC). The WWSC sends a discrete signal to CDC 5 to provide power to the water pumps.

The water pump continuously monitors its flow and temperature while running. In case of an overheat or overcurrent, the water pump reports the failures to the WWSC. The WWSC shuts down the pump and switches to the other pump. If the overheat or overcurrent discrete is set three consecutive times, the WWSC sends a fault message to the onboard maintenance system (OMS) and does not switch back to the defective pump.

The water pressure switch provides a low water pressure indication to the WWSC. When the water pressure switch indicates low-pressure, the WWSC turns off the operating pump and turns on the idle pump. If the water pressure continues to indicate low-pressure, the WWSC faults the water pressure switch and reverts the pump to the previously operating pump.

When the FILL/DRAIN switch is selected to the FLIGHT position, the WWSC switches to the idle pump. The WWSC operates one pump during water system draining.

The WWSC alternates system operation between water pump 1 and water pump 2 after every servicing. One pump operates when the potable water tank is draining.

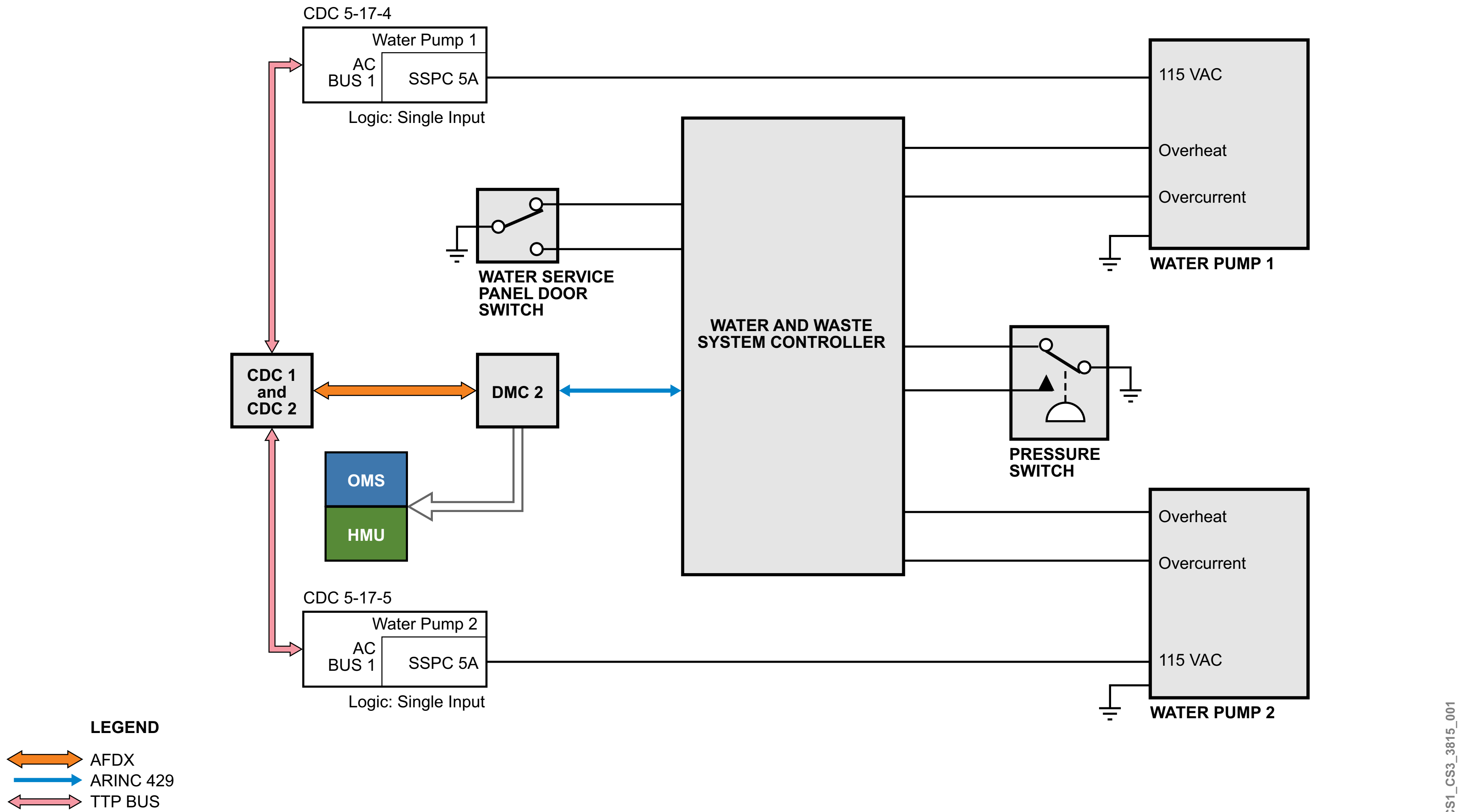


Figure 13: Water System Pressurization Schematic

30-70 WATER AND WASTE HEATERS

GENERAL DESCRIPTION

The water and drain line heaters are provided to prevent ice formation in the water and drain systems in unheated areas of the aircraft. 115 VAC power is used for heating. The heaters are provided for the potable water service panel fittings, the waste service panel fittings, drain masts, water distribution, and drain lines.

The water line system includes the in-line heaters and the integrally heated hose assemblies. The main distribution water line is heated by three in-line heaters. The branch water lines to the monuments, and from the water tank to the main line, are heated by integrally heated hose assemblies. The heated hose assemblies have built-in temperature sensors that provide input to the water and waste system controller (WWSC) to control the heater operations.

Two external electric heated blankets are installed on the potable water tank. The FILL/DRAIN valve is heated and covered with an insulation blanket.

The two water pumps, and the inlet and discharge manifolds are heated. Each pump has an insulation blanket installed over it.

The drain masts are provided with a heater to prevent freezing, and to allow water drainage at low temperatures.

The forward and aft drain valves are heated and covered with an insulation blanket.

Temperature sensors are installed in a waste rinse fitting heater, water fill fitting heater, water tank blanket heaters, in-line heaters, heated hose assemblies, drain valves, pumps, and associated manifolds, to monitor the temperature and control the power applied to the heaters. The temperature sensors installed in the drain masts only monitor the temperature.

The waste tank drain valve and rinse fitting are heated. The water tank fill valve is freeze protected by a separate gasket heater that is mounted between the service panel and the valve flange.

COMPONENT LOCATION

The following components are installed in the water and waste heater system:

- Water tank heater
- Forward and aft drain mast heaters
- Water line heaters
- In-line heaters
- Integrally heated hoses
- Gasket heaters

WATER TANK HEATER

The electric heater blankets are installed on the potable water tank.

FORWARD AND AFT DRAIN MAST HEATERS

The two heated drain masts are located on the external underside of the aircraft at the longitudinal centerline.

WATER LINE HEATERS

The water line heaters are located in areas of the aircraft that are susceptible to freezing. Heaters are also installed at the connection points on the Y and T fittings.

GASKET HEATERS

Gasket heaters are located at the water service panel fill fitting, waste service panel drain valve, and rinse valves.

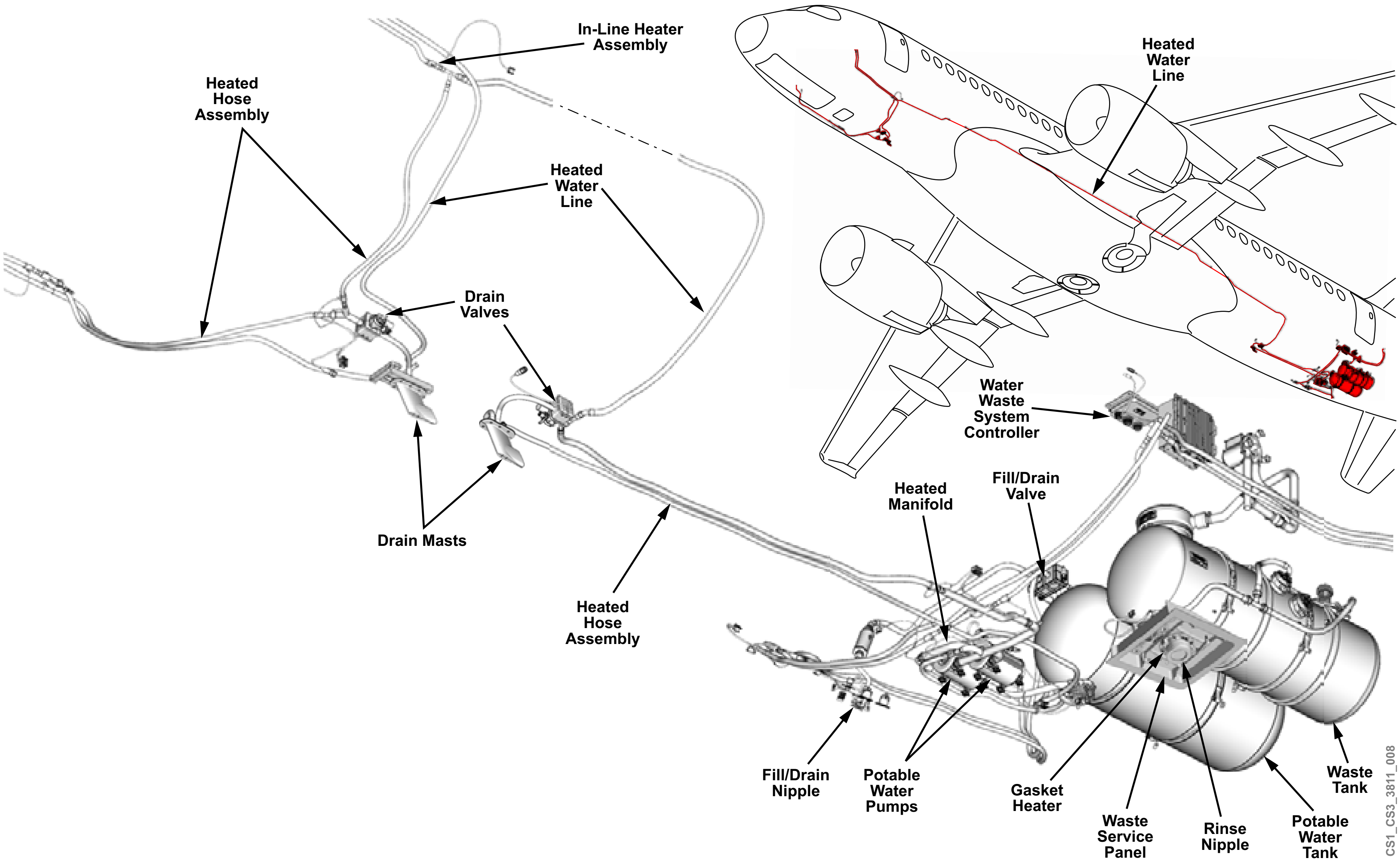


Figure 14: Water and Waste Heaters

COMPONENT INFORMATION

All heaters receive power from AC BUS 1. The heating elements sensors provide a signal to turn power on when the temperature sensed is less than 5°C (41°F). The power is removed when temperature increases above 10°C (50°F).

WATER TANK HEATER

The electric heater blankets are installed on the potable water tank. The blanket is attached to the water tank with adhesives. One heated blanket is wrapped around the forward portion of the tank, and one around the aft portion of the tank.

FORWARD AND AFT DRAIN MAST HEATERS

The two heated drain masts are located on the external underside of the aircraft at the longitudinal centerline.

A low-temperature thermostat is embedded in the drain mast to prevent draining through the forward drain valve in case of a drain mast heater failure. The safety thermostat opens the power circuit to the forward drain valve and prevents the draining of potable water.

The WWSC monitors the temperature of the drain mast through a temperature sensor in the drain mast.

WATER LINE HEATERS

There are two types of water line heaters used in the water system: in-line heaters and integrally heated hoses. There are also heated Y and T fittings at connection points.

In-line Heaters

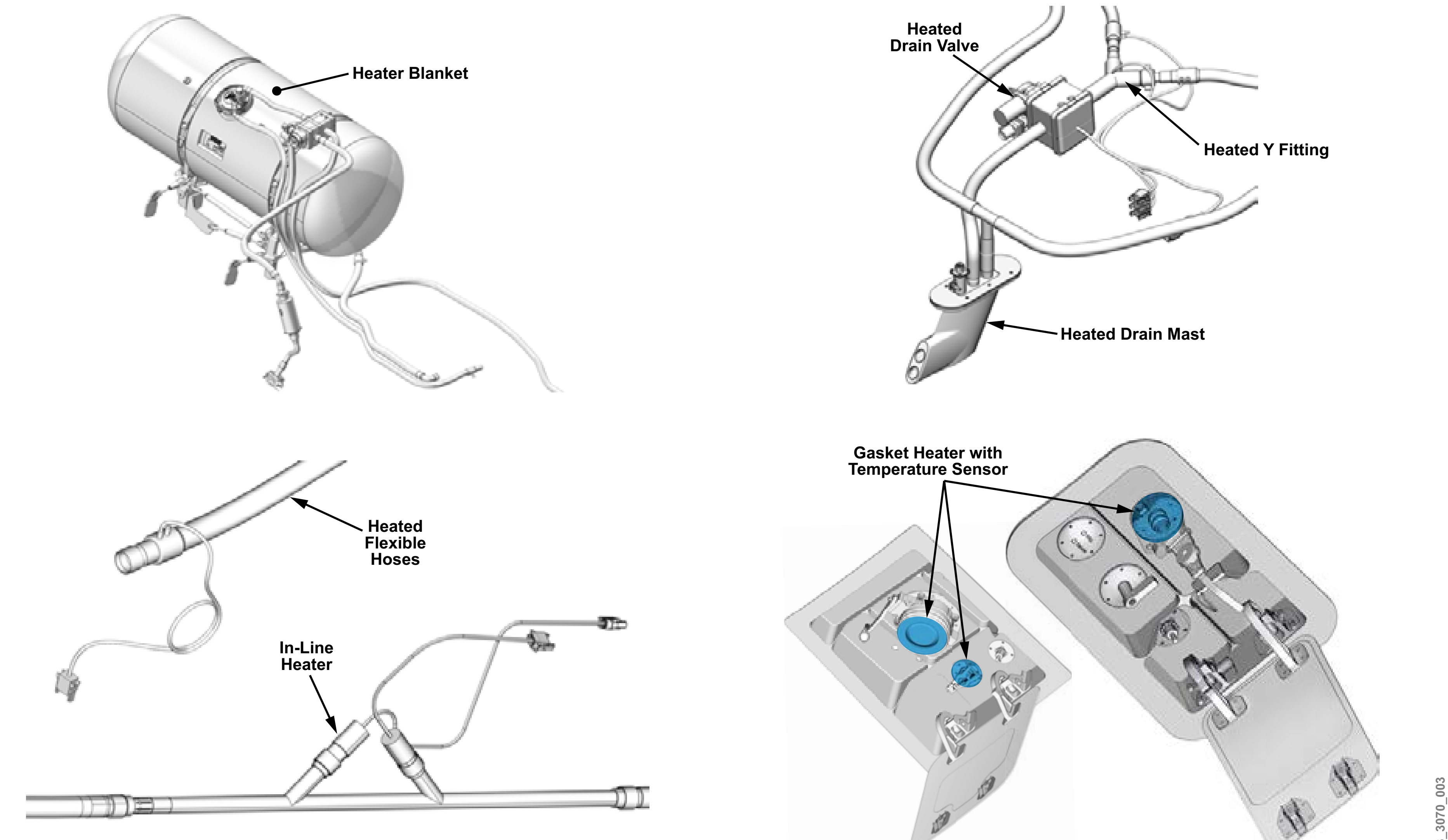
The heater element is inside a sealed Teflon tube that gets inserted inside the water line. The assembly installs in such a way that the heating element is physically located inside the hose, providing direct contact with the water. The in-line heater does not have an internal thermostat. The heater is switched on and off by the WWSC, which commands the CDC to switch power based on external temperature sensors.

Integrally Heated Hoses

The heated flexible hoses have the heater element built into the hose assembly. The heated hoses meet the same pressure requirements as the non-heated flexible hoses. The heated hoses have built-in temperature sensors and are controlled in the same manner as the in-line heater described above.

GASKET HEATERS

Gasket heaters with built-in temperature sensors are used on the water service panel fill fitting, waste service panel drain valve, and rinse valves. The gasket heaters are installed between the panels and the fittings or valves.



NOTE

All heaters are powered from AC BUS 1. Heaters turn on at 5°C (41°F) and turn off at 10°C (50°F).

Figure 15: Water and Waste Heater Components

DETAILED DESCRIPTION

Temperature sensors are installed in a waste rinse nipple heater, water fill nipple heater, water tank blanket heaters, in-line heaters, heated hose assemblies, drain valves, pumps and associated manifolds, and drain masts to monitor temperatures and control the power applied to the heaters. The temperature sensors provide a discrete signal to the water and waste system controller (WWSC).

The heaters are divided into seven heater groups and are controlled independently. A heater group is turned on when any sensor in a heater group indicates the temperature is falling below 5°C (41°F), and is turned off when all sensors in the heater group indicate the temperature is above 10°C (50°F).

All of the water and waste system heaters receive 115 VAC power from either control and distribution cabinet (CDC) 3 in the forward zone, or CDC 5 in the aft zone. The WWSC sends a heating command through DMC 2 to CDC 1 then from CDC 1 over the TTP BUS to CDC 3 to power the forward heaters and CDC 5 to power the aft heaters.

The WWSC monitors the drain mast temperature through a temperature sensor within the drain mast.

In case of heater failure, including thermostats out of range or no signal, the WWSC shuts down the related heater and sends a message to the onboard maintenance system (OMS).

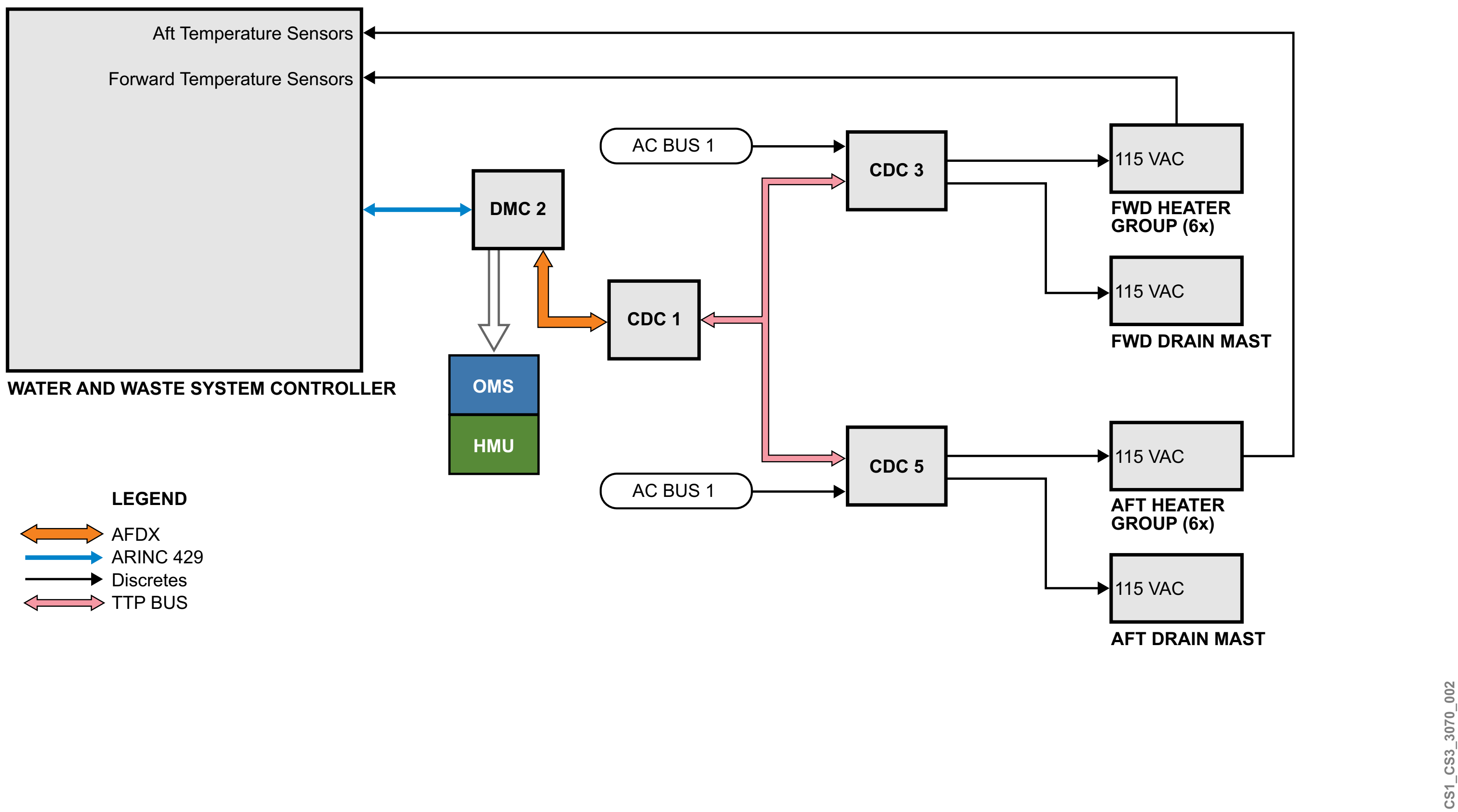


Figure 16: Water Heater - Detailed Description

MONITORING AND TEST

The following page provides the cabin management system (CMS) water and waste heaters displayed messages.

CABIN MANAGEMENT SYSTEM MESSAGES

Table 2: Cabin Management System Messages

MESSAGE	LOGIC
FWD DRAIN MAST FAULT	The low-temperature thermostat indicates a temperature less than 5ºC (41°F).
AFT DRAIN MAST FAULT	The low-temperature thermostat indicates a temperature less than 5ºC (41°F).

38-12 GALLEY AND LAVATORY POTABLE WATER

GENERAL DESCRIPTION

The galley is supplied with water through a shutoff valve. The shutoff valve isolates the galley water system from the distribution system. The manually-operated valve is open when the handle is in-line with the water lines.

The galley water distribution is based on the galley configuration. A typical galley has a spigot for dispensing water, and provisions for coffee makers.

An optional replaceable filter is installed in the galley structure.

COMPONENT LOCATION

The following component is installed in the lavatory and galley water system:

- Water shutoff valve

WATER SHUTOFF VALVE

The water shutoff valve is located on the galley structure.

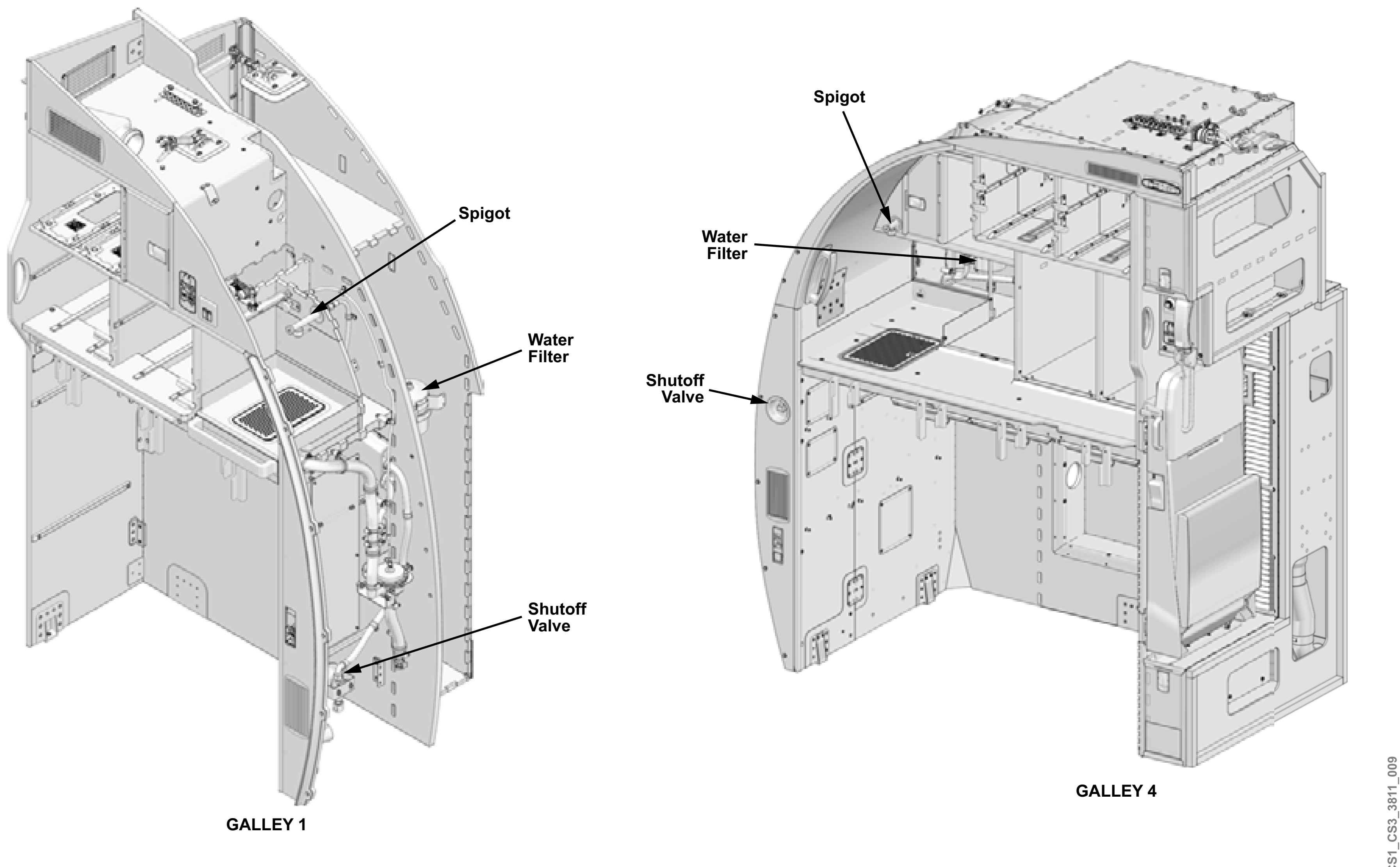


Figure 17: Galley Potable Water

LAVATORY POTABLE WATER SYSTEM

The lavatory potable water system provides water to the wash basin and toilet.

The water is supplied through a shutoff valve and is supplied to the water heater and cold water inlet of the water mixer. The water supply is also routed to the toilet to provide rinse water during the toilet flush cycle.

The shutoff valve is located at the base of the lavatory cabinet. The lavatory water shutoff valve isolates the lavatory potable water system from the distribution system. The manually operated valve is open when the handle is in-line with the water lines. Access to the valve is through the cabinet door.

The potable water heater is installed in the lavatory below the sink. The water heater is powered by 115 VAC. The water temperature is controlled by thermostats. A water level sensor provides water level information to the control circuit. When the water level sensor detects enough water inside the heater tank, the heater turns on.

The faucet has one button on the top to activate the water flow. When the button is pushed, a signal is sent to the lavatory potable water mixer, which opens the water flow.

The mixer has two inlets: one for cold water coming from the potable water distribution system, and one for the hot water coming from the water heater. There is one outlet connected to the lavatory faucet. The mixer is powered by 28 VDC.

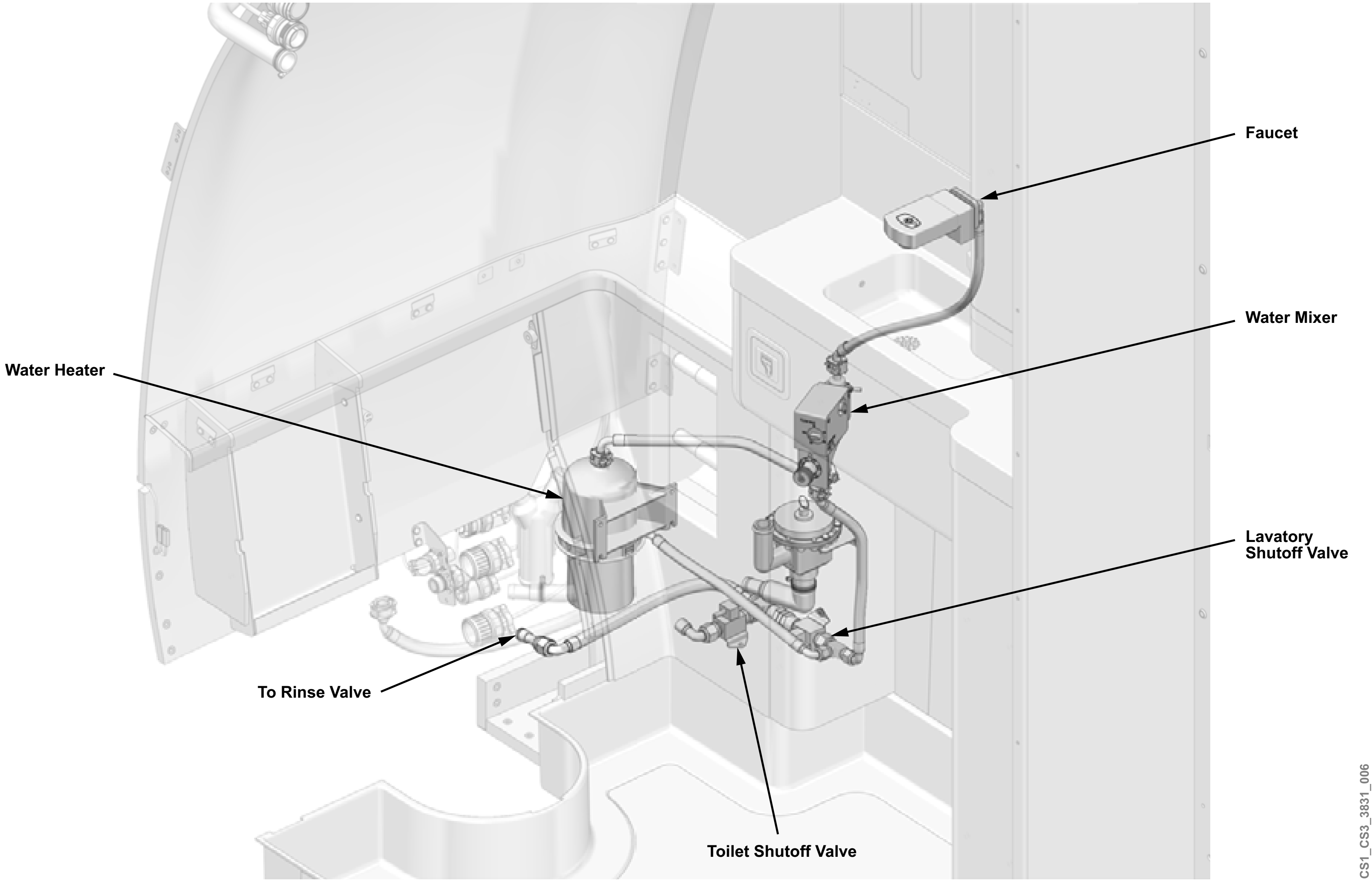


Figure 18: Lavatory Potable Water System

COMPONENT INFORMATION

WATER HEATER

There is one power switch used to turn the water heater on and off, and an light-emitting diode (LED) that indicates power is on. The water heater is powered by 115 VAC. The water temperature is controlled by thermostats. Three temperature switches provide backup protection and shut the heater off if the water temperature goes out of the normal temperature range. The water heater control circuit also shuts off the heater for an overcurrent or open load condition. A water level sensor provides water level information to the control circuit. When the water level sensor detects enough water inside the heater tank, the heater turns on.

FAUCET

In case the solenoid mechanism fails, the rinse valve closes. The water inlet and the electrical harness are connected to the lavatory potable water mixer. The faucet receives 28 VDC from the mixer.

WATER MIXER

When the water faucet open/close button is pushed, a signal is sent to the mixer. The rinse valve opens, letting the mixed water flow to the water outlet.

The water mixer has a knob located on the side of the body and adjusts a thermostat to set the outlet temperature. A second knob located in the front of the main body sets the water flow time.

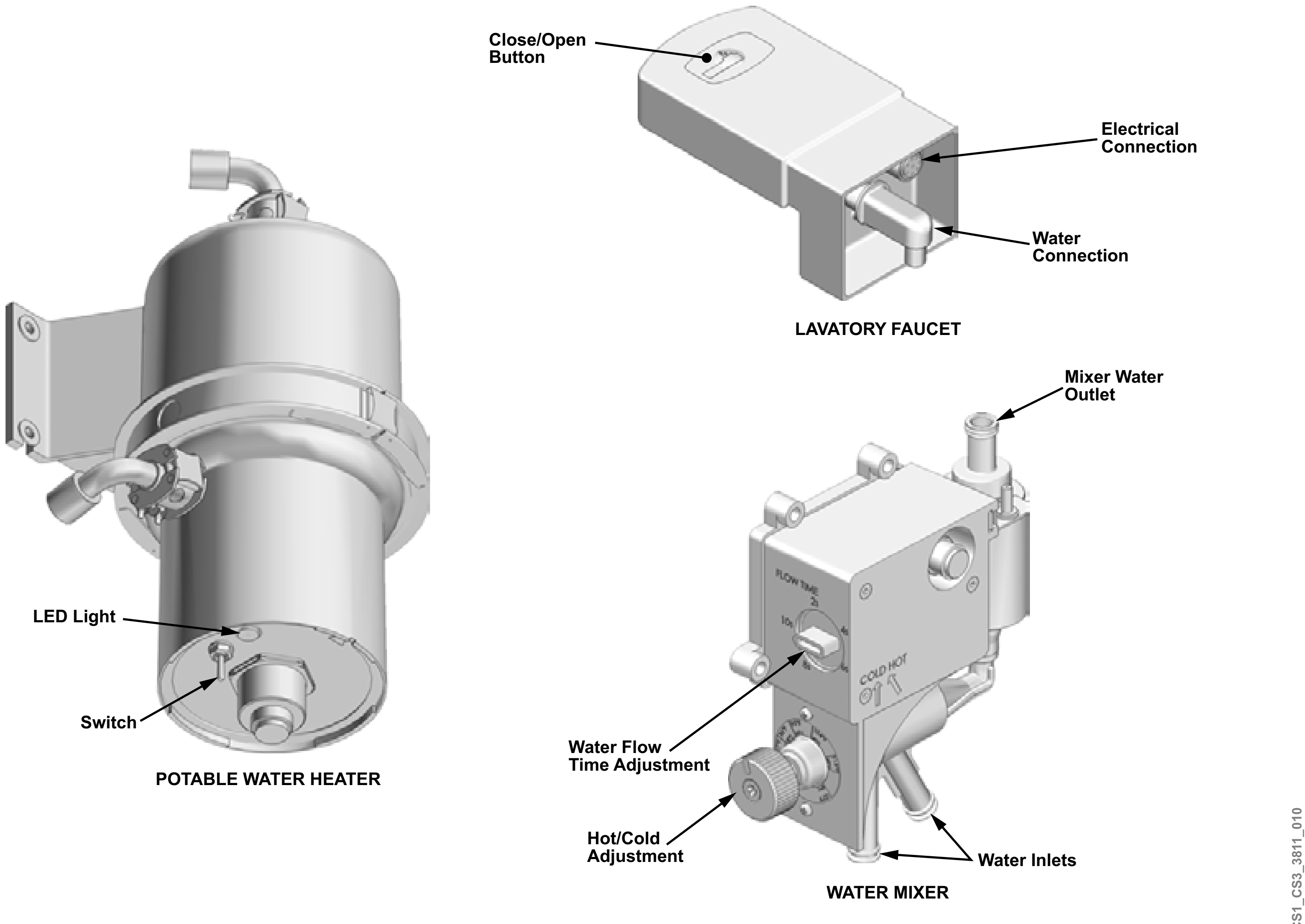


Figure 19: Lavatory Potable Water - Component Information

38-14 WATER QUANTITY CONTROL AND INDICATION

GENERAL DESCRIPTION

A capacitance level sensor measures the water tank quantity. The sensor consists of a capacitance rod, electronics, and housing.

The water quantity is reported to the water and waste system controller (WWSC) for display on the cabin management system (CMS) crew terminal.

The WWSC also uses the sensor signal to close the FILL/DRAIN valve when the water quantity reaches the preselected level.

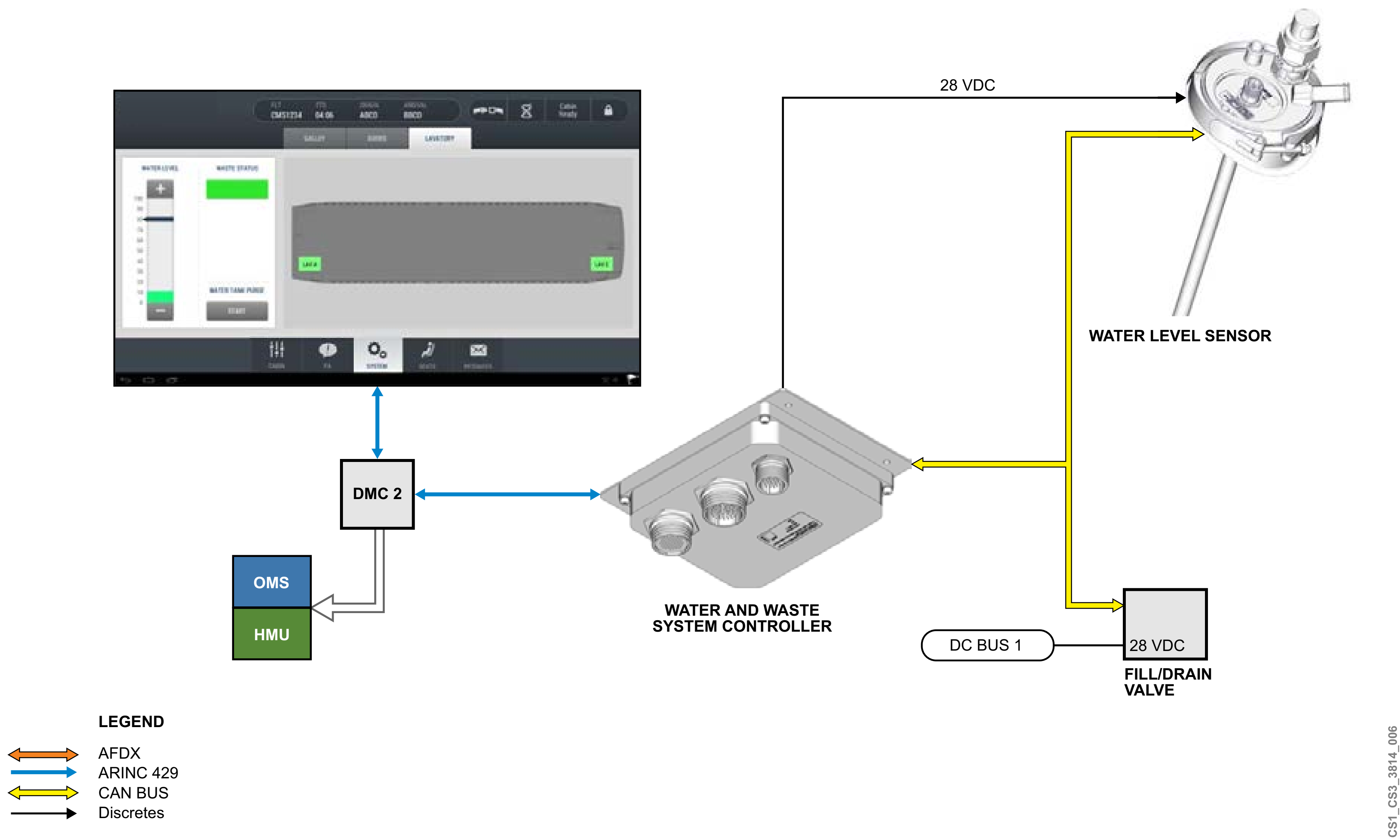


Figure 20: Water Quantity - General Description

COMPONENT LOCATION

The following component is installed in the water quantity indication system:

- Water level sensor

WATER LEVEL SENSOR

The water level sensor is installed on top of the water tank.

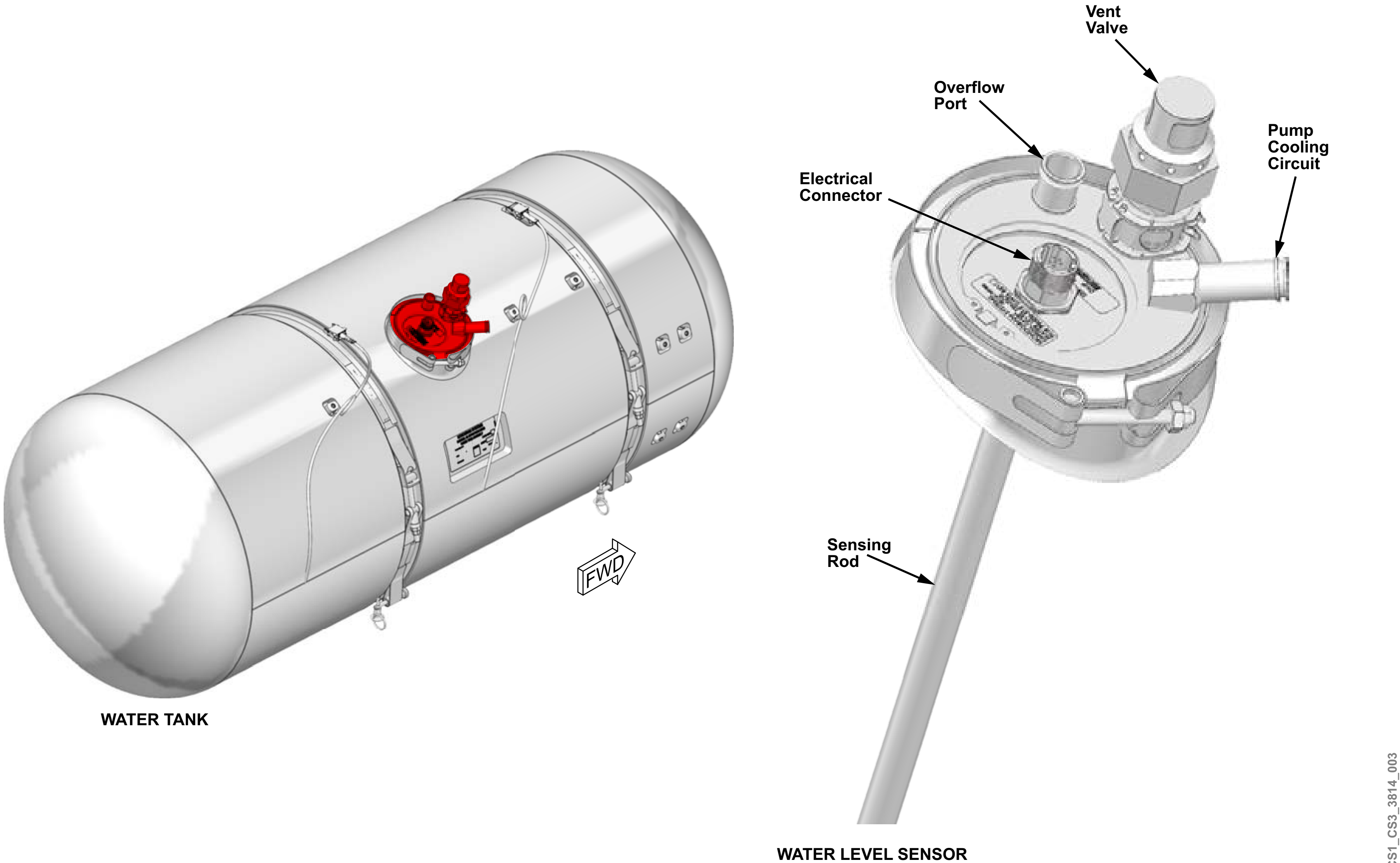
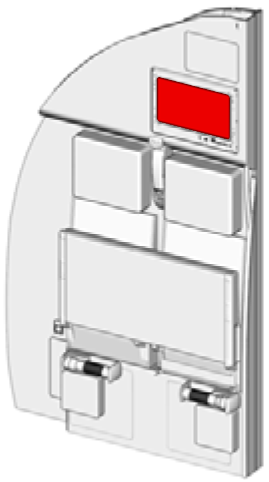


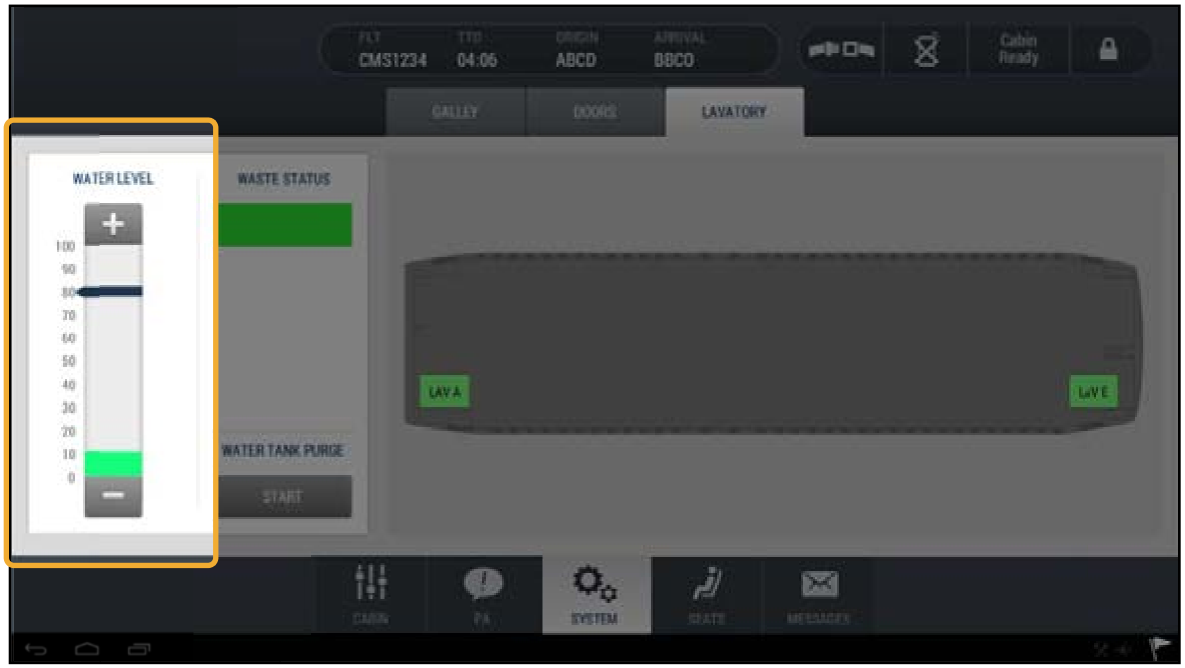
Figure 21: Water Quantity Components

CONTROLS AND INDICATIONS

The water level sensor sends water level information to the WWSC. This information is sent through the right data concentrator unit module cabinet (DMC) to the cabin management system (CMS). On the LAVATORY page, the water level is indicated by a green graduated bar that displays the level in 10% increments.



FORWARD FLIGHT ATTENDANT SEATS



CREW TERMINAL

CS1_CS3_3811_016

Figure 22: Cabin Management System Lavatory Page

DETAILED DESCRIPTION

The sensor is powered by DC BUS 1 and interfaces with the WWSC via a CAN BUS. The water level is displayed on the CMS water and waste page.

The measured capacitance changes proportionally with the length of the rod that is submerged in water. The sensor uses the capacitance output to calculate the quantity of water in the tank.

The WWSC also uses the sensor signal to close the FILL/DRAIN valve when the water quantity reaches the preselected level.

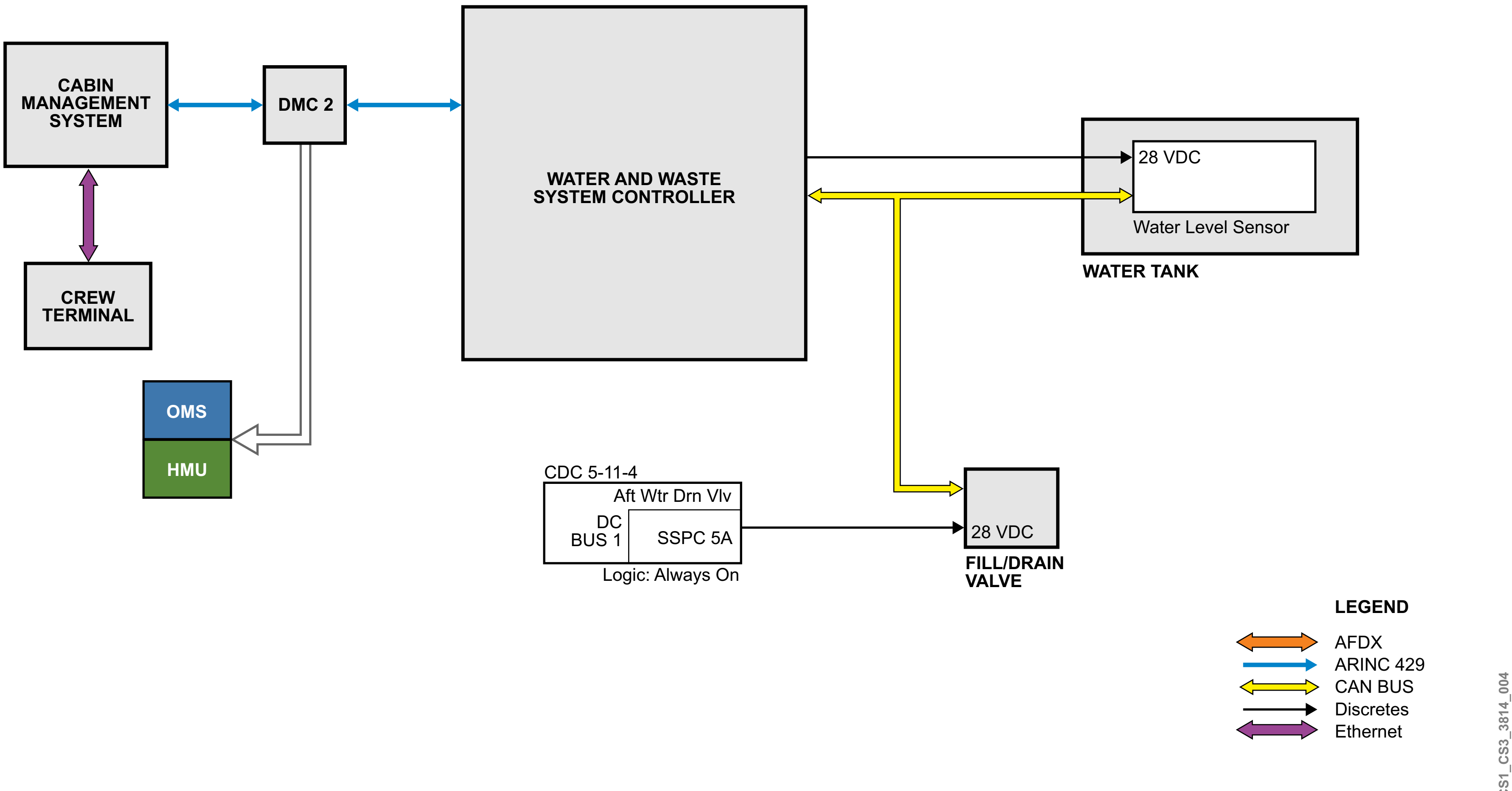


Figure 23: Water Quantity - Detailed Description

38-33 GRAY WATER DRAINS

GENERAL DESCRIPTION

The gray water drain system drains waste water from lavatory and galley sinks through electrically heated drain masts.

The strainer at the bottom of the sink prevents large particles from entering the drain lines. The gray water flows through the autodrain valve and exits the aircraft from the drain mast. The aircraft is divided into two sections, forward and aft, which have their own respective drain lines, valve, and drain mast.

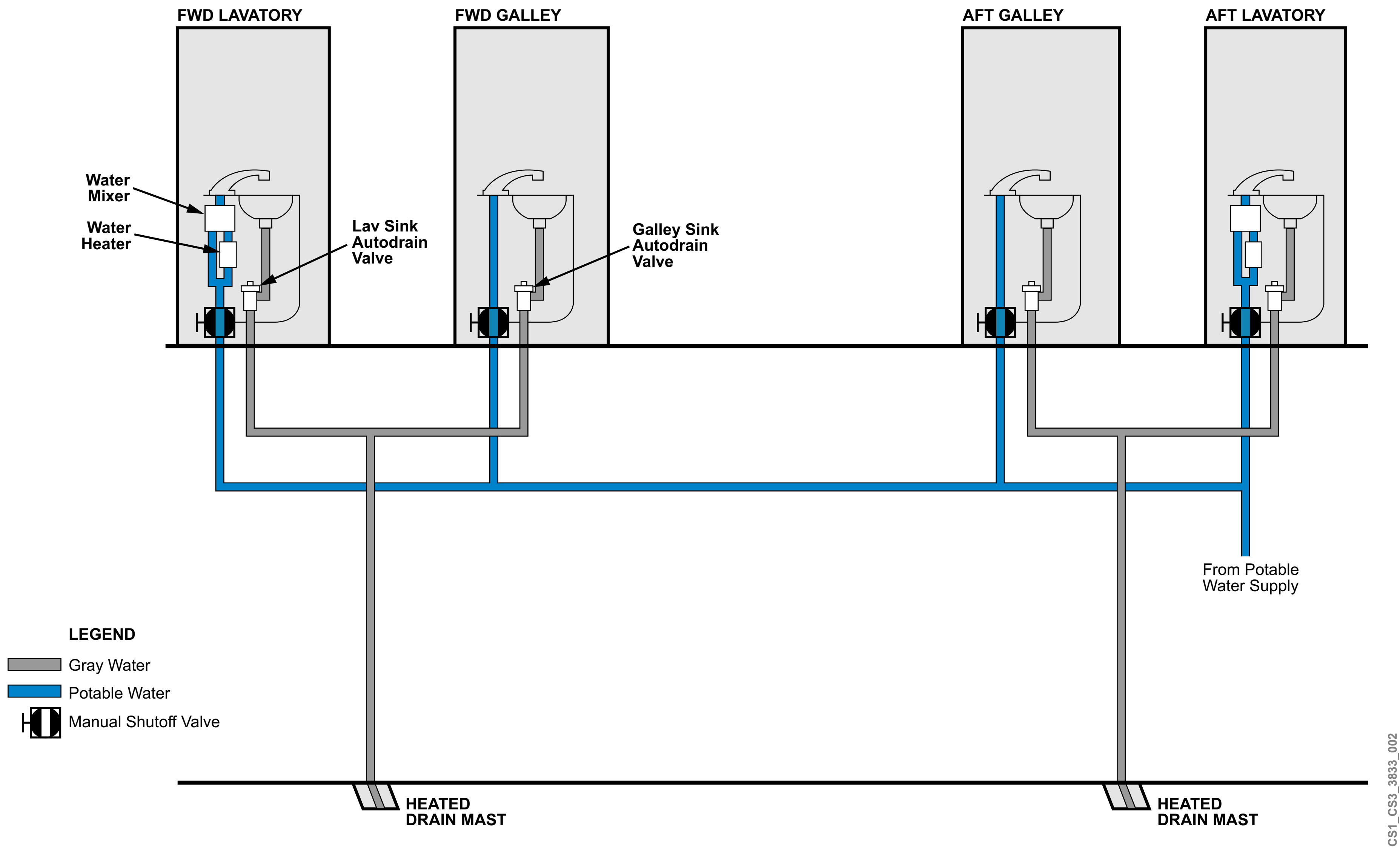


Figure 24: Gray Water Drains

COMPONENT LOCATION

The following components are installed in the gray water drain system:

- Drain mast
- Autodrain valves

DRAIN MAST

The two heated drain masts are located on the external underside of the aircraft at the longitudinal centerline.

AUTODRAIN VALVES

The autodrain valves are located in each lavatory and galley under the respective sinks.

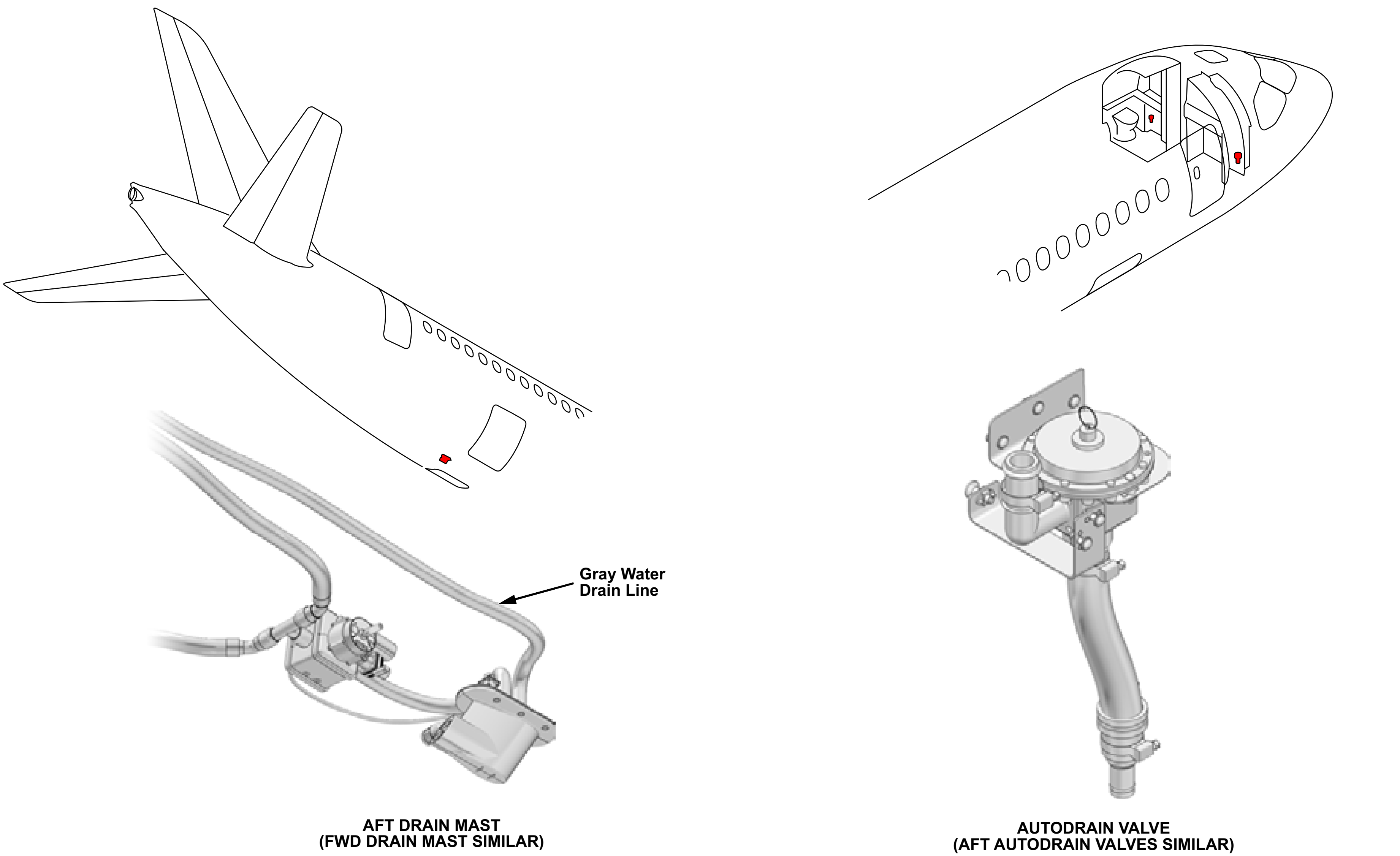


Figure 25: Gray Water Drains Component Location

COMPONENT INFORMATION

AUTODRAIN VALVE

The autodrain valve has a ball float that closes off the drain and reduces noise in the galley and lavatories. During normal conditions, gray water lifts the ball float and drains freely from the valve.

The autodrain valve also provides backflow protection. During backflow conditions, if gray water from other parts of the system enters the outlet, a ball float within the valve shuts off the autodrain outlet, preventing water from flowing into the galley or lavatory sink drains.

The valve has a manual release lever to open the ball valve in the event of a blockage.

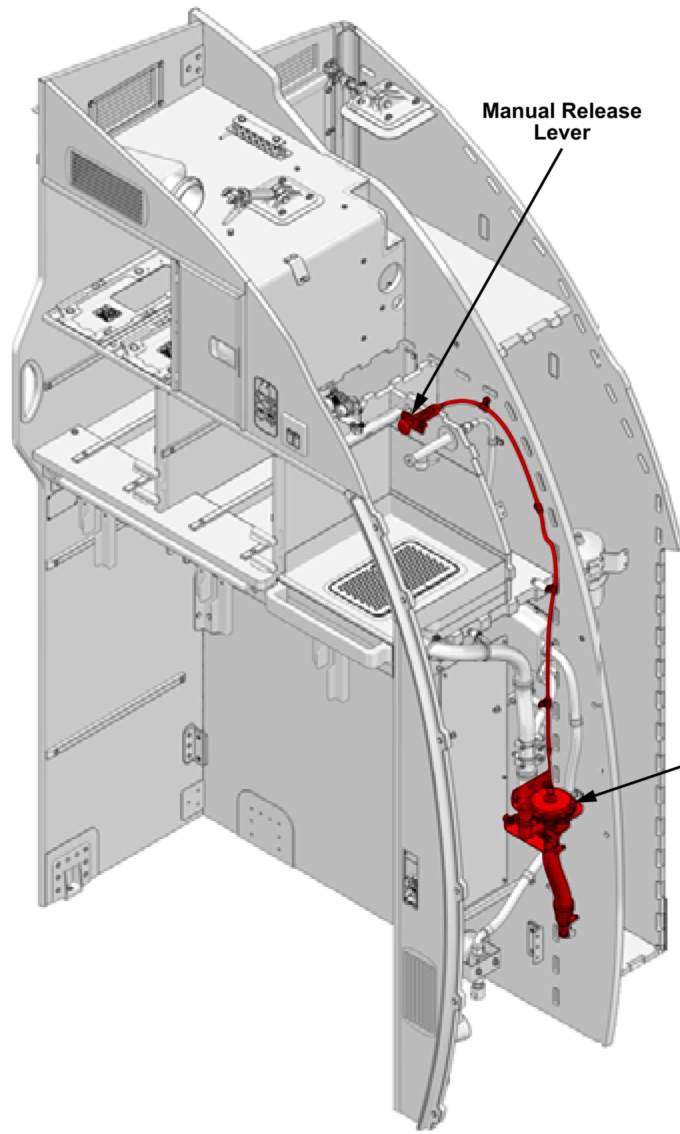


Figure 26: Autodrain Valve

38-31 VACUUM WASTE SYSTEM

GENERAL DESCRIPTION

The vacuum waste system provides servicing interfaces, waste storage, level indication, and system control for collecting and holding toilet waste from lavatory toilets. The waste tank is located behind the aft cargo compartment with a usable volume of 144 L (38 gal).

The waste tank provides storage for waste. The toilet assemblies are connected to the waste tank by 5 cm (2 in.) diameter waste transport lines located under the main deck floor beams of the aircraft.

The waste system uses vacuum suction to draw waste products from the toilets into a waste tank. The vacuum is generated by cabin differential pressure in flight, or a vacuum generator, when the aircraft is on the ground or at low altitudes.

The vacuum generator is powered by a 115 VAC 3-phase, 400 Hz motor. The vacuum generator has an impeller and cooling fan mounted on a common drive shaft. The fan provides cooling when the vacuum generator is operating. The vacuum generator has overheat protection that shuts down the generator until it cools. A vacuum generator motor drive (VGMD), powered by AC BUS 1, receives inputs from the WWSC and the toilet assembly via a CAN BUS. When a flush command is received, the VGMD powers the vacuum generator.

A spring-loaded flapper check valve is installed in the overboard vent line parallel with the vacuum generator, to control airflow so that the vacuum generator does not recirculate its own discharge air.

At altitudes above 16,000 ft, the check valve allows air from the waste tank to vent overboard without having to pass through the vacuum generator.

When the flush switch is pressed, the toilet is supplied with rinse water from the potable water system. After a short delay to ensure there is sufficient vacuum, the flush valve opens and the waste is transported from the toilet to the waste tank. The toilet assemblies receive power from DC BUS 1.

The waste arrives at the waste inlet and is deposited in the waste tank. The waste tank has components to decelerate the waste and separate water from the airstream. These components prevent waste from going overboard through the overboard vent. The vacuum air exhausts through the water separator before being vented overboard.

The waste tank level is monitored by ultrasonic point level sensors. If the waste tank is full, the vacuum waste system is disabled.

The waste system is serviced from a waste service panel located below and adjacent to the waste tank. The waste system service panel has a manually-operated ball valve installed to permit waste draining. A rinse fitting is installed in the service panel to allow waste tank rinsing. Two rotating rinse nozzles in the waste tank distribute the rinse water to clean the interior surfaces of the waste tank. The fittings are protected from freezing by gasket heaters.

If the drain valve is not in the fully closed position, the service panel door does not close.

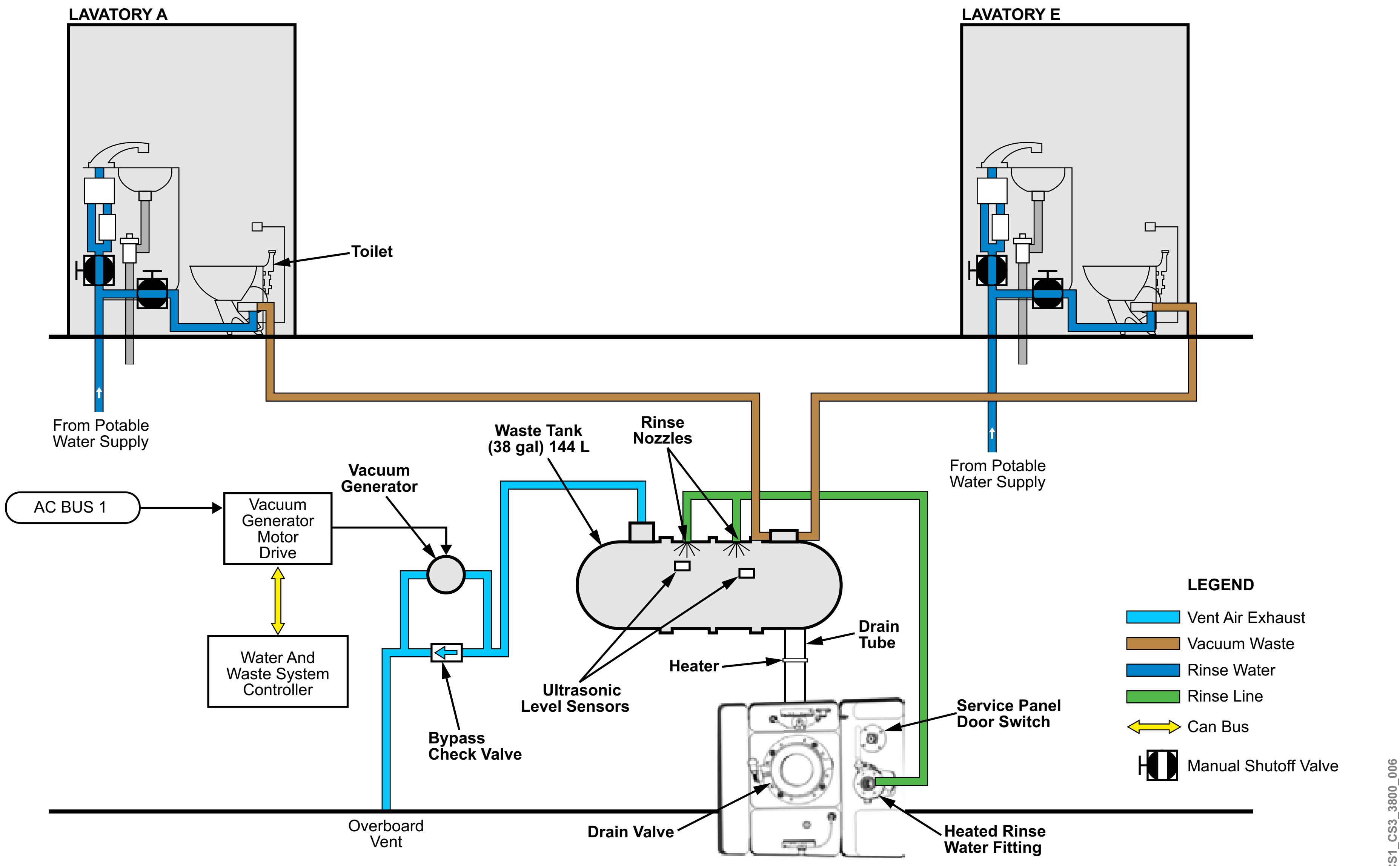


Figure 27: Vacuum Waste System

WASTE SERVICE PANEL

The vacuum waste system service panel is located on the aft lower section of the aircraft. The service panel drain valve and waste tank rinse fitting are installed on the service panel.

Drain Valve

A drain valve is used to drain the waste tank assembly during servicing. The drain valve is a ball valve, manually operated by a handle. A gasket heater prevents freezing of the valve.

A flexible, preformed drain line connects the waste tank drain to the service panel ball valve located in the waste service panel.

Rinse Fitting

The heated service panel rinse fitting is used to rinse the waste tank after drainage, and to provide a means of adding precharge to the waste tank during ground servicing.

The fitting has a lever lock with sealing cap to prevent leakage of any residual rinse water that remains in the rinse line after servicing.

The rinse fitting includes an integral heater and temperature sensor to keep any residual water in the rinse line from freezing.

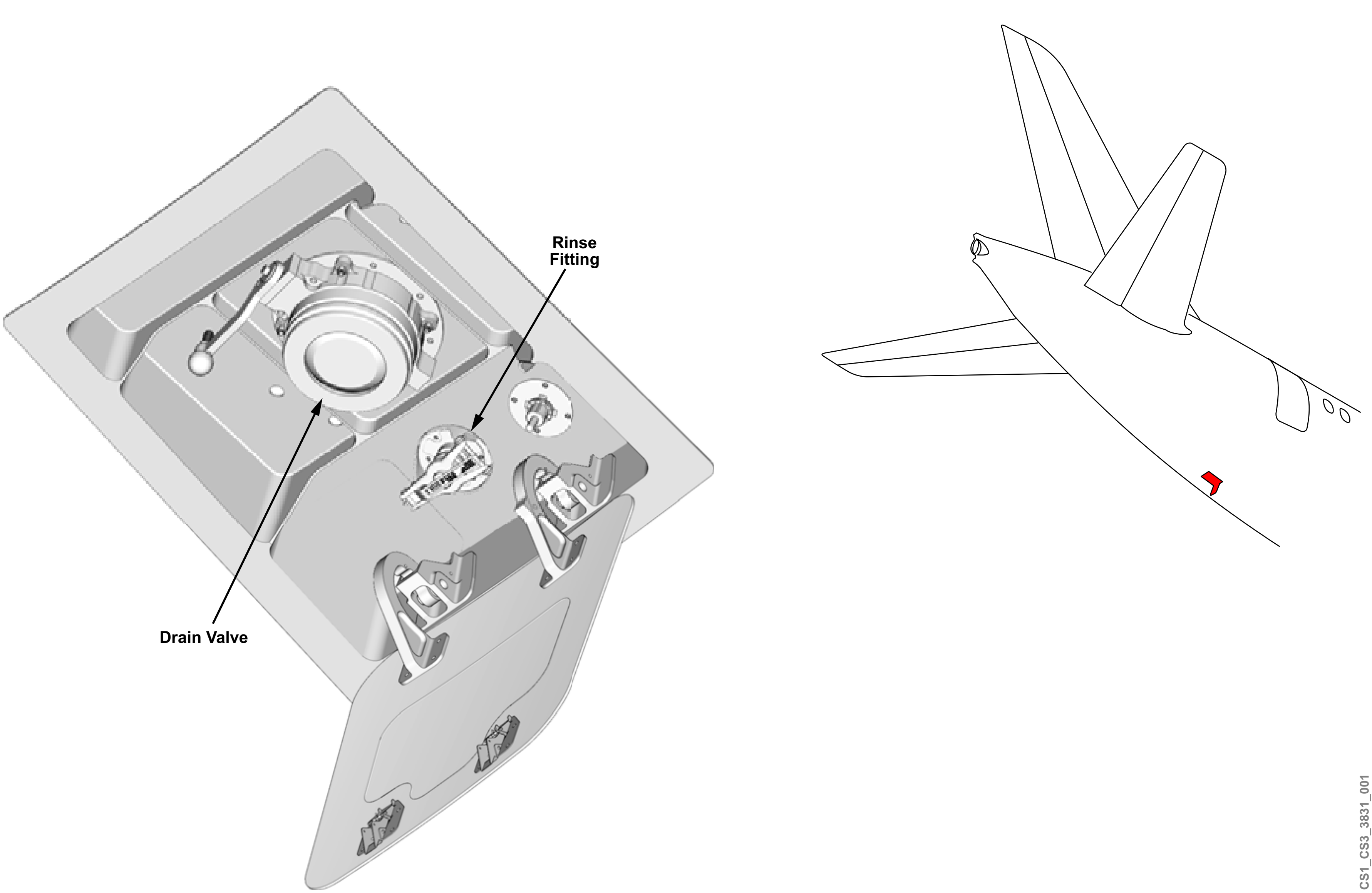


Figure 28: Waste Service Panel

COMPONENT LOCATION

The following components are installed in the vacuum waste system:

- Waste tank assembly
- Waste service panel
- Vacuum generator
- Vacuum generator motor drive unit
- Vacuum generator bypass check valve
- Toilet assembly

WASTE TANK ASSEMBLY

The waste tank is located behind the aft cargo compartment.

WASTE SERVICE PANEL

The vacuum waste system service panel is located on the lower aft section of the aircraft.

VACUUM GENERATOR

The vacuum generator is located behind the aft cargo compartment.

VACUUM GENERATOR MOTOR DRIVE UNIT

The vacuum generator motor drive (VGMD) unit is located behind the aft cargo compartment.

VACUUM GENERATOR BYPASS CHECK VALVE

The vacuum generator bypass check valve is located in the overboard discharge line.

TOILET ASSEMBLY

The toilet assembly is installed in each lavatory.

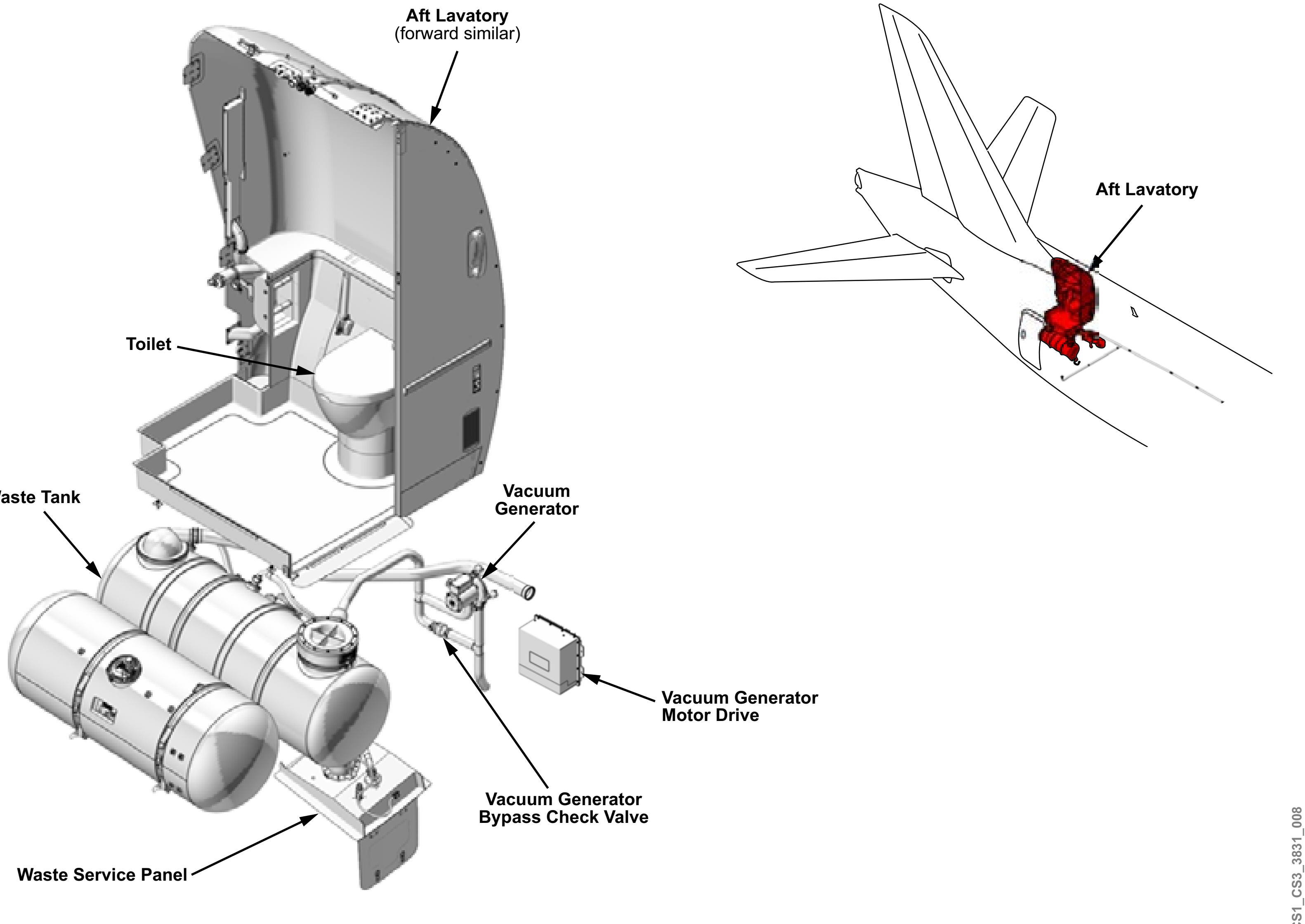


Figure 29: Vacuum Waste System Component Location

COMPONENT INFORMATION

WASTE TANK ASSEMBLY

The waste tank is corrosion resistant stainless steel with a total capacity of 194 L (51.2 gal) and a usable capacity of 144 L (38 gal). The waste tank assembly consists of an inlet assembly, a water separator, and two rinse nozzles. Two ultrasonic point level sensors monitor the waste tank quantity, The tank is mounted to the aircraft structure with two band straps. A drain tube connects the tank to the waste service panel.

Inlet Assembly

During a flush sequence, waste material and air enters the waste tank through the waste tank inlet assembly, located near the aft end of the waste tank. The waste inlet assembly reduces the kinetic energy of the incoming waste by separating the waste material from the air rushing in. An inner deflector directs the waste towards the back end of the waste tank parallel to the surface of the fluid in the tank to minimize splashing.

Water Separator

The water separator removes moisture from the air expelled overboard from the waste tank during a flush sequence. The separator inside the waste tank is held in place by the outer cap. Air passes from the tank, through the separator, and through the mesh, before exiting from the outlet port. Drain holes allow fluid to drain out of the separator and back into the tank. The mesh filter mounts inside the outer cap.

Rinse Nozzles

Two rotating style rinse nozzles are mounted through the tank wall for the purpose of rinsing the tank interior during servicing. The rinse nozzles are positioned to ensure rinse coverage of the interior of the waste tank wall, waste inlet, and level sensors. The rinse nozzles are also used to supply precharge fluid to the waste tank.

Ultrasonic Point Level Sensors

Two ultrasonic point level sensors (UPLS) provide waste tank fill status. One sensor is installed to detect 75% full level. The other sensor is installed to detect 100% full. The sensors operate on 28 VDC. Lavatory toilet operation is inhibited when the 100% sensor indicates Full.

The sensor uses an ultrasonic transducer and receiver molded into a plastic housing. The sensor provides a signal when the contents of the tank rise to fill the sensor’s V-shape gap.

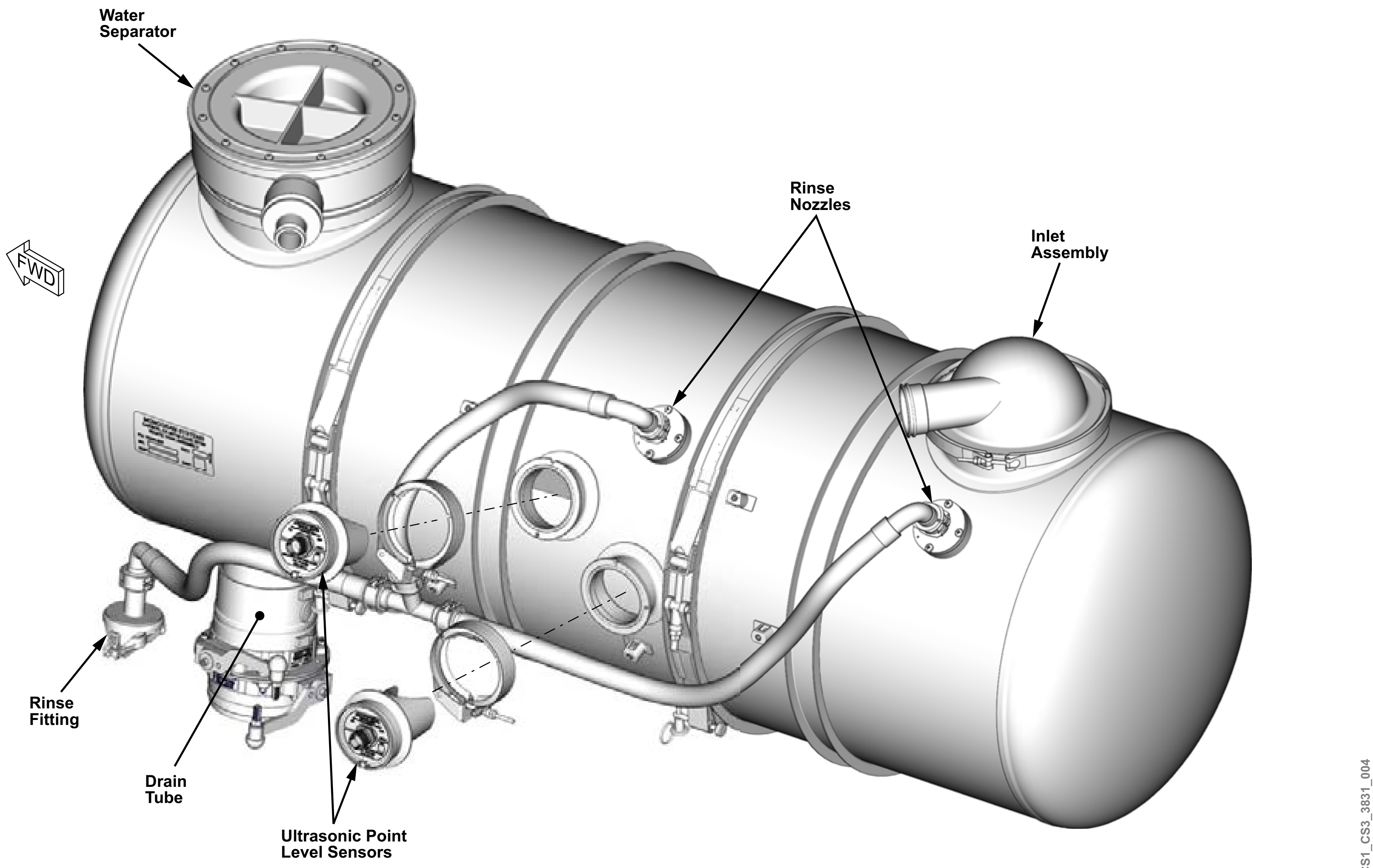


Figure 30: Waste Tank Assembly Components

TOILET ASSEMBLY

The toilet assembly is designed to collect waste material and control the transport to the storage tank. The toilet assembly consists of a toilet bowl, a flush system to empty the bowl, a rinse system to clean the bowl, and a controller to provide system controls.

The bowl is designed to be rinsed clean after each use by a single flush of approximately 237 ml (8 oz) of potable water at 38 psig. The capacity of the bowl is 9.4 L (2.5 gal). The bowl includes an overflow to limit the maximum level of fluid in the bowl.

Rinse Valve

The rinse valve is an electronically-controlled solenoid valve. Upon activation, water flows through the valve to the antisiphon valve. The rinse valve is equipped with an integral water inlet fitting and an inlet screen.

Flush Valve

The flush valve assembly connects the toilet bowl to the vacuum discharge line. The valve is an electrically actuated, reversible, rotating disk valve. The actuator is a DC type motor with built-in thermal overload protection.

Antisiphon Valve

The antisiphon valve is designed to prevent contamination of the aircraft potable water supply by breaking contact of the fluid and providing an air gap, which prevents reverse flow from the toilet bowl. Rinse water is supplied to the antisiphon valve by the rinse valve.

Flush Ring

The flush ring is mounted on the upper rim of the bowl. Spray nozzles direct an even spray of rinse water onto the sides of the toilet bowl. The flush ring inlet connects to the antisiphon valve.

Flush Control Unit

The flush control unit (FCU) is a modular electronic assembly mounted on the toilet pedestal. A discrete logic circuit monitors and controls operation of the toilet by timing and sequencing the various components during the flush cycle.

Flush Switch

The flush switch is mounted on the wall next to the toilet assembly. The flush switch initiates a timed sequence controlled by the FCU that includes the introduction of water into the vacuum toilet assembly bowl.

Maintenance Switch

The toilet assembly has a maintenance mode that is entered by pressing a switch on the FCU. The maintenance mode inhibits the operation of the vacuum generator.

When in maintenance mode:

- Pressing the maintenance switch opens the flush valve. The valve remains open until the maintenance mode is exited
- Pressing and holding the maintenance switch for more than 1 second commands the vacuum generator to run for 6 seconds

The maintenance mode can be exited by pressing the flush switch. The flush valve closes and the toilet returns to normal operation.

Toilet Manual Shutoff Valve

The toilet shutoff valve is a manually operated valve mounted under the lavatory countertop next to the toilet. It is positioned between the toilet water supply and the toilet. It is used to isolate the toilet from the water supply for maintenance purposes.

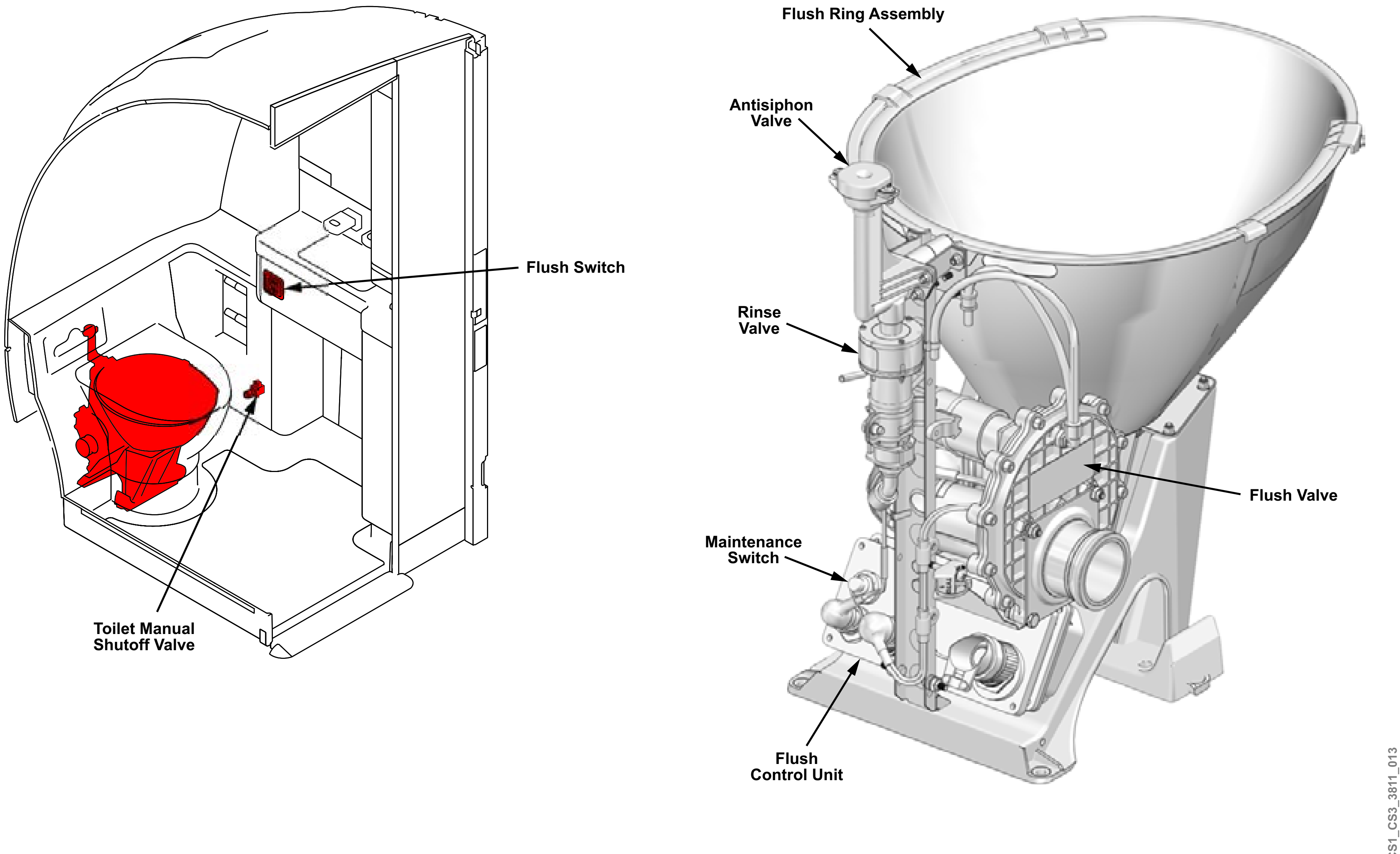


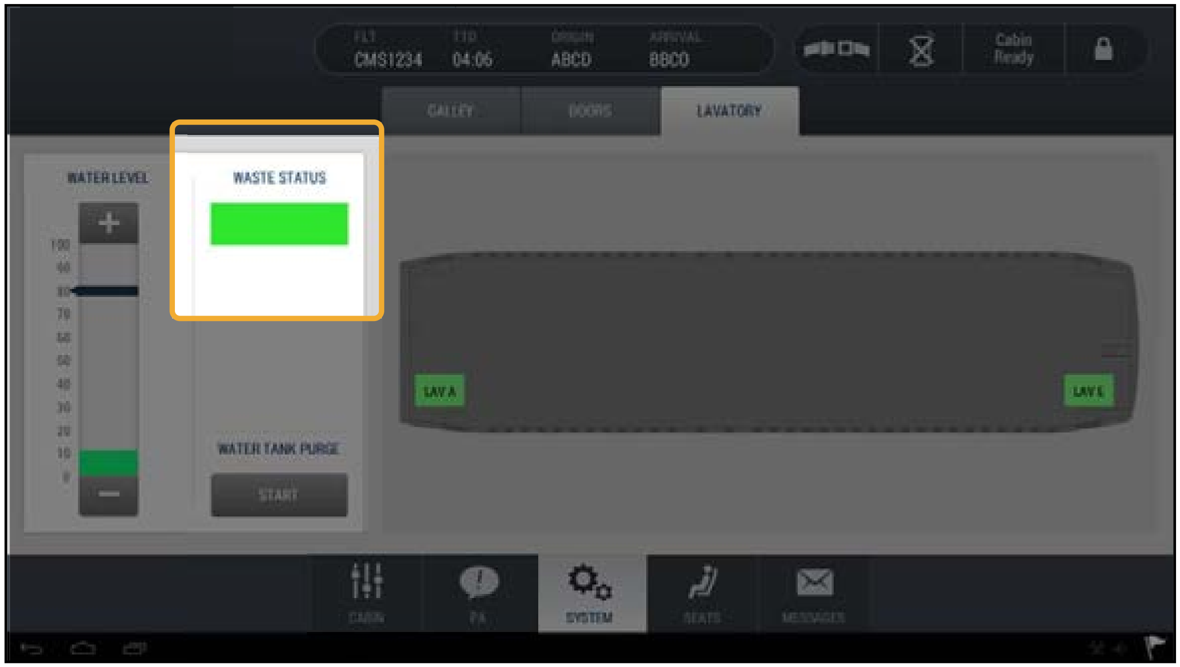
Figure 31: Toilet Assembly

CONTROLS AND INDICATIONS

The waste level is indicated on the lavatory screen on the cabin management system (CMS) crew terminal. The WASTE STATUS indicates the serviceability of the waste tank. The status is displayed using the following colors:

- GREEN – The level is < 75% of capacity
- AMBER – The level is 75% and 99% of capacity
- RED – FULL

WASTE STATUS	
Symbol	**Condition**
	Normal
	75% Full
	100% Full



CREW TERMINAL

CS1_CS3_3832_001

Figure 32: Cabin Management System Lavatory Page

DETAILED DESCRIPTION

The water and waste system controller (WWSC) controls the vacuum waste system. Waste system information is displayed on the cabin management terminal. The WWSC communicates with the cabin management system (CMS) through the DMC, using ARINC 429 BUS The WWSC communicates with the components of the waste system through a CAN BUS.

A vacuum generator provides system vacuum on demand when the aircraft is on the ground or below 16,000 ft. The vacuum generator is powered via a motor drive unit that converts 115 VAC 3-phase variable frequency to 400 Hz fixed frequency power.

On the ground or below 16,000 ft, when a lavatory flush switch is pressed, the vacuum generator is turned on. After a 3 second delay to allow vacuum to be generated, the toilet flush valve is commanded open and then closed. Above 16,000 ft, cabin to outside differential pressure is utilized to transport waste. The integrated air system controllers (IASCs) provide altitude information to the WWSC.

A waste service panel door switch provides a discrete signal to the VGMD to inhibit the vacuum generator whenever the service panel door is open. The VGMD sends this information through the CAN BUS to the WWSC. The controller sends this information to the CMS display to show a WASTE SERVICE DOOR OPEN message.

The quantity of waste inside the tank is measured using two point level sensors are installed in the waste tank. One is located at the 75% level and the other is located at the 100% level. When the 100% sensor detects FULL, the waste system is shut down and a corresponding message is sent to the CMS.

The ultrasonic point level sensor (UPLS) communicates with the water and waste system controller (WWSC) over a CAN BUS. The sensors each have their own source of power from DC BUS 1. If both UPLS sensors are detected to be faulty, full tank condition is assumed and the toilet operation is disabled. If only the 100% UPLS sensor is faulty, the WWSC allows 40 additional flushes once the 75% UPLS becomes full.

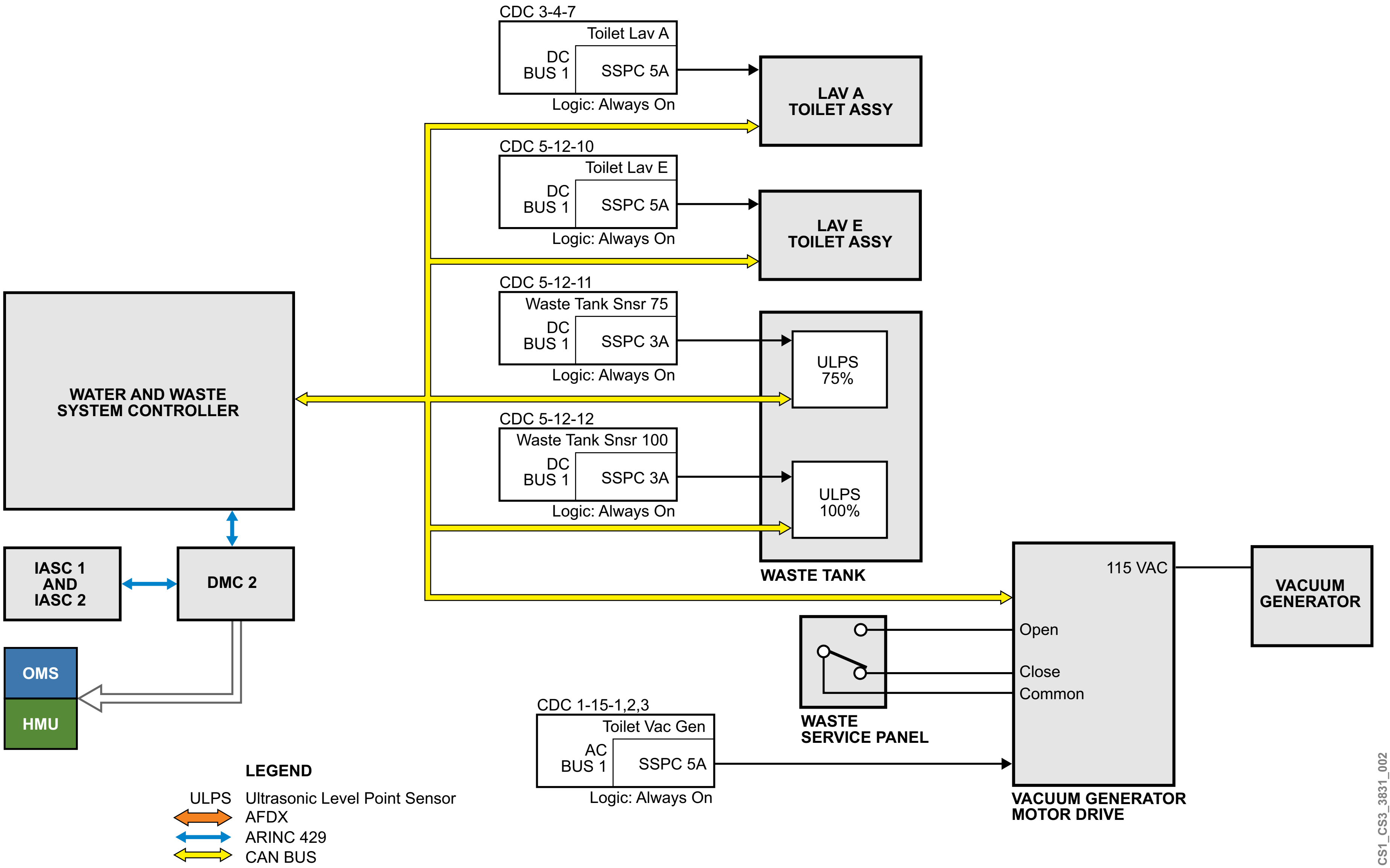


Figure 33: Waste System Schematic

MONITORING AND TESTS

The following page provides the cabin management system (CMS) waste quantity and waste system displayed messages.

CABIN MANAGEMENT SYSTEM MESSAGES

Table 3: Cabin Management System Messages

MESSAGE	LOGIC
WASTE TANK 75%	The message is posted when the waste level reaches 75%.
WASTE TANK FULL	The message is posted when waste level indicates 100%.
WASTE SERVICE DOOR OPEN	The waste service door is open. The CMS displays the message and generates a low chime.
WASTE SYSTEM INOPERATIVE	If the waste tank is full or if the waste level sensor 100% is failed and the waste level is greater than 75%, or both waste level sensors inop, or all vacuum toilet assemblies inop, the system is considered inoperative and the WWSC reports to DMC. The CMS receives a WASTE SYSTEM INOP message and generates a low chime for crew awareness action.
LAVATORY A OR E INOP	The message is posted whenever the related flush valve is failed (overcurrent or high current) or the related flush switch jammed, or loss of communication with the related FCU.

PRACTICAL ASPECTS

WASTE TANK SERVICING

When the service panel door is opened, the door switch removes power from the vacuum generator. This prevents creating a vacuum in the tank during waste tank servicing.

To service the waste tank, the cap on the rinse line port is opened before attaching the drain hose to ensure that the tank is properly vented. After the drain hose is attached to the drain valve, the valve is opened by rotating the lever 90°, and the tank drains by gravity into the waste service truck. Rinse water is then connected to the aircraft service panel rinse nipple. The rinse water is turned on while the drain valve is open. The rinse water is used to rinse the interior of the tank and the level sensors inside the tank.

Rinsing should continue until the tank is drained completely. It is recommended that the tank be rinsed for 2 to 5 minutes at 40 psi, each time the tank is drained. After rinsing has been completed, the service panel ball valve should be closed and 7.5 L (2 gal) precharge fluid should be added. The drain and rinse hoses can be removed and the service panel door closed.

If the aircraft needs to be prepared for a cold soak, first assure that the waste tank is completely drained by following the waste tank servicing procedures. Initiate a dry flush from the forward toilet. When the water system is shut down, no water should flow from the rinse nozzles. After 15 seconds, initiate a second dry flush from the forward toilet.

NOTE

When the rinse hose is removed, remaining rinse water in the tank rinse line must be allowed to drain from the rinse nipple before it is capped to avoid freezing water in the line.

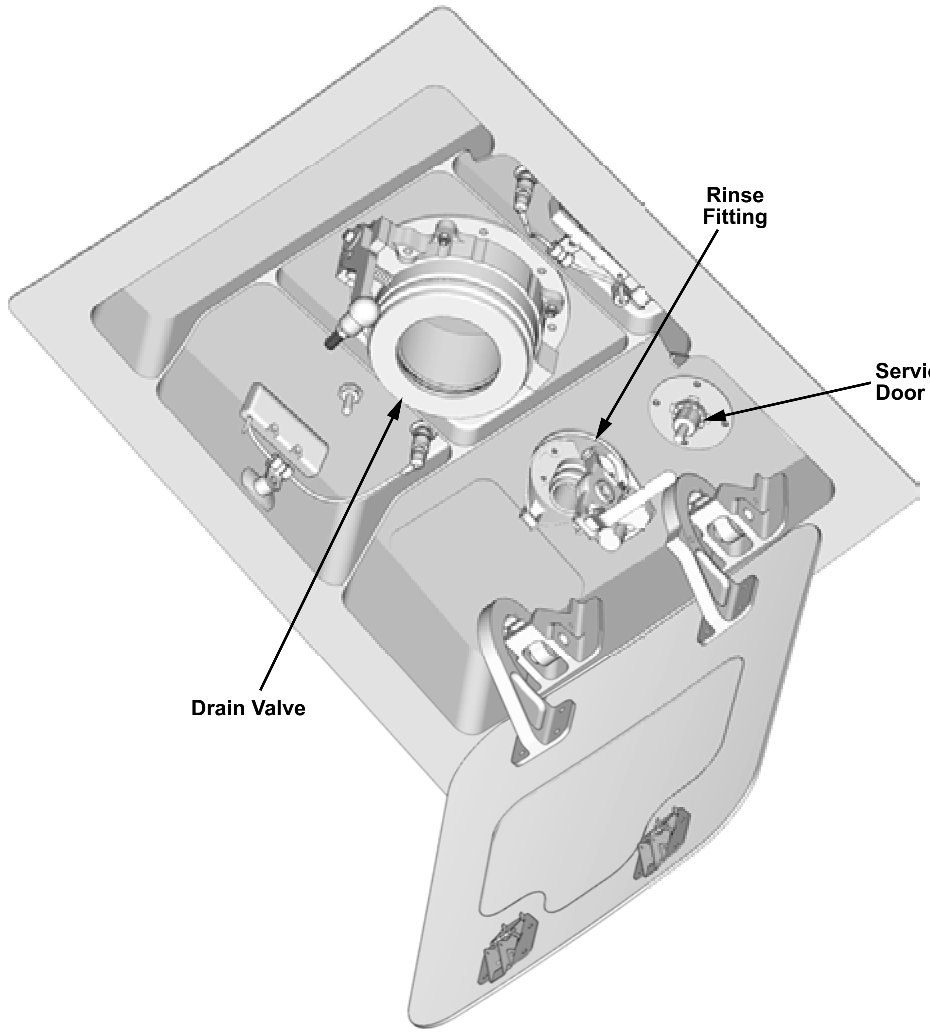


Figure 34: Waste Tank Servicing

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ATA 50 - Cargo Compartments

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

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CARGO COMPARTMENTS - CHAPTER BREAKDOWN

Forward Cargo Compartment 1

Aft Cargo Compartment 2

Cargo Compartment Nets 3

Equipment Bays
(Forward, Mid and Aft) 4

50-10 FORWARD CARGO COMPARTMENT

GENERAL DESCRIPTION

The forward cargo compartment is located below the floor structure of the forward fuselage and the forward/mid fuselage section. It is located between fuselage frames FR 20 and FR 36, and positioned between the forward equipment bay and the environmental control system (ECS) bay. The volume of the forward cargo compartment is as follows:

Table 1: Forward Cargo Compartment Volume

	CS100	CS300
Forward Cargo Compartment	10.3 m^3 (365 ft^3)	14.8 m^3 (524 ft^3)

CS1_CS3_TB50_001

The cargo compartment consists of the following primary components:

- Ceiling panels
- Sidewall panels
- Forward and aft bulkhead panels
- Flat reinforced floor panels
- Curved reinforced floor panels
- Decompression panels
- Pressure equalization valves
- Air inlets

Fiberglass ceiling, sidewall, forward and aft bulkhead panels, door liners, surround liners, and floor panels are secured to the aircraft structure using standard threaded aircraft fasteners.

Gaps between panels are sealed with special fiberglass tape to provide a fire and smoke barrier. Special rubber P-seals are installed at floor panel joints to provide an improved liquid leak barrier.

The forward cargo compartment is equipped with the following components:

- Fire detection and extinguishing (FIDEX) nozzles
- Cargo ceiling lights
- Smoke detectors

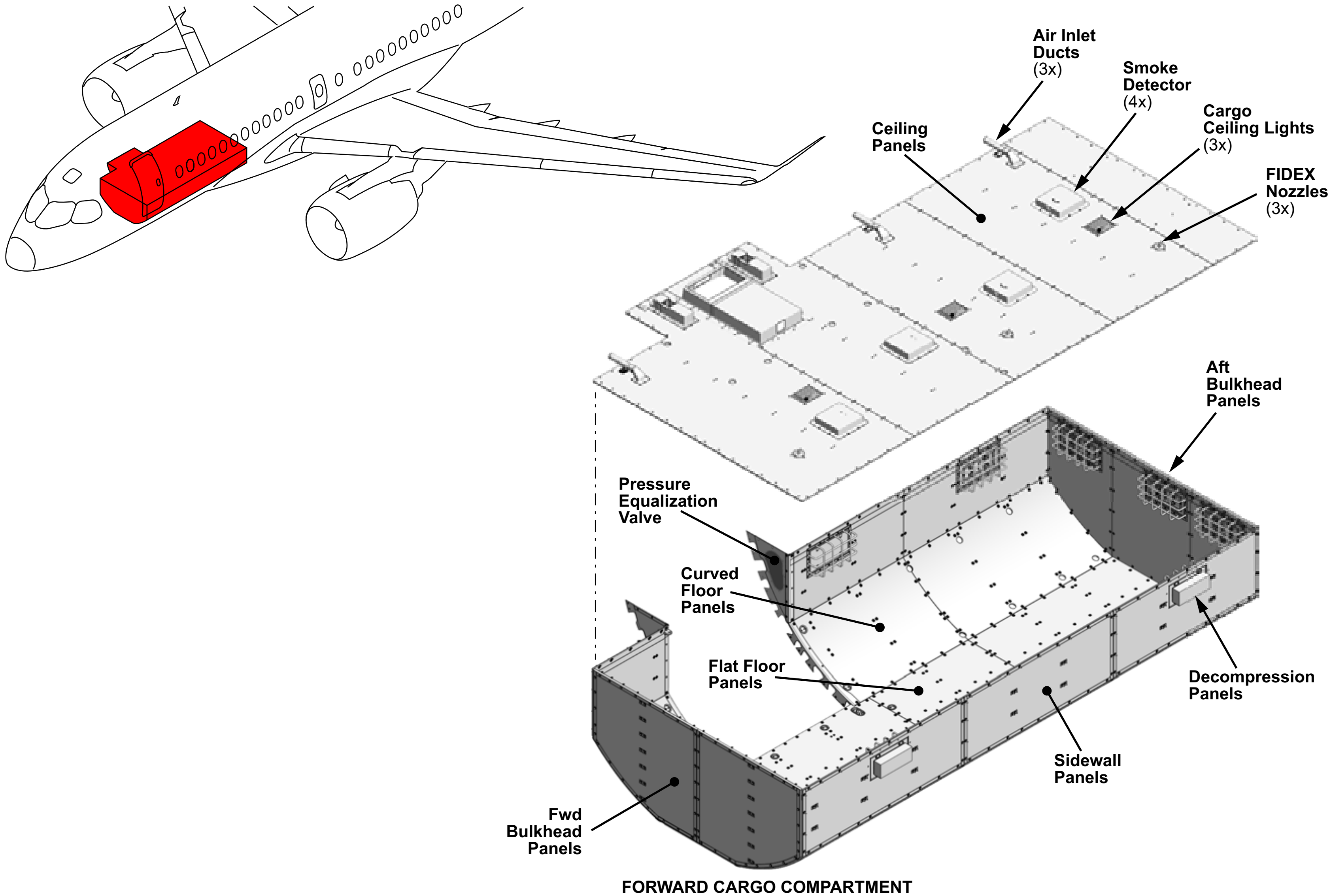

Figure 1: Forward Cargo Compartment

50-10 AFT CARGO COMPARTMENT

GENERAL DESCRIPTION

The aft cargo compartment is located below the floor structure of the aft mid and the rear fuselage sections. It is located between fuselage frames FR 54 and FR 72, and positioned between the mid equipment bay and the water system bay. The volume of the aft cargo compartment is as follows

Table 2: Aft Cargo Compartment Volume

	CS100	CS300
Aft Cargo Compartment	13.4 m^3 (474 ft^3)	16.8 m^3 (594 ft^3)

CS1_CS3_TB50_002

The aft compartment shares the same ceiling, sidewall, bulkhead and floor covering materials as the forward compartment. Similar fire detection and extinguishing (FIDEX) nozzles, smoke detectors, and lighting are used in the aft compartment.

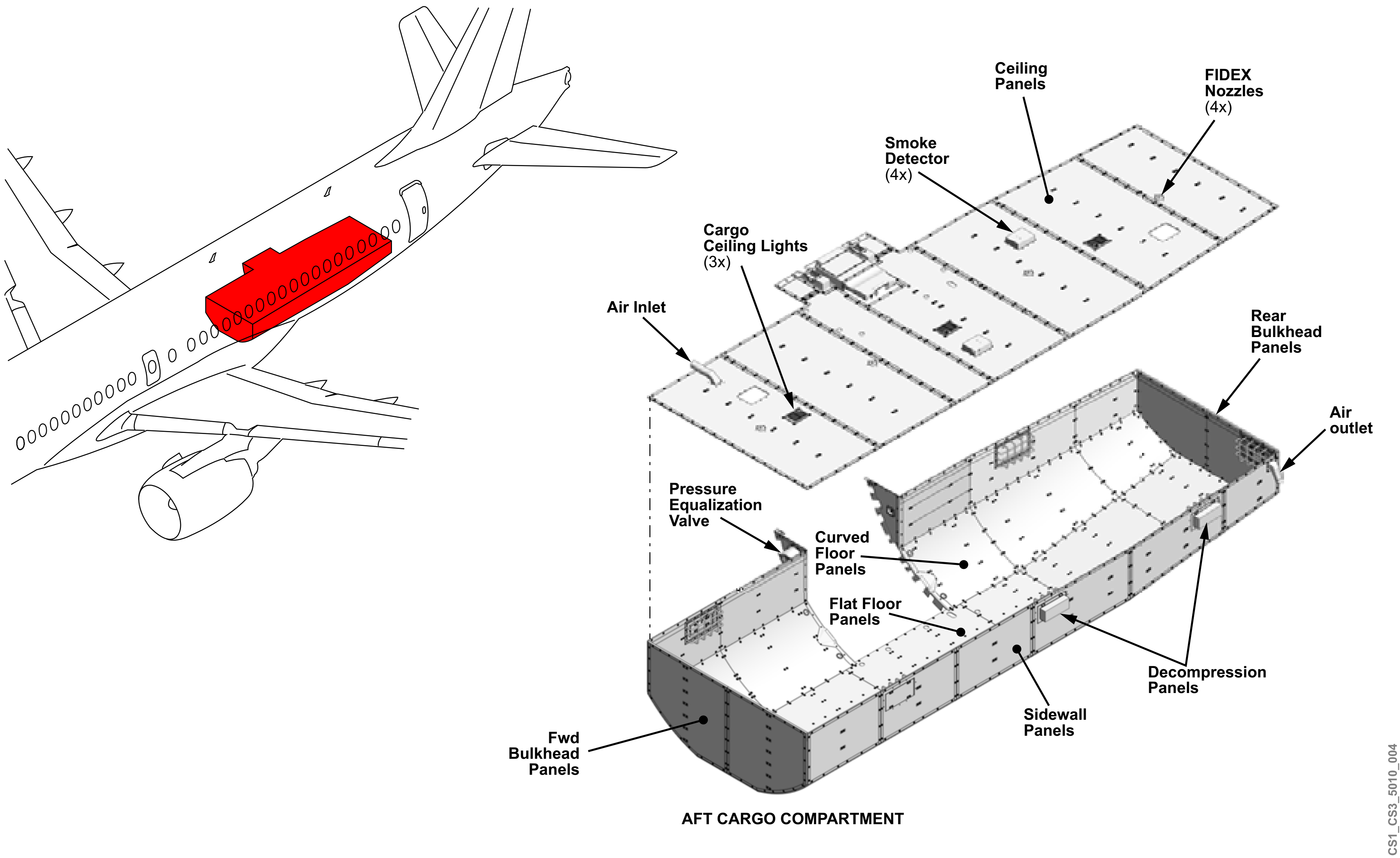

Figure 2: Aft Cargo Compartment

DETAILED COMPONENT INFORMATION

CARGO COMPARTMENT PANELS

The forward and aft cargo compartment linings include the sidewall panels, forward and rear bulkhead panels, ceiling, and floor panels.

The cargo compartment linings provide fire containment inside the forward and aft cargo compartments. The linings are installed to cover the fuselage structure and the components of the electrical, hydraulic, and air conditioning system installed in this part of the aircraft.

The ceiling, bulkhead, and sidewall panels are made of fiberglass sandwiched panels with a Nomex core. All joints between the ceiling panels and sidewall panels are tape sealed with a fiberglass flame penetration proof seal.

The reinforced flat and curved floor panels consist of a honeycomb core sandwiched between fiberglass plies. The lower surface consists of a single ply, and the top surface has an overlay of six plies that wrap around the edges of the panel.

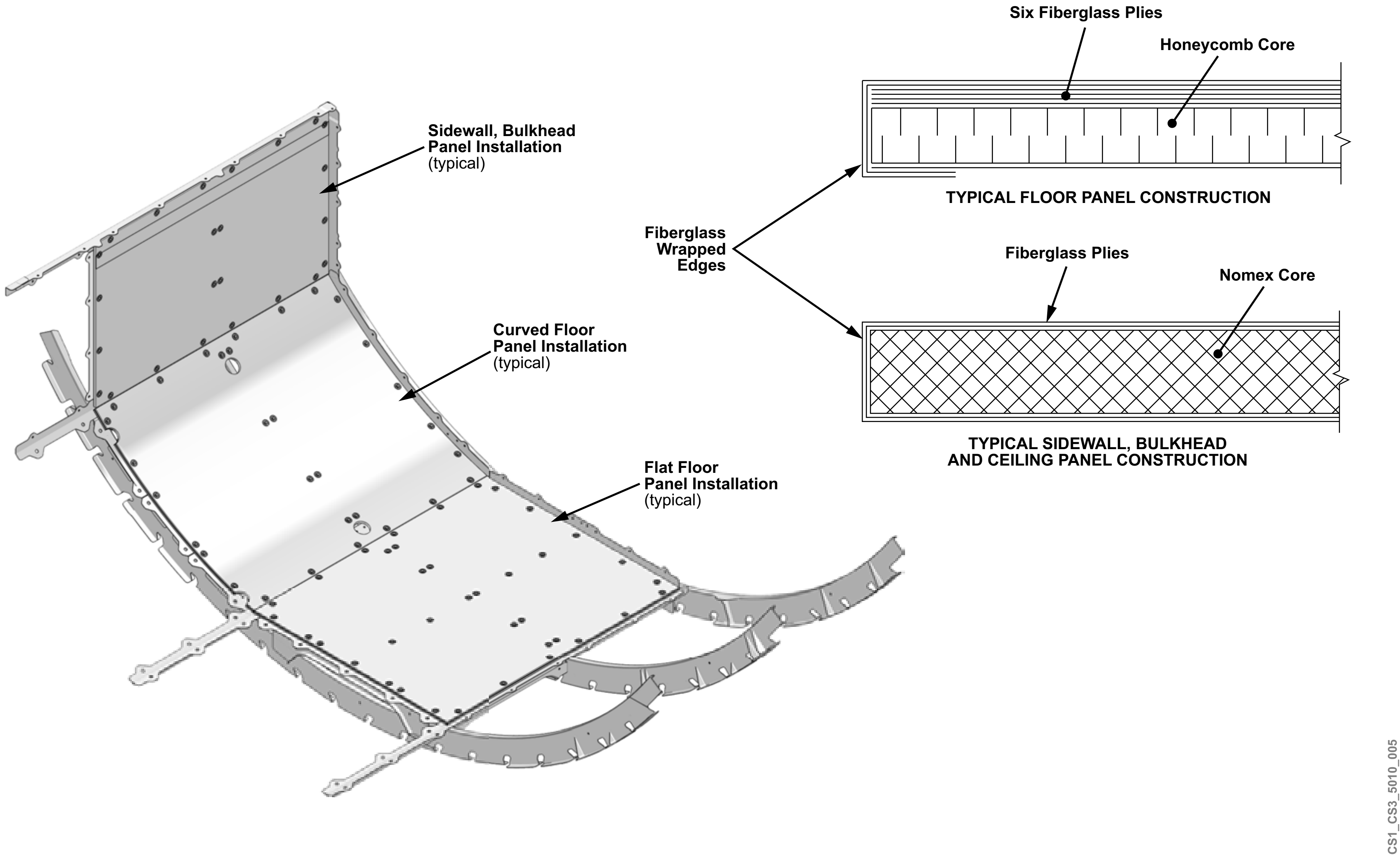

Figure 3: Cargo Compartment Panels

CARGO COMPARTMENT DECOMPRESSION PANELS

In case of a decompression event, the decompression panels release pressure from the cargo compartment. Both one-way and two-way decompression panels are used. There is a single one-way panel and six two-way panels in the forward compartment, and two one-way and four two-way panels in the aft cargo compartment.

The one-way decompression panel area is 322.58 cm^2 (50 in^2). It has a 2-3 ply front liner with a 1.02 cm (0.40 in.) thick core. Six tabs are located along the outer edge of the front liner. The panels are installed in an aluminum frame, and secured with a thin aluminum front plate. The panel/frame assembly is affixed to a cargo panel. A protective screen is installed to protect the panel from being blocked with cargo.

The two-way panels are installed with spring-hinges. Should a decompression event occur, the spring-hinges allow the panel to open and release cargo compartment pressure, and close afterwards. It is designed to open at 0.10 bar (1.5 psi) to 0.17 bar (2.5 psi). Silicone seals are placed between the panels and frames, and provide flammability protection and air leakage control.

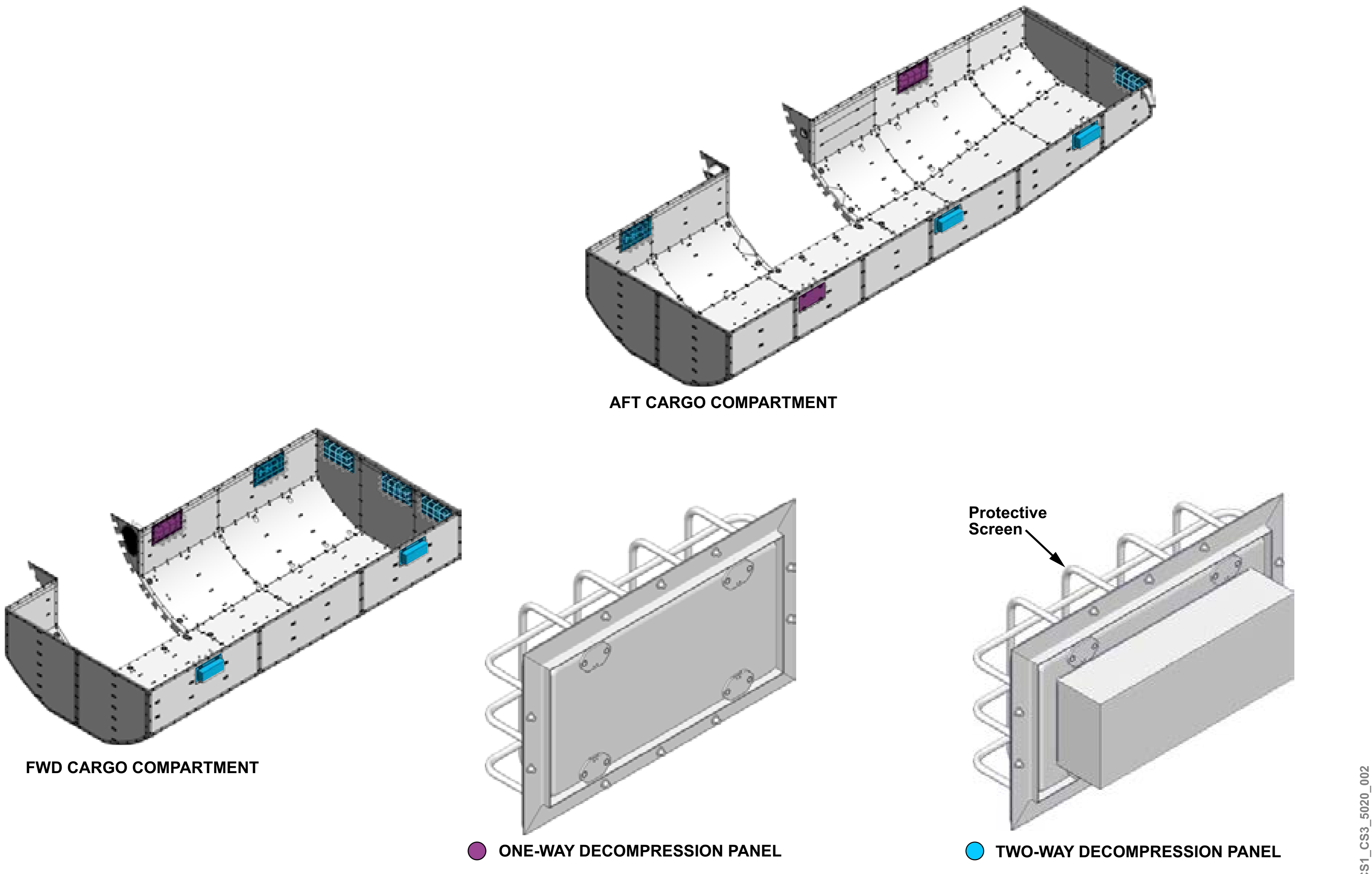

Figure 4: Cargo Compartment Decompression Panels

50-22 CARGO COMPARTMENT NETS

DETAILED COMPONENT INFORMATION

CARGO RESTRAINT NETS AND DOOR NETS

The cargo door nets are mounted on the forward and aft cargo compartment door surroundings to keep the baggage from hitting the door and jamming. The cargo door net is installed vertically just in front of the cargo compartment doors.

Cargo restraint nets and cargo door nets are manufactured from polyester. Each net consists of adjustable and removable webbing. Quick disconnect attachment fittings are attached to the ends of the cargo nets and cargo door nets for fast and easy installation and removal.

Removable and replaceable quick disconnect fittings are attached to the ends of the cargo nets and cargo door nets. These fittings are attached to cup fittings located on the cargo floor, ceiling, and sidewalls.

CARGO NET ATTACHMENT FITTINGS

Aluminum alloy cup fittings are installed to the structure of the cargo compartments by screws and anchor nuts. The cup fittings are mainly situated at the cargo panels split joint locations. Anchor plates are installed in the cup fittings. The nets are connected to the anchor fittings with push and slide type cargo net retainer fittings.

TIE DOWN ANCHORS

Tie down anchors are located on the floor throughout the cargo compartment to help secure heavy cargo. There are 16 tie down anchors in the forward cargo compartment, and 24 tie down anchors in the aft cargo compartment on the CS 100. The CS300 has 22 tie down anchors in the forward cargo compartment and 28 tie down anchors in the aft cargo compartment.The tie down anchors are similar to the cargo net attachment fittings and consist of a net cup and anchor plate.

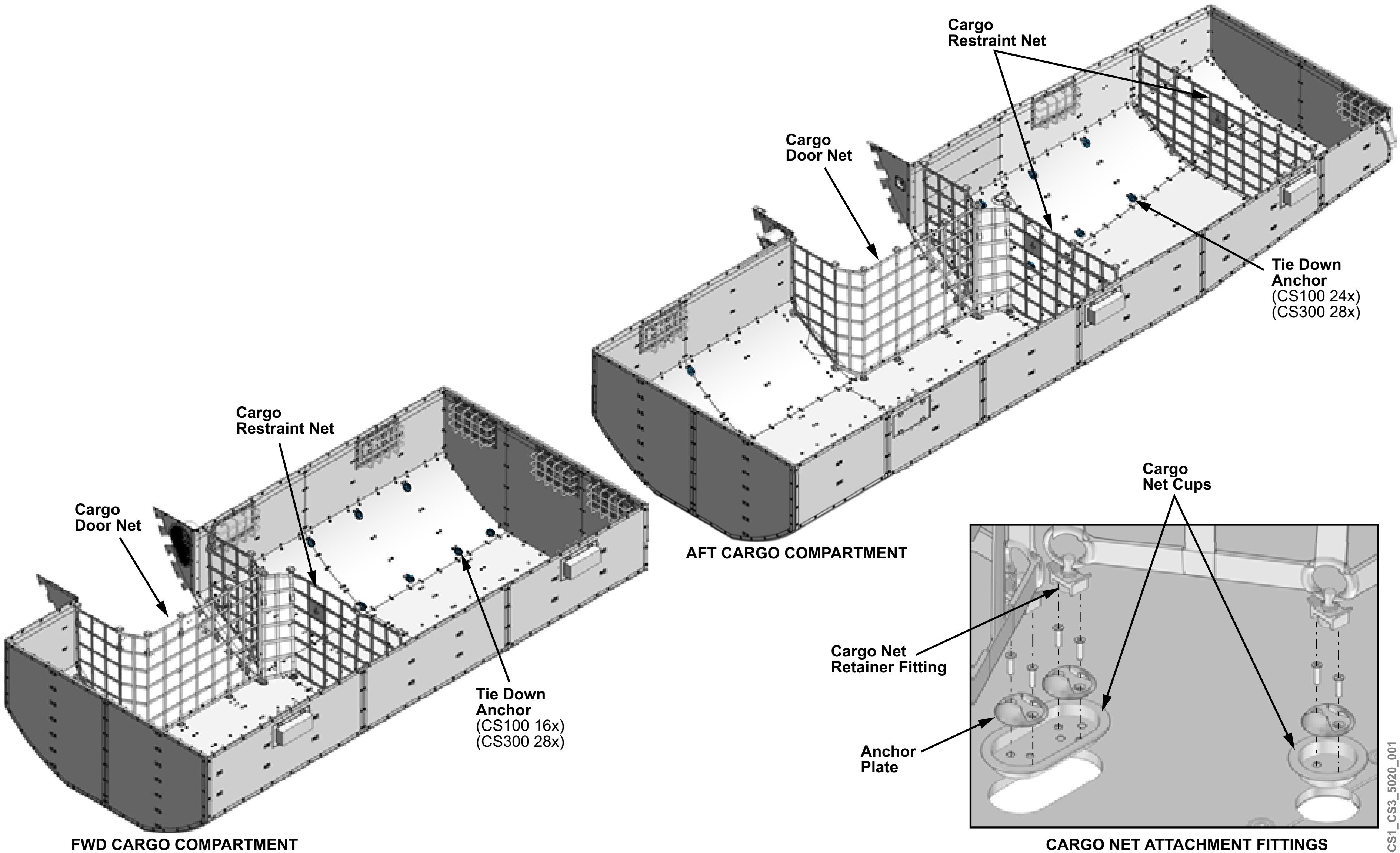

Figure 5: Cargo Compartment Nets and Attachment Fittings

50-10 EQUIPMENT BAYS

GENERAL DESCRIPTION

FORWARD EQUIPMENT BAY

The forward equipment bay is located below the floor structure in the forward fuselage forward section. Located between the forward semi-pressure bulkhead at FR 15, and the forward cargo forward bulkhead at FR 22. Aluminum alloy racks to house the avionics and electrical equipment are installed in the forward equipment bay.

MID EQUIPMENT BAY

The mid equipment bay is located below the floor structure, aft of the center wing box, located between the aft mid fuselage semi-pressure bulkhead at FR 51, and the rear cargo forward bulkhead at FR 54. Aluminum alloy racks are used to house the avionics and electrical equipment installed in the mid equipment bay.

AFT EQUIPMENT BAY

The aft equipment bay is located in the aft fuselage section and lies within the unpressurized zone of the aircraft. An equipment rack mounted on the right side of the bay contains the components for hydraulic system no. 3. Another rack, on the left side contains avionics components.

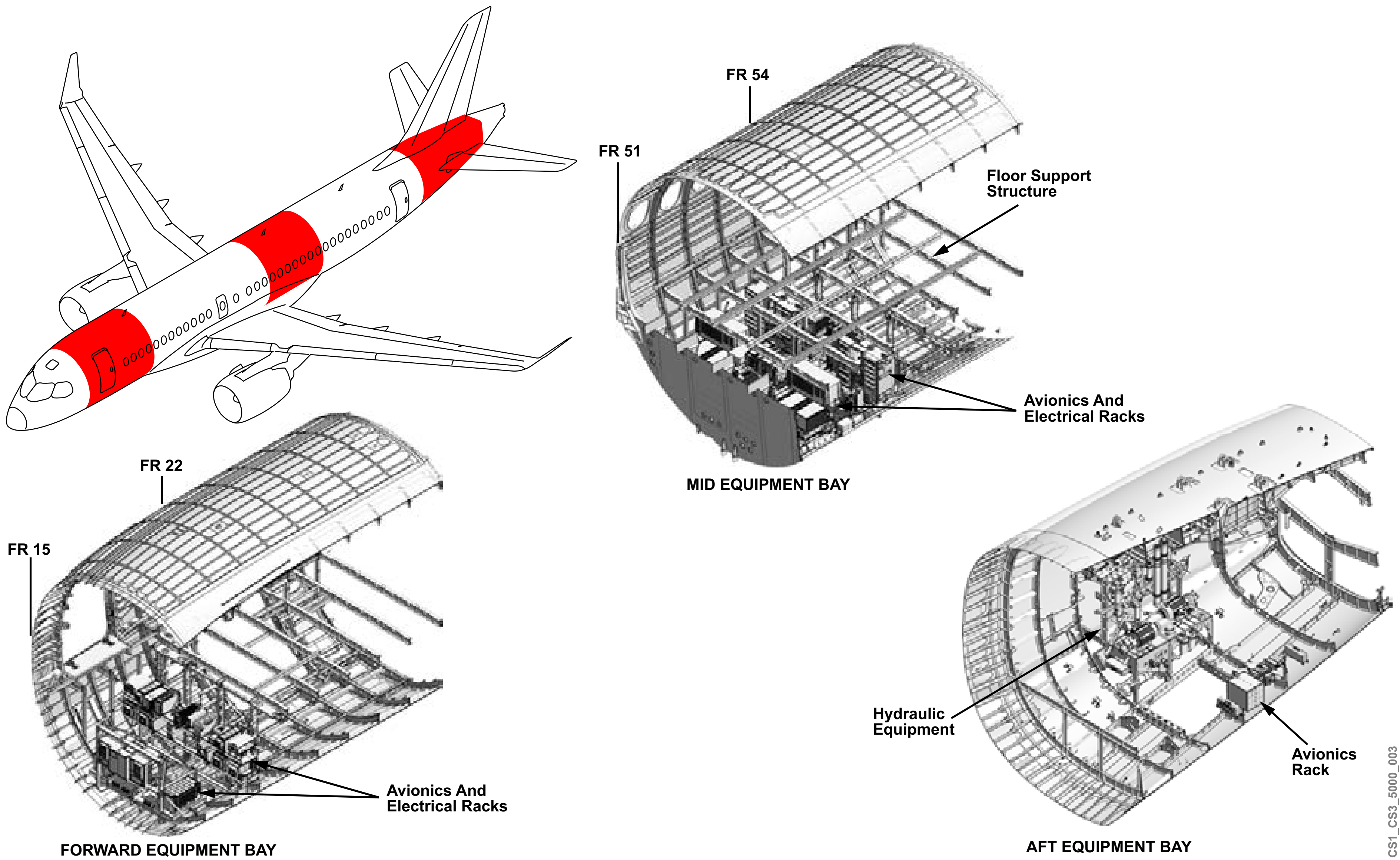

Figure 6: Equipment Bays General Description

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ATA 51 - Standard Practices - Structures

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

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STANDARD PRACTICES - STRUCTURES
CHAPTER BREAKDOWN

51-10 STRUCTURES

GENERAL DESCRIPTION

The aircraft structure is divided into primary and secondary classifications to show the relative structural importance. This classification of the structure helps to make decisions about how and when to repair damage.

PRIMARY STRUCTURE

The primary structure is the structure that transmits loads that are caused by:

- Pressurization
- Wind gusts
- Flight maneuvers
- Takeoff
- Landing
- Ground maneuvers

Structural Significant Item

The primary structure is made up of structural significant items (SSIs). The SSIs are structural parts that are most important in the transmission of flight, ground, and pressurization loads. Failure in the structural significant items of the primary structure can have an unwanted effect on the safety of the aircraft.

When repairs are made to structural significant item (SSI), make sure the repair is damage tolerant.

Principal Structural Element

Principal structural elements (PSEs) are those individual elements of the SSIs whose failure could result in catastrophic failure of the aircraft.

Airworthiness Limitation Item

It is possible that all or part of an SSI is classified as an airworthiness limitation item (ALI). The ALIs are those parts of the structure where mandatory maintenance and inspection are necessary to prevent the loss of airworthiness due to fatigue cracking. Some ALIs are replaced before fatigue cracks can occur. Other ALIs are inspected for cracks with a time-controlled inspection program.

Fatigue Critical Baseline Structure

Fatigue critical baseline structure (FCBS) describes elements of the primary structure, which if repaired or altered, could become susceptible to fatigue cracking, and require a damage tolerance evaluation (DTE) to determine if additional damage tolerance inspections (DTIs) are required. Any additional inspection resulting from a repair to a fatigue critical baseline structure (FCBS) is known as a specific airworthiness limitation (SAL).

SECONDARY STRUCTURE

The secondary structure is the structure that is made to:

- Transmit local aerodynamic loads and inertia to the primary structure
- Keep the aerodynamic shape
- Attach equipment

Loads applied to the secondary structure are transmitted to the primary structure. Damage to the secondary structure can cause unwanted effects to the primary structure, engine performance, aerodynamic qualities, or control surface stiffness.

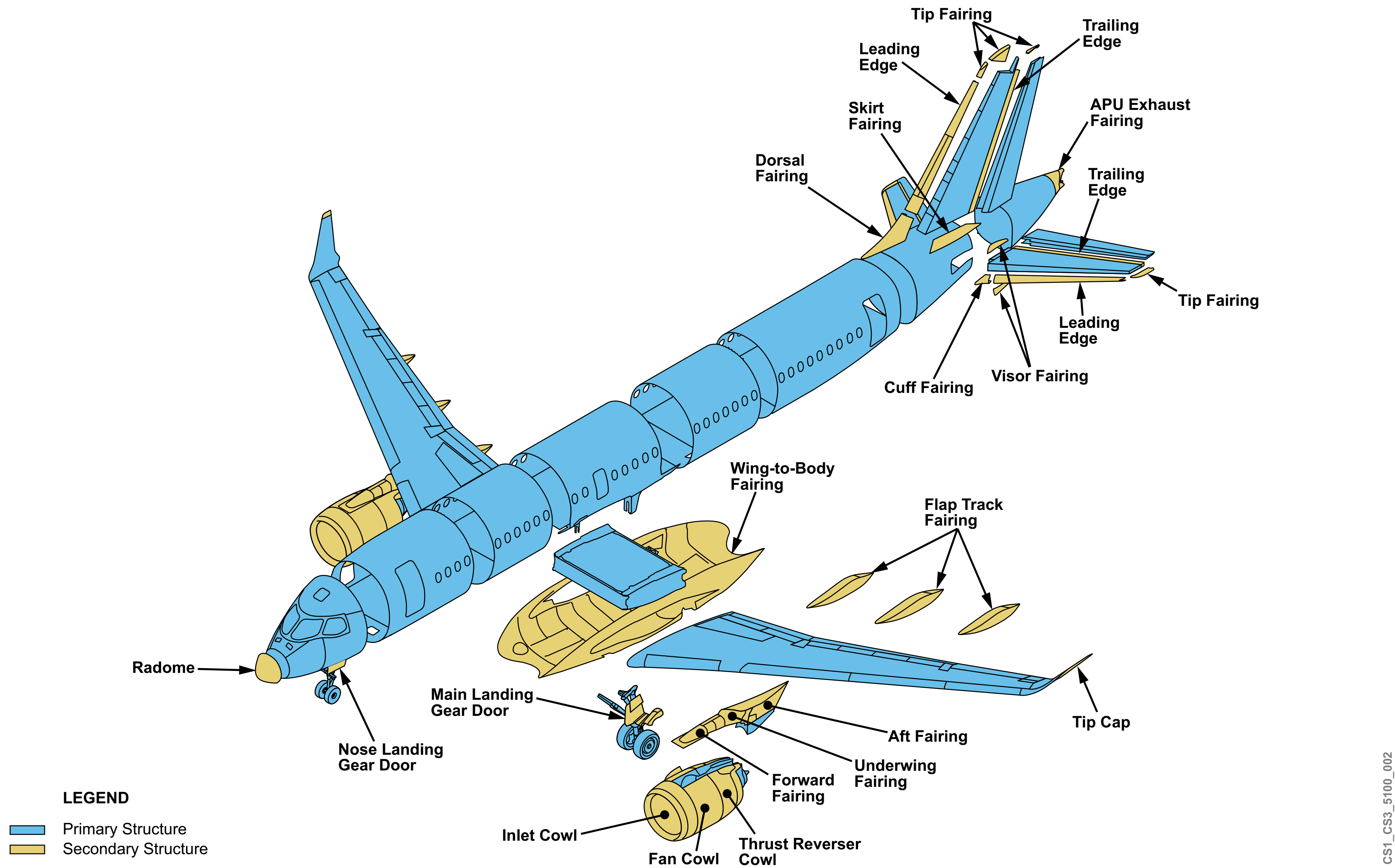

Figure 1: Aircraft General Structures

DAMAGE ASSESSMENT

Composite structure typically wears out from disbonding, delamination, environmental degradation, and inter-laminar cracking, and for composite honeycomb sandwich failure mechanisms, including facesheet disbonding of the core.

Composite structures behave differently from metallic structures when subject to impact damage as there may not be any external evidence, such as dents or buckles, to indicated the need for maintenance action.

In terms of damage assessment, there are five categories of damage.

CATEGORY 1

Category 1 includes manufacturing defects as well as barely visible impact damage (BVID), which is the likely impact damage at the threshold of reliable visible detection. Such damage must be durable up to the operational life limit of the aircraft.

CATEGORY 2

Category 2 is visible impact damage (VID), which is the damage that can be reliably detected by scheduled or directed field inspections performed at specified intervals. Visible impact damage (VID) up to the aircraft structure repair publication (ASRP) permitted damage limits (PDL) is durable up to the operational life limit of the aircraft. Damage beyond permitted damage limit must be repaired when detected.

CATEGORY 3

Category 3 is damage that can be reliably detected within a few flights of occurrence by operations or ramp maintenance personnel without special skills in composite inspection.

CATEGORY 4

Category 4 is discrete source damage from a known incident that flight maneuvers would be limited until the aircraft lands. Some examples of Category 4 damage include rotor burst, bird-strikes, tire bursts, and severe in-flight hail.

CATEGORY 5

Category 5 is severe damage created by ground or flight events, which are not covered by design criteria or structural substantiation procedures. This type of damage requires immediate repair. Some examples of Category 5 damage include severe service vehicle collisions with aircraft, anomalous flight overload conditions, abnormally hard landings, maintenance jacking errors, and loss of aircraft parts in flight, including possible subsequent high-energy, wide-area impact with adjacent structure.

Some Category 5 damage scenarios have no clear visual indications of damage, particularly in composite structures. A complete report of actual and possible damage is essential after a Category 5 event.

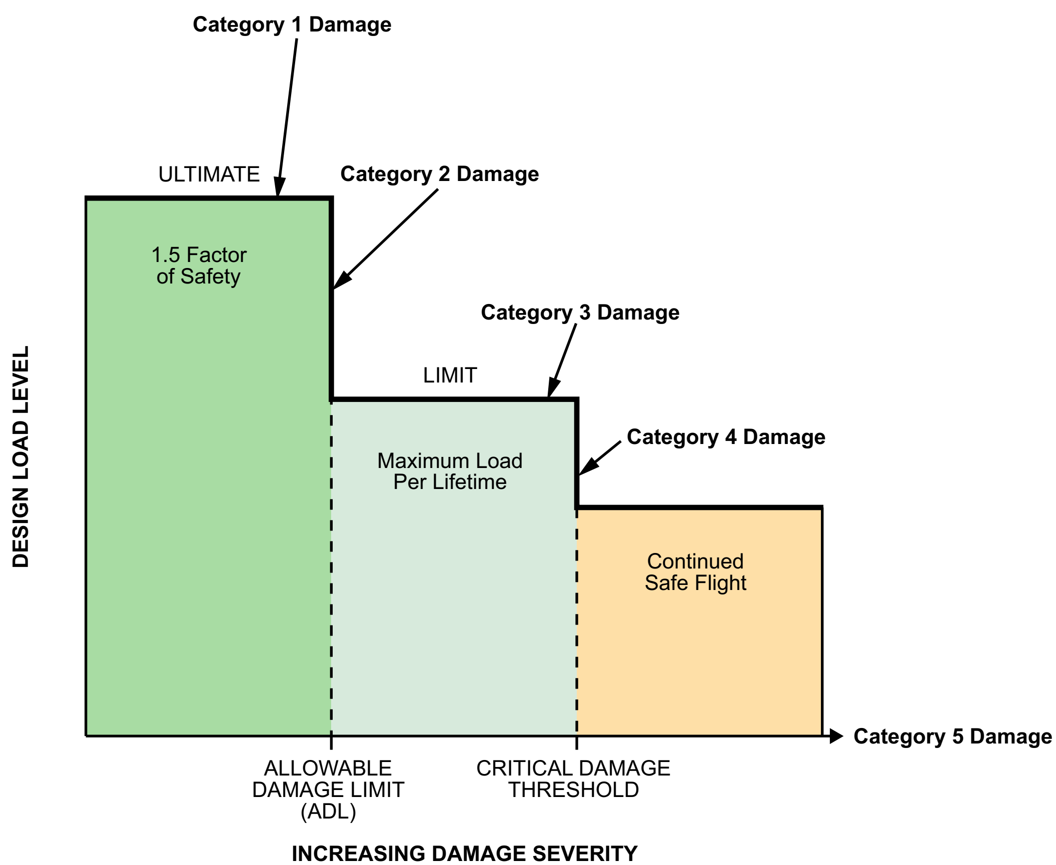

Figure 2: Damage Assessment Categories

REPAIRS

TYPES OF REPAIRS

Repair instructions are given in the aircraft structure repair publication (ASRP) for repairs that keep the structure airworthy.

PERMITTED DAMAGE REPAIRS

Permitted damage repairs are covered in ATA 51-12. These repairs permit small amounts of damage to be repaired by simple procedures, such as blending a scratch for example. It usually requires removal of material and restoration of the protective finish. The quantity of material that can be removed must not be more than what is specified in the permitted damage limits.

TYPICAL REPAIRS

Typical repairs are covered in ATA 51. Use typical repairs to the primary structures only when they are specified. Typical repairs are suitable for use on typical structural components covered in more than one chapter. They are meant for repairing minor damage to secondary structure which is beyond allowable damage limits.

TEMPORARY REPAIRS

Temporary repairs are sometimes used to rapidly restore the structural airworthiness. These time-limited repairs must be replaced after a specified time with a permanent repair. This time is usually specified as flight hours, flight cycles, or letter checks.

GENERIC REPAIRS

Generic repairs are covered in selected sections of ATA 51-71 and apply to a special type of structure, such as fuselage frames. They are intended to repair damage which is beyond allowable damage limits. These repairs can be used as a guide when repairs of the same type are necessary in another location of the same type of structure.

SPECIFIC REPAIRS

Specific repairs are found in ATA 52-ATA 57 and apply to damage in a specific area of the aircraft. They are intended to repair damage which is beyond allowable damage limits for the applicable area. As an alternative, an applicable repair engineering order (REO), or service bulletin (SB) can be used, or the damaged parts can be replaced.

APPROVED REPAIRS

Structural repairs that are done using approved data are called approved repairs. Approved repairs can be done by using the ASRP, approved REO, SB, or by replacing the damaged components.

NOT-APPROVED REPAIRS

Structural repairs that are done without using approved data or by changing approved data are called not-approved repairs. Additional airworthiness authority approval may be required for not-approved structural repairs to principal structural elements (PSEs), or fatigue critical baseline structures (FCBSs).

CLASSIFICATION

The structural repair approval logic diagram is used to find if a repair is an approved structural repair or a not-approved structural repair, to find the correct approval procedure, and to make a decision if changes to the damage tolerance inspection requirements are necessary.

DAMAGE TOLERANCE APPROVAL

The damage tolerance analysis of the repair must be sent to the applicable airworthiness authority for approval. If the damage tolerance analysis indicates that additional inspections are required, the repaired structure becomes a specific airworthiness limitation which must be recorded in the maintenance requirements publication (MRP) supplementary structural deviation inspection requirements (SDIR) document for that specific aircraft serial number.

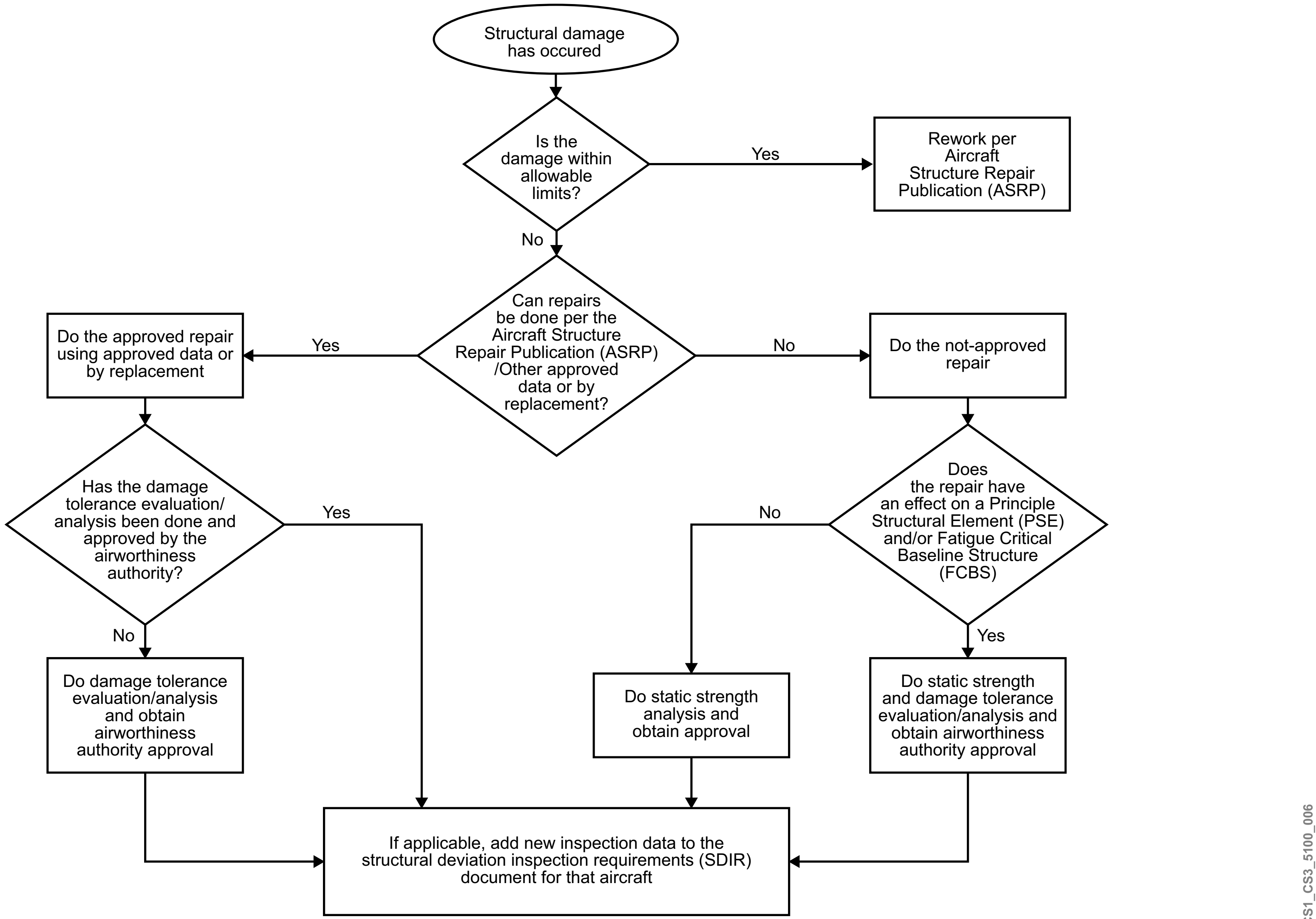

Figure 3: Structural Repair Approval Logic Diagram

51-20 PROCESSES

GENERAL DESCRIPTION

The C Series aircraft incorporates additional processes for protective finishes and corrosive prevention materials.

PROTECTIVE TREATMENT

Some parts of the aircraft structure have additional finishes applied to them during manufacturing or after assembly to provide supplementary protection against corrosion during service. Additional finishes such as fluid resistant primer or fuel tank coatings must be applied after any repair or rework of these parts or areas is carried out.

Water-displacing corrosion inhibiting compound (CIC) must be reapplied to the repair areas after application of all other finishes.

CORROSION PREVENTION

The design and manufacture of the C Series aircraft incorporates many corrosion preventive features. These include the use of corrosion resistant materials wherever possible, protective treatments and finishes during manufacture of parts, extensive use of sealants during assembly, and the application of protective treatments and finishes to complete assemblies.

It is important that corrosion damage is found and repaired as soon as possible. If corrosion damage is not found and repaired, structural failure can occur. Corrosion damage can cause deterioration of the structure to the point where replacements must be made in order to prevent serious structural failure.

CORROSION PREVENTION AND CONTROL PROGRAM (CPCP 51-21)

The CPCP is established to maintain corrosion protection of the aircraft against structural degradation caused by environmental interaction. There are three levels of corrosion that relate to the degradation extent that affect the primary structure:

Level 1: Corrosion occurring between successive inspections that can be reworked or blended out within allowable limits and does not require structural reinforcement or replacement.

Level 2: Corrosion occurring between successive inspections that requires a single rework or blending which exceeds allowable limits and requires a repair/reinforcement or replacement of an PSE but not a critical airworthiness concern.

Level 3: Corrosion found during the first or successive inspections which is found to be an urgent airworthiness concern requiring immediate action.

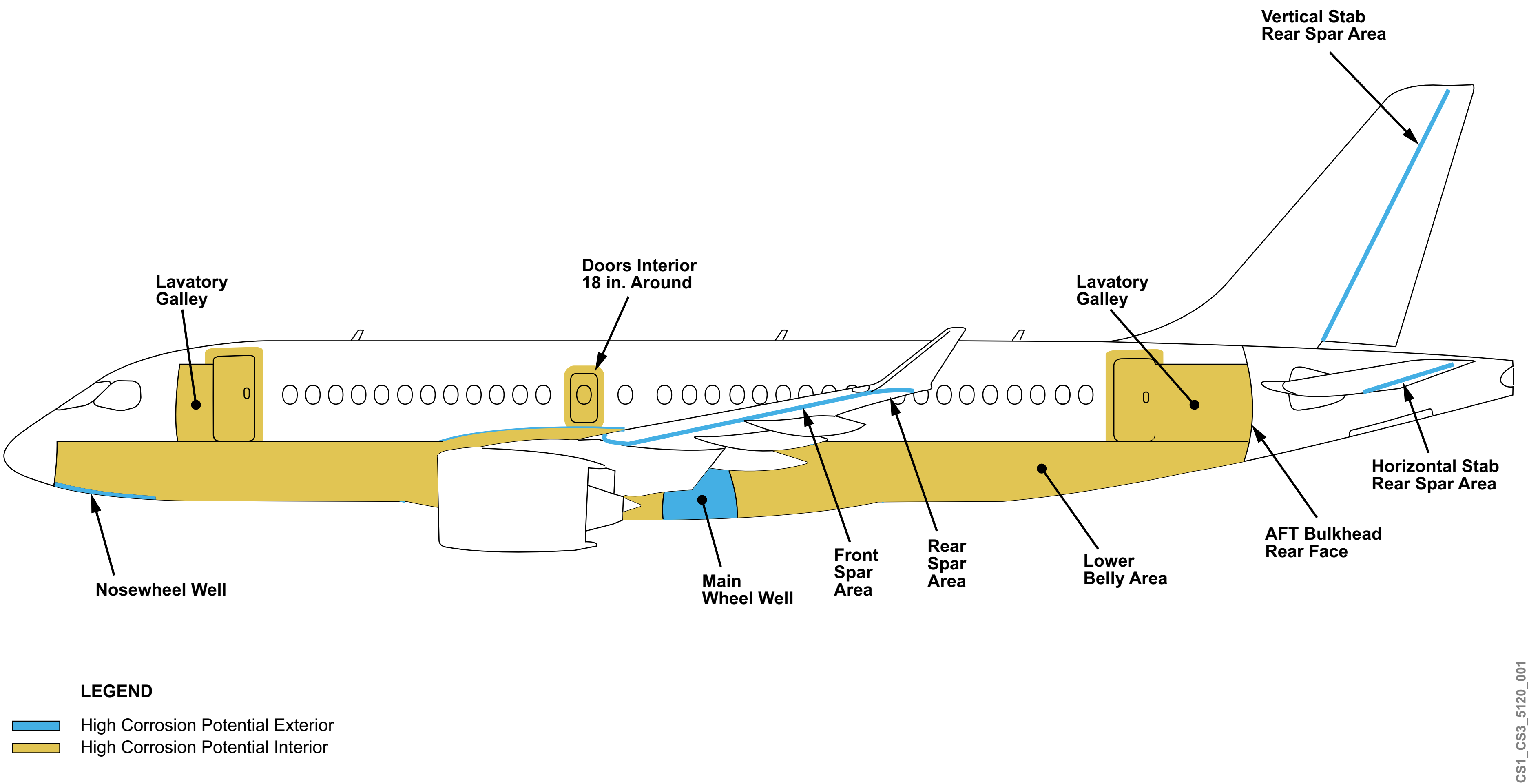

Figure 4: Protective Treatment and Corrosion Prevention

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ATA 51A -Standard Practices – Structures (Damage Classification, Assessment, and Repair) – Example

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

C SERIES STANDARD PRACTICES AND STRUCTURES

For dimension and areas information, refer to:

Aircraft Maintenance Publication (AMP) 06-10-00

For structure classification, refer to ASRP 51-00-00

For zones information, refer to AMP 06-30-00

For stations information, refer to AMP 06-20-00

For panels, floors, fairings information, refer to AMP 06-40-00

HOW TO USE THE STRUCTURE AND REPAIR PUBLICATION

1. Find out the damaged location on the aircraft.
   Do this with the help of a station diagram for the concerned ATA−chapter (AMP 06-20-00). Find out the type of damage (dent, scratch,...). Measure the size of the damage (length, width, depth and distance). Do this as exact as possible.
2. Select the aircraft applicable tail number (serial) on the Navigator IETP for your aircraft. As Bombardier Navigator is accessible online and maintained by Bombardier, this gives you access to the latest revision of the technical publications.
3. Identify the material of the damaged part.
   Refer to the applicable material identification data in the ASRP 51-00-00. This gives information about the material and the thickness in the damaged area.
4. Find the allowable damage data for the damaged structure.
   This can be found in ASRP 51-12-00. A link is given to the applicable action, depending on whether the damage is within allowable limits, or outside allowable limits.
5. Find the repair data for the damaged structure.
   This can be found in the ASRP for the applicable ATA sub-section. There are different repair options (repair at stringer, between stringer..., external or flush repairs). ASRP 51-12-01 gives treatment of damage within allowable limits. These are detailed in the ASRP 51-71. Depending on the situation, the maintenance staff can choose one of these repairs. Repair evaluation guidelines are given in ASRP 51-13-02 to help determine the proper repair applicable for your situation.

NOTE

It is necessary to read all information including all notes, warnings and cautions before starting any action. For example, some times a restriction is given at the end of the instructions or as an insignificant note.

DAMAGE CLASSIFICATION AND REPAIR

CLASSIFICATION OF DAMAGE

(Refer to ASRP 51-10-06)

To agree with the loss of strength in a structural part at a given location, damage is classified as follows:

Permitted damage:

Permitted damage is damage that is not important to the structural safety or safe operation of the aircraft. This type of damage is divided into two groups, as follows:

1. Corrective procedures or repairs are not necessary
2. Small corrective procedures or repairs that are necessary such as:
    - Removal of nicks and scratches
    - Removal of light surface corrosion
    - Corrosion preventive treatment

Repairable damage:

For damage that is more than the permitted damage limits, it is necessary to add repair parts or replace the part. Make repairs as shown by the repair instructions in the ASRP, or by a repair instruction approved by the applicable engineering authority

Damage where replacement of parts is necessary:

Replacement of parts is in general the most satisfactory repair procedure. It is necessary when:

- Parts are damaged too much to repair
- Damaged parts are too short to repair by insertion
- Forged, cast, and machined fittings are damaged more than permitted damage limits
- After a repair to highly-stressed parts, the allowance for safety will not be sufficient
- Parts that have their strength decreased by the effects of fire or heat

REPAIR TYPES

When repairs are done, they are usually done with patches (reinforcement) or insertion, as follows:

Patch repair

To repair damage (usually cracks, creases, or holes) with patches, the damaged structure is removed and reinforcement parts are put on the damaged area. The patch is installed using fasteners so that the necessary structural strength of the part is put back with no effect on the function of the structure.

Repair by insertion

To repair parts by insertion, it is necessary to replace the damaged portion of a part with a new portion. This inserted part transmits the structural loads that the surrounding structure by fastening both ends of the insertion to the structure with splices.

External patches on aerodynamic surfaces

With the approval of the applicable engineering authority, external repair patches that protrude above the aerodynamic moldlines are used. This quick repair is applied and is used to decrease the downtime of the aircraft. The fastener locations must be structurally correct for subsequent use for aerodynamic (flush-type) repairs. It can also be used as a permanent repair if it is aerodynamic and visually satisfactory.

Aerodynamic (flush-type) repairs

Use this type of repair on external aerodynamic surfaces whenever possible and where a repair patch that protrudes is not satisfactory. The damaged area is cut away and an internal doubler (patch) is installed to transmit the structural loads. The skin area that was cut away is made aerodynamic with fillers made from the same material as the skin. Alternatively, epoxy-type aerodynamic filler is used.

Continued Airworthiness of Structure General (Refer to ASRP 51-13-00)

Repair approval and release

A repair can be approved in three stages: an interim, second and third stage approval.

Interim approval and release

Interim approval can be given for repairs to Principal Structure Element (PSE) and/or Fatigue Critical Baseline Structure (FCBS) on a single aircraft or component serial number that have the necessary ultimate static strength where it is not possible to complete the Damage Tolerance Analysis (DTA) immediately

Second stage approval and release

1. For specific repairs applicable to a single aircraft or component serial number, a DTA and the Damage Tolerance Inspection (DTI) intervals for the repair must be approved by the applicable airworthiness authority, usually within 12 months of repair incorporation.
2. Multiple aircraft applicability ASRP and Generic Repair Engineering Order (GREO) repairs must be at stage two or stage three when first released.

Final stage 3 approval and release

Final approval and release of a repair occurs when the complete damage tolerance inspections including the nondestructive test (NDT) technique has been developed and approved.

Static strength approval (SSA)

For not-approved repairs, you can get approval for the static strength analysis from one of the following authorities:

- Bombardier Aerospace (BA)
- Other specified/approved persons by the applicable airworthiness authority to approve STAs
- The operator or repair facility, if they are approved by the applicable airworthiness authority to approve static strength analyses
- The applicable airworthiness authority

Allowable damage

The information to be found within allowable damage page block enables the operator to define whether a damaged airplane may be returned into service without repair.

A permitted allowable damage has no significant effect on the strength or fatigue life of the structure, which must still be capable of fulfilling its function.

Allowable damage may require minimal rework such as cleanup of small scratches or minor corrosion damages.

Allowable damage is damage for which an immediate structural repair is not necessary. You must blend out the allowable damage and protect the surface on the component.

EXAMPLE:

53−21−01 Allowable Damage − SKIN CROWN PANEL FR15 TO FR30

This data module gives the Permitted Damage Limits (PDL) for the forward fuselage crown skin panel.

C SERIES STANDARD PRACTICES AND STRUCTURES

Following are the most useful ATA references from ASRP

51-00-00 STRUCTURAL CLASSIFICATION

51-12-00 PERMITTED DAMAGE

Treatment of permitted damage on metallic structure

Treatment of permitted damage on composite structure

51-12-01 TREATMENT OF ALLOWABLE DAMAGE

Scratches

Light damage

Nicks and gouges

Dents

Cracks

Small holes

51-71-12 STANDARD REPAIR

Repair of attached flanges in frames or ribs

Repair of stringer cut-outs in frames or ribs

Repair of lightening holes

Repair of attached flanges with angles clips

Repair of rib corners

51-71-28 STANDARD WEB REPAIR

Rectangle web repair

Circular web repair

Example web repair

51-71-59 MACHINED, EXTRUDED AND FORMED SECTION STANDARD REPAIR

Flange standard repair

Extruded and machined free flange standard repair

Formed free flange standard repair

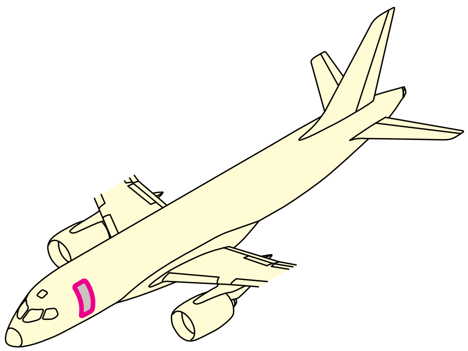
Attached flange, extruded and machined sections standard repair

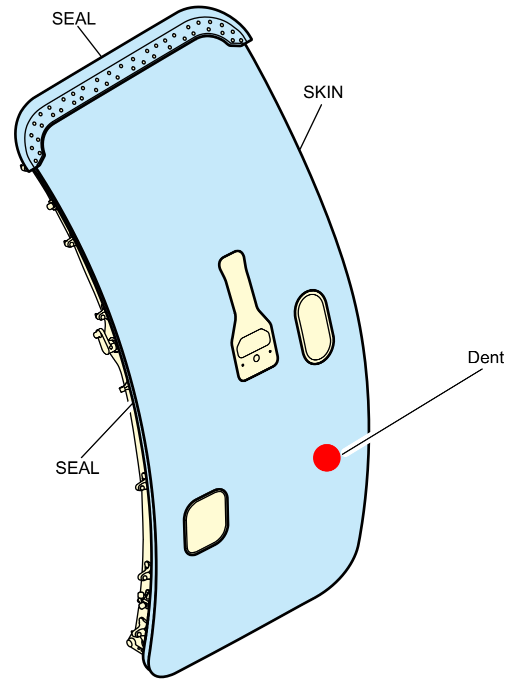
Formed section splices standard repair

Extruded section splice standard repair

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

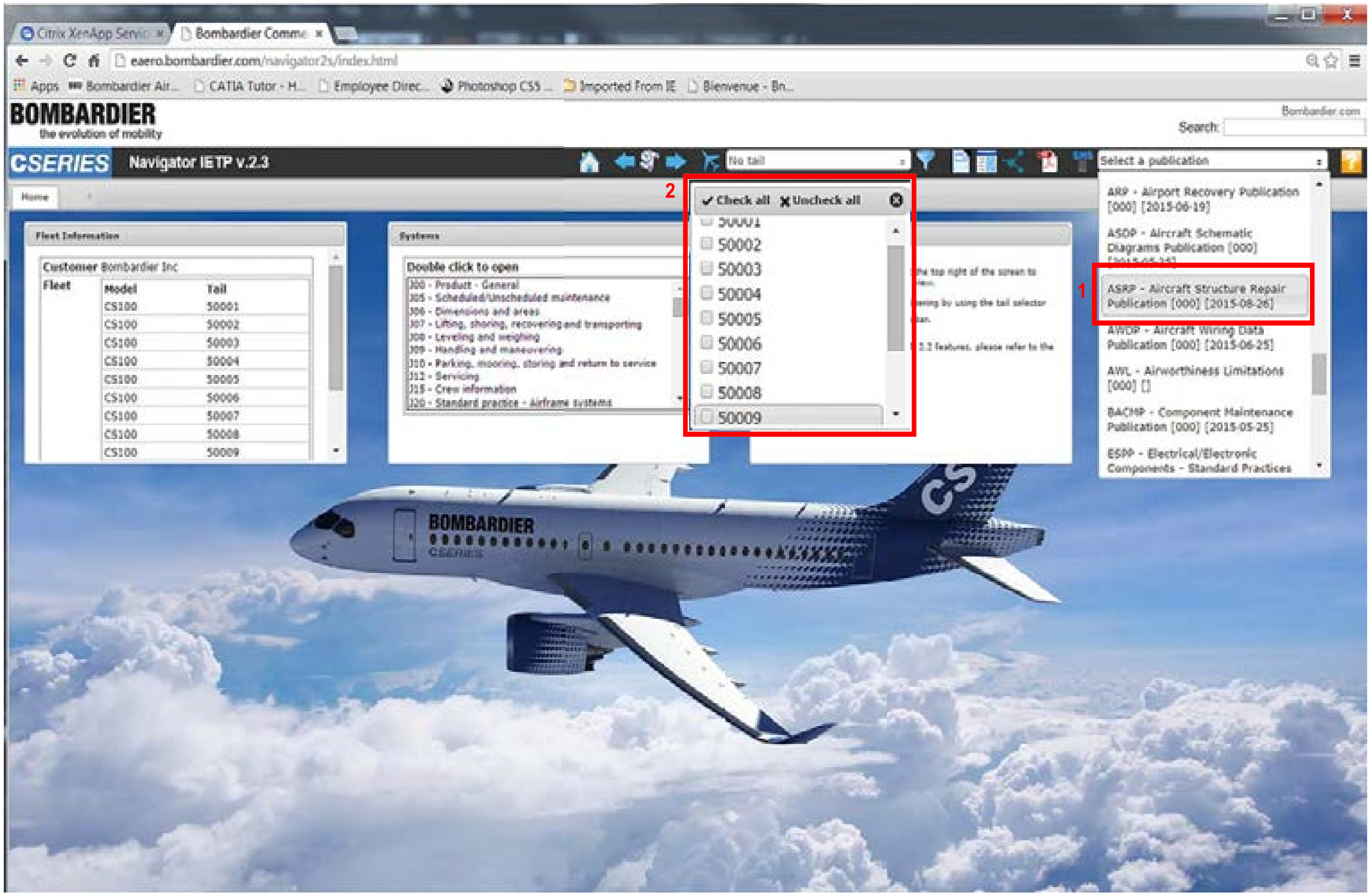
Damage: Dent on L/H forward passenger door outer skin, lower aft section.





DAMAGE ASSESSMENT AND REPAIR EXAMPLE

Navigator IETP



1. Choose Aircraft Structure Repair Publication (ASRP) from the pull down menu (revision date is in brackets)
2. Choose the aircraft tail # for applicability

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

3. Identify damaged structure (primary/secondary)
   (Refer to ASRP 51-00-00)

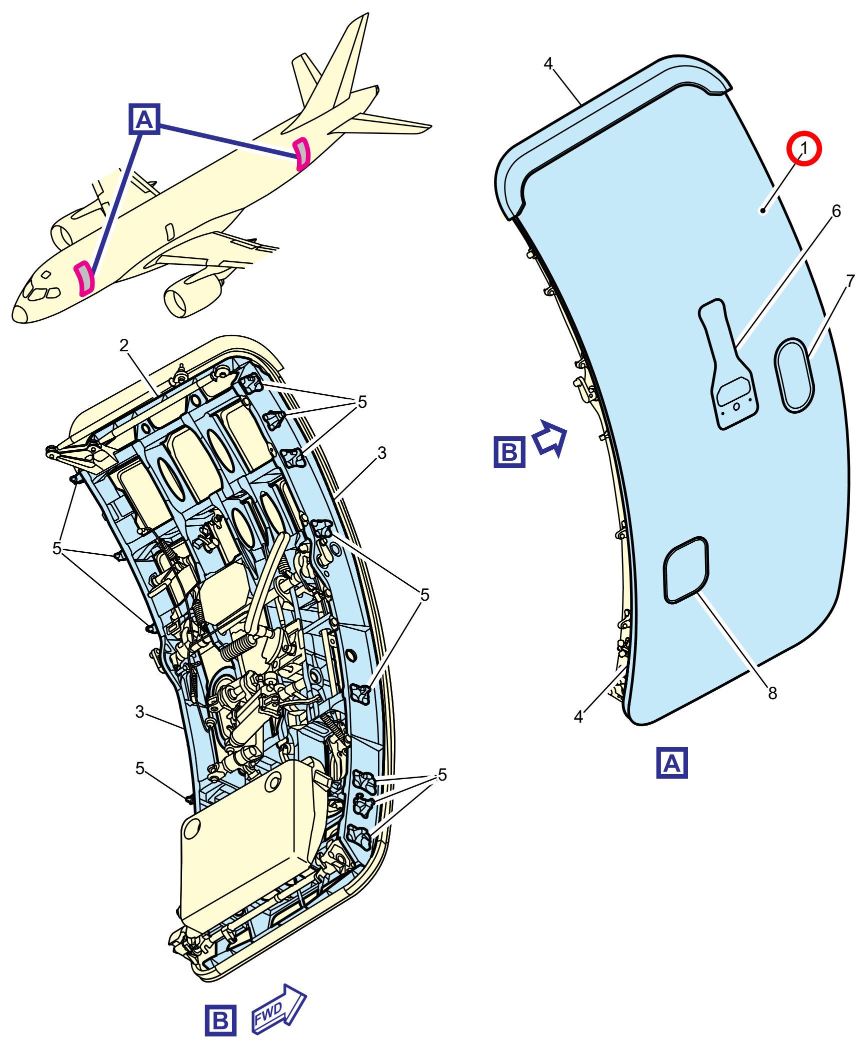


Table 2 Forward and aft passenger doors, structural classification

Item	ATA Number	Nomenclature	Primary		FCBS	Secondary
			PSE	Others		
1	52-11-01	Skin, forward passenger door (external skin only)	X	-	X	-
	52-12-10	Skin, aft passenger door (external skin only)				
2	52-11-02	Structure, forward passenger door	X	-	-	-
	52-12-02	Structure, aft passenger door				
3	52-11-03	Edge structure, forward passenger door	X	-	X	-
	52-12-03	Edge structure, aft passenger door				
4	52-11-07	Seals, forward passenger door	-	-	-	X
	52-12-07	Seals, aft passenger door				
5	52-11-27	Fittings, forward passenger door	X	-	X	-
	52-12-27	Fittings, aft passenger door				

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

4. Identify permitted damage

   General damage limits (ASRP 52-11-01)
   Refer to Table 6 for general damage limits on all areas except the areas specified for local areas of table and adjacent to cutouts on Fig. 1.

NOTE

Use an approved repair procedure to repair damage that does not agree with the permitted damage limits. Refer to BD500-A-J51-13-01-00AAA-913A-A.

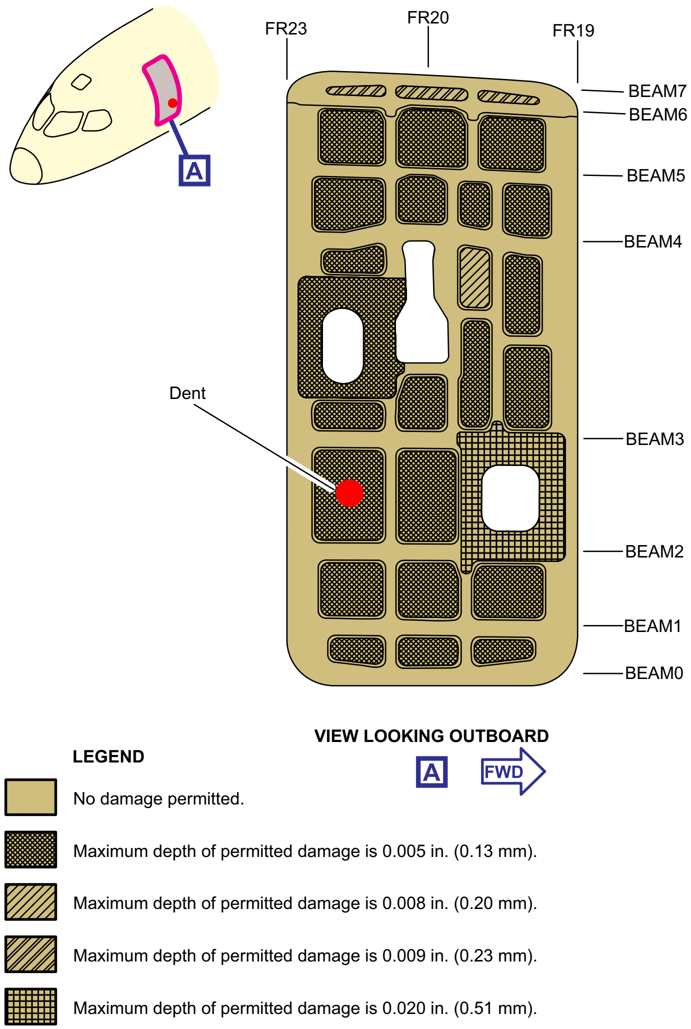


Fig. 1

(Refer to ASRP 52-11-01)

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

PERMITTED DAMAGE

(Refer to ASRP 52-11-01)

Dents		
Location of damage	Limit	Remarks
Skin joints	Not permitted	--
Other	Refer to Figure 2. Dents must be separated by at least one-half (1/2) the width of the largest adjacent dent. Dents must not be closer than 1.0 in. (25.40 mm) or more from the fastener line in the skin panel non- riveted area. The ratio W/D must not be less than 20. Depth limit given applies to all areas but does not include Zone 1 areas, refer to BD500-A-J51-14-06-00AAA-361A-A. The depth of dents in Zone 1 areas must agree with the aerodynamic smoothness, refer to BD500-A-J51-14-06-00AAA-361A-A. Dents must not be larger than 2.0 in. (50.8mm) in diameter.	Dent must be smooth and have no sharp creases, gouges, or cracks. Dent must not cause damaged or loose fasteners, or elongated fastener holes. Refer to the logic diagram in BD500-A-J52-00-01-00AAA-661A-A to determine if the removed damage needs to be repaired. For dents with sharp creases, refer to (Fig TBD) for the flight operation limits

Table 6

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

Damage depth: 0.050 in (D)

Damage length: 1.5 in (W)

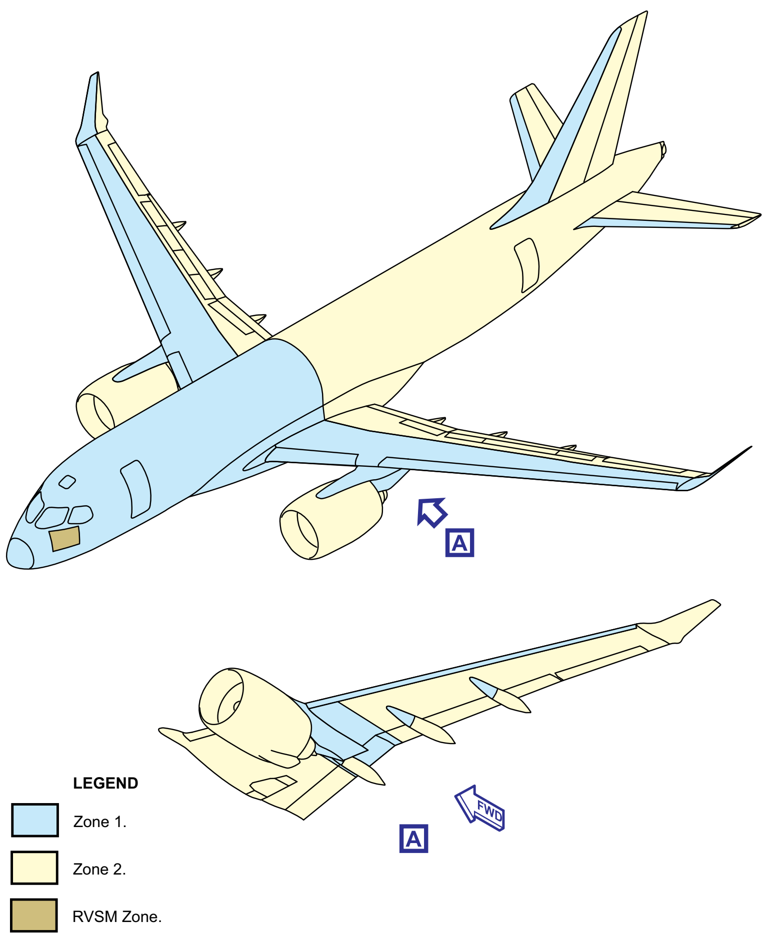


Fig. 1

(Refer to ASRP 51-14-06)

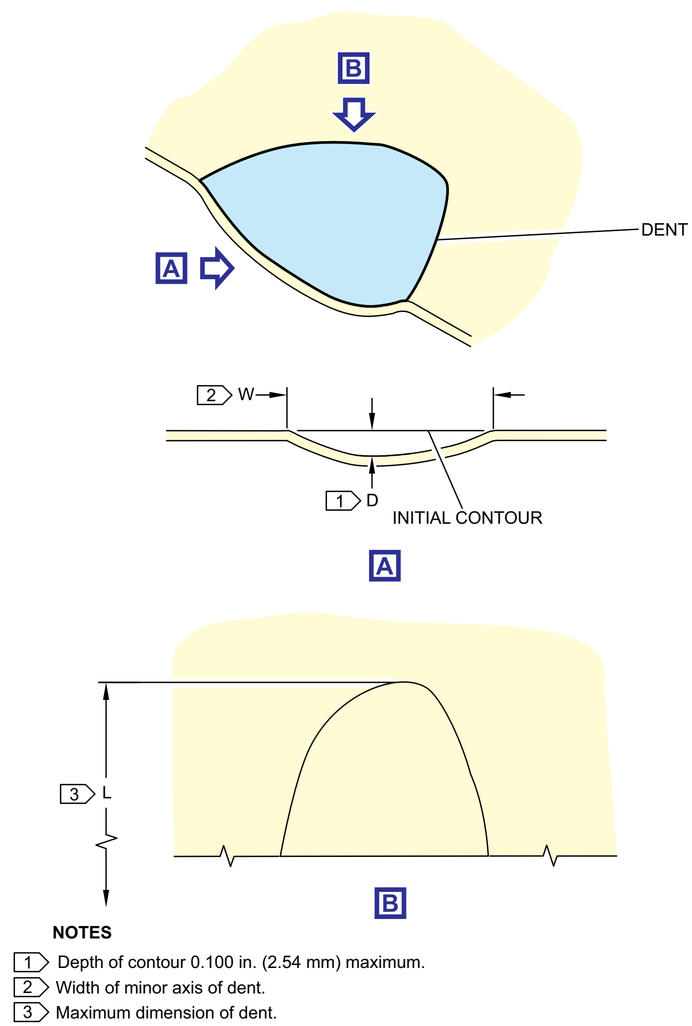


Fig. 2

(Refer to ASRP 52-11-01)

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

5. Identify damaged material and thickness
   ASRP 52-11-01

Item	Nomenclature	MFC and Part #	Material specification	Thickness In. (mm)	Finish code
	Structure assy. Stage 1 FWD PASS DR	88308 C01518110			
1	SKIN, FWD PASS DR	88308 C01511011-003	BAMS516-015 Aluminum alloy sheet 2524-T3	0.125 (3,18)	A001

Table 2 Key to Fig. 1

Chem-mil skin and pocket thickness	
Code	Skin thickness in. (mm)
A	0.045 (1.14)
B	0.068 (1.73)
C	0.086 (2.18)
D	0.125 (3.18)

Table 3 Key to Fig. 2

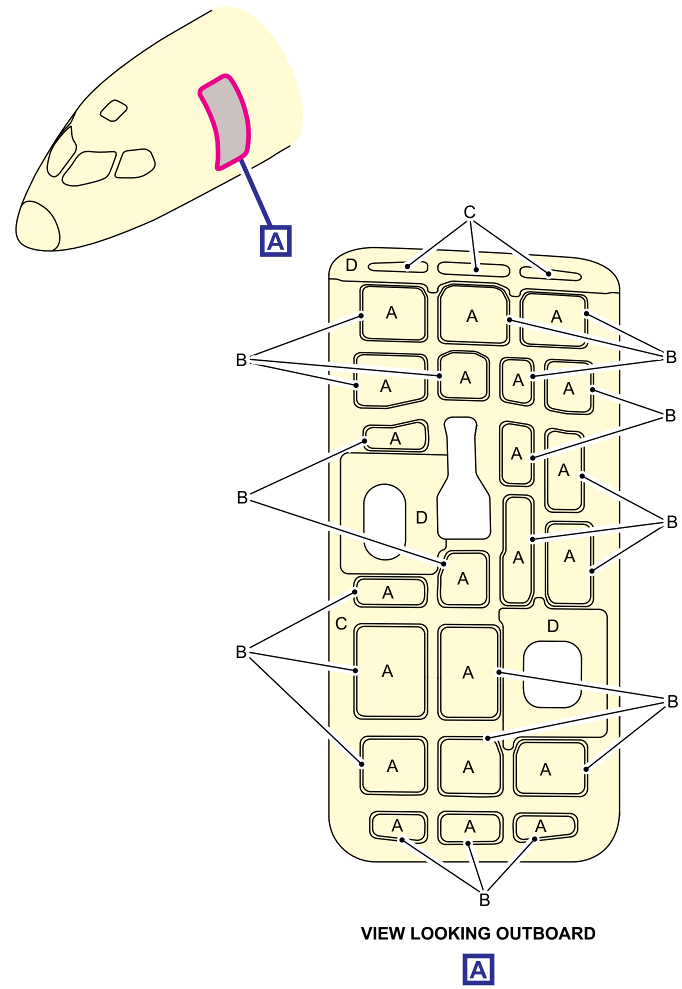


Fig. 2

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

6. Repair procedure (Refer to ASRP 51-12-01)

When dents are found, refer to the permitted damage limit (PDL) in the applicable data modules in ATA 52 to 57 and continue as follows:

- If the dent dimensions or quantity of dents are less than the maximum permitted in the applicable PDL, the area can be filled with aerodynamic filler, if necessary for aerodynamic reasons or for appearance (repair per ASRP 51-12-01).
- If the dents are larger or the quantity of dents are more than permitted by the applicable PDL, do a structural repair.

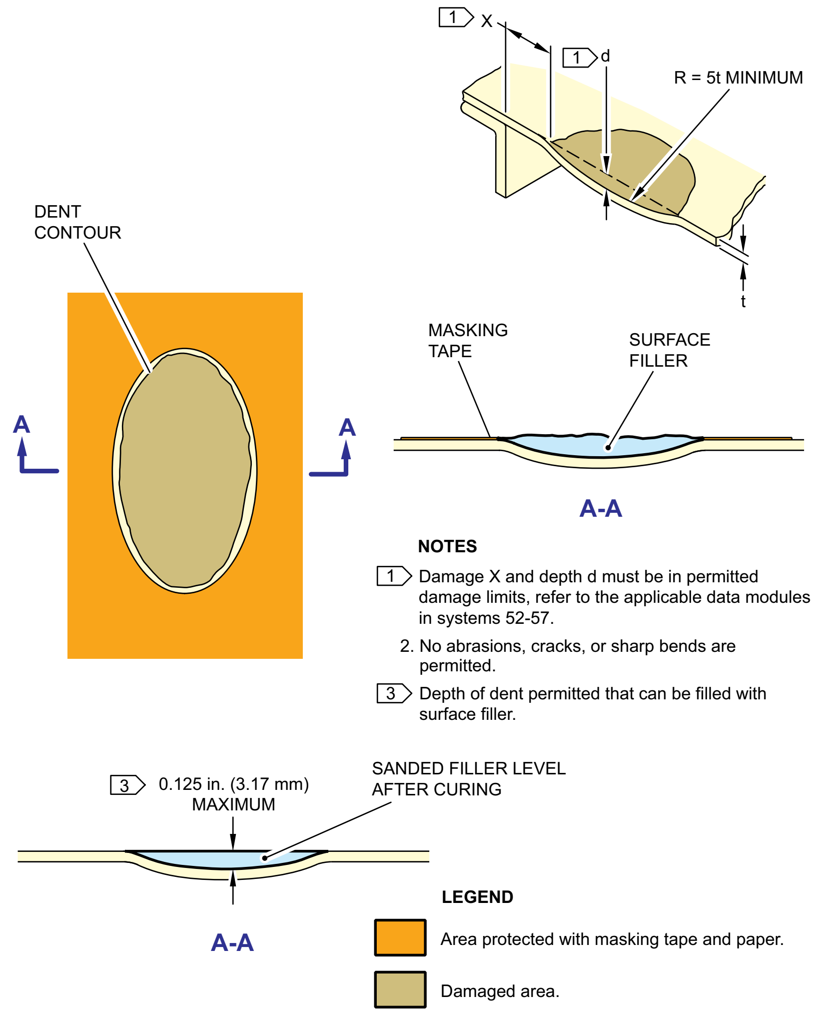


Fig. 1

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

If dent is found **outside permitted damage limits**, repair per ASRP 52-11-01.

Single Bay Repair

This data module gives the approved generic repair procedure for the Forward Passenger Door (FPD) skin for damages that exceeds the Permitted Damage Limits (PDL).

This data module complies with the Repair Engineering Order (REO) 500-52-11-005.

When damage occurs outside the boundaries of this data module, report the damage to Bombardier Aerospace Customer Response Centre. Refer to BD500-A-J51-13-00-00AAA-028A-A

CAUTION

For damage beyond the specified limits, you must do more strength and damage tolerance evaluations (refer to BD500-A-J51-13-00-AAA-028A-A). If you do not do this, you can cause more damage to the aircraft structure.

WARNING

Be careful when you do work with or near cutting equipment. You can cause damage to equipment and/or injury to persons.

A Damage Tolerance Evaluation (DTE) has determined that a special repeat inspection is necessary for this repair at every TBD flight hours/TBD flight cycles.

The inspection method is TBD.

Applicability

This repair procedure is applicable to the FPD skin. Refer to Fig. 1.

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

CAUTION

Before and during repairs, make sure you obey the precautions that follow:
- Protect the wiring, electrical and electronic equipmentfrom metal debris or other contaminants
- Make sure that you do not damage the insulation when you move or disconnect the electrical wiring
- Make sure that you do not damage the wiring when drilling or cutting the structure
- Make sure that you do not damage the structure parts which are adjacent and below when you cut the damaged structure
- Clean and vacuum the repair area before job completion.
If you damage the structure or electrical wiring, you must stop the repair and contact Bombardier Aerospace Customer Response Centre before you continue with the repair.

Preparation of damaged structure

Cut and discard the damaged area of the skin. Refer to BD500-A-J51-24-36-00AAA-659A-A.

- Do a detail inspection to make sure that there is no damage to the underlaying structure below the skin cutout. Refer to BD500-A-J51-11-00-00AAA-311A-A.
- Do a detail inspection to make sure that there is no damage to the edges of the skin cutout. Refer to BD500-A-J51-11-00-00AAA-311A-A.
- Do a liquid penetrant inspection (refer to NDT DM TBD) or eddy current inspection (refer to NDT DM TBD) to make sure that there is no crack on the edges of the cutout.
- Clean the repair area.
- Touch up cut edges with the finish code TBD. Refer to BD500-A-J51-21-16-00AAA-030A-A.

Prepare repair parts per table below

Repair Materials				
Item	Description	Material specification	Thickness in. (mm)	Finish code
1	DOUBLER	Aluminum Alloy Clad Seet 2024-T3	0.063 (1.60)	TBD

Table 6 Repair Materials

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

INSTALLATION OF REPAIR PARTS

(Refer to ASRP 52-11-01)

Make the repair items to fit the adjacent structure.

Refer to Fig. 1 & Fig. 2 for the shape, dimensions of repair materials, and type of fasteners necessary for the repair.

- Drill holes in the repair parts to match the existing hole pattern in the skin. Refer to Fig. 2 to follow the new holes pattern.
- Maintain the existing fastener pitch at the repair area unless specified differently.
- Make sure that there is a distance of 4D to 9D (D = fastener diameter) between all new fasteners unless specified differently.
- Make sure that there is a minimum edge distance of 2D for the metal parts unless specified differently.
- Make sure that there is a minimum edge distance of 3D to skin cutline unless specified differently.
- Remove nicks and burrs, and make the holes smooth.
- Apply finish code TBD. Refer to BD500-A-J51-21-16-00AAA-030A-A.
- Drill applicable new holes in the original skin through the repair parts. Refer to BD500-A-J51-40-11-00AAA-652A-A.

NOTE

- Quantity of fasteners and pitch can vary in each area
- It is permitted to use the one higher size fastener on the existing holes if necessary
- Spotfacing is necessary but should be limited to corner areas

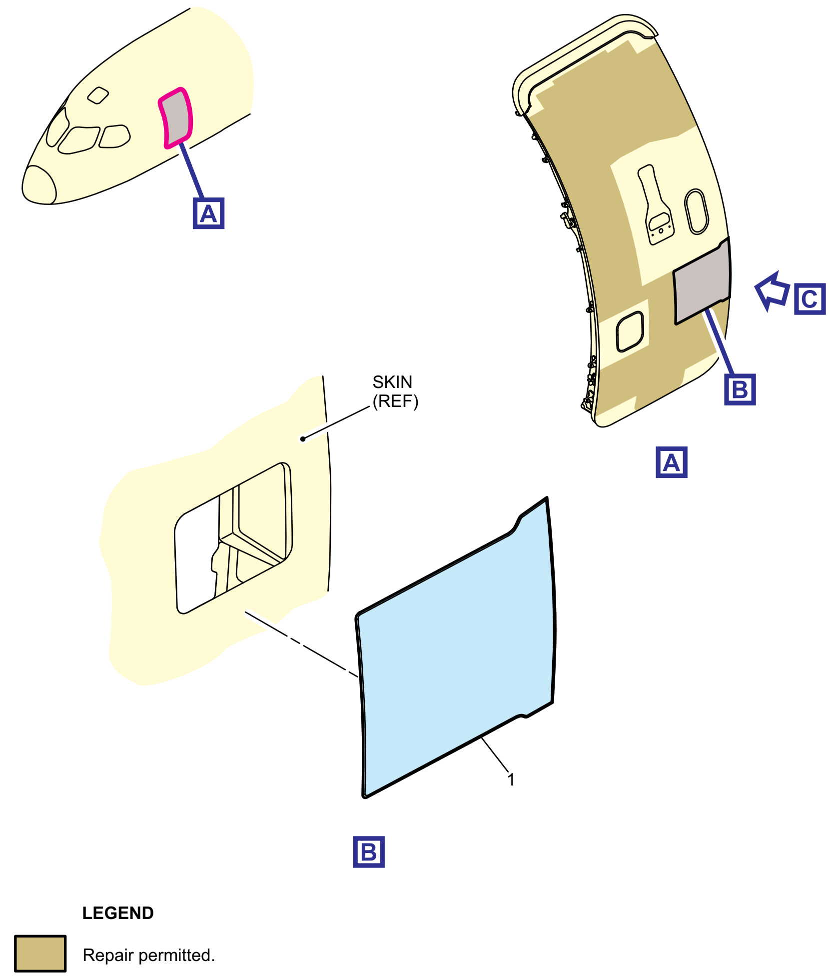


Fig. 1

(Refer to ASRP 52-11-01)

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

- Disassemble the repair parts, remove nicks and burrs, and make the holes smooth.
- Apply finish code TBD. Refer to BD500-A-J51-21-16-00AAA-030A-A.
- Wet assemble all items, seal faying surfaces, fillet seal, and fill gaps. Refer to BD500-A-J51-23-00-00AAA-259B-A.
- Install the solid rivets. Refer to BD500-A-J51-42-06-00AAA-078A-A.
- Apply the aerodynamic sealant on the chamfered edges of the doubler (1) and the skin. Refer to BD500-A-J51-23-00-07AAA-259B-A.
- Restore the finish TBD. Refer to BD500-A-J51-21-16-00AAA-030A-A.

Requirements after job completion

Action/Condition	Data module/Technical publication
Remove all tools, equipment, and unwanted materials from the work area	
Install the removed components	
Do the aerodynamic smoothness check	BD500-A-J51-14-06-00AAA-361A-A

Table 7

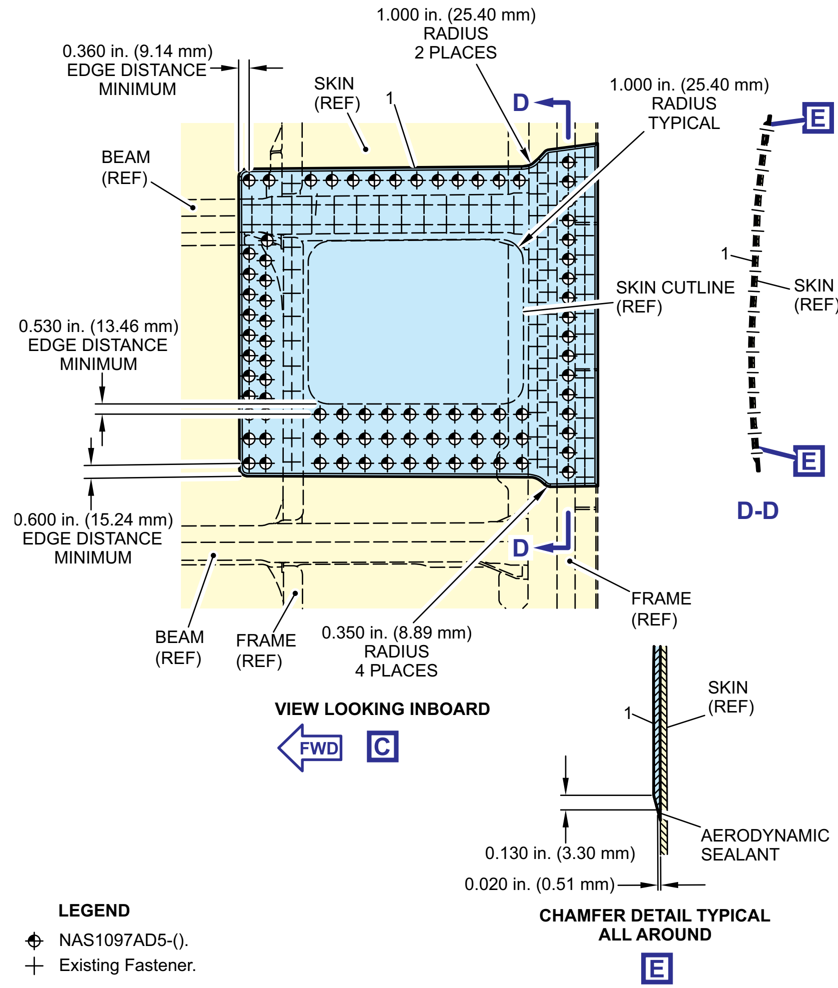


Fig. 2

(Refer to ASRP 52-11-01)

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

7. Aerodynamic smoothness check (Refer to ASRP 51-14-06)

General

This data module gives limits and procedures that will help to keep airframe, nacelle, and pylon in an acceptable aerodynamically-clean condition.

The aircraft needs an aerodynamically clean and smooth shape to keep its high performance. In general, a change from the initially manufactured aerodynamic shape of the aircraft makes it aerodynamically unclean. Although some decrease of the initial aerodynamic smoothness must be expected during regular service, operators must make full effort to keep the initial shape and smoothness of the aircraft.

The designed contour of the airplane must be fair and smooth. The outline of the assembled airplane must follow this contour within the given tolerances.

Contour tolerances

For straight surfaces, the straight edge used for checking must have a length equal to surface up to a maximum of 36.000 in. (914.40 mm).

For curved surfaces, the appropriate contour boards or a curved spline must be used. Minimum length of 36.000 in. (914.40 mm) or total length if surface is less than 36.000 in. (914.40 mm).

All control surfaces or moveable aerodynamic structure must be adequately sealed to minimize leakage flow in cruise flight.

External non-flush repairs are commonly used to return the aircraft to service quickly. However, installation of a number of external non-flush repairs can significantly reduce the performance of the aircraft. In addition, external repairs must respect limitations of certain areas where pitot-static tubes and other airflow sensors are located. Skin repairs in the more aerodynamically critical areas must be of the flush type in order to keep the best aircraft performance.

Aerodynamic zones

The external surfaces of the aircraft are divided into two major areas to specify limits that agree with the aerodynamic importance of each area. These areas are (ref Fig. 1):

- Zone 1 & Zone 2

In zone 1 the airflow is most sensitive to the imperfections or sudden changes in the shape. It is not permitted to install any type of lights, antennas, or devices that protrudes out of Outer Mold Line (OML) in Zone 1.

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

Aerodynamic smoothness check

Contour deviations

General

Refer to Figure 2

The contours of the aircraft must be smooth and must agree with specified tolerances. Refer to contour tolerances (previous page).

Definition

H - Maximum loft contour deviation.

h - Height or depth of local contour deviation (elevation or depression).

L1 & L2 - Length of local contour deviation from edge to maximum deviation.

h/L1 or h/L2 - Rate of change of local contour deviation in table 6

Loft contour deviation

The final contour of the airplane must not deviate from the outer mold line (OML) by more than the values (H) in table 6.

Local contour deviation

Unless specified differently, any local contour of the airplane must be fair and smooth with elevations, depressions, or flat spots not more than the specified values as given in table 6.

Contour deviations are not additive. The local contour deviations must be within the maximum permitted loft contour deviation as specified in table 6.

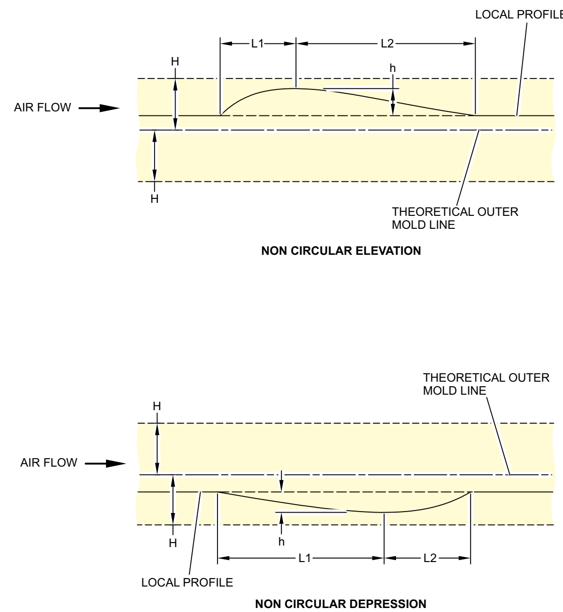


Fig. 2

Table 6 Local & loft contour deviation

Zone	Maximum OML deviation (H) in. (mm)	Maximum height of depth (h) in. (mm)	Maximum rate of change (slope) (h/L1 or h/L2) in. (mm)
Zone 1	+/- 0.030 (0.76)	+/- 0.020 (0.51)	0.003 (0.08)
Zone 2	+/- 0.660 (1.52)	+/- 0.030 (0.76)	0.005 (0.13)
Zone 2 (fuselage)	+/- 0.060 (1.52)	+/- 0.030 (0.76)	0.010 (0.25)

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

Aerodynamic smoothness check

(Refer to ASRP 51-14-06)

Gaps and mismatches (steps)

Gaps and mismatches that occur at an angle of not more than 5 degrees to the forward and aft axis of the aircraft are known as longitudinal. For those that occur at an angle of more than 5 degrees to the forward and aft axis of the aircraft are known as lateral.

Gaps

Fill all gaps between skins that are permanently attached and between skins and structure with aerodynamic sealant. Refer to BD500-A-J51-23-00-00AAA-259B-A.

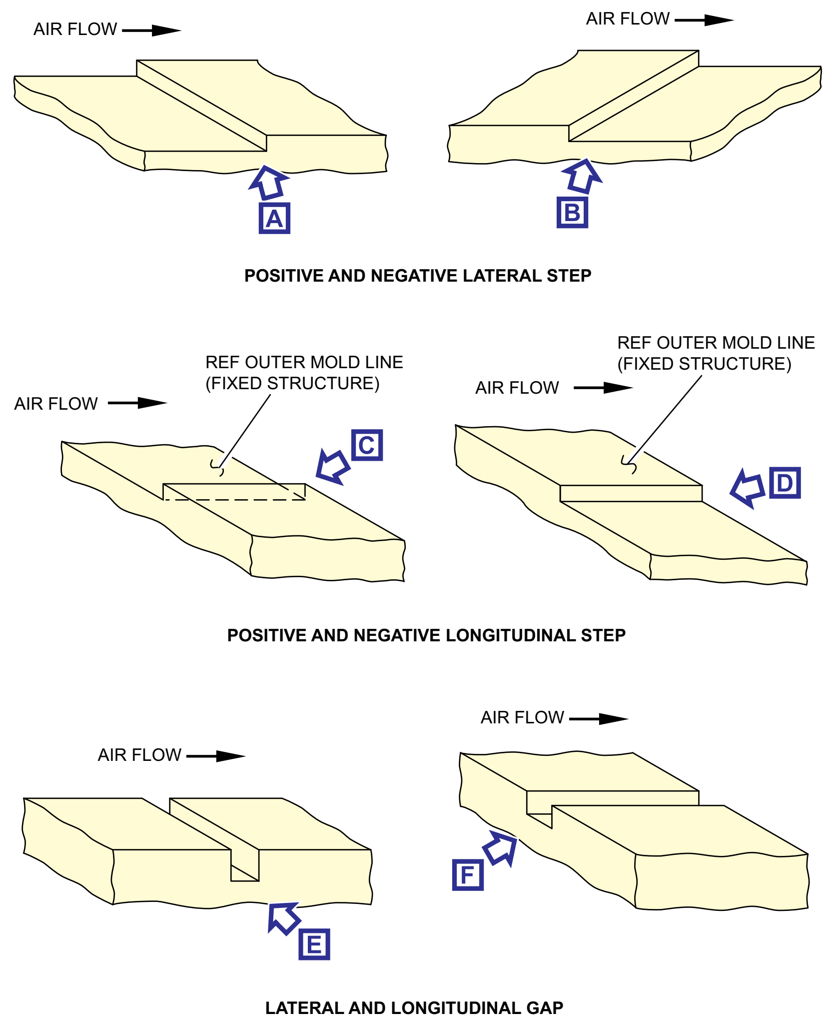


Fig. Gaps

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

Aerodynamic smoothness check

Mismatches

The contours of aircraft must be as free of mismatches as possible. Where mismatches occur due to tolerance build up, chamfer may be employed to reduce the effect of the step.

Alternatively, the permitted step may be increased if the use of chamfer is feasible.

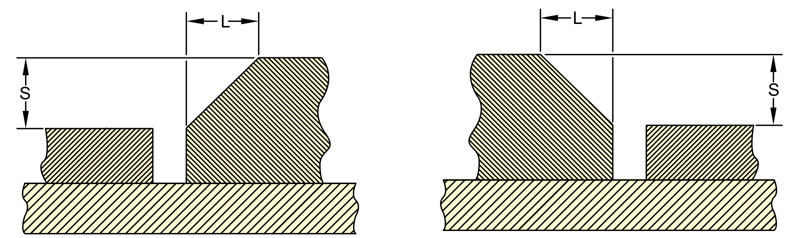
The conditions for the use of chamfer are given below.

The length of the chamfer (L) must not be less than two times of step (2S) and not larger than five times of step (5S). Within these limits the length of the chamfer must be as large as practically possible.

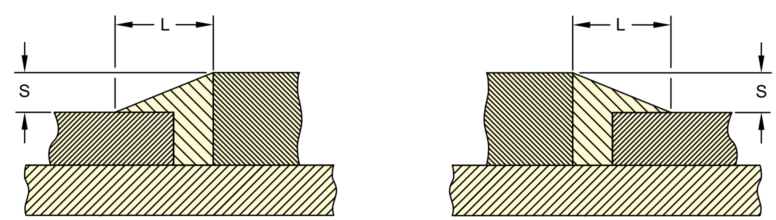
Where the sufficient chamfer is not possible at the junction of fixed skins or structures, aerodynamic sealant can be used.

The length of the sealant (L) must not be less than two times of step (2S) and not larger than five times of step (5S).

The approaches of gaps and steps can be used in conjunction. Under these conditions, the permitted limits for gaps and mismatches for each division of the wing and each zone of the remaining surfaces are given as follows:



USE OF CHAMFER



USE OF AERODYNAMIC SEALANT

Fig. Mismatches

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

Mismatches (cont’d)

(Refer to ASRP 51-14-06)

Doors

The gaps necessary at the skin panel must be as small as is practically possible, within the limits specified in Table 12.

A positive step must be interpreted as door protruding from structure OML and negative step as a door recessing from structure OML.

The step requirement do not apply in the lap joint area for all the doors.

Doors are controlled-replaceable parts that are supplied with a trim allowance on the skin.

The step and gap must be measured when the aircraft is in the fully pressurized condition for doors refer to Table 13.

DAMAGE ASSESSMENT AND REPAIR EXAMPLE

Mismatches (cont'd) (Refer to ASRP 51-14-06)

Doors

Table 12 Maximum allowable steps and gaps

Type of function	Aerodynamic zone	Maximum gap in. (mm)	Max. step in (mm)		
			Lateral in. (mm)	Lateral chamfer to be used in. (mm)	Longitudinal in. (mm)
Between fixed skins or removable structures (sealant used) ex: fuselage skins, wing plank, and Fairings	1	Not applicable: the gap is sealed. Refer to the Engineering drawing for the gap value.	±0.020 (±0.51)	±0.030 (±0.76)	±0.040 (±1.02)
	2		±0.030 (±0.76)	±0.040 (±1.02)	±0.040 (±1.02)
Removable structures (not sealed) open gap ex: access doors, spoilers	1	0.050 (1.27)	±0.020 (±0.51)	±0.030 (±0.76)	±0.040 (±1.02)
	2	0.070 (1.78)	±0.030 (±0.76)	±0.040 (±1.02)	±0.040 (±1.02)

Table 13 Step and gap for pressurized doors

Type of door	Maximum gap in. (mm)		Maximum steps (1) in. (mm)	
	Lateral	Longitudinal	Lateral	Longitudinal
Forward Passenger Door (FPD)	Forward 0.200 (5.08) Aft 0.250 (6.35)	Upper NA Lower 0.200 (5.08)	±0.040 (±1.02)	±0.040 (±1.02)

1. A positive step to be interpreted as door protruding from structure OML, and negative step as a door recessing from structure OML The step between doors and structure should not be measured at lap joint area. In location where a scuff plate is installed the step tolerance is +0.000 to -0.0096 in. (+0.00 to 2.44 mm).

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ATA 52 - Doors

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

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Flight Crew Emergency Exit Hatch 5

Flight Deck Door 6

52-10 PASSENGER AND SERVICE DOORS

GENERAL DESCRIPTION

There are two passenger doors located on the left side of the aircraft, and two service doors on the right side of the aircraft. Both passenger and service doors are similar in design and operation, but differ in size.

Each door is a type-C plug-type door that can be operated from the inside, or outside of the aircraft using handles connected to the internal mechanisms. The internal mechanisms sequence the opening and closing of the doors.

Each door is used as an emergency exit and is equipped with an escape slide that can be deployed in an emergency to evacuate the aircraft. The escape slides are installed on the lower part of the door under a bustle.

The slide is armed for deployment or disarmed using the mode select handle. When the slide is armed, the slide automatically deploys when the door is opened. A mechanical slide arm/disarm flag provides the status of the escape slide.

When the door is closed and locked, a door locked flag indicates green. Each door has a viewport to observe the external surroundings before opening the door.

With the door closed and locked, the door can be opened from the inside by rotating the internal handle. To open the passenger doors, the internal handle rotates in a counterclockwise direction. On the service doors the handle rotates clockwise. The initial movement of the door is inward and upward. When the door is lifted, the door swings outward and forward on the hinge arm, clear of the opening. When fully open, the door locks in that position and its weight is supported by the hinge arm.

To close the door, the hold open handle is pulled to unlock the door and then the door is pulled into the opening. The internal handle is rotated in the opposite direction to close, latch, and lock the door.

To open the door from the outside, the external handle is used. The handle flap is pushed in and the handle is grasped and lifted up. The internal mechanisms sequence the door operation in the same manner as using the internal handle. In addition, if the door mode handle is in the armed position, operating the external handle also returns the mode select handle to the disarmed position, ensuring the escape slide does not deploy. When the door is opened, a vent panel on the door opens to relieve any residual cabin pressure.

An aerodynamic seal, installed on the top of the door, covers the gap between the fuselage and the door.

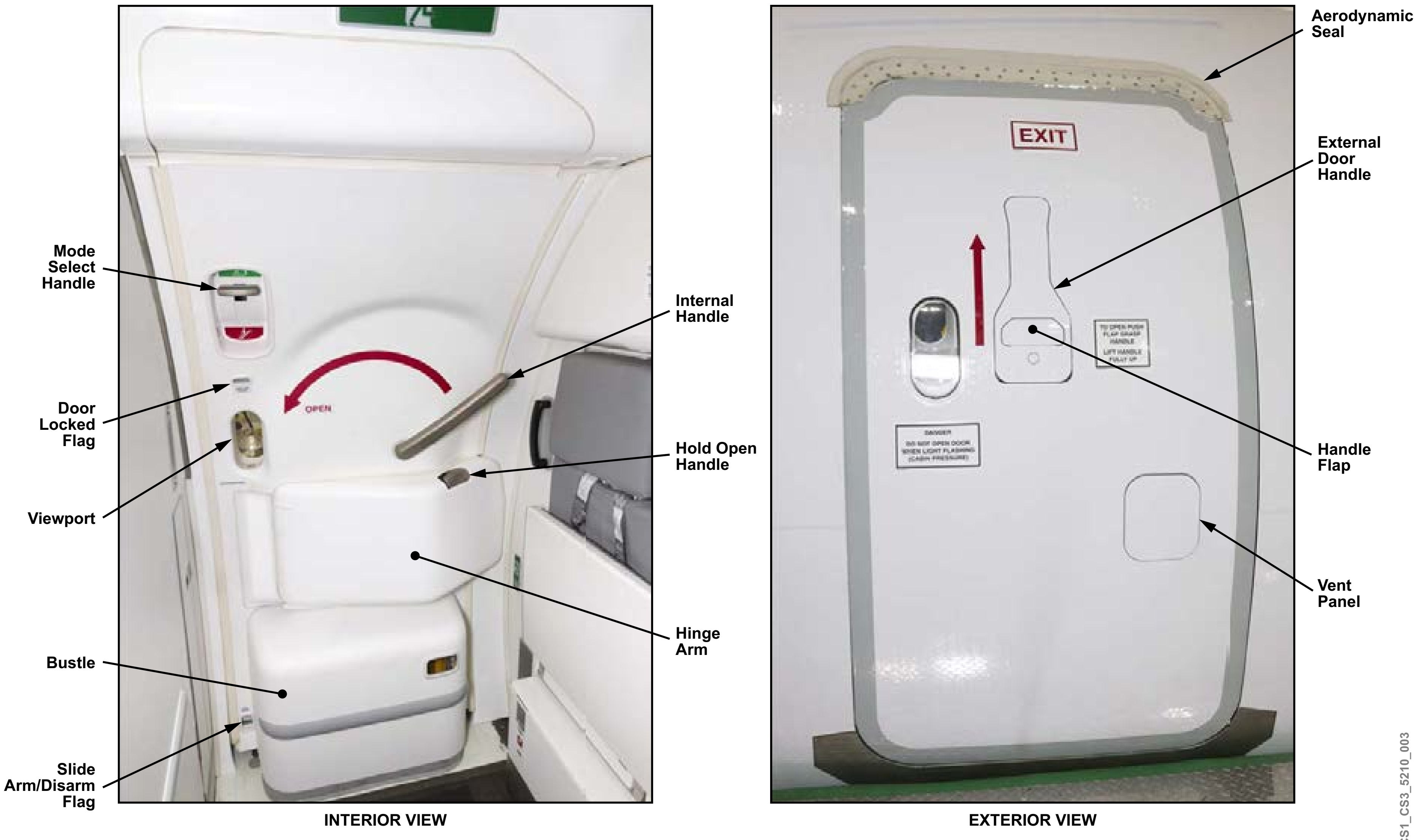


Figure 1: Passenger and Service Doors

STABILIZER BARS

Two stabilizer bars are connected between the top of the door and the fuselage structure to stabilize the door during opening and closing.

EMERGENCY OPENING ASSIST MEANS

The emergency opening assist means (EOAM) is mounted inside the door hinge of the passenger and service doors. The EOAM consists of a hydraulic actuator and a pneumatic reservoir. During normal door operation, the EOAM provides a hydraulic damping effect to limit the speed of the door and prevent aircraft structure damage. When the door is opened in an emergency, the pneumatic reservoir provides the energy to drive the door open rapidly.

DOOR MECHANISMS

The door mechanisms ensure a specific sequence is followed when opening, or closing the doors. The passenger and service door mechanisms include:

- Lock, latch, and lift mechanisms
- Hold open hook
- Door handles
- Mode select mechanism
- Door interlocks
- Vent panel mechanism

Lock, Latch, and Lift Mechanisms

The lock mechanism initiates the opening and closing of the door. Two locks, located on each side of the lock shaft, secure the latches in a latched condition when the door is closed.

The latch mechanism unlatches the door and provides the initial lift of the door. Two latches, located on each side of the latch shaft secure the door in the closed condition during flight.

When the door is unlocked and unlatched, it is lifted on the hinge via the lift mechanism. Weight compensation springs to help raise and lower the door.

Hold Open Hook

The hold open hook engages a pin on the fuselage structure when the door reaches the full open position.

Door Handles

The internal handle is directly connected to the lock and lift mechanisms. The door weight is partially compensated via the internal handle spring rod during lifting and lowering.

The external handle is connected to the lock shaft. The external handle is also connected to the mode select mechanism to disarm the door automatically should it be armed.

Mode Select Mechanism

The mode select mechanism connects to the girt bar mechanism in order to arm and disarm the slide. The mode select mechanism is also used to actuate the EOAM when the slide is armed.

Door Interlocks

Door interlocks prevent actuation of the internal door handle and mode select mechanisms to prevent the door mechanisms from going out of sequence.

Vent Panel Mechanism

The vent panel mechanism prevents the aircraft from pressurizing unless the door is properly latched, locked, and closed. The vent panel mechanism is spring-loaded to the open position.

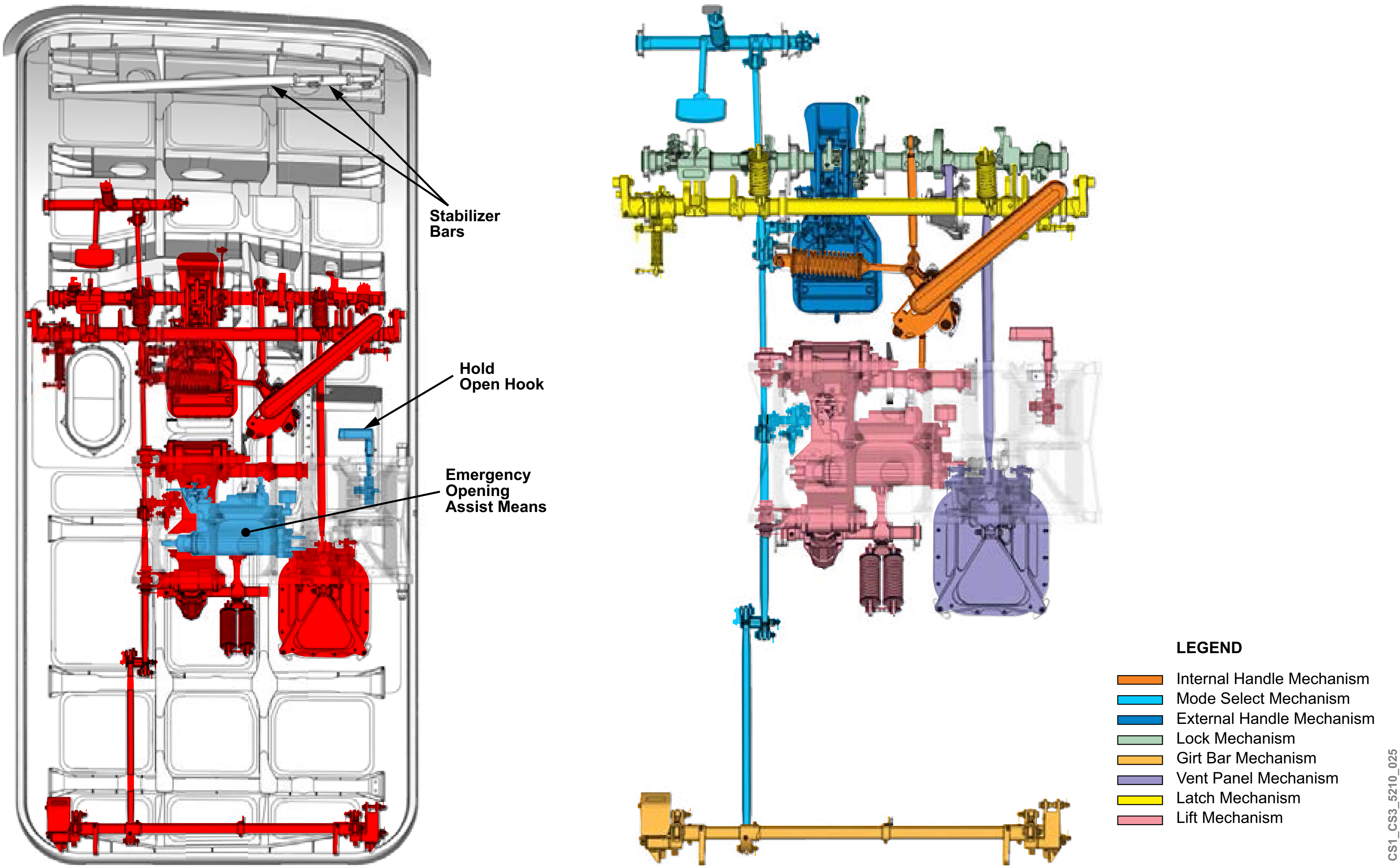


Figure 2: Internal Mechanisms

DOOR INDICATIONS

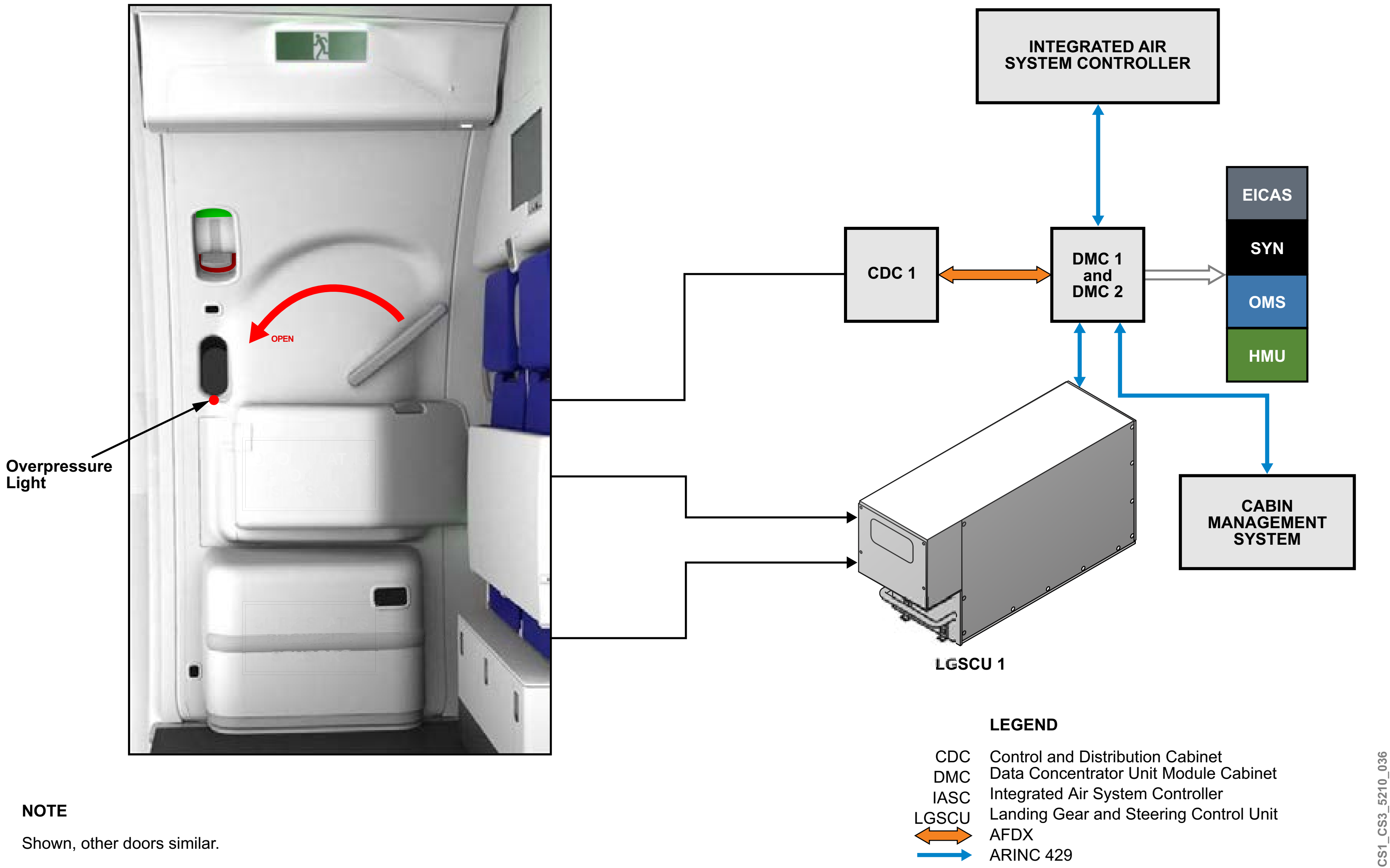
The passenger and service doors provide door status indications to the engine indication and crew alerting system (EICAS) and the cabin management system (CMS) through the landing gear and steering control unit (LGSCU).

Each door has three proximity sensors that monitor the door latching, locking, and closing functions. Additional proximity sensors on each door monitor the slide status.

The LGSCU reads the status of the passenger and service doors proximity switches and provides the data to the data concentrator unit module cabinets (DMCs). The DMCs display the doors status on the DOOR synoptic page and on EICAS.

The LGSCU also monitors the health status of the proximity sensors and supplies the information to the onboard maintenance system (OMS) and the health management unit (HMU).

A red overpressure light located in the viewport indicates a high differential pressure exists between the cabin and the outside of the aircraft. The light is turned on by the control and distribution cabinet (CDC) 1 when the integrated air system controller (IASC) 1 detects a high differential cabin pressure.



NOTE

Shown, other doors similar.

Figure 3: Door Indications

COMPONENT LOCATION

The following components are installed on the forward and aft passenger and service doors:

- Stabilizer bars
- Door seals
- Roller fittings
- Stop fittings
- Hold open handle
- Emergency opening assist means (EOAM)
- Mode select handle
- Hinge arm
- Girt bar mechanism

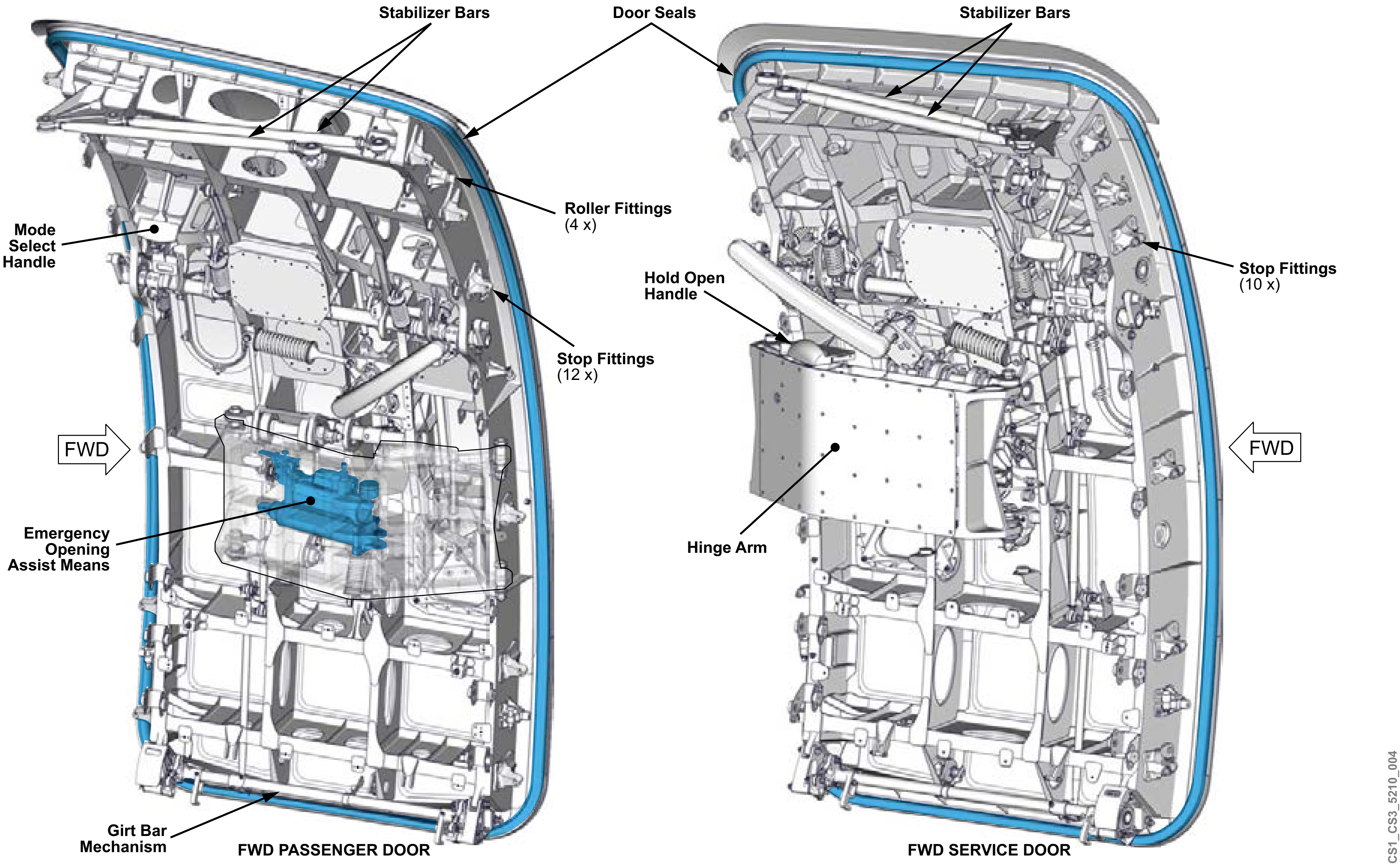


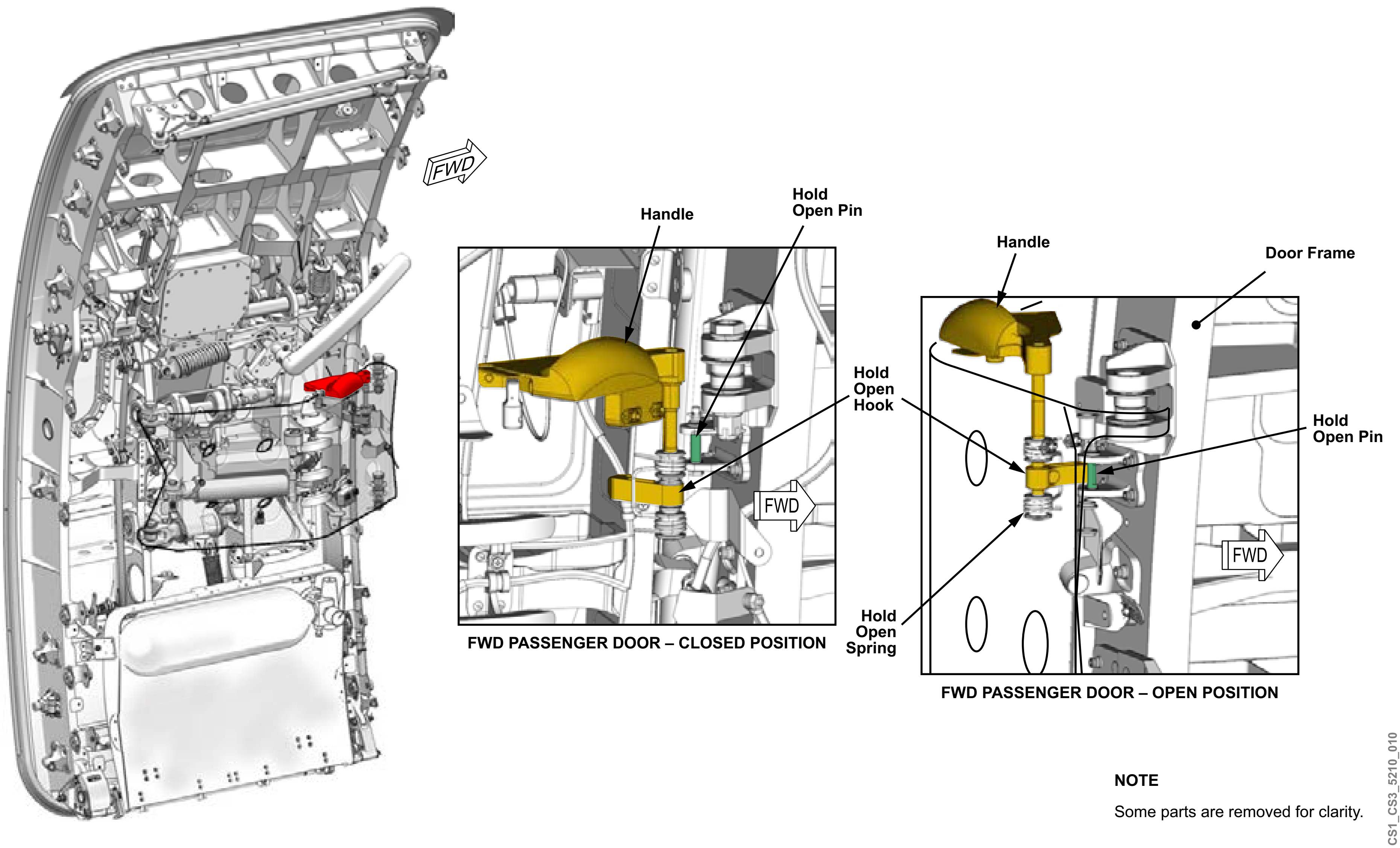
Figure 4: Passenger and Service Doors Component Location

COMPONENT INFORMATION

HOLD OPEN HOOK

The passenger and service doors are held in the open position by a hold open hook. The spring-loaded hold open hook engages the hold open pin installed on the fuselage door frame. The hook retains the door in the fully open position even in high gust loads.

When closing the door, the hold open hook is released from the pin by pulling the handle located on top of the hinge arm.



NOTE

Some parts are removed for clarity.

Figure 5: Hold Open Hook

EMERGENCY OPENING ASSIST MEANS

The main components of the EOAM are:

- Pneumatic assembly for powering the actuator during an emergency
- Hydraulic actuator for damping
- A mechanical percussion or trigger device used to release the nitrogen during an emergency
- A temperature-compensated pressure gauge is available to check the pressure in the pneumatic assembly

The trigger device has an actuating lever that interfaces with the door mode select mechanism when the door mode select handle is in the arm position. If the door is opened in an emergency, the lever is lifted and pushes down on a striker needle, which then perforates a burst disc. The nitrogen gas then powers the actuator to rapidly push the door open.

A filling valve allows servicing of the pneumatic assembly with nitrogen. The actuator is serviced to 2175 psi.

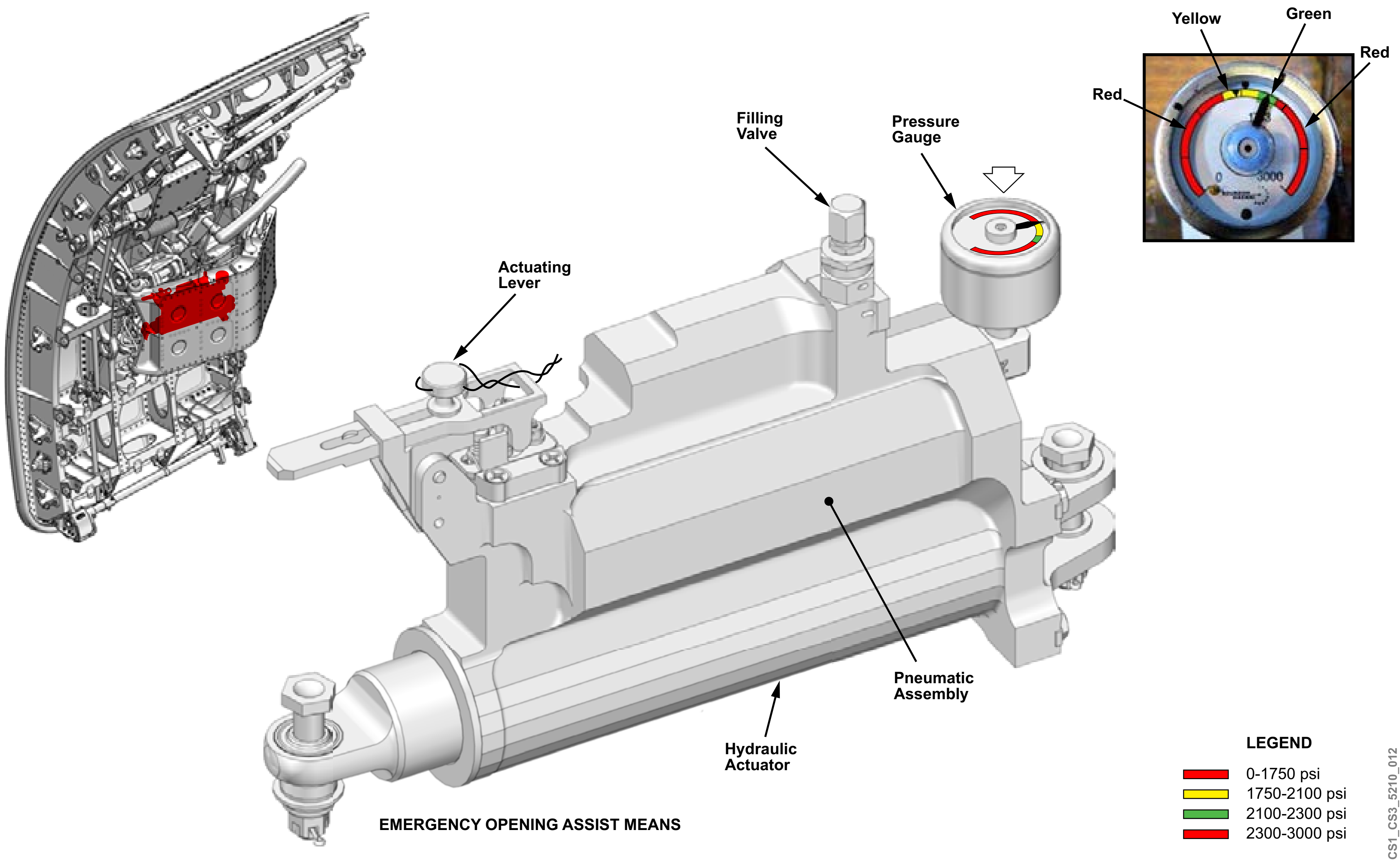


Figure 6: Emergency Opening Assist Means

DETAILED COMPONENT INFORMATION

LOCK MECHANISM

When the door is operated, the lock shaft is the primary mechanism for controlling and sequencing the door operation. The lock mechanism prevents any rotation of the latch shaft when it is latched. It is not possible to engage the locks in the locked position if the latches are not latched.

Operation of the door using the external handle is done through a clutch installed on the lock shaft. When using the internal handle, rotation of the internal handle shaft is transmitted to the lock shaft via a rod to lift crank located on the lock shaft.

The locks are disengaged at 30° of the internal handle rotation followed by the latch rotation and unlatching at 35°. The locks rotate and engage with a counterlock mounted on the latch shaft to prevent any rotation of either shaft in the locked or unlocked positions. The latching and unlatching of the door is performed through the lock system by the geneva wheel lever which engages the latch geneva wheel.

A visual indicator monitors the aft lock and shows if the door is latched and locked. It consists of an idler installed on the lock shaft that follows a cam located on the latch shaft. The position of the visual indicator idler is controlled simultaneously by the position of the lock shaft and latch shaft.

The forward lock is monitored by a sensor installed on the door structure. The lock sensor target moves with the lock shaft.

On the passenger and service doors a vent panel is used to prevent pressurization if the door is not latched and locked. The lock shaft vent panel cam controls the opening or closing, but does not close until the door is latched and locked.

The lock securing means spring prevents movement of the lock shaft once it is in the lock position. When the door is in the unlocked configuration, the lock securing means spring exerts a force on the lock shaft against its stop and maintains this position.

The mode select interlock consists of a cam that blocks the mode select mechanism once the door is in either the armed or disarmed configuration and lifted. The interlock prevents slide arming when the door is not in the closed position.

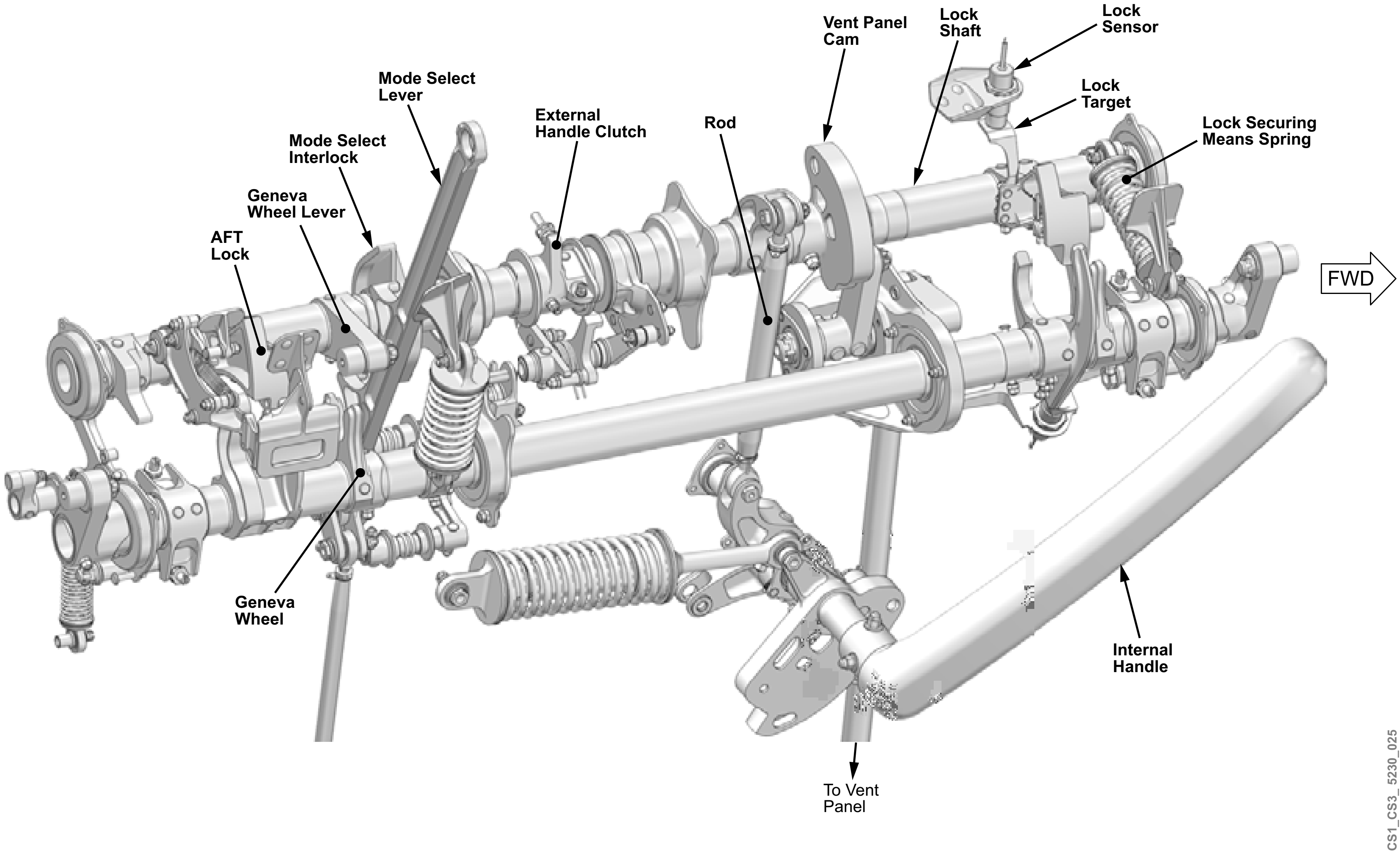


Figure 7: Lock Mechanism

LATCH MECHANISM

The purpose of the latch mechanism is to sequence the deployment of the latches during the opening and closing of the door.

Two latch cranks are located at each end of the latch shaft. When the latch shaft is rotated, the latch crank rollers are engaged into the latch guide fittings, mounted on the fuselage door surround. When the door is closed, the latch cranks prevent any upward movement of the door that would cause it to open.

When the door is opened, the contact between the latch crank roller and the lower surface of the guide fitting provides an initial lift of the door.

The rotation of the lock shaft geneva wheel allows the latch shaft to be unlocked, then unlatched and then locked in the unlatched position by the continuous rotation of the lock shaft in the opening sequence. The sequence is reversed for closing.

The rotation of the latch shaft is limited by an adjustable latch stop, located on each side of the latch shaft. The latch stop contacts a stop on the door structure at the end of the latch shaft travel.

Two latch securing means springs lock the shaft in the latched and unlatched position.

A latch sensor monitors the forward counterlock and is located on the latch shaft.

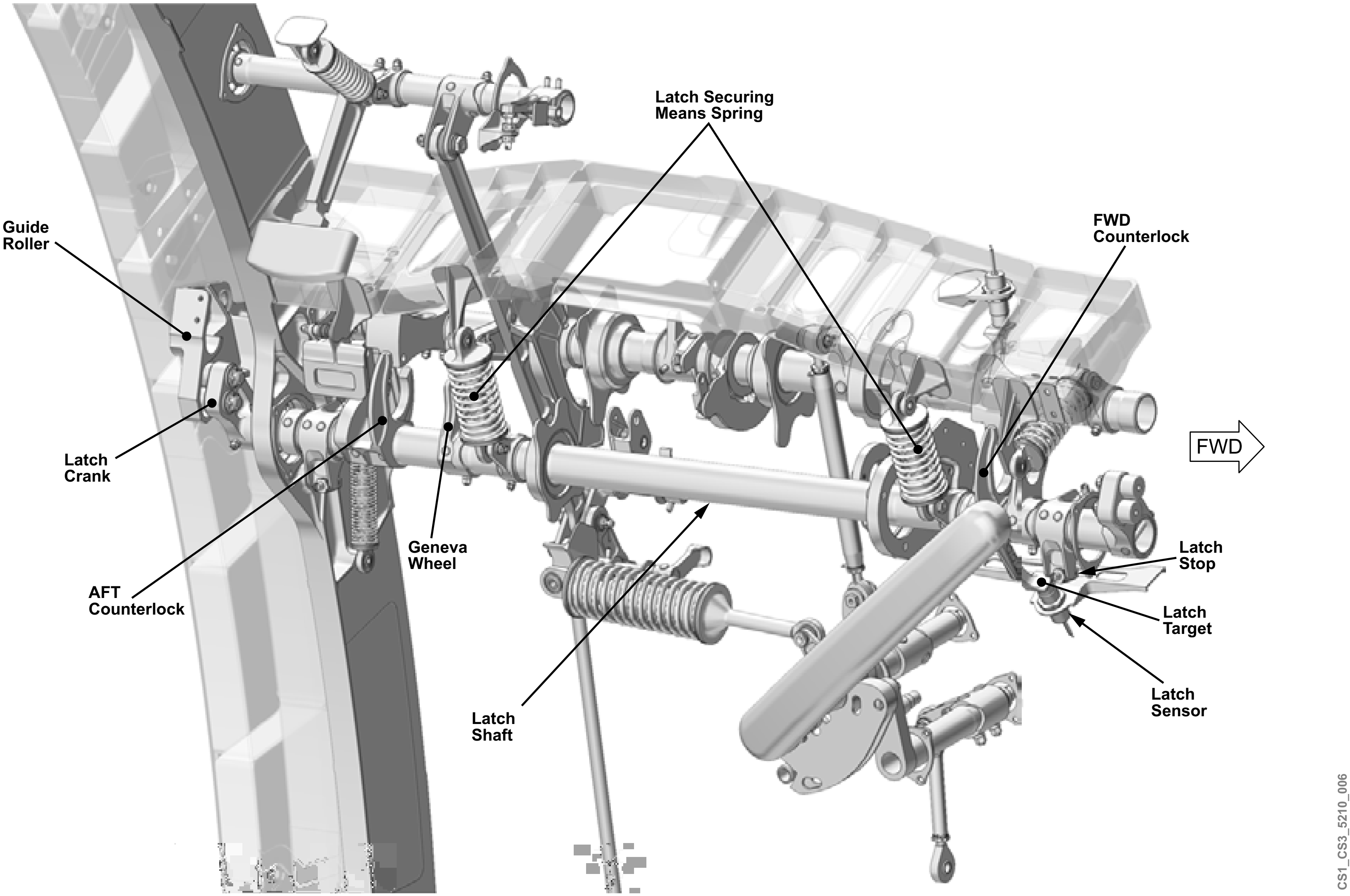


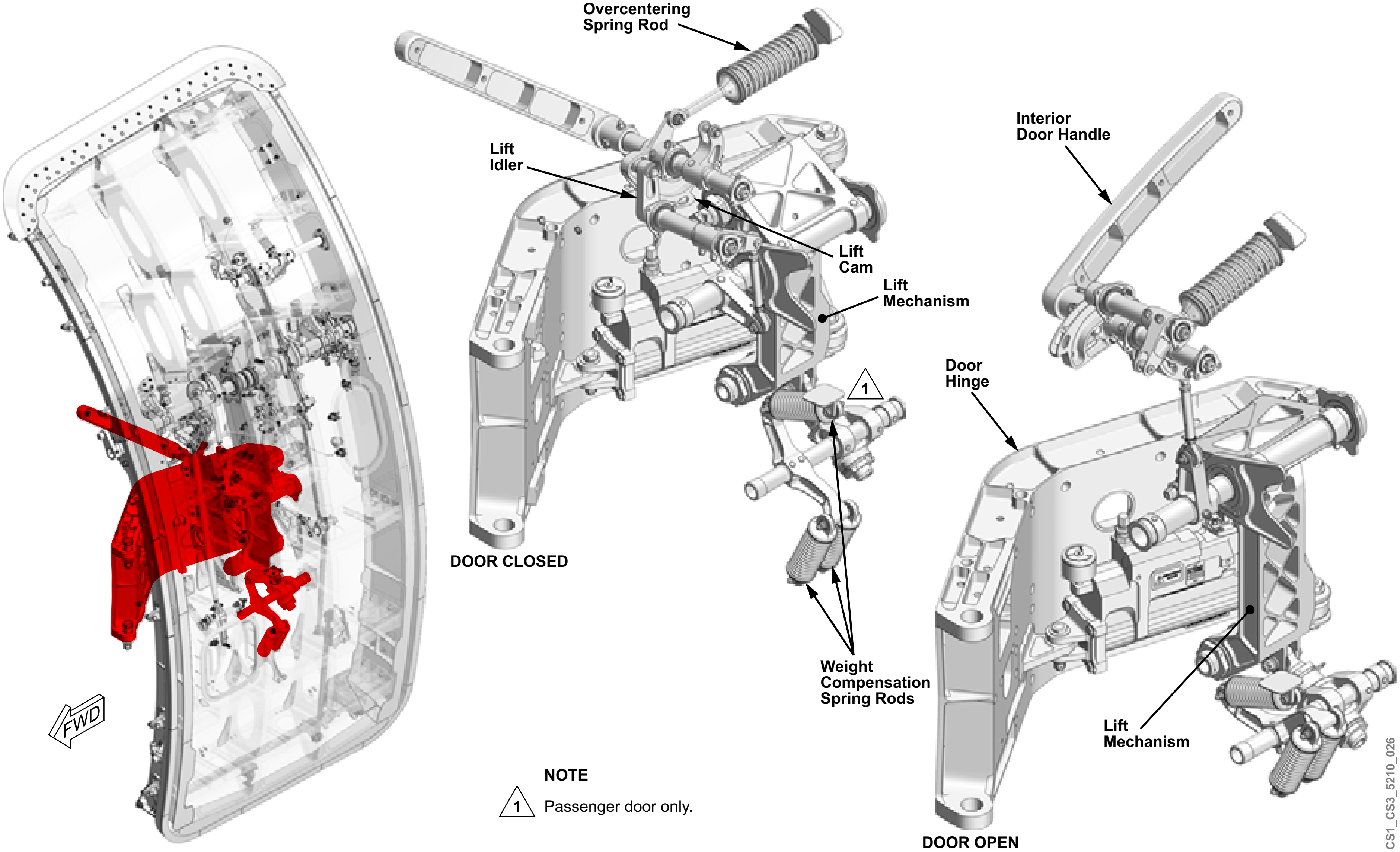
Figure 8: Latch Mechanism

LIFT MECHANISM

When the door is unlocked and unlatched, the door is lifted on the hinge arm by a lift mechanism. The lift mechanism is located between the door structure and the hinge arm. The lift mechanism lifts the door in order that the door stop pins clear the fuselage stop pads. The lifting path is defined by the guide fittings mounted on the door surround structure.

After the initial lift of approximately 0.2 in., generated by the latches reacting on the door surround latch fittings, the remaining lift is done by the lift mechanism reacting directly on the hinge arm. The lift cam sets the timing of the door lift and controls the rotation of the lift idler so that the movement is transmitted to the lift system.

The weight compensation spring rods are located near the vent panel at the lower side of the door. It consists in an assembly of three spring rods which lift the door by generating a torque around the lower link axis. On the service doors, only two springs are used because the service doors weigh less than the passenger doors.



NOTE

1 Passenger door only.

Figure 9: Lift Mechanism

EXTERNAL HANDLE

The external handle is connected to the lock system by a clutch installed on the lock shaft. The main role of the clutch is to allow separate use of the internal and external handle. When the external handle is used to operate the door, the internal handle is engaged and moves to the selected position.

When the internal handle is used to open the door, the external handle does not move. If the external handle was used to open the door and the door is closed using the internal handle, the external handle is engaged through the clutch and closes when the internal handle moves to the closed position.

If the mode select mechanism in the armed position when the external handle is used, the door is automatically disarmed. The disarming roller on the external handle contacts the disarming crank on the mode select mechanism. This moves the mode select handle to the disarm position.

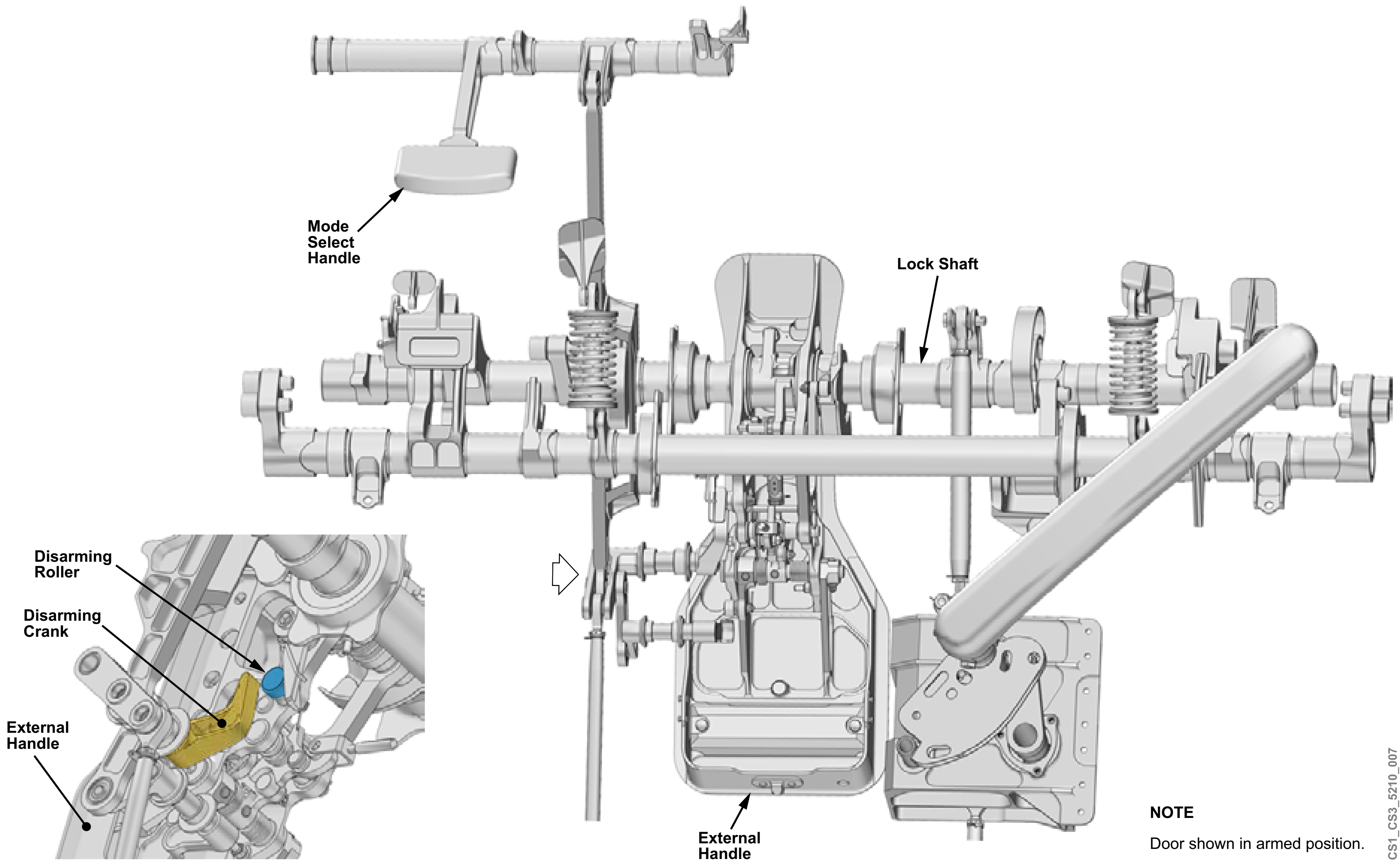


Figure 10: External Handle

MODE SELECT MECHANISM

The mode select mechanism consists of the mode select handle, push-pull rods, and the girt mechanism. An interlock prevents operation of the mode select mechanism unless the door is fully closed.

The mode select mechanism performs two functions:

- Arms and disarms the slide
- Actuates the emergency opening assist means (EOAM)

Slide Arming and Disarming

The mode select handle arms and disarms the slide through the girt mechanism. The girt mechanism connects to the mode select handle through push-pull rods. The girt mechanism has hooks that engage the girt bar. The girt bar is part of the slide, and sits in the floor fittings when the door is closed.

When the mode select handle is in the disarmed position, the grit bar hooks engage the girt bar and lift it clear of the floor fittings, allowing the door to be opened without deploying the slide.

When the mode select handle is moved to the armed position, the girt mechanism engages the girt bar in the floor fittings. The hooks move clear of the girt bar, allowing the slide to be pulled from the door and deploy when it is opened.

The girt bar mechanism has a slide status indicator that shows the girt bar armed or disarmed status. In the armed position the slide status is red, and in the disarmed position is green.

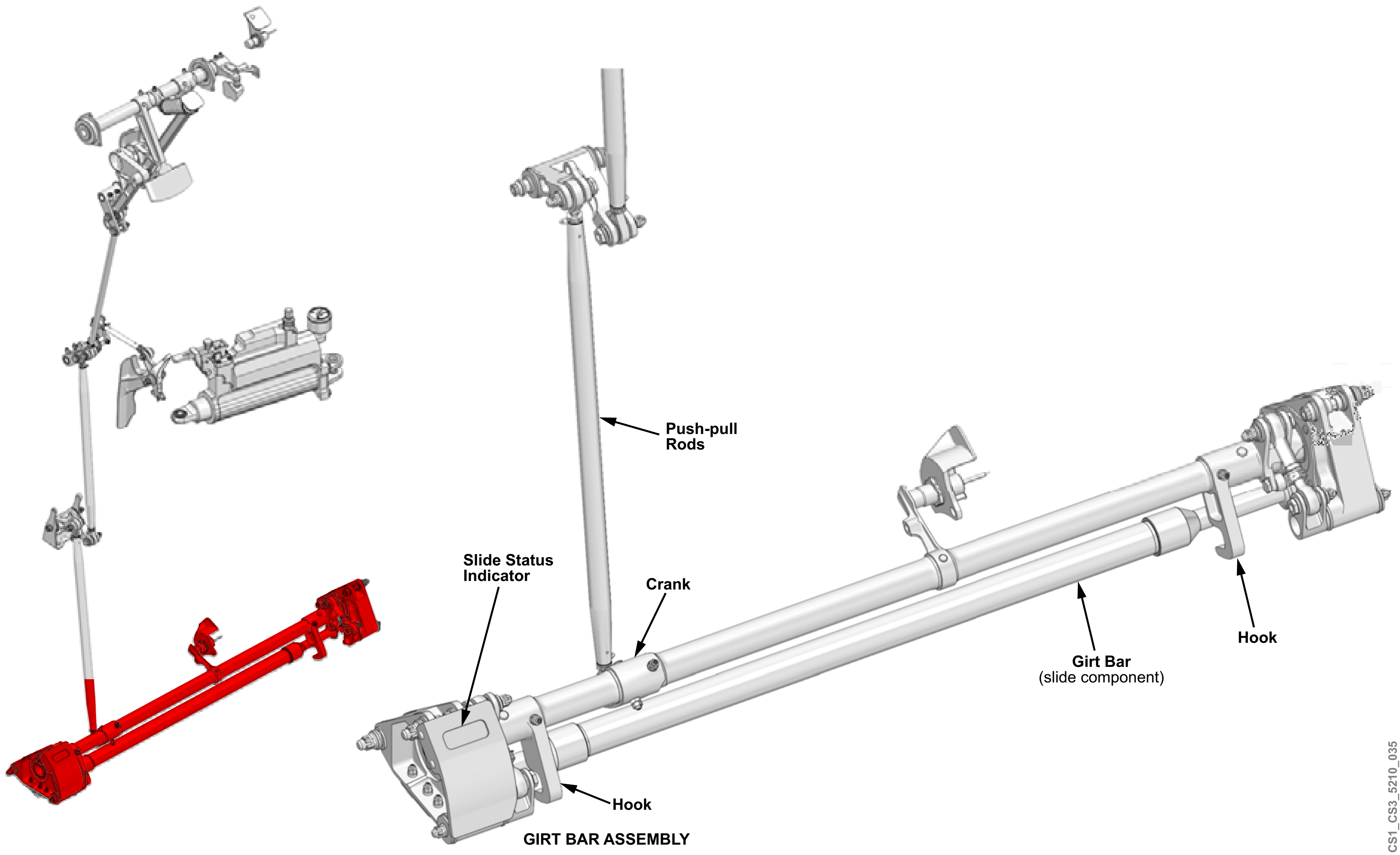


Figure 11: Mode Select Mechanism Slide Arming and Disarming

Emergency Opening Assist Means Actuation

With the mode select handle in the armed position, the mode select EOAM lever also moves to the armed position. When the lever is in the armed position, it contacts the EOAM actuating lever if the door is opened. The EOAM is triggered and assists in driving the door to the full open position.

When the door is unlocked, the mode select interlock on the lock shaft rotates to prevent the mode select handle from being operated.

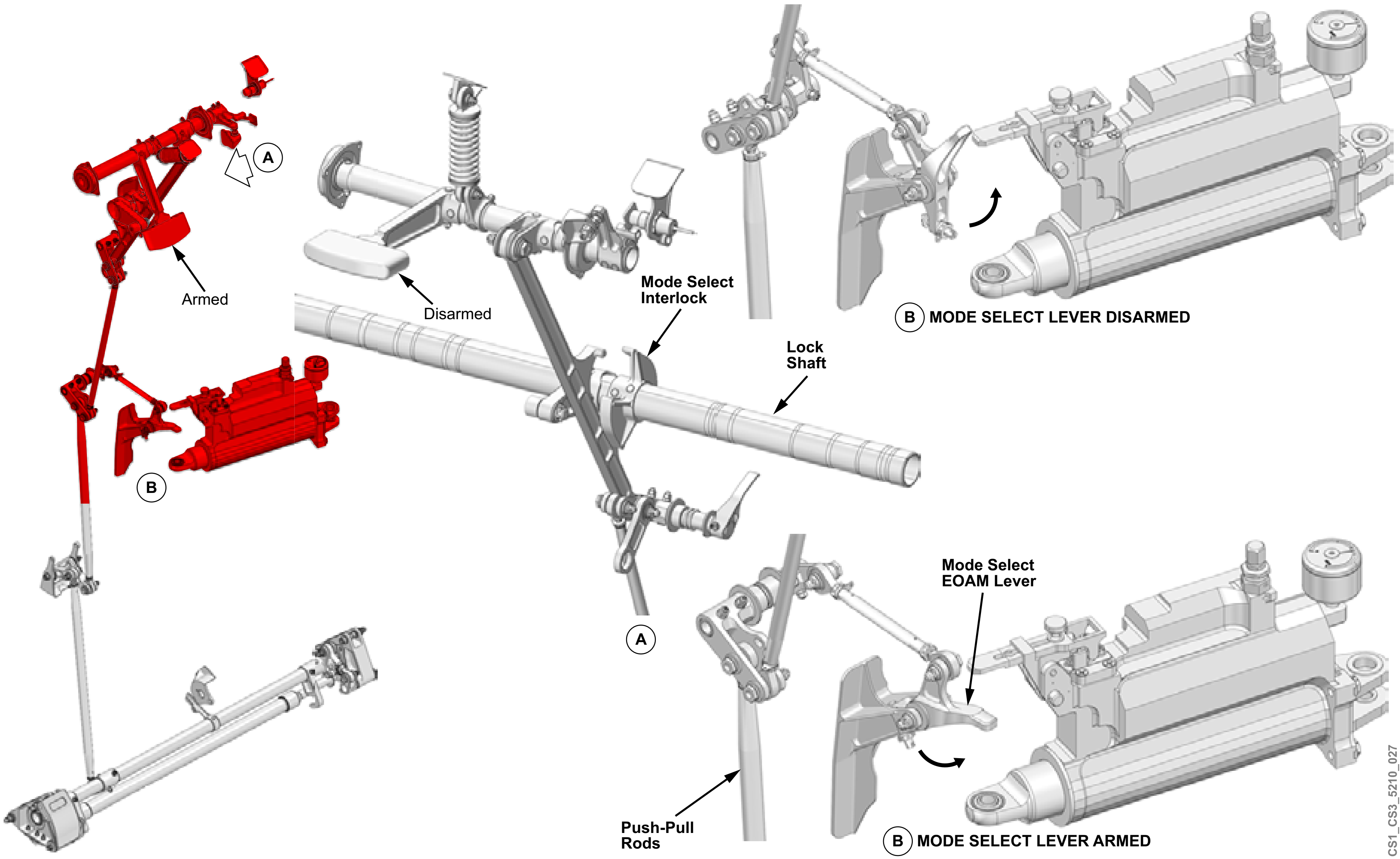


Figure 12: Mode Select Mechanism EOAM Actuation

VENT PANEL

The vent panel ensures that any pressure differential is equalized before a door is opened.

The vent panel mechanism consists of a vent panel and operating mechanism. It is connected to the door lock shaft and operates through a cam mounted on the lock shaft. The vent panel cam sequences the operation of the vent panel mechanism.

The vent panel is spring-loaded to the open position. During door closing, the rod pushes the vent panel to overcome the spring force and shut the panel. If the vent panel mechanism fails or does not fully close, the aircraft is not able to pressurize.

If the vent panel is frozen shut, the vent panel eccentric crank allows movement of the vent panel longitudinally to shear the ice.

The door can be opened on the ground, with a residual pressure of 0.125 psi or less. When the cabin pressure differential reaches 2 psi the door handle can not be moved due to forces acting on the vent panel cam. A vent panel seal helps maintain cabin pressurization.

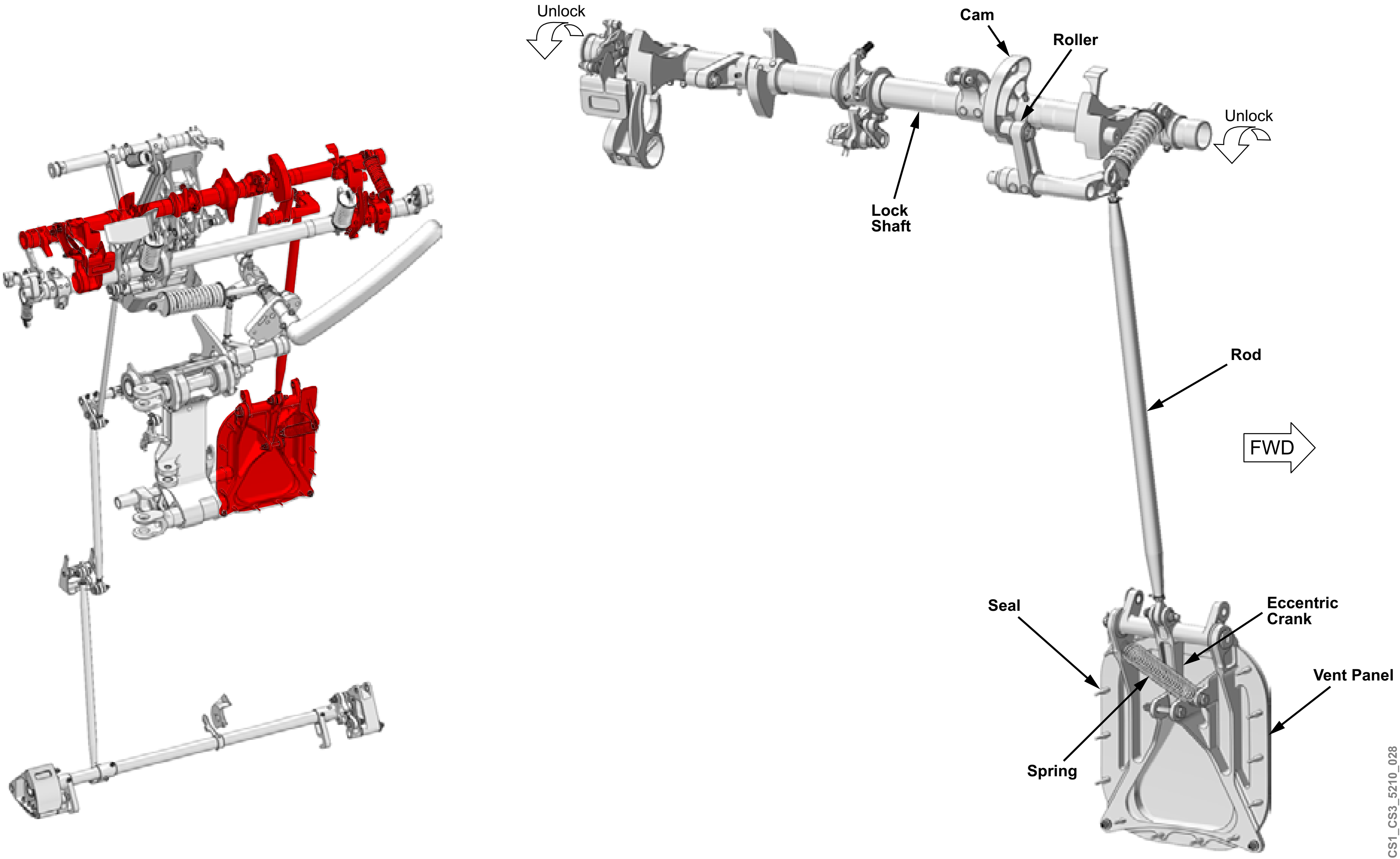


Figure 13: Vent Panel Mechanism

INTERLOCK MECHANISM

The interlock mechanism prevents door handle operation when the door is opened.

The interlock mechanism is installed on the aft side of the door, below the lock shaft. The interlock mechanism consists of an interlock roller, interlock lever, and interlock spring installed on a common shaft. It interacts with the interlock stop on the lock shaft. When engaged, it prevents the rotation of the lock mechanism and handle.

As the door opens, the interlock roller moves out of the fuselage roller guide. Once it is clear of the roller guide, the spring pushes up and rotates the interlock lever into the interlock stop, preventing movement of the lock shaft and handle.

When the door is closing the roller engages the interlock roller guide and forces the spring to compress and pull the interlock lever clear of the interlock stop. The door handle can now be used to close and lock the door.

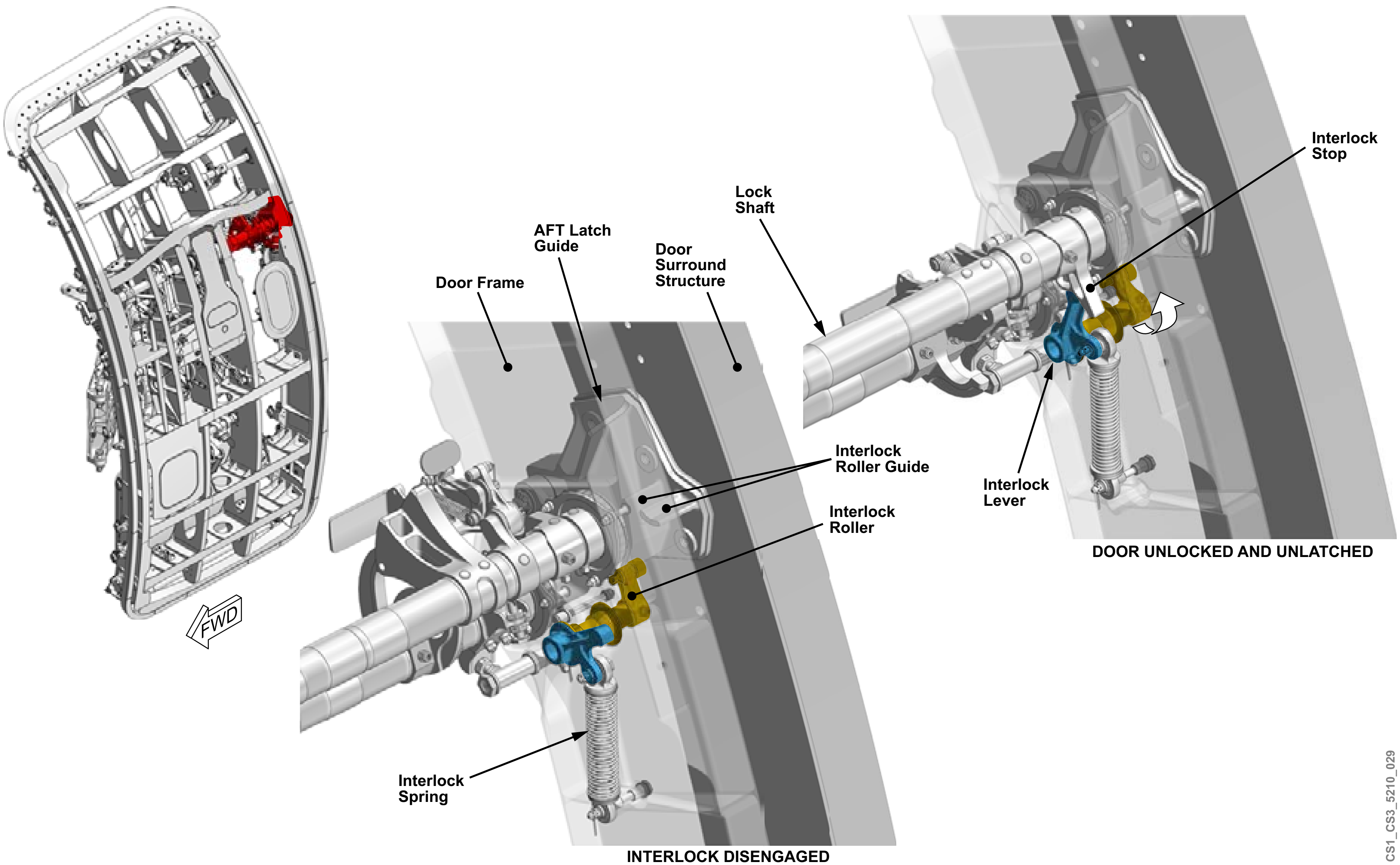


Figure 14: Interlock Mechanism

DOOR SEALS

The door seals ensure positive pressurization of the aircraft. Seals are installed around the door, and on the vent flap door. Each seal has inlet air holes for inflation. The pressure seals are spliced in multiple sections for maintenance, and are retained by sections of metal seal retainers.

STOP FITTINGS

The passenger doors are equipped with 12 stop fittings and the service doors are equipped with 10 stop fittings. When the door is closed they contact the fuselage stop pads. The stop fittings transfer the door pressurization loads to the aircraft structure. each stop fitting has an adjustable stop pin.

ROLLER FITTINGS

The rollers fittings guide the door during opening and closing.

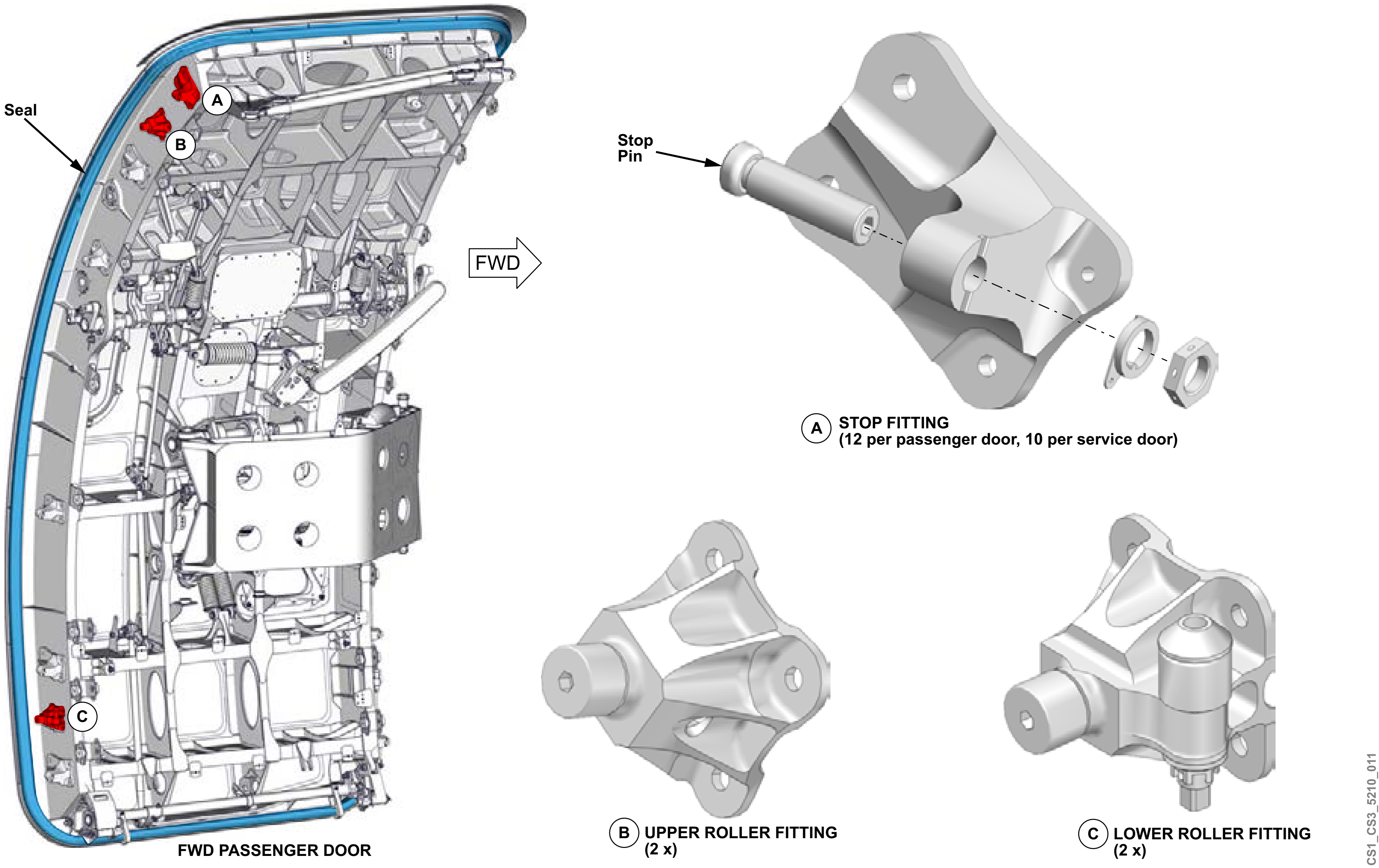


Figure 15: Door Seals, Stop Fitting, and Roller Fittings

DOOR FRAME STRUCTURE

Fuselage Stop Fittings

The passenger and service doors surround structure carries pressurization loads transmitted from the door stop fittings through the fuselage stop fittings. Each fitting has a replaceable stop pad.

There are 12 fuselage stop fittings on each passenger door, and 10 stop fittings on each service door.

Guide Fittings and Door Centering

Four fuselage guide fittings are installed on the door structure. Two are located on the upper structure, and two are located on the lower structure. The guide fittings provide the path the door follows as it is opened or closed. The lower guide fittings also center the door during opening and closing.

Two latch guides provide a path for the latch cranks and provide an overlatch condition, ensuring the latch shaft is secured in the latched position.

Bridge Assembly

The door hinge arm is attached to the fuselage structure through the bridge assembly. The bridge assembly increases the opening angle for the hinge arm.

Safety Strap

A safety strap is installed in a bracket assembly on one side of the door frame. The safety strap can be extended across to a hook on the opposite side door frame.

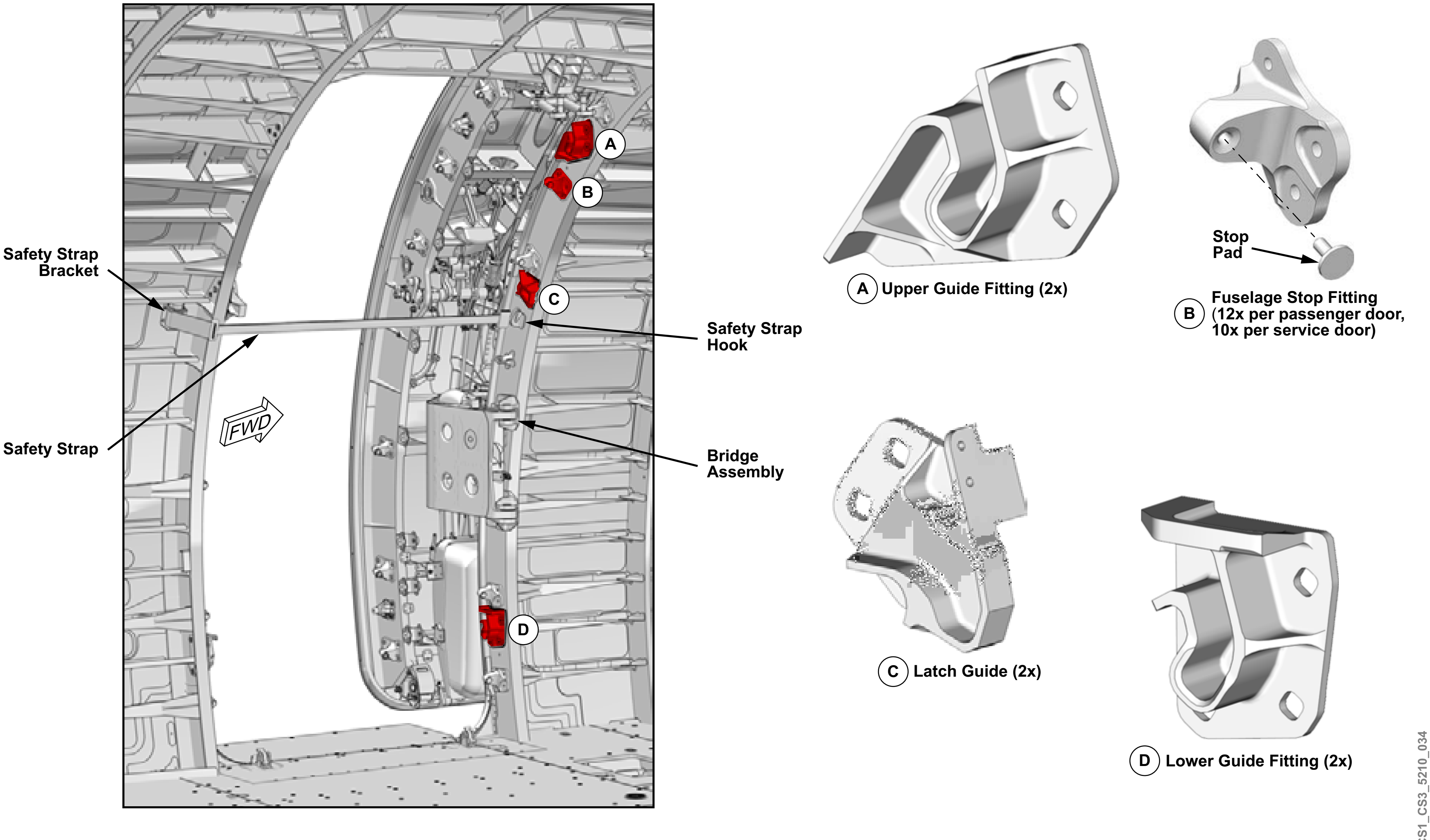


Figure 16: Door Frame Structure

CONTROLS AND INDICATIONS

DOOR INDICATIONS

Door Locked Status indicator

The door locked status indicator is green if the door is latched and locked. When the door is unlocked and unlatched, the indicator is red.

Overpressure Light

The red overpressure light is located in the viewport. When a high cabin differential pressure exists, the light turns on.

Slide Status Indicator

The slide status indicator is green if the slide is disarmed and red if the slide is armed.

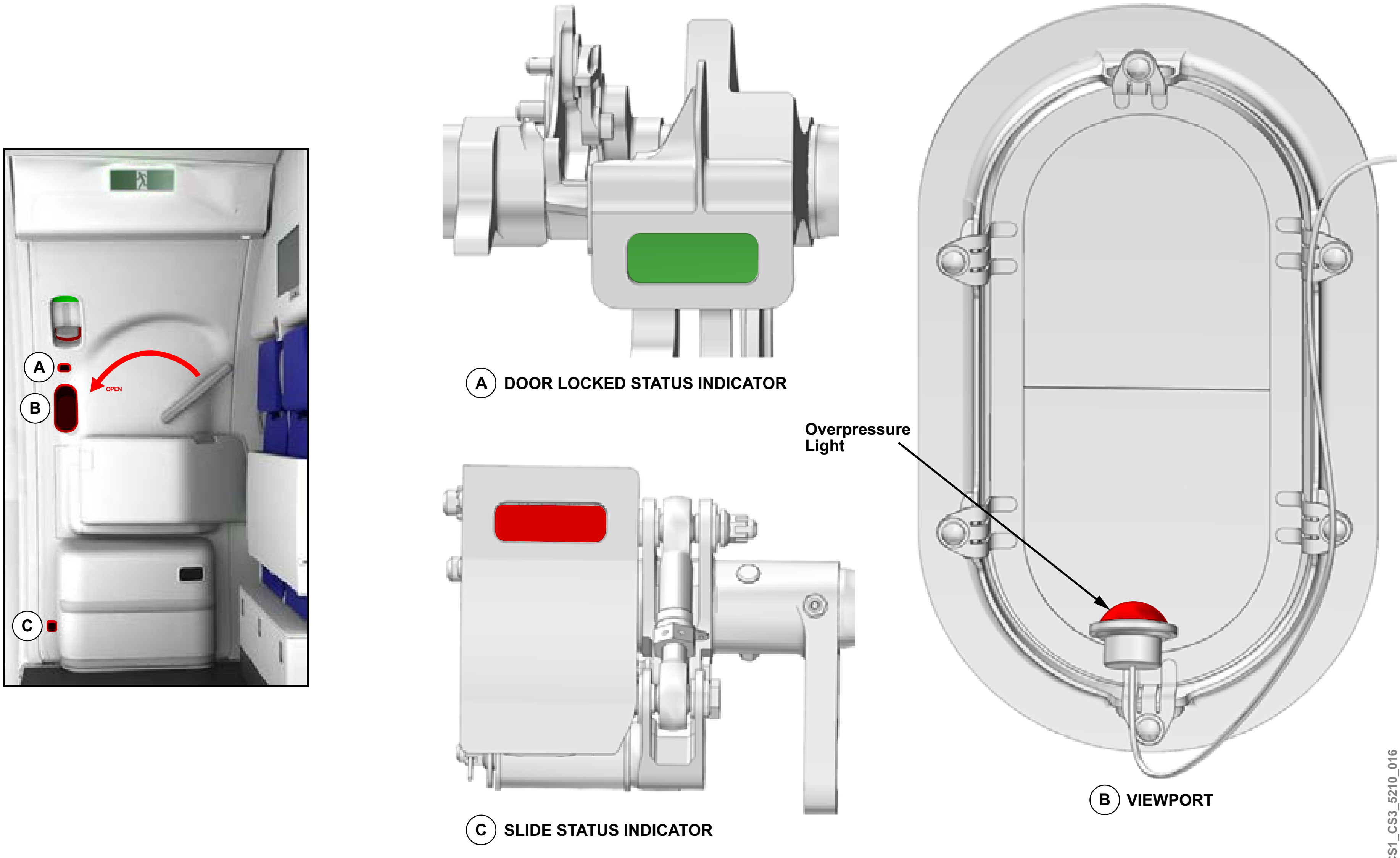
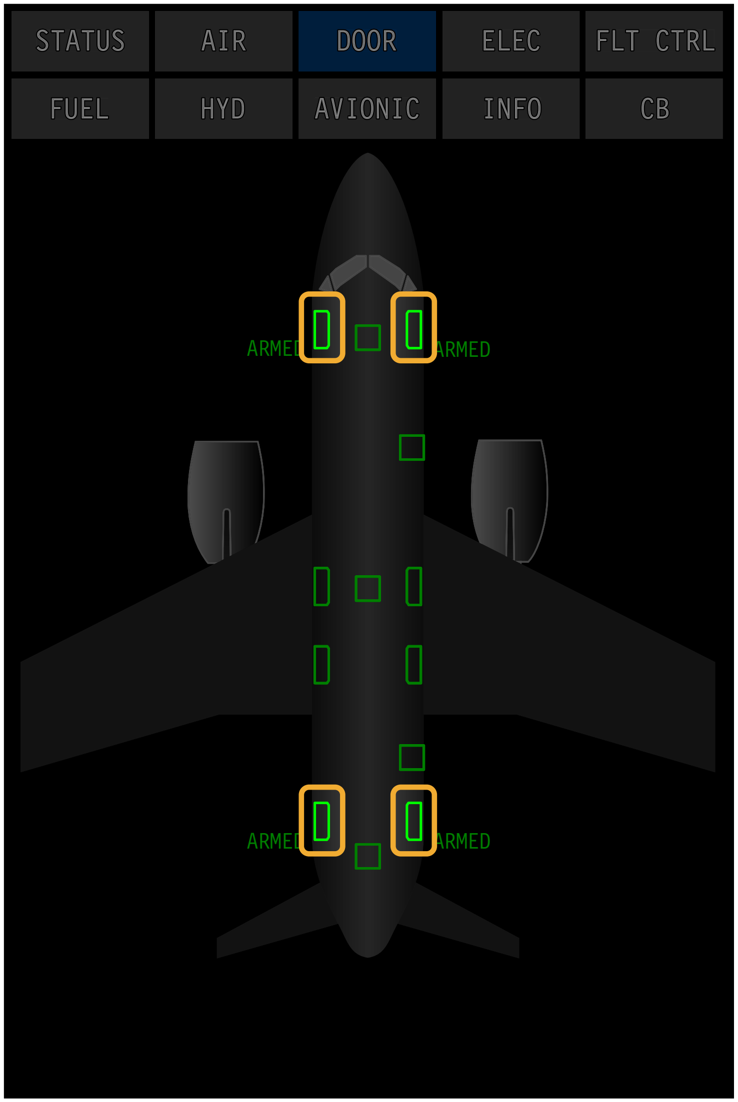


Figure 17: Passenger and Service Doors Controls and Indications

SYNOPTIC PAGE

The status of the passenger and service doors is shown on the DOOR synoptic page.



SYNOPTIC PAGE – DOOR

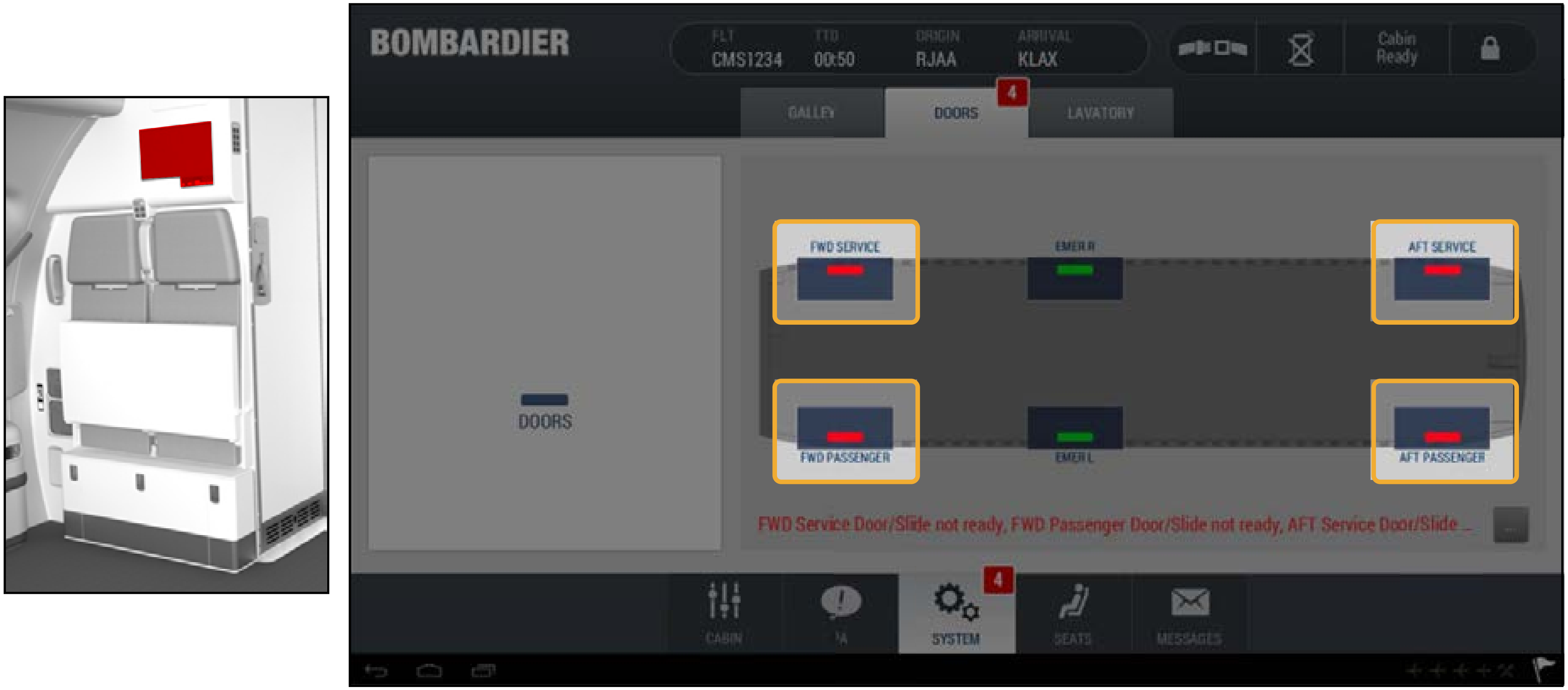
DOOR STATUS	
Symbol	**Condition**
FWD PAX	At least one engine running and the door is not closed/latched/locked. **OR** Door fault. Note: Label of door is shown in amber.
	Door closed/latched/locked. Note: Label of door is not shown.
FWD PAX	Door open on ground, engine is not running. Note: Label of door is shown in white.
FWD PAX	Invalid or failed LGSCU or invalid door sensor. Note: Label of door is shown in amber.

CS1_CS3_5210_017

Figure 18: Synoptic Page

CABIN MANAGEMENT SYSTEM CREW TERMINAL

The DOORS screen on the cabin management system (CMS) displays the status of all doors, overwing emergency exits on the aircraft.



CS1_CS3_5210_032

Figure 19: Cabin Management System Crew Terminal

OPERATION

OPENING THE PASSENGER OR SERVICE DOOR FROM THE OUTSIDE

Open the passenger or service door from outside as follows:

1. Push the flap in and grasp the external handle.
2. Pull the external handle up to lift the door. Ensure vent panel opens.
3. Pull the door outward and forward.
4. Make sure that the hold open mechanism locks the door in the open position.

CAUTION

The viewport overpressure indicator turns on when the cabin differential pressure is high. Operating the door may cause the door to open quickly, causing injury to personnel outside the aircraft.

NOTE

Opening the door from the outside automatically disengages the escape slide mechanism.

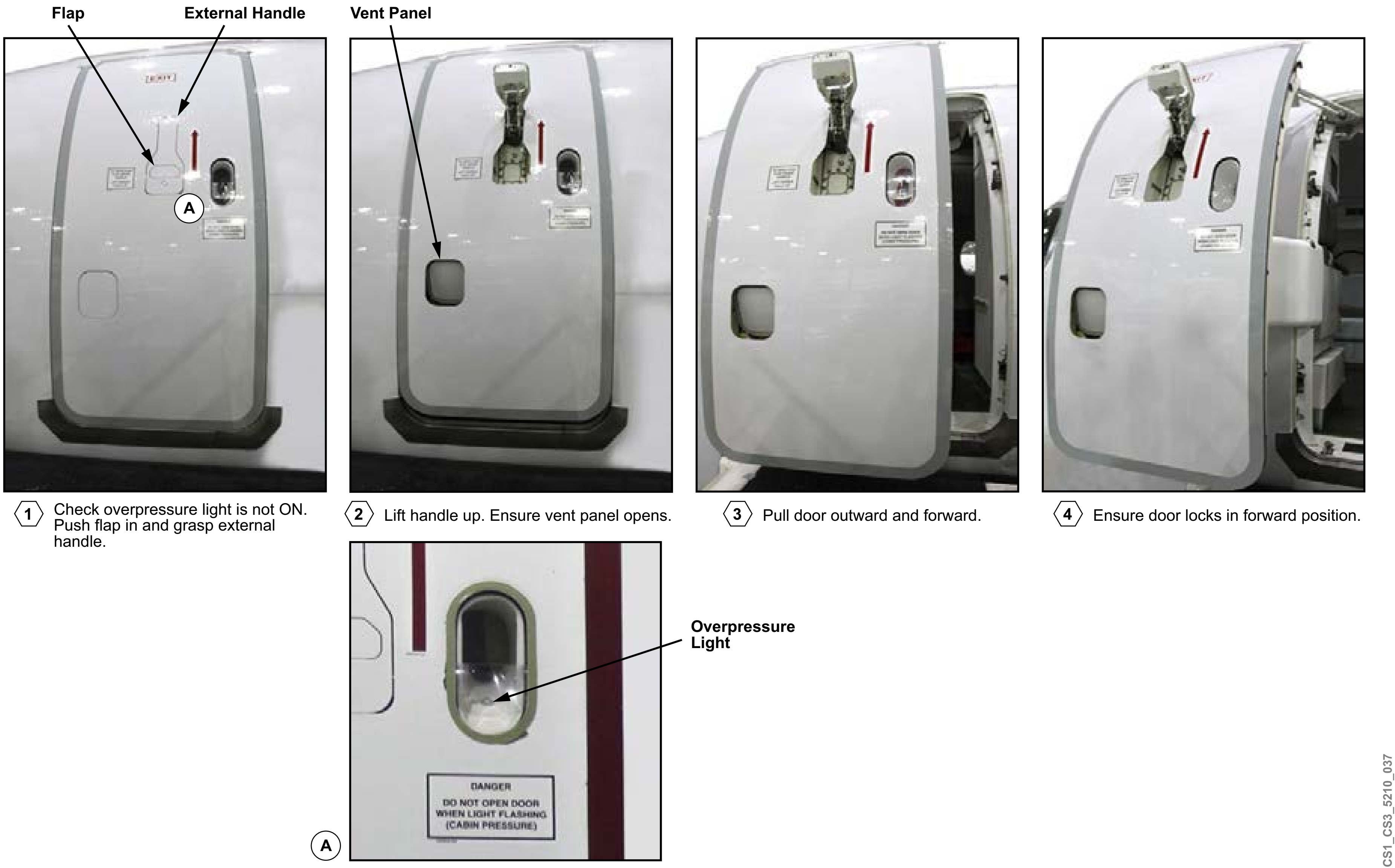


Figure 20: Opening the Passenger and Service Door from the Outside

CLOSING THE PASSENGER OR SERVICE DOOR FROM THE OUTSIDE

Close the passenger or service door from outside as follows:

1. Pull the hold open handle to disengage the door from the fuselage.
2. Pull the door rearward and push into the door frame.
3. Pull the external handle down.
4. Make sure the door is fully closed and flush with the fuselage.
5. Make sure the external handle is in the stowed position and flush with the door.
6. Make sure the vent flap is flush with the door.

Figure 21: Closing the Passenger and Service Door from the Outside

OPENING THE PASSENGER OR SERVICE DOOR FROM THE INSIDE

Open the passenger or service from inside as follows:

1. Push the mode select handle up to disarm the door.
2. Make sure the slide status indicator is green.
3. Turn the internal handle counterclockwise for the passenger doors or clockwise for the service doors.
4. Make sure that the door is in the fully up position and the door status indicator is red.
5. Push the door outward and forward.
6. Make sure that the hold open mechanism locks the door in the open position.

WARNING

BEFORE OPENING THE PASSENGER OR SERVICE DOOR, MAKE SURE THAT THE DOOR IS DISARMED. FAILURE TO DO SO ACTIVATES THE EMERGENCY OPENING ASSISTING MEANS (EOAM) AND DEPLOYS THE ESCAPE SLIDE. THIS CAN CAUSE INJURY TO PERSONNEL AND DAMAGE TO EQUIPMENT.

CAUTION

The viewport overpressure light turns on when the cabin differential pressure is high. Operating the door may cause the door to open quickly, causing injury to personnel outside the aircraft.

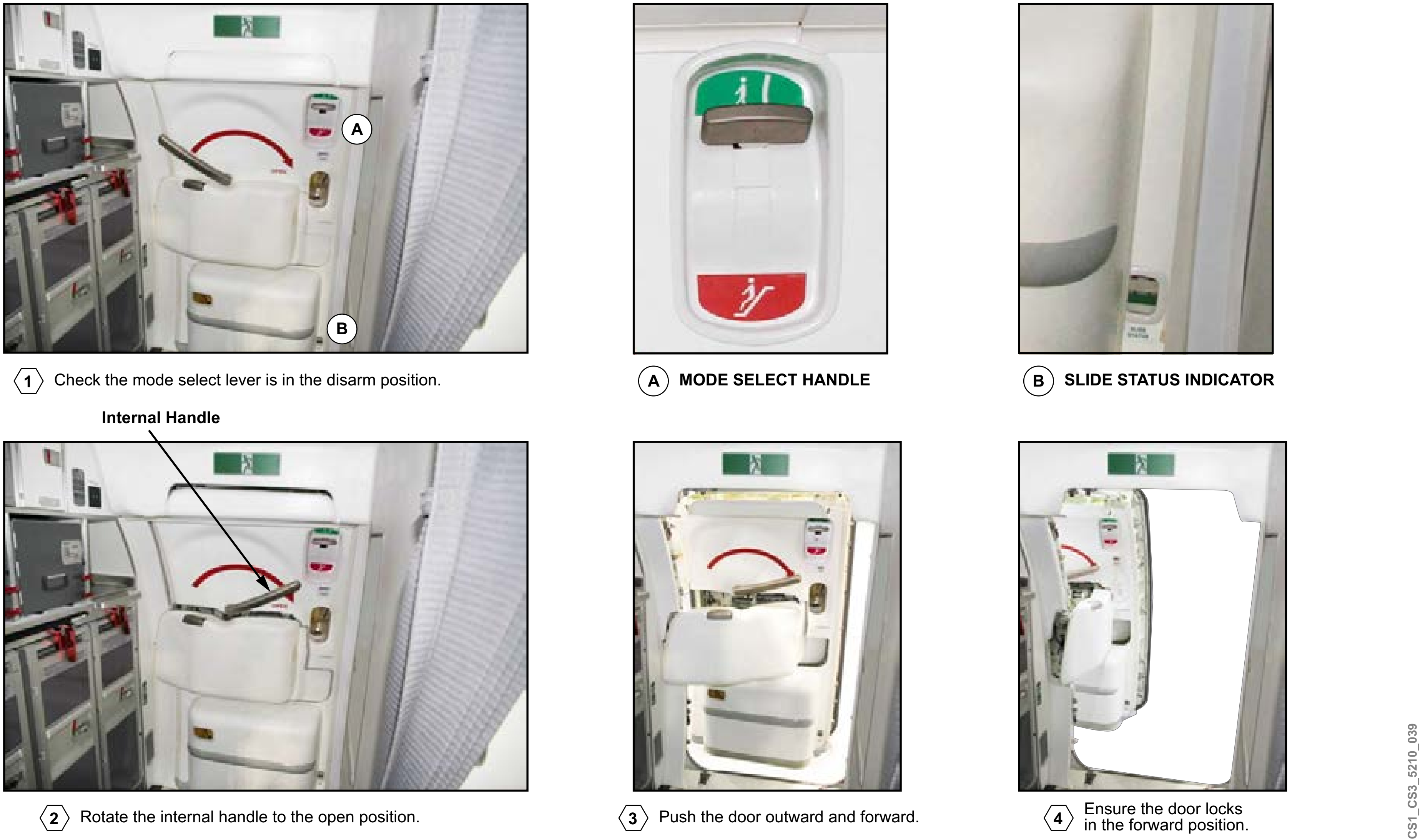


Figure 22: Opening the Passenger and Service Door from the Inside

CLOSING THE PASSENGER OR SERVICE DOOR FROM THE INSIDE

Close the passenger or service from inside as follows:

1. Pull the hold open handle up to disengage the door from the fuselage.
2. Pull back the door rearward and into the door frame.
3. Turn the internal handle clockwise for the passenger doors and counterclockwise for the service doors.
4. Make sure that the door is fully closed and the door lock status indicator is green.

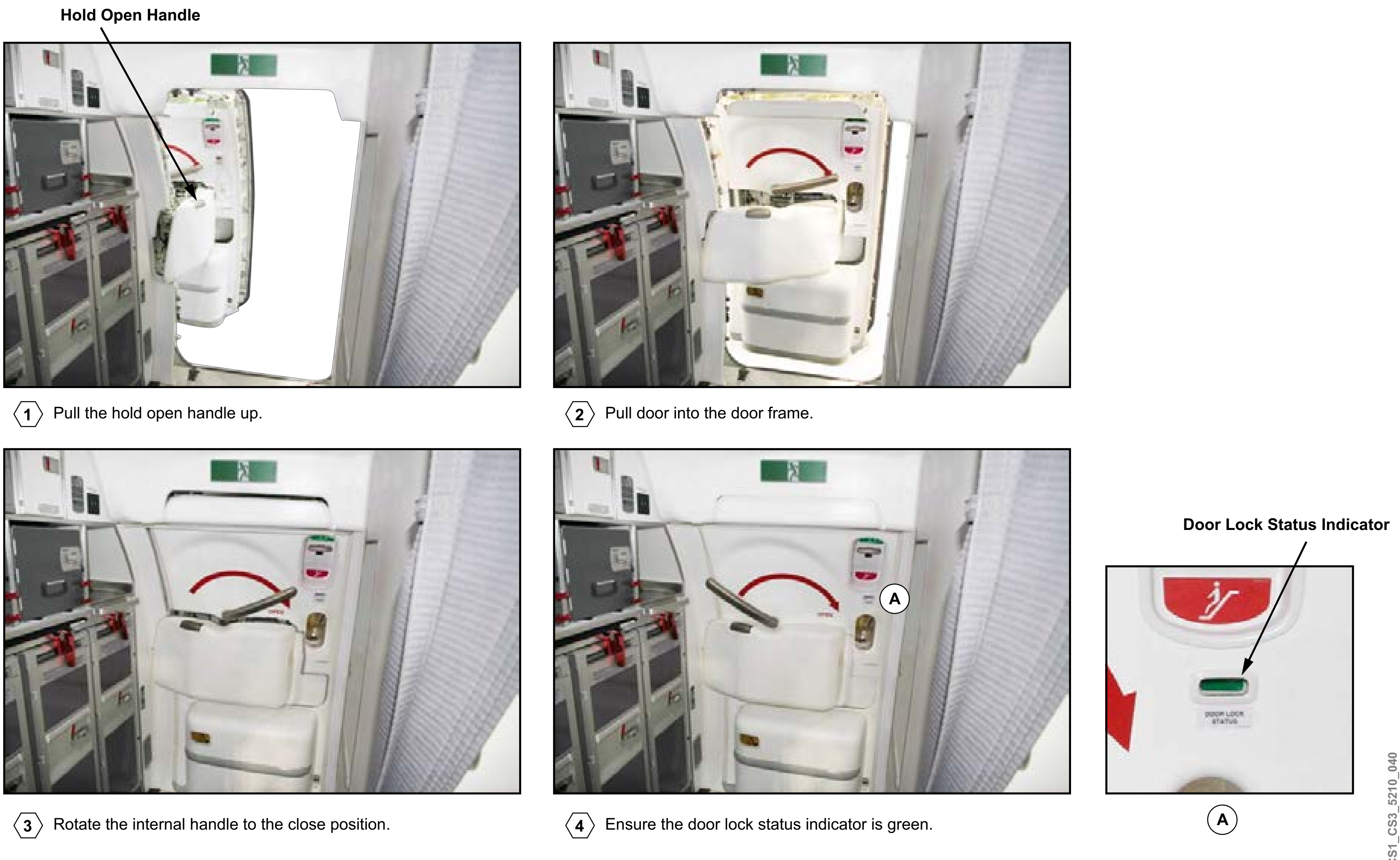


Figure 23: Closing the Passenger and Service Door from the Inside

DETAILED DESCRIPTION

CLOSED, LATCH, AND LOCK PROXIMITY SENSORS

Closed Sensor

The door closed sensor is installed in the door cut-out and the sensor target is installed on the door structure. When the door is closed the sensor target is in proximity with the sensor. When the door lifts, the door closed sensor target moves out of proximity from the door closed sensor.

Latch Sensor

The latch shaft is equipped with a latch proximity sensor monitoring the forward counterlock located on the latch shaft.

Lock Sensor

While the aft lock is monitored directly by the visual indicator, the monitoring of the forward lock is made by a lock sensor installed on the door structure. The sensor target is installed directly on the lock and moves with the lock shaft.

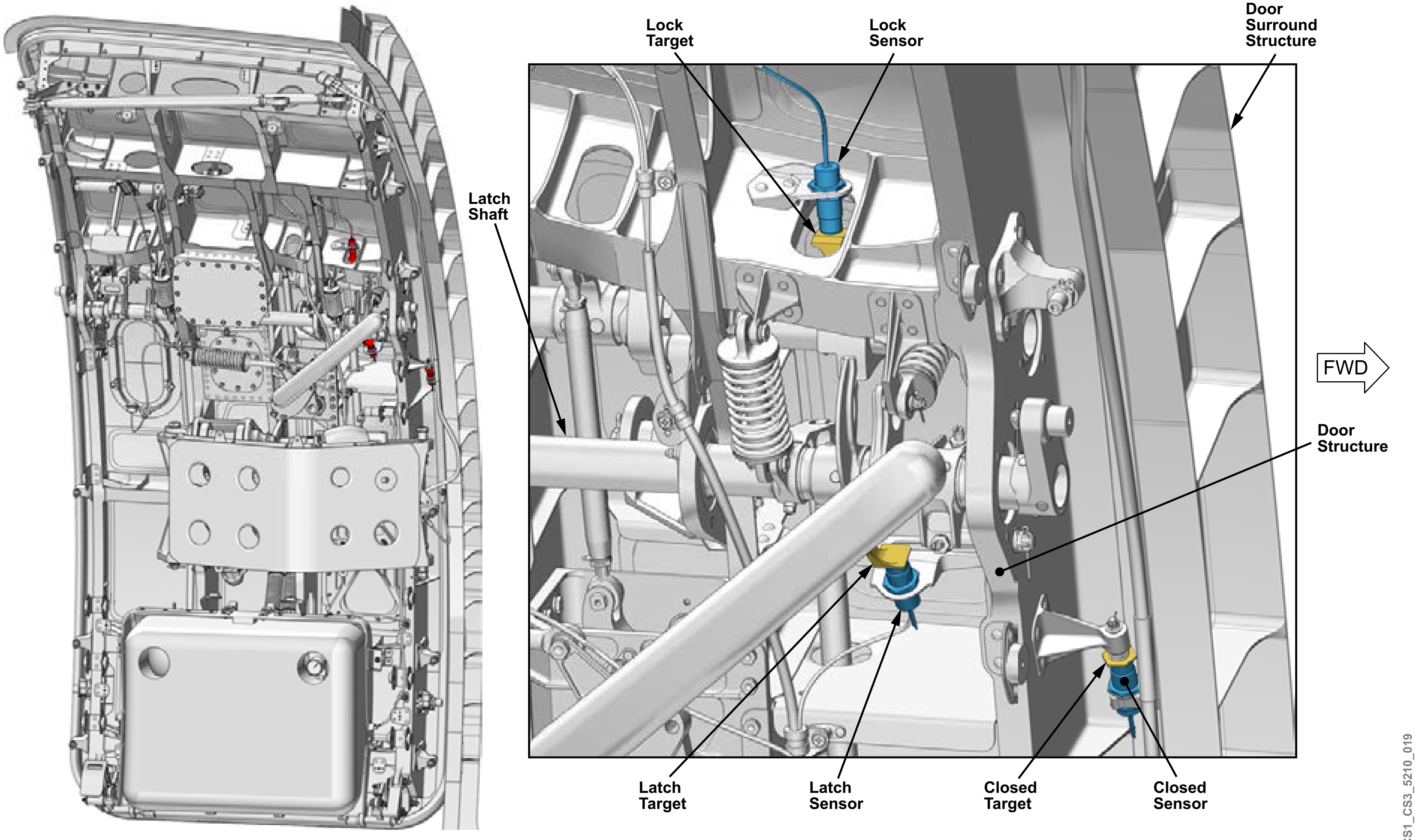


Figure 24: Closed, Latch, and Lock Proximity Sensors

GIRT BAR AND MODE SELECT PROXIMITY SENSORS

Girt Bar Proximity Sensor

The girt bar mechanism is equipped with a target that actuates a girt bar proximity sensor located on the door. This provides an arm or disarm signal to EICAS.

When the mode select handle is in the armed position, the girt bar target is in proximity to the girt bar proximity sensor. A rig pin is used to adjust the target with respect to the sensor.

Mode Select Proximity Sensor

The mode select mechanism is equipped with a target that actuates a mode select proximity sensor located on the door. This provides an arm or disarm signal to EICAS.

The mode select target is located directly on the mode select shaft together with the mode select handle. A rig pin is used in order to adjust the target with respect to the sensor.

All of the proximity sensors are adjustable. The proximity sensors have the same gap adjustment of 0.7620 - 1.2700 cm (0.30 in. - 0.50 in.).

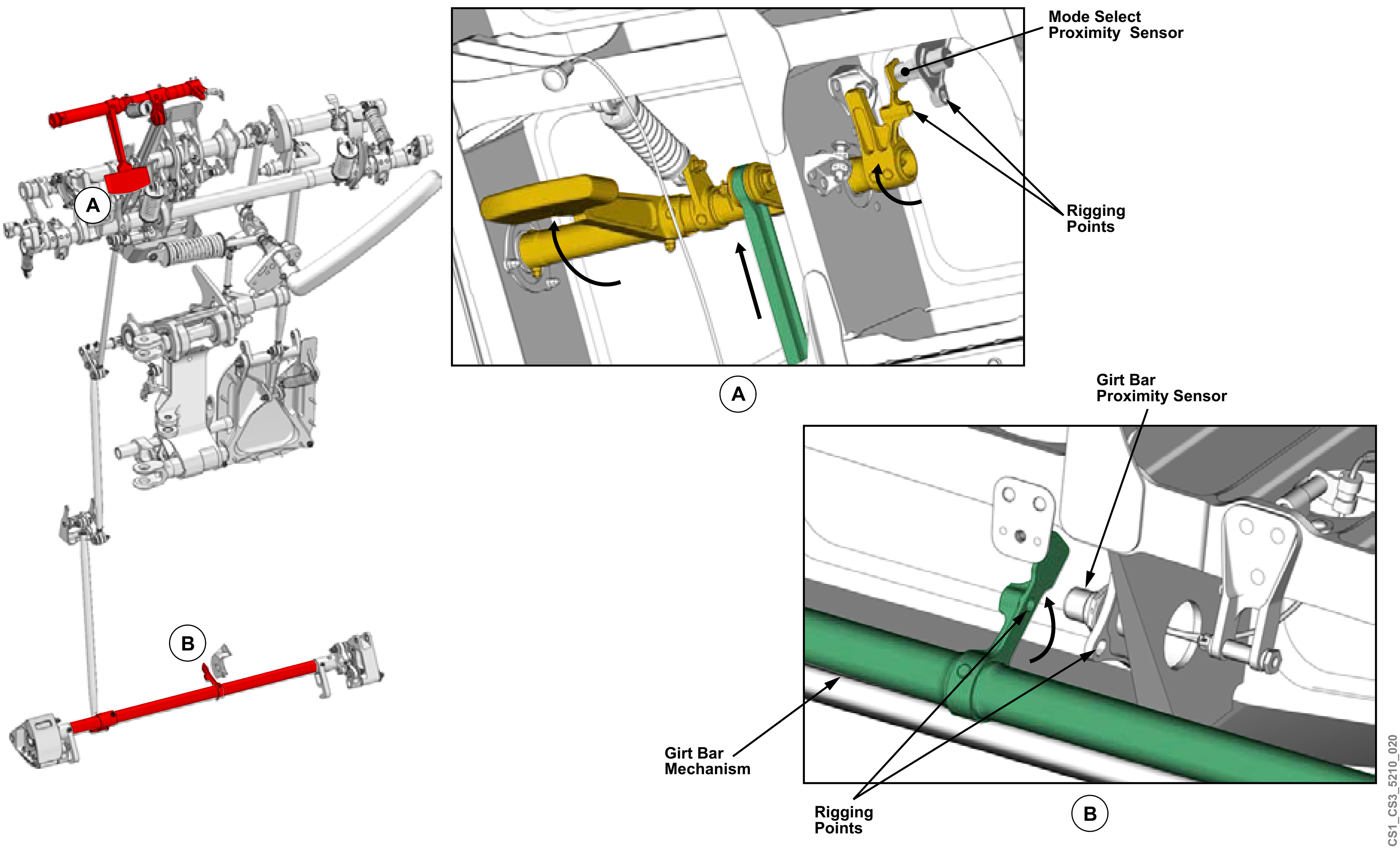


Figure 25: Girt Bar and Mode Select Proximity Sensors

PASSENGER AND SERVICE DOORS ELECTRICAL SCHEMATIC

The door proximity sensors indicate the door position and slide status. The latch, lock, and closed proximity sensors monitor the door. The armed and disarmed proximity sensors monitor the slide status. The forward passenger and service doors are monitored by landing gear and steering control unit (LGSCU) 1. The aft passenger and service doors are monitored by LGSCU 2.

The door status is provided to the DOOR synoptic page and cabin management system (CMS) DOORS screen.

If a door sensor or target fails with the aircraft on the ground and no engines running, a DOOR FAULT advisory message is displayed on EICAS.

If the engines are running, a FWD DOOR or AFT DOOR caution message is displayed when any door is open or has a fault.

A pressure differential visual indicator, located on the viewport and visible from inside and outside the aircraft, indicates a differential pressure exists between the cabin and the exterior of the aircraft. The light turns on when there is a pressure imbalance.

The cabin pressure differential is provided by the integrated air system controller (IASC). If this pressure has exceeded 0.125 psi after a rejected takeoff or 0.3 psi on ground the light flashes on all the passenger and service doors.

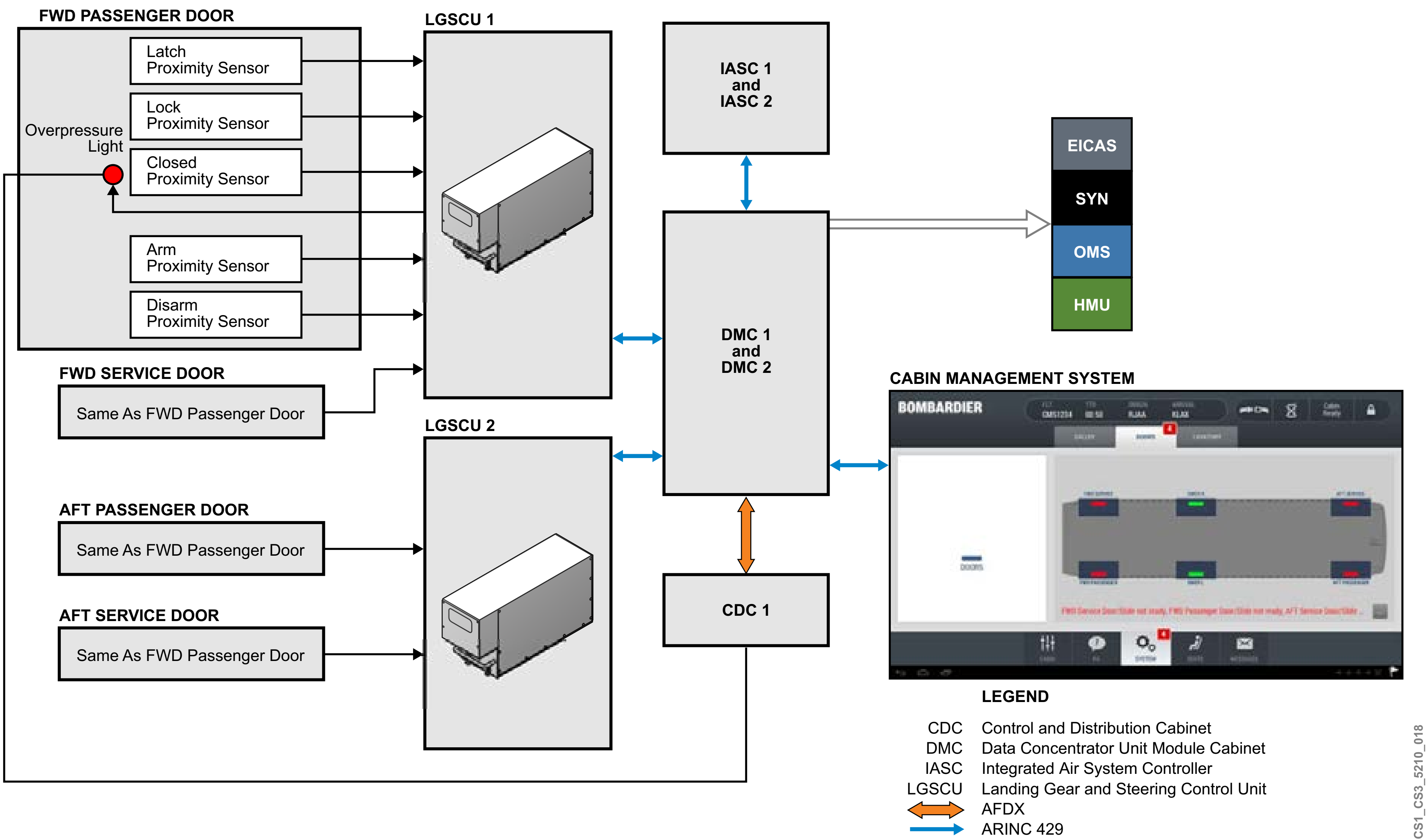


Figure 26: Passenger and Service Doors Electrical Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the passenger and service doors.

CAS MESSAGES

Table 1: CAUTION Messages

MESSAGE	LOGIC
FWD DOOR	FWD passenger or service door failed or open with at least one engine running.
AFT DOOR	AFT passenger or service door failed or open with at least one engine running.

Table 2: ADVISORY Messages

MESSAGE	LOGIC
DOOR OPEN	Any door open while on ground and no engine is running.
DOOR FAULT	Any electrical or mechanical failures of any door while on ground and no engine is running. Refer to INFO messages.

Table 3: INFO Messages

MESSAGE	LOGIC
52 DOOR FAULT - FWD PAX DOOR SNSR INOP	Electrical problem has been detected with at least one (up to three) of the FWD passenger door proximity sensors.
52 DOOR FAULT - AFT PAX DOOR SNSR INOP	Electrical problem has been detected with at least one (up to three) of the AFT passenger door proximity sensors.
52 DOOR FAULT - FWD SERV DOOR SNSR INOP	Electrical problem has been detected with at least one (up to three) of the FWD service door proximity sensors.
52 DOOR FAULT - AFT SERV DOOR SNSR INOP	Electrical problem has been detected with at least one (up to three) of the AFT service door proximity sensors.

Table 3: INFO Messages

MESSAGE	LOGIC
52 DOOR FAULT - FWD PAX DOOR TRGT INOP	Mechanical problem (target FAR) has been detected with at least one (up to three) of the FWD passenger door proximity sensors.
52 DOOR FAULT - AFT PAX DOOR TRGT INOP	Mechanical problem (target FAR) has been detected with at least one (up to three) of the AFT passenger door proximity sensors.
52 DOOR FAULT - FWD SERV DOOR TRGT INOP	Mechanical problem (target FAR) has been detected with at least one (up to three) of the FWD service door proximity sensors.
52 DOOR FAULT - AFT SERV DOOR TRGT INOP	Mechanical problem (target FAR) has been detected with at least one (up to three) of the AFT service door proximity sensors.

PRACTICAL ASPECTS

EMERGENCY OPENING ASSIST MEANS SAFETYING PROCEDURE

To prevent inadvertent actuation during transportation or maintenance, the emergency opening assist means (EOAM) must be deactivated. To deactivate the EOAM:

1. Install a safety pin to prevent rotation of the actuation lever.
2. Remove and discard the lock wire from the knurled screw.
3. Loosen the knurled screw.
4. Slide the actuating lever to disengage it from the percussion device.
5. Tighten the knurled screw.

WARNING

EXTREME CAUTION SHOULD BE TAKEN WITH THE EOAM, IT IS PRESSURIZED EQUIPMENT. AN ACCIDENTAL PRESSURE RELEASE CAN CAUSE A SERIOUS INJURY OR DEATH.

EMERGENCY OPENING ASSIST MEANS SERVICING

If the EOAM has been used, it can be serviced on the aircraft. A filling valve allows servicing after reconditioning of the EOAM.

Do the EOAM reconditioning as follows:

1. Remove the actuation lever.
2. Replace the diaphragm and the perforator of the mechanical percussion device. Refer to Component Maintenance Publication CMP 52-11-08.
3. Remove the dust cap from the filling valve and connect the supply hose to the filling valve.

NOTE

Use only nitrogen gas to inflate the pneumatic assembly.

Make sure that the nitrogen supply hose and the filling valve are clean before connecting the supply hose. Contamination inside the pneumatic assembly can cause a leak.

4. Loosen the valve nut.
5. Inflate the pneumatic assembly to 2175 psi by steps of 290 psi with a minimum of 10 seconds between each step. Make sure that there is no pressure drop while inflating.

NOTE

A small pressure drop less than 72 psi can occur just after inflation because of the nitrogen temperature and pressure stabilization. If there is a sudden drop during servicing of the EOAM, refer to the CMP 52-11-08

6. If there is no leakage, adjust the pressure to the nominal value of 2175 psi at the room temperature of 20°C (68°F).
7. Tighten the valve nut and torque to 88.50 lbf·in.
8. Install the dust cap on the filling valve.
9. Reinstall the actuating lever and secure it with the knurled screw in the armed position.
10. Install new shear pin.
11. Tighten the knurled screw and safety with lock wire.

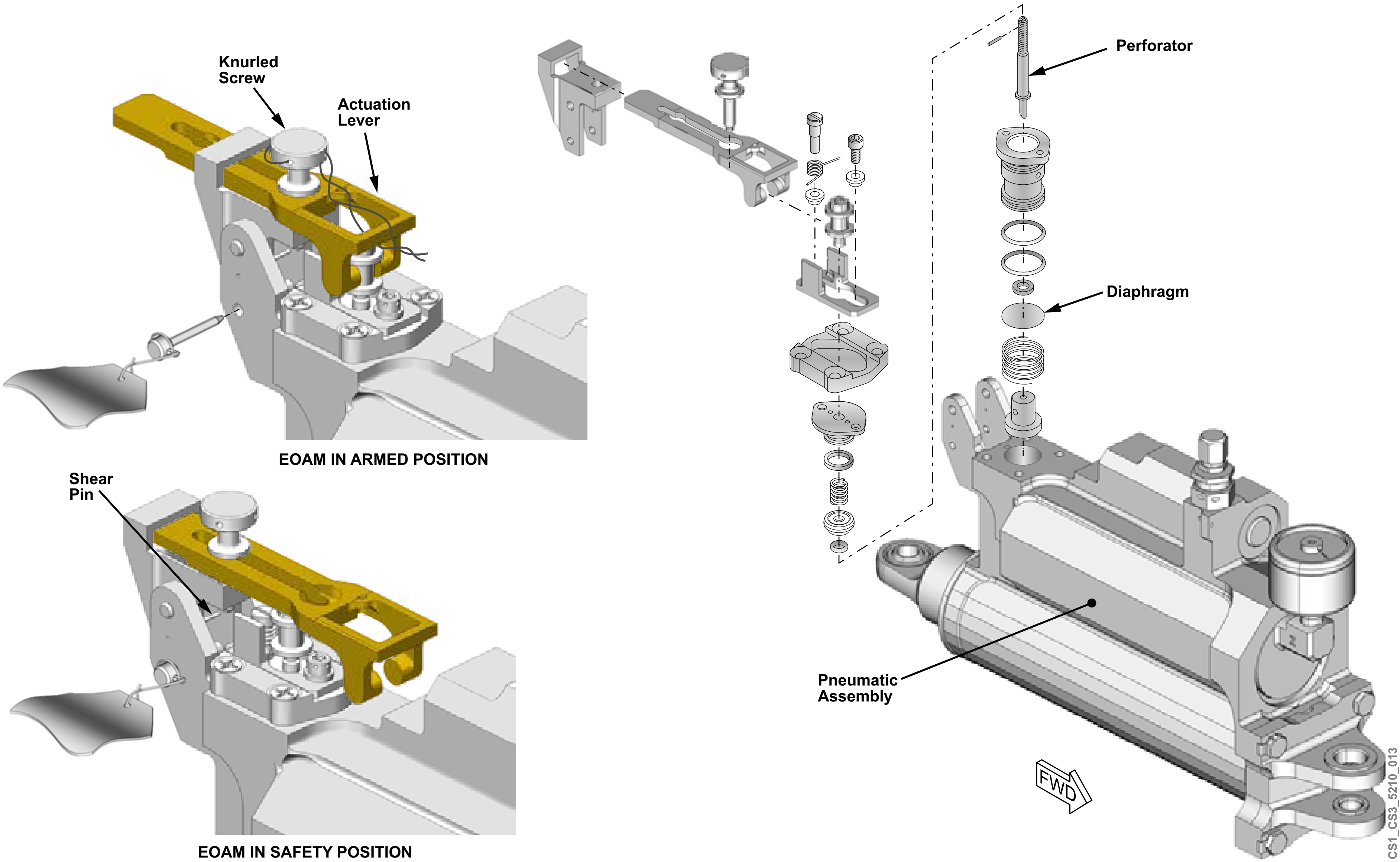


Figure 27: Emergency Opening Assist Means

52-21 OVERWING EMERGENCY EXIT DOORS

GENERAL DESCRIPTION

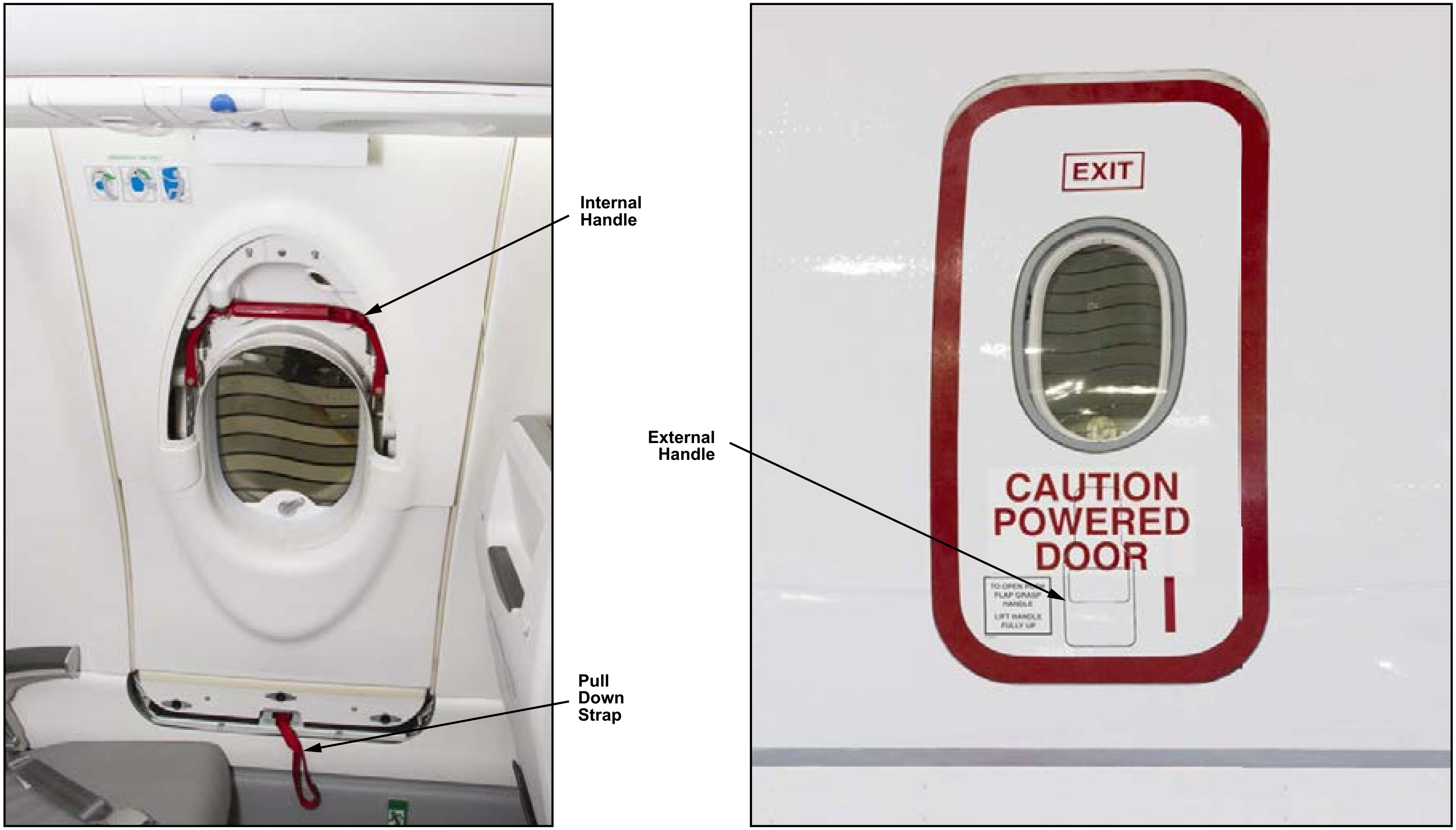
The overwing emergency exit (OWEE) doors are located in the center fuselage above the wings. The OWEE door is a type III emergency exit. Two additional OWEE doors, installed aft of the existing OWEE doors are available as an option on the CS300. When the additional OWEE doors are not installed, plugs, constructed in a similar manner to the OWEE doors, are installed.

The OWEE door is designed to be installed on the left or right side of the aircraft. The door is an inward type semi-plug door. The door is classified as an emergency exit and triggers an overwing escape slide automatically when opened. (Refer to chapter 25 for additional information.) The OWEE door can be opened from the inside or outside, but can only be closed from inside the cabin.

The door is manually opened using an internal handle or an external handle. The initial door opening motion is inward and upward. When the door is lifted, it rotates outward and upward clear of the emergency exit.

The OWEE door has a window to check the status of the area around the door.

A pull down strap is used to pull the door into the opening before using the internal handle to lock the door closed. When the internal handle is in the locked position, a removable cover is installed over it.



NOTE

OWEE door shown with handle cover and lower liner removed.

Figure 28: Overwing Emergency Exit Doors

OWEE DOOR INSTALLATION

The OWEE door is mounted on a hinge arm. The hinge arm is attached to the fuselage structure and supports the weight of the door when lifted.

An actuator fitted between the OWEE door and the fuselage assists in opening the door. The actuator provides a hydraulic damping function to control the opening speed of the door and a pneumatic charge to assist in opening the door. An actuator lock plate secures the door in the fully opened position. The actuator operates as soon as either door handle is operated and the door stop fittings have lifted clear of the fuselage stop fittings.

Each OWEE door has six door stop fittings that transmit door pressurization loads through the fuselage stop fittings to the fuselage structure. Four guide fittings direct the door mounted guide rollers to control the inward upward movement of the door.

Each OWEE door has two proximity sensors. The closed proximity sensor, located in the door cutout, monitors the door closed status. The latch proximity sensor, installed on the internal handle mechanism, monitors the door latched status.

CAUTION

The actuator pushes the door fully open within 3 seconds. If the door is opened from the outside, stand clear of the door as the handle is lifted.

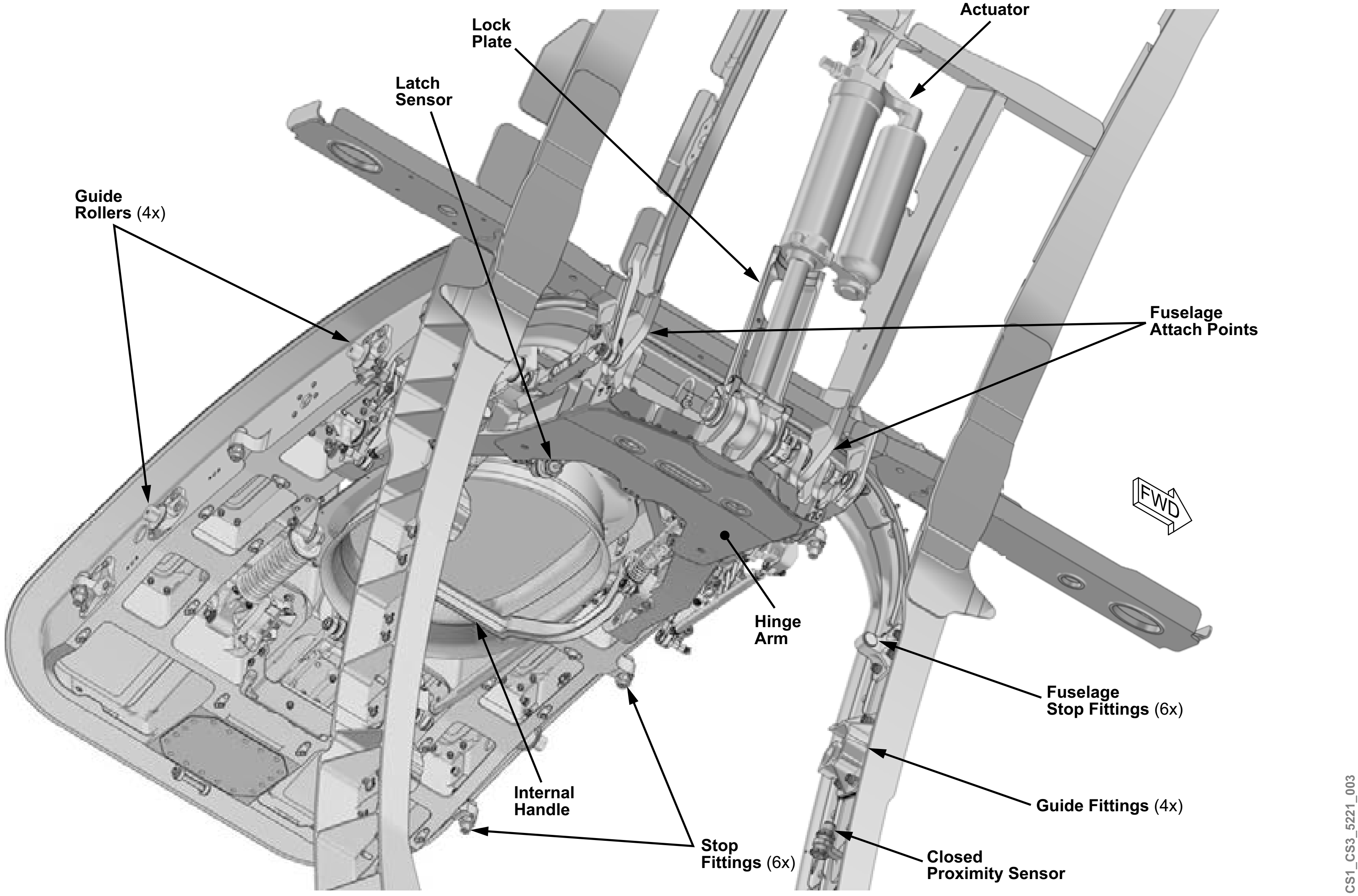


Figure 29: Overwing Emergency Exit Door Installation

OWEE DOOR MECHANISMS

Interior and Exterior Handles

The interior handle mechanism consists of a handle and two spring rods, which are used to help keep the handle in an overcentered and closed position. Pulling down on the handle overcomes the spring pressure to open the door, and rotates the two latch shaft cranks connected to the handle through rods. The rotation of the latch shaft cranks lifts the door free of its stops, allowing it to swing open.

The exterior handle is connected to the interior handle by a series of gears. The external handle mechanism contains a handle coupled an external handle shaft. Opening the door from the outside does not disarm the off-wing slide. The slide can only be disabled from the inside.

When the door is closed from the inside, the external handle follows the interior handle, and closes flush with the door when the door is fully closed. When the door is opened from the inside, the external handle shaft rotates, while the external handle remains flush with the door.

OWEE Door Lockout

The OWEE doors can be locked from the interior using a pin. The pin locks the internal handle to the hinge.

Lift and Latch Mechanism

The latch mechanism is comprised of a latch lift cam, two latch operating mechanisms, and two latch safeguard means. The latch mechanism is operated by either the internal or external handle mechanism. If a failure occurs in the latch lift cam, the latch safeguard means prevents the door from lifting.

The lift mechanism helps lift the OWEE door on its hinge when the OWEE door is unlatched. When the door is lifted, it opens outward and upward. The lift latch cam on the latch shaft controls the lifting sequence.

Interlock Mechanism

The OWEE door interlock mechanism prevents door mechanism operation when the door is in the open position. When the door is in the opened position, the interlock mechanism is engaged on the latch mechanism to prevent the handle from being operated and putting the door out of sequence. The interlock mechanism engages automatically when the door opens and must be manually operated when the door handle is moved to the closed position.

Pressure-Sensing Flap

The spring-loaded pressure-sensing flap is installed on the internal structure of the OWEE door. The pressure-sensing flap acts in a similar manner to the vent panel on the passenger and service doors. The pressure-sensing flap is closed when the door is closed and opens when the door is opened to relieve any cabin differential pressure. The pressure-sensing flap also provides a locking function when the door is closed and the cabin pressure differential is greater than 2 psi.

Self-Opening Mechanism

In order to avoid the possibility of the door being left in a slightly lifted position, a self-opening mechanism continues to lift the door to the fully lifted position using spring force.

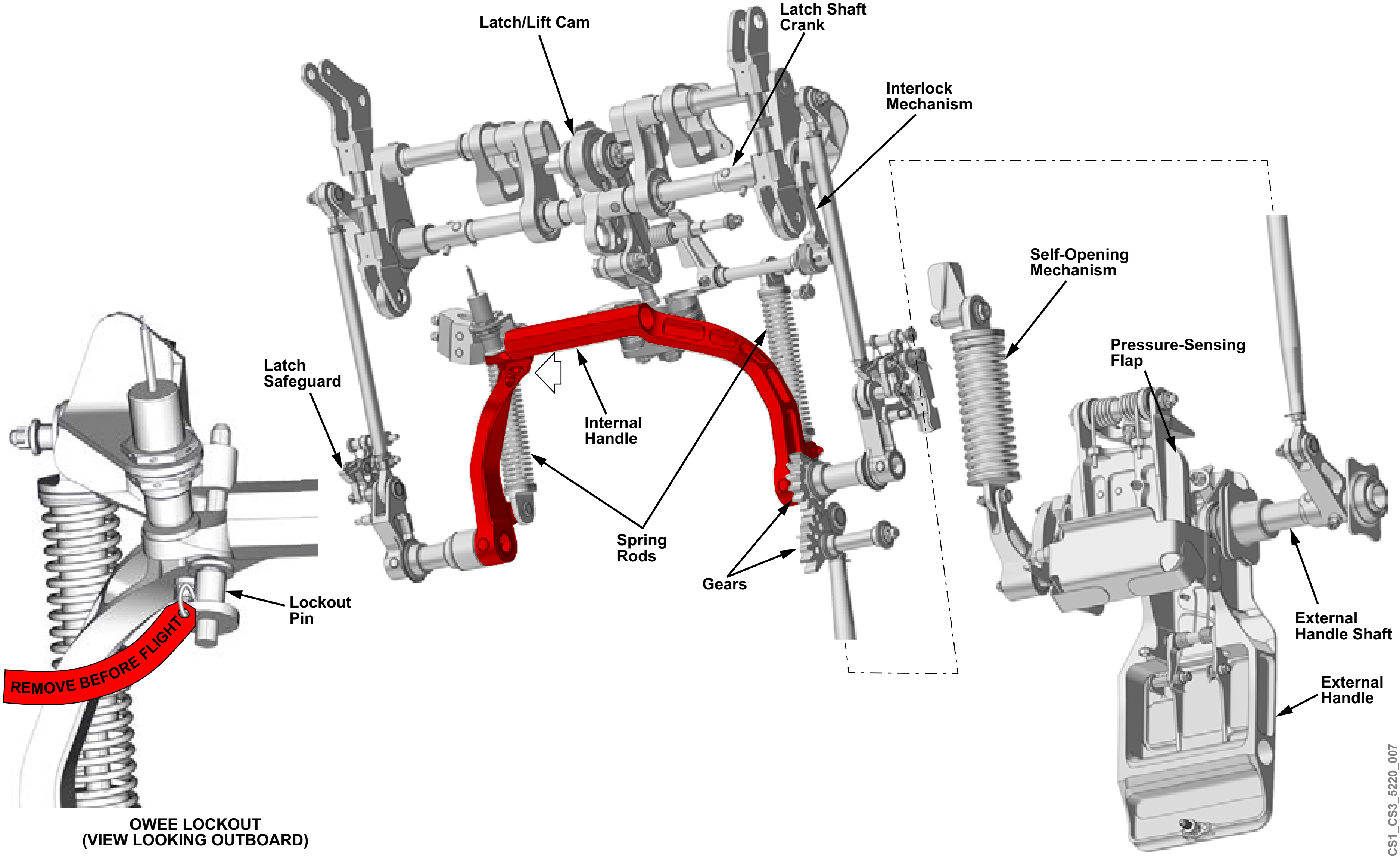


Figure 30: OWEE Door Mechanisms

ESCAPE SLIDE MECHANISM

Off-wing escape slides, located in the wing-to-body fairing (WTBF), deploy when the OWEE door opens. (Refer to ATA 25 for additional information.) The escape slide is normally deployed through the slide crank assembly.

When the OWEE door opens, the command link actuates the slide crank assembly. The crank assembly pulls on the escape slide cable and triggers the deployment of the escape slide automatically.

If the slide fails to deploy automatically, a manual backup cable handle pulls the escape slide cable through the manual slide cable.

The OWEE door can be operated without deploying the escape slide by inserting a key in the slide mode select mechanism. Rotating the key operates the disarming rod which disables the crank assembly.

The operation and mechanisms of the additional OWEE door, if installed, are identical to the existing doors. The only difference is the routing of the activation cables between the doors and the slide and is located in the WTBF.

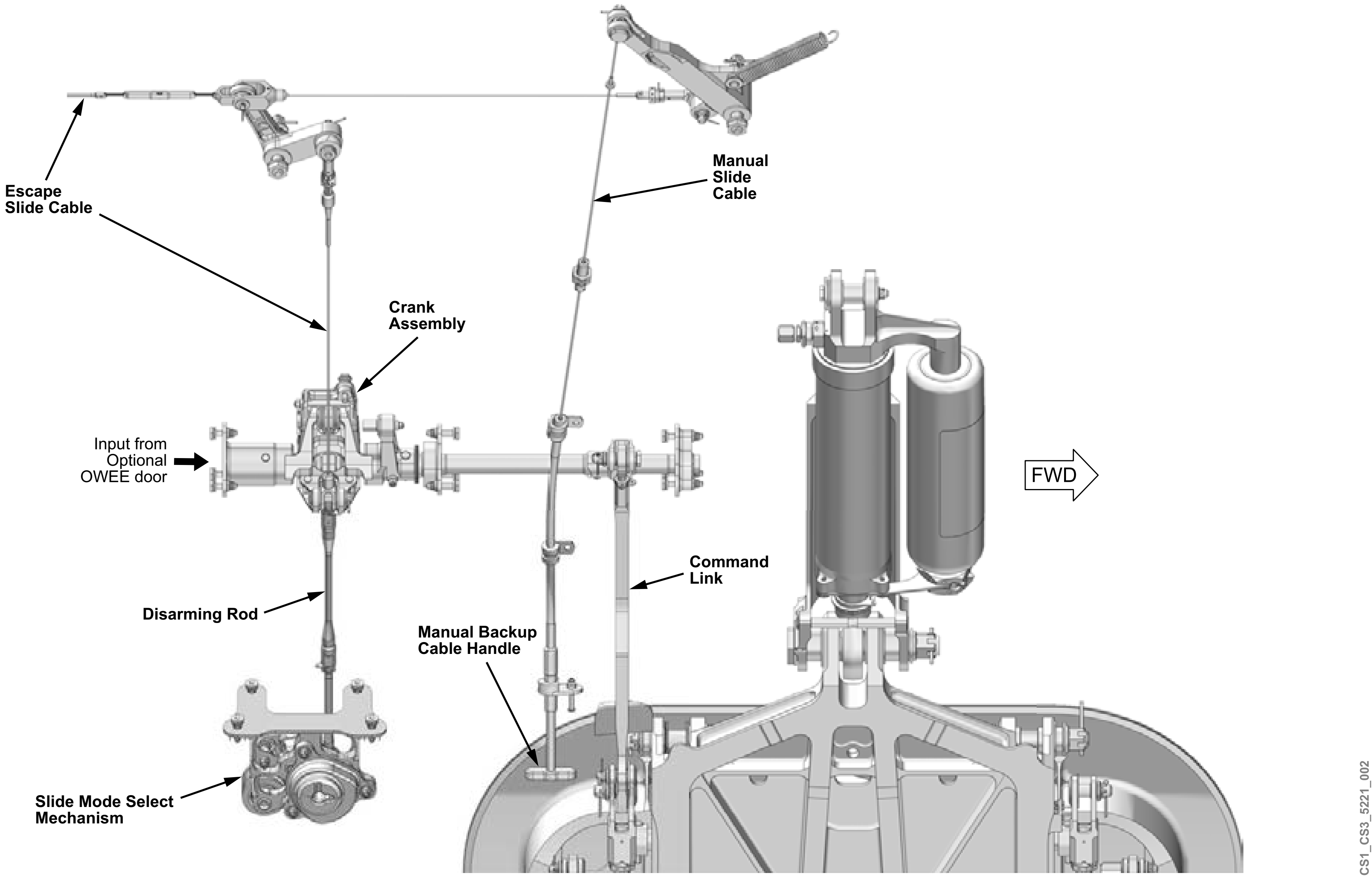


Figure 31: Escape Slide Deployment

COMPONENT LOCATION

The OWEE door has the following components:

- Manual slide cable
- Actuator
- Hinge
- Internal handle
- Closed proximity sensor target
- Pressure-sensing flap
- Shield
- Latch proximity sensor
- Guide fitting
- Stop fitting
- Slide mode select mechanism
- Crank assembly

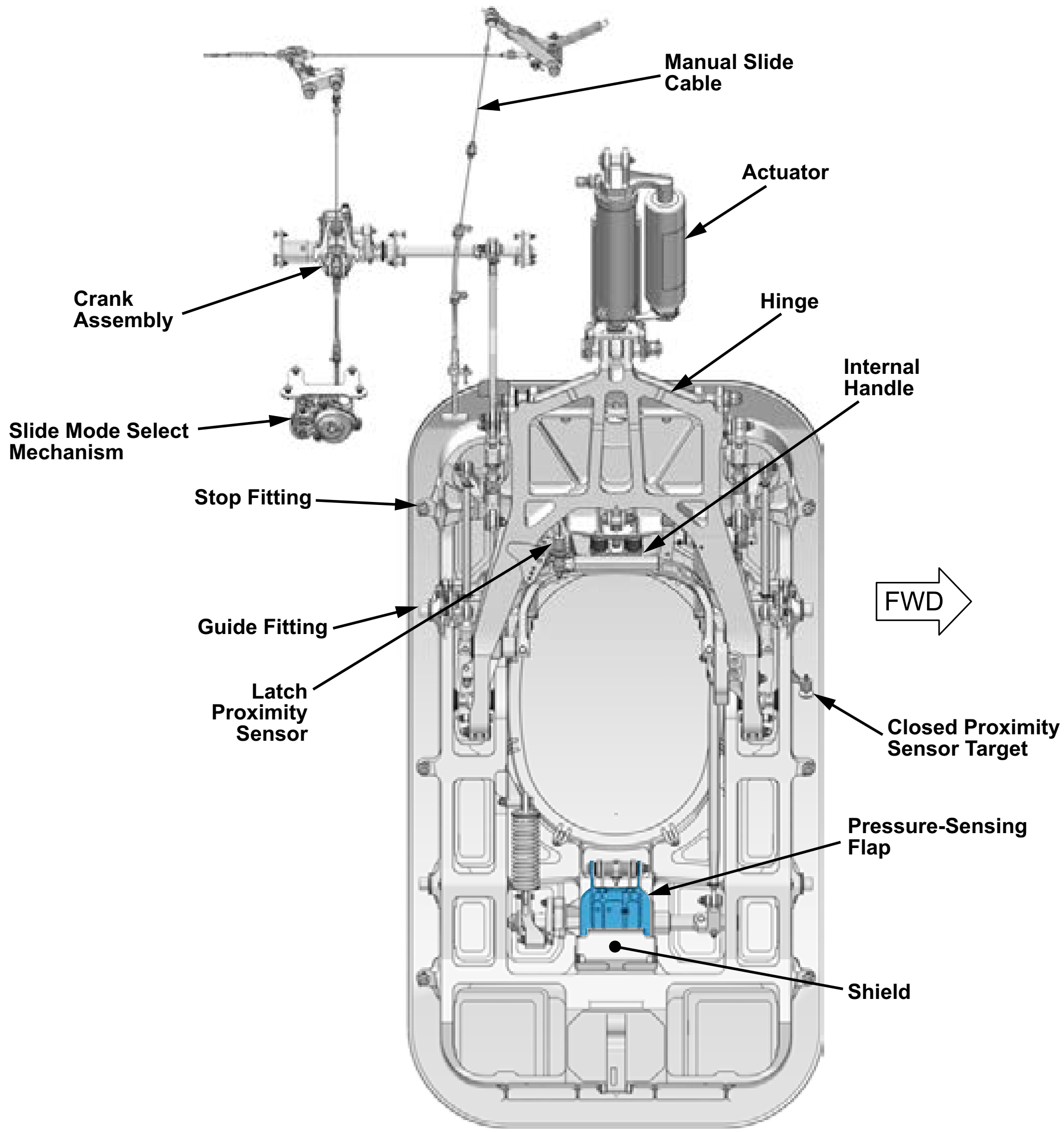


Figure 32: Overwing Emergency Exit Door Component Location

COMPONENT INFORMATION

OWEE DOOR ACTUATOR

The OWEE door actuator has an accumulator that provides the energy to push the door open. It is automatically triggered as the door is lifted by the internal or external door handles.

The actuator is attached between the OWEE door and the aircraft structure. An accumulator on the actuator assists in opening the door. A hydraulic damper limits the door opening speed. The accumulator is charged with nitrogen and does not require recharging after the door has been opened.

A lock plate on the actuator is used to hold the door in the open position. The spring-loaded lock plate is fixed to the side of the actuator. The lock plate rotates inward against the actuator when the actuator has reached the open position.

To close the OWEE door, the locking plate is released using a screwdriver or similar hand tool, inserted into the base of the locking plate. Rotate the locking plate to disengage it from the actuator and then pull the door closed.

CAUTION

The actuator pushes the door fully open within 3 seconds. If the door is opened from the outside, stand clear of the door as the handle is lifted.

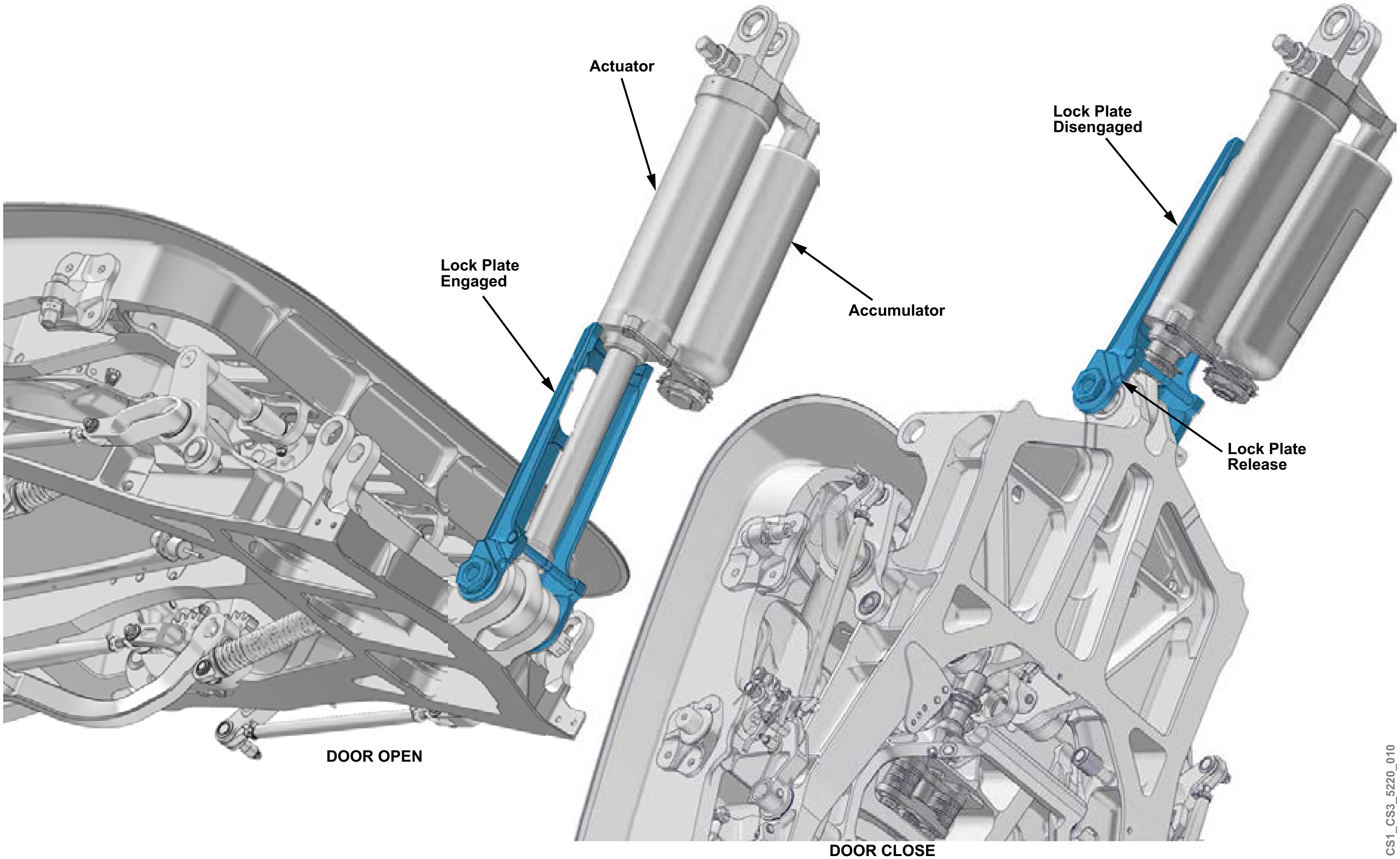


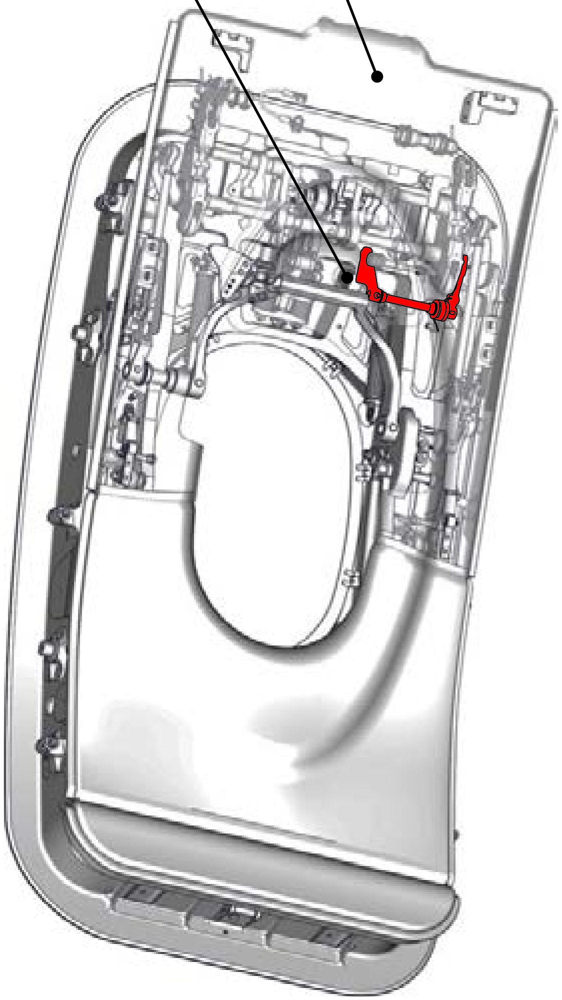
Figure 33: OWEE Door Actuator

INTERLOCK MECHANISM

The interlock mechanism prevents the door handle from being operated once the door is opened. The spring-loaded interlock hook engages the latch shaft when the door is fully raised.

The hook is disengaged using the interlock lever that is accessed through a cutout in the liner.

During closing, the door is pulled into the frame. When the door is engaged with the door frame, the interlock lever is manually actuated, allowing the door handle to lock the door in the closed position.



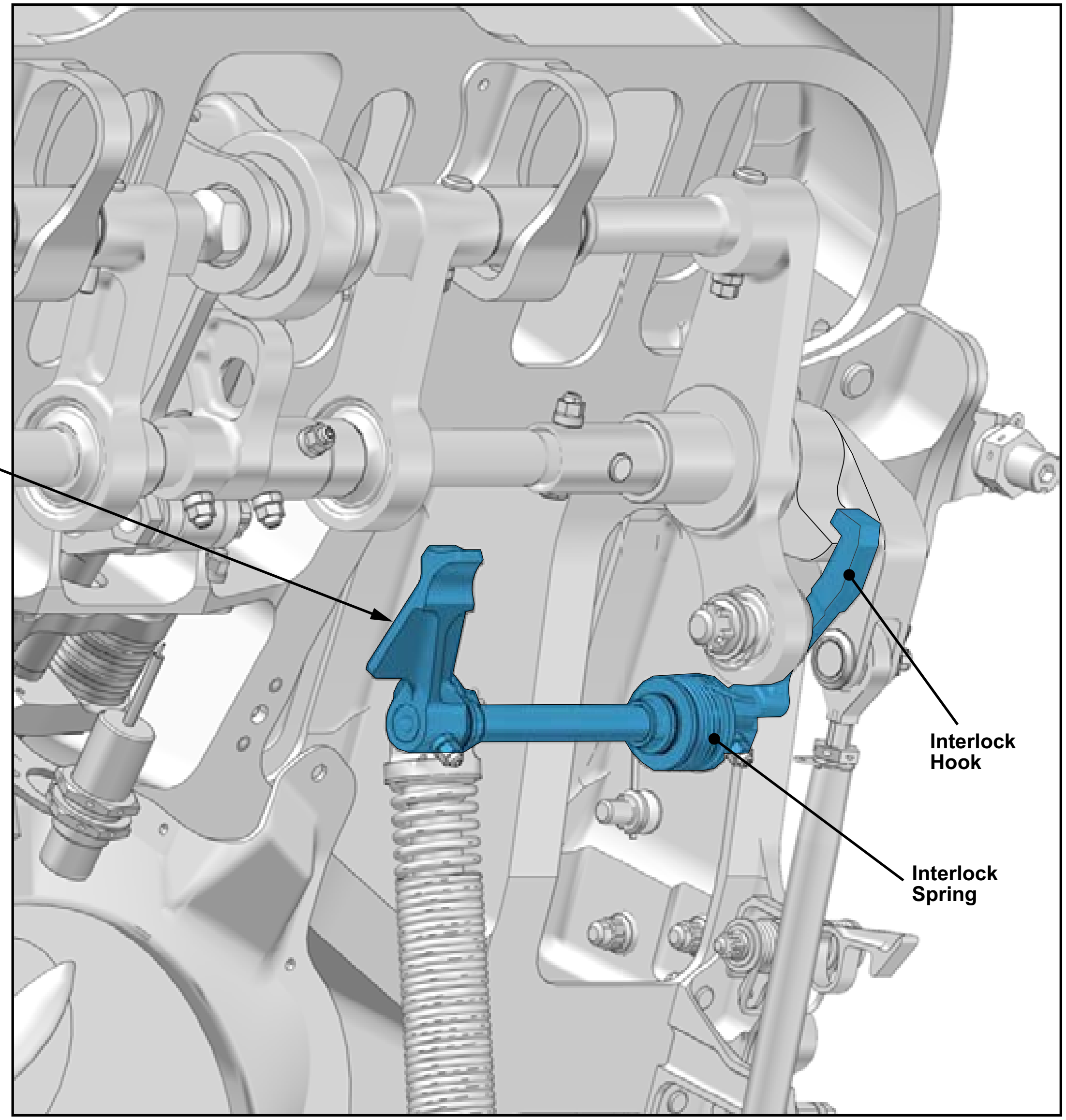


Figure 34: Interlock Mechanism

INTERIOR DOOR PANEL

The interior door panel is attached to the top of the OWEE door by piano hinges and spring hinges at the bottom. As the door is opened a hinge lining hook unlatches, allowing the interior panel to be pushed open by two spring hinges. The purpose of this is to provide clearance for the door mechanisms to operate. An open panel is indicated by a red stripe on the side of the panel.

When the door is closed, the panel is restowed by pushing on it to latch the hinge lining hooks.

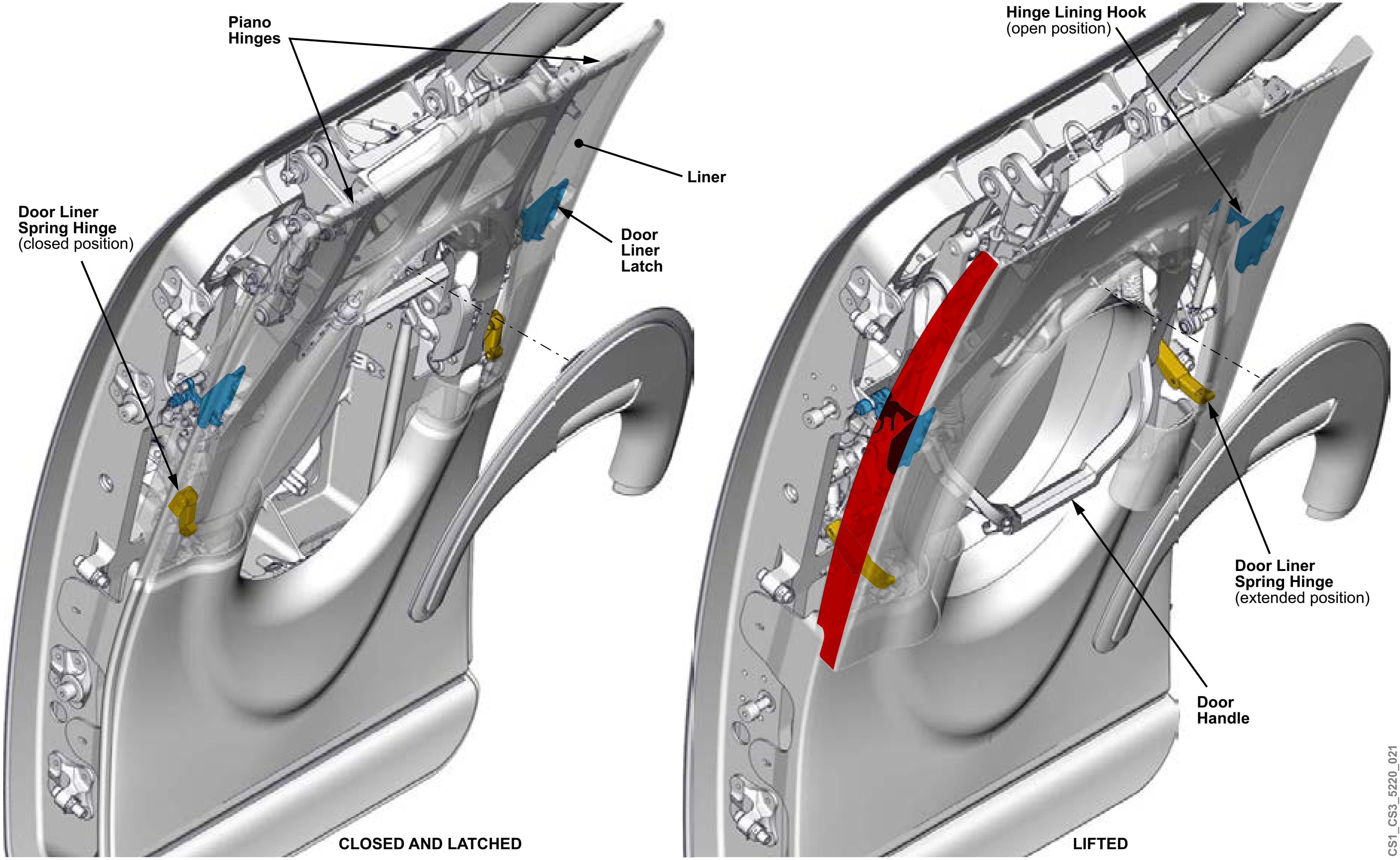


Figure 35: Interior Door Panel

ESCAPE SLIDE MODE SELECT MECHANISM

The slide mode select mechanism is used to disarm the OWEE slide if the OWEE door has to be opened. The slide mode select mechanism is located behind the emergency exit sign located next to the OWEE door.

The slide is locked out using the OWEE armed/disarmed emergency key. The key is inserted into the slide mode select mechanism and rotated counterclockwise to disarm the slide through the disarming rod. When the key is rotated to the disarm position removal from the slide mode select mechanism is not possible. The mode select mechanism is held in the disarm position by a spring and lever assembly.

The mode select mechanism has red and green visual indicators. When the key is in the disarm position, a green stripe is shown. When the key is returned to the armed position, the red mark is shown.

The operation and mechanisms of the optional OWEE door, if installed, are identical to the existing doors. The only difference is the routing of the cables between the doors and the slide and are located in the wing-to-body fairing (WTBF).

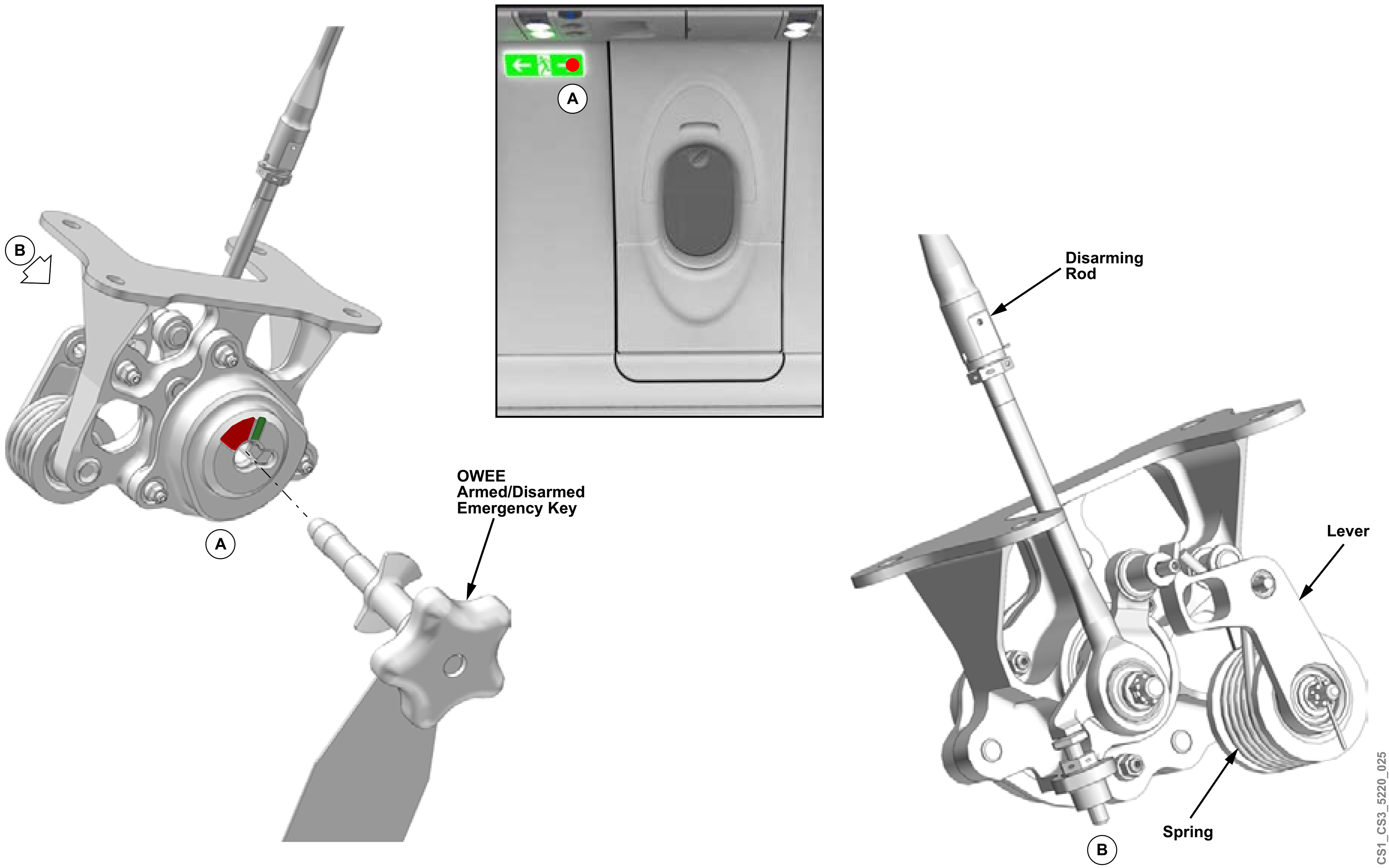


Figure 36: Escape Slide Mode Select Mechanism

DETAILED COMPONENT INFORMATION

DOOR LIFT MECHANISM

The door lift mechanism is designed to lift the door above the door pressure stops and clear the fuselage stop fittings. The latch lift cam contacts the lift roller during rotation and lifts the door.

The lift roller is part of the door structure and lifts the door along a path defined by the guide fittings. The door lifting guides and holdup links help control the lift of the door.

The lift jam load fitting, composed of multiple spring washers, limits the load on the door structure in case of door jamming.

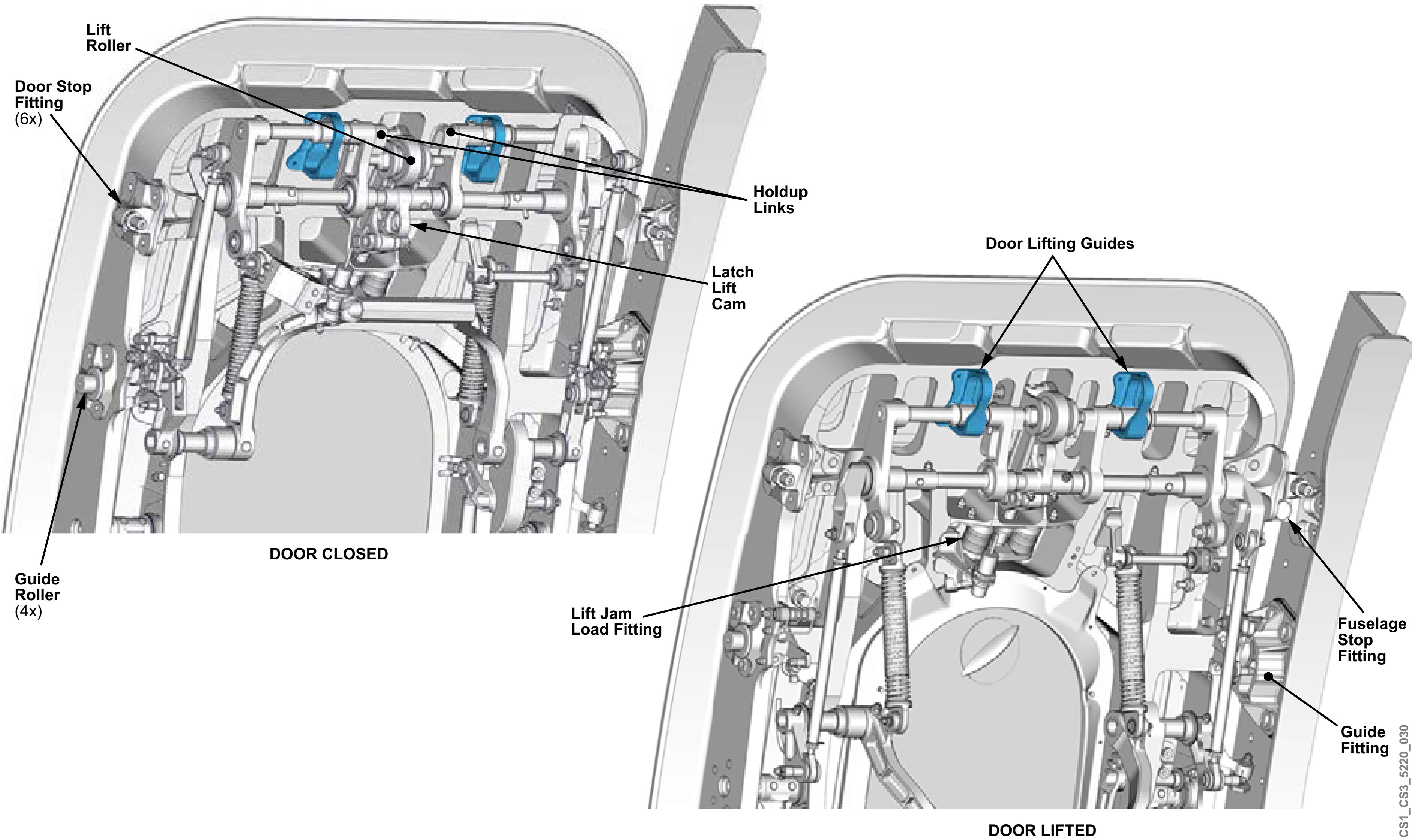


Figure 37: Door Lift Mechanism

PRESSURE-SENSING FLAP

A lock roller on the external handle shaft and lock cam on the flap is used to close the flap. As the OWEE door is closed, the external handle shaft rotates and the lock roller rides along the lock cam to pull the pressure-sensing flap closed. When the external handle is fully closed the lock roller is engaged with the lock cam and holds the pressure-sensing flap closed.

When the door is opened and the aircraft is not pressurized, the rotation of the external handle shaft disengages the lock roller from the lock cam and the spring opens the pressure-sensing flap inward. The pressure-sensing flap full open travel is limited by the shield assembly installed on the door structure.

If the aircraft is pressurized above 2 psi, the pressure acting on the pressure-sensing flap is greater than the spring force. The pressure-sensing flap remains closed even if the lock roller and lock cam disengage. If an attempt is made to open the door, the external handle clutch crank engages the pressure-sensing device lock and prevents either handle from opening the door.

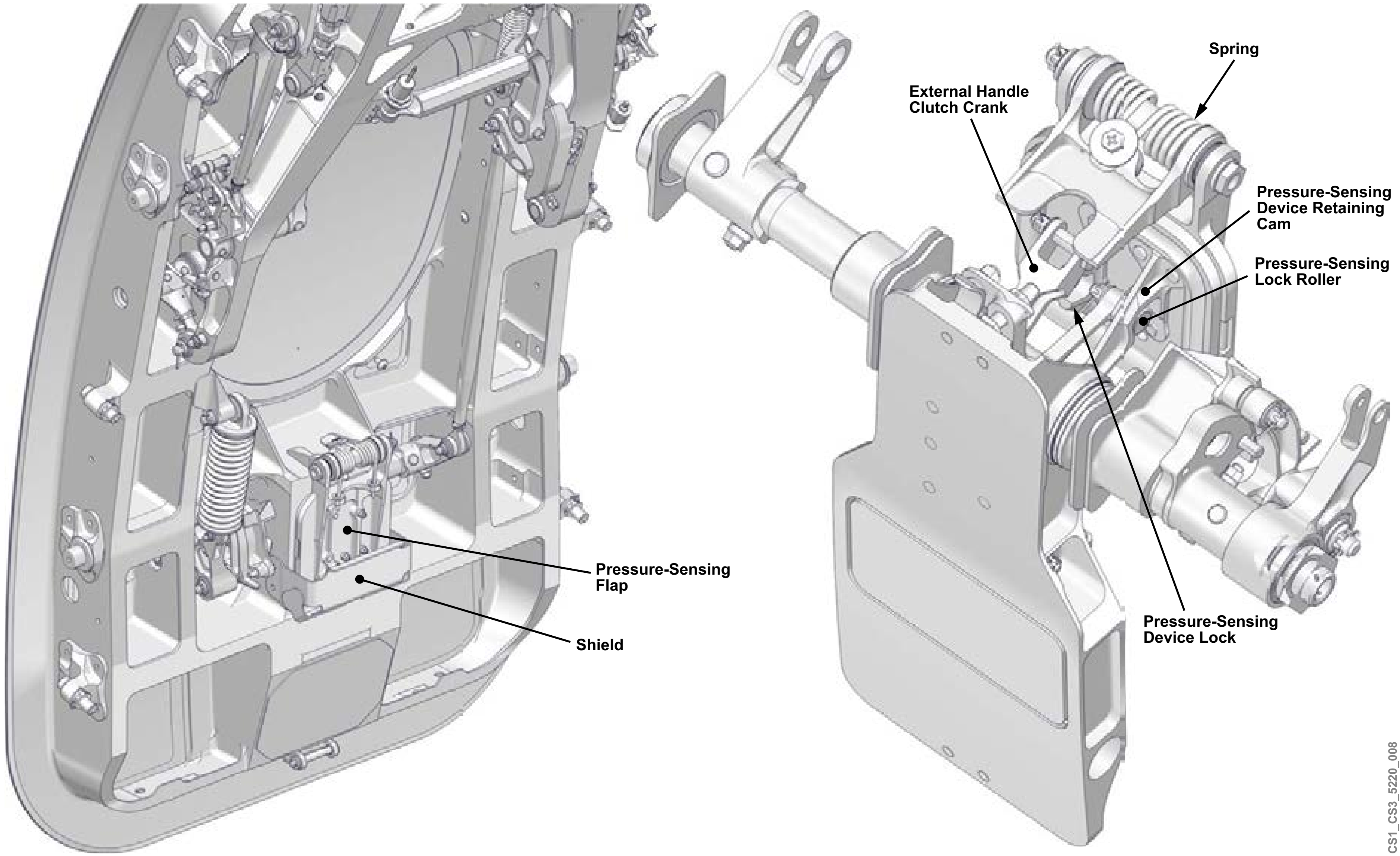


Figure 38: Pressure-Sensing Flap

CRANK ASSEMBLY

The crank assembly is installed in line with the command shaft. The crank has a spring-loaded striker and a connection point for the slide cable. The crank is free to pivot independent of the command shaft. The command shaft has a hammer installed on one end and is connected to the OWEE door through the command link. When the OWEE door opens, the command link is pulled down, rotating the command shaft.

When the command shaft rotates, the hammer contacts the striker causing the crank to rotate and pull the slide cable. The slide cable triggers the deployment of the escape slide automatically.

When the slide is disarmed, the disarming rod moves up allowing the striker to disengage from the disarming rod hook and pivot downward clear of the hammer. If the door is opened, the command link rotates the command shaft and hammer. The striker is no longer in the path of the hammer, and the slide cable is not pulled.

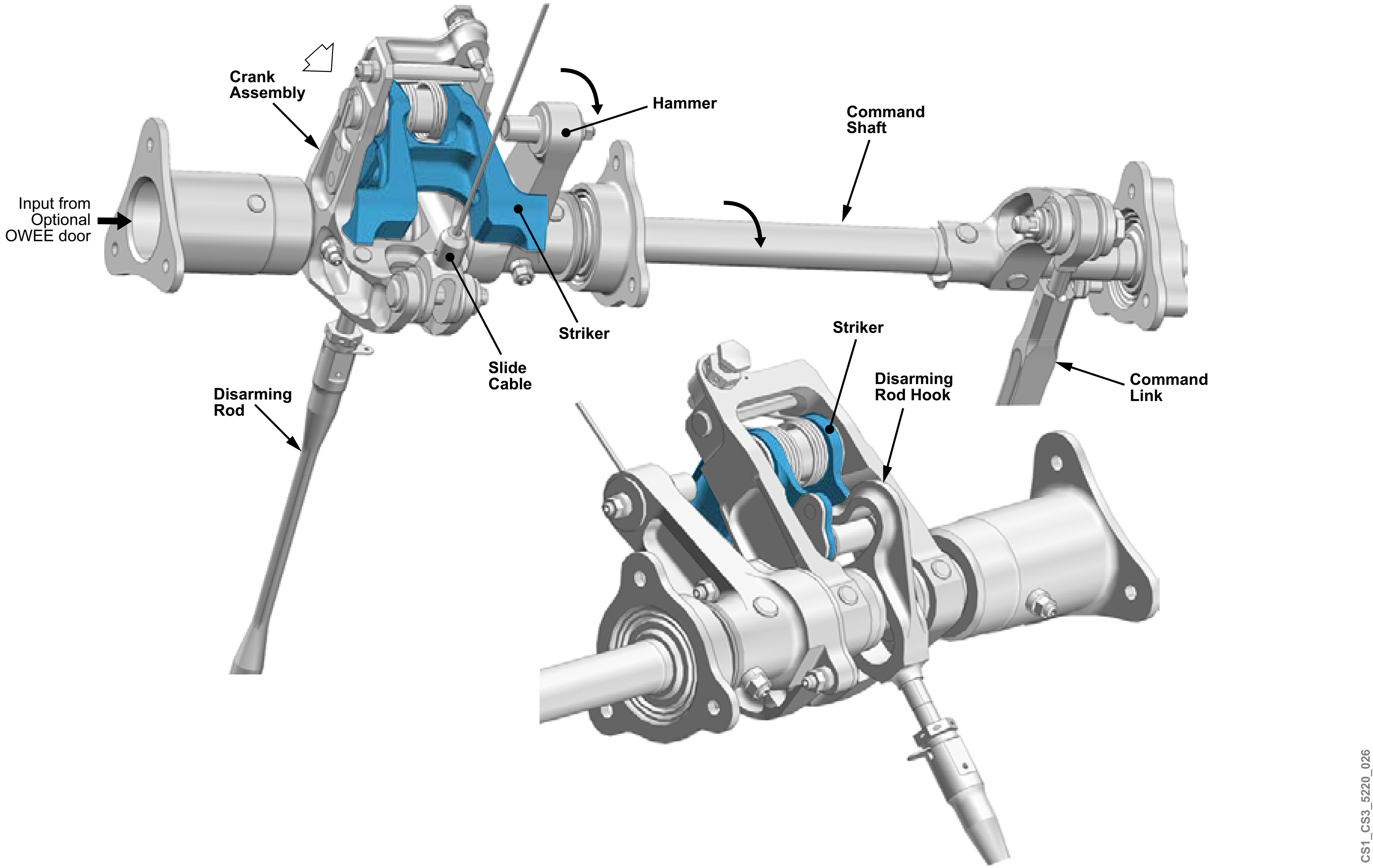
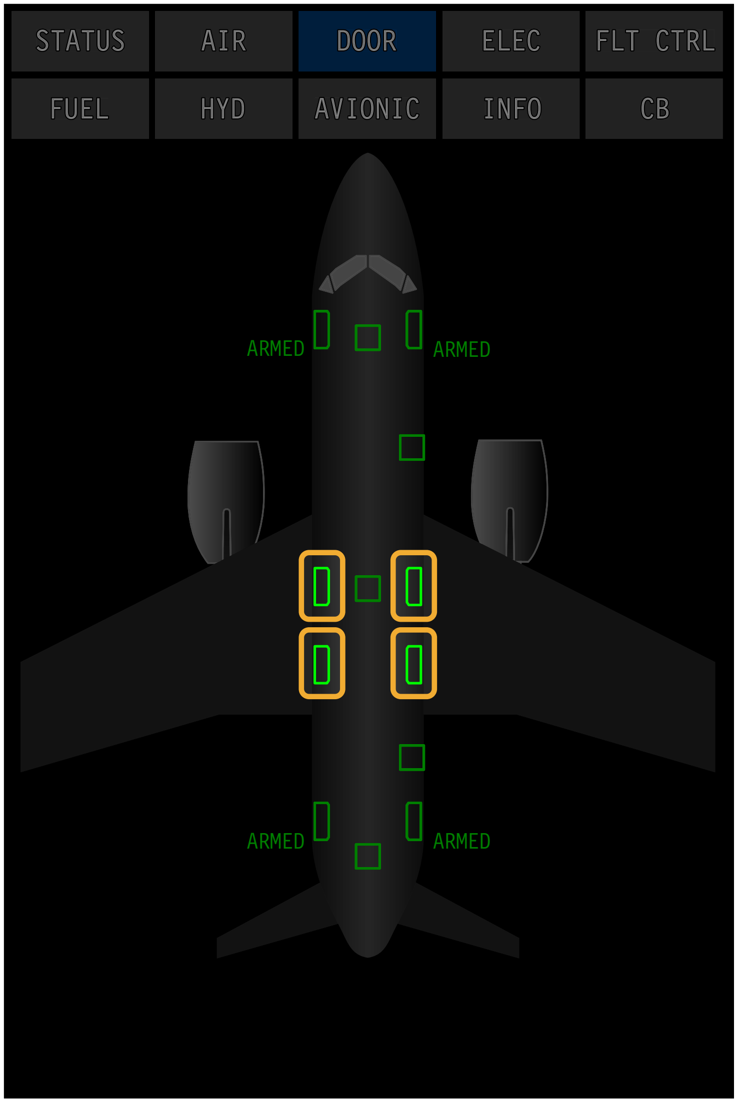


Figure 39: Crank Assembly

CONTROLS AND INDICATIONS

SYNOPTIC PAGE

The status of the overwing emergency exit doors are shown on the DOOR synoptic page.



SYNOPTIC PAGE – DOOR

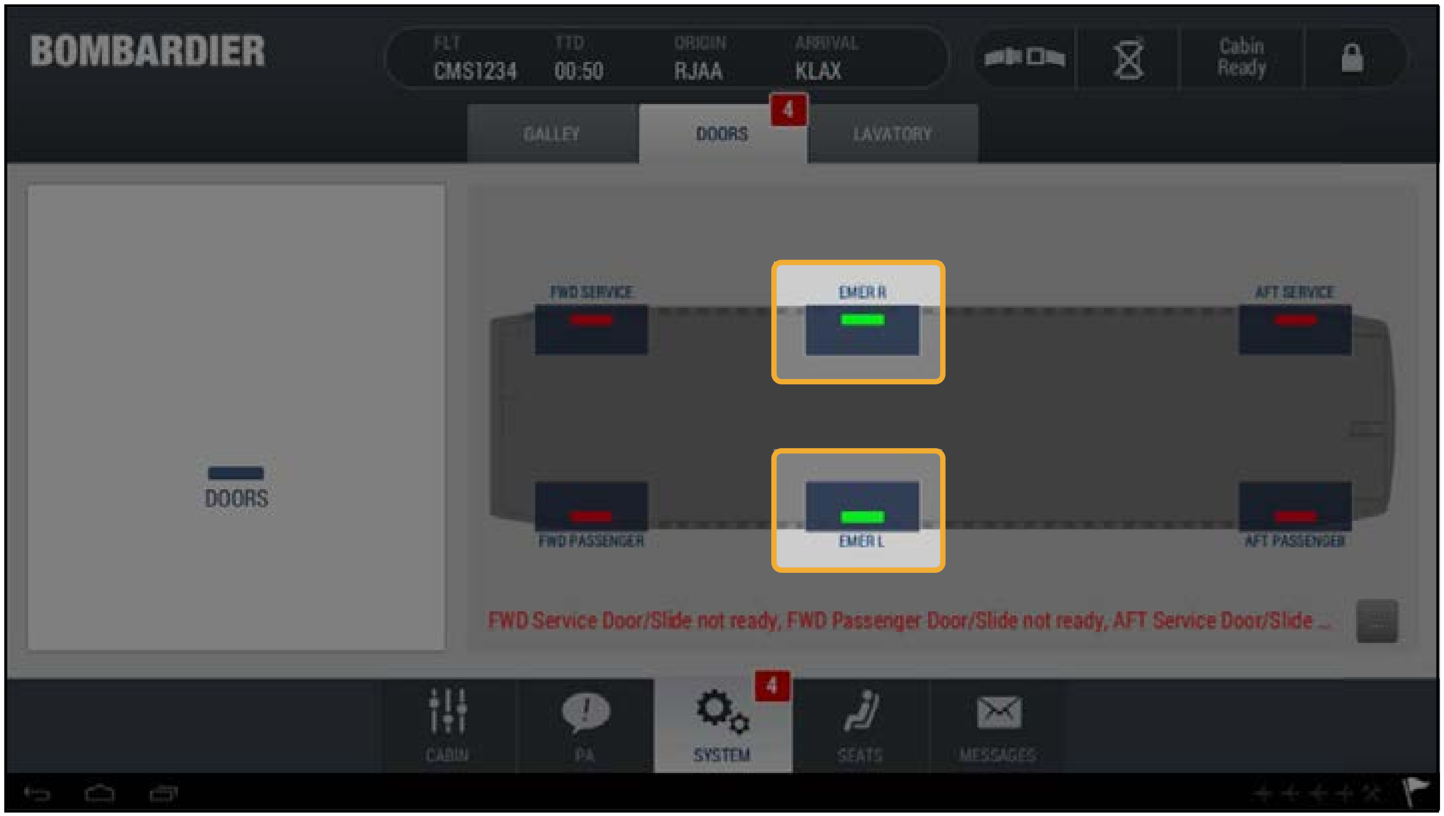
DOOR STATUS	
Symbol	**Condition**
L OVERWING	At least one engine running and the door is not closed/latched/locked. **OR** Door fault. Note: Label of door is shown in amber.
	Door closed/latched/locked. Note: Label of door is not shown.
L OVERWING	Door open on ground, engine is not running. Note: Label of door is shown in white.
L OVERWING	Invalid or failed LGSCU or invalid door sensor. Note: Label of door is shown in amber.

CS1_CS3_5220_015

Figure 40: Synoptic Page

CABIN MANAGEMENT SYSTEM DOORS SCREEN

The status of the overwing emergency exit doors are shown on the cabin management system DOOR screen.



CS1_CS3_5220_023

Figure 41: Cabin Management System Door Screen

OPERATION

INTERNAL EMERGENCY OWEE DOOR OPENING

To open the door from the inside:

1. Check overwing area is clear.
2. Remove handle cover.
3. Pull internal handle down.
4. Release handle as door opens.

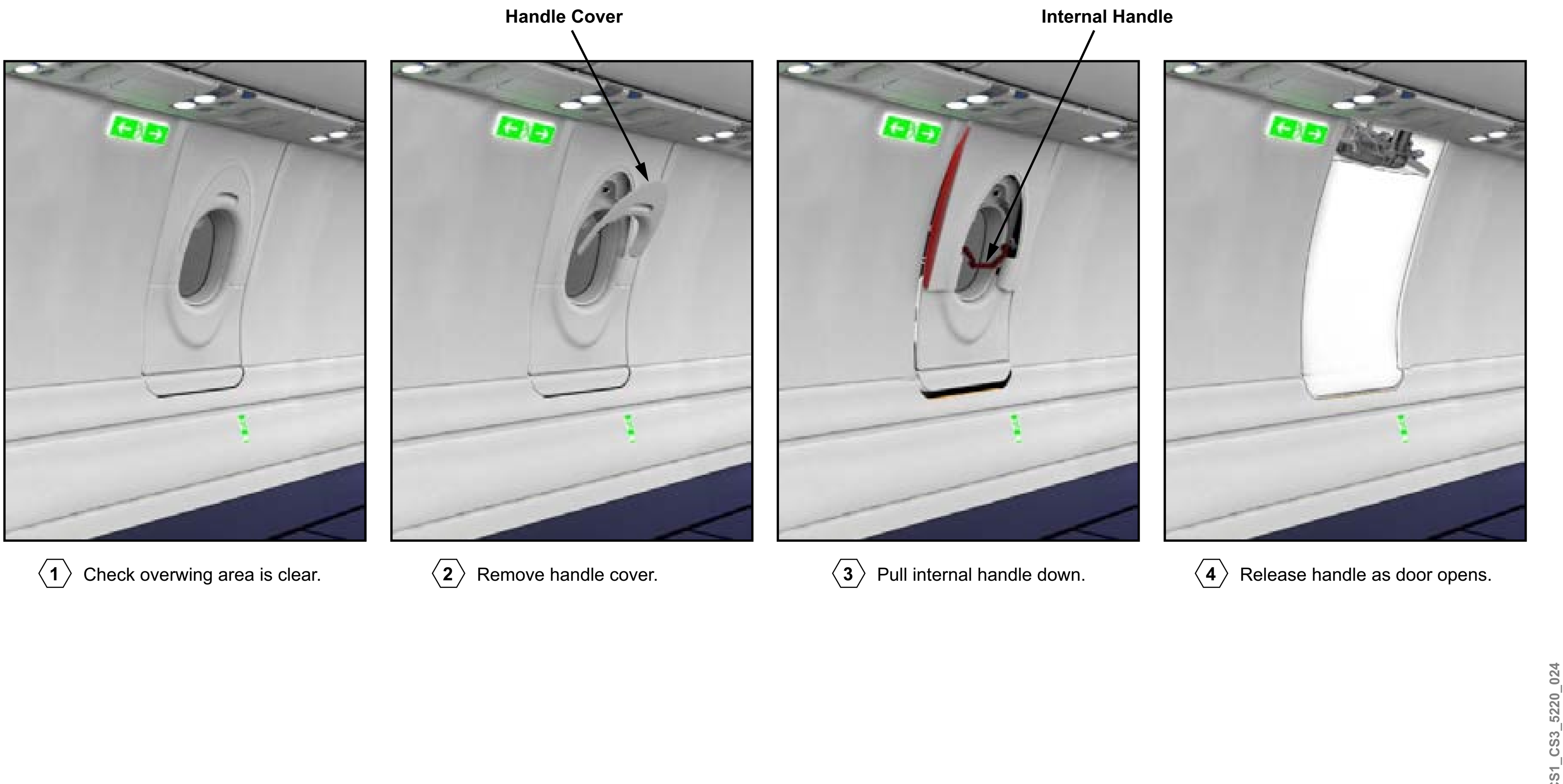


Figure 42: Internal Emergency OWEE Door Opening

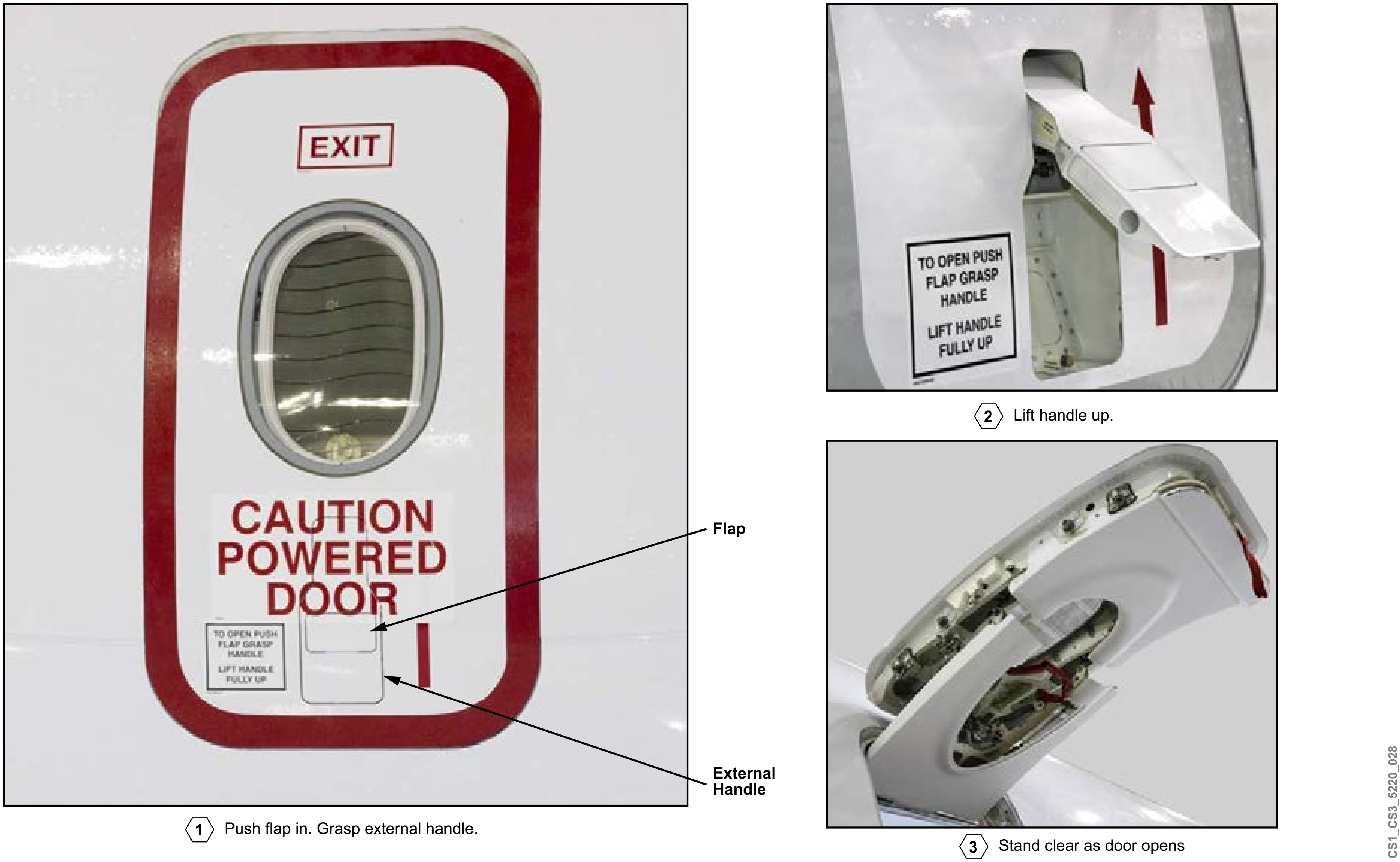
EXTERNAL EMERGENCY OWEE DOOR OPENING

To open the door from the outside:

1. Push external handle flap in.
2. Grasp external handle and pull up.
3. Release handle as the door actuator drives the door open.
4. Stand clear of door and overwing slide assembly.

CAUTION

The actuator pushes the door fully open in less than 3 seconds and the overwing slide deploys. If the door is opened from the outside, stand clear of the door and overwing slide as the handle is lifted.



1 Push flap in. Grasp external handle.

2 Lift handle up.

3 Stand clear as door opens

Figure 43: External Emergency OWEE Door Opening

DOOR OPENING FOR MAINTENANCE

A slide lockout mechanism is used to lockout the overwing slide in order that the OWEE door can be opened without the slide deploying. The slide must be deactivated, even if opening the door from the outside.

A key is inserted into the slide mode select mechanism, located behind the emergency exit sign, disarms the slide. Removal of the key is not possible until the mechanism is in the armed condition.

The visual marks for off-wing condition after key rotation are as follows:

- Red for slide armed
- Green for slide disarmed

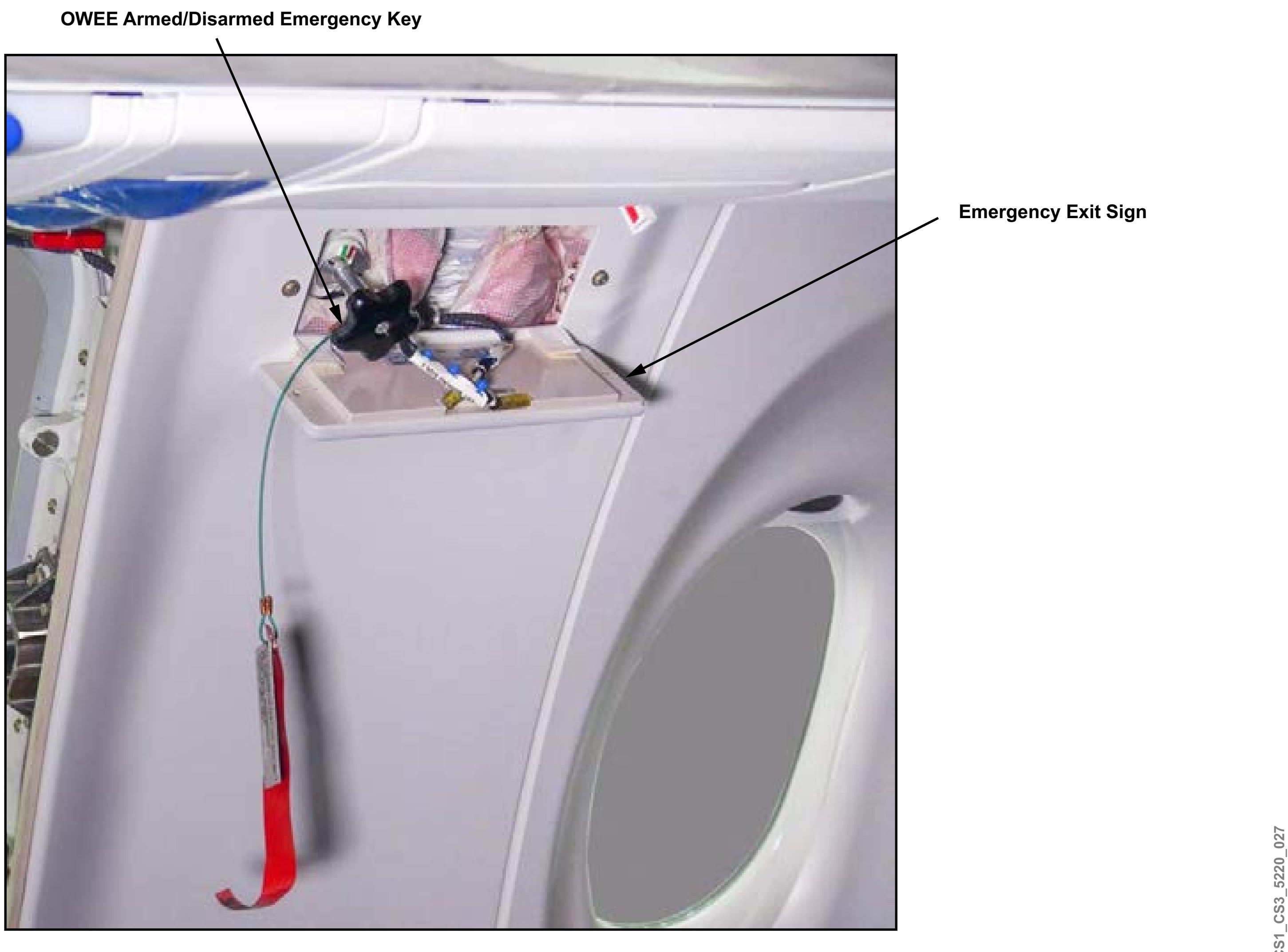


Figure 44: Door Opening for Maintenance

OWEE DOOR CLOSING

To close the OWEE door:

1. Push upward on the door from outside.
2. Insert a screwdriver into the lock plate release.
3. Rotate the screwdriver upwards to disengage the lock plate.
4. Pull the door downwards a few degrees using the door pull strap.
5. Remove screwdriver.
6. Pull the door into the guide rollers from inside the cabin. Push the door into guide rollers from outside.
7. Operate the interlock mechanism to release the internal handle.
8. Push the internal handle up until it locks.
9. Close the door liner.
10. Install the door handle cover.
11. If required, rearm the slide mechanism.

CAUTION

1. **Closing the OWEE door requires two people.**
2. **Do not use the interlock mechanism to move the handle unless the door is in the door frame. If you move the handle without the door being in the frame, damage to the door mechanisms may occur.**

1 Push up on door from outside.
Release the actuator lock plate.

2 Pull the door into the frame closing the door strap.

3 Release the interlock and pull the internal handle up.

A

NOTE

Closing the OWEE door requires two people.

CS1_CS3_5220_019

Figure 45: OWEE Door Closing

DETAILED DESCRIPTION

PROXIMITY SENSORS

The OWEE doors are monitored for latch and closed status.

Latch Proximity Sensor

The latch proximity sensor is installed in the door closing mechanism to confirm that the door latches are secured. The sensor is attached to the door hinge and the target is attached to the internal door handle.

Closed Proximity Sensor

The closed proximity sensor is located in the door frame and used to detect and confirm that the door is closed. The sensor is on the airframe and the target is on the door.

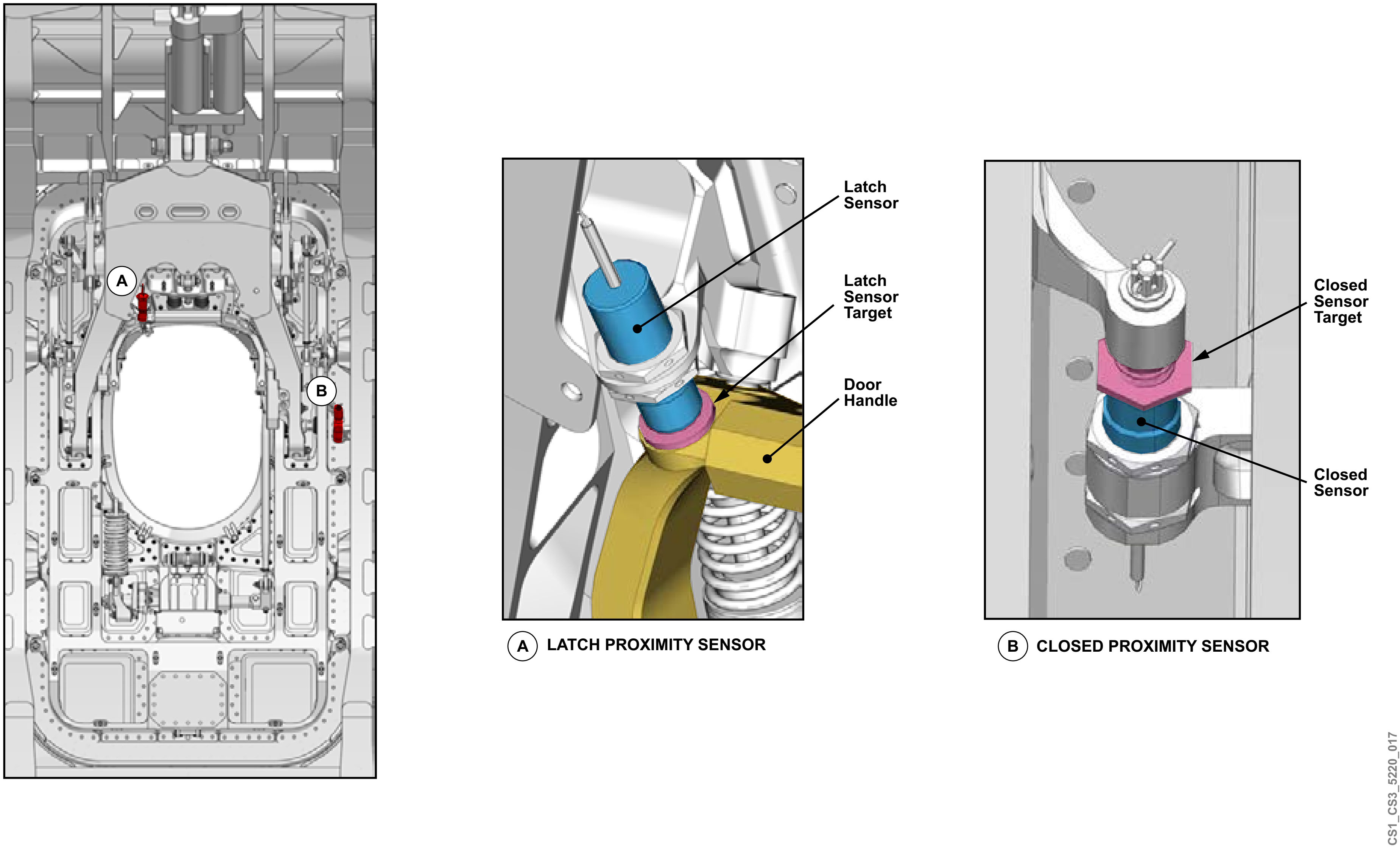


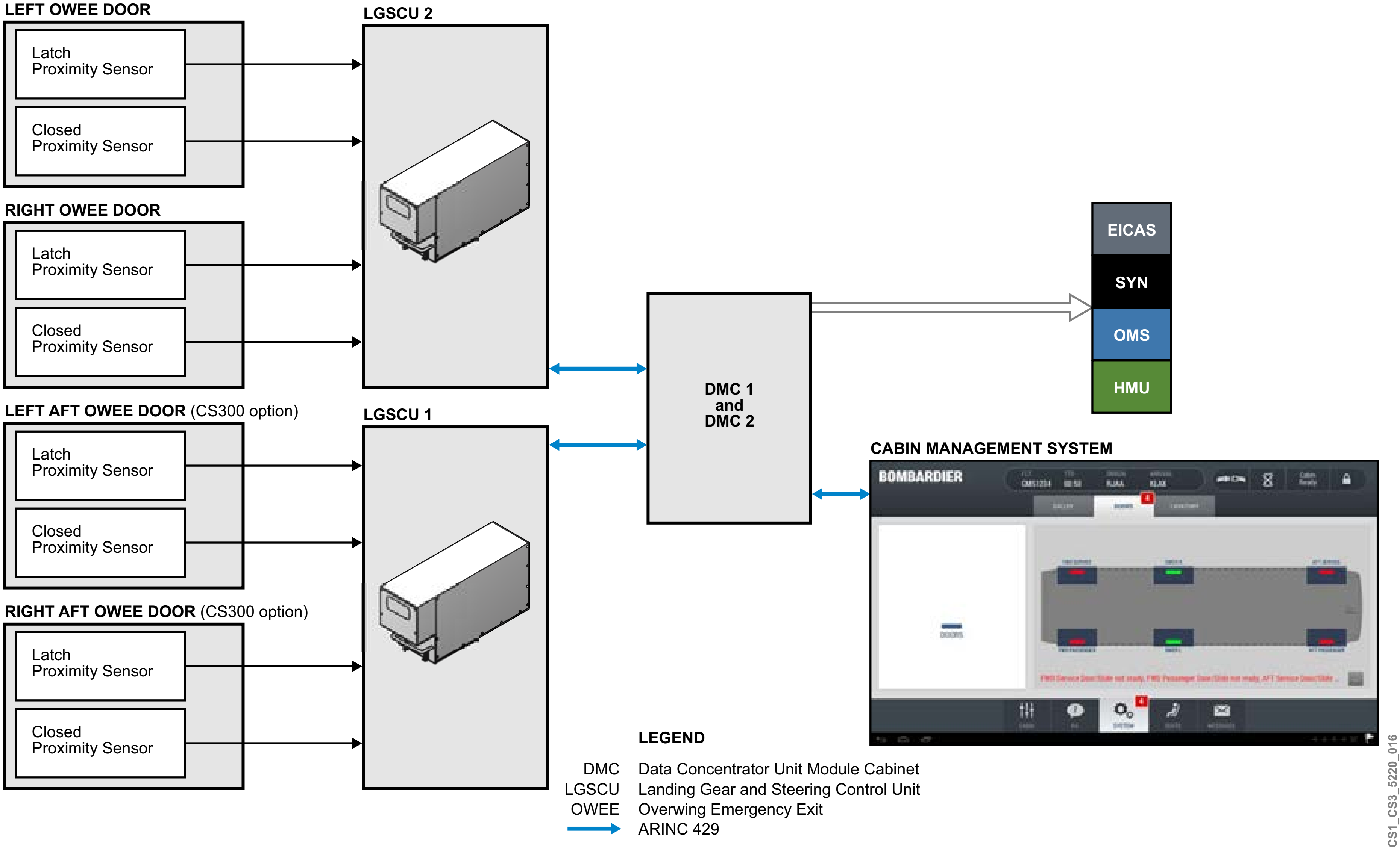
Figure 46: Proximity Sensors

OWEE DOOR POSITION INDICATION

The latch and closed proximity sensors monitor the door position. The proximity sensors for the OWEE doors are connected to landing gear and steering control unit (LGSCU) 2. If the optional OWEE doors are installed, they are connected to LGSCU 1. The door indications are displayed on the DOOR synoptic page and the cabin management system (CMS) DOOR screen. The status of the sensors is monitored by the onboard maintenance system (OMS).

If an OWEE door sensor or target fails with the aircraft on the ground and no engines running, a DOOR FAULT advisory message is displayed on EICAS.

If the engines are running, an OVERWING DOOR caution message is displayed if an OWEE door is open or has a fault.



LEGEND

DMC	Data Concentrator Unit Module Cabinet
LGSCU	Landing Gear and Steering Control Unit
OWEE	Overwing Emergency Exit
(blue arrow)	ARINC 429

Figure 47: Overwing Emergency Exit Doors Indicating System

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the overwing emergency exit doors.

CAS MESSAGES

Table 4: CAUTION Message

MESSAGE	LOGIC
OVERWING DOOR	L or R overwing doors that failed or open with at least one engine running.

Table 5: ADVISORY Messages

MESSAGE	LOGIC
DOOR OPEN	Any door open while on the ground and no engine running.
DOOR FAULT	Any electrical or mechanical failures of any door while on the ground and no engine running. Refer to INFO messages.

Table 6: INFO Messages

MESSAGE	LOGIC
52 DOOR FAULT - L OVERWING DOOR SNSR INOP	The DOOR FAULT L OWING SNSR INOP info message are associated with the generic DOOR FAULT CAS advisory whenever an electrical problem has been detected with one or both LHS overwing emergency exit door proximity sensors.
52 DOOR FAULT - R OVERWING DOOR SNSR INOP	The DOOR FAULT R OWING SNSR INOP info message are associated with the generic DOOR FAULT CAS advisory whenever an electrical problem has been detected with one or both RHS overwing emergency exit door proximity sensors.
52 DOOR FAULT - L OVERWING DOOR TRGT INOP	The DOOR FAULT L OWING TRGT INOP info message are associated with the generic DOOR FAULT CAS advisory whenever a mechanical problem (target FAR) has been detected with at least one or both of LHS overwing emergency exit door proximity sensors.

Table 6: INFO Messages

MESSAGE	LOGIC
52 DOOR FAULT - R OVERWING DOOR TRGT INOP	The DOOR FAULT R OWING TRGT INOP info message are associated with the generic DOOR FAULT CAS advisory whenever a mechanical problem (target FAR) has been detected with at least one or both of RHS overwing emergency exit door proximity sensors.

52-30 CARGO COMPARTMENT DOORS

GENERAL DESCRIPTION

The cargo compartment doors are opened using the external door handle. Lifting the handle unlocks and unlatches the door, and opens a vent panel that relieves any differential pressure that may exist.

The locked or unlocked status of the door is displayed by a visual indicator. A green flag is shown when the door is closed, latched, and locked. A red indication is displayed for any other conditions.

When the door handle is opened, the cargo door actuator is powered through a door open switch, which is operated from the control panel.

The actuator pushes on the door hinge to rotate the door outward and upward clear of the cargo compartment opening. Limit switches within the actuator shut off power when the door reaches the full open position.

When the door is closed, the actuator lowers the door to its initial unlocked and unlatched position. The handle is used to lock and latch the door. The vent flap closes and power is removed from the actuator when the handle is in the stowed position.

The door seals installed around the door, ensure positive pressurization of the aircraft. Each seal has inlet air holes for inflation. An aerodynamic seal, installed on the top of the door, covers the gap between the fuselage and the door.

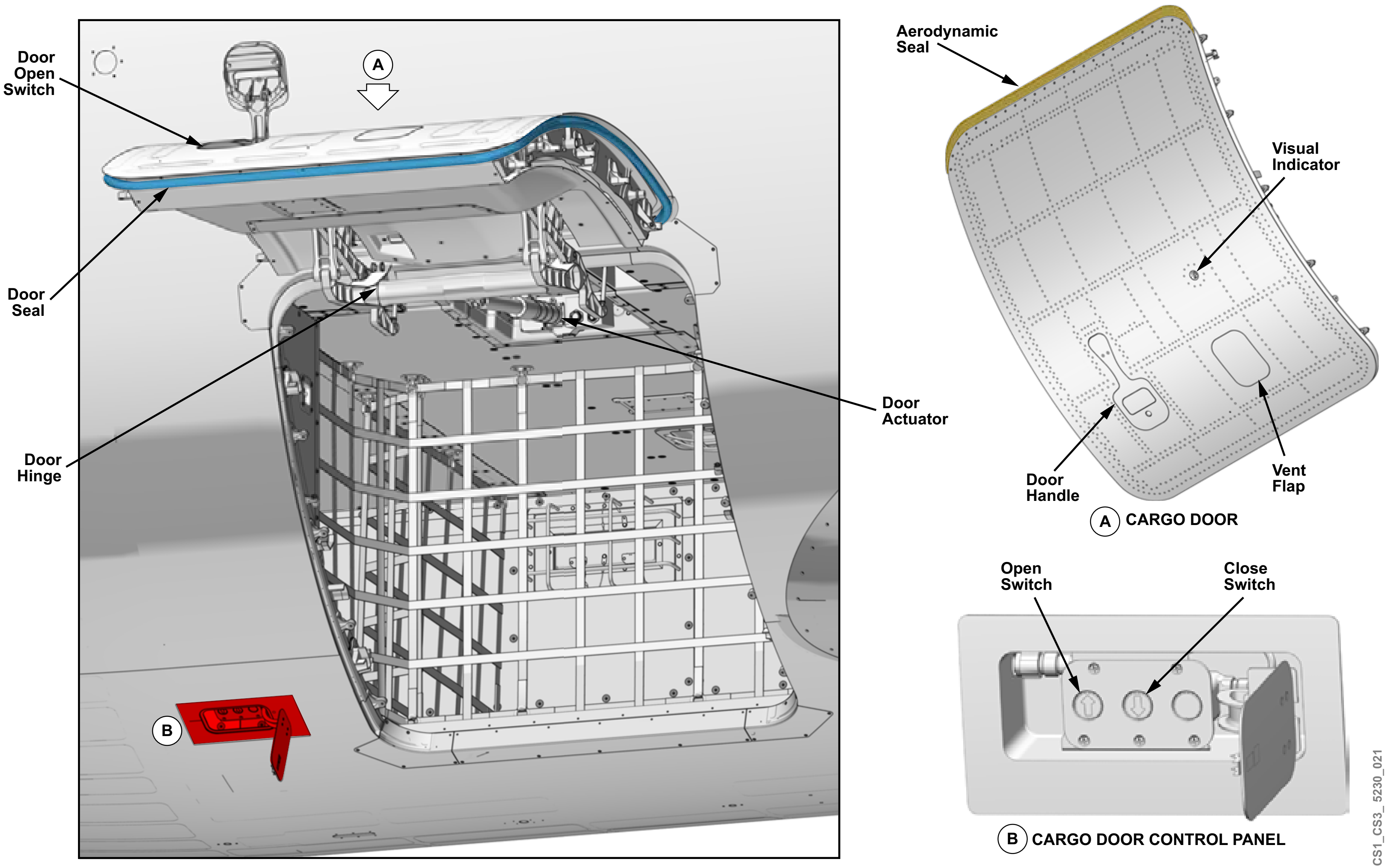


Figure 48: Cargo Compartment Doors

CARGO DOOR CONTROL

When the cargo door handle is operated, the door open switch is actuated and the cargo door actuator is powered from its respective BATT DIR BUS. The forward cargo door is powered from BATT DIR BUS 1, and the aft cargo door is powered from BATT DIR BUS 2.

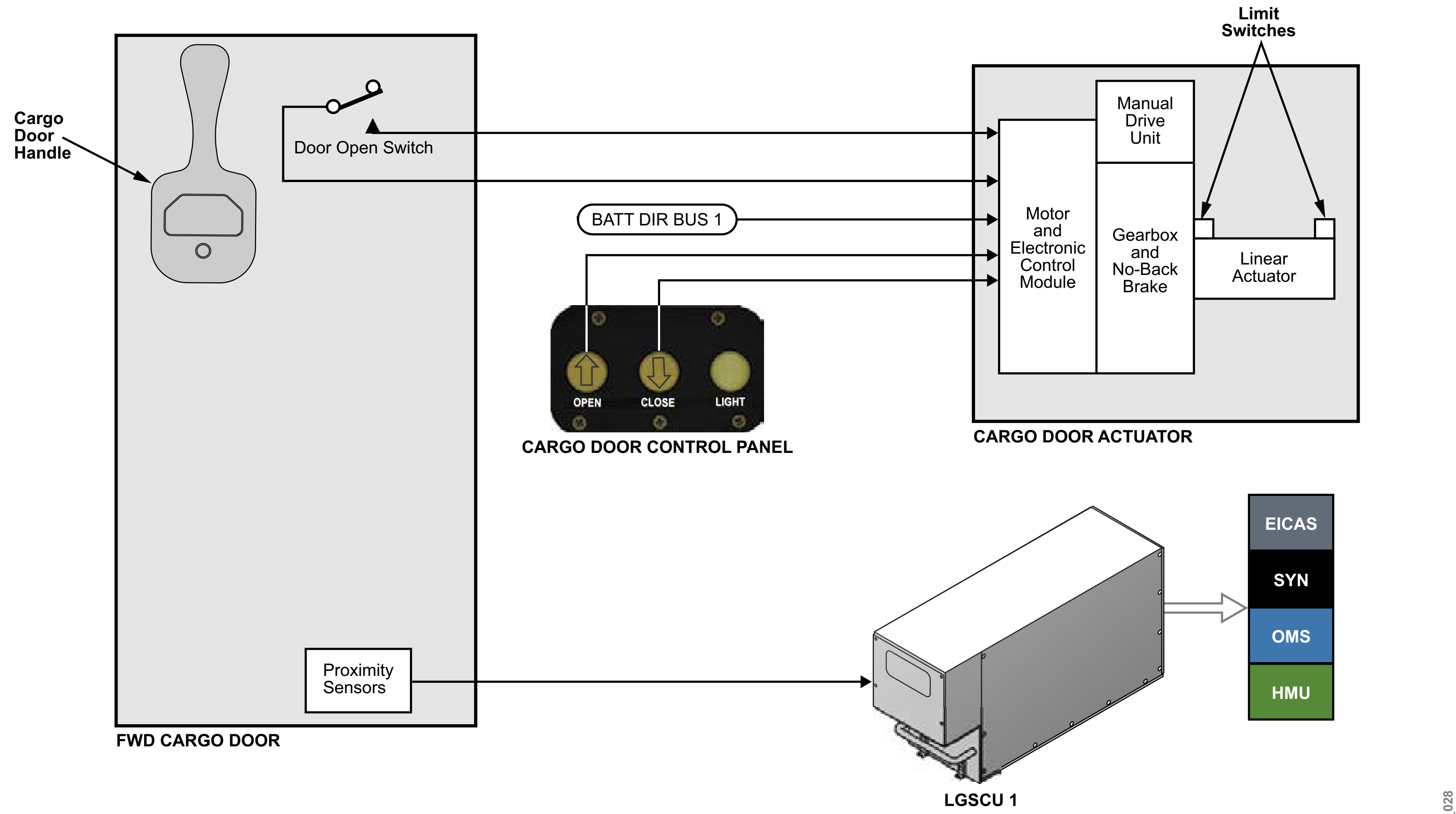
The cargo door actuator has an electronic control module that controls a DC motor. It controls the motor speed and direction based on the control panel OPEN and CLOSE switch inputs.

The gearbox drives the linear actuator through a no-back brake. The no-back brake engages and holds the door in its existing position when power is removed from the actuator. The linear actuator is monitored by limit switches that signal the electronic controller when the actuator reaches its limit of travel in either the open, or closed direction.

The actuator has a manual drive unit that allows the cargo door to be opened in the event an actuator failure or power loss. The manual drive unit has a standard 3/8 in. drive and built-in overload protection.

The electronic controller has built-in test (BIT) and fault diagnostic capabilities. The electronic controller monitors the motor for underspeed, overcurrent, undervoltage, and power supply faults. Any of these conditions inhibit motor operation.

The cargo door position is monitored by proximity sensors. The proximity sensors provide position information to the landing gear and steering control unit (LGSCU) for display on the engine indication and crew alerting system (EICAS).



LEGEND

LGSCU Landing Gear and Steering Control Unit

NOTE

FWD cargo door shown, AFT similar.
AFT cargo door is powered by BATT DIR BUS 2.

Figure 49: Cargo Door Control

CARGO DOOR MECHANISMS

The cargo door has internal mechanisms that control the sequencing of the locking, unlatching, and lifting of the door.

Lock Mechanism

The lock mechanism provides the main control of the door. The external handle is connected directly to the lock shaft. When the external handle is operated, the lock shaft rotates to initiate the following sequence:

- Vent flap opens
- Latch shaft unlocks
- Door lift mechanism begins to lift the door

Latch Mechanism

When the lock shaft unlocks, the latch operating cam rotates the latch shaft. At the same time, the lift mechanism begins to lift the door. The movement of the door is controlled by the latch cranks, located at each end of the latch shaft.

Lift Mechanism

When the external handle has reached full travel, the door is ready to be lifted open. An actuator, attached between the hinge arm and the fuselage structure, lifts the door. The geometry of the hinge arm ensures that the door is clear of the cargo compartment opening.

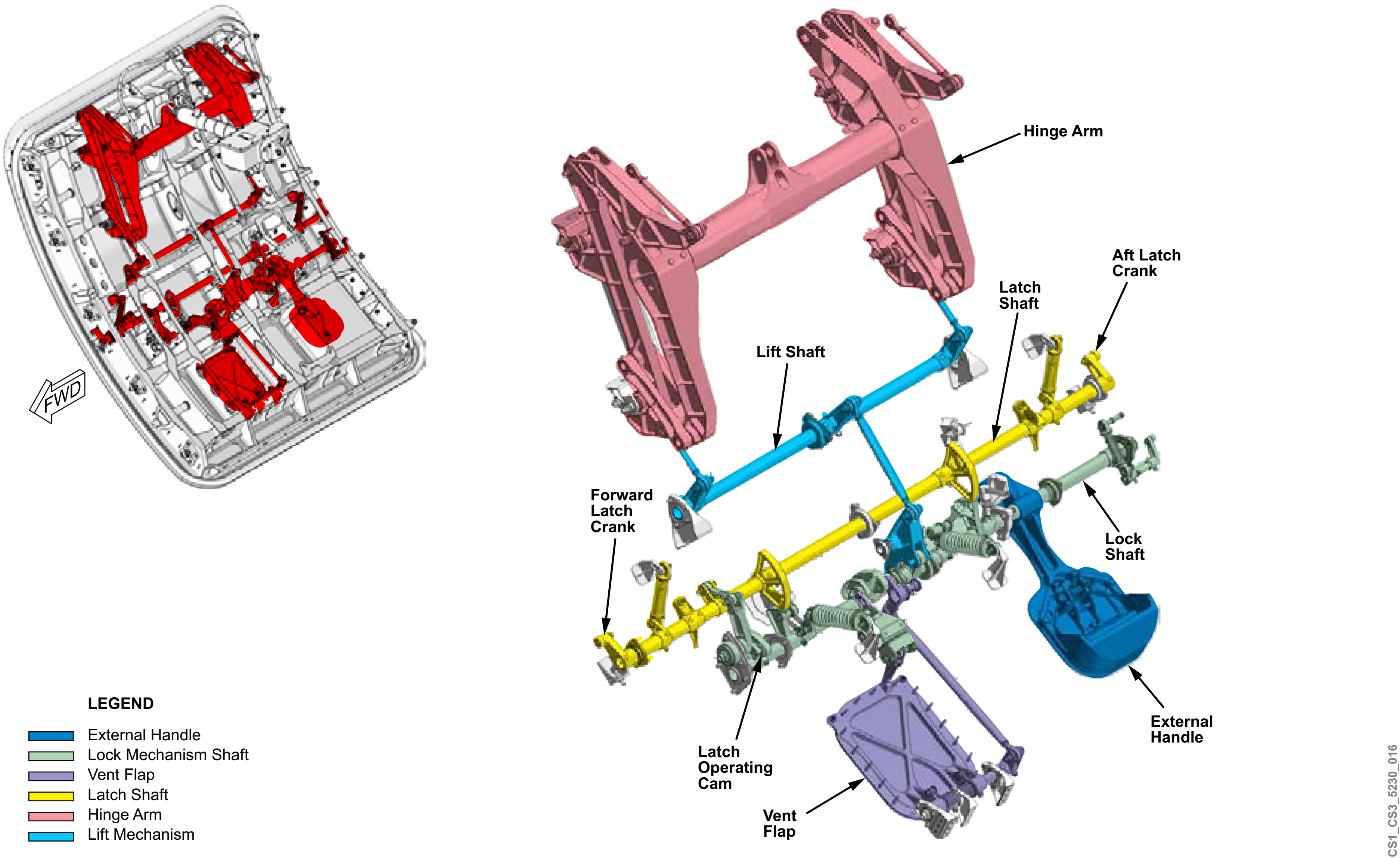


Figure 50: Cargo Compartment Door Mechanisms

COMPONENT LOCATION

The following components are located on the cargo door:

- Closed sensor target
- Stop fittings
- Door roller fittings
- Actuator
- Door open switch

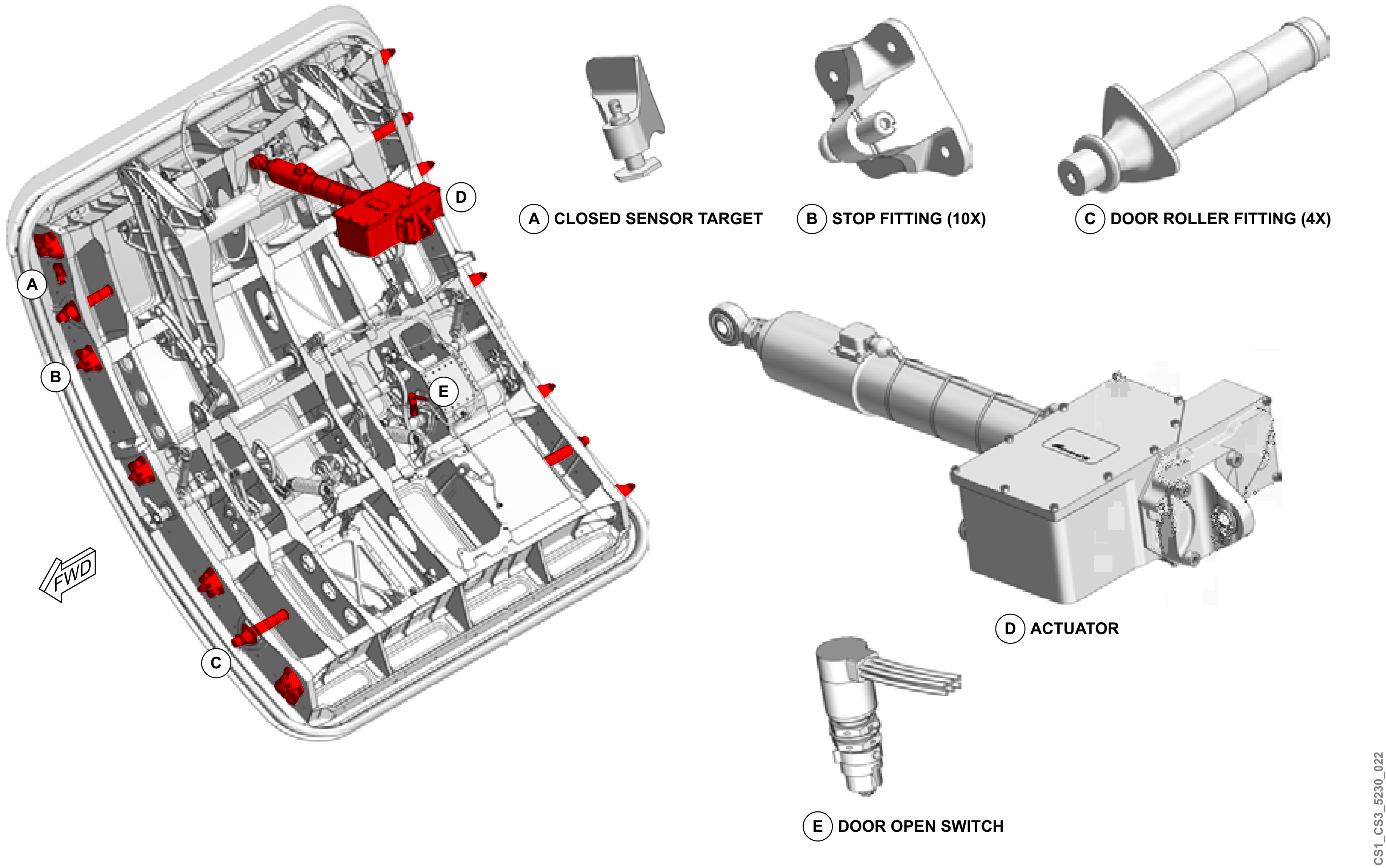


Figure 51: Cargo Compartment Door Component Location

DETAILED COMPONENT INFORMATION

LOCK MECHANISM

The lock mechanism ensures that the cargo door does not become unlatched. The lock mechanism consists of the lock shaft, the forward lock, and the aft lock. The lock shaft also has a lift cam, ditching cam and a latch cam to control the sequencing of the cargo door.

The lock shaft is rotated when the external handle is operated. The external handle, consisting of the external handle body and a the spring-loaded external handle flap, is attached directly to the lock shaft.

When the door is operated, the lock shaft is the primary mechanism for controlling the door operation. It is not possible to engage the locks in the locked position if the latches are not latched. The lock system prevents any rotation of the latch shaft when it is latched.

The ditching cam moves the ditching stop clear of the vent flap through the ditching idler. The vent cam operates the vent flap to relieve any excess differential pressure. If the vent flap fails to close, the aircraft can not be pressurized.

The lock shaft has a forward lock and an aft lock that interact with the latch shaft forward counterlock and aft counterlock. The aft lock is monitored by the lock proximity sensor. The aft lock also has a cam for the door open switch. The forward lock is monitored by the visual indicator through the visual indicator idler installed on the forward lock body.

A visual indicator is used to confirm the door is closed. The visual indicator has a spring-loaded idler that pivots around an axis on the forward lock. A green mechanical flag is installed on the idler. The idler is controlled by a cam on the forward counterlock. When the forward lock and forward counterlock are in the door closed position, the green mechanical flag is displayed. When the lock shaft moves towards open, the green mechanical flag moves away from the visual indicator and a red indicator is displayed.

A lift cam provides timing for the initial lift of the door. When the external handle is operated, the lifting cam provides the initial lift of the door.

The latch cam operates through the latch idler. When the vent flap is open, the latch idler rotates the latch shaft.

Two lock-securing means springs secure the lock shaft in the locked position.

An interlock mechanism prevents the operation of the door handle when the door is clear of the door frame.

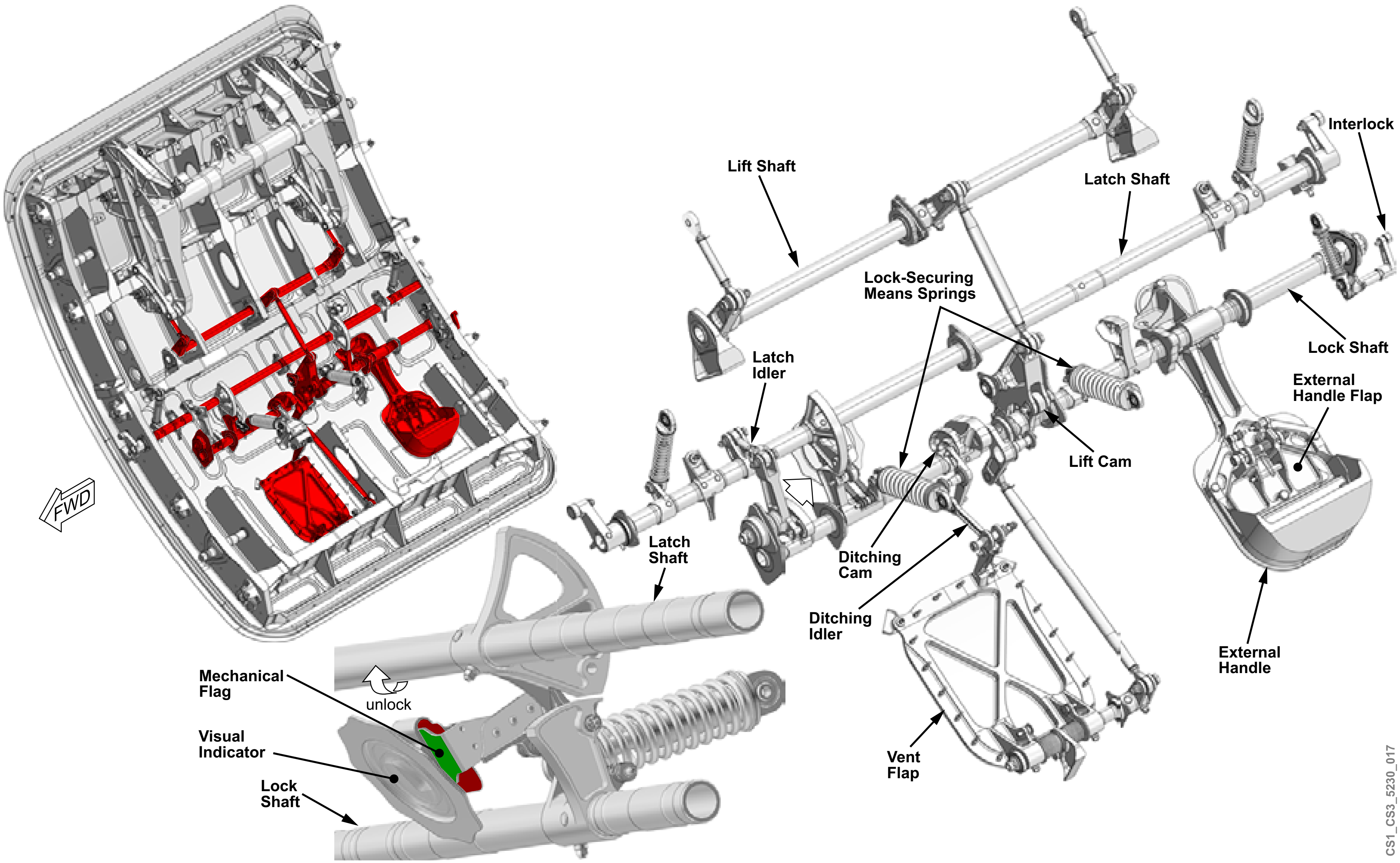


Figure 52: Lock Mechanism

INTERLOCK MECHANISM

The interlock mechanism prevents the cargo door handle from moving when the door has moved away from the fuselage. This prevents the door mechanisms from going out of sequence.

The interlock mechanism consists of an interlock crank, interlock lever, and an interlock spring installed on a common shaft.

The interlock mechanism is spring-loaded toward the locked position, and is activated by the door moving away from the surround structure in the fuselage. As the door rotates open, the interlock stop engages a stop on the lock shaft, preventing movement of the shaft.

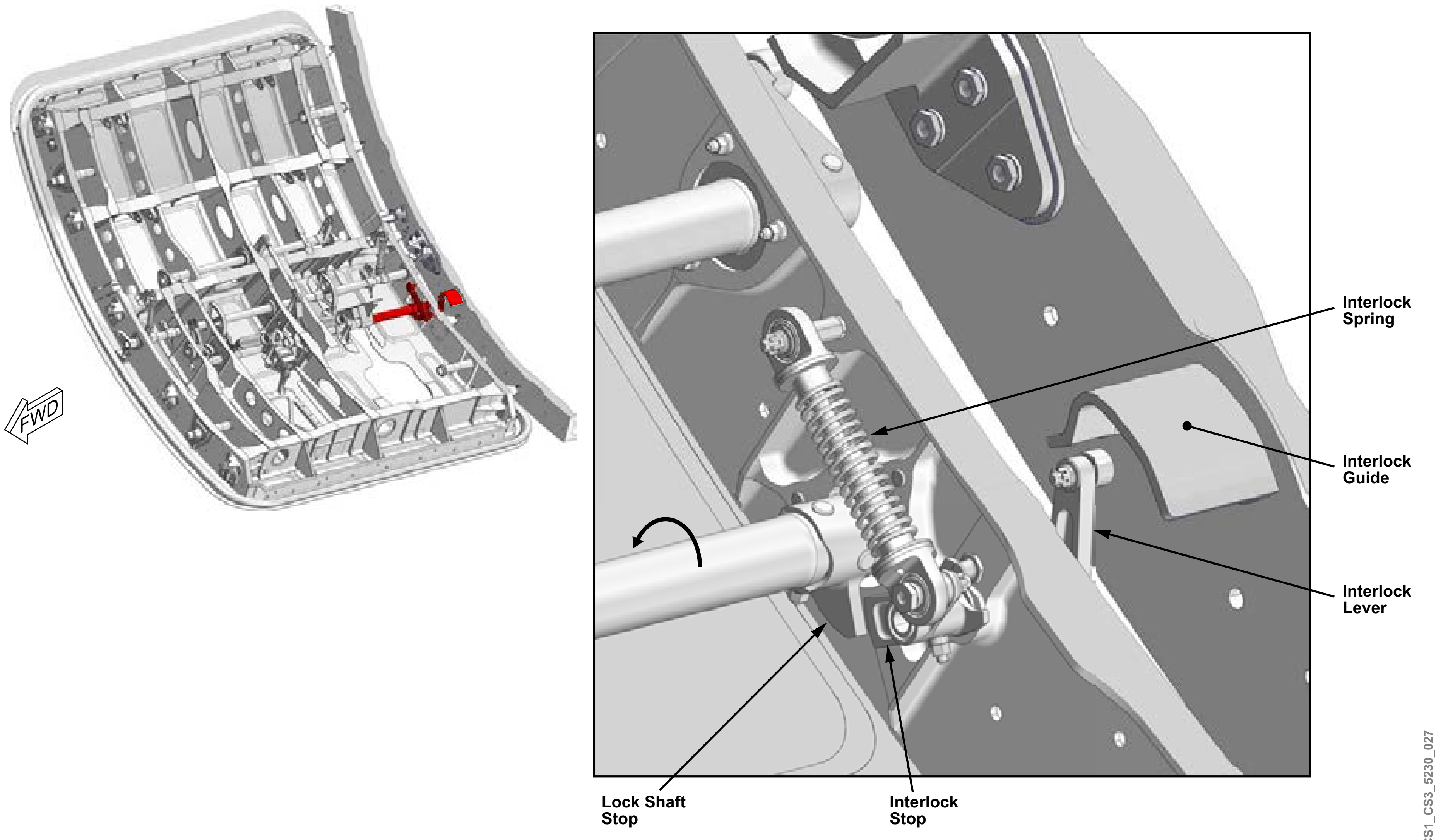


Figure 53: Interlock Mechanism

LATCH MECHANISM

The latch mechanism ensures the cargo door remains in the latched condition. The latch mechanism consists of the latch shaft, forward counterlock, aft counterlock, forward latch crank, and aft latch crank.

The latch shaft operates through the latch idler driven by the lock shaft. It consists of two counterlocks that work with the lock shaft locks to ensure the door remains locked when it is closed. The aft latch counterlock position is monitored by a proximity sensor. The forward counterlock position is monitored by the visual indicator on the door. When the indicator is green, the door is locked and latched. Under any other condition, the visual indicator is red.

Latch cranks, installed at each end of the latch shaft, mate with the latch guide fittings installed on the cargo door surround structure. The latch function is provided by the 15° overlatched position of the latch crank in the fully closed configuration. Each latch crank has a ditching stop to prevent the door from unlocking in a water landing.

The travel of the latch shaft in the fully closed configuration is controlled by adjustable stops. The stops contact a corresponding stop on the cargo door structure.

Two latch-securing means springs maintain the latch shaft in either the latched, or unlatched position.

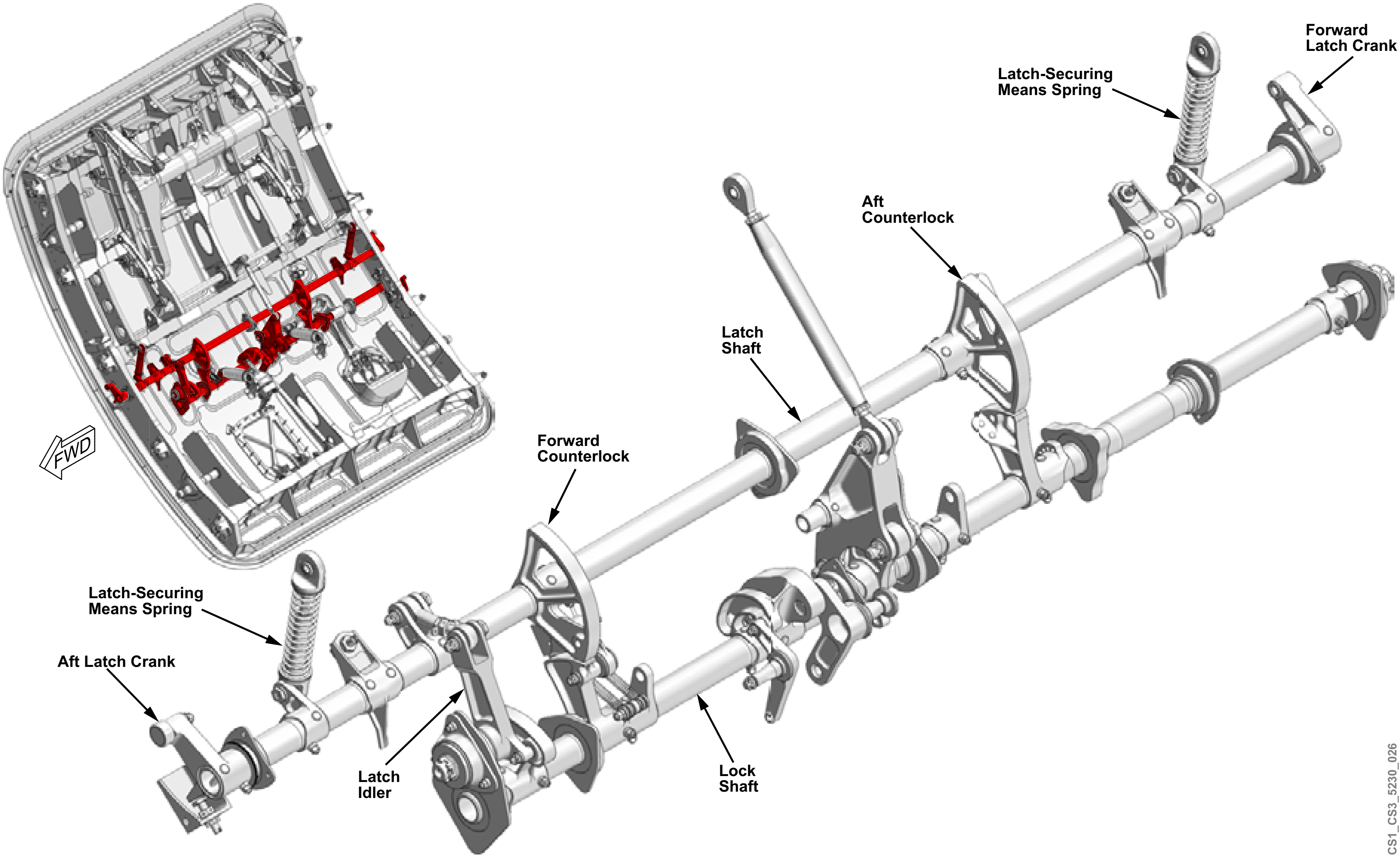


Figure 54: Latch Mechanism

LIFT MECHANISM

The lift mechanism raises the door enough for the door stop fittings to clear the fuselage stop pads. The door lift is controlled by the lift cam mounted on the lock shaft. The lift cam acts on the lift idler to raise the door following a path defined by the fuselage guide fittings and door guide rollers. As the lift cam rotates, the lift idler pulls the main lift rod. This rotates the lift shaft and pulls the two lift rods installed at each end, allowing the door to lift.

When the door is lifted, the lift cam supports the door in this position and prevents the door from closing. The door actuator mounts between the fuselage structure and the hinge tube of the hinge mechanism. The actuator pushes on the hinge tube extending, it outward and upwards

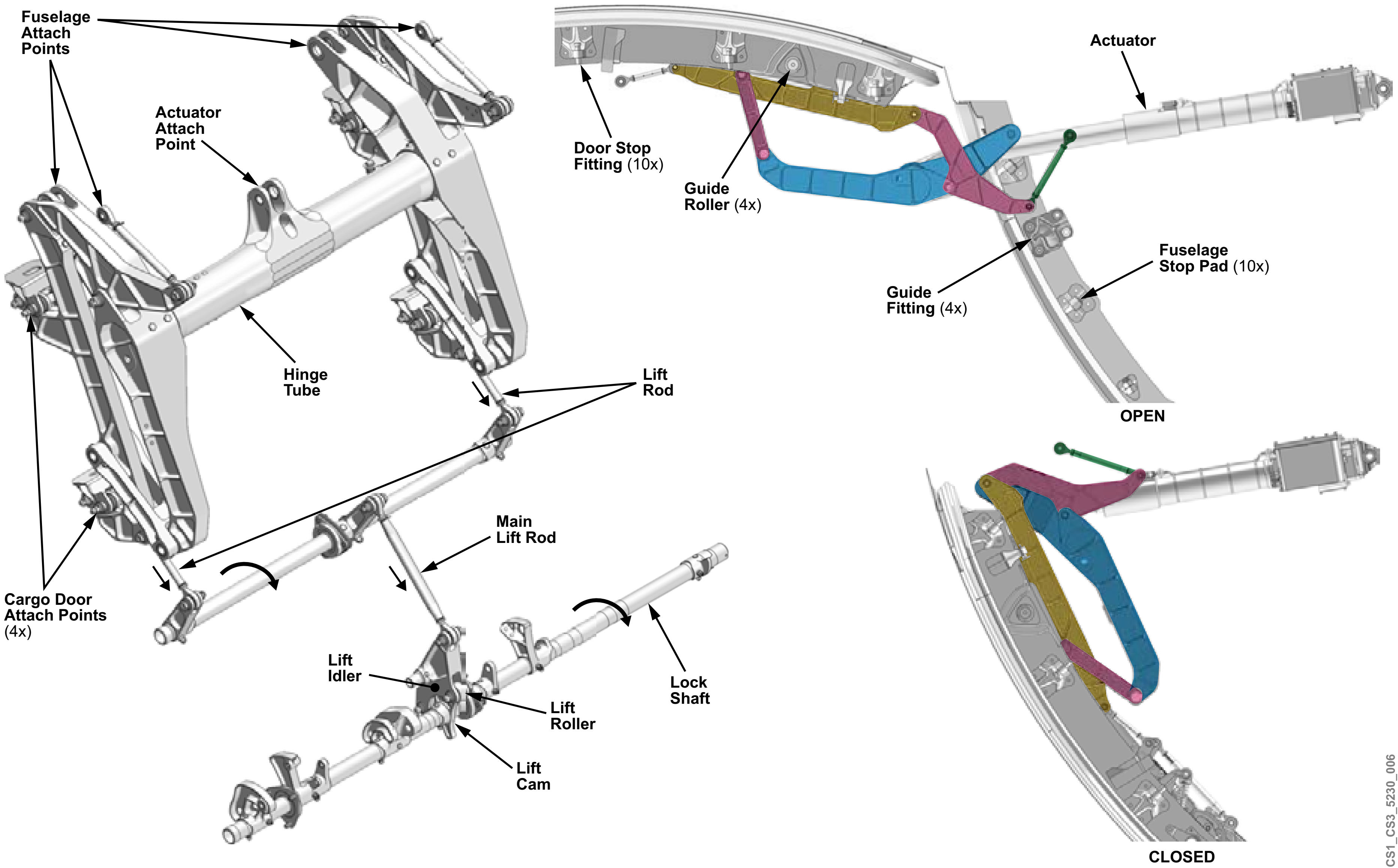


Figure 55: Lifting Mechanism and Cargo Door Hinge Mechanism

CARGO DOOR FRAME

Fuselage Stop Fittings

Each cargo door has ten door stop fittings that contact the fuselage stop fittings when the door is closed. The fuselage stop fittings carry the pressurization loads into the fuselage structure.

Guide Fittings

The guide fittings are mounted on the door surround structure to provide a lift path for the door to follow. The upper guide fittings and lower guide fittings each have a unique lift path and are not interchangeable. The door guide rollers follow the path provided by the guide fittings in order to direct the door open or closed.

Latch Crank Guide Fittings

Two latch crank guide fittings provide the latch crank locking when the door is closed.

Interlock Guide

The interlock guide fitting is installed on the aft door surround only, and prevents the external handle from being operated before the cargo door is in the door frame.

Door Centering Guide Rollers

There are two door centering guide rollers mounted on the door surround structure that contact the door mounted centering guide fittings. The centering guides ensure the door does not tilt as it is closing.

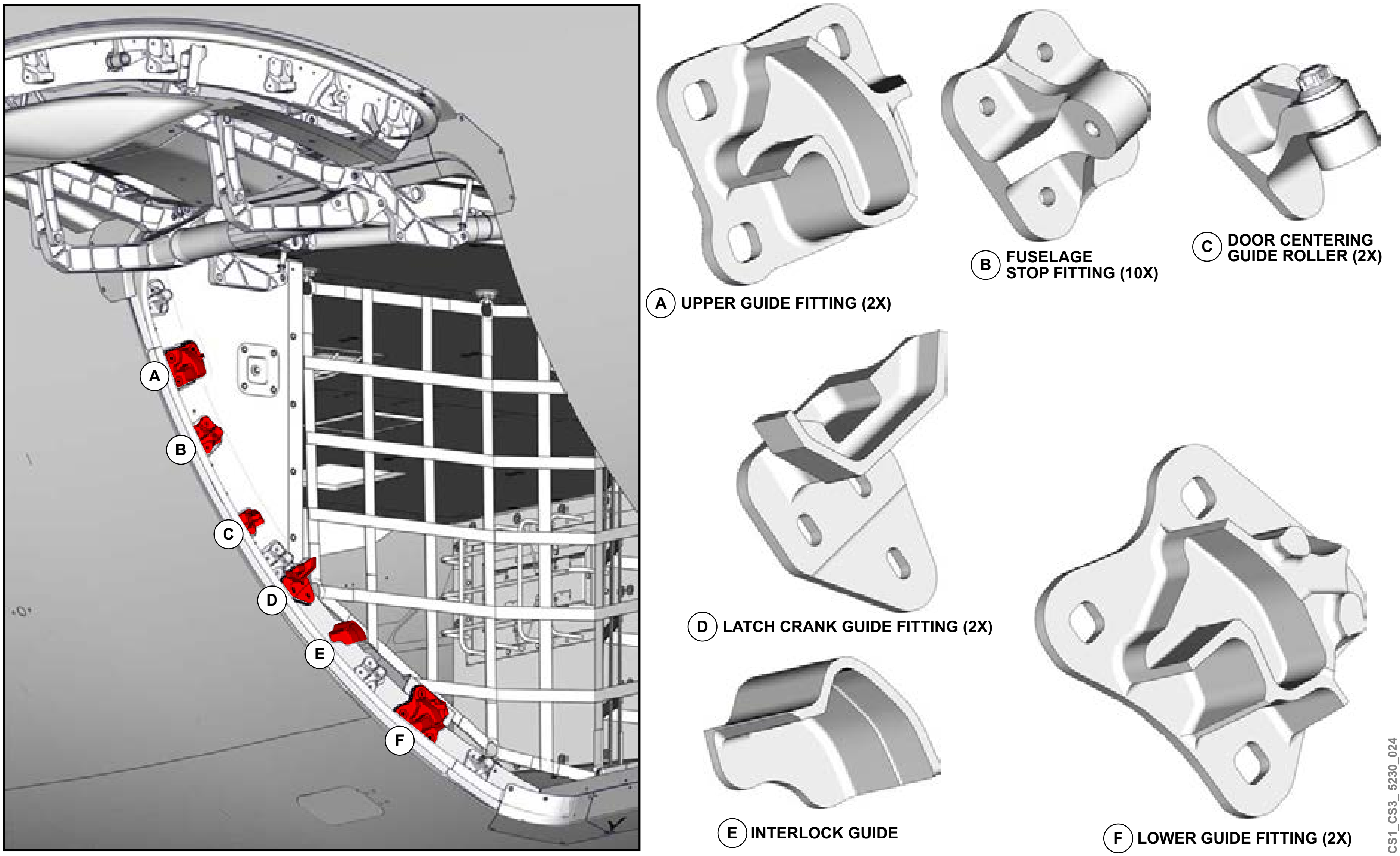


Figure 56: Cargo Door Frame

CONTROLS AND INDICATIONS

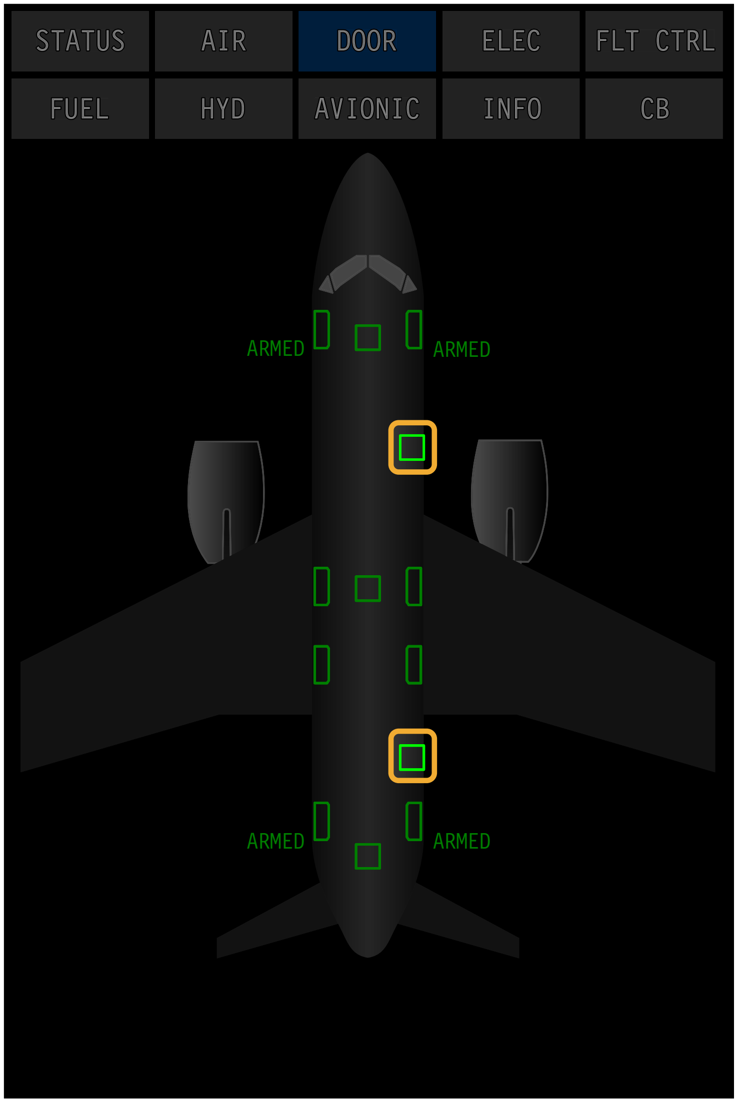
Each cargo door has a separate control panel, located aft of its respective door. Each panel has an open and close switch and a light switch for the loading area light.

CS1_CS3_5230_007

Figure 57: Cargo Compartment Doors Controls

SYNOPTIC PAGE

The status of the cargo doors are shown on the DOOR synoptic page.



SYNOPTIC PAGE – DOOR

DOOR STATUS	
Symbol	**Condition**
FWD CARGO	At least 1 engine running and door not closed, or not latched or not locked. **OR** Door fault. Note: Label of door is shown in amber.
	Door closed,lactched, locked. Note: Label of door is not shown.
FWD CARGO	Door open on ground, engine not running. Note: Label of door is shown in white.
FWD CARGO	Invalid or failed LGSCU or invalid door sensor. Note: Label of door is shown in amber.

CS1_CS3_5230_020

Figure 58: Cargo Door Indications

OPERATION

OPENING THE CARGO DOOR

Open the cargo door as follows:

1. Push the flap to grasp the external handle.
2. Pull the handle up to unlock the door.
3. Ensure the vent flap is open.
4. On the cargo door, make sure that the visual indicator is red.
5. On the control panel, press the open button to raise the cargo door. The cargo door automatically stops when the door reaches the full limit of travel.

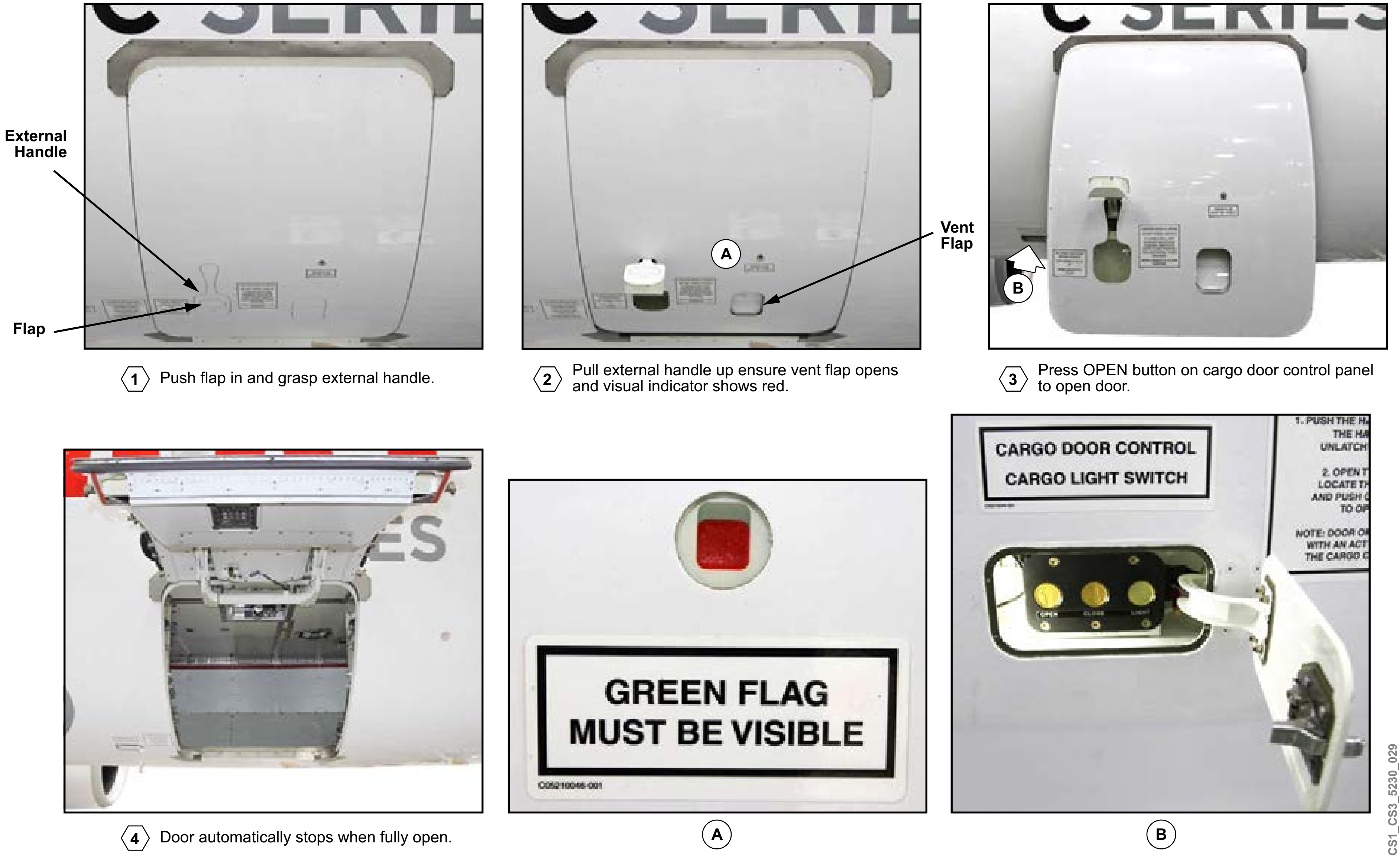


Figure 59: Cargo Compartment Door Opening

CLOSING THE CARGO DOOR

Close the cargo compartment door as follows:

1. On the control panel, press the close button to lower the cargo door.
2. When the door reaches the limit of travel, release the switch.
3. Pull the external handle down into its housing to lock the door
4. On the cargo door, make sure that the visual indicator is green
5. Make sure that the vent flap and the external handle are flush with the cargo door.

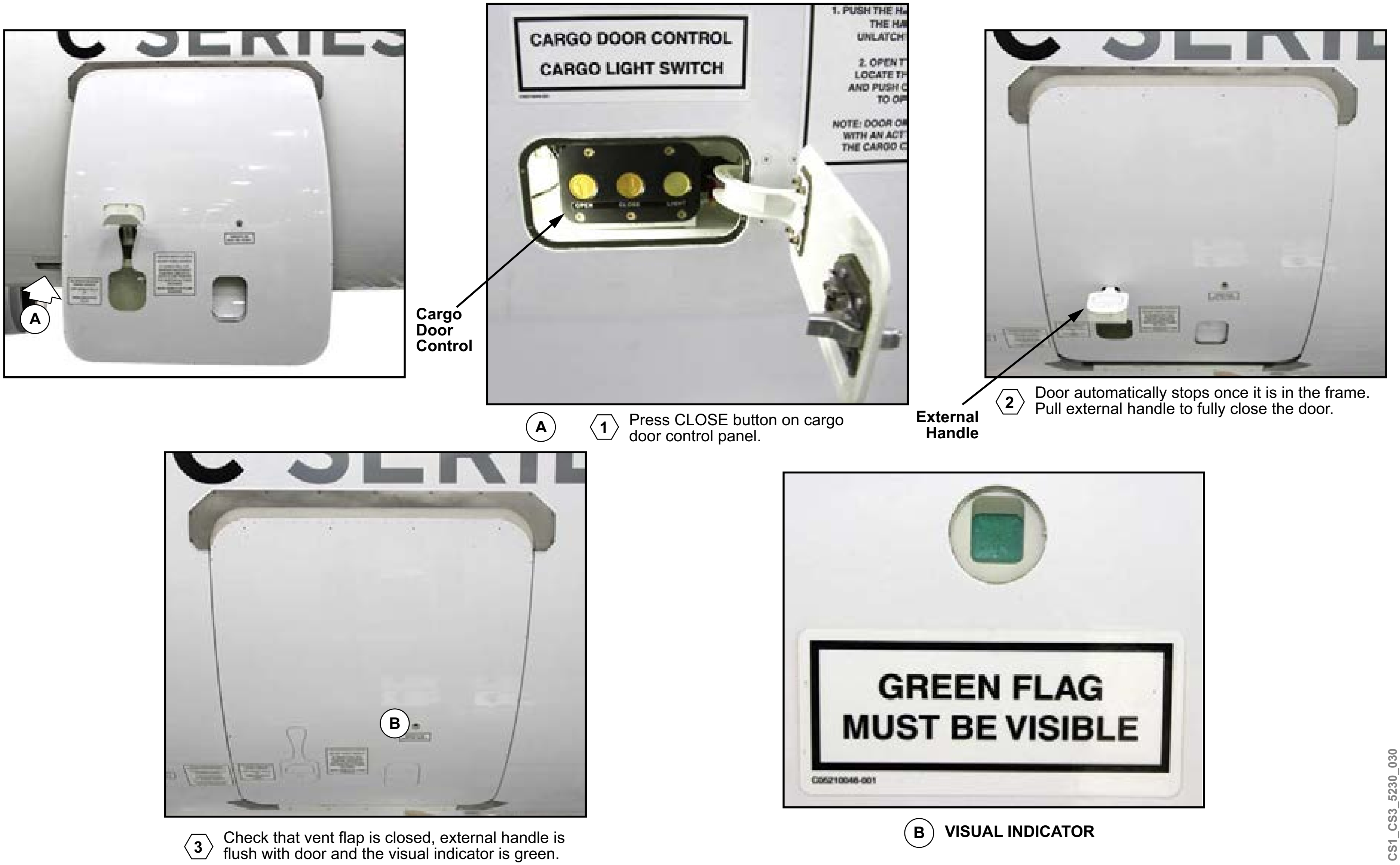


Figure 60: Cargo Compartment Door Closing

DETAILED DESCRIPTION

PROXIMITY SENSORS

Each cargo compartment door has three proximity sensors used to monitor the closed, latched, and locked states of the door.

Closed Sensor

The closed sensor is installed on the fuselage door frame, and the target is installed on the door structure.

Latch Sensor

The latch sensor is mounted on the door structure and monitors the aft counterlock. The latch sensor target is fixed to the latch shaft.

Lock Sensor

The lock sensor is installed the door structure. The target is mounted on the aft lock.

DOOR OPEN SWITCH

The door open switch is activated when the door handle is lifted. A cam on the aft lock engages the switch as the handle is lifted. Lifting the door handle rotates the lock shaft, and activates the switch. The open switch provides power to the door actuator.

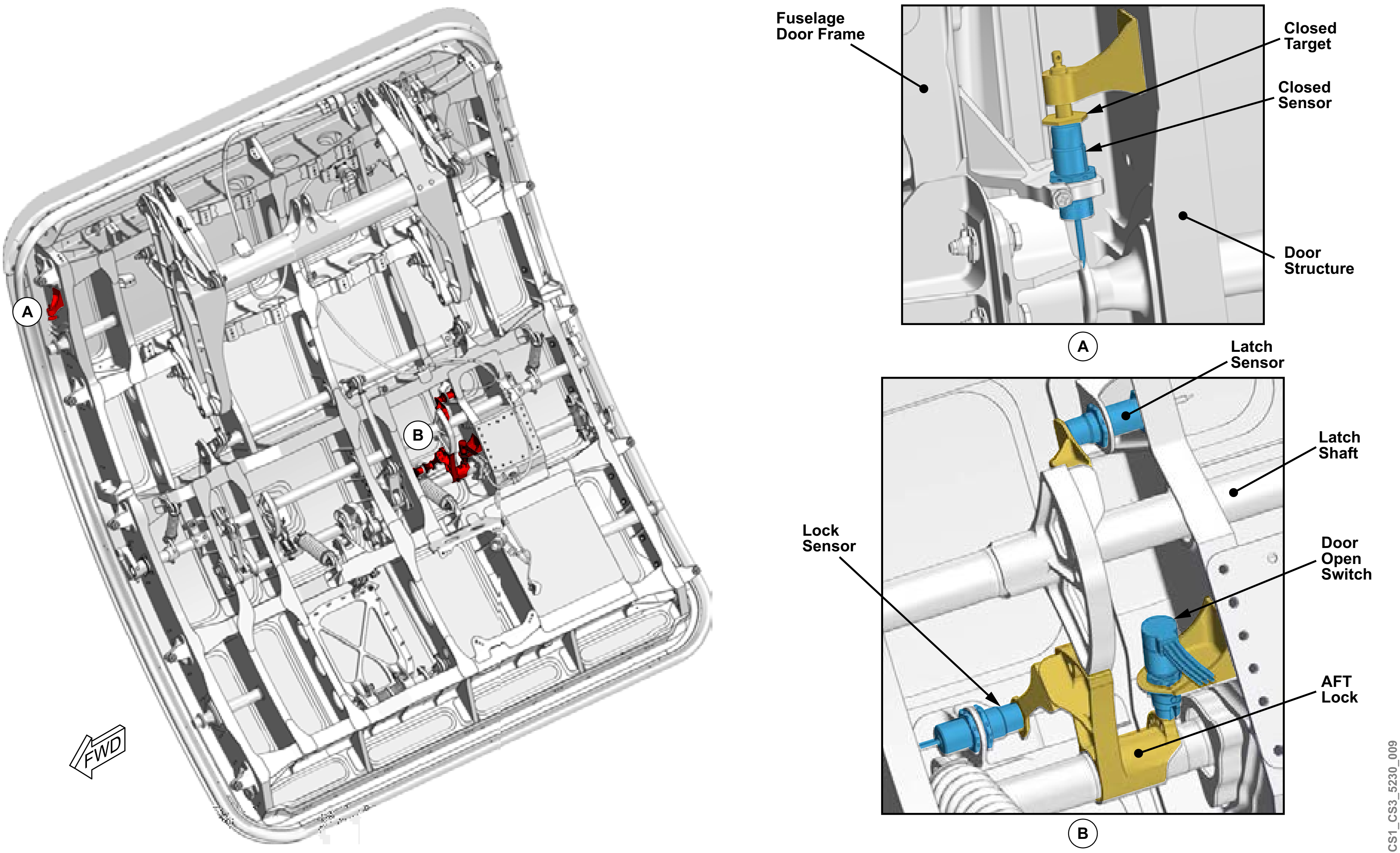


Figure 61: Proximity Sensors and Door Open Switches

CARGO DOOR POSITION INDICATION

The closed, latch, and lock proximity sensors monitor the cargo door position. The proximity sensors for the forward cargo door are connected to landing gear and steering control unit (LGSCU) 1. The aft cargo door proximity sensors are connected to LGSCU 2. The door indications are displayed on the DOOR synoptic page. The status of the sensors is monitored by the onboard maintenance system (OMS).

If a cargo door sensor or target fails with the aircraft on the ground and no engines running, a DOOR FAULT advisory message is displayed on EICAS.

If the engines are running, a CARGO DOOR caution message is displayed when either cargo door is open or has a fault.

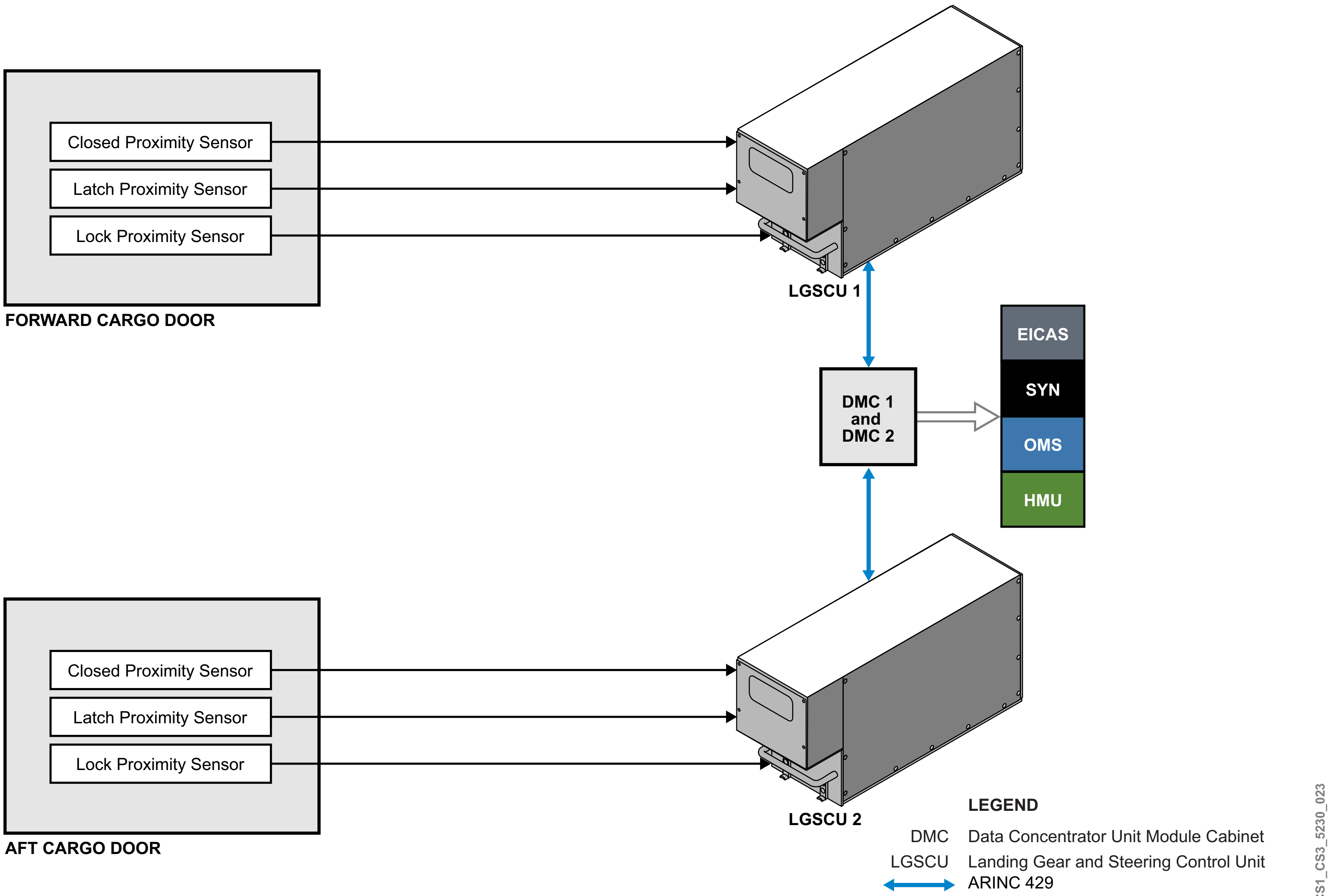


Figure 62: Cargo Door Position Indication

MONITORING AND TESTS

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

CAS MESSAGES

Table 7: CAUTION Message

MESSAGE	LOGIC
CARGO DOOR	FWD or AFT cargo door failed or open with at least one engine running.

Table 8: ADVISORY Messages

MESSAGE	LOGIC
DOOR OPEN	Any door open while on the ground and no engine running. Refer to INFO messages.
DOOR FAULT	Any electrical or mechanical failures of any door while on the ground and no engine running. Refer to INFO messages.

Table 9: INFO Messages

MESSAGE	LOGIC
52 DOOR FAULT - FWD CARGO DOOR SNSR INOP	Message is associated with the generic DOOR FAULT CAS advisory whenever an electrical problem has been detected with at least one (up to three) of the FWD cargo door proximity sensors.
52 DOOR FAULT - AFT CARGO DOOR SNSR INOP	Message is associated with the generic DOOR FAULT CAS advisory whenever an electrical problem has been detected with at least one (up to three) of the AFT cargo door proximity sensors.
52 DOOR FAULT - FWD CARGO DOOR TRGT INOP	Message is associated with the generic DOOR FAULT CAS advisory whenever a mechanical problem (target FAR) has been detected with at least one (up to three) of the FWD cargo compartment door proximity sensors.
52 DOOR FAULT - AFT CARGO DOOR TRGT INOP	Message is associated with the generic DOOR FAULT CAS advisory whenever a mechanical problem (target FAR) has been detected with at least one (up to three) of the AFT cargo compartment door proximity sensors.

PRACTICAL ASPECTS

CARGO DOOR ACTUATOR ELECTRICAL FAILURE

Open the Cargo Door

Open the cargo door as follows:

1. Unlock the door by pulling the handle.
2. Make sure the cargo door visual indicator is red.
3. Remove the cargo liner access panel located behind the vent flap.
4. Put the override tool in position through the opening of the cargo door vent flap and the liner access panel, and engage the square end of the tool to the manual drive.
5. Turn the tool handle until the cargo door is open enough to get inside the cargo compartment.
6. Remove the tool.
7. Go into the cargo compartment.
8. Turn the manual drive in the clockwise direction using a 3/8 in. standard drive to open the cargo door.

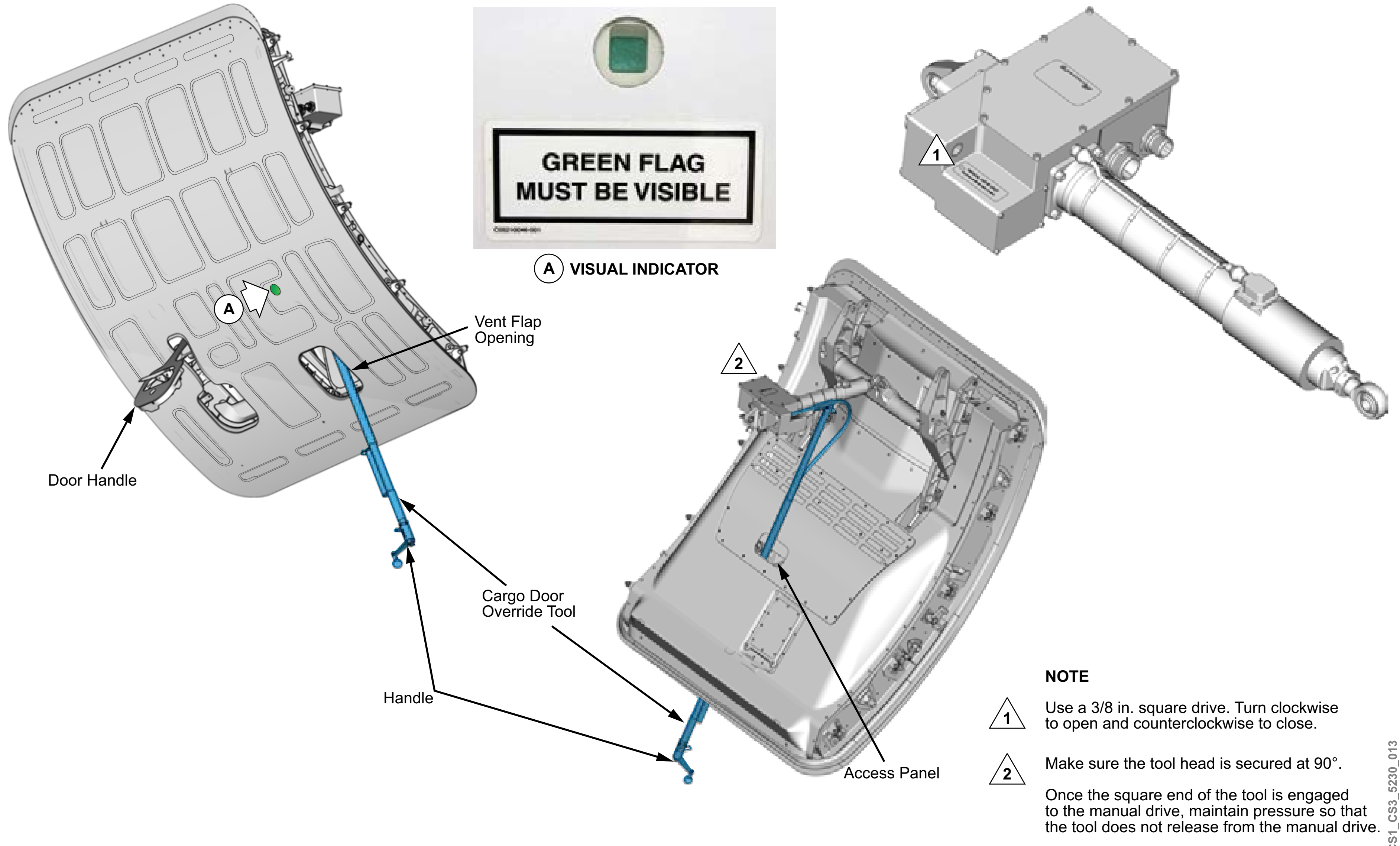
Close the Cargo Door

Close the cargo door as follows:

1. Go into the cargo compartment.
2. Turn the manual drive in the counterclockwise direction using a 3/8 in. standard drive until the cargo door is open enough to get out of the cargo compartment.
3. Get out of the cargo compartment.
4. Put the tool in position through the opening of the cargo door vent flap and the liner access panel.
5. Turn the tool handle in counter-clockwise direction to close the cargo door.
6. Remove the tool.
7. Install the liner access panel through the cargo door vent flap.
8. Pull down the handle down to lock the cargo door.
9. On the cargo door, make sure that the visual indicator is green.
10. Make sure that the flap and the handle are flush with the cargo door.

NOTE

1. The screw at the pivot point of the tool must be tightened to keep the tool head at 90°.
2. Once the square end of the tool is engaged to the manual drive, maintain pressure so that the tool does not release from the manual drive.
3. The override tool handle can be removed and an electric drill can be connected to the tool shaft. Do not turn the shaft at a speed more than 200 rpm.
4. Decrease the speed of the electric drill when you get near the full open or full closed position.



NOTE

1. Use a 3/8 in. square drive. Turn clockwise to open and counterclockwise to close.

2. Make sure the tool head is secured at 90°.

   Once the square end of the tool is engaged to the manual drive, maintain pressure so that the tool does not release from the manual drive.

Figure 63: Cargo Door Actuator Electrical Failure

CARGO DOOR ACTUATOR MECHANICAL FAILURE

Open the Cargo Door

Open the cargo door as follows:

1. Unlock the door by pulling the handle.
2. Make sure the cargo door visual indicator is red.
3. Disconnect the actuator from the cargo door hinge.
4. Lift the door up to allow entry into the cargo compartment.
5. Remove the actuator from the actuator fitting.
6. Install the dummy actuator in the actuator fitting and secure with a quick release pin.
7. Adjust the dummy actuator length by turning the hand wheel until the dummy actuator aligns with the hinge fitting and secure with the quick disconnect pin.

Close the Cargo Door

Close the cargo door as follows:

1. Bring the door close to the door cut out using the dummy actuator.
2. Support the door in this position.
3. Disconnect the dummy actuator from the door hinge fitting and actuator fitting.
4. Slowly push the door until all the rollers are engaged in their respective guide fittings.
5. Pull down the handle in its housing to lock the cargo door.
6. Make sure that the vent flap and the handle are flush with the cargo door.

NOTE

1. The door needs to be opened a minimum of 25.4 cm (10 in.) to install the dummy actuator.
2. With no actuator installed, the door is free to move once unlocked and unlatched. The door must be supported until it is in the vertical position or injury and damage to aircraft can occur.

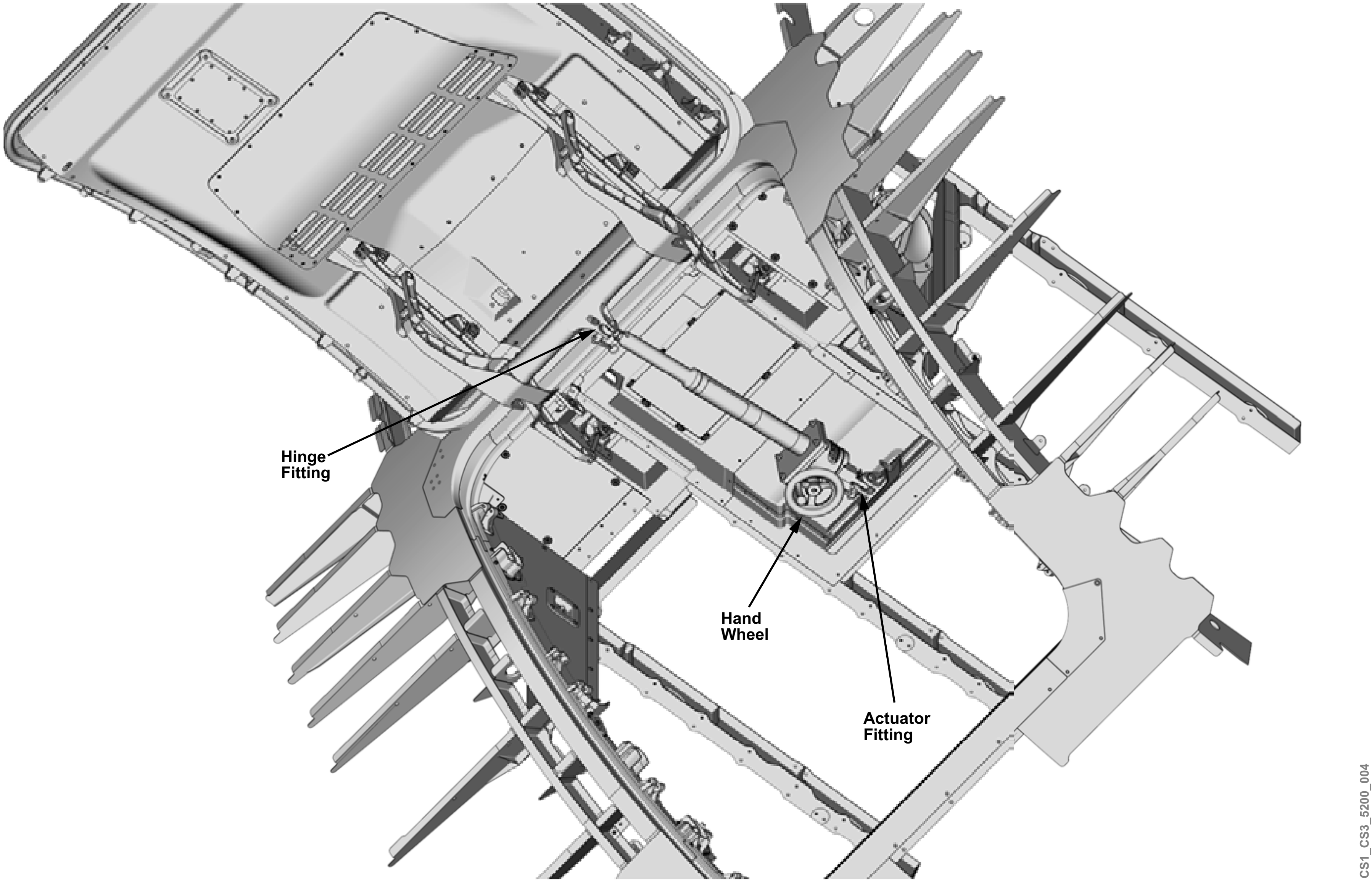


Figure 64: Cargo Door Actuator Mechanical Failure

52-40 EQUIPMENT BAY DOORS

GENERAL DESCRIPTION

The equipment bay doors consist of:

- Fwd equipment bay door
- Mid equipment bay door
- Aft equipment bay door

FORWARD EQUIPMENT BAY DOOR

The forward equipment bay door is a plug-type sliding door that opens upward then slides rearward into the forward equipment bay. When the door is fully rearward, the door is stowed under the aft equipment rack. A pressure seal maintains cabin pressurization. Four stop fittings transmit pressurization loads to the fuselage.

The door includes lift, latch, and hook mechanisms operated by an external handle. The latch mechanism has two latch pins that latch the door with the fuselage. A press trigger releases the handle from the stowed position and the handle ejects from the door recess. The external handle is rotated 90° counterclockwise to unlatch the door. The forward and aft latch pins retract into the door from the door frame latch fittings.

The lift mechanism provides the upward and rearward motion of the door during opening. Four lift fittings mounted on the door rotate 90° and raise the door. Each lift fitting has rollers installed in rails that guide the door aft to the stowed position. Four weight compensation springs, located at each lift fitting, assist in door opening and closing. A hook maintains the door in the up position.

Once the door is in the up position, the handle must be returned to the closed position and stowed in the door recess so that the door can slide aft. If the handle is not pushed into the door recess, a lock mechanism engages the rail to prevent the door from sliding aft.

A proximity sensor mounted in the door surround structure monitors the forward latch position and provides door indication to the engine indication and crew alerting system (EICAS).

Door Opening

The forward equipment bay door can only be operated from the outside of the aircraft.

To open the door:

1. Push the trigger to release the handle.
2. Turn the handle counterclockwise to release the door.
3. Push up on the door until it locks in place.
4. Turn the handle clockwise and then push the handle to stow it.
5. Slide the door aft on the rails.

Door Closing

To close the door:

1. Slide the door forward on the rails.
2. Push the trigger to release the handle and turn the handle counterclockwise.
3. Pull the door down and turn the handle in the clockwise direction.
4. Push the handle up to stow it.

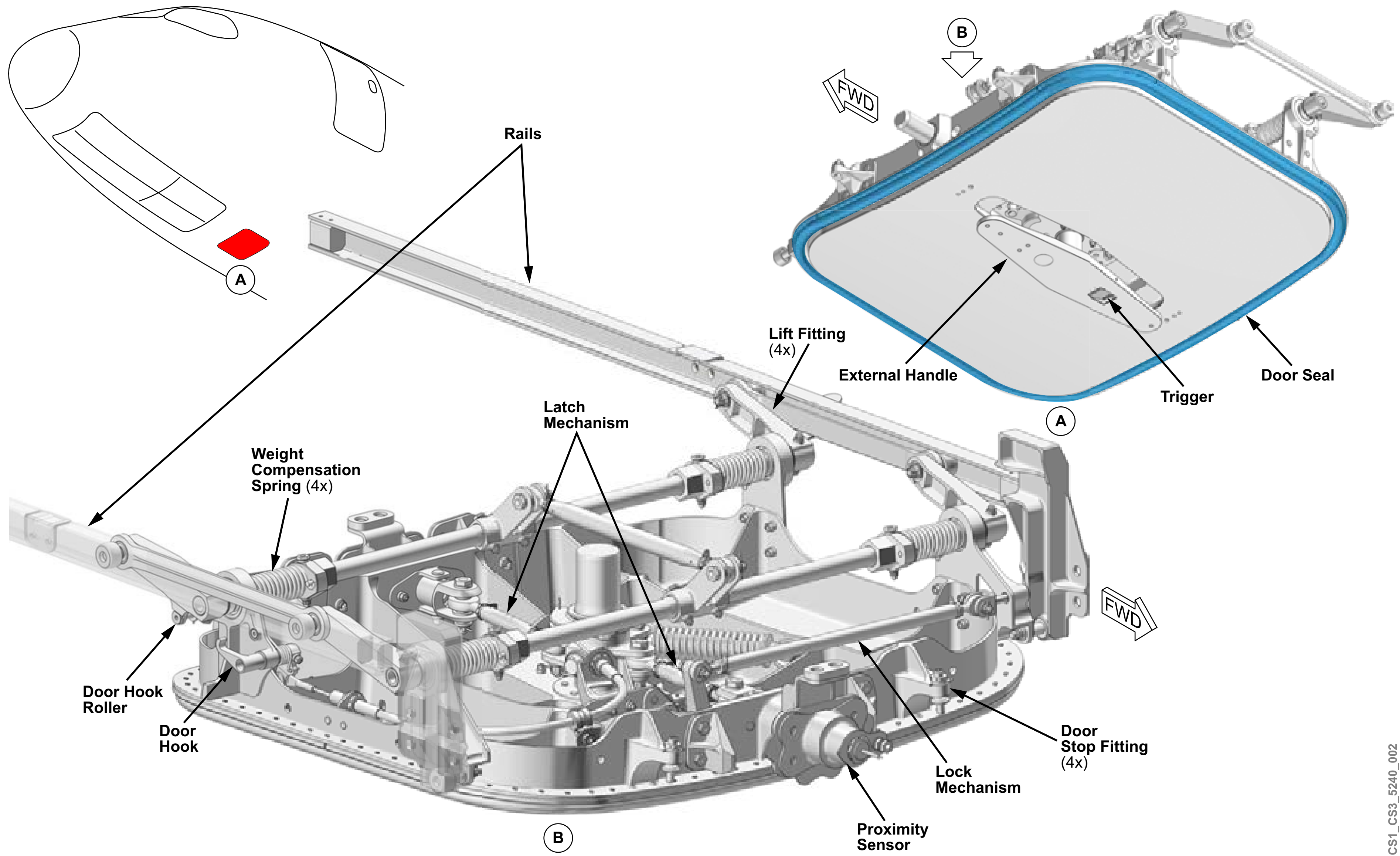


Figure 65: Forward Equipment Bay Door

MID EQUIPMENT BAY DOOR

The mid equipment bay door is an inward plug-type door. The door is operated using a handle on the outside of the door. An additional grab handle is also available to help support the door during opening and closing.

The mid equipment bay door is attached to the fuselage by two hinge arms that connect to hinge fittings mounted on the door surround. The door is held closed by a latch mechanism consisting of two spring-loaded latch pins. The handle mechanism has a spring-loaded handle and a handle flap. It is connected to the latch mechanism of the door. Both the handle and handle flap are spring-loaded to the close position. A pressurized door seal helps maintain cabin pressurization. Pressurization loads are carried by four door stop fittings.

Latch bumpers and fuselage bumpers provide protection to the surrounding structure and door as the door is opened. When the door is fully opened it rests on the fuselage bumpers and the mid equipment bay surround structure. When the door is not fully closed the latch pins rest on the latch bumpers.

There is one proximity latch sensor located on the door surround that provides door indication to EICAS.

Door Opening

The door can only be opened from the outside. To open the door, the handle flap is pushed in and the handle is pulled down. The latches retract, allowing the door to be pushed inward. The door handle is released and the latches extend, at this point the door can be rested on the latch bumpers. The door is then rotated to the door fully open position until the door latches contact the fuselage bumpers.

Door Closing

The door is rotated closed until the latches contact the latch bumpers. The door handle is pulled down, retracting the latch pins and can be lowered into the closed position.

The grab handle helps to maintain control of the door and allows sufficient force to be applied to compress the door seal during the latch pin engagement.

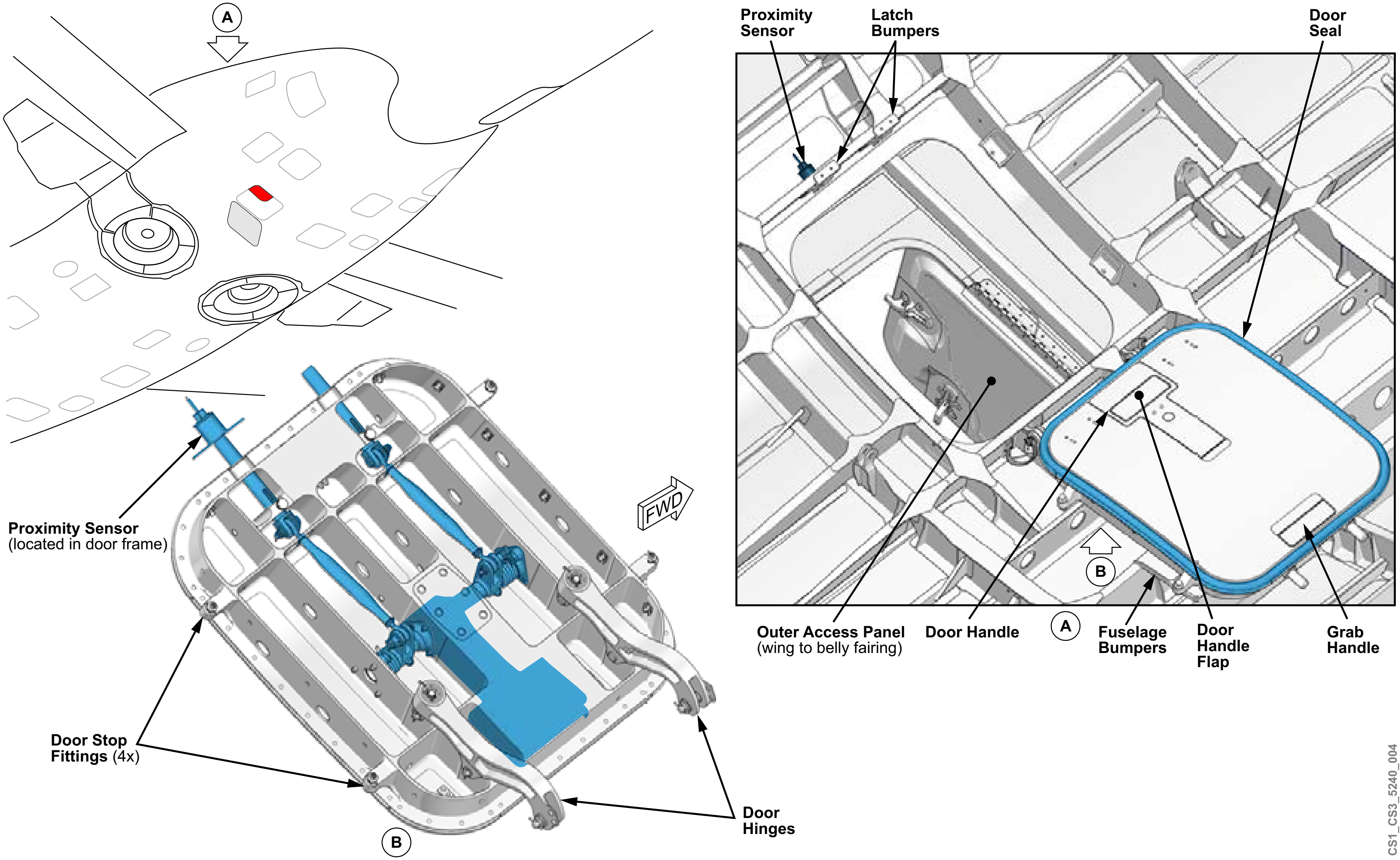


Figure 66: Mid Equipment Bay Door

AFT EQUIPMENT BAY DOOR

The aft equipment bay door is a nonplug-type door located aft of the pressure bulkhead and provides access to the equipment installed in that zone. It also acts as a pressure blowout panel in case of pressure bulkhead failure.

The door is attached to the fuselage by two hinges. The hinges are located in the forward direction of the aircraft, allowing the aerodynamic forces to keep the door closed. The door hinges are secured to the aircraft using pit pins. A strut is attached between the door and the fuselage. When the strut is extended, a lock mechanism engages and prevents the door from swinging due to gust loads. A door seal prevents water and dust ingression.

The latch mechanism consists of two rods that transfer the rotary motion of the handle to two roller arms. The roller arms sit against their stopper fittings to enable latching of the door.

A proximity sensor provides door position indication to EICAS. The proximity sensor is mounted on the fuselage and the target is located on the door.

Door Opening

The door is opened from the outside. To open the aft equipment bay door, depress the trigger plate located on the door latch. This releases the handle, which is pulled back to release the door.

Door Closing

To close the door, disengage the strut lock mechanism and push the door closed. Push the external handle until the handle latches under the trigger plate.

Overpressure Condition

In an overpressure condition, the pressure builds up in the internal face of the door. A load is generated between the two roller arms and the structure mounted stop fittings. When the pressure builds, the springs compress allowing the shaft to rotate. The roller arms then rotate away from the stop fittings and the door opens.

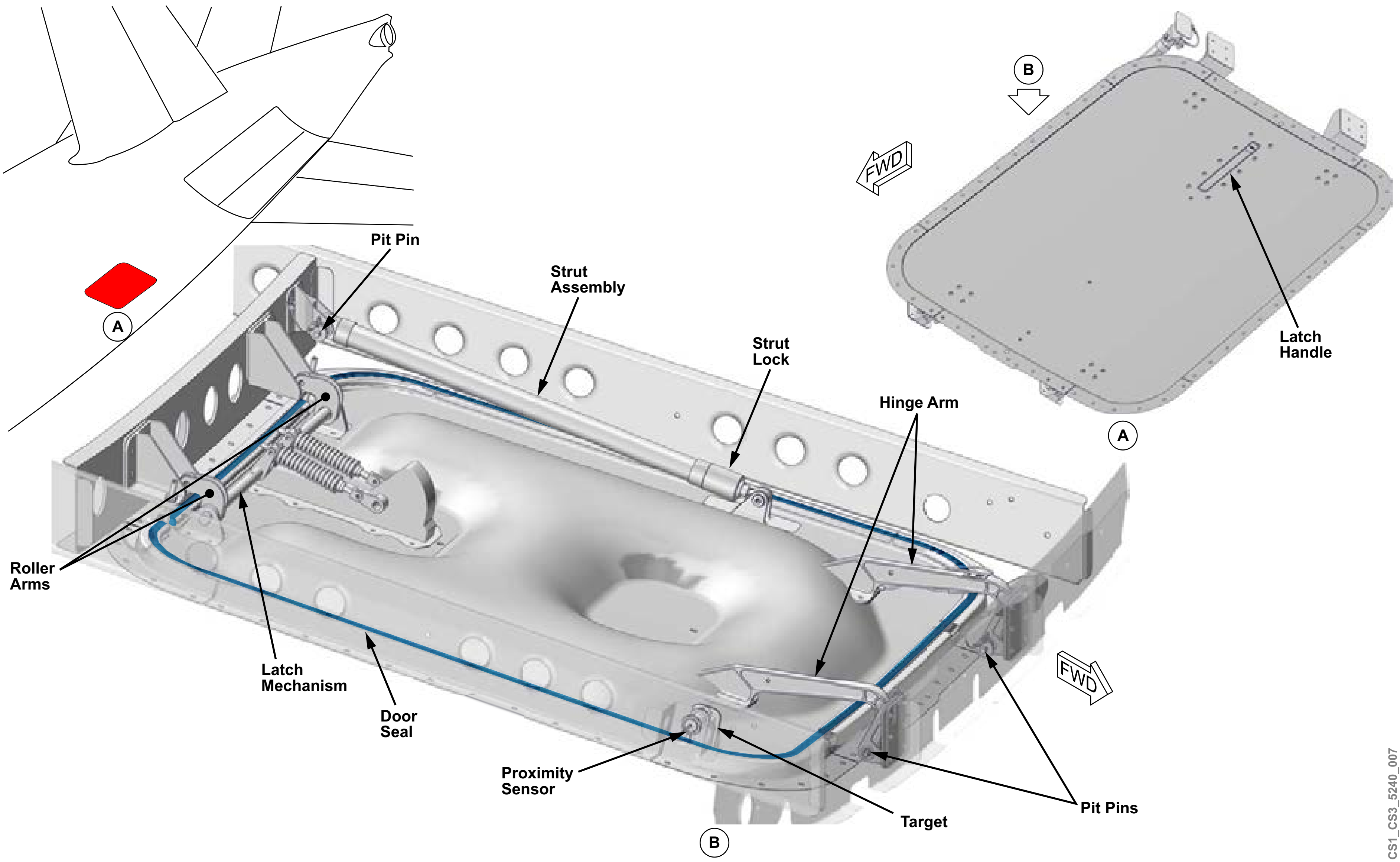
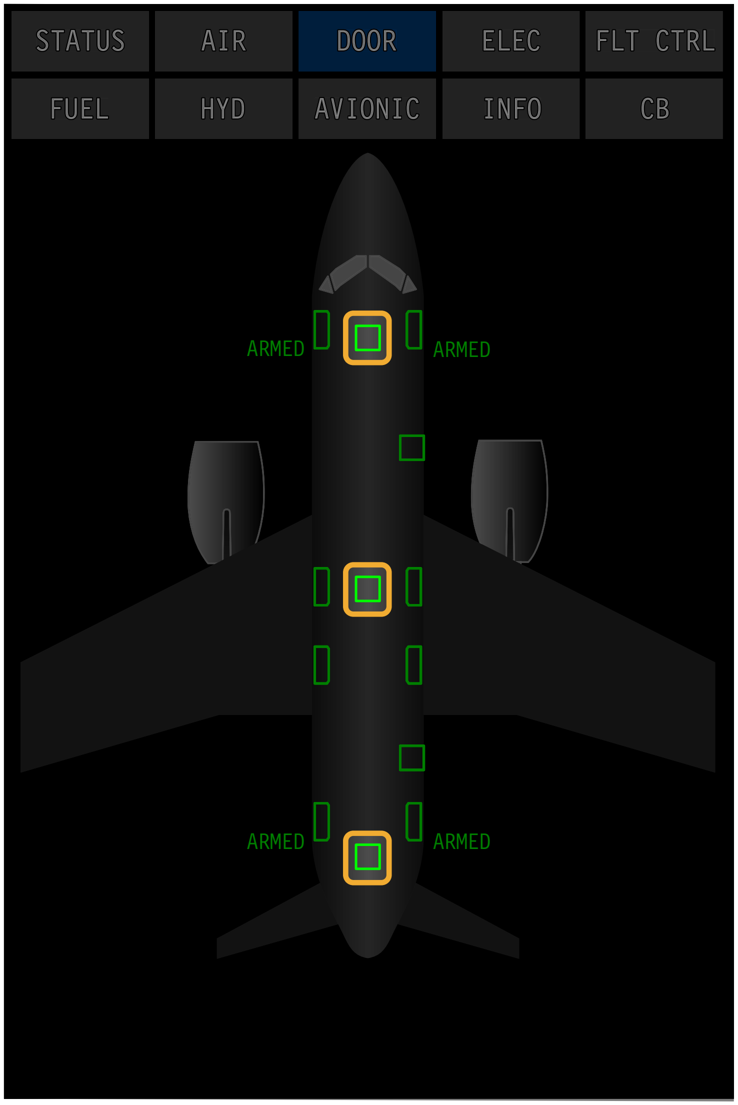


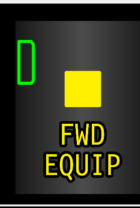
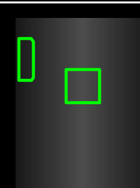
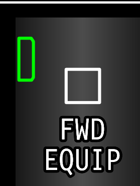
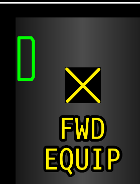
Figure 67: Aft Equipment Bay Door

CONTROLS AND INDICATIONS

The equipment bay door status information is displayed on the DOOR synoptic page.



SYNOPTIC PAGE – DOOR

DOOR STATUS	
Symbol	**Condition**
	At least one engine running and the door is not closed/latched/locked. **OR** Door fault. Note: Label of door is shown in amber.
	Door closed/latched/locked. Note: Label of door is not shown.
	Door open on ground, engine is not running. Note: Label of door is shown in white.
	Invalid or failed LGSCU or invalid door sensor. Note: Label of door is shown in amber.

CS1_CS3_5230_014

Figure 68: Equipment Bay Doors Controls and Indications

DETAILED DESCRIPTION

EQUIPMENT BAY DOOR INDICATING SYSTEM

Each of the three equipment bay doors have one closed proximity sensor to detect the door condition, and provide an indication to the flight deck.

The proximity sensors on the fwd and mid equipment bay doors detect the position of the latches. The forward and mid equipment bay sensors are monitored by landing gear and steering control unit (LGSCU) 1.

The aft equipment bay door is equipped with a proximity sensor located in the door cutout, and a sensor target fitted to the composite structure. The aft equipment bay door is monitored by LGSCU 2.

If an equipment bay door sensor or target fails with the aircraft on the ground and no engines running, a DOOR FAULT advisory message is displayed on EICAS.

If the engines are running, an EQUIP BAY DOOR caution message is displayed when any equipment bay door is open or has a fault.

The LGSCUs provide door status information to EICAS and the DOOR synoptic page.

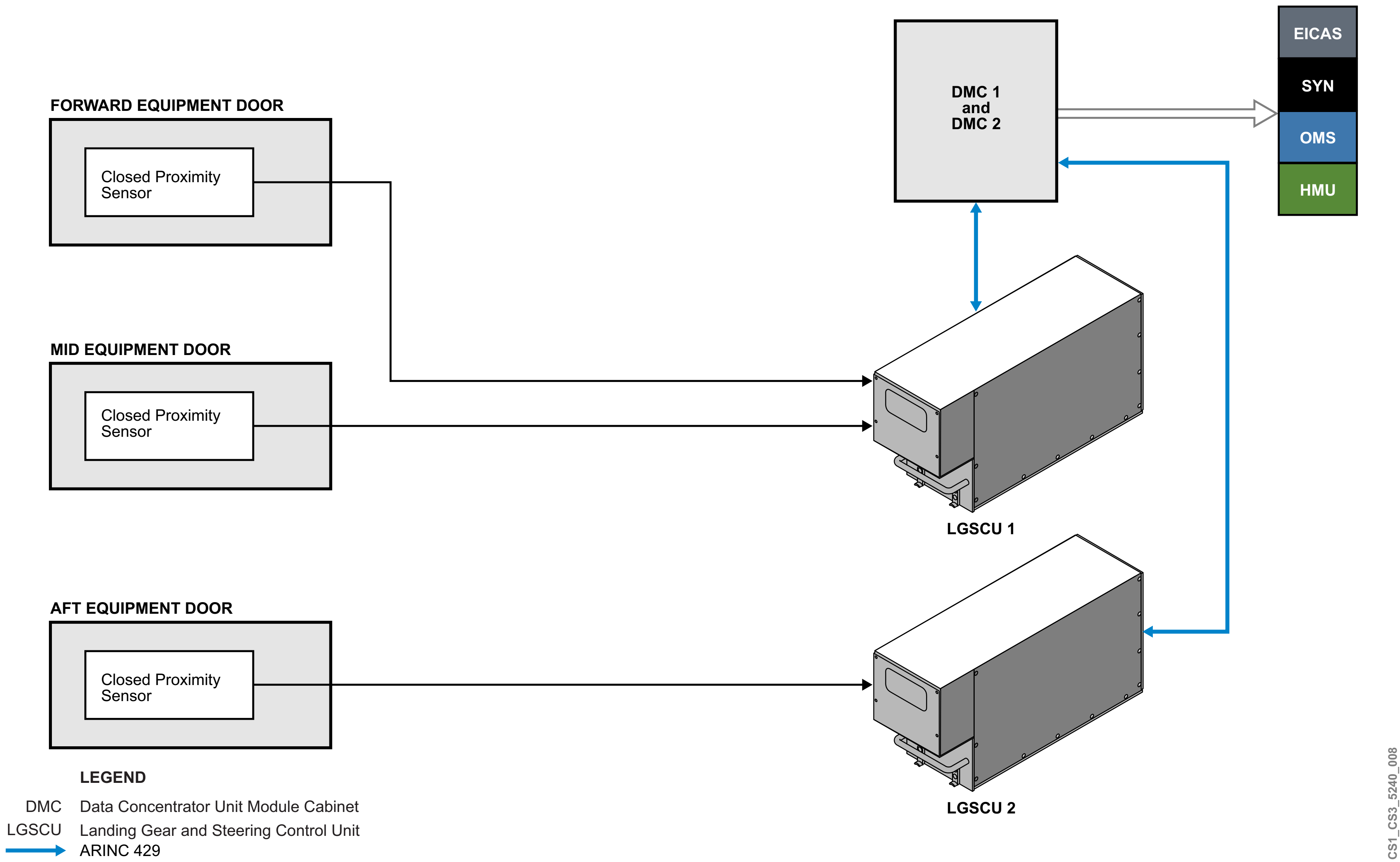


Figure 69: Equipment Bay Door Indicating System

MONITORING AND TESTS

The following crew alerting system (CAS) and INFO messages are associated with the equipment bay doors.

CAS MESSAGES

Table 10: CAUTION Messages

MESSAGE	LOGIC
EQUIP BAY DOOR	Aft, mid, or fwd equipment bay door failed or open with at least one engine running.

Table 11: ADVISORY Messages

MESSAGE	LOGIC
DOOR OPEN	Any door open while on ground and no engine running.
DOOR FAULT	Any electrical or mechanical failures of any door while on the ground and no engine running. Refer to INFO messages.

Table 12: INFO Messages

MESSAGE	LOGIC
52 DOOR FAULT - FWD EQUIP BAY DOOR SNSR INOP	Electrical problem has been detected with the fwd equipment bay door proximity sensor.
52 DOOR FAULT - MID EQUIP BAY DOOR SNSR INOP	Electrical problem has been detected with the mid equipment bay door proximity sensor.
52 DOOR FAULT - AFT EQUIP BAY DOOR SNSR INOP	Electrical problem has been detected with the aft equipment bay door proximity sensor.

52-22 FLIGHT CREW EMERGENCY EXIT HATCH

GENERAL DESCRIPTION

The flight crew emergency exit (FCEE) hatch is located on the centerline of the flight deck upper fuselage. It is an inward opening removable plug hatch providing a 498 mm x 549 mm (19.6 in. x 21.6 in.) opening for the crew to exit the flight deck.

The FCEE hatch is equipped with a seal that inflates once the aircraft is pressurized. Four door stop fittings transmit pressurization loads into the fuselage structure. The FCEE hatch is attached to the structure by two hinge arms. The hatch opens downwards and towards the aft of the aircraft.

There is a drain valve mounted on the external handle box. The valve connects to a drain tube. The valve drains water from the external handle box when the aircraft is not pressurized. This minimizes the chance of the handle being stuck due to icing. When the cabin is pressurized, the valve closes.

The latch mechanism has two latches, a torque tube and a rod. The rod is pulled or pushed by the internal or external handle. Moving either handle extends or retracts the latches. The latch mechanism is spring-loaded to ensure the handles remain in the stowed position.

The internal handle is connected directly to the latch mechanism. The internal handle operation does not affect the external handle. When the handle is stowed in the locked position, two green flags indicate the latches are closed.

The external handle mechanism consists of the handle, and handle flap. The handle flap has a latch that locks the external handle in place. The external handle shaft has two override mechanisms that act on the internal handle, which in turn operates the latch mechanism and opens the hatch.

FCEE HATCH OPENING

To open the FCEE hatch:

1. Remove the handle cover to access the internal handle.
2. Pull down the internal handle and pull the crew escape hatch down.

To open the FCEE hatch from the outside:

1. Push the handle flap and pull up on the external handle.
2. Lower the FCC hatch and restow the handle.

FCEE HATCH CLOSING

To close the FCEE hatch:

1. Push the crew escape hatch up to align with the structure.
2. Push the internal handle to latch the FCC hatch.
3. Install the handle cover and make sure the green indication on the latch pins are visible through the lining hole.

NOTE

The FCEE hatch can only be closed using the internal handle.

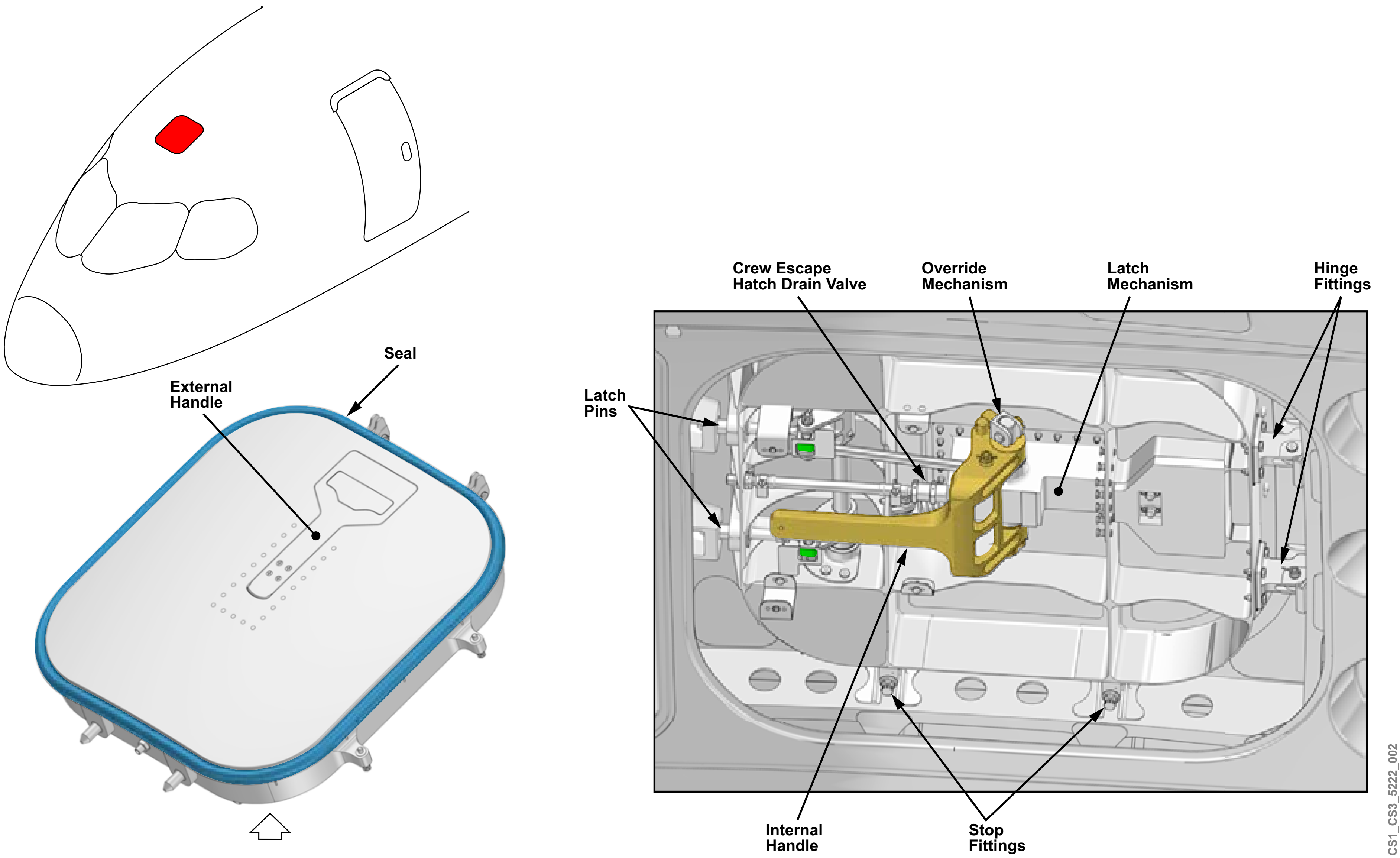


Figure 70: Flight Crew Emergency Exit Hatch

25-17 FLIGHT DECK DOOR

GENERAL DESCRIPTION

The reinforced flight deck door assembly provides security and protects against unauthorized access to the flight deck. The door is controlled by a flight deck remote access system (FDRAS) that allows locking and unlocking the door without leaving the pilots seat. The FDRAS is controlled by an electronic door control module (EDCM) controlled by the COCKPIT DOOR panel.

The EDCM controls the locking and unlocking of the door based on inputs from the keypad and the COCKPIT DOOR panel and is powered by DC ESS BUS 1. The FDRAS can be deactivated for maintenance using the LOCK SYSTEM switch on the EDCM. The FDRAS operation can be programmed using the PROGRAM MODE button and the keypad.

During normal operation, the door is always locked. A request for entry can be made from the keypad at the flight deck entranceway. When a code is entered on the keypad, the flight deck receives a door bell chime and DOOR icon on the engine indication and crew alerting system (EICAS). The door can be unlocked from the COCKPIT DOOR panel DOOR UNLOCK switch.

In an emergency, an emergency code is entered on the keypad. Access to the flight deck can be given using the DOOR UNLOCK switch or denied using the EMER ACCESS PBA. The emergency access request is ignored, the door unlocks for 5 seconds after a pre-programmed time period. The status of the access request is shown on the color-coded status indicator on the keypad.

In case of an FDRAS failure, a dead bolt is used for locking and unlocking the flight deck door. A key is required to lock and unlock the door from the passenger compartment.

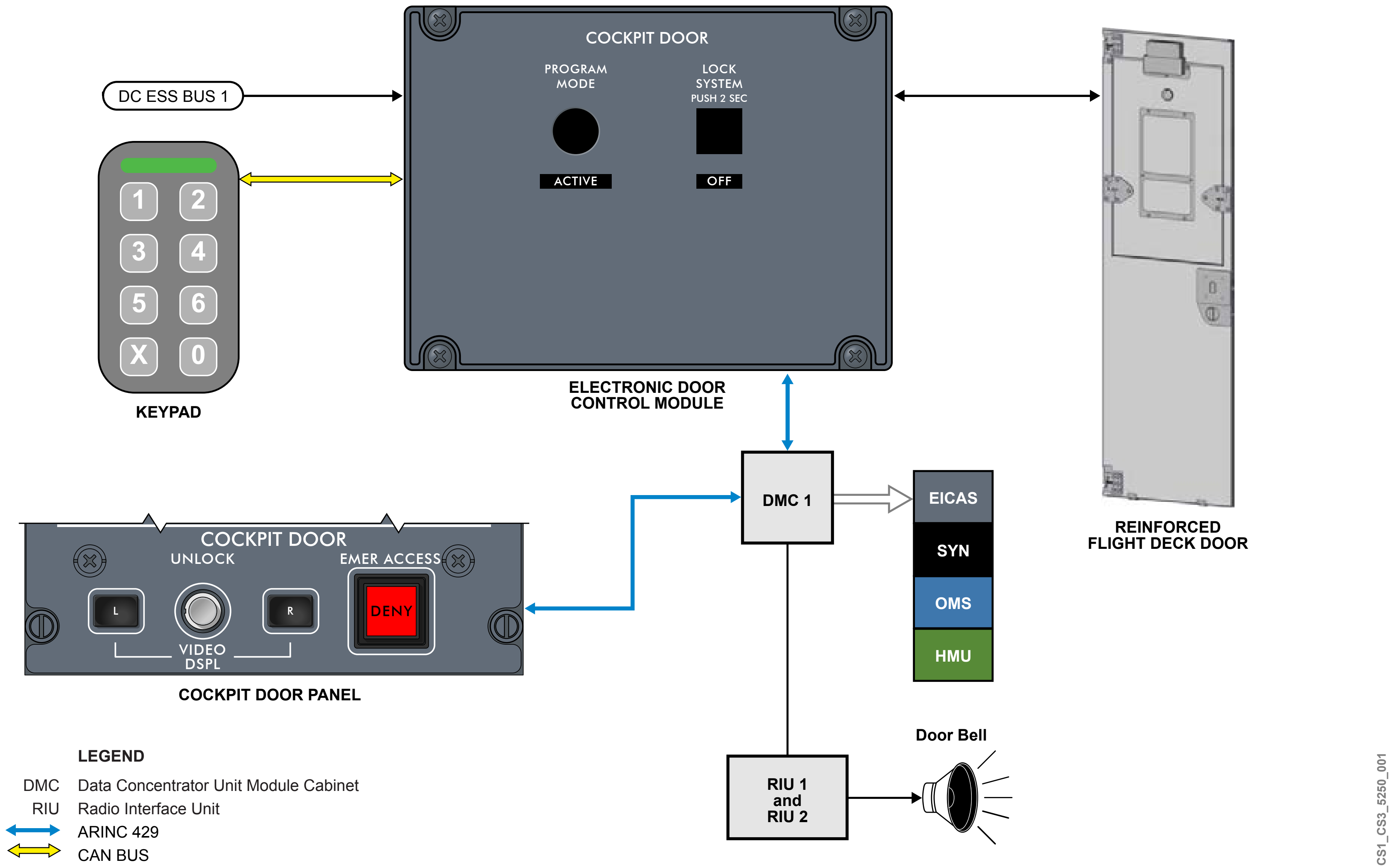


Figure 71: Flight Deck Door System

COMPONENT LOCATION

The flight deck door system includes:

- Keypad
- Electronic door control module
- Cockpit door panel
- Flight deck door (Refer to figure 73)

KEYPAD

The flight deck door keypad is located on the forward lavatory bulkhead.

ELECTRONIC DOOR CONTROL MODULE

The electronic door control module (EDCM) is located on the bulkhead behind the pilots seat.

COCKPIT DOOR PANEL

The cockpit door panel is located on the pedestal.

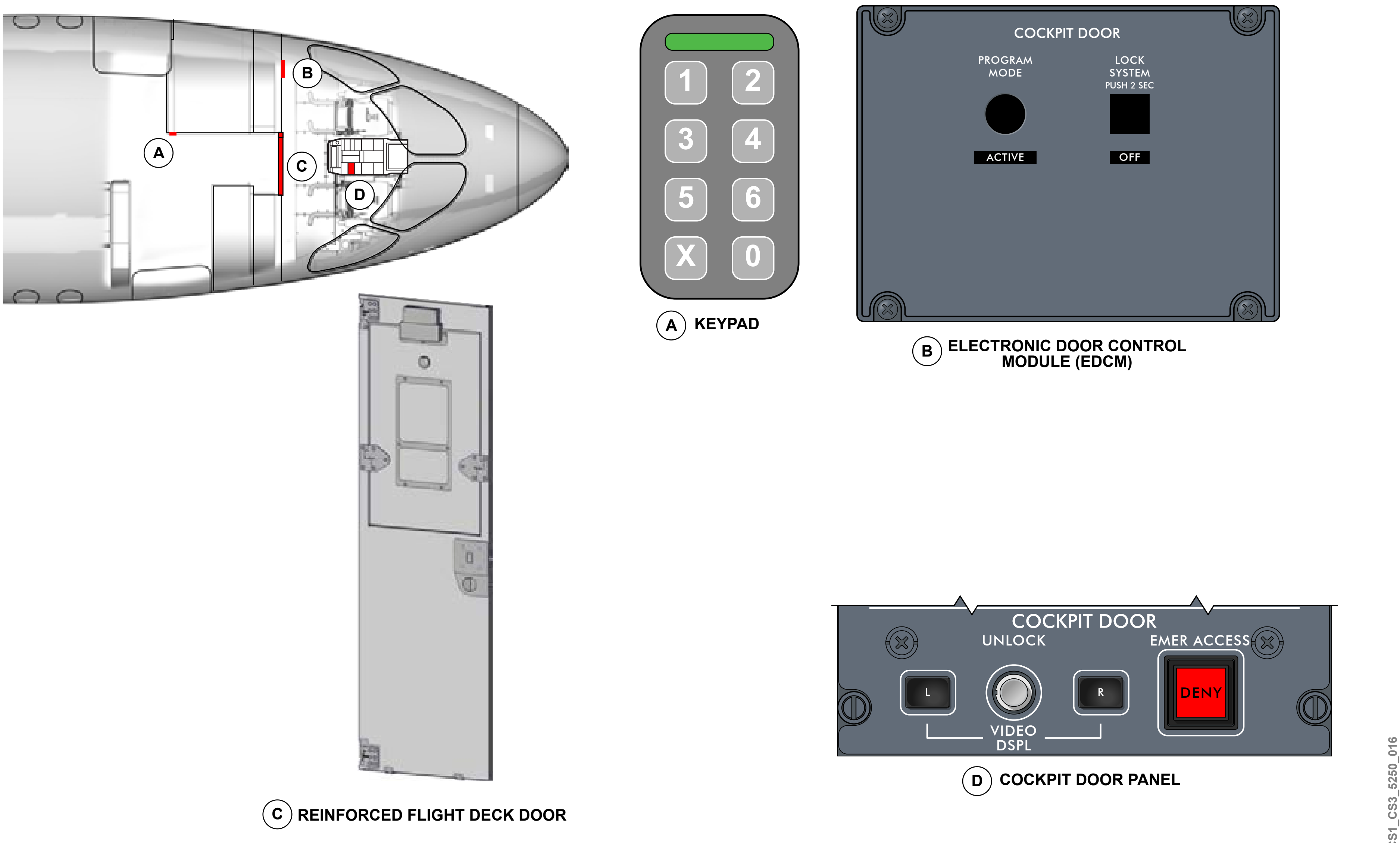


Figure 72: Flight Compartment Door System Component Location

FLIGHT DECK DOOR AND DOOR FRAME COMPONENTS

The flight deck door components include:

- Peephole
- Hinges
- Decompression panel
- Decompression latch
- Door latch
- Deadbolt
- Door handle
- Key lock

The door frame components include:

- Striker
- Upper plunger assembly
- Lower plunger assembly

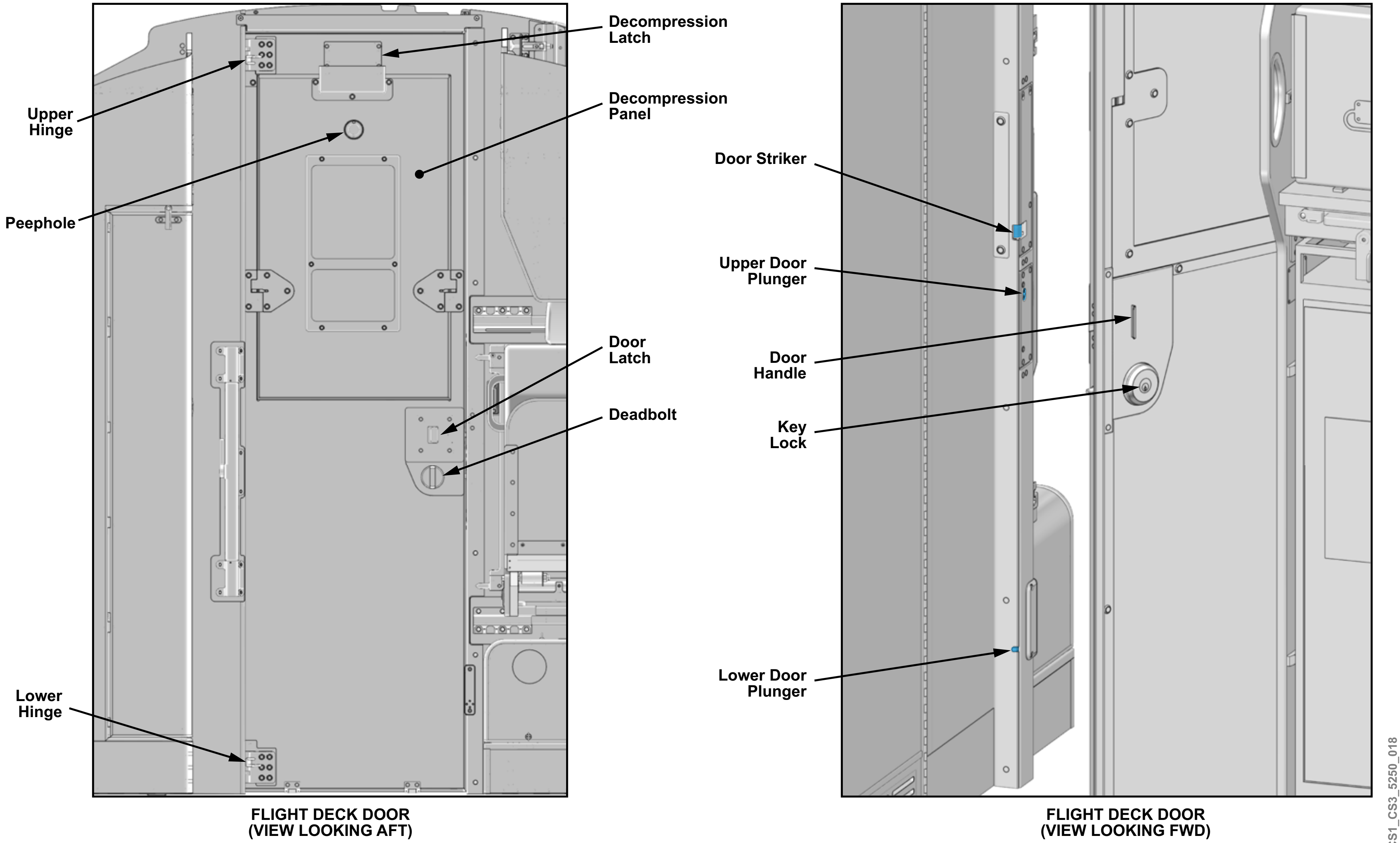


Figure 73: Flight Deck Door and Door Frame Components

COMPONENT INFORMATION

FLIGHT DECK DOOR CONSTRUCTION

The flight deck door is made of Nomex honeycomb core panels, sandwiched in the middle with a bulletproof insert. It is attached to the structure with two quick-release hinges that can be removed from inside the flight deck in an emergency.

Peephole

The door has a bullet resistant peephole to view the main passenger cabin. The peephole has two lenses to provide a field of view of 135°.

Hinges

The flight deck door is mounted by two hinges located at the upper and lower sides of the door. These hinges enable the flight deck door to be open or close as needed. The door is secured in place by two spring-loaded pins that are inserted into upper and lower hinges. In an emergency situation, the flight deck crew can quickly dislocate the whole door from its hinges by removing the two hinge pins.

Door Latch

The flight deck door latch is used to open and close the flight deck door. The door latch consists of a spring-loaded bolt. The springs keep the door bolt extended in the door closed position.

Deadbolt

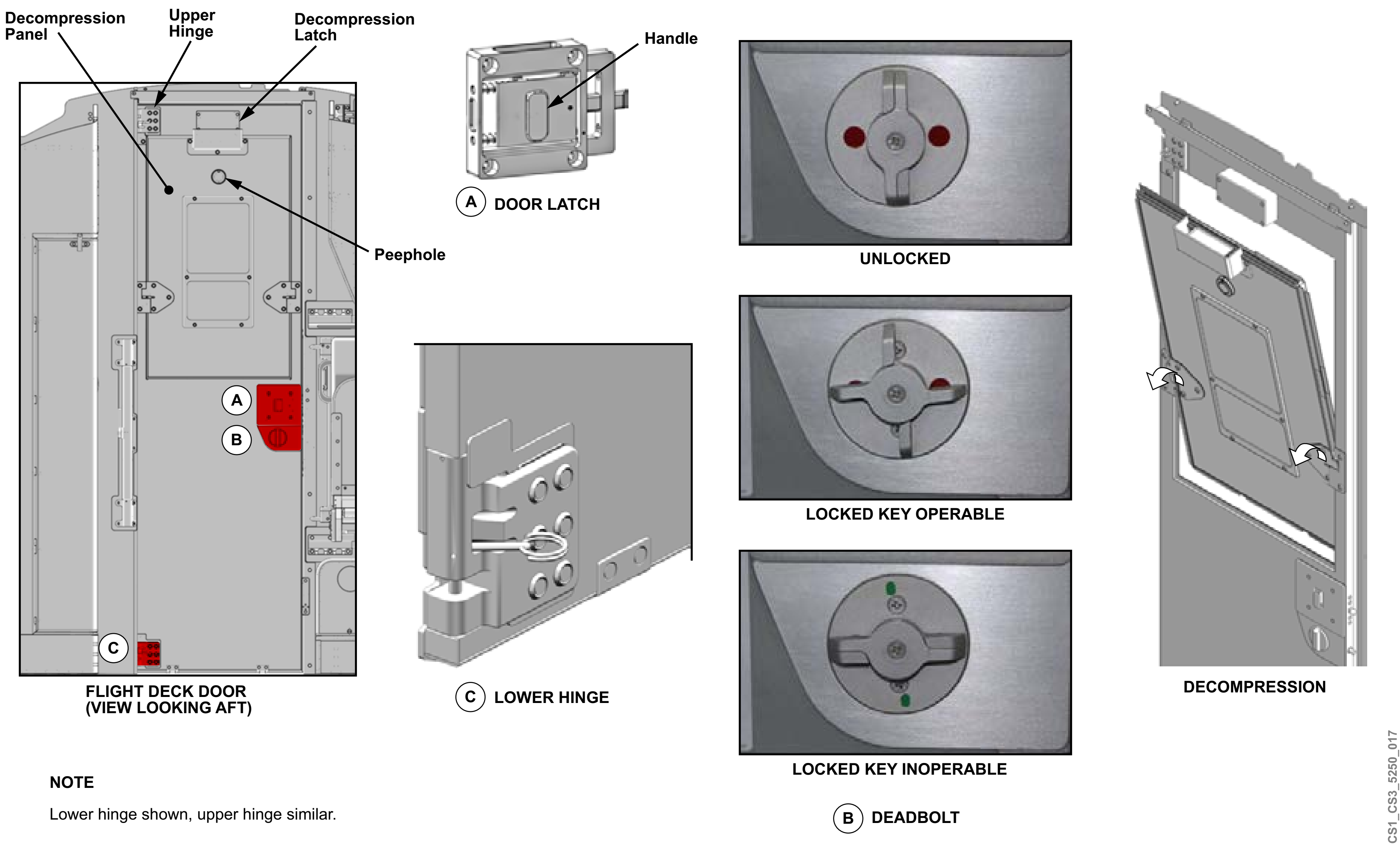
The manual deadbolt is used as a mechanical backup to the FDRAS for locking and unlocking the door. It is designed to lock and unlock the door manually from inside the flight deck. The flight deck door can be unlocked from the main cabin by using a key if the deadbolt is in the locked key operable position.

The deadbolt has three positions:

- Unlocked: The door is unlocked and two red dots show on the deadbolt
- Locked key operable: The door can be unlocked with a key from outside the flight deck. Two red dots are partially shown
- Locked key inoperable: The door is locked and the key cannot be used to open the door. Two green dots are shown

Decompression Latch

During an emergency decompression, the decompression latch releases the decompression blowout panel on the flight deck door allowing it to pivot on the two hinges. The decompression panel is then free to swing open, equalizing the pressure between the flight deck and passenger cabin. The decompression panel remains open until it is manually closed.



NOTE

Lower hinge shown, upper hinge similar.

Figure 74: Flight Deck Door Construction

Door Striker

The door striker assembly has a solenoid operated lock pin, a spring-loaded striker, and a microswitch. The solenoid operated lock pin extends when power is available and locks the door. It prevents the striker from rotating when force is applied to the passenger cabin side of the flight deck door. The microswitch is actuated when power to the solenoid is removed and the lock pin retracts.

When power is removed for the solenoid and the lock pin retracts, the spring-loaded striker moves to the unlock position and the door can be opened.

Lower Door Plunger Assembly

The lower door plunger assembly actuates a microswitch when the door is closed. The microswitch provides the signal to the electronic door control module (EDCM).

Upper Door Plunger Assembly

The upper door plunger assembly actuates a microswitch when the deadbolt is used to lock the door. The microswitch provides a signal to the data concentrator unit module cabinet (DMC).

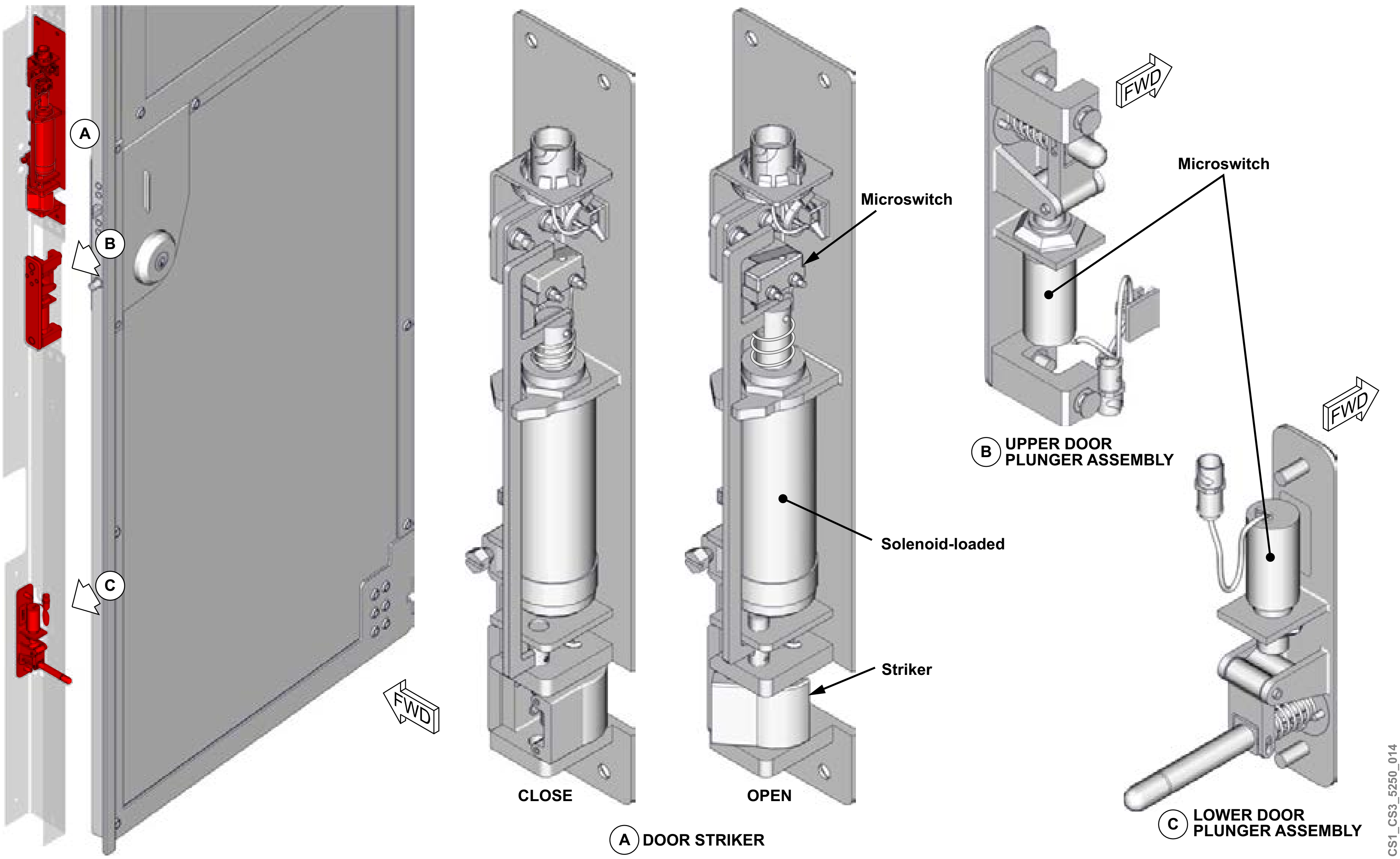


Figure 75: Door Striker and Door Plunger Assemblies

DETAILED COMPONENT INFORMATION

ELECTRONIC DOOR CONTROL MODULE

The EDCM has a maintenance mode and programming mode that are selected using switches on the face of the EDCM.

In the maintenance mode, the flight deck door is unlocked and any keypad inputs or COCKPIT DOOR panel inputs are ignored. To enter the maintenance mode, press the LOCK SYSTEM switch for 2 seconds. To exit the maintenance mode, press the LOCK SYSTEM switch again. When the system is in the maintenance mode, a CKPT DOOR LOCK OFF status message is displayed.

The programming mode is used to customize the flight deck remote access system (FDRAS). The programming mode can only be entered if the system is already in maintenance mode. To enter the programming mode the PROGRAM MODE switch is held in for 2 to 5 seconds. When the programming mode is entered, the ACTIVE indication is displayed on the unit and the EDCM monitors the keypad for inputs. The programming is carried out from the keypad.

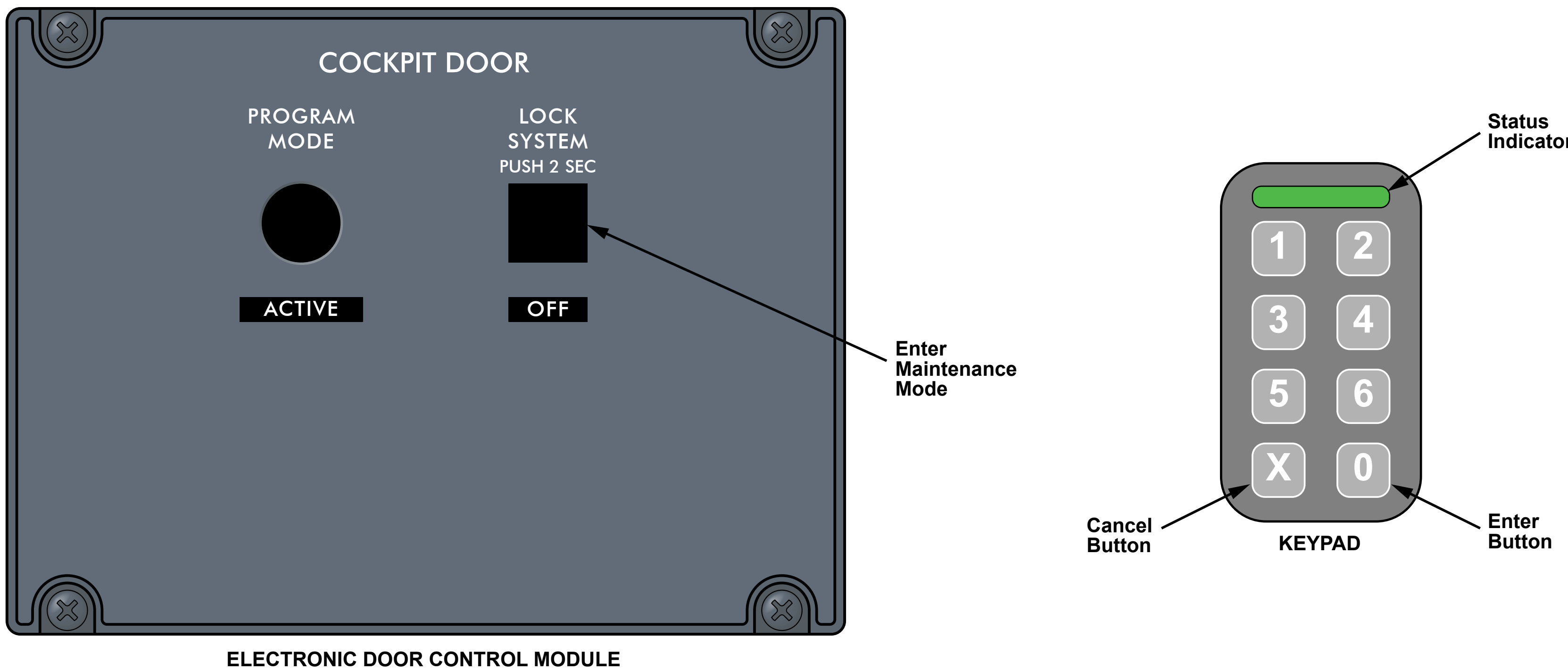
The programming mode is used to program the following functions:

- Emergency access code. A three to five digit code to initiate an emergency request
- Emergency audio alert time with a programmable duration of 30 to 60 seconds
- Emergency reaction time with a programmable duration 30 to 60 seconds
- Deny access inhibit time with a programmable duration 5 to 30 minutes
- Doorbell time with a programmable duration of 1 to 5 seconds
- Doorbell button numeric key selection to define bell key

To return to the maintenance mode, press the PROGRAM MODE switch.

KEYPAD

The keypad is numbered from 1 to 6. The key marked O is used to enter a request. The key marked X cancels a request. Access codes are programmable and can be customized by the operator. A status indicator indicates the status of the entry request made. The keypad is backlit as soon as any key is pressed.



PROGRAMMING FUNCTIONS	
Emergency Access Code	Three to five digit code to initiate an emergency request
Emergency Audio Alert Time	Programmable duration of 30 to 60 seconds
Emergency Reaction Time	Programmable duration of 30 to 60 seconds
Deny Access Inhibit Time	Programmable duration of 5 to 30 minutes
Doorbell Time	Programmable duration of 1 to 5 seconds
Doorbell Button Numeric Key Selection	Define bell keypad selection

CS1_CS3_5250_021

Figure 76: Electronic Door Control Module and Keypad

CONTROLS AND INDICATIONS

KEYPAD

The keypad is the interface for the FDRAS. The keypad is used for making normal and emergency entry requests, as well as, programming the system. The keypad status indicator show the status of the entry request.

ELECTRONIC DOOR CONTROL MODULE

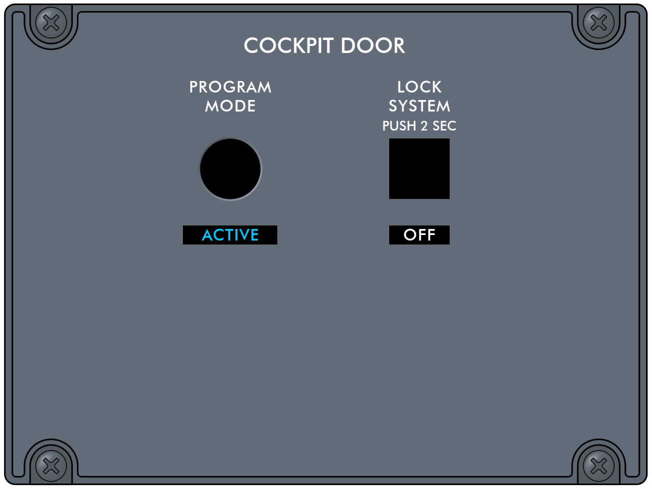
The EDCM ACTIVE indication illuminates when the system is in the programming mode.

COCKPIT DOOR PANEL

The COCKPIT DOOR panel UNLOCK button is used to unlock the flight deck door. The EMER ACCESS button is used to deny entry into the flight deck.

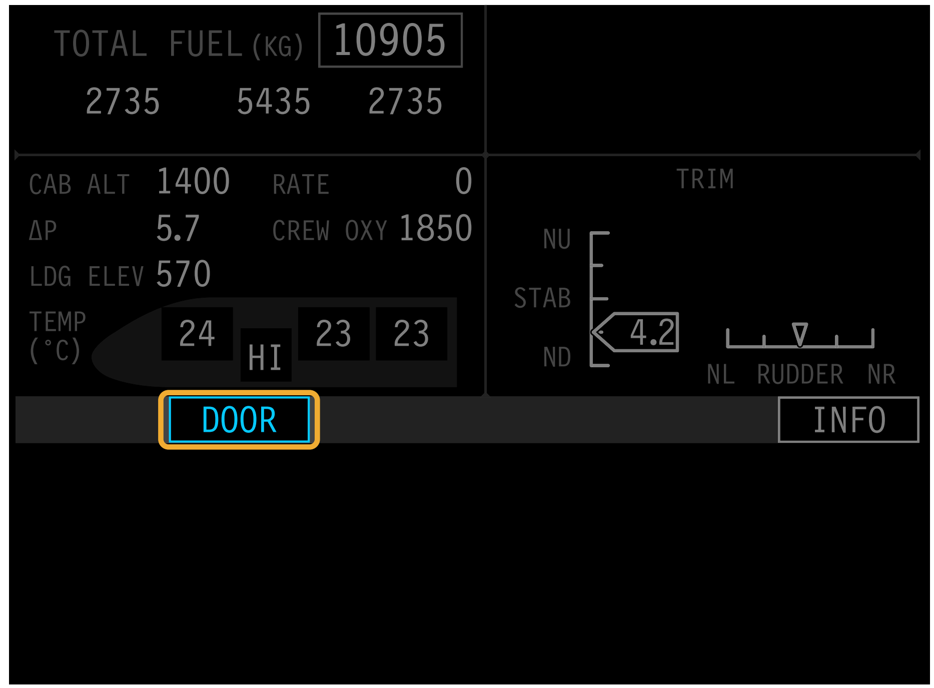
EICAS

The EICAS DOOR icon comes on when an access request has been made from the keypad.

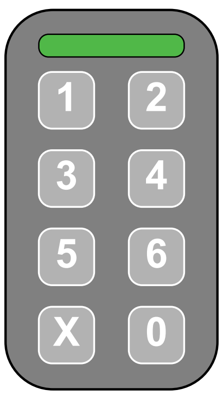


ELECTRONIC DOOR CONTROL MODULE

COCKPIT DOOR PANEL



EICAS



KEYPAD

STATUS ILLUMINATION	
Symbol	**Condition**
	Normal
	Awaiting Response
	Access Granted
	Access Denied

CS1_CS3_5250_009

Figure 77: Flight Deck Door Controls and Indications

OPERATION

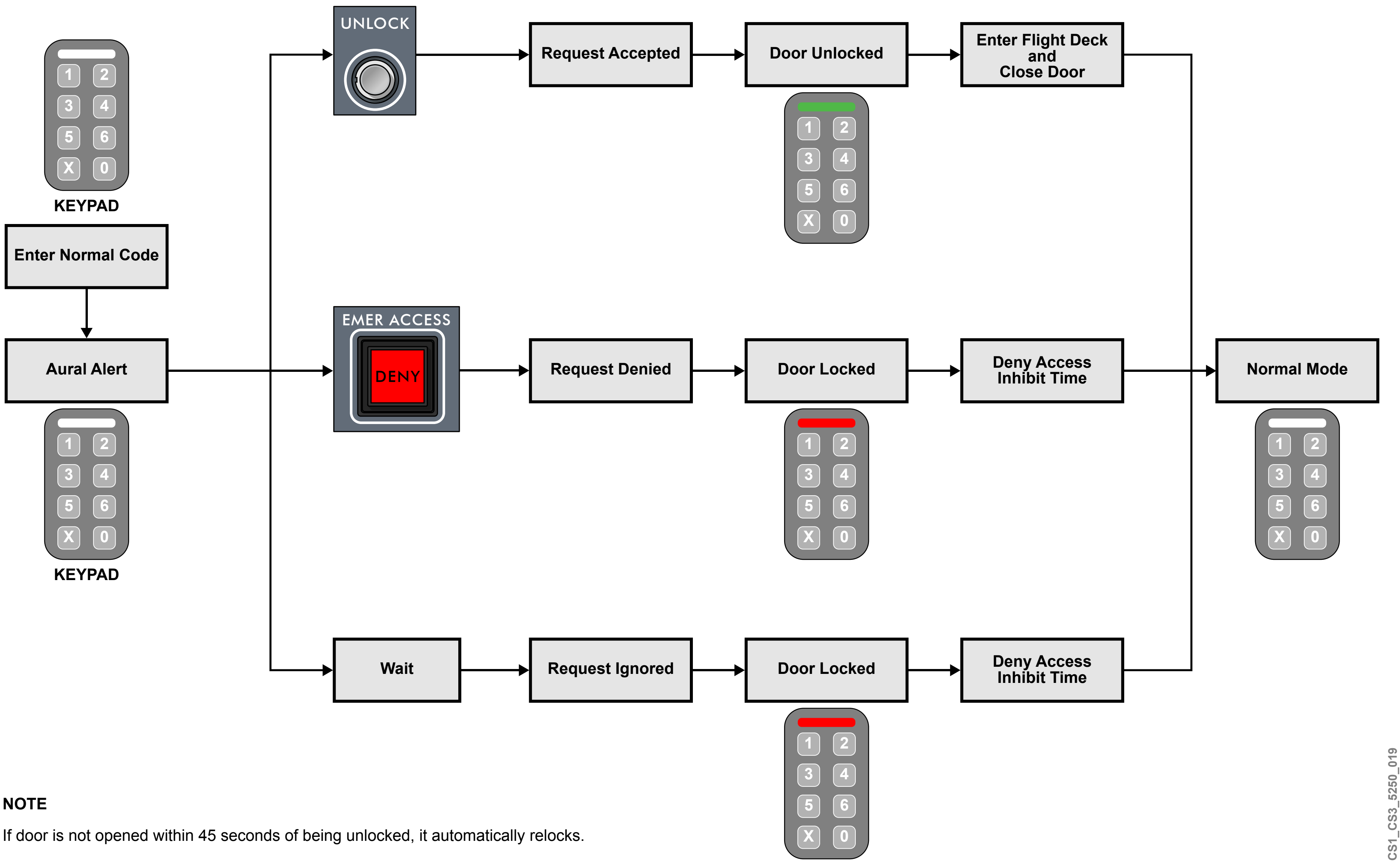
NORMAL ENTRY REQUEST

A normal access request to the flight deck is made by pressing the doorbell key on the keypad followed by the O button. The doorbell key could be any of the numeric buttons 1 to 6 on the keypad. The default doorbell key is 1. This activates the aural doorbell for a programmable time between 1 to 5 seconds. The keypad status indicator turns amber, indicating the request is being processed.

When the UNLOCK button is pressed, the keypad status indicator turns green, indicating that the door is unlocked. Once the door is opened and then closed, the keypad status indicator goes blank, indicating it is ready to accept a new request.

If the request is denied or ignored, the status indicator turns red, indicating access is denied. The system times out after the pre-programmed time period and returns to normal operation.

A request for access can be canceled by pressing the X button.



NOTE

If door is not opened within 45 seconds of being unlocked, it automatically relocks.

Figure 78: Normal Entry Request

EMERGENCY ENTRY REQUEST

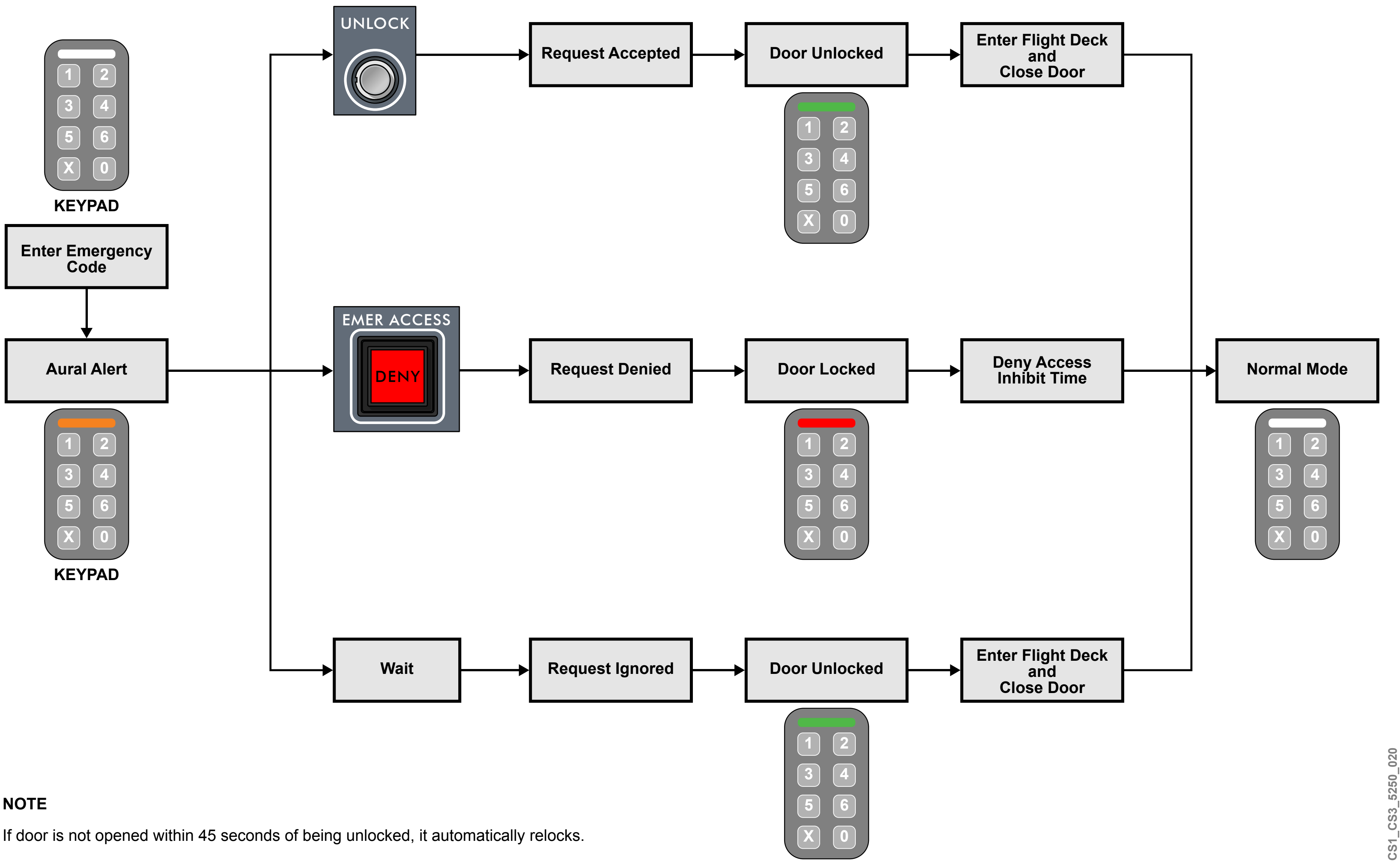
An emergency access request to the flight deck is made by entering the correct 3 to 5 digit code on the keypad followed by the O key. The default emergency access code is 1 2 3 4 5. This activates the aural doorbell for 30 to 60 seconds depending on the program settings. In addition, a CKPT DOOR EMER ACCESS warning message is displayed on EICAS. The request must be responded to within the 30 to 60 seconds programmable reaction time. The keypad status indicator turns amber, indicating the request is being processed.

When the UNLOCK button is pressed, the keypad status indicator turns green, indicating that the door is unlocked. Once the door is opened and then closed, the keypad status indicator blanks, indicating it is ready to accept a new request.

The request can be denied using the EMER ACCESS PBA. If the request is denied within the reaction time, any further request is ignored by the system for a programmable time of 5 to 30 minutes. The keypad status indicator turns red and remains red for the duration of the deny access time. Once the system times out, the keypad status indicator blanks, indicating a new request can be made.

If the request is ignored and the programmed reaction time period ends, the door unlocks for 5 seconds as indicated by the green keypad status indicator. Once the flight deck is entered and the door is closed, the system returns to normal mode operation, as indicated by the blank status indicator on the keypad.

A request for emergency access can be canceled by pressing the X button.



NOTE

If door is not opened within 45 seconds of being unlocked, it automatically relocks.

Figure 79: Emergency Entry Request

DETAILED DESCRIPTION

The EDCM communicates with the keypad via a CAN BUS. The keypad provides inputs to the EDCM for entry requests and programming.

When the EDCM is powered, it provides 28 VDC to the door striker to lock the door. A microswitch monitors the door striker solenoid position to provide a locked or unlocked indication to the EDCM.

The lower door plunger assembly microswitch provides a door closed signal to the EDCM.

The COCKPIT DOOR panel provides unlock and deny commands over an ARINC 429 BUS.

The EDCM communicates with the data concentrator unit module cabinet (DMC) via an ARINC 429 BUS. The DMC provides EDCM information to:

- Radio interface unit (RIU) for aural alerts
- EICAS for the door icon and CAS messages
- Onboard maintenance system (OMS) for fault reporting

If the FDRAS fails a CKPT DOOR LOCK FAIL caution message is displayed indicating the door lock has failed. When the deadbolt is used to lock the door, the upper plunger microswitch signals the DMC that the door is manually locked and the CKPT DOOR LOCK FAIL caution message is removed.

The EDCM performs self-tests after power-up. During operation it performs a continuous BIT (CBIT), monitors the solenoid driver and the status of the solenoid. Any keypad faults are monitored via the CAN BUS. All detected faults are stored in the non-volatile memory and reported to the OMS.

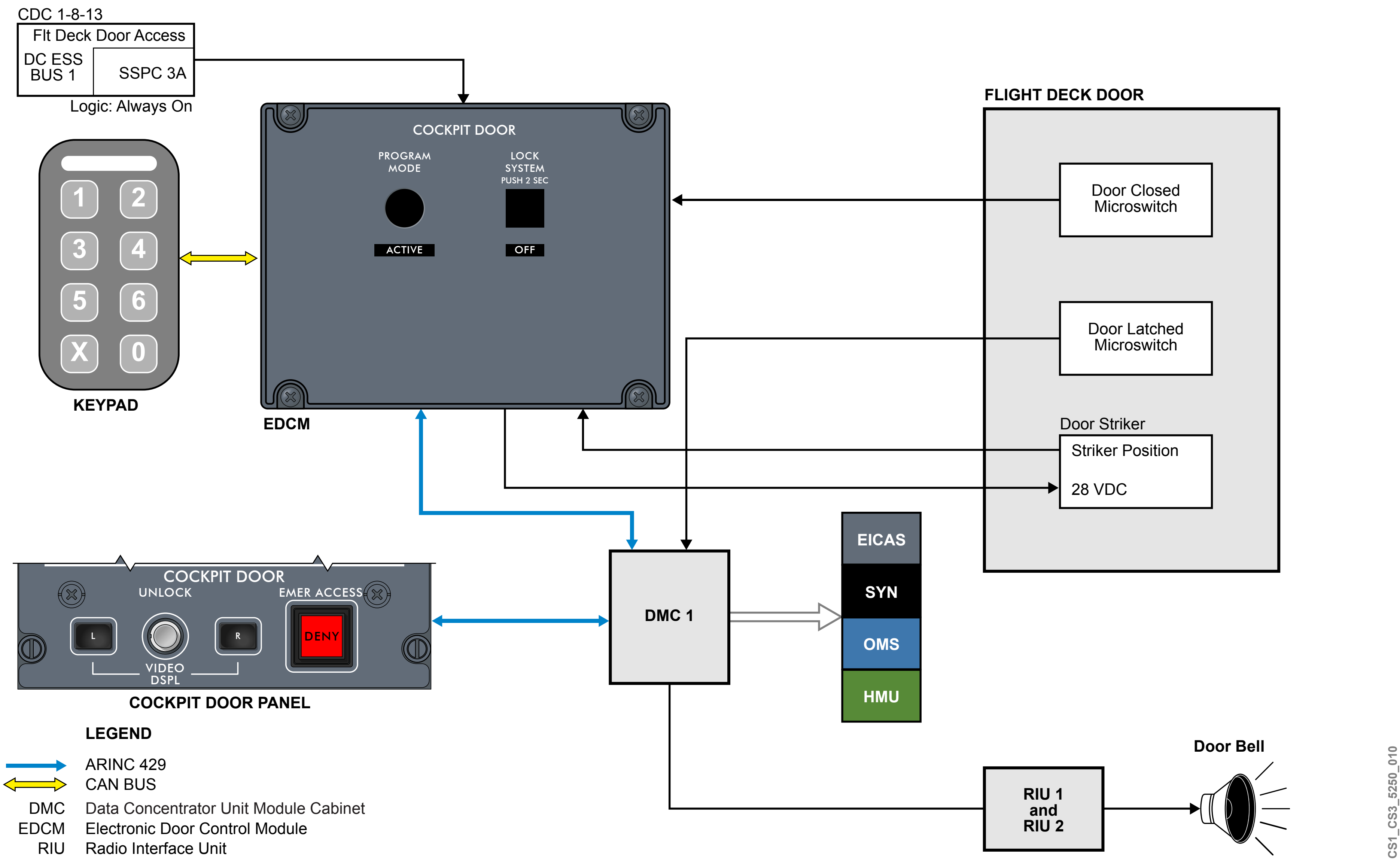


Figure 80: Flight Deck Door Remote Access System Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages associated with the FDRAS.

CAS MESSAGES

Table 13: WARNING Message

MESSAGE	LOGIC
CKPT DOOR EMER ACCESS	Emergency request from cabin. Associated aural COCKPIT DOOR.

Table 14: CAUTION Message

MESSAGE	LOGIC
CKPT DOOR LOCK FAIL	Failure of the FDRAS to lock door or loss of communication to the DMC.

Table 15: ADVISORY Message

MESSAGE	LOGIC
CKPT DOOR OPEN	Cockpit door is open.

Table 16: STATUS Message

MESSAGE	LOGIC
CKPT DOOR LOCK OFF	Cockpit door system is in maintenance mode or when associated solid-state power controller (SSPC) is PULLED and LOCKED.

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ATA 53 - Fuselage

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

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FUSELAGE - CHAPTER BREAKDOWN

53-00 FUSELAGE STRUCTURE

GENERAL DESCRIPTION

CS100

The fuselage is a semi-monocoque structure made of eight separate assemblies, attached to make the complete fuselage shell. The assemblies, except the tailcone, are considered primary structures. All sections, except the aft fuselage and tailcone, consist of aluminum-lithium (Al-Li) alloy frames, stringer, and skins. The aft fuselage and tailcone are complete carbon fiber reinforced polymer (CFRP) structures.

The fuselage sections, except the tailcone, are attached with circumferential butt joints. The tailcone is installed using attachment fittings and bolts.

Frames 15, 30, 39, 51, 61, and 79, are the locations of the circumferential butt joints for the assemblies. The tailcone is installed at FR 85.

The assemblies are identified as follows:

- Nose fuselage
- Forward fuselage
- Forward mid fuselage
- Center mid fuselage
- Aft mid fuselage
- Rear fuselage
- Aft fuselage
- Tailcone

The pressure bulkheads and the fuselage skin make the basic pressurized zone. The flight deck, cabin, forward equipment bay, and the forward and aft cargo compartments are included in the pressurized zone. Pressure bulkheads are installed at FR 2 and FR 79.

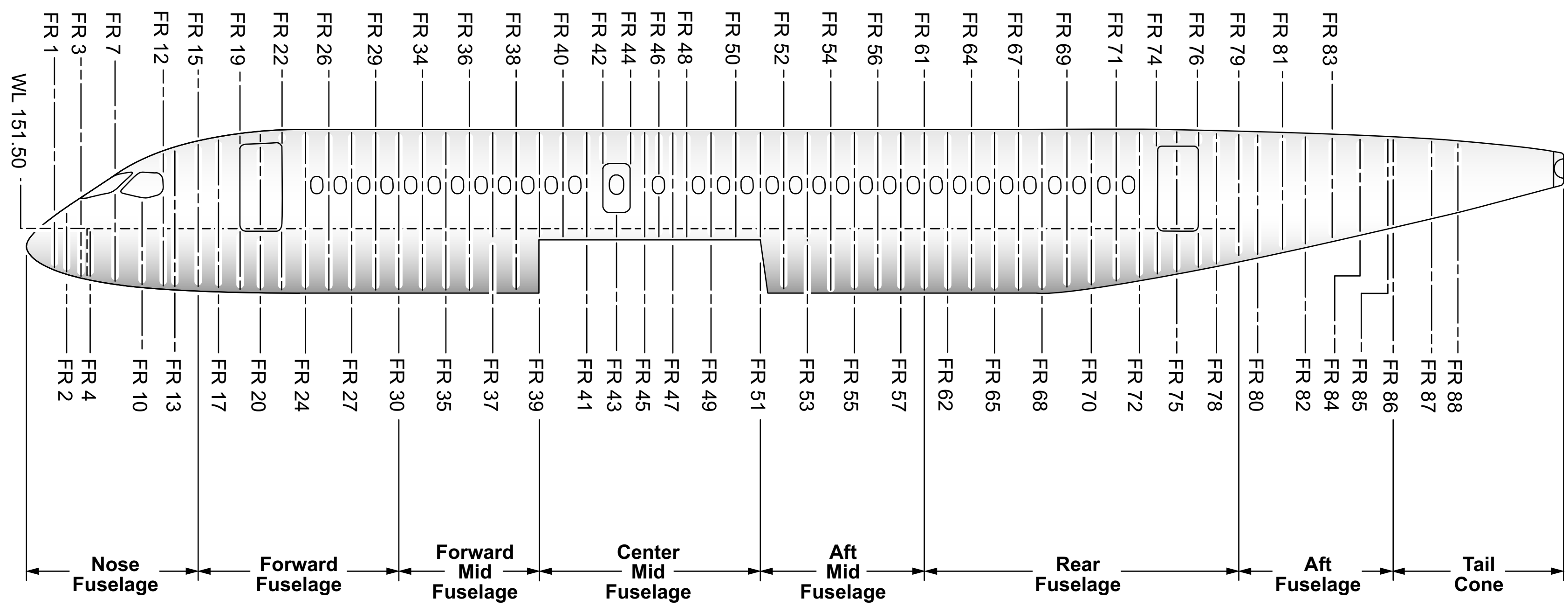
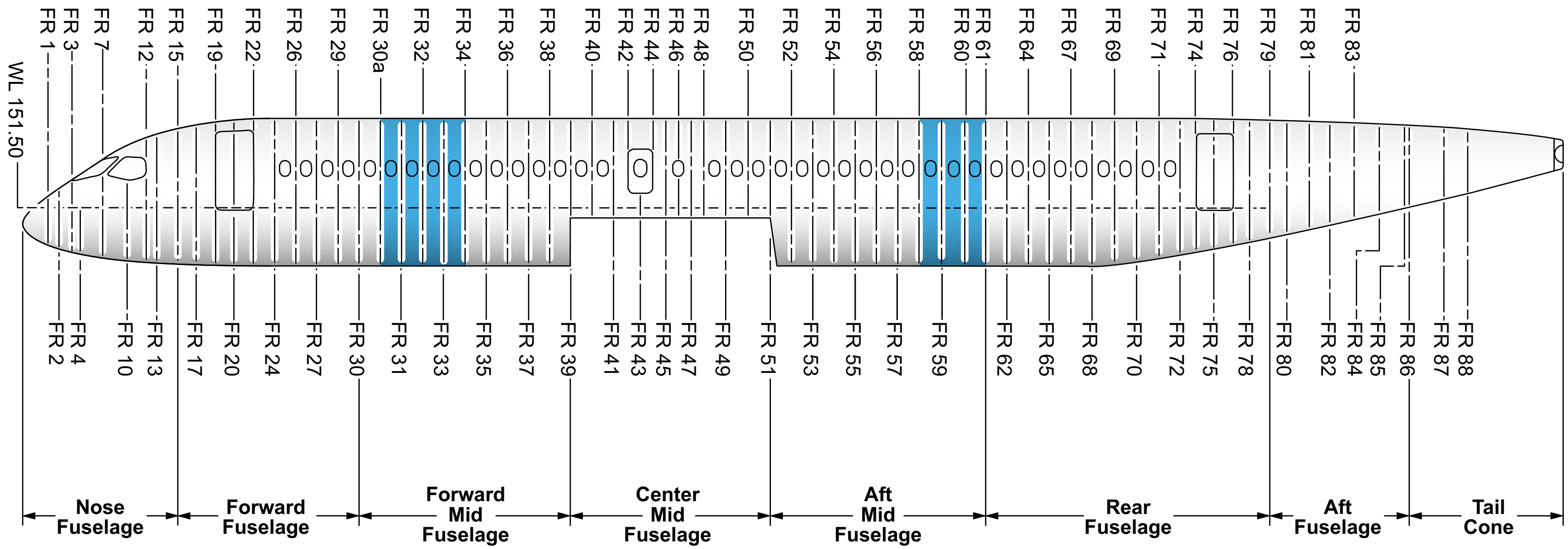


Figure 1: CS100 Fuselage Structure

CS300

The CS300 fuselage construction is the same as the CS100. The CS300 has four additional frames added between frame 30 and frame 34, and three additional frames between frame 57 and frame 61.



LEGEND

Additional Frames added for CS300

CS1_CS3_5300_021

Figure 2: CS300 Fuselage Structure

PRIMARY STRUCTURAL COMPONENTS

The primary structural sections of the fuselage consist of the following components:

- Frames
- Frame splices
- Longitudinal stringers
- Stringer splices
- Intercostals
- Skin panels
- Shear ties
- Straps

These items are located throughout the fuselage sections, depending on the structural requirements. All of these components are manufactured from varying grades of aluminum alloys. The frames, stringers, and skin panels are manufactured almost entirely using Al-Li alloys.

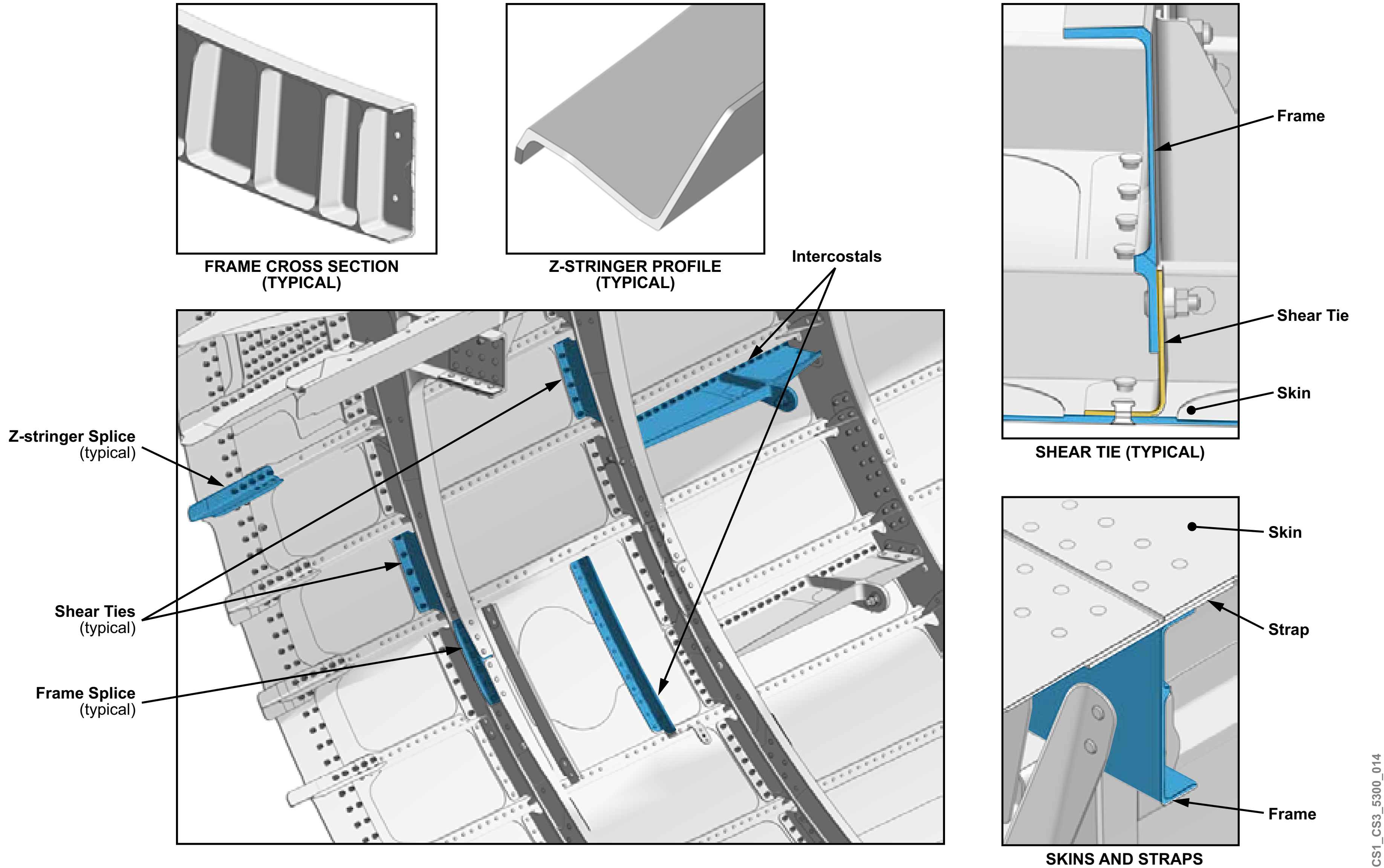


Figure 3: Primary Structural Components

DETAILED COMPONENT INFORMATION

FRAMES

The frames provide the fuselage cross section throughout the length of the aircraft. They are manufactured from either conventional 7000-series aluminum (nose fuselage), or Al-Li alloys (all other primary structural sections). All frames are C profile extrusions.

The frames are made in sections and are assembled to produce a continuous circumferential structure. They are mounted vertically, and are attached to the fuselage skin panels.

FRAME SPLICES

Frame splices have a similar profile to the frames, and are manufactured from similar materials. The frame splices are used to attach the frame sections to produce a single structure.

LONGITUDINAL STRINGERS

The longitudinal stringers are manufactured from Al-Li alloys, and are mounted horizontally, running the length of the fuselage from FR 15 to FR 79. There are no stringers in the nose fuselage or the tailcone.

STRINGER SPLICES

Stringer splices are used at the main butt joints between the fuselage sections. They are used to tie the forward and aft stringers together at the fuselage frames.

INTERCOSTALS

Intercostals are used throughout the aircraft to provide additional strength between frames and stringers. Some intercostals are simply extrusions, which provide a secondary structure for attachment of the skin. Others are machined or forged parts, which assist in supporting other structural members or aircraft components.

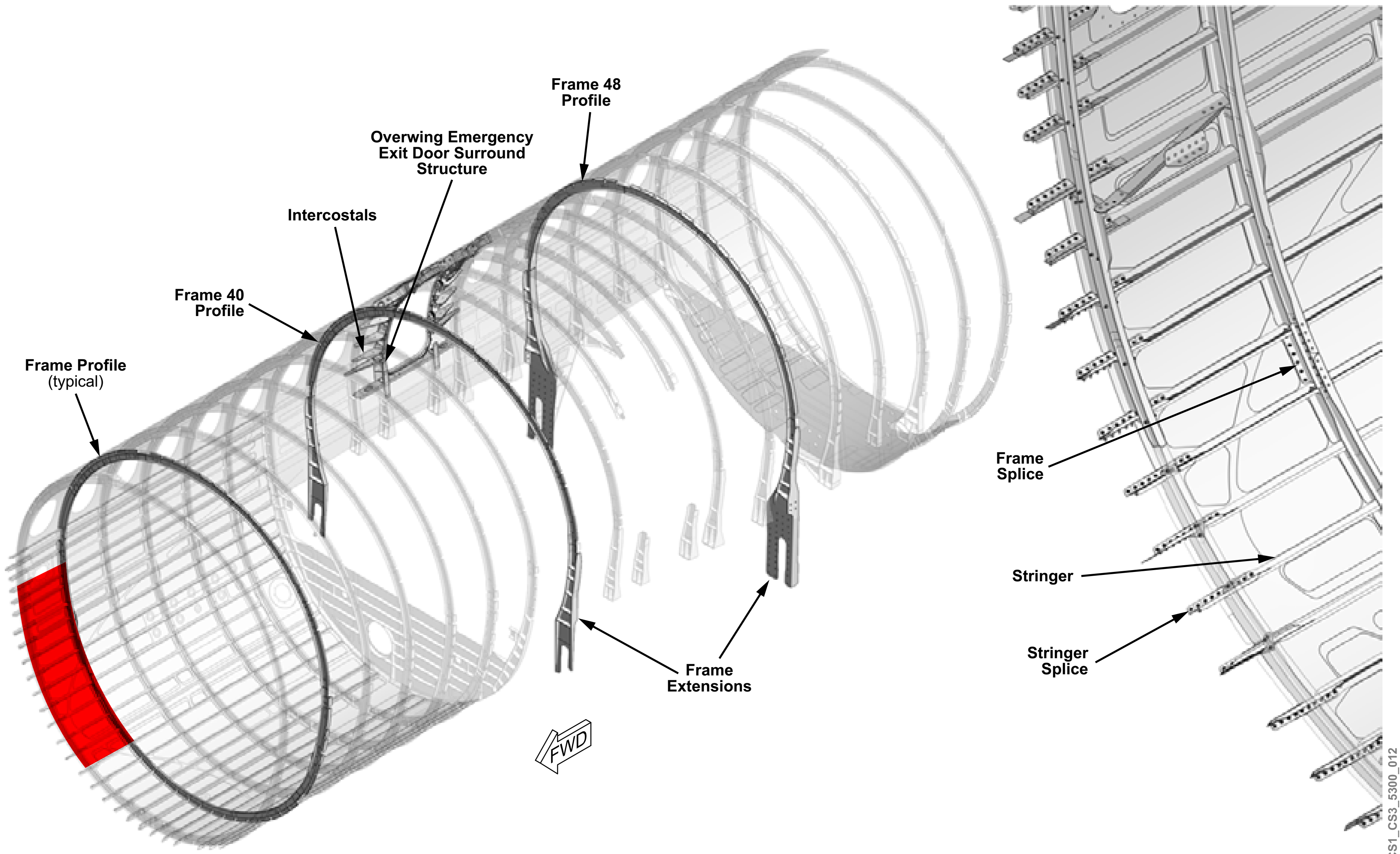


Figure 4: Detailed Structural Component Information

SKIN PANELS

The skin panels are manufactured from three main types of material:

- Al 2524-T3 (damage tolerant) - Nose fuselage skins
- Al-Li alloy - Main fuselage section skins
- CFRP - Lower rear fuselage, and tailcone skins

Due to its formability, Al 2524-T3 is used for the nose fuselage skins, allowing complex curvatures in the panels.

The Al-Li alloy panels used in the main fuselage sections are chemically milled in the pockets between frames and stringers to reduce overall weight, while maintaining required strength.

The CFRP panels are used in fuselage sections that are not subjected to pressurization cycles. These fuselage areas do not use longitudinal stringers. The CFRP panels are manufactured with integral stringers for added strength.

The skin panels have cutouts for passenger, service, cargo, emergency exit and equipment bay doors, windows, antennas, vents, drains, and other items requiring access to the interior or exterior of the fuselage.

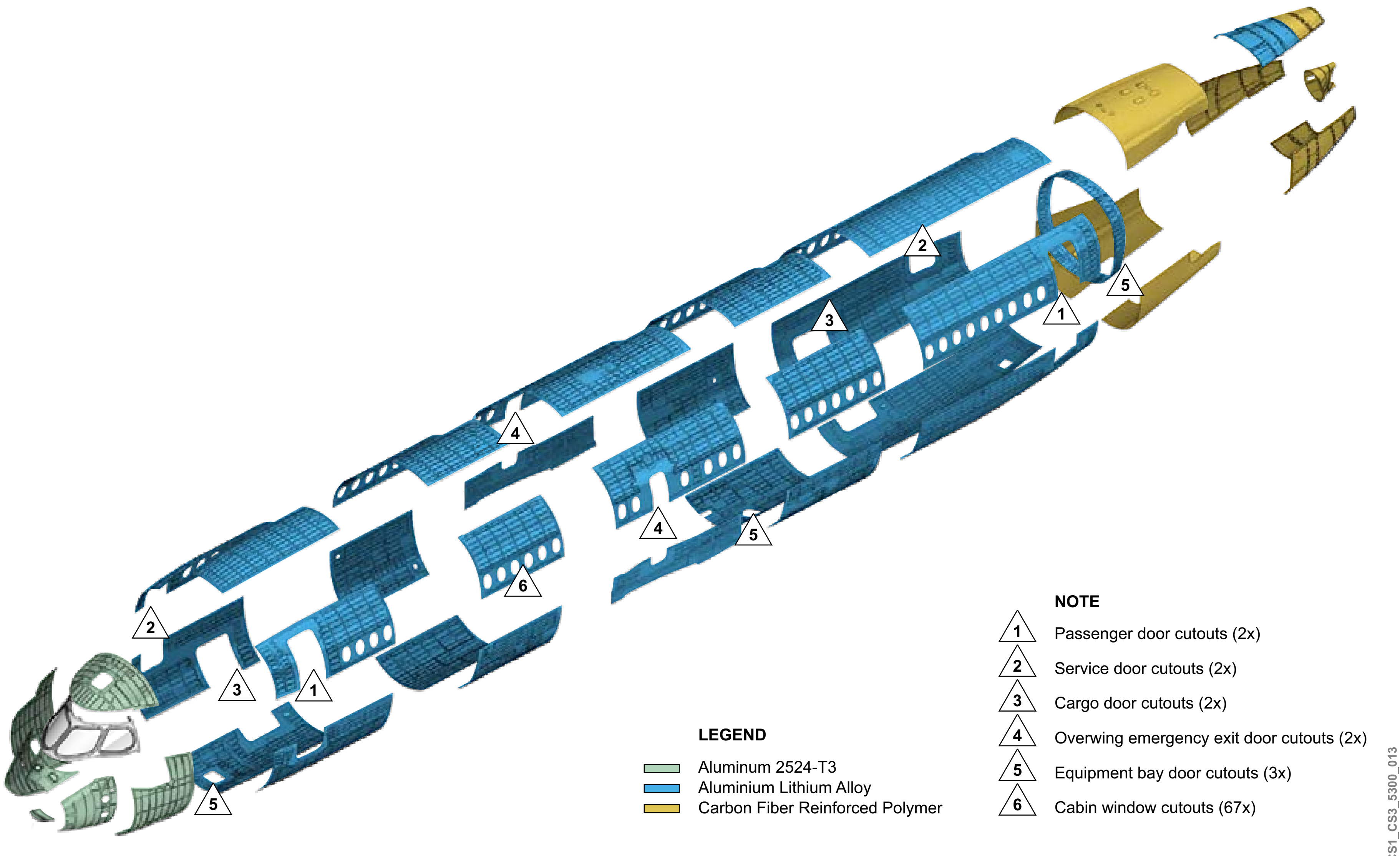


Figure 5: Fuselage Skin Panels

SKIN PANEL ATTACHMENT

Lap joints are used for attaching the skin panels to each other and the fuselage. All lap joints use the triple fastener row method for added strength.

Butt joints are used for attaching the skins to the frames. The butt joints use the double row fastener method, and straps to strengthen the joints.

Depending on the location, some skin panels are attached to the frames with shear ties. This is known as a floating installation. Shear ties are used where attachment directly to the frame could cause weakening of the frame.

In other locations, the panels are attached directly to the frames, known as a standalone or non-floating installation.

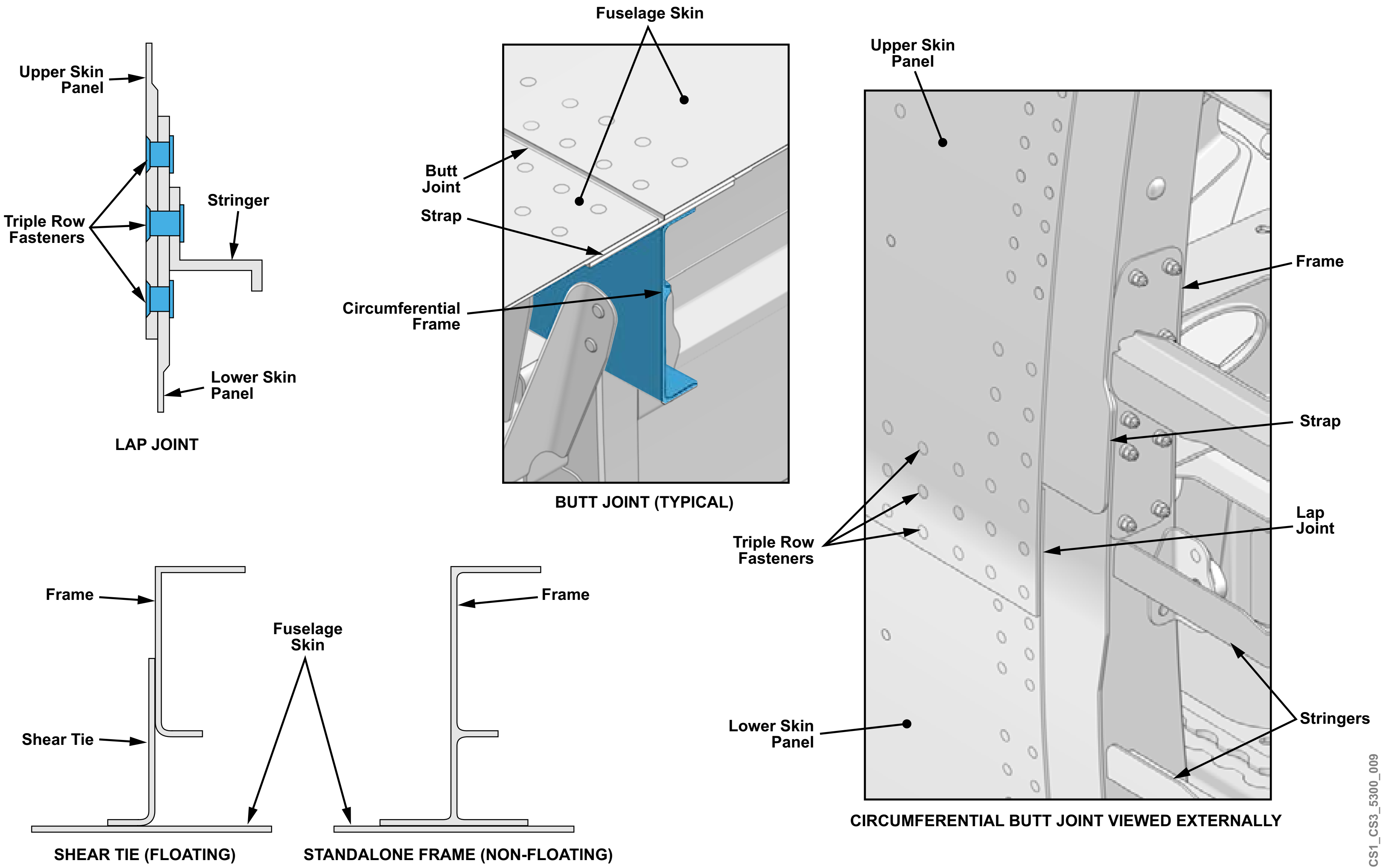


Figure 6: Skin Panel Attachment

53-10 NOSE FUSELAGE

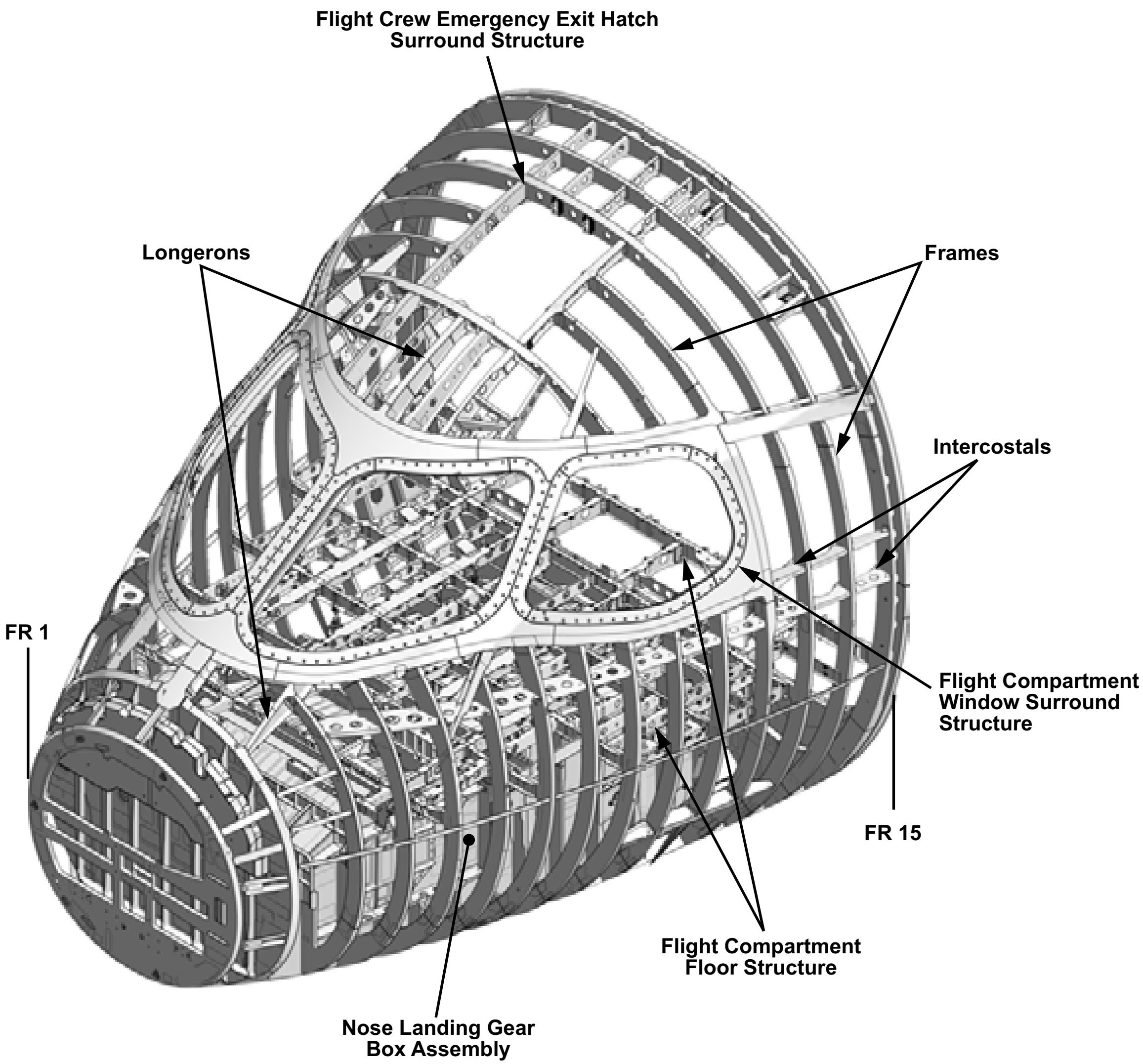
GENERAL DESCRIPTION

The nose fuselage extends from FR 1 to FR 15, where it attaches to the forward fuselage with a circumferential butt joint.

The nose fuselage section is constructed of Al 2425-T3 alloy. The nose fuselage primary structure is a frame/longeron/intercostal structure. The frame pitch is reduced, the skin panels are long circumferentially, and there are relatively few axial stiffeners.

The nose fuselage houses the following structures:

- Nose landing gear box assembly
- Flight crew emergency exit hatch surround structure
- Flight deck window surround structure
- Flight deck floor structure



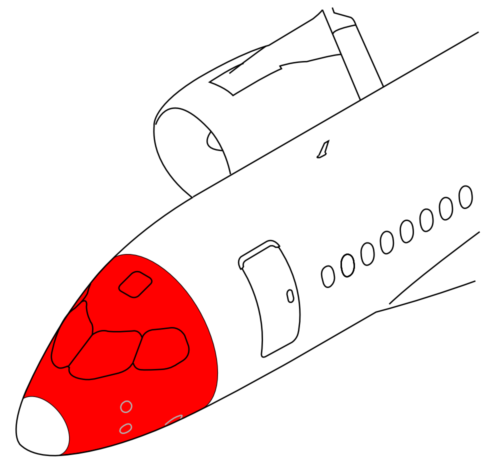


Figure 7: Nose Fuselage Structure

DETAILED COMPONENT INFORMATION

FRAMES

The fifteen nose section frames are machined from conventional 7000-series aluminum with a C-shape profile and radial ribs. Due to a reduced frame pitch, and the use of longerons for axial support, stringers are unnecessary.

INTERCOSTALS AND LONGERONS

Intercostals are located between frames where needed, and together with the longerons provide longitudinal stability to the nose fuselage. The intercostals and longerons are mainly C-shaped.

FORWARD BULKHEAD

The forward bulkhead is located at FR 1 and is made from machined aluminum. The forward bulkhead supports the radome and radome systems. Aft of the forward bulkhead are two large bird strike barriers made of glass fiber shells, aluminum webs, and honeycomb cores for energy absorption.

FORWARD PRESSURE BULKHEAD

The forward pressure bulkhead is located at FR 2 and is machined from 7000-series aluminum plate. It features integrally-machined stiffeners (both sides), and an integral attachment flange. It also makes up the forward end of the nose landing gear box.

NOSE LANDING GEAR BOX AFT PRESSURE BULKHEAD

The nose landing gear box aft pressure bulkhead is located at FR 15, and is machined from aluminum plate. It makes up the lower section of the last frame in the nose fuselage, and is the aft end of the nose landing gear (NLG) box.

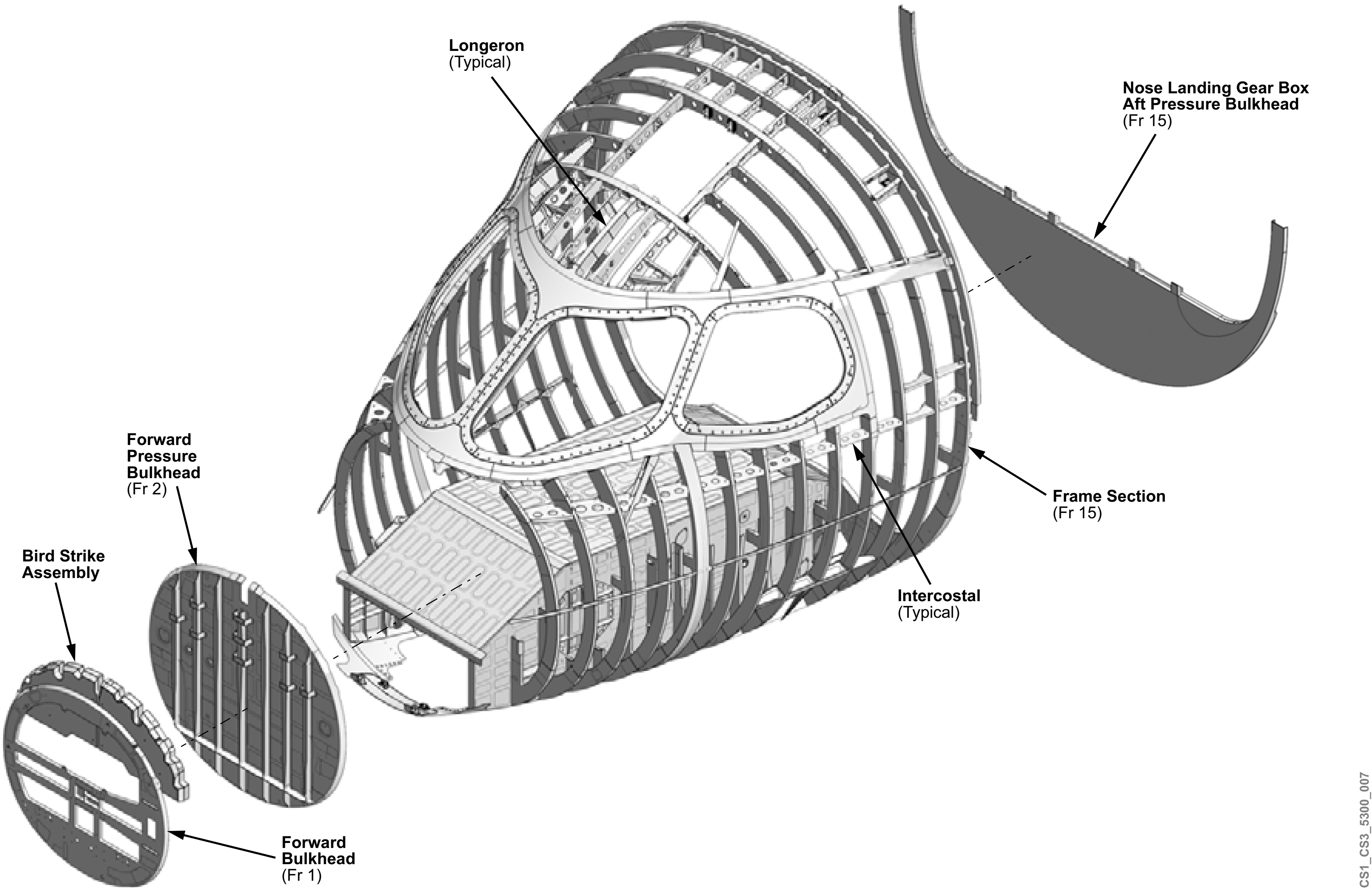


Figure 8: Nose Fuselage Components

NOSE LANDING GEAR BOX STRUCTURE

The nose landing gear box structure consists of two machined 7000-series aluminum sidewalls with integral stiffeners, and 12 side beams. The top panel (pressure floor) is composed of six integrally-stiffened machined plates. The front and rear of the box are enclosed by the nose section forward (FR 2), and aft (FR 15) pressure bulkheads, which absorb the vertical loads from the nose landing gear structure, and the pressure imbalance between the pressurized flight deck and unpressurized nose landing gear box.

FLIGHT CREW EMERGENCY EXIT HATCH SURROUND

The box-type escape hatch surround structure ensures that fuselage loads pass around the flight crew emergency exit hatch. It forms part of the canopy assembly and is made up of two machined 7000-series aluminum frames, and two side beams. The forward frame is FR 10, and the aft frame is FR 13.

FLIGHT DECK WINDOW SURROUND

The window surround structure consists of the following structural components:

- Upper sills
- Lower sills
- Center post
- Side posts
- Aft posts

All of the structural components are machined from 7000-series aluminum plate.

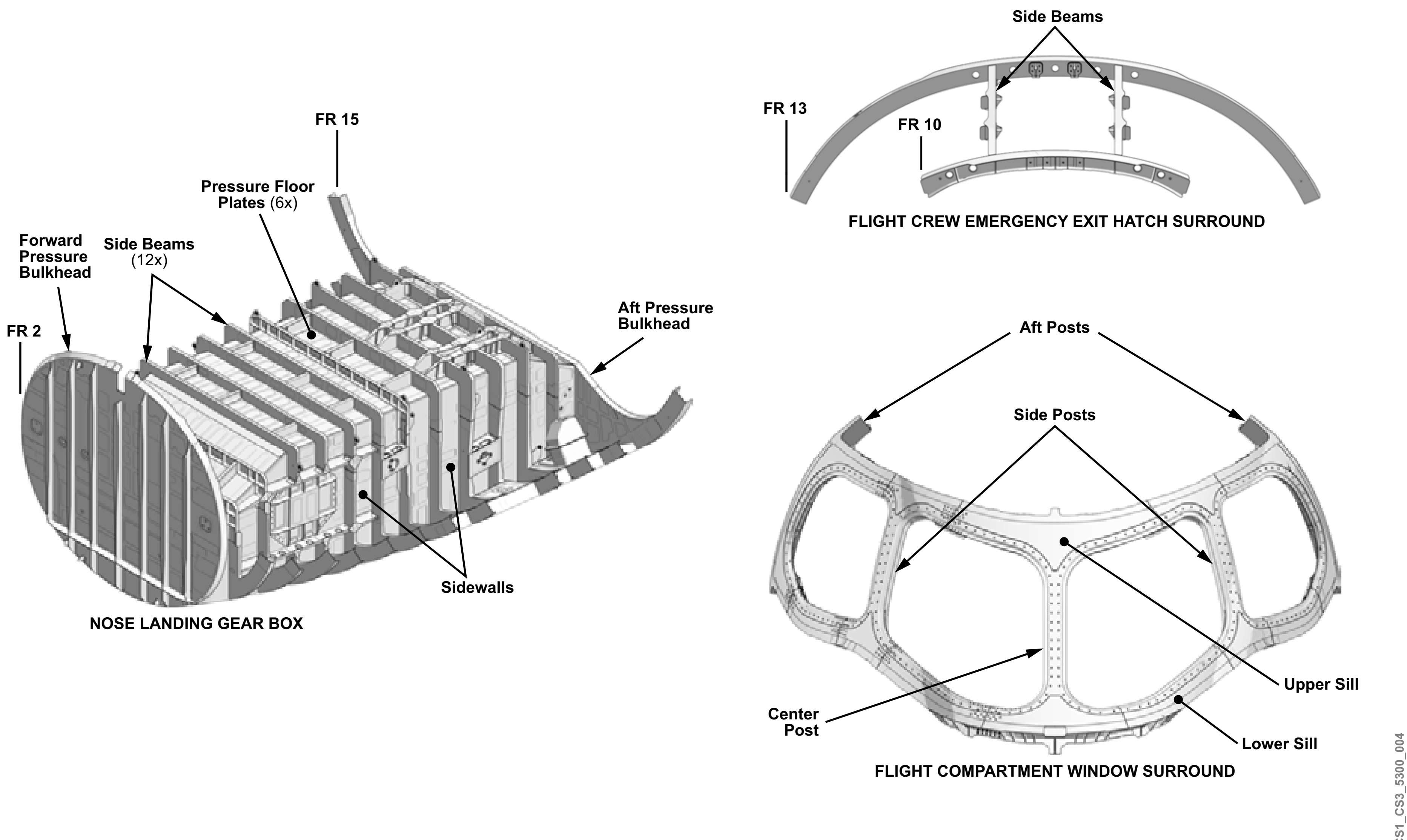


Figure 9: Nose Fuselage Surround Structures

NOSE FUSELAGE FLOOR SUPPORT STRUCTURE

The floor support consists of longitudinal and transverse floor beams, intercostals, floor posts, and shear clips. These components support the flight deck floor panels, flight crew seats, and flight deck equipment. Due to the nature of the loading, the longitudinal beams contain cutouts, while the transverse beams are continuous. They are made of a combination of 2000-series aluminum sheet, and 7000-series machined plate.

FLIGHT DECK FLOOR PANELS

The flight deck floor panels are made of a CFRP honeycomb structure, and are mechanically fastened to the nose fuselage floor support structure. The floor panels have cutouts for the center pedestal, rudder pedal assemblies, and flight deck seat tracks.

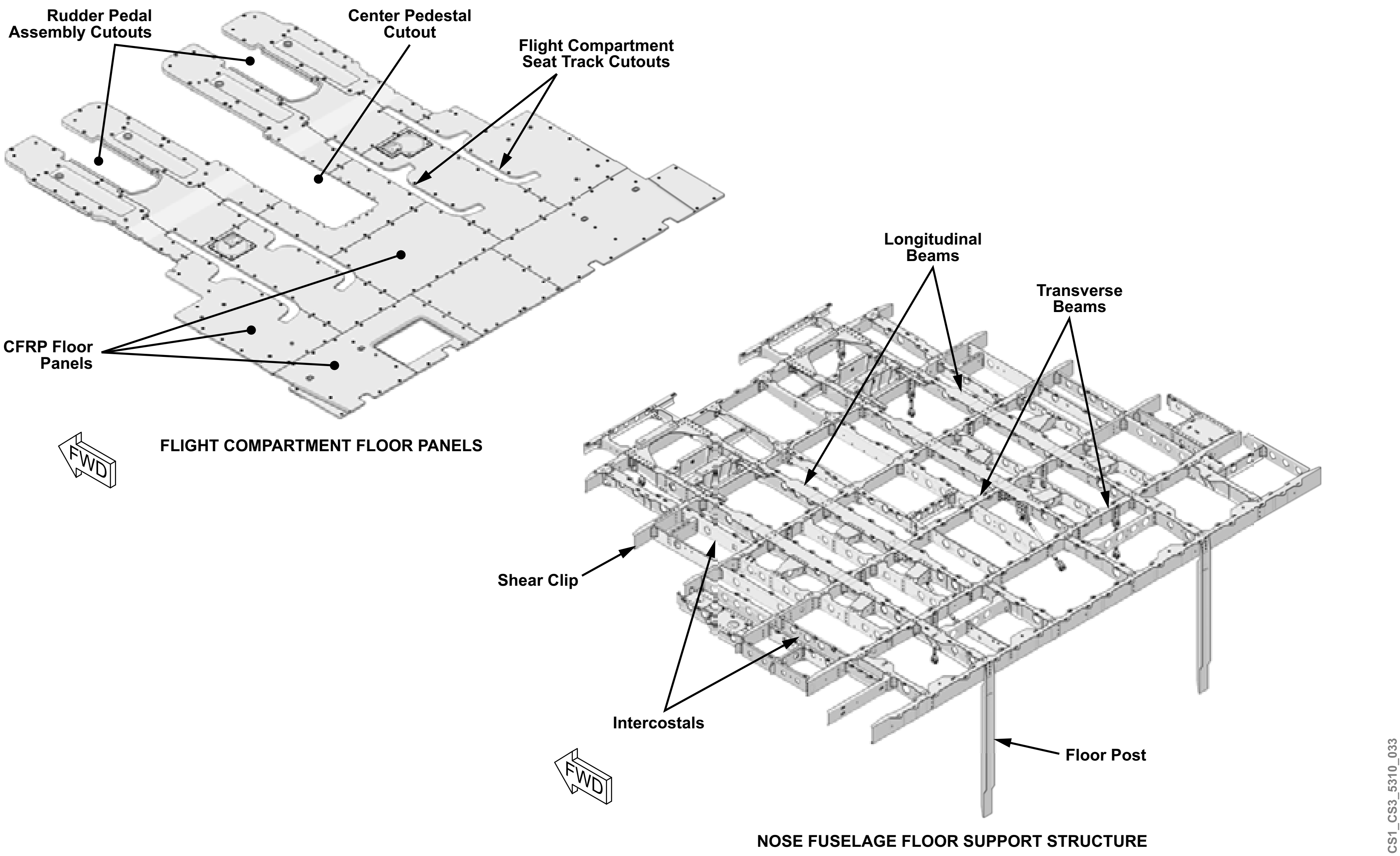


Figure 10: Flight Deck Floor

53-00 PASSENGER CABIN FLOOR SUPPORT STRUCTURE

GENERAL DESCRIPTION

The floor structure, from FR 15 to FR 79, is essentially the same. The only exception is the floor structure in the mid fuselage, which uses support webs between FR 39 and FR 51, above the wing box structure. All of the structures consist of lateral crossbeams and longitudinal floor beams, which are supported vertically by floor posts.

The floor posts are joined to the fuselage frames at their lower ends. The upper ends of the posts are connected to the lateral crossbeams. The crossbeams are machined Al-Li I-beam extrusions and attach to the left and right side frames with shear attachments.

The longitudinal floor beams are machined Al-Li I-beam extrusions and attach to the upper surface of the crossbeams. Additional floor beams are installed where hard points are required for installation of different furnishings in the entrance area. The beams are mounted to the lateral crossbeams.

The floor beams are positioned in the following locations:

- The center floor beam is located at buttock line (BL) 0
- The inboard floor beams are located at BL 23.00 left and right
- The outboard floor beams are located at BL 43.50 left and right

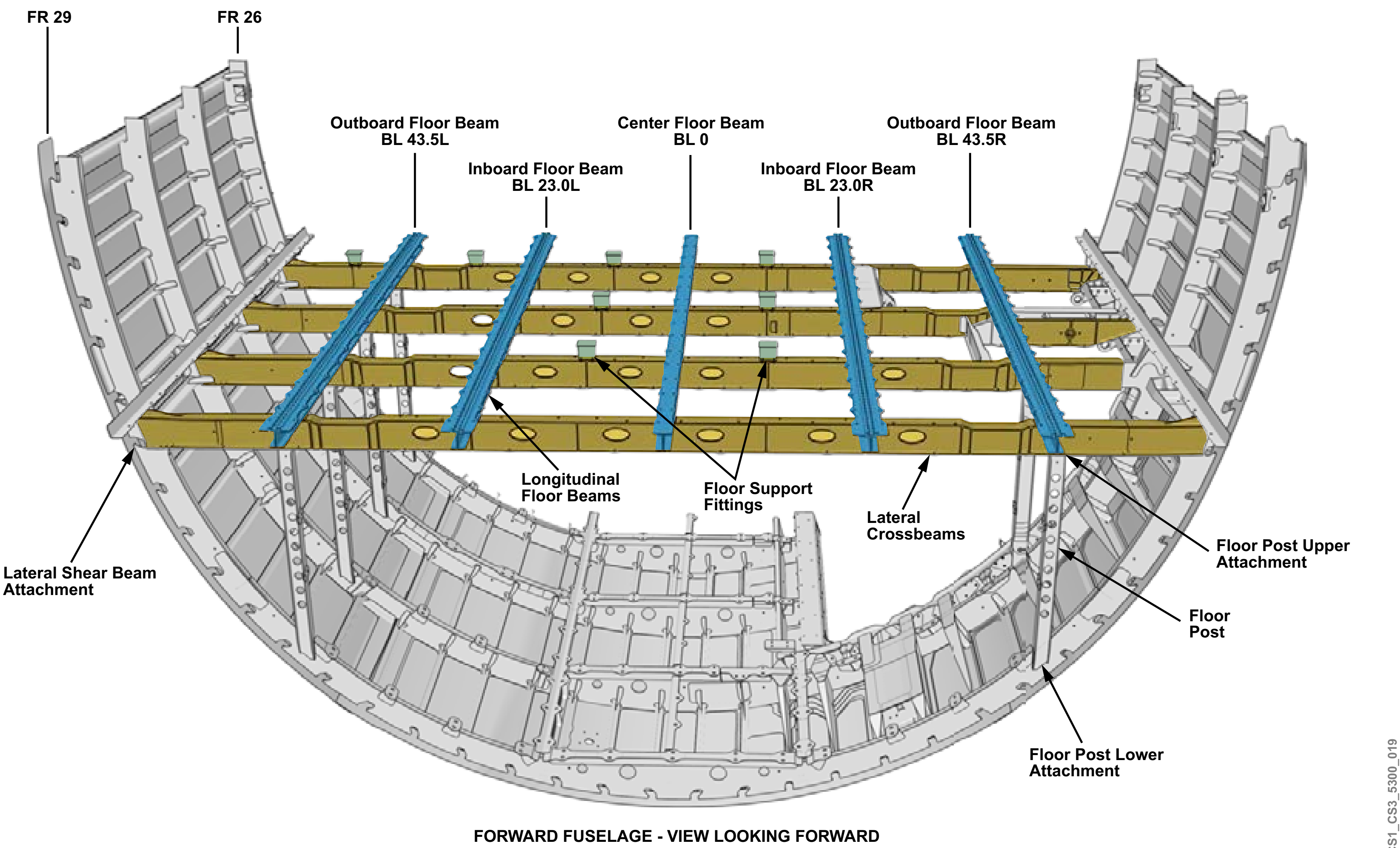


Figure 11: Passenger Cabin Floor Support Structure

DETAILED COMPONENT INFORMATION

CENTER MID FUSELAGE FLOOR STRUCTURE

The center mid fuselage floor support structure differs from the other fuselage sections, due to the mid section being located directly above the wing center box structure. For this reason, three pressure floor assemblies are used to seal the pressurized cabin from the unpressurized wing box cavity.

The floor structure is located between FR 39 and FR 51. The mid fuselage floor structure uses longitudinal floor beams, similar to other fuselage sections. Only two lateral floor beams are used, while titanium angles are used to maintain alignment of the longitudinal beams.

In the center mid fuselage section, support webs are used to support the longitudinal beams rather than floor posts.

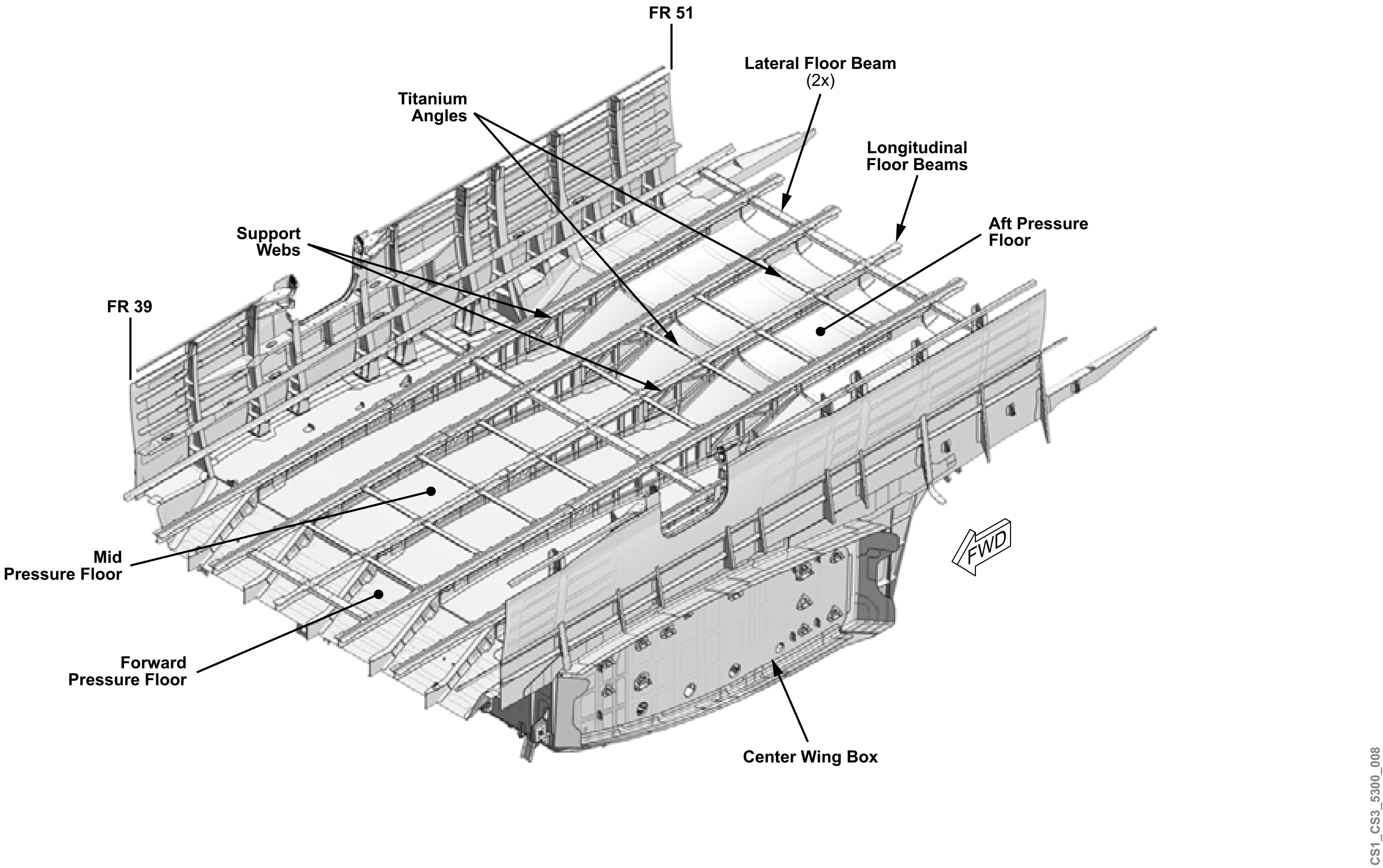


Figure 12: Center Mid Fuselage Floor Structure

FLOOR PANELS

There are three grades of floor panels used throughout the aircraft. The floor panels are industry standard sandwich panels. The grading of the floor panels differs per location. The highest loading grade (Grade III) panels are used in the galley and high-traffic entrance areas. Medium loading grade (Grade II) panels are used in the aisles, lavatories, and the aft entrance area. The lowest loading grade (Grade I) panels are used in the sections were the passenger seats are installed.

The floor panels are mounted with a layer of polyurethane tape installed on the floor beam, and sealing plates. The plates provide an interface between the adjacent panels. The floor panels are secured with bolts which pass through the panel and the sealing strips, and are secured with anchor nuts. An injectable sealant is used between the seams of the panels to seal against moisture and debris.

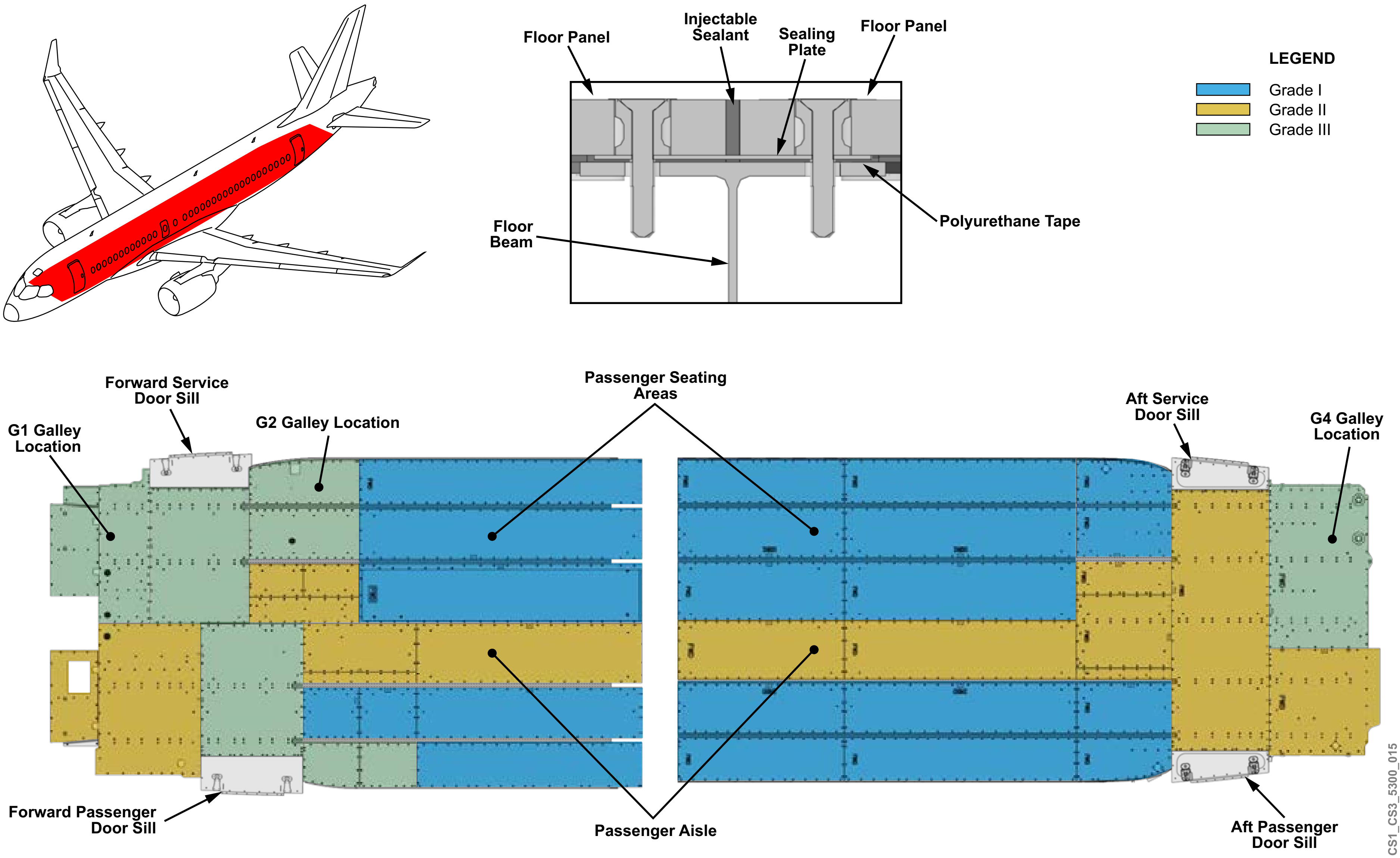


Figure 13: Floor Panels

53-20 FORWARD FUSELAGE

GENERAL DESCRIPTION

The forward fuselage starts at the circumferential splice at FR 15, where it connects to the nose fuselage. The forward fuselage ends at the circumferential splice at FR 30.

The aerodynamic curvature of the nose fuselage continues in the forward fuselage section up to FR 24. Aft of FR 24 the forward fuselage has a constant cross section.

This fuselage section contains the forward equipment (avionics) bay, located between FR 15 and 22, as well as the forward portion of the forward cargo compartment, between FR 22 and FR 30.

The forward fuselage has surround structures for the following items:

- Forward passenger door
- Forward service door
- Forward cargo door
- Forward equipment bay door

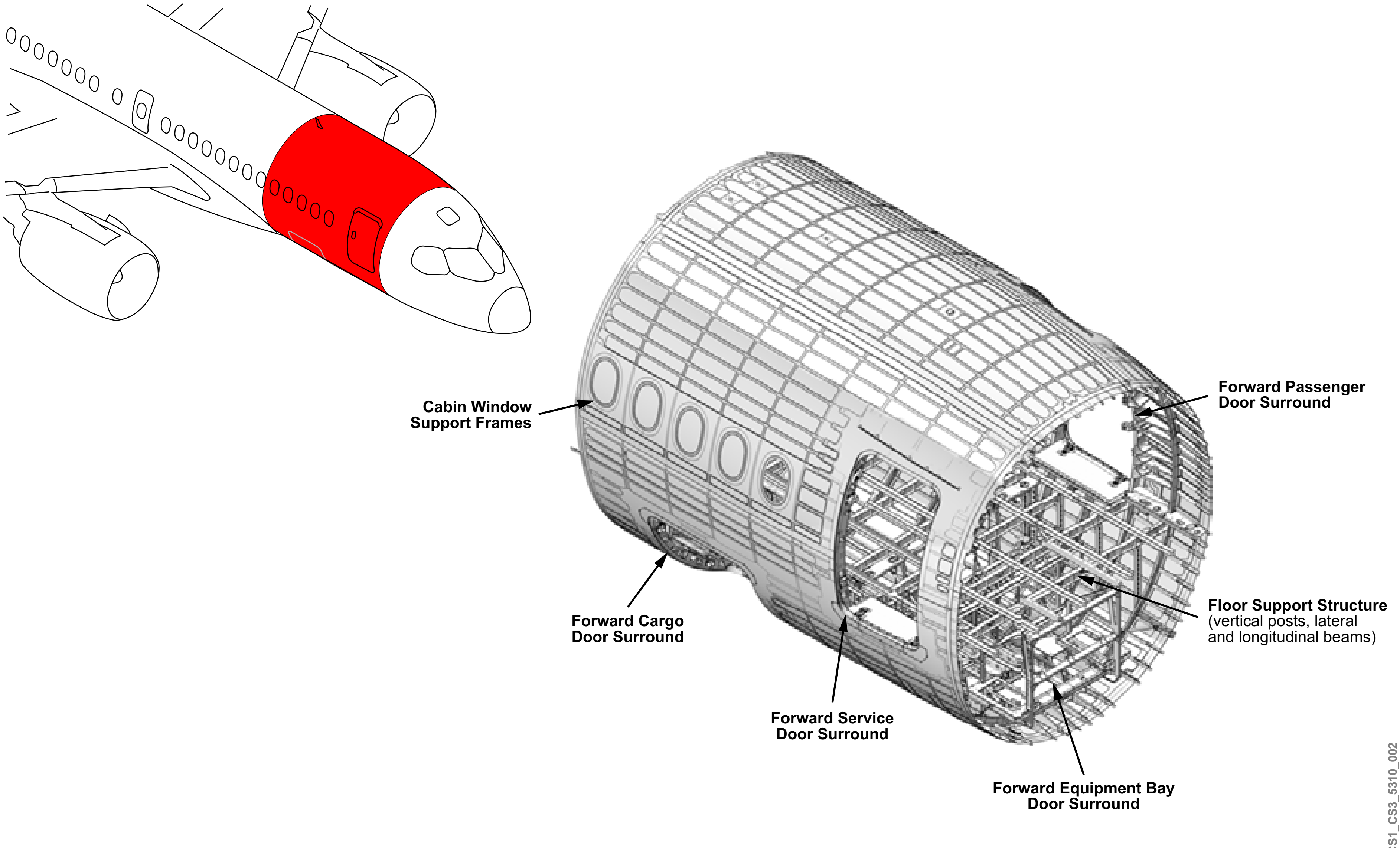


Figure 14: Forward Fuselage

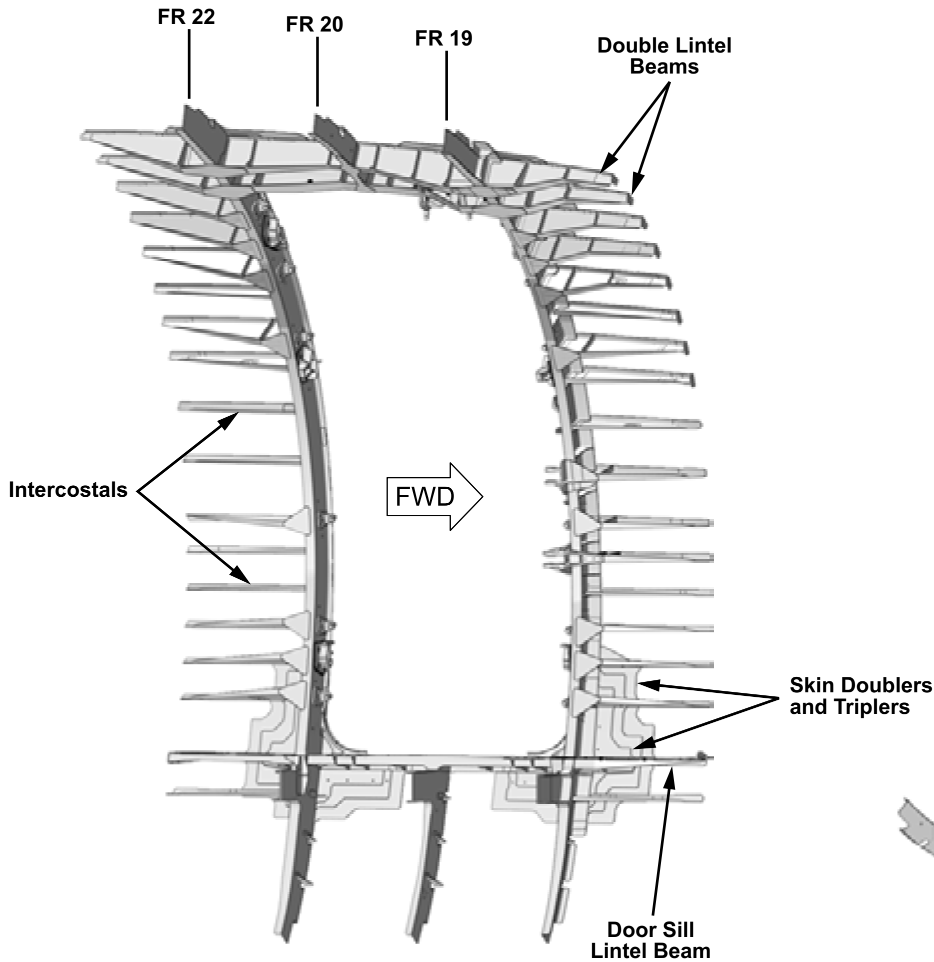
DETAILED COMPONENT INFORMATION

PASSENGER AND SERVICE DOOR SURROUND STRUCTURES

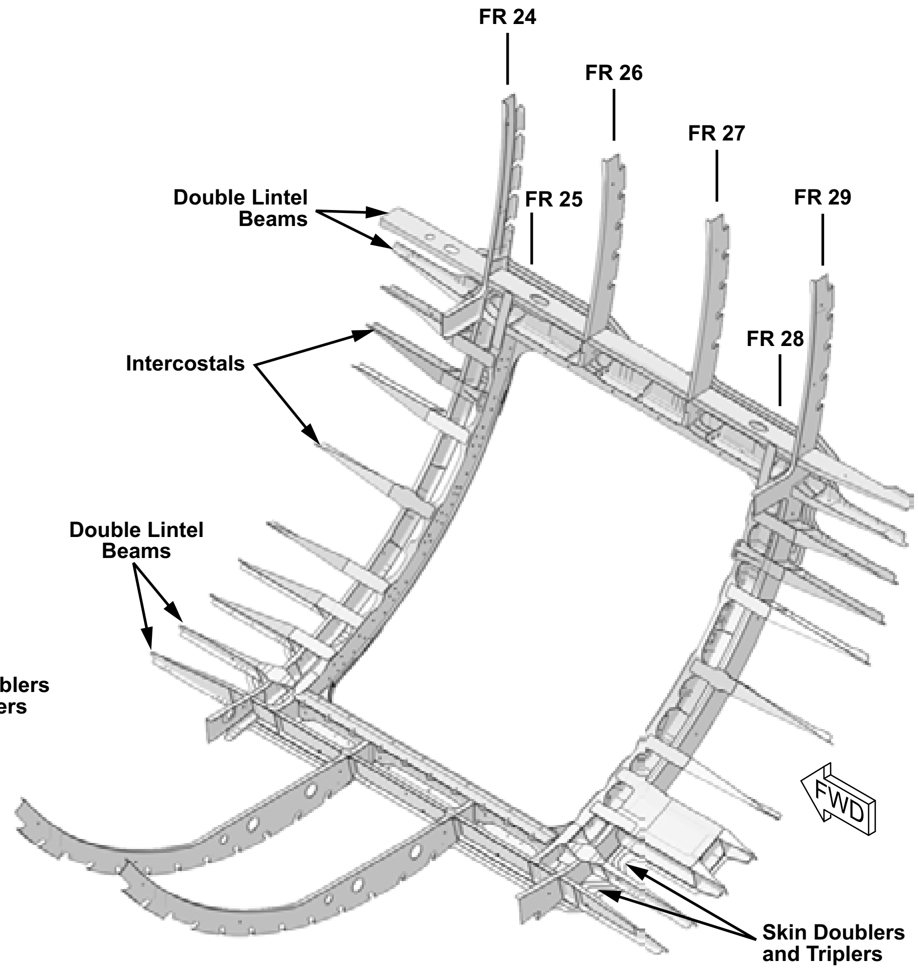
The structure around the passenger and service door is reinforced by skin doublers and triplers, lintel beams, and intercostals. Two lintel beams reinforce the upper section of the door cutout, and a single lintel beam reinforces the lower section of the door cutout. Doublers and triplers are used to reinforce the fuselage skin around the cutout.

CARGO DOOR SURROUND STRUCTURE

A double frame is used for the reinforcement of the fuselage at the cargo door cutout. The two additional local frames are numbered FR 25 and FR 28. Two lintel beams are used at the upper and lower sill of the cargo door surround structure. The fuselage skin is locally reinforced around the cargo door cutout with a doubler and a tripler.



FORWARD PASSENGER DOOR SURROUND STRUCTURE
(FORWARD SERVICE DOOR SURROUND STRUCTURE SIMILAR)



FORWARD CARGO DOOR SURROUND STRUCTURE

CS1_CS3_5320_002

Figure 15: Forward Passenger, Service, and Cargo Door Surround Structures

FORWARD EQUIPMENT BAY DOOR SURROUND

This surround structure is located below the cabin floor, and spans from FR 15 to FR 22.

The opening to the forward equipment bay is located between FR 17 and FR 19. The opening structure has an auxiliary frame at FR 19, a skin doubler, and a shim for door alignment and sealing.

Machined aluminum lintels (beams) are located on the left and right sides of the surround structure to provide additional strength. Intercostals, which attach to the surround structure with aluminum gussets, are located forward and aft of the equipment bay opening.

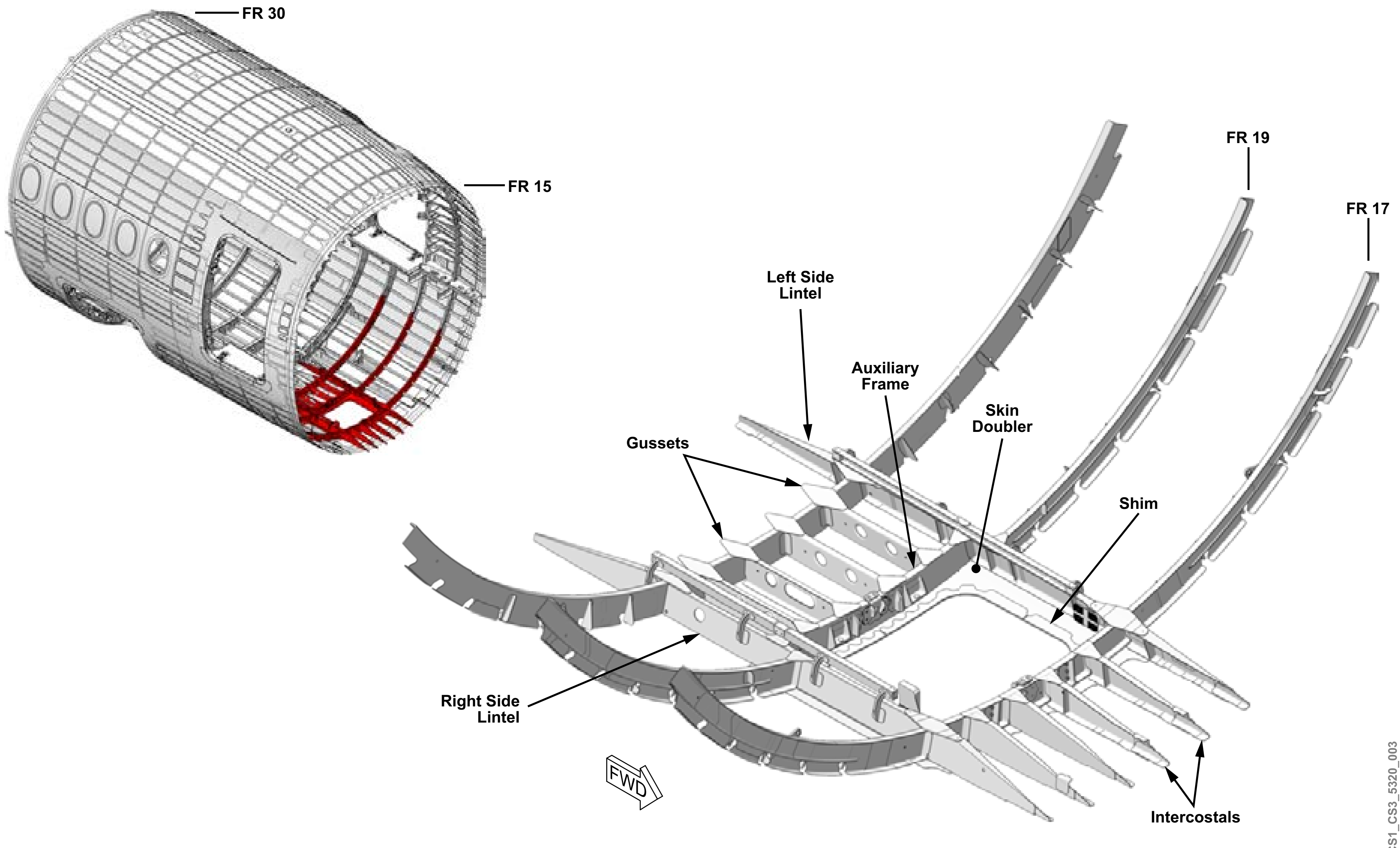


Figure 16: Forward Equipment Bay Door Surround

53-30 FORWARD MID FUSELAGE

GENERAL DESCRIPTION

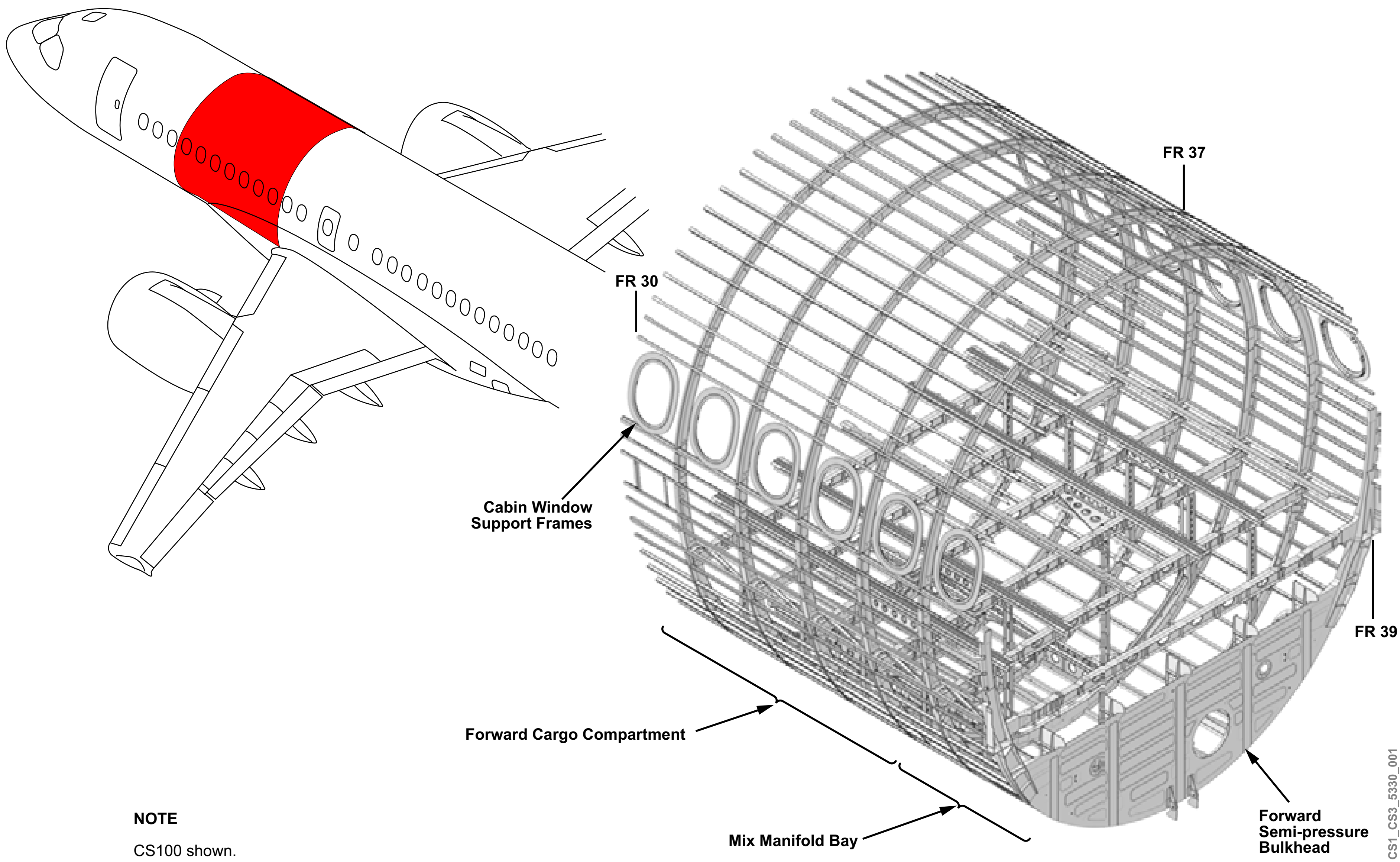
The forward mid fuselage starts at the circumferential splice at FR 30, where it connects to the forward fuselage section. The forward mid fuselage ends at FR 39. This fuselage section has a constant fuselage cross section between FR 30 and FR 39.

The forward mid fuselage section contains a portion of the forward cargo compartment, from FR 30 to FR 37. The air conditioning system mix manifold bay is located between FR 37 and FR 39. The forward semi-pressure bulkhead is located at FR 39.

The forward mid fuselage includes:

- Cabin window support frames
- Forward semi-pressure bulkhead

On the CS300, four additional frames are added to the front of the forward mid fuselage.



NOTE

CS100 shown.

Figure 17: Forward Mid Fuselage

DETAILED COMPONENT INFORMATION

EXTERNAL LONGERON

The external longeron distributes the load from the wing into the fuselage skin, and is tapered into the fuselage skin between FR 38 and FR 41. This longeron is an Al-Li I-beam section installed with bolts. The longeron is used to relieve the load from the connection between the flat side skin panel, the center wing box cruciform fitting, and the forward semi-pressure bulkhead.

The external stringer profile distributes the load longitudinally from the flat side skin panel above the center wing box in the mid fuselage to the forward mid fuselage at stringer 23, the location of the kink in the flat side skin panel.

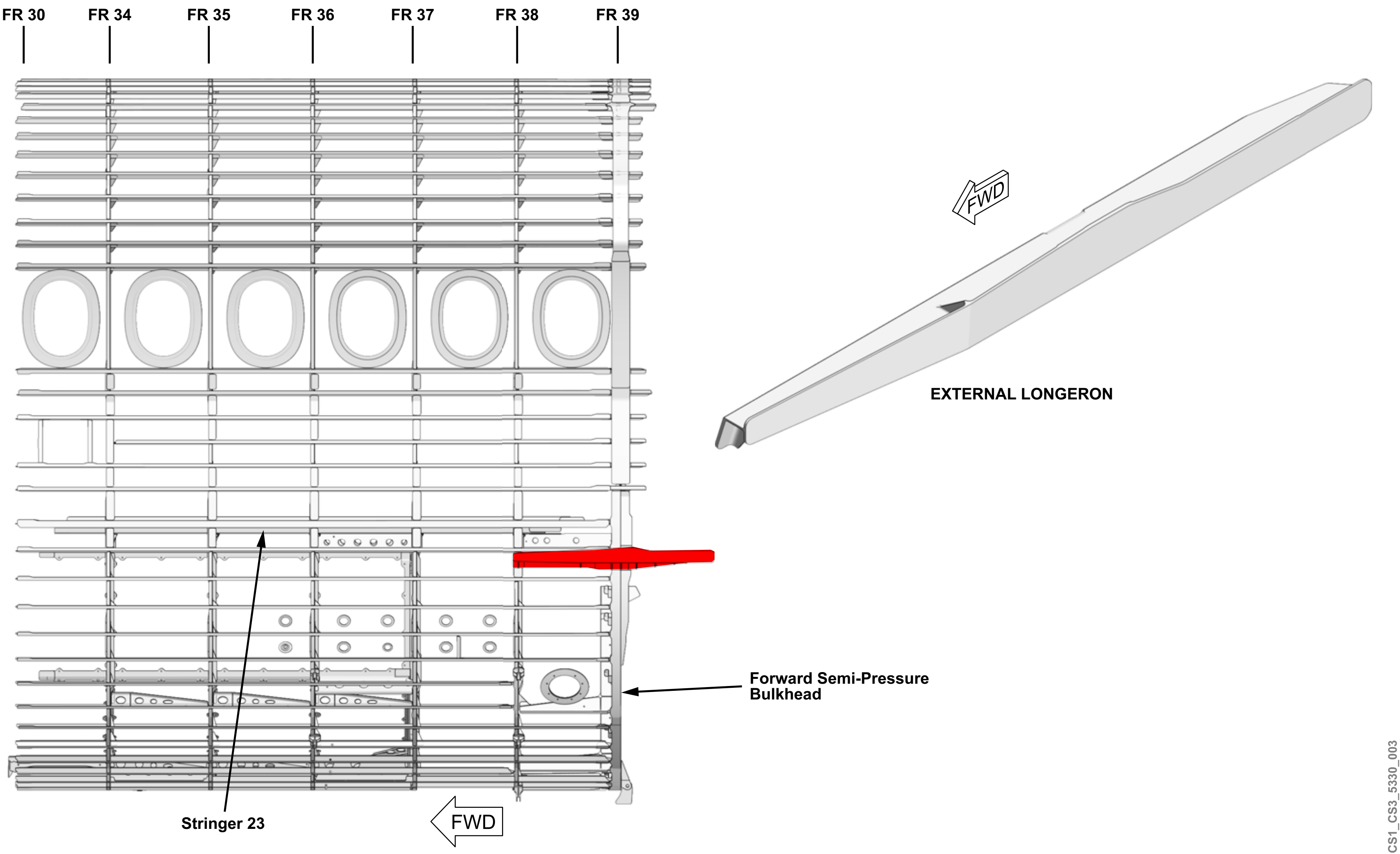


Figure 18: External Longeron

FORWARD SEMI-PRESSURE BULKHEAD

The forward semi-pressure bulkhead is a flat machined Al-Li plate with integrally machined stiffeners. The bulkhead is stiffened by reinforcing beams. The upper beams interface with the floor structure over the center wing box, while the lower beams connect to the mid fuselage under floor structure aft of FR 39. Chemically milled pockets are provided to reduce the bulkhead's thickness in areas between structural members. The bulkhead includes a circular reinforced hole for the installation of the outflow valve.

A sloped forward pressure floor assembly is installed above the semi-pressure bulkhead. The pressure floor is a flat machined plate with integral stiffeners, which is reinforced by reinforcing beams. The sloped pressure floor is made of Al-Li alloy.

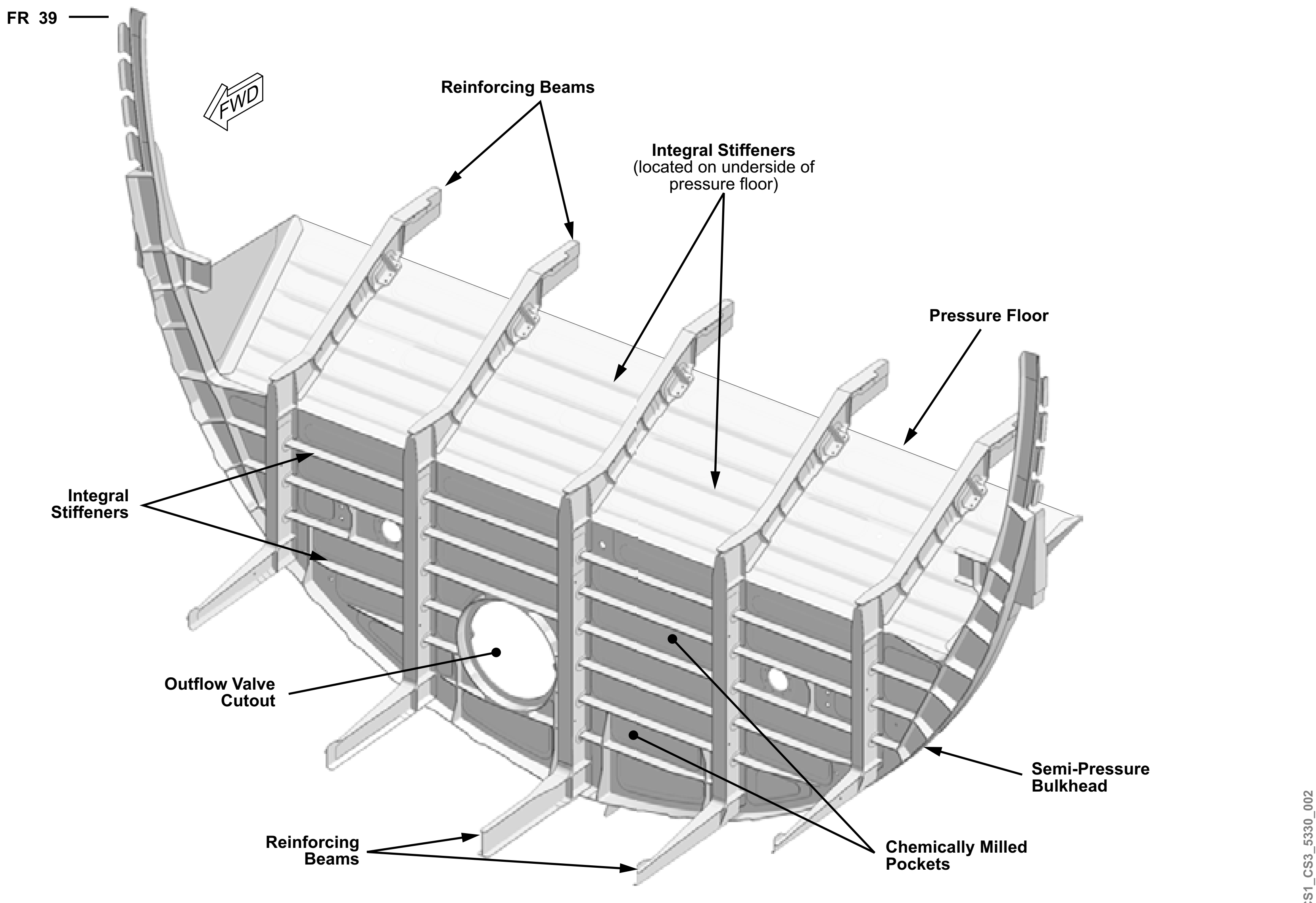


Figure 19: Forward Semi-Pressure Bulkhead

53-40 CENTER MID FUSELAGE

GENERAL DESCRIPTION

The center mid fuselage starts at FR 39, ends at the circumferential joint at FR 51, and has a constant cross section.

There are two door surround structures in the center mid fuselage, for the left and right overwing emergency exit doors.

The center mid fuselage is located directly above the wing center box structure. The center box structure attaches to the fuselage with left and right upper cruciform fittings, as well as frame extensions at FR 40 and FR 48.

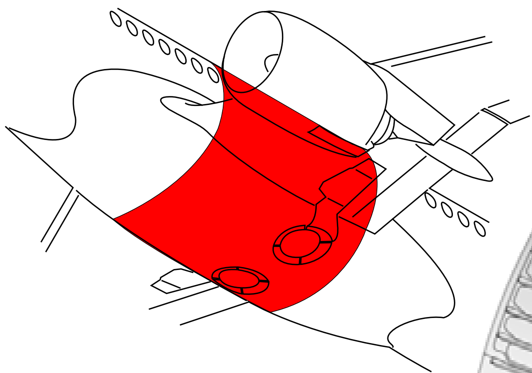
A single keel beam is located under the wing box to provide support for the wing-to-body (WTBF) fairing. A pressure floor is located between FR 48 and FR 51, to separate the unpressurized wing box cavity area from the pressurized cabin.

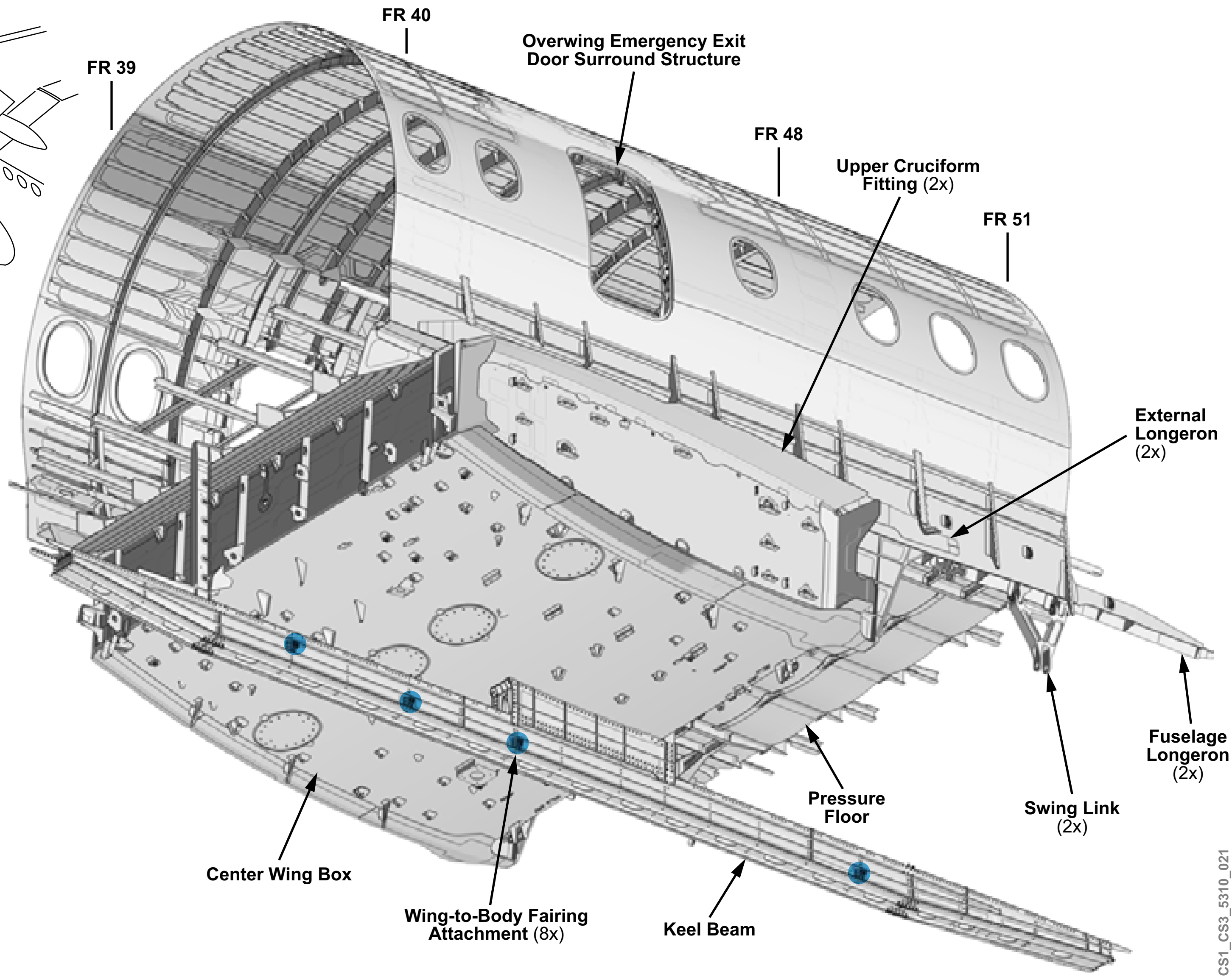
Two fuselage longerons are secured to the mid fuselage structure, on the left and right sides. Two external longerons are located between FR 46 and FR 49.

Swing links are attached at FR 50 and FR 51, to support the main landing gear (MLG) support beam.

This section includes:

- Floor support webs
- Wing attach fittings
- Left and right overwing emergency exit door surrounds
- Cabin window support frames
- Pressure floor assembly





NOTE

CS100 shown.

Figure 20: Center Mid Fuselage

DETAILED COMPONENT INFORMATION

LONGERONS

The center mid fuselage longerons are attached to the fuselage trap panels. The trap panels transfer wing thrust loads to the longerons, which transfer the loads to the fuselage structure. The flat side skin is sandwiched between the mid fuselage longeron, and the external longeron.

The external longeron runs at the level of the kink in the transition area of the side skin, where it changes from flat to curved. The external longerons are similar to the external longerons installed in the forward/mid fuselage structure.

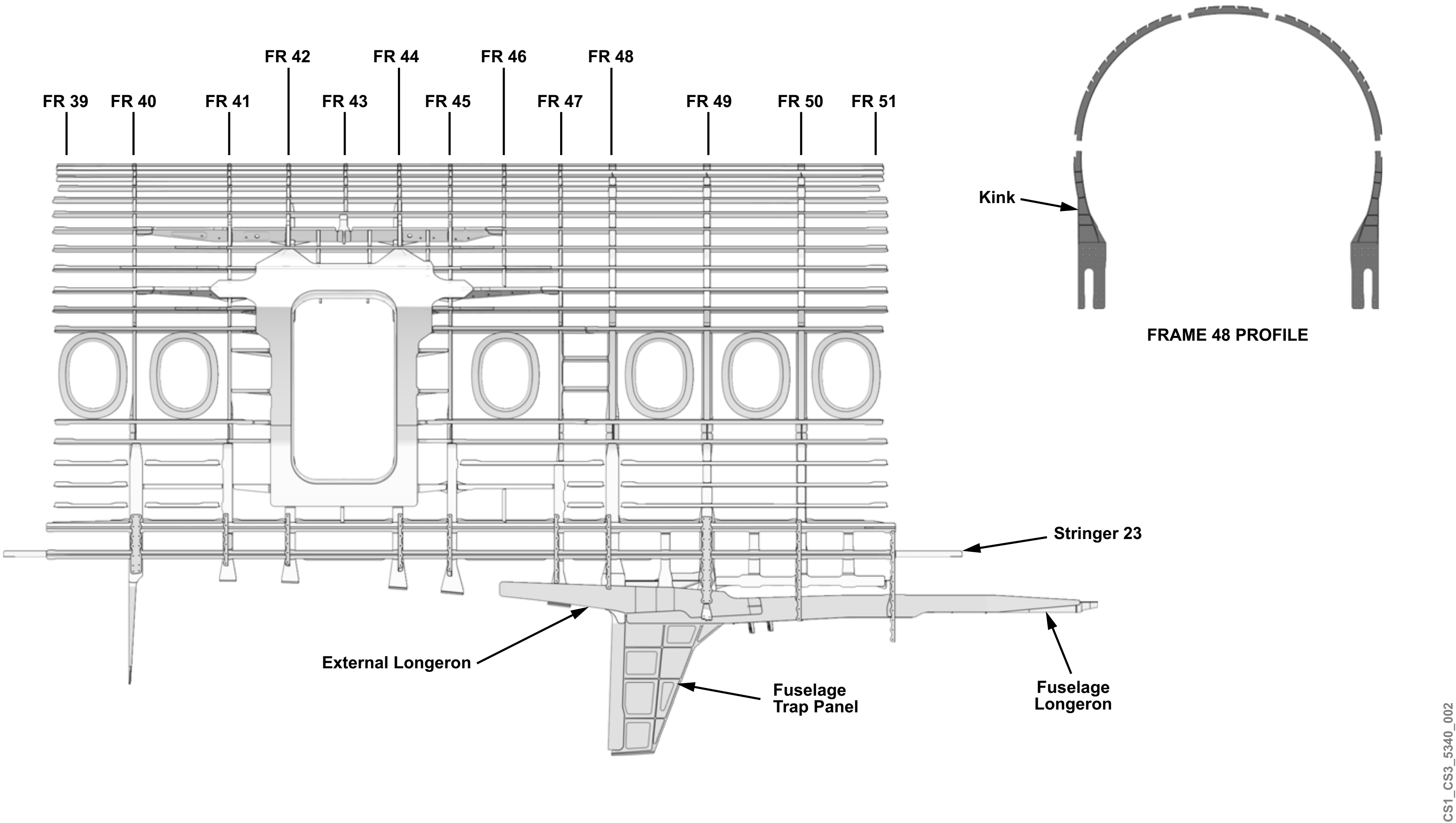


Figure 21: Center Mid Fuselage Longerons

OVERWING EMERGENCY EXIT DOOR SURROUNDS

The overwing emergency exit door (OWEE) surrounds are located on the left and right center mid fuselage section, between FR 42 and FR 44. These two frames comprise the fore and aft sides of the surround structure. The frames are strengthened by intercostals which extend forward and aft of the surround, and attach to the fuselage frames and stringers.

The surround structure consists of two upper lintel beams, with a single lower lintel beam. The lintels extend beyond the surround structure to FR 40 forward and FR 47 aft. Two additional sub frames are located at the upper surround structure for added strength in supporting the OWEE when in the open position. A structural seal is attached to the perimeter of the surround, which is strengthened by double and triple skins.

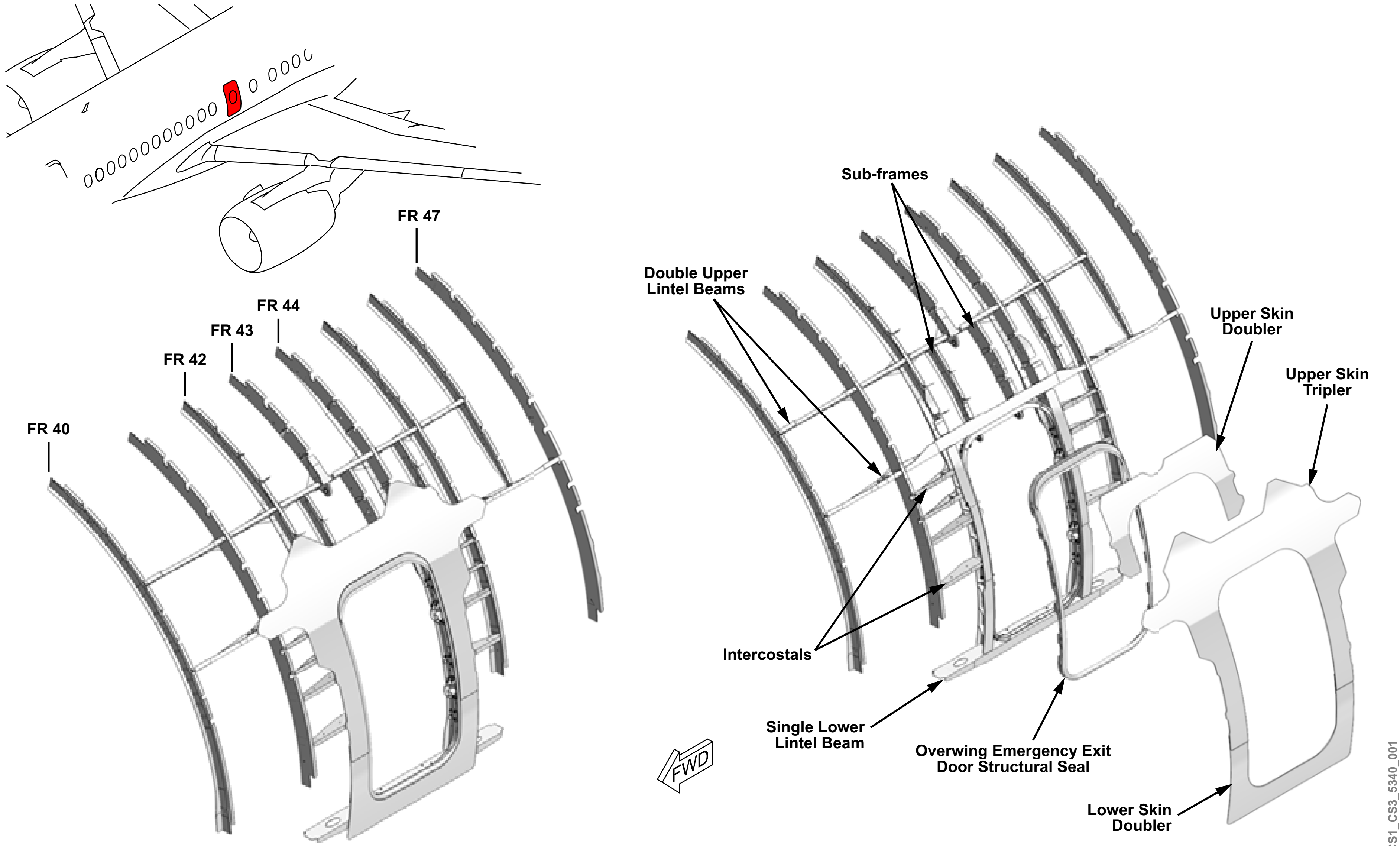


Figure 22: Overwing Emergency Exit Door Surrounds

KEEL BEAM

The keel beam is located beneath the lower mid fuselage (FR 37 to FR 54). It is attached to the rear spar and central rib of the center wing box and to the fuselage front and aft semi-pressure bulkheads. The keel beam section is a box-type assembly of four parts:

- An extruded aluminum upper cap
- A central rib
- Machined aluminum sidewalls
- An extruded aluminum lower plate

The keel beam box assembly also has several extruded inner ribs for added strength. The wing-to-body fairing (WTBF) interfaces with the keel beam at four places on each side of the sidewalls. The lug-type fittings are located at the fairing tie-rods, and machined ribs are added to back it up.

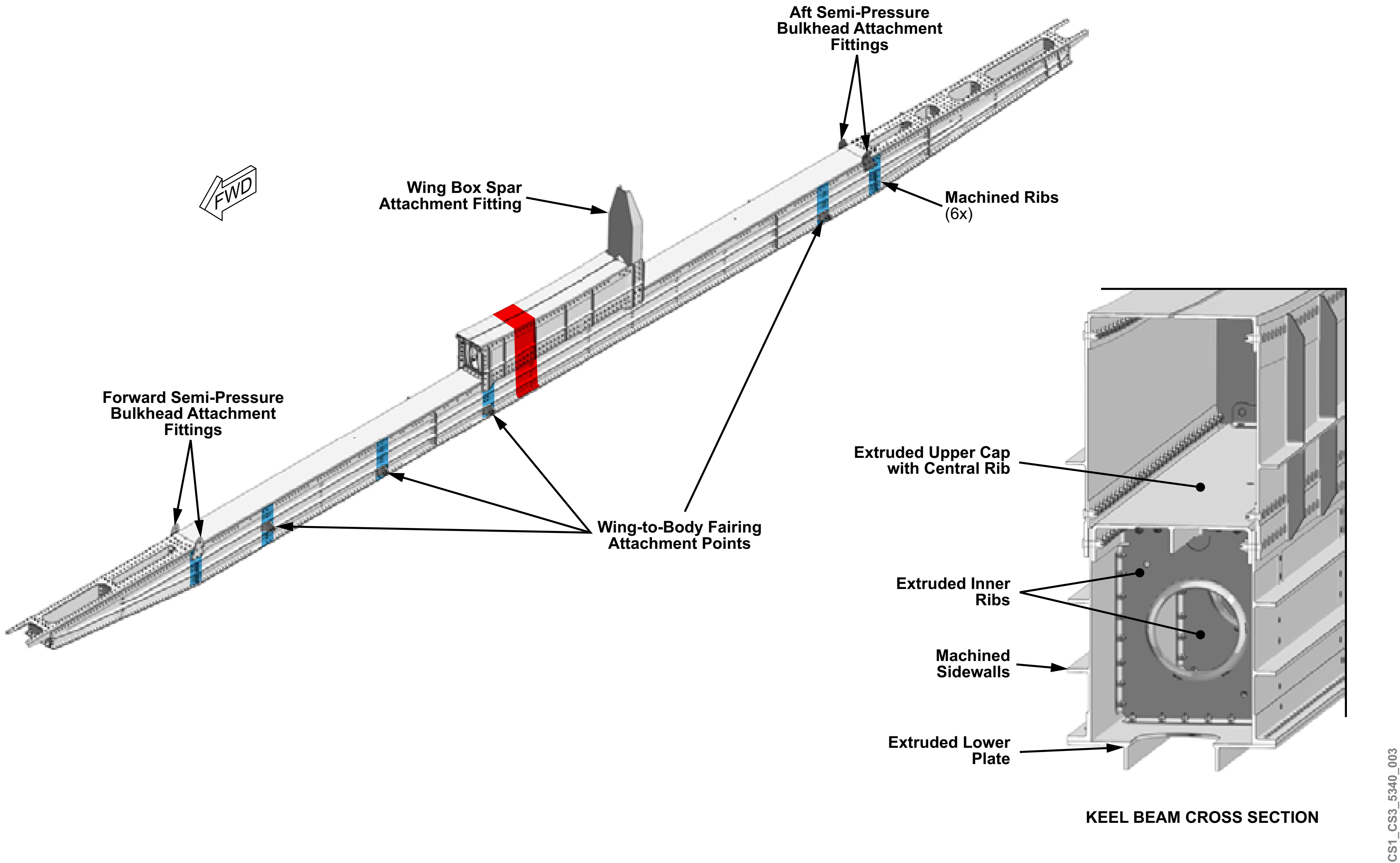
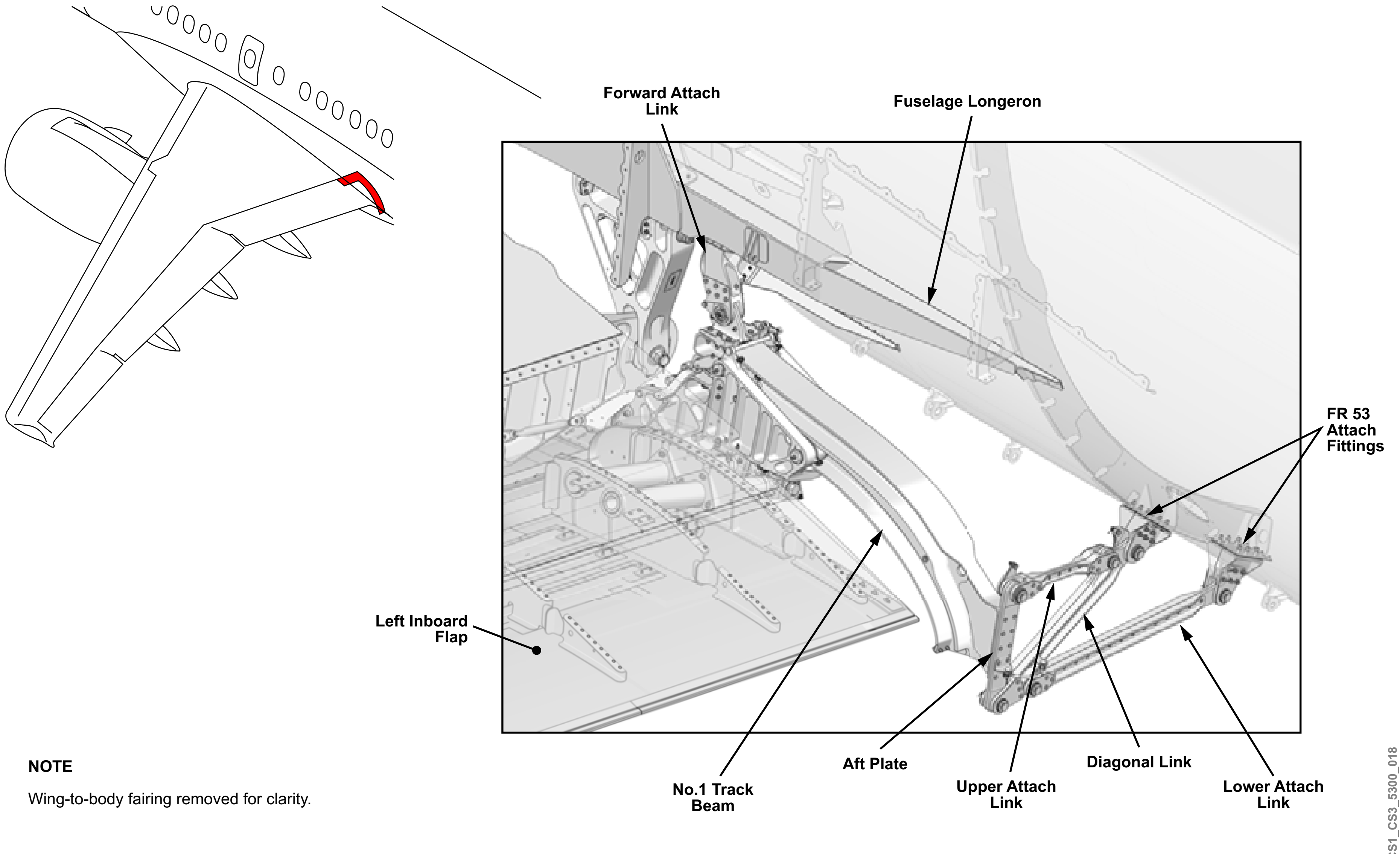


Figure 23: Keel Beam

NO. 1 FLAP TRACK BEAM ATTACHMENT

The left and right no. 1 flap track beams are attached to the fuselage inside the WTBF, and the MLG wheel wells. The track beam forward lug is attached to the forward attach link, which is secured to the fuselage longeron on each side of the fuselage.

The track beam is attached to a plate at the aft end. The plate interfaces with a diagonal link, upper attach link, and a lower attach link. The upper and lower links have attach fittings, which are secured to FR 53 of the fuselage structure.



NOTE

Wing-to-body fairing removed for clarity.

Figure 24: No. 1 Flap Track Beam Attachment

53-50 AFT MID FUSELAGE

GENERAL DESCRIPTION

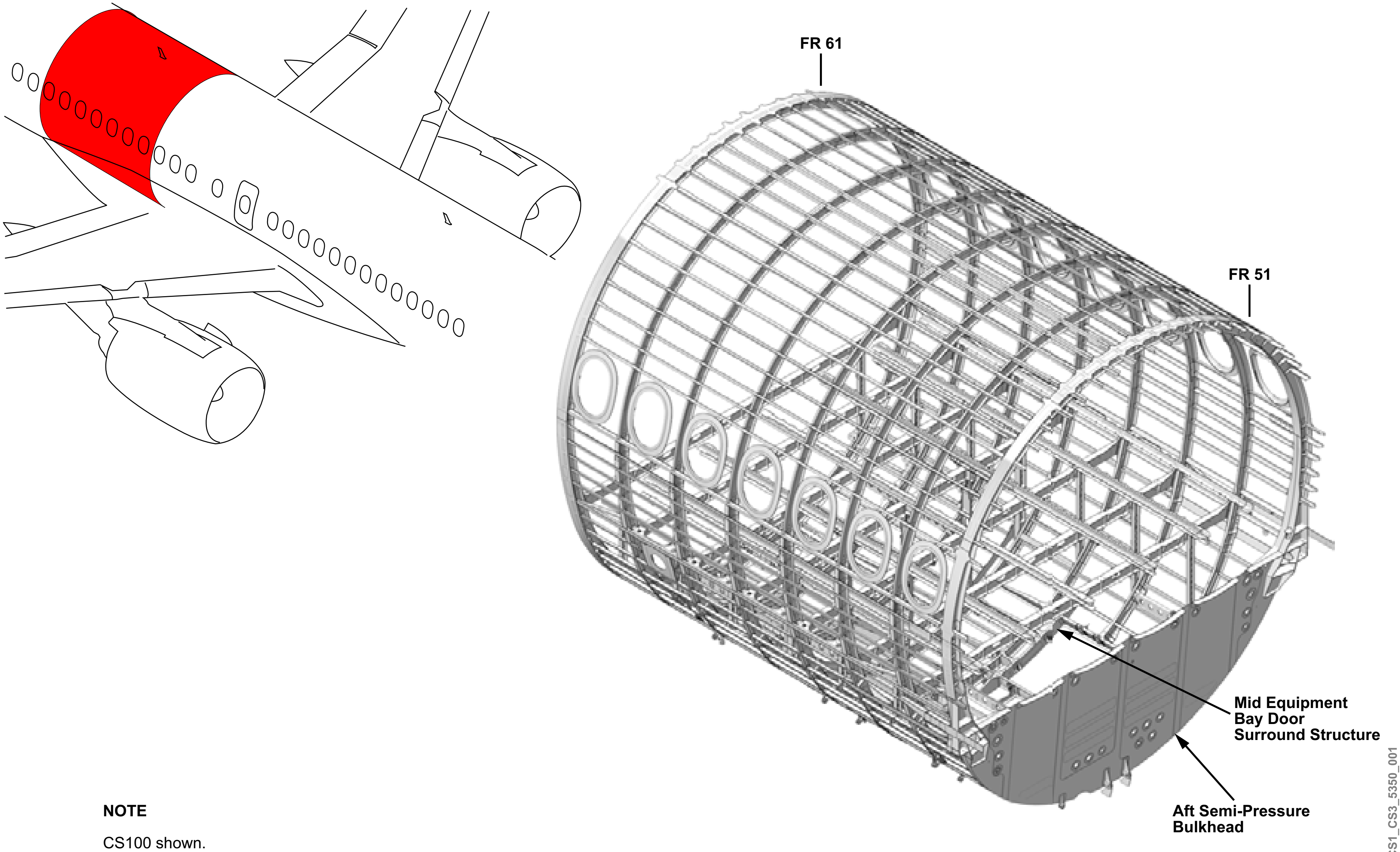
The aft mid fuselage starts at FR 51, and ends at the circumferential splice at FR 61. The aft mid fuselage has a constant fuselage cross section between FR 51 and FR 61.

There is one door surround structure in the aft mid fuselage for the mid equipment bay. The aft semi-pressure bulkhead is located at FR 51.

The aft mid fuselage section includes:

- Cabin window support frames
- Mid equipment bay door surround
- Aft semi-pressure bulkhead

On the CS300, four additional frames are added to the end of the aft mid fuselage.



NOTE

CS100 shown.

Figure 25: Aft Mid Fuselage

DETAILED COMPONENT INFORMATION

AFT SEMI-PRESSURE BULKHEAD

The aft semi-pressure bulkhead is a flat machined Al-Li plate with integrally machined stiffeners. The bulkhead is stiffened by reinforcing beams. Where possible, chemically milled pockets are placed between major structures, locally reducing the bulkhead thickness.

The bulkhead attaches to the floor structure at stringer 23 at the upper section, and to the fuselage under floor structure at the lower section. The bulkhead provides a solid divider between the unpressurized, and pressurized areas under the floor. The bulkhead has various holes machined for system plumbing and wiring installations.

RAT BOX AND RAT SHIELD

The ram air turbine (RAT) box and RAT shield provide structural protection should a blade detach from the RAT.

MID EQUIPMENT BAY DOOR SURROUND

The mid equipment bay door surround structure is a rectangular frame, which is located between FR 52 and FR 54. The frame is offset, and the forward and aft frames do not align with FR 53 and FR 54.

The surround structure has an upper and lower sill, forward and aft auxiliary frames, and an inner reinforcing strap. The frame assembly is strengthened by intercostals and gussets.

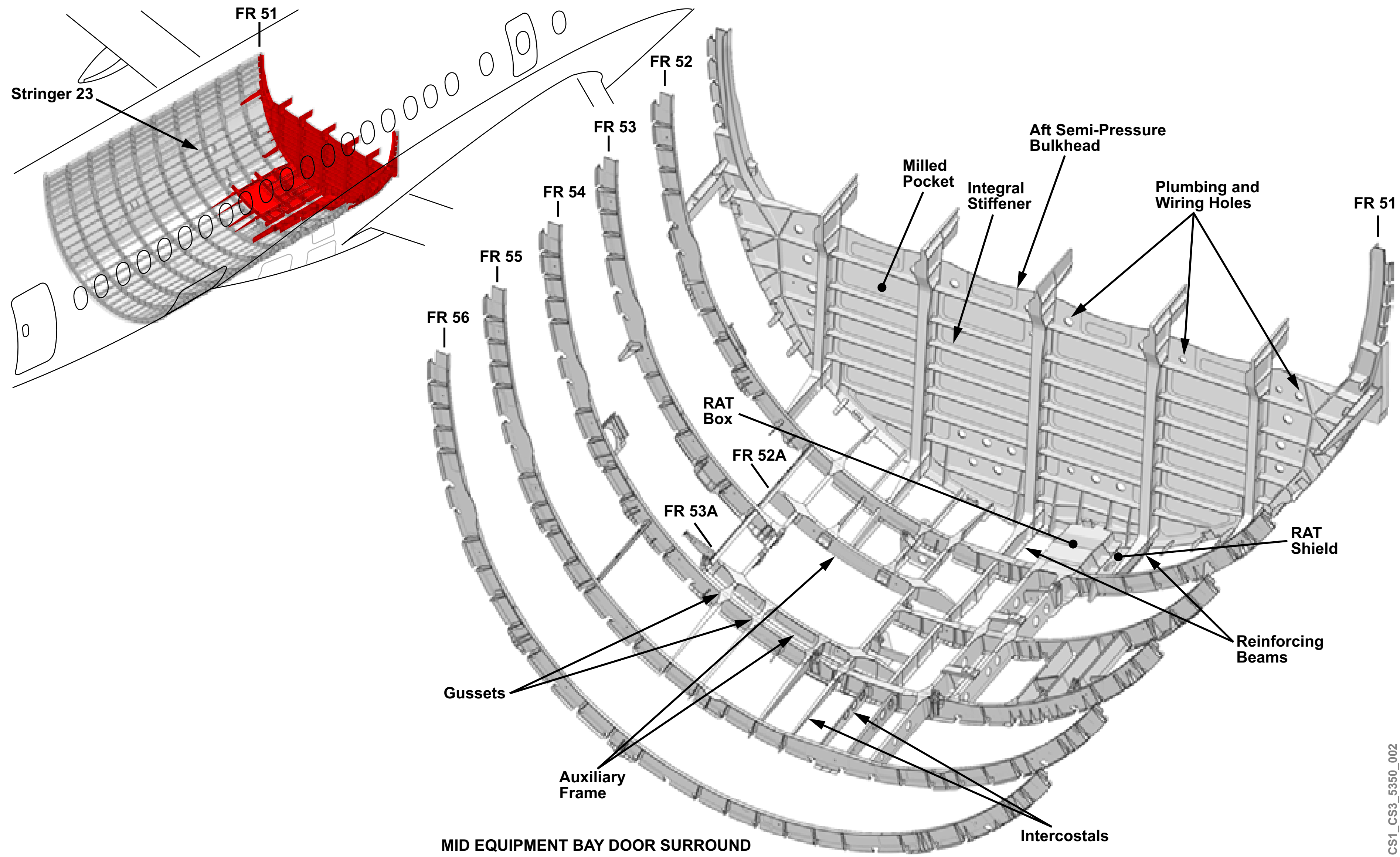


Figure 26: Aft Semi-Pressure Bulkhead and Mid Equipment Bay Door Surround

53-60 REAR FUSELAGE

GENERAL DESCRIPTION

The rear fuselage starts at the circumferential splice at FR 61, where it joins the mid fuselage section. The rear fuselage ends at the circumferential splice at FR 79.

Between FR 61 and FR 68, the rear fuselage has a constant cross section. From FR 68 the rear fuselage cross section tapers to the rear pressure bulkhead.

The rear fuselage section includes:

- Aft passenger and service door surround structures
- Aft cargo door surround structure
- Cabin window support frames

The cabin floor support structure ends at FR 79.

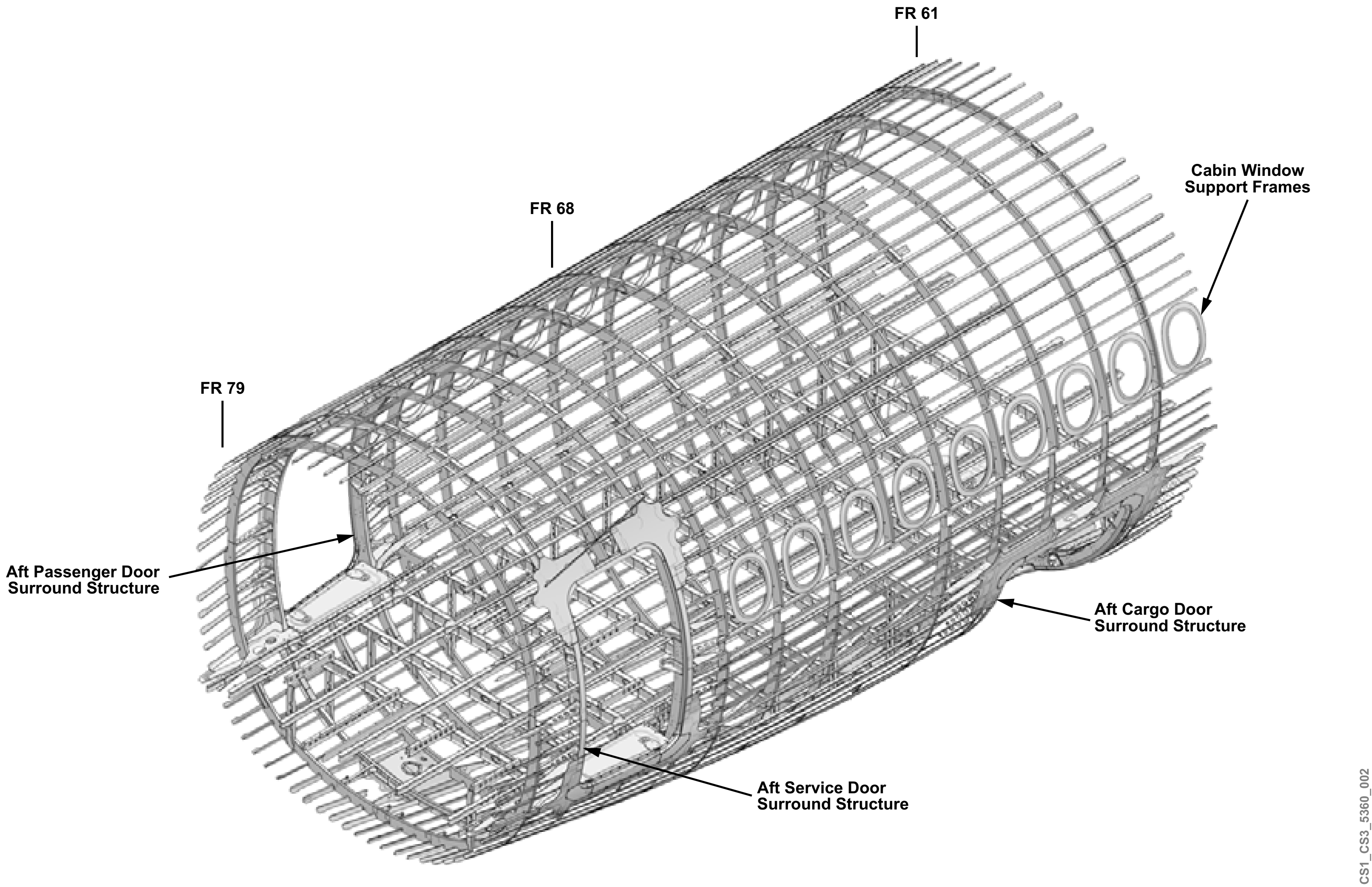


Figure 27: Rear Fuselage

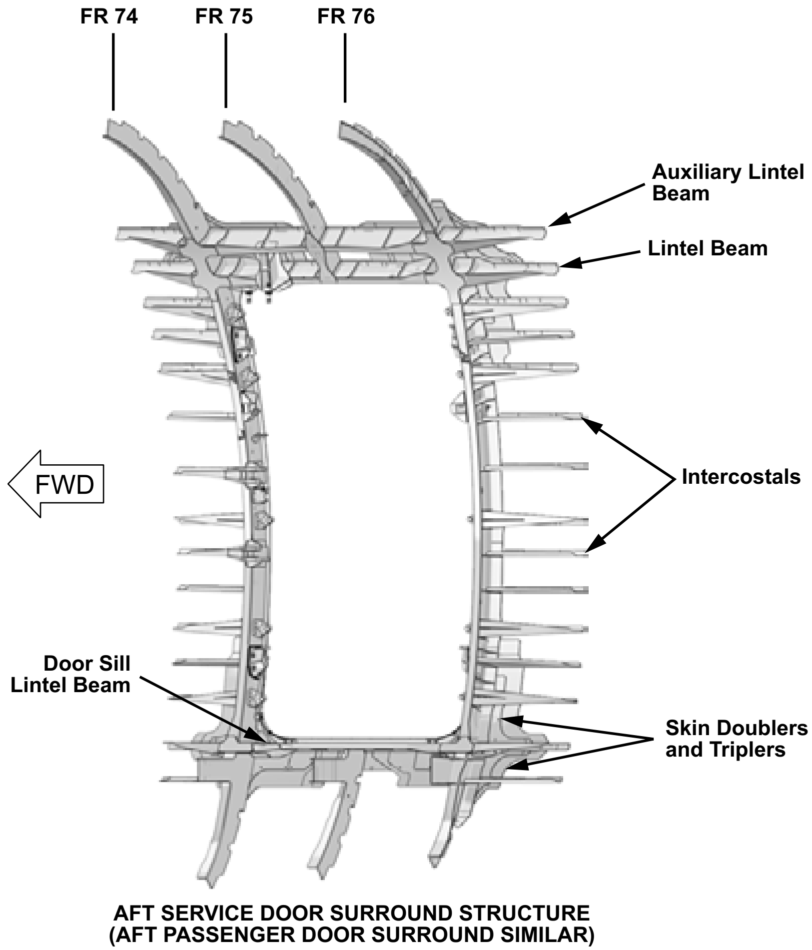
DETAILED COMPONENT INFORMATION

PASSENGER AND SERVICE DOOR SURROUNDS

The structure around the passenger and service doors is reinforced by skin doublers and triplers, beams and intercostals. Two lintel beams reinforce the upper section of the door surround, and one lintel beam comprises the lower sill of the door surround. Doublers and triplers are used to reinforce the fuselage skin around the cutout.

CARGO DOOR SURROUND

A double frame is used for the reinforcement of the fuselage cargo door surround. The two additional local frames are numbered FR 63 and FR 66. Two lintel beams are used at the upper and lower sections of the double frame structure. The fuselage skin is locally reinforced around the cargo door cutout with doublers, and triplers.



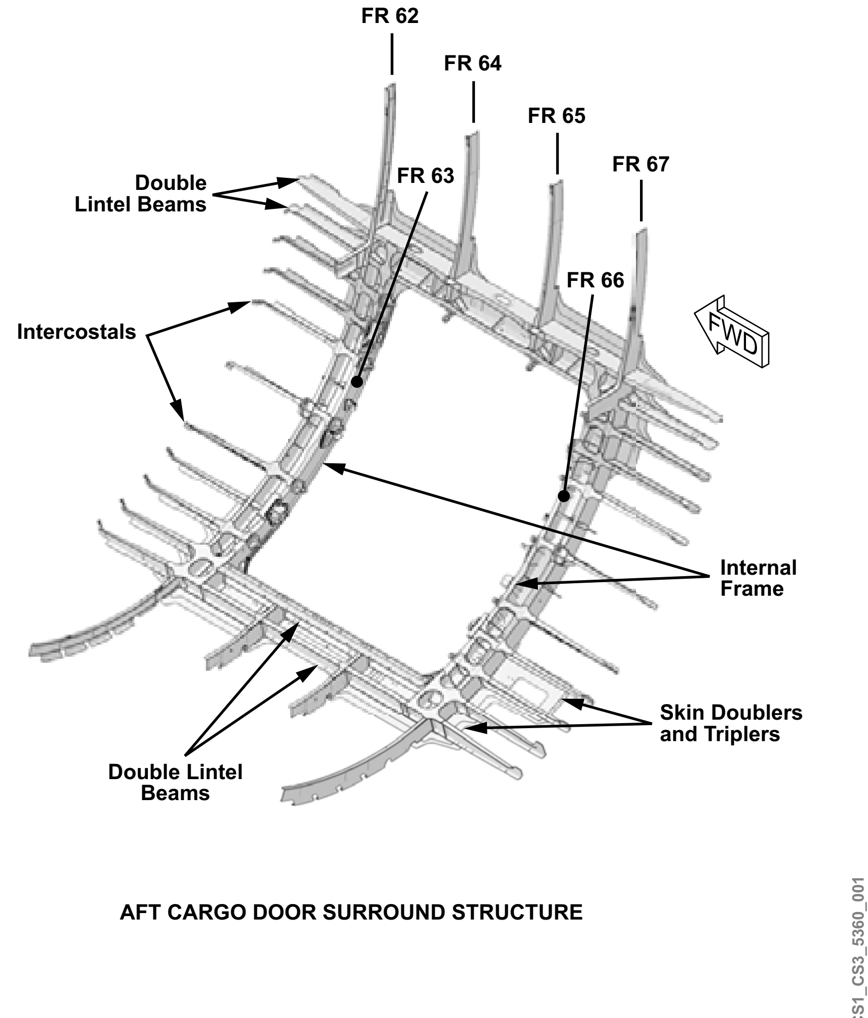


Figure 28: Rear Fuselage Passenger, Service, and Cargo Door Surrounds

53-70 AFT FUSELAGE

GENERAL DESCRIPTION

The aft fuselage extends from FR 79 to FR 85. The aft fuselage is constructed primarily of composite materials. It houses the aft equipment bay component support racks, the aft equipment bay door and surround structure, and the horizontal stabilizer trim actuator (HSTA) support structure.

At the forward end it joins with the rear barrel of the rear fuselage, where the rear pressure bulkhead is located, and at the aft end it joins with the tailcone. Structures on the top and sides of the aft fuselage connect the vertical and horizontal stabilizers to the aircraft.

Machined aluminum longerons are located on each side of the aft fuselage structure. The longerons transfer forward and aft loads from the horizontal stabilizer, and provide support at the stabilizer cutouts.

The C-section frames are made from a combination of aluminum sheet metal and machined plate, and are mechanically fastened to the skins. Frame 85 has upper and lower frames connected by the pivot hinge fittings that attach the horizontal stabilizer to the aircraft. The lower frame is wrapped with fire blankets due to its proximity to the APU compartment. Four horizontal stabilizer load rods connect the pivot fittings with FR85, which also interfaces with the tailcone. and transfers loads from both the horizontal stabilizer and tailcone onto the aft fuselage.

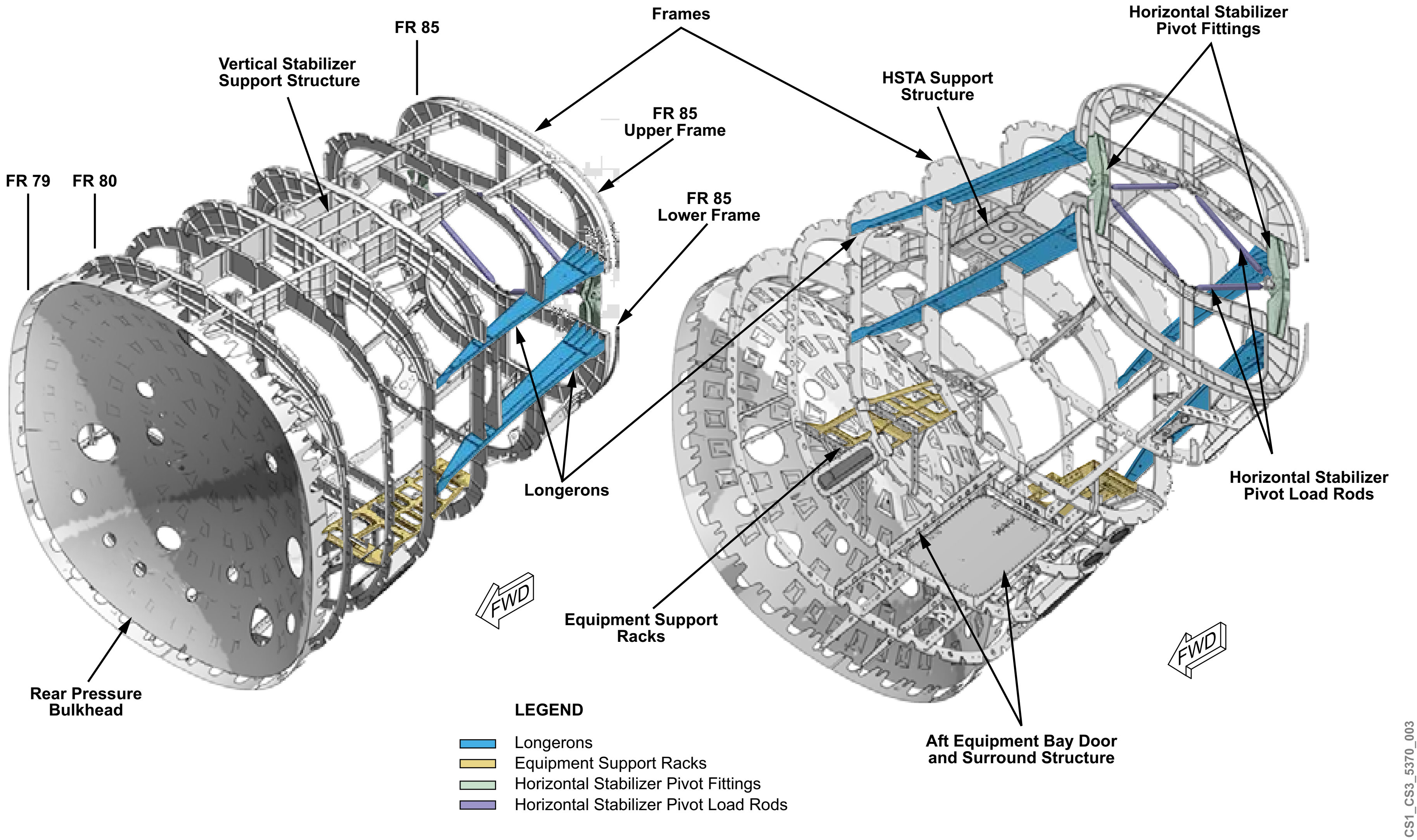


Figure 29: Aft Fuselage

DETAILED COMPONENT INFORMATION

SKINS, STRINGERS, AND CUTOUTS

The aft fuselage integrally-stiffened skin is composed of solid carbon fiber reinforced polymer (CFRP) laminate. The skin panels are manufactured using automated fiber placement (AFP), and co-cured with delta and omega shaped CFRP stringers. There are three panels: one upper, one lower left, and one lower right, which are connected by butt joints.

Cutouts are provided for the following major items:

- Vertical stabilizer attach fittings
- Horizontal stabilizers
- Aft equipment bay door

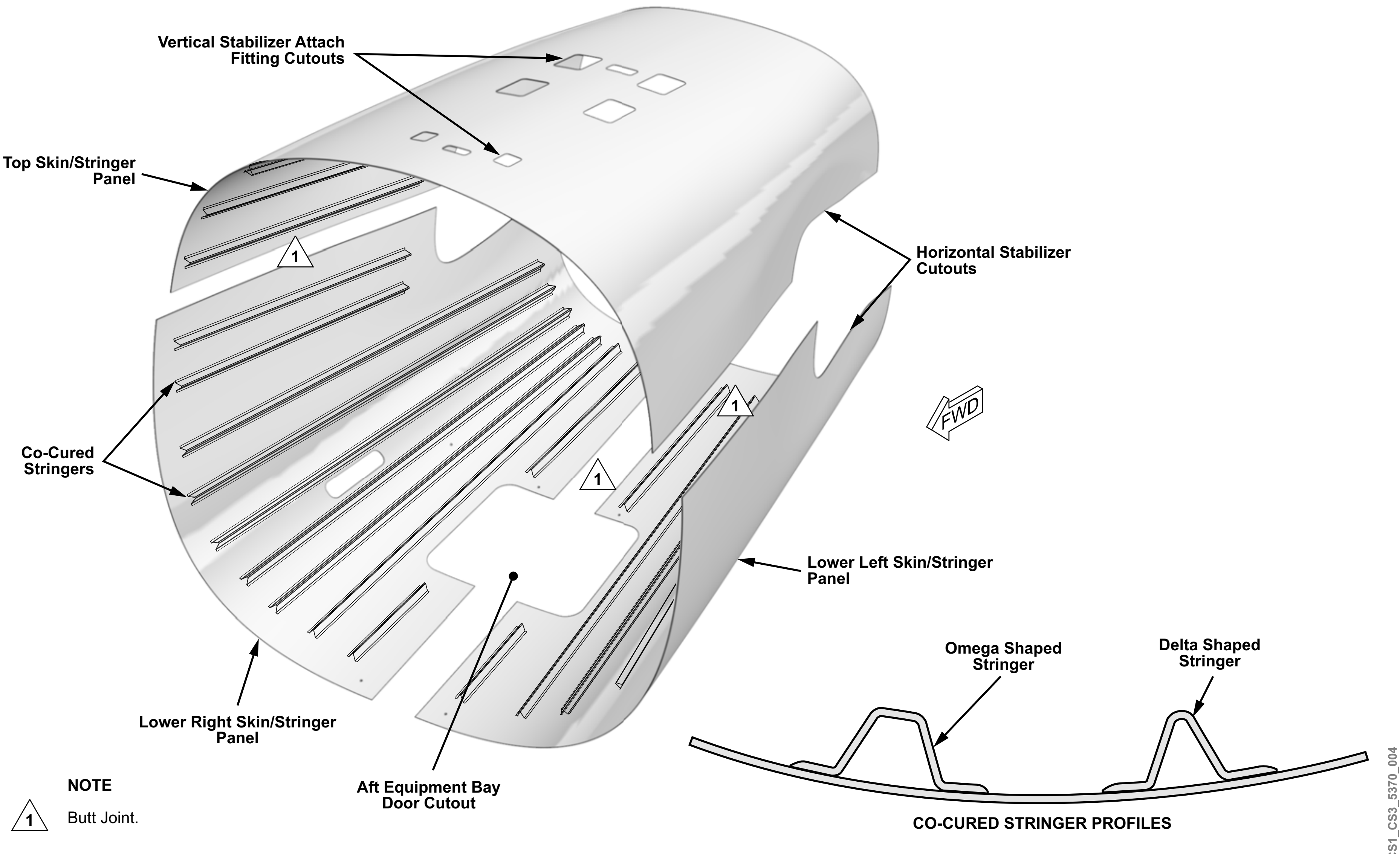


Figure 30: Aft Fuselage Skins, Stringers, and Cutouts

VERTICAL STABILIZER SUPPORT STRUCTURE

The vertical stabilizer support structure extends from FR 80 to FR 84, and consists of 7000-series aluminum intercostals. These transfer the loads from the vertical stabilizer into the aluminum aft fuselage frames and composite skins through mechanical fasteners.

VERTICAL STABILIZER SHEAR FITTINGS

The vertical stabilizer attaches to the upper aft fuselage with six 7000-series aluminum shear fittings.

HORIZONTAL STABILIZER TRIM ACTUATOR SUPPORT STRUCTURE

The horizontal stabilizer trim actuator (HSTA) support structure consists of a closed box made up of two intercostals between FR 82 and FR 83, and a shear plate to form the box. The box transfers the loads from the HSTA into the aft fuselage, specifically the frames, through the HSTA attachment fittings. It is in close proximity to the vertical stabilizer support.

HORIZONTAL STABILIZER TRIM ACTUATOR ATTACH FITTINGS

The HSTA attach fittings connect the HSTA to the aft fuselage frames. A single fitting plate contains three fittings in one assembly, consisting of two primary load path fittings, and one secondary load path fitting. It is a failsafe design made from 7000-series aluminum.

HORIZONTAL STABILIZER PIVOT FITTING ASSEMBLY

The horizontal stabilizer pivot fitting assembly consists of two large fittings, attached at FR 85 to the upper and lower half frames. The fittings and four diagonal rods are designed to support the horizontal stabilizer, and transfer the loads into the aft fuselage.

The diagonal rods are attached to the upper and lower half frames at their innermost point. The rods are attached to the failsafe horizontal stabilizer pivot fittings at their outermost points.

AFT EQUIPMENT BAY SUPPORT RACKS

The aft equipment bay is located between FR 80 and FR 83. The support structure consists of an assembly of 2000 and 7000-series aluminum posts and horizontal supports. It provides a light, but strong support, for the equipment located in the bay. The right equipment bay rack supports hydraulic system no. 3 components. The left rack supports primarily avionics equipment.

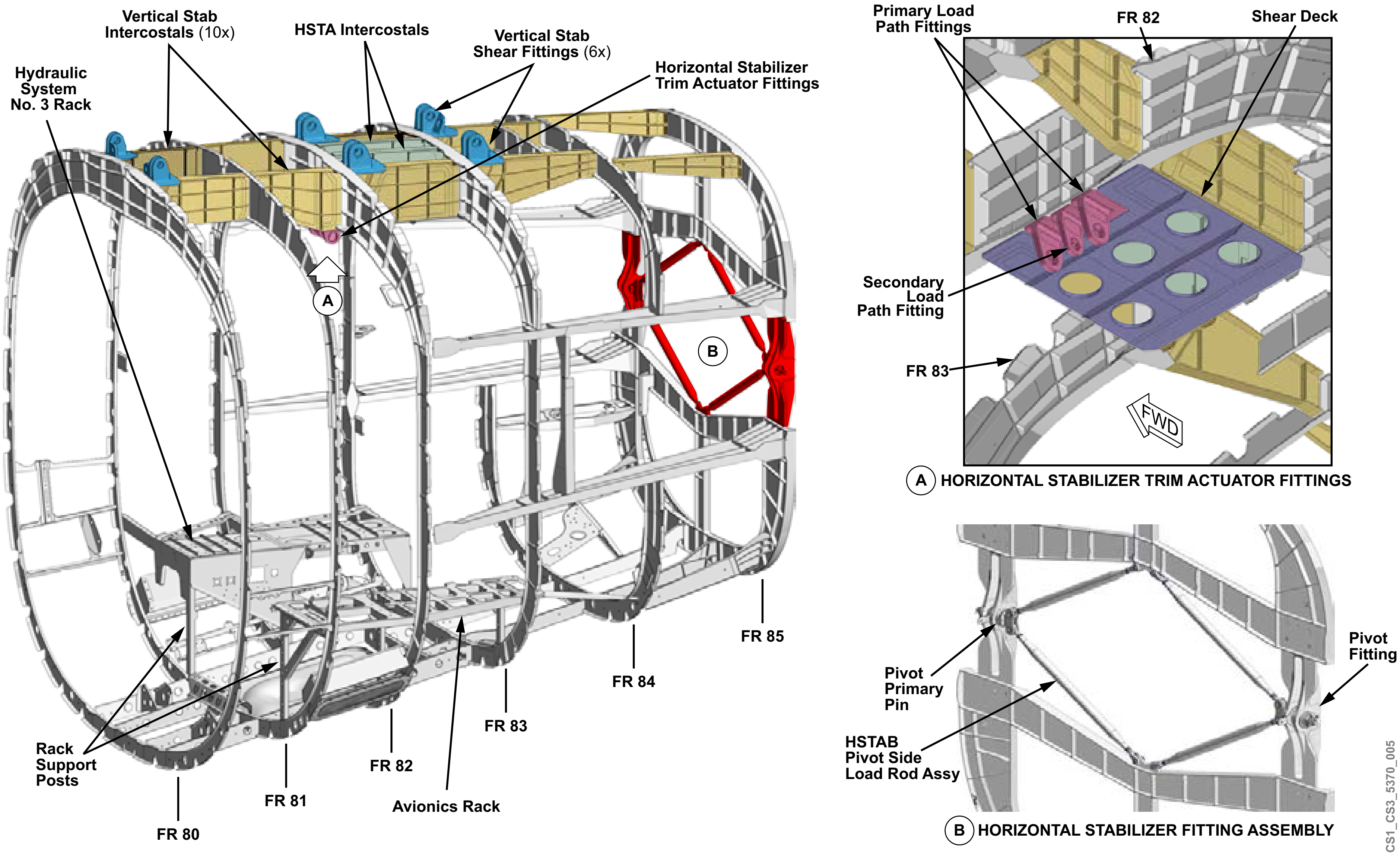


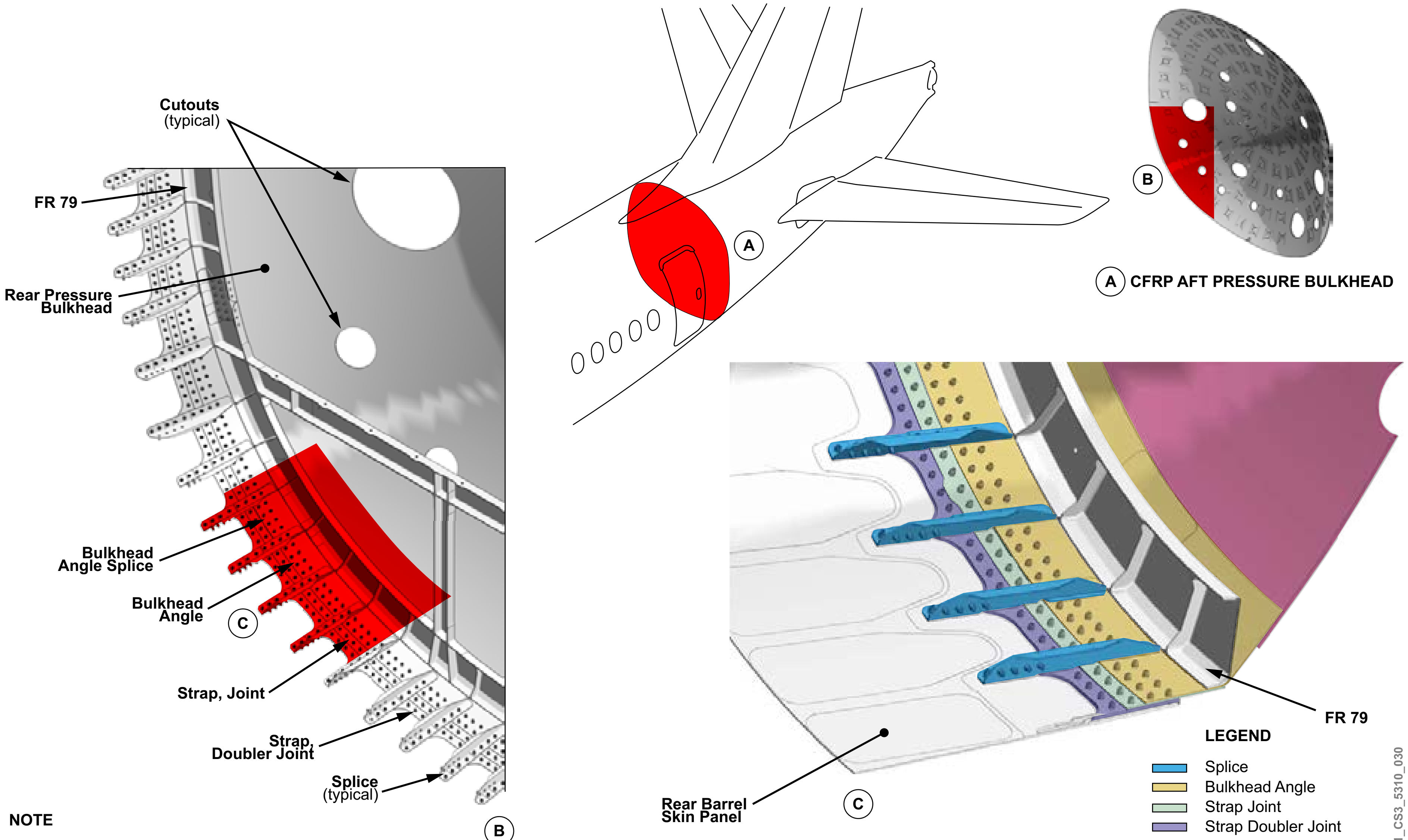
Figure 31: Aft Fuselage Support Structures and Fittings

REAR PRESSURE BULKHEAD

The dome-shaped rear pressure bulkhead is a CFRP solid laminate panel installed at FR 79.

The rear pressure bulkhead is attached to the fuselage with a four piece titanium strap joint, and circumferential splices. Due to the extreme atmospheric pressure differentials applied to the bulkhead in flight, the four piece strap joint is backed-up by bulkhead angles and splices, and doublers.

The pressure bulkhead has cutouts for hydraulic, pneumatic, fuel, and electrical system plumbing, ducting, and wiring. The cutouts around the pass through items are sealed to protect the pressurized cabin from the unpressurized aft fuselage area.



NOTE

Rear pressure bulkhead is located at frame 79.

Figure 32: Rear Pressure Bulkhead

53-80 TAILCONE

GENERAL DESCRIPTION

The tailcone is the most aft section of the fuselage. It is constructed almost entirely of carbon fiber reinforced polymer (CFRP) frames, stringers and skins, and extends from FR 86 to the canted FR 91. Its primary function is to accommodate the auxiliary power unit (APU). A taillight fairing is attached to the tailcone at FR 91.

The tailcone is attached to the aft fuselage at FR 86 with four primary attach fittings, and lower center fittings.

Three main compartments can be identified in the tailcone:

- **APU enclosure -** A heat resistant structure providing protection from the APU to the rest of the tailcone structure. It is located between FR 86 and FR 89. The enclosure provides access panels for internal inspection of the tailcone section
- **Forward upper tailcone compartment** - Houses the APU inlet duct and door. This section is located directly above the APU enclosure, and can be accessed through APU enclosure access panels
- **APU muffler compartment -** Contains the APU exhaust duct. This section is located between FR 89, and ends at FR 91

NOTE

The current design of tailcone skins and stringers has been modified from composite to aluminum parts. There are no major changes to form, fit, or function. Refer to the Aircraft Maintenance Publication for effectivity details.

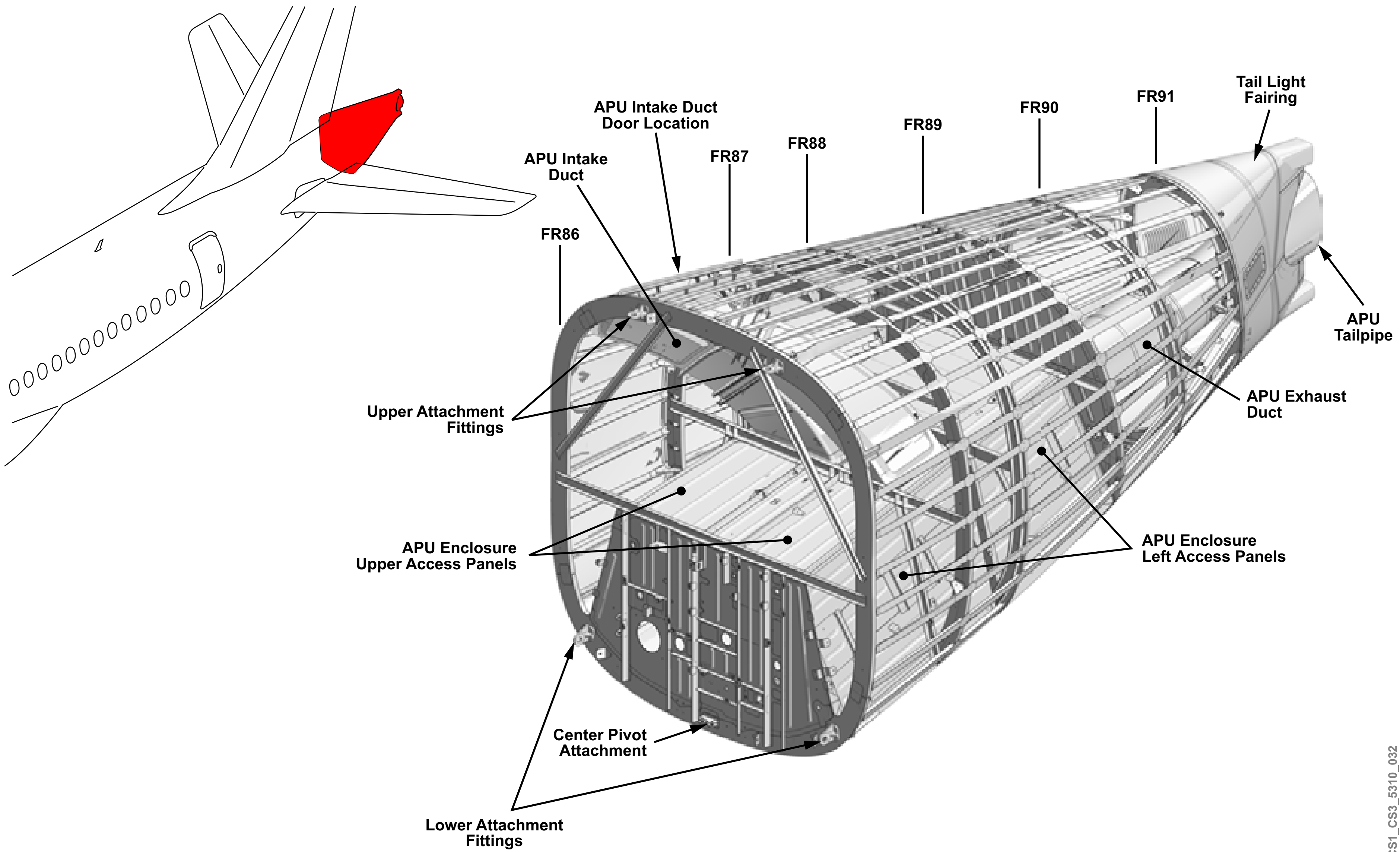


Figure 33: Tailcone

DETAILED COMPONENT INFORMATION

SKINS AND CUTOUTS

The tailcone has the following four composite CFRP solid laminated skin panels between FR 86 and FR 91:

- Upper skin panel
- Left side skin panel
- Right side skin panel
- Lower skin panel

The skin panels feature the following:

- A multiple CFRP buildup
- A copper mesh ply for electrical continuity
- Glass fiber top ply in locations where composite is in contact with a metal structure

The number of plies of the skin panels varies with the location. The tailcone upper skin panel has a cutout for the APU intake duct.

STRINGERS

The stringers are composite CFRP solid laminated one piece T-section stringers, co-bonded to the skin panels. The exceptions are two stringers, which are made from aluminum. These metallic strap stringers are used to ensure the electrical continuity through the tailcone by forming the bonding from the skin panels.

The tailcone has 34 stringers with a pitch that reduces from FR 86 to FR 91, due to the taper of the tailcone.

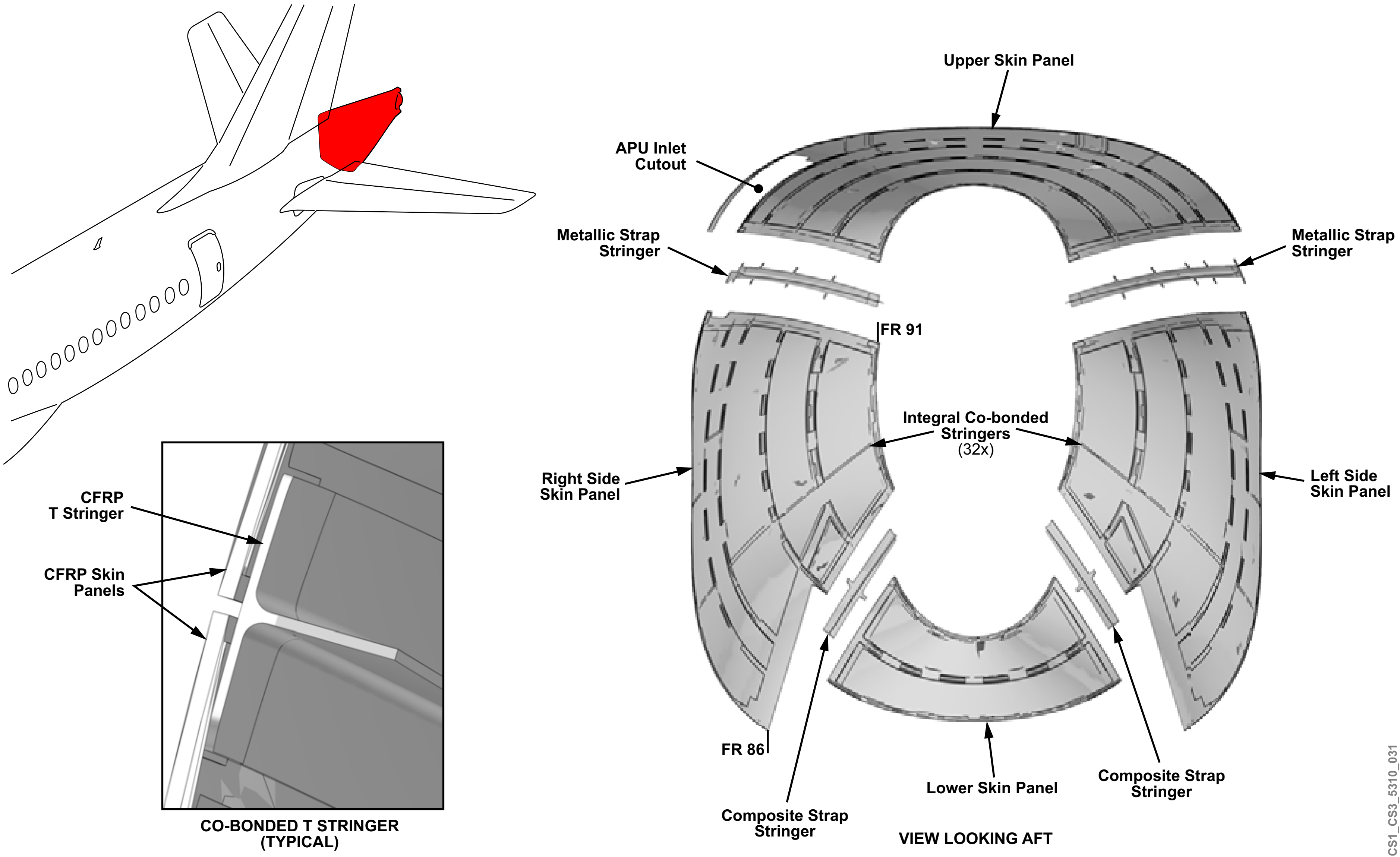


Figure 34: Tailcone Skins, and Stringers

FRAMES

The tailcone features six frames. The frames are numbered FR 86 through FR 91. Frames 89 through 91 are canted frames, therefore they have no identifying fuselage station number.

All frames are made of aluminum, except for the lower section of FR 86. The lower section of FR 86 is located in the APU compartment. It is not shielded from high temperatures, and therefore it is made of titanium. FR 86 and FR 89 are machined frames. The other frames are made from extruded sheet metal.

AUXILIARY POWER UNIT ENCLOSURE ASSEMBLY

The enclosure of the APU serves as the structural mounting framework for the APU, as well as the protection for the tailcone against an APU fire. The skin plating and access panels are made of titanium. A fire resistant seal is installed on the access panels to ensure adequate sealing against fire penetration. The titanium enclosure panels are reinforced by vertical and horizontal beams at the locations of the frames. In addition to the access panels, the APU enclosure features a number of penetrations for system ducting, plumbing, and wiring feed through.

TAILCONE LIGHT FAIRING ASSEMBLY

The last section of the tailcone is an aerodynamic fairing. It covers the exhaust/tail pipe section of the APU muffler. The light fairing is easily removable in two sections, and installed by bolts to FR 91. Made from single curvature aluminum sheet, the light fairing is easily repairable. It provides the mounting structure for the navigation and strobe lights. No direct connection is made between the light fairing and the APU muffler tail pipe. A seal mounted on the light fairing provides the interface.

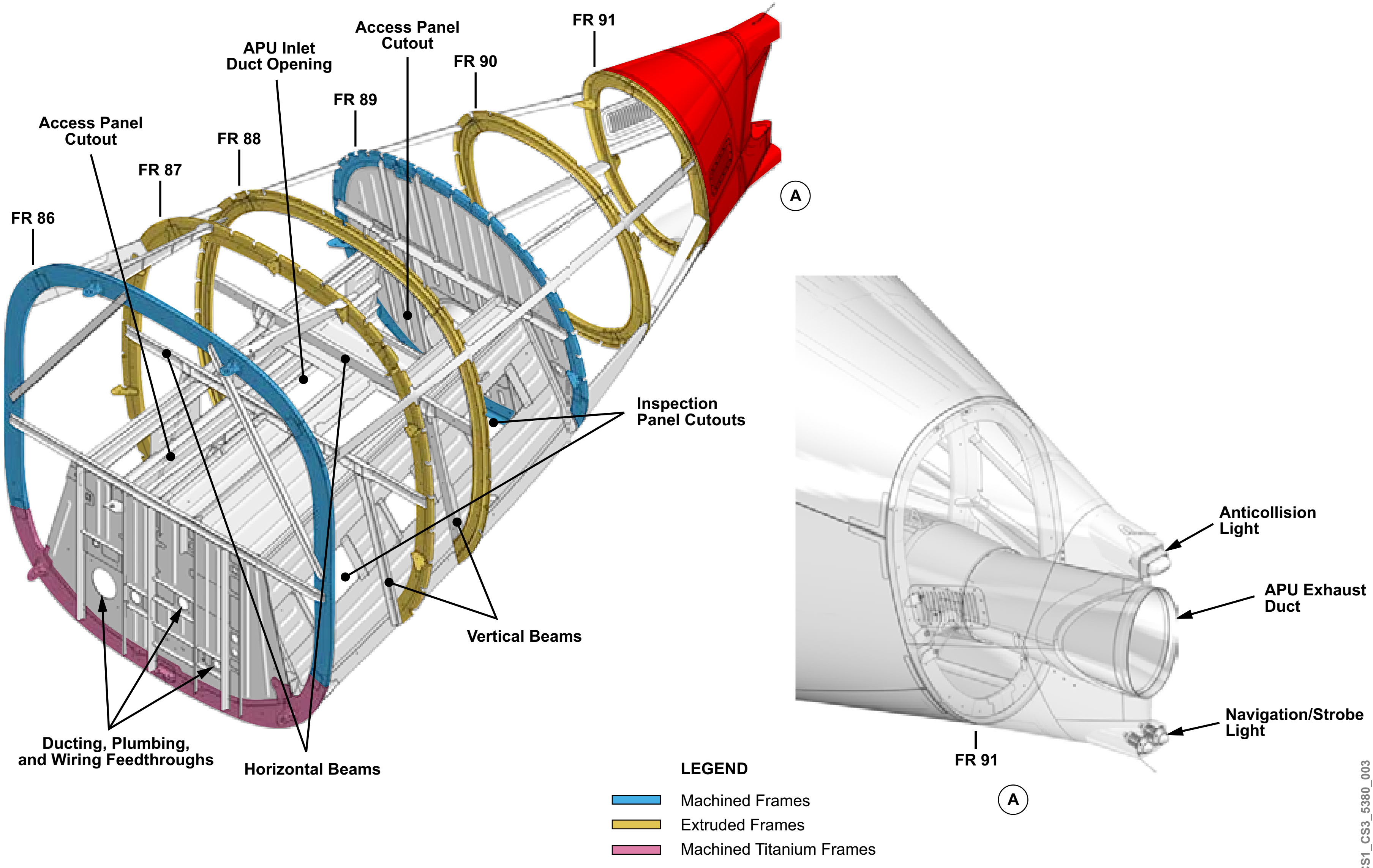


Figure 35: Auxiliary Power Unit Enclosure

53-80 WING-TO-BODY FAIRING

GENERAL DESCRIPTION

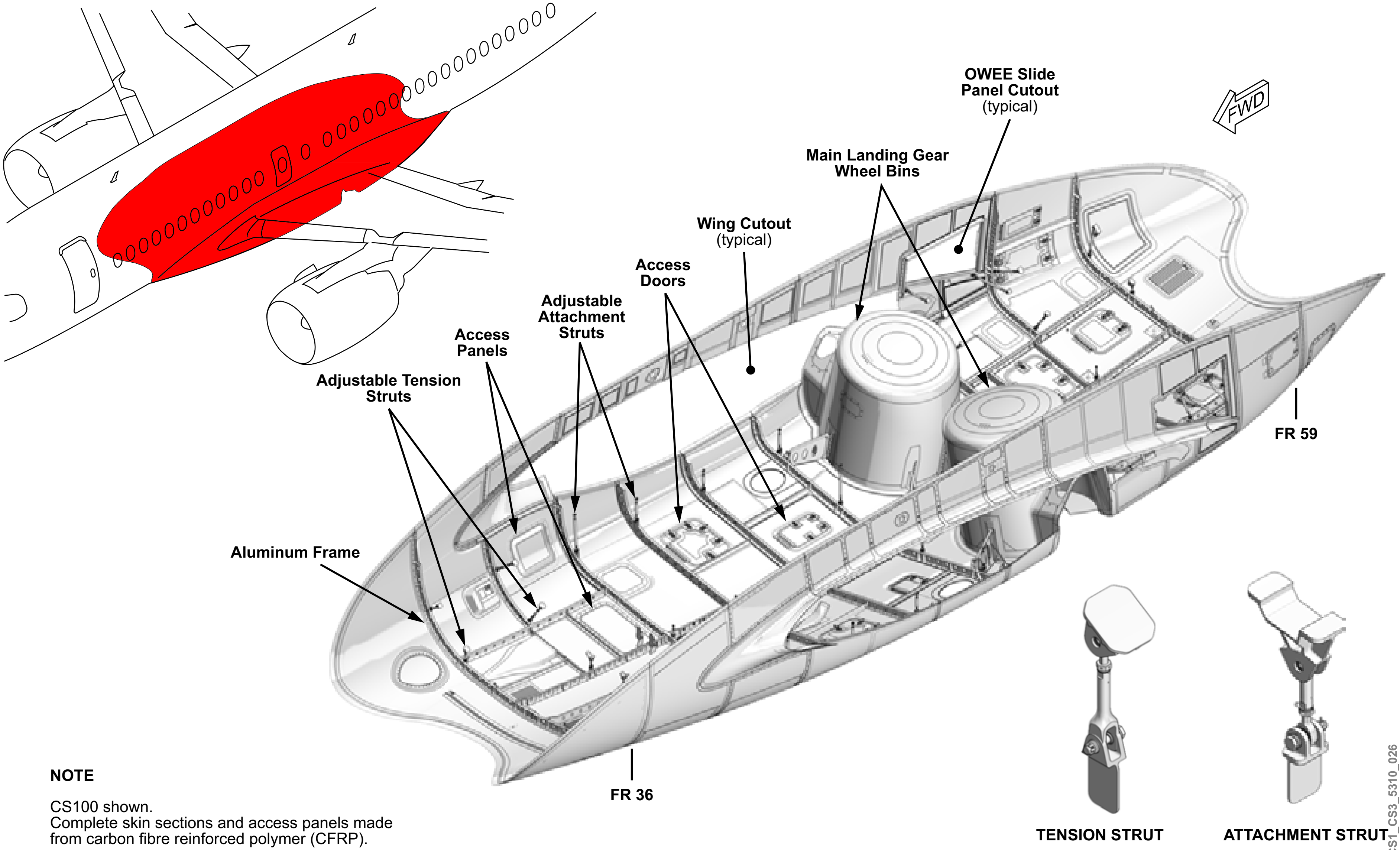
The wing-to-body fairing (WTBF) provides aerodynamic smoothness at the wing-to-fuselage interface. It is made of a combination of carbon fiber reinforced polymer (CFRP), skinned honeycomb, and solid CFRP laminate, attached to an aluminum alloy frame. The aluminum frame extends from FR 36 to FR 59. The WTBF extends beyond the aluminum frame. The WTBF covers, and provides access to many of the aircraft systems, including the environmental control system (ECS), hydraulic, fuel, avionic, electrical, fire protection, water, and high lift system components.

The WTBF is divided into large, separately removable panels, which feature smaller access panels. The small access doors and panels provide quick access to components for inspection and servicing. The larger panels enable the removal/replacement of larger system components without having to remove the entire fairing.

The complete fairing assembly is attached to the fuselage structure with screws, adjustable struts, adjustable tension struts, and support fittings.

The WTBF also houses the main landing gear (MLG) wheel bin enclosures.

The WTBF has cutouts for the aircraft wings, and the overwing emergency exit slide assemblies.



NOTE

CS100 shown.
Complete skin sections and access panels made from carbon fibre reinforced polymer (CFRP).

Figure 36: Wing-to-Body Fairing

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ATA 54 - Nacelles and Pylons

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

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NACELLES AND PYLONS - CHAPTER BREAKDOWN

54-10 NACELLES

GENERAL DESCRIPTION

The nacelle on each wing includes the engine inlet cowling, left and right fan cowlings, and the left and right reverser cowlings.

The inlet cowling is mounted to the front section of the engine. It directs intake air into the engine fan section. The lip of the inlet cowling is heated with hot engine bleed air to prevent icing.

The fan cowlings enclose the engine fan section. They are hinged to the pylon and held closed with latches at their lower ends.

Two thrust reverser (T/R) cowlings enclose the engine core section. They are hinged to the pylon, and held closed with latches at their lower ends. The outer surface of the thrust reverser have translating sleeves, which are part of the thrust reverser system.

CAUTION

The T/R cowling translating sleeves are moveable. They slide aft to deploy, and forward to stow.

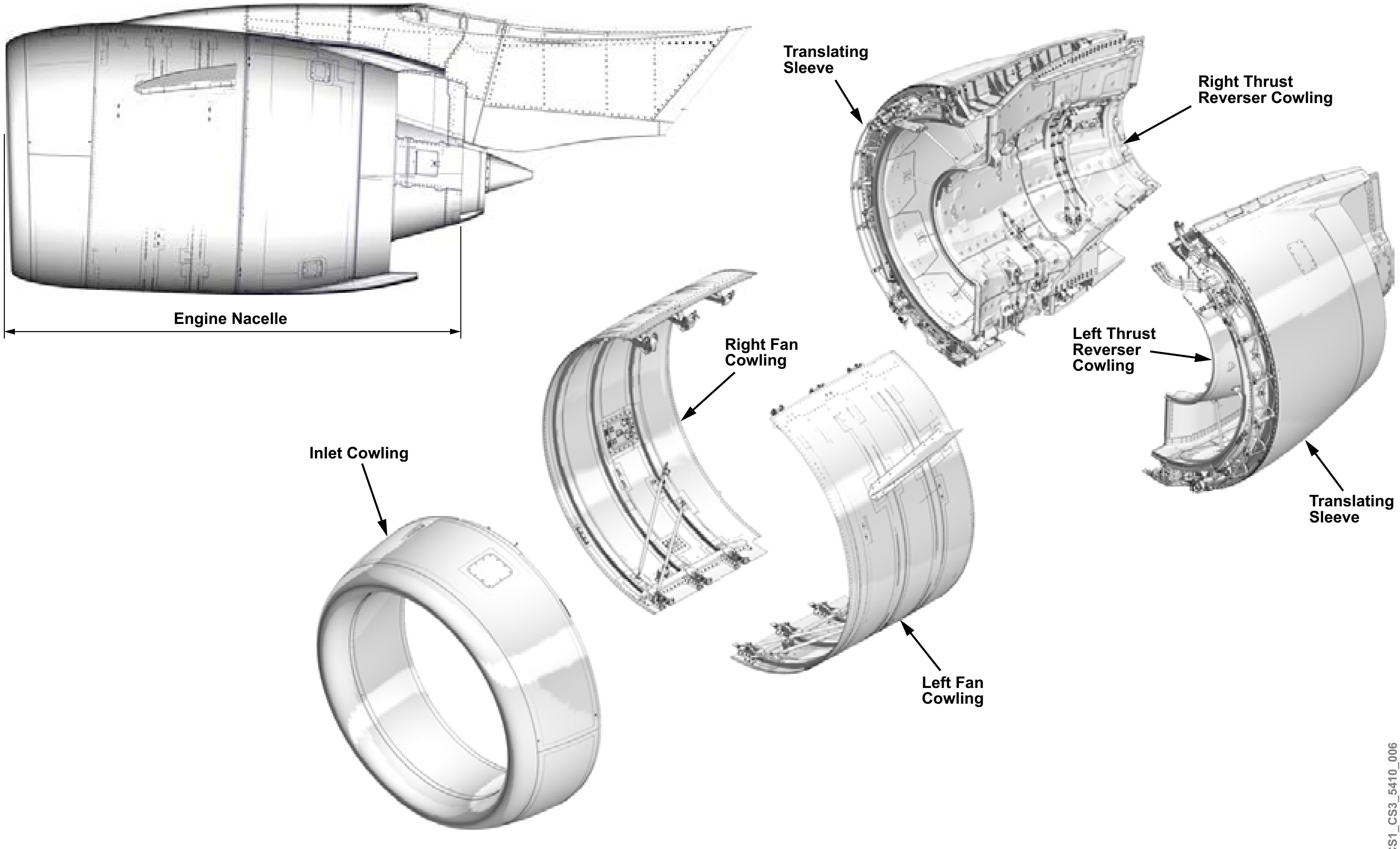


Figure 1: Nacelle General Description

DETAILED COMPONENT INFORMATION

INLET COWLING

The inlet cowling skins are manufactured from composite material. The lip section of the inlet cowling is made from aluminum alloy. Internal structural members such as the forward and aft bulkheads, stiffeners, and rings are made from aluminum alloy. The inlet cowling is attached to the engine fan case section with 40 bolts and self-locking nuts. The inlet cowling can be installed on either engine.

FAN COWLINGS

The fan cowlings are manufactured from carbon fiber reinforced polymer (CFRP). Each fan cowling is hinged to the pylon with three hinge fittings. The engine fan cowlings are secured together at their lower ends by three latches.

THRUST REVERSER COWLINGS

The thrust reverser (T/R) cowling consists of the core cowl and the translating sleeve. The core cowl is a stationary part and encloses the engine core section. The translating sleeve moves aft when deployed, and forward to stow.

The skin sections of the thrust reverser cowling are manufactured from CFRP. Various structural parts are manufactured from aluminum alloy.

The core cowl inner skin surface, that is exposed to the engine core compartment, is covered with a heat resistant insulation blanket.

The thrust reverser cowling is hinged to the pylon by three hinges. The thrust reverser cowlings are secured together at their lower ends by five latches.

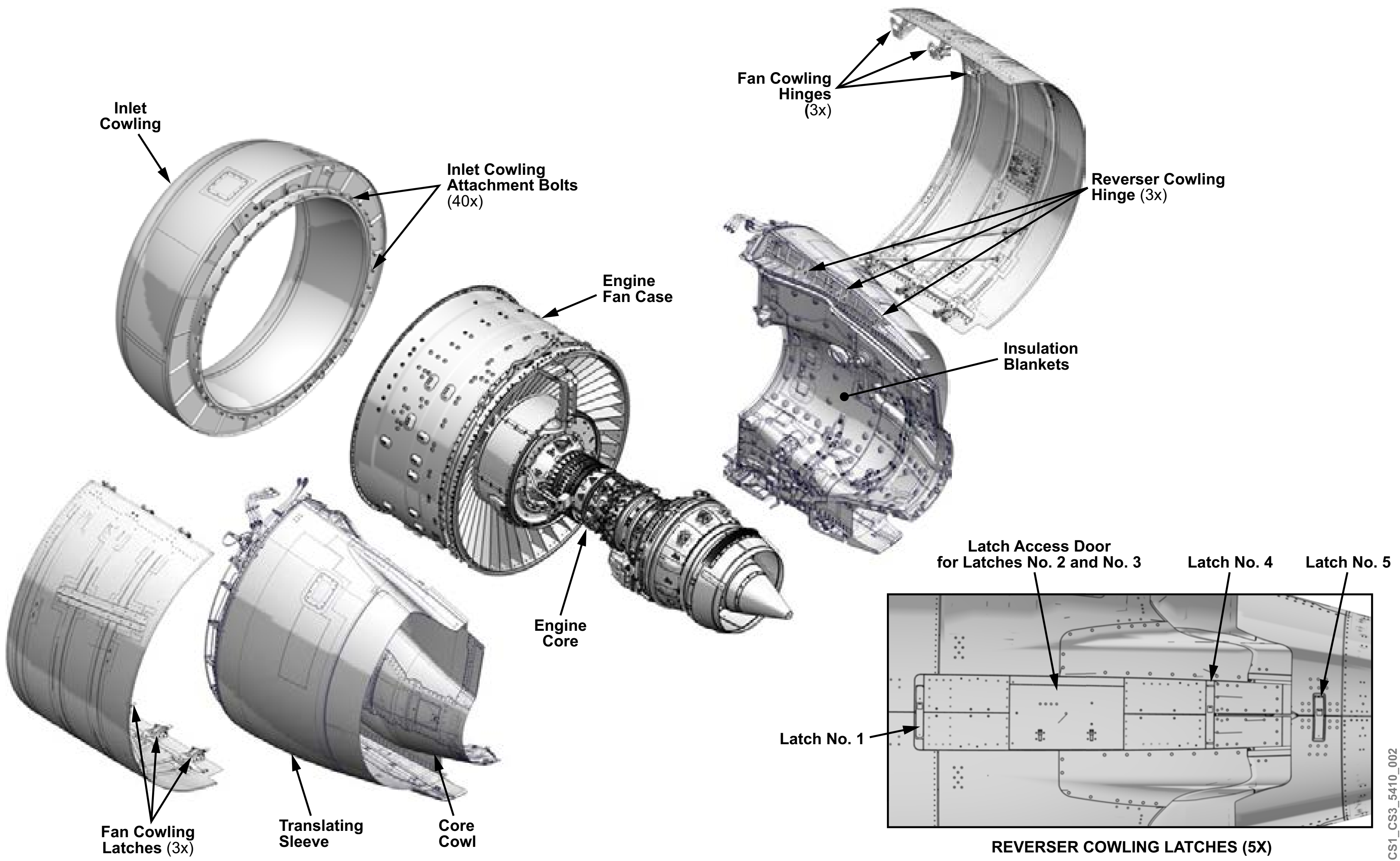


Figure 2: Inlet, Fan, and Thrust Reverser Cowlings

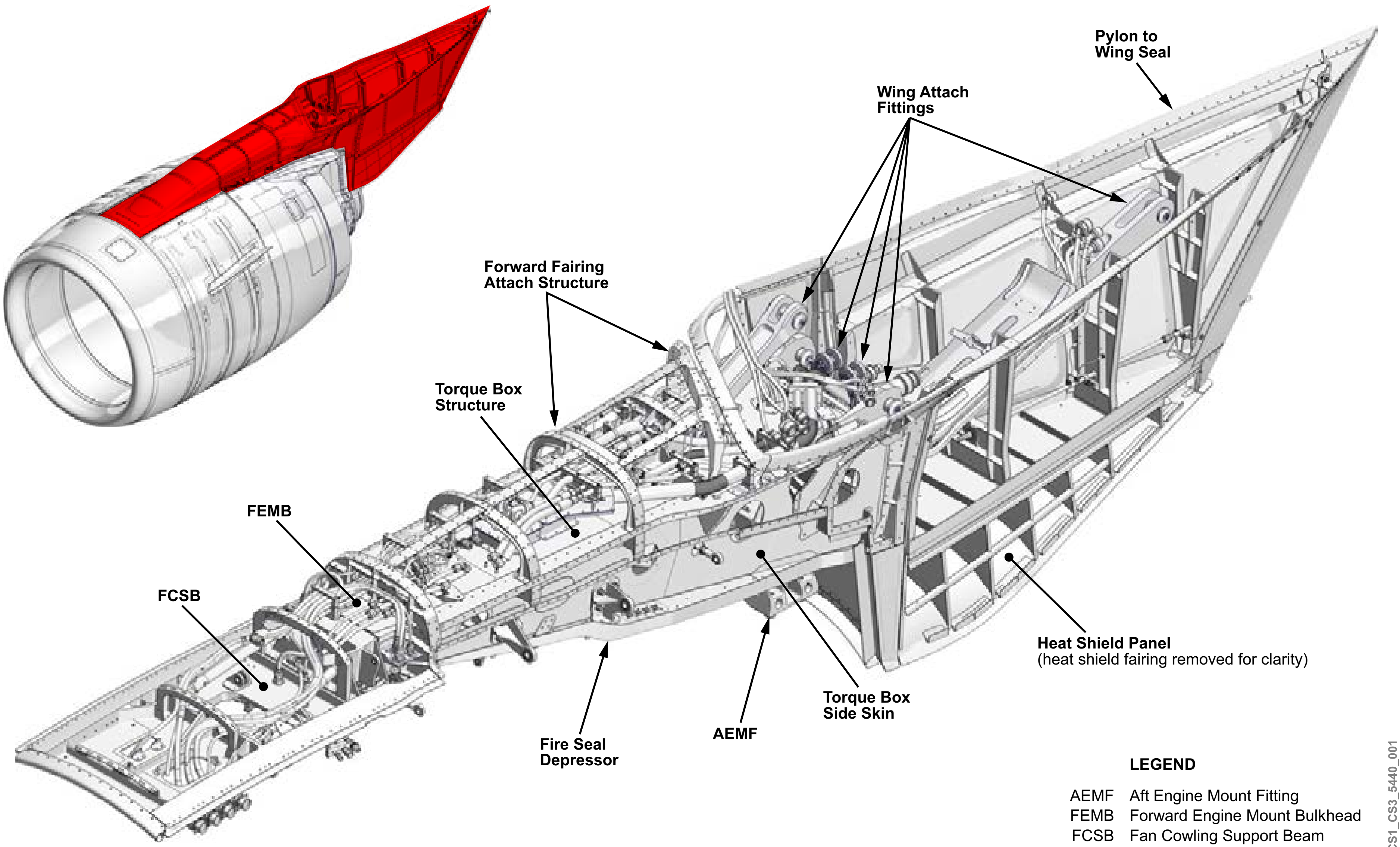
54-50 PYLONS

GENERAL DESCRIPTION

The pylons are the support structure for the engines and nacelles. The primary structural element of the pylon is the torque box structure. The torque box transmits lateral and longitudinal loads to the wing attach points, which transfer the loads to the wing.

The fan cowling support beams (FCSBs) are located at the forward end of the torque box assembly. The forward engine mount bulkhead, and aft engine mount fittings are attached to the torque box structure. A support structure is attached to the pylon structure to allow installation of exterior panels. Due to contact with the engine nacelle, the lower section of the pylon contains fire seals.

The aft portion of the pylon is aerodynamically faired using an interior frame and external fairing panels. A heat shield is located at the base of the aft pylon section to protect against engine heat. A pylon-to-wing blade seal is located around the upper perimeter of the pylon aft portion for aerodynamic and environmental sealing.



LEGEND

AEMF	Aft Engine Mount Fitting
FEMB	Forward Engine Mount Bulkhead
FCSB	Fan Cowling Support Beam

Figure 3: Pylon

DETAILED COMPONENT INFORMATION

PYLON TORQUE BOX

The pylon torque box is a closed-end structure consisting of four spar skins, four bulkheads, and three frames for stability.

The titanium alloy upper spar consists of a spar web (skin), and left and right aluminum spar chords. The upper spar web has cutouts at various locations to accommodate systems that run through the box.

The torque box left and right titanium side skins absorb torsion, vertical shear, and lateral bending loads applied to the torque box at various locations, including the engine mount attachments. The side skins incorporate holes for access, and structural inspections.

The lower spar consists of a titanium web and chords. The lower spar serves as a firewall for the pylon along with a fire seal depressor. The lower spar web has cutouts at various locations to accommodate systems that run through the box.

The mid bulkhead fitting is located between frame 3 and the aft engine mount bulkhead. The aft engine mount bulkhead is made from forged titanium alloy, and redistributes aft engine mount fitting loads into the pylon torque box structure.

The aluminum pylon aft strut bulkhead closes the aft end of the pylon, and redistributes loads from the R2 fitting, the aft upper spar fittings (R3/R4), and the dual-side links (R7/R8) into the pylon structure.

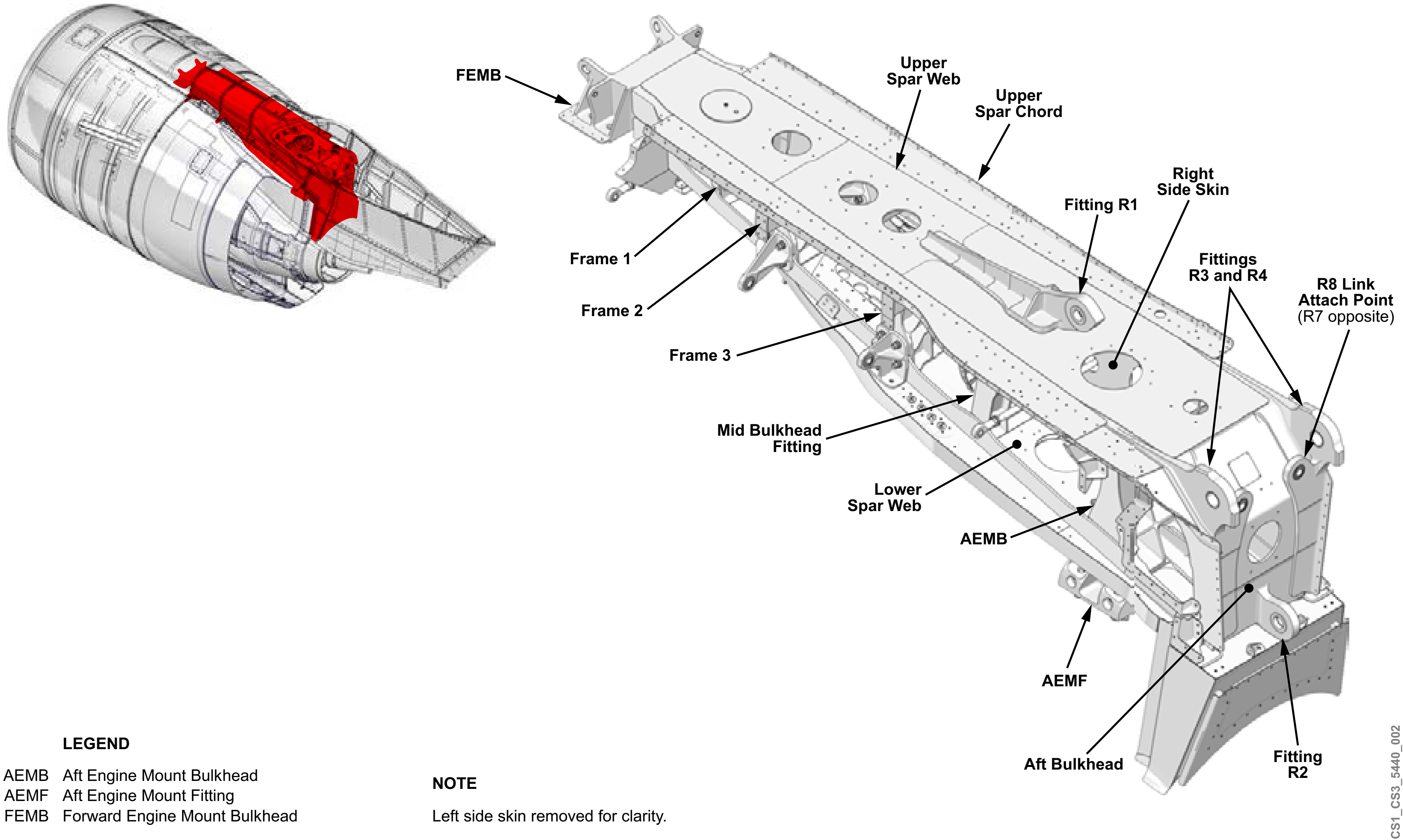
The aluminum forward engine mount bulkhead (FEMB) closes the forward end of the pylon, and redistributes the forward engine mount vertical and lateral loads into the pylon structure.

The three internal frames are attached between the four spar skins to form the box structure. Frames 2 and 3 react to the loads from the thrust reverser hinges and structural ducts and transfer them to torque box panels.

The fittings R1, R2 R3/4, and R7/8 are part of the wing attach arrangement. Each of these items transfer loads that affect the torque box.

The R1 fitting is attached to the upper spar. The aft end of the fitting has a lug that transmits loads from the pylon through the upper link into the wing front spar.

The titanium R2 fitting is an integral part of the torque box aft bulkhead. It makes up the aft lower closeout of the torque box, and runs forward to the aft engine mount bulkhead. The R2 fitting transfers loads from the torque box into an aluminum diagonal brace link, which then transmits those loads into the wing lower surface.



LEGEND

AEMB	Aft Engine Mount Bulkhead
AEMF	Aft Engine Mount Fitting
FEMB	Forward Engine Mount Bulkhead

NOTE

Left side skin removed for clarity.

Figure 4: Pylon Torque Box

PYLON-TO-WING ATTACHMENTS

Each pylon is attached to the wing at six points. The structural arrangement utilizes multiple load paths to transmit axial, lateral, and vertical shear and moments from the nacelle/pylon to the wing. The fittings are designed to carry limit load after failure of any single structural element (failsafe). Each major load path has a structural fuse pin to prevent damage to the wing structure in the event of a severe overload.

The pylon-to-wing installation includes the upper spar fitting (R1), diagonal brace (R2), aft upper fittings (R3/R4), and side links (R7/R8). The R7 and R8 links use corrosion resistant steel bolts. All other attachment fittings and links use corrosion resistant steel fuse and non-fuse pins. The upper link and side links are made from titanium. The diagonal brace link is made from aluminum.

R1 Fitting

The R1 fitting is bolted to the upper spar web, frame 3, and the mid pylon bulkhead. The fitting is manufactured from titanium. Its primary purpose is to transfer the forward/aft and vertical loads of the torque box to the wing via the titanium R1 fitting link.

R2 Fitting

The R2 fitting absorbs the loads from the aft engine mount bulkhead, and the torque box structure, and transfers them to the wing via the aluminum diagonal brace. The diagonal brace is connected to the R2 fitting with a double shear non-fuse type pin. The fitting is manufactured from titanium.

R3 and R4 Fittings

The R3 and R4 fittings make up the chords along the aft section of the upper spar. The fittings extend downward as they attach to the aft strut bulkhead, R2 fitting, and lower spar chords. The fittings are made of titanium and attach the pylon assembly to the mid spar of the wing by two double shear fuse pins. These fittings transfer the lateral loads, as well as loads due to yaw, roll, and torque, to the wing structure.

R7 and R8 Links

At their lower ends, the titanium R7 and R8 links attach to the R3 and R4 fittings respectively. The upper link ends attach to wing fittings, and transfer lateral loads to the wing fittings.

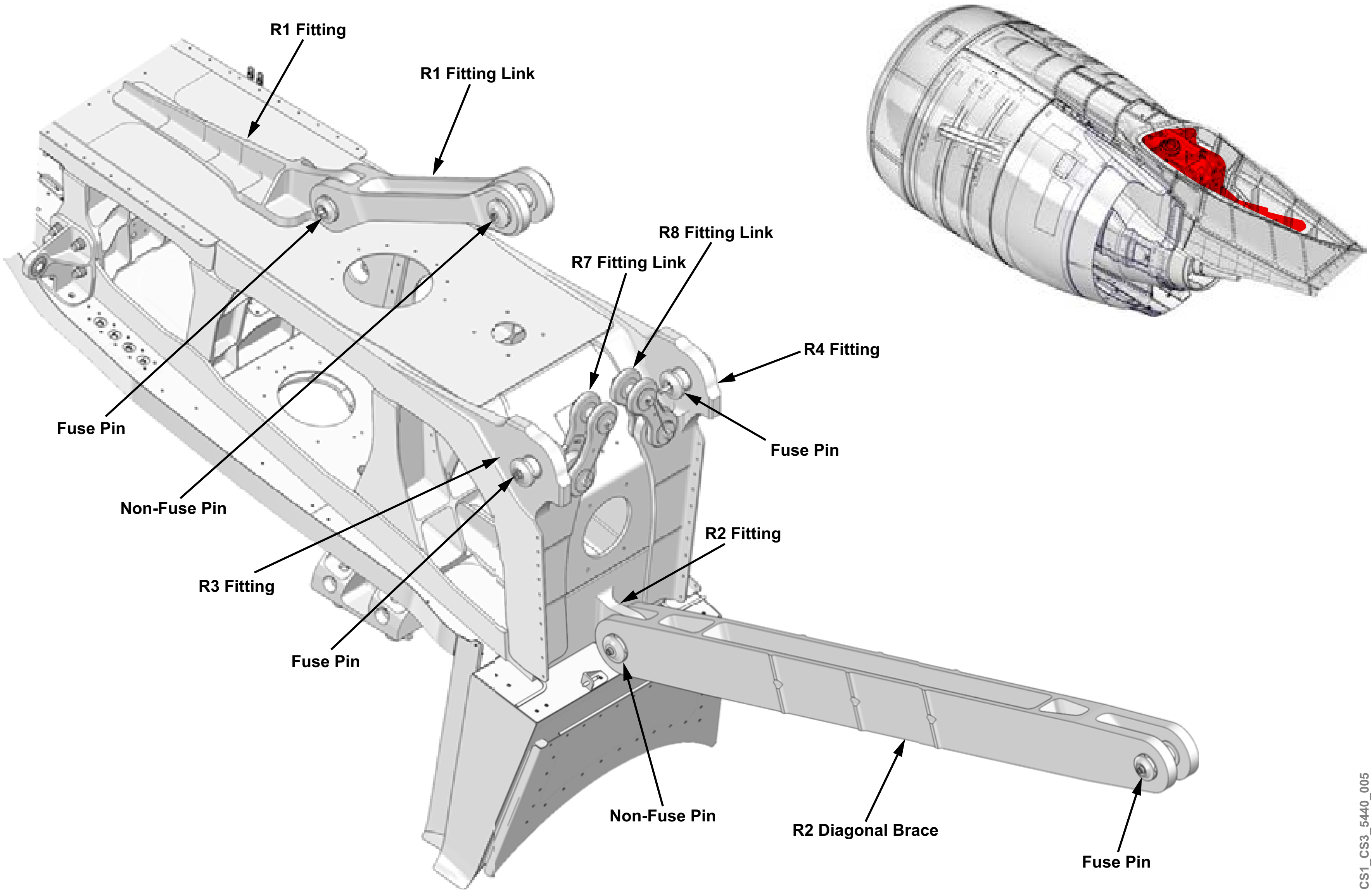


Figure 5: Pylon-To-Wing Attachments

ENGINE MOUNTS

Forward Engine Mount

The forward engine mount bulkhead (FEMB) is manufactured from aluminum alloy plate. The aft end of the FEMB is secured to the pylon torque box upper and lower spars, and the side skins. The forward end is connected to the fan cowling support beam through tie rods.

The main function of the FEMB is to absorb engine loads and transfer them into the pylon torque box. The fitting also provides supports for engine change ground support equipment (GSE).

Vertical and torsional loads are transferred through the FEMB by four 1.90 cm (0.75 in) diameter tension bolts. Lateral loads are transferred through the FEMB by two 1.90 cm (0.75 in) diameter shear pins.

Aft Engine Mount Fitting

The aft engine mount fitting (AEMF) is manufactured from titanium. The AEMF reacts to the engine mount loads and distributes them into the pylon torque box.

The lateral loads are transferred through the AEMF by 2.22 cm (0.875 in) diameter shear pins. Vertical and torsional loads are transferred through four 1.90 cm (0.750 in) diameter tension bolts. Two lugs, on the forward side of the AEMF, serve as supports for precooler fittings. Lugs on the aft side of the AEMF are for engine change GSE.

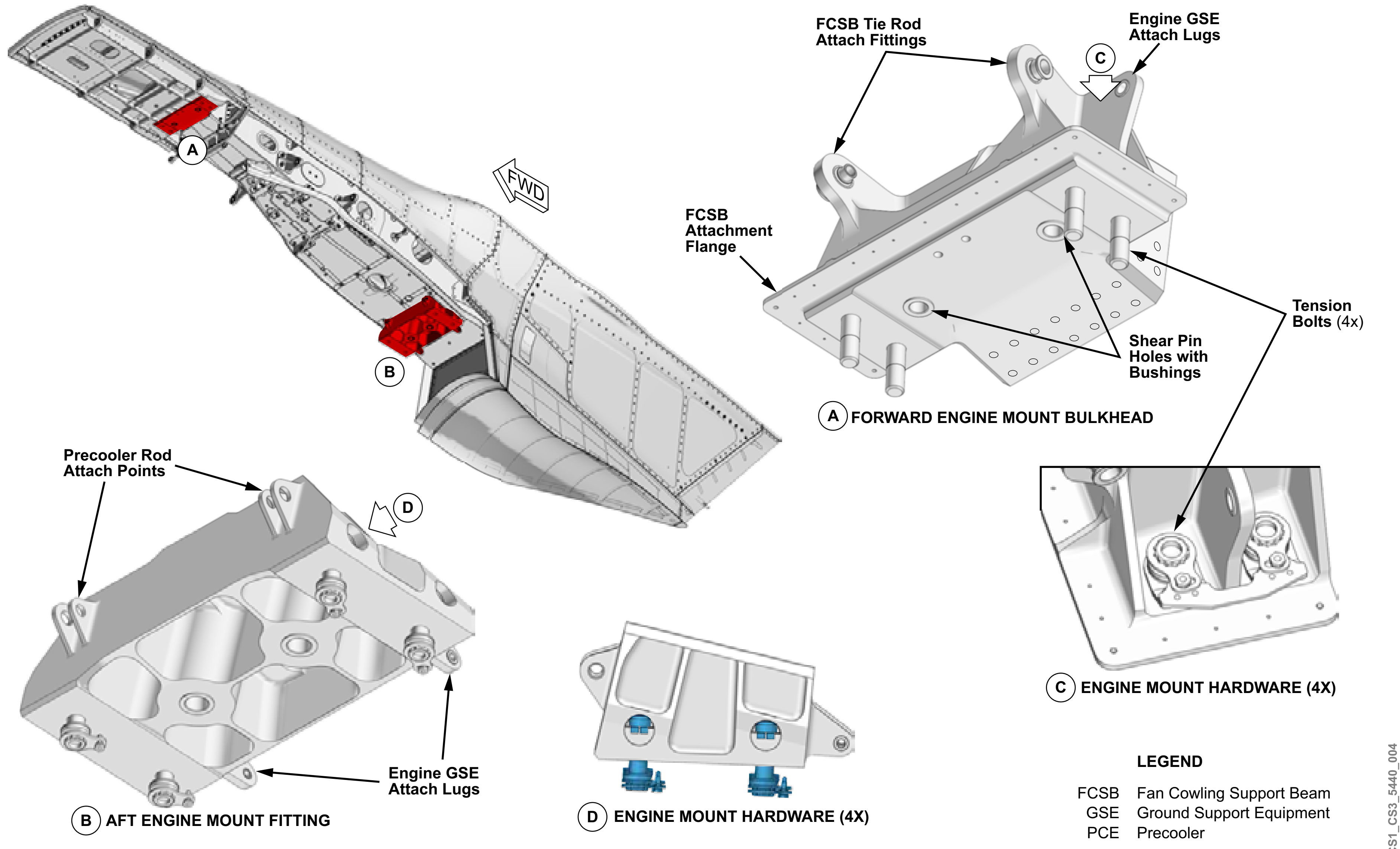


Figure 6: Engine Mounts

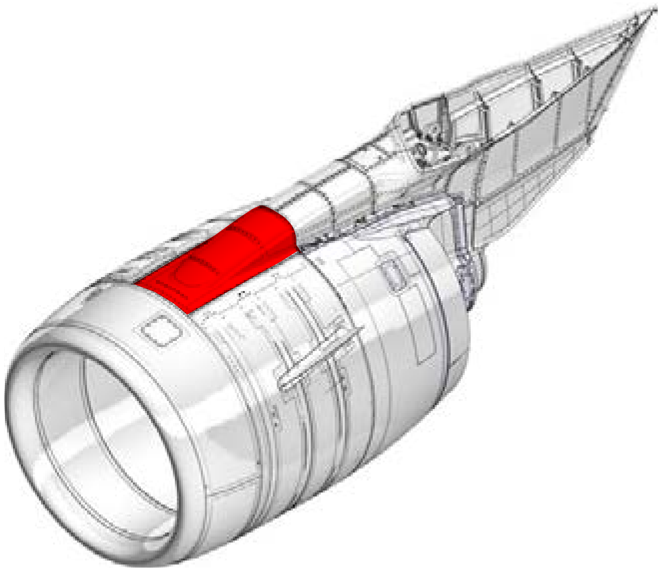
FAN COWLING SUPPORT BEAM

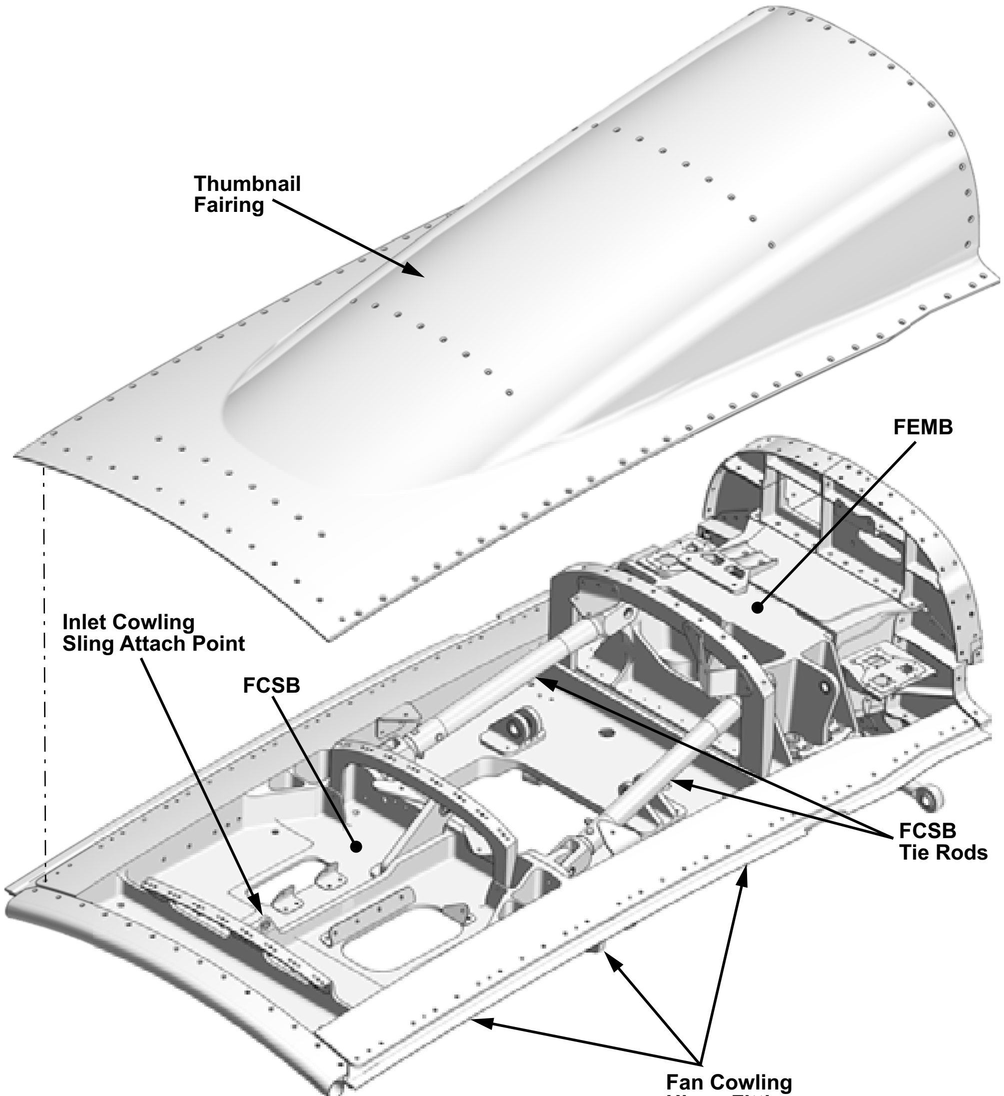
The fan cowling support beam (FCSB) structure provides support for both the fan cowling doors of the nacelle, and the forward upper thumbnail fairing.

The FCSB also provides support for electrical and hydraulic systems lines. The FCSB structure is machined out of a single aluminum alloy plate. The plate attaches to the front and sides of the forward engine mount bulkhead (FEMB) through a single shear joint.

Two titanium tie rods with machined aluminum alloy end fittings connect the FCSB to lugs on the top the FEMB. These attachments transfer loads from the FCSB to the pylon torque box.

The support beam also has a fitting for the attachment of the inlet cowling sling assembly.





LEGEND

FEMB	Forward Engine Mount Bulkhead
FCSB	Fan Cowling Support Beam

Figure 7: Fan Cowling Support Beam

PYLON FAIRINGS AND PRESSURE-RELIEF DOOR

The fairings that enclose the pylon provide an aerodynamic surface. The thumbnail fairing and both aft fairings are manufactured from CFRP, and are secured to the pylon with screws.

The inboard and outboard heat shield fairings, and lower heat shield are manufactured from titanium sheet.

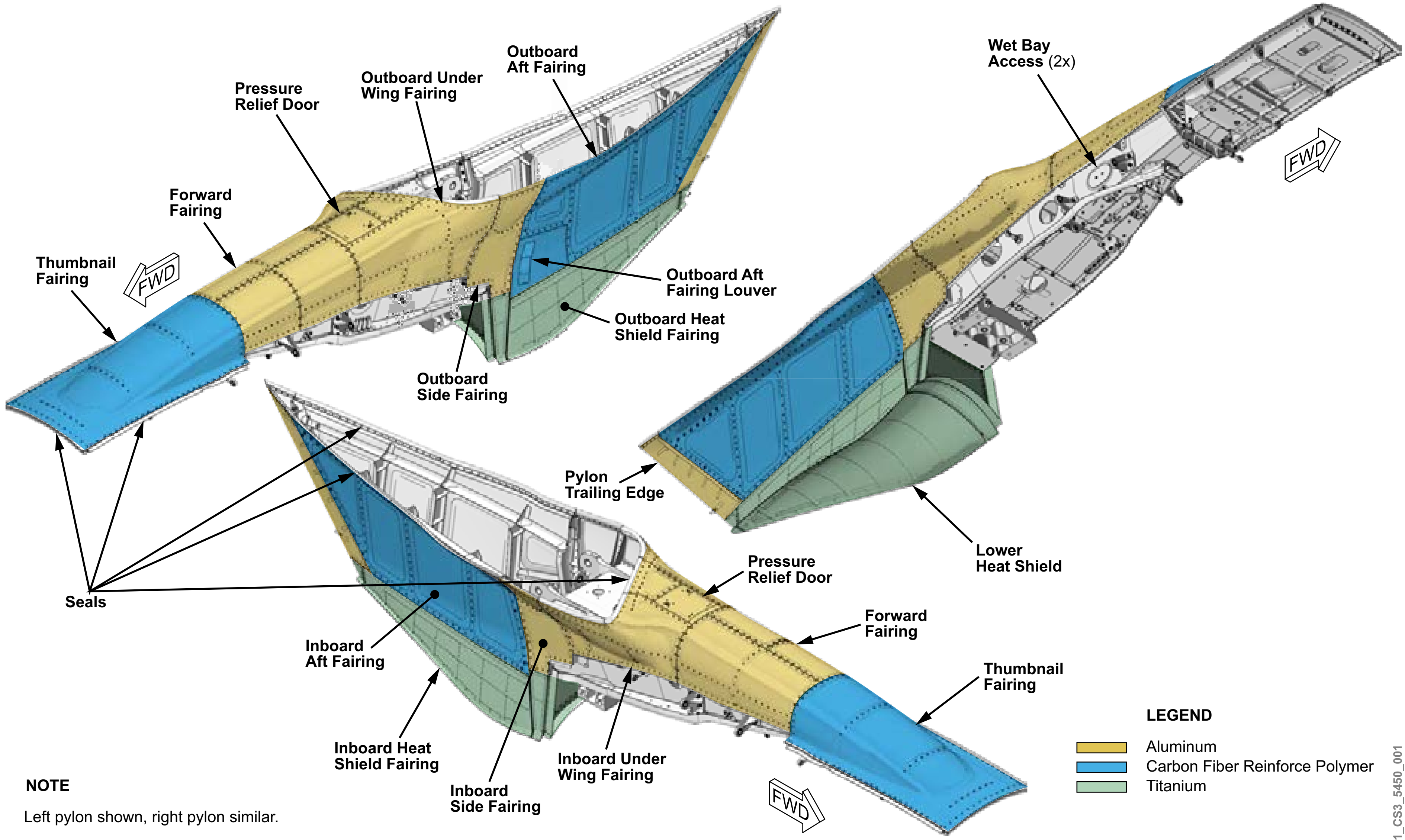
The forward fairing, under wing fairings, pressure-relief door, side fairings and pylon trailing edge are of aluminum construction.

A single pressure-relief door is located on the top of the pylon. The purpose of the door is to relieve excessive air pressure build-ups in the pylon and engine nacelle areas.

The aft fairing louver, located on the outboard side of the pylon, ventilates the aft pylon fairing and front leading edge of the wing.

Two access panels are provided on each sides of the torque box to access the inside of the wet bay for maintenance purposes.

All fairing panels are fitted with different type of seals to ensure aerodynamic airflow around them and sealing of the applicable compartment.



NOTE

Left pylon shown, right pylon similar.

PYLON FAIRINGS AND DOOR

Figure 8: Pylon Fairings and Pressure-Relief Door

PYLON DRAINAGE

The pylon drainage is divided into four paths:

FUEL LINE AND WET BAY

The engine feed and motive flow fuel lines in the pylon are double walled with an inner line and outer shroud. The inner line carries the fuel to the engine while the outer shroud line collects any fuel leakage from the inner line. Any fuel collected in the shroud can drain into the wet bay or through a drain line in the aft pylon.

The wet bay is a sealed area formed by the pylon torque box and frame 2 and frame 3. The wet bay has single walled fuel and hydraulic lines. The wet bay drains through a line on frame 3. The wet bay drain line and fuel shroud drain lines connect to a T-fitting in the aft pylon.

The drain line extends through the aft pylon and drains overboard. A flame arrestor in the line prevents flames from coming in contact with fuel.

TORQUE BOX CONDENSATION DRAINS

The interior compartments of the pylon torque box permit fluid from condensation to flow aft and down through the space between the lower spar chords, web, and frames. This drainage path is through two tubes in the wet bay section to keep it sealed from the other torque box bays.

Two similar tubes, located at the aft end of the pylon torque box, route the condensation drainage fluid out the side of the pylon through the torque box condensation overboard drain.

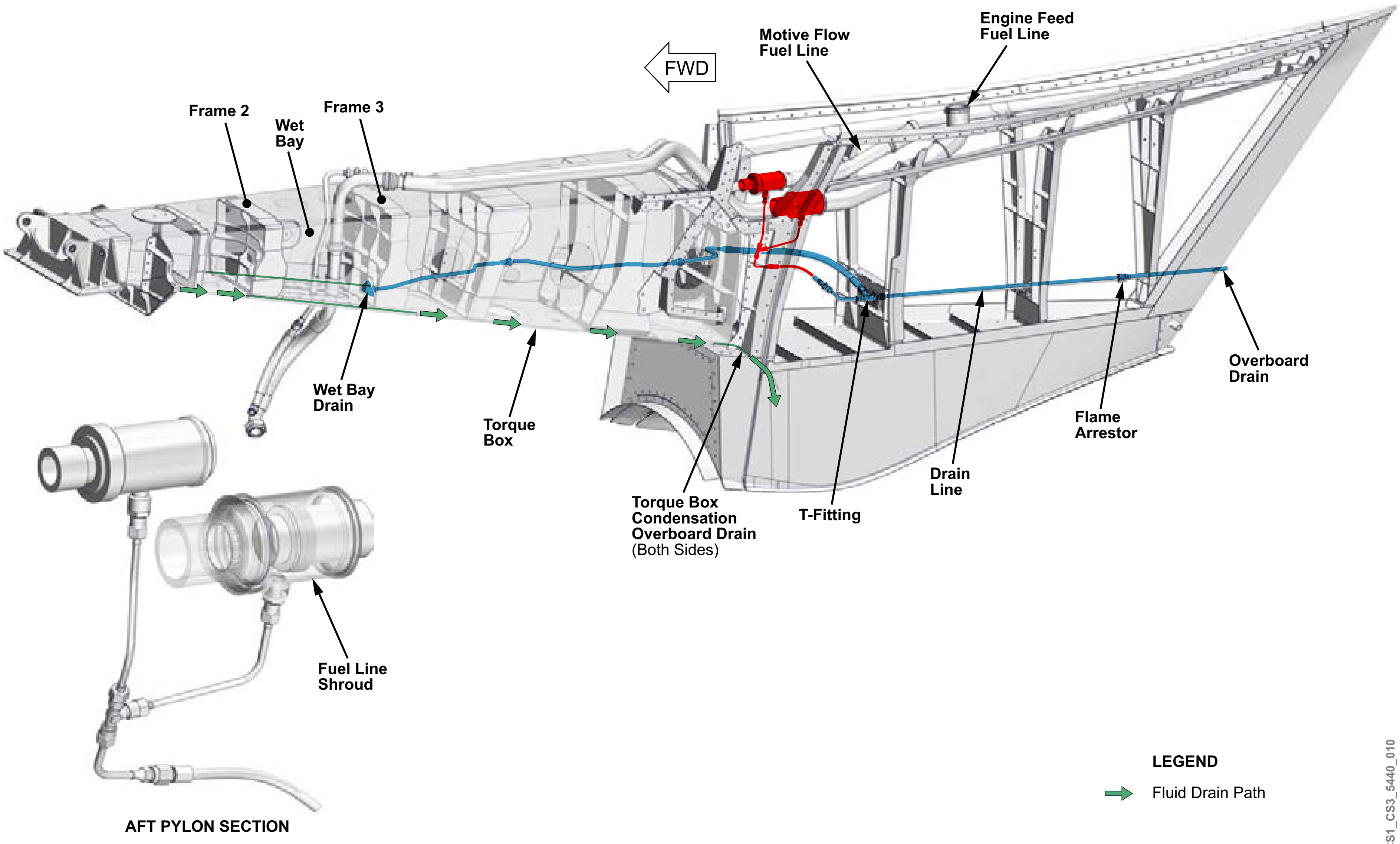


Figure 9: Fuel Line, Wet Bay, and Torque Box Drainage

PYLON UPPER WEB AND AFT PYLON FAIRING

Condensation or single wall hydraulic line leak from beneath the pylon fairing panels or aft pylon fairing under the wing is drained overboard at the aft low end of the pylon aft fairing.

Fluid drains aft along the upper pylon web and over the aft pylon bulkhead, down onto the upper surface of the pylon aft fairing upper web. From the pylon aft fairing upper web forward end, drainage path is along the upper web surface aft to the outboard drain outside the pylon aft end.

FAN COWLING SUPPORT BEAM

Openings in the fan cowling support beam (FCSB) lower surface permit condensation and hydraulic fluid leak drainage down over the fan case area. This drainage is structural only and there are no tubes nor lines involved.

NOTE

Any condensation entering inside the pylon aft lower heat shield would be evaporated as this area gets above water boiling temperature from the exhaust gas path radiation.

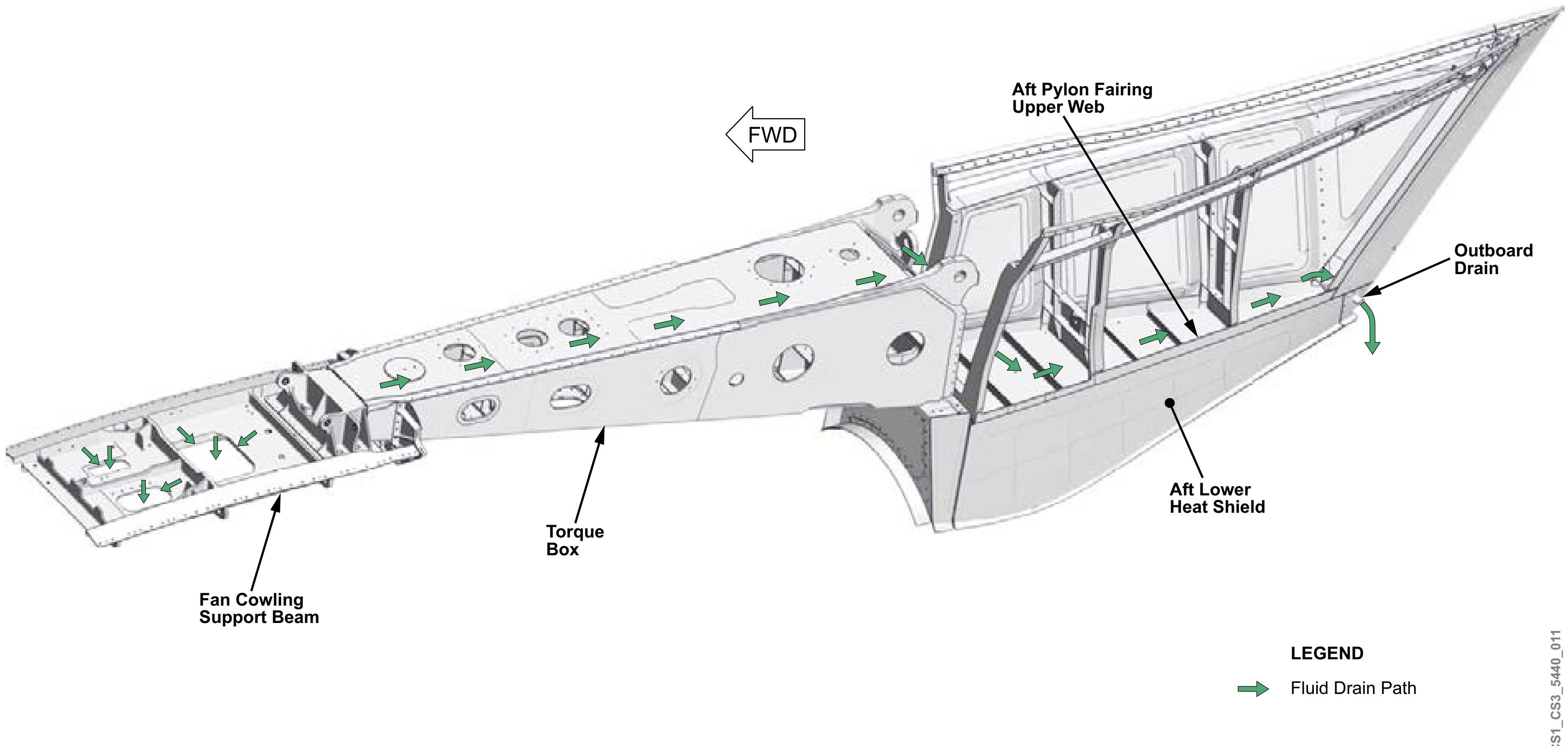


Figure 10: Pylon Upper Web, Aft Pylon, and Fan Cowling Support Beam Drainage

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ATA 55 - Stabilizers

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

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STABILIZERS - CHAPTER BREAKDOWN

55-10 HORIZONTAL STABILIZERS

GENERAL DESCRIPTION

The left and right horizontal stabilizers are connected at their inboard sections to the horizontal stabilizer center box. Each of the three horizontal stabilizer assemblies are manufactured from carbon fiber reinforced polymer (CFRP). A CFRP elevator is hinged to the trailing edge of each horizontal stabilizer.

The horizontal stabilizer assembly is connected to the fuselage structure by two failsafe type pivot pins, located at the center box rear spar. The leading edge of the center box structure has a two-piece fitting, for attachment to the horizontal stabilizer trim actuator.

The left and right horizontal stabilizer multispar boxes have front and rear spars, leading edge and trailing edge assemblies. Logo lights are mounted on the upper surfaces of the left and right trailing edge assemblies.

A tip assembly closes the outboard end of each multispar box and provides aerodynamic shape. Visor fairings that follow the movement of the horizontal stabilizer are located on the upper and lower inboard ends of the left and right multispar box. The visor fairings provide closeout and aerodynamic shape.

COMPONENT LOCATION

The horizontal stabilizer consists of the following principal components:

- Multispar box
- Leading edge assembly
- Tip assembly
- Trailing edge assembly
- Visor fairing
- Horizontal stabilizer center box

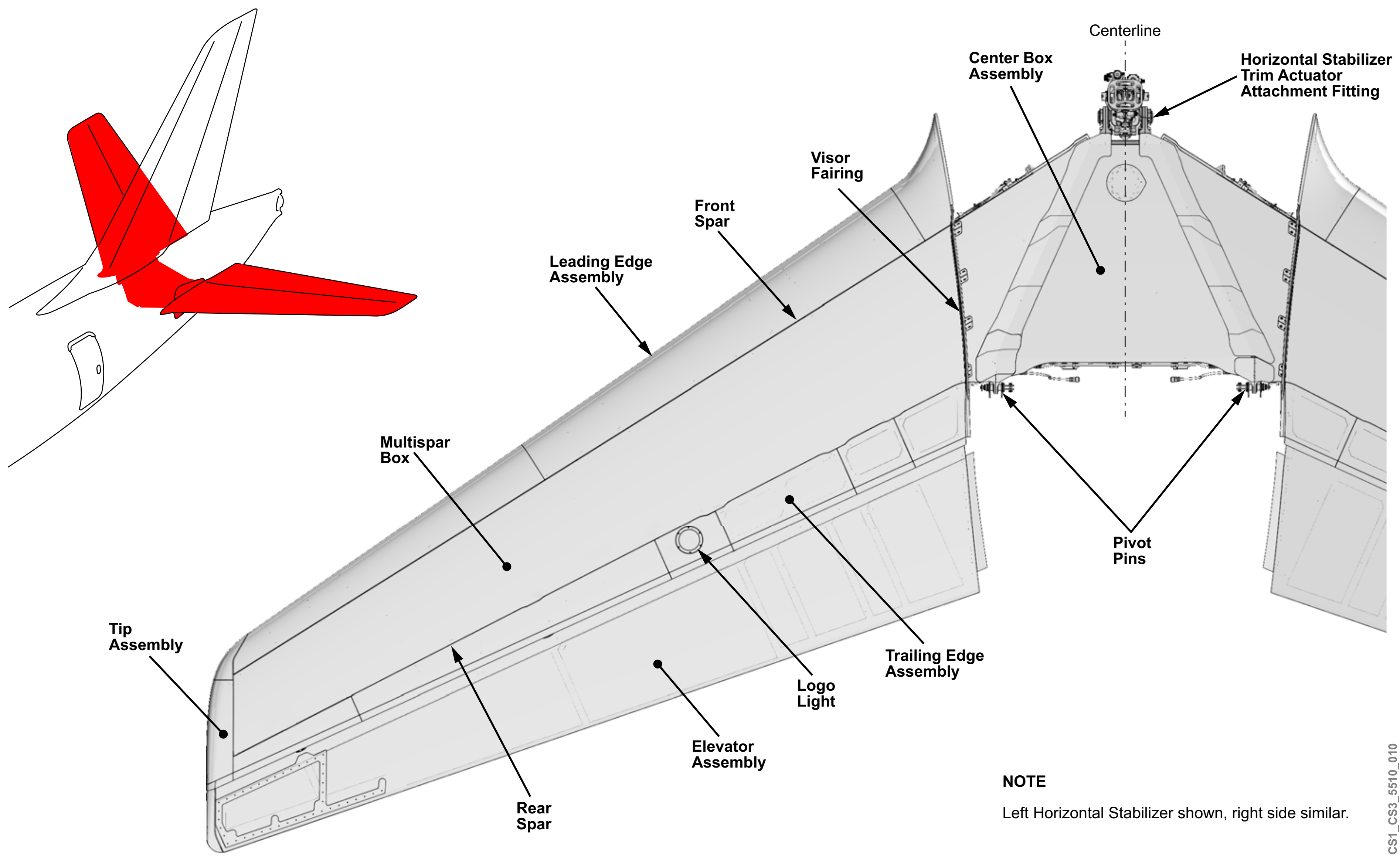


Figure 1: Horizontal Stabilizer Assembly

DETAILED COMPONENT INFORMATION

MULTISPAR BOX

The multispar box is a CFRP multispar structure. It consists of a forward spar, rear spar, no. 1 mid spar, and a no. 2 mid spar. The spars are sandwiched between an upper and lower skin.

The multispar box is a co-cured assembly. The uncured spars are bonded to the uncured skins, and the complete assembly is allowed to cure at the same time. This type of construction greatly enhances the strength of the multispar box.

The assembly is further strengthened by fastening the skins to the spars with antipeel bolts, using a spacing of approximately 30.5 cm (12 in).

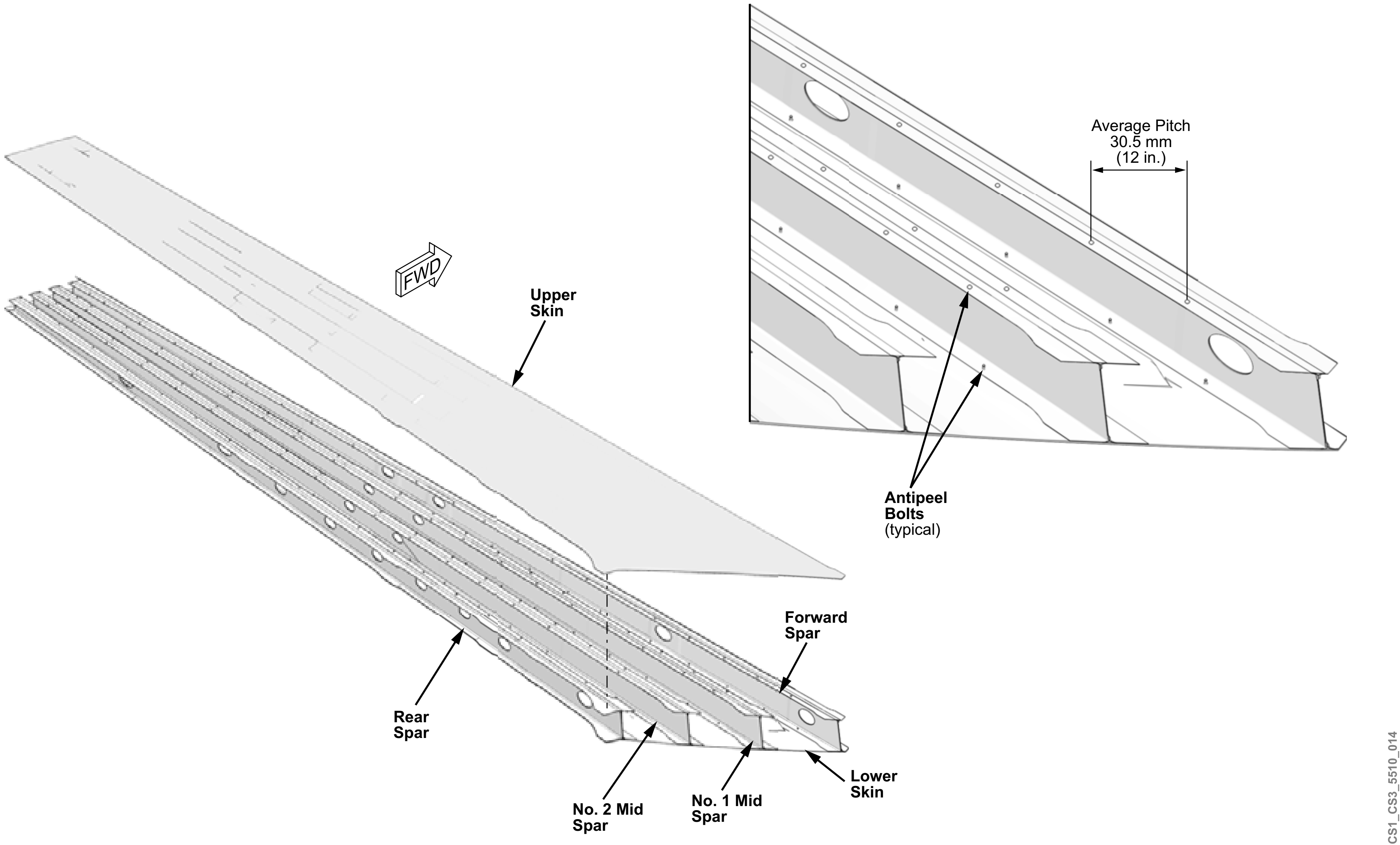


Figure 2: Multispar Box

LEADING EDGE ASSEMBLY

The left and right horizontal stabilizers are provided with a leading edge assembly. The aluminum alloy leading edge assembly consists of three segments:

- Outboard leading edge
- Inboard leading edge
- Cuff fairing

The assembly is fastened to the upper and lower flanges of the horizontal stabilizer forward spar with standard fasteners and nut plates. Aluminum alloy splice rib 1 and splice rib 2, provide a means of mechanically securing and splicing the three leading edge segments.

The cuff fairing at the inboard end of the leading edge assembly provides a smooth aerodynamic transition toward the horizontal stabilizer surfaces. It consists of a preformed aluminum alloy skin, and a single rib. The cuff fairing is fastened to splice rib 1 and to the forward spar of the multispar box.

TIP ASSEMBLY

Each left and right horizontal stabilizer has a tip assembly at the outboard end. The aluminum alloy tip assembly consists of a tip closure rib, tip casting, and a fairing assembly. The tip assembly is fastened to its respective horizontal stabilizer with standard aircraft screws and nut plates.

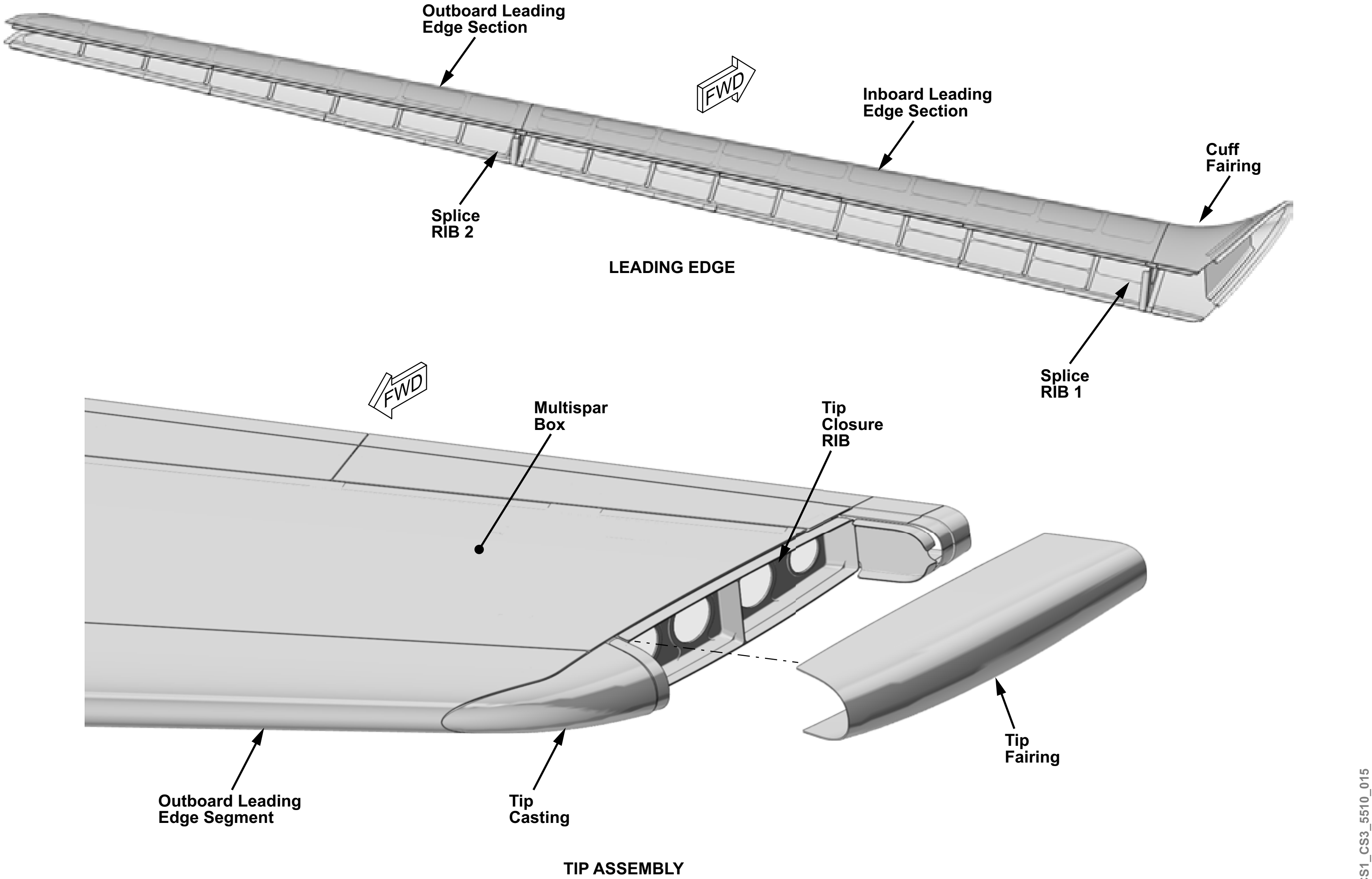


Figure 3: Leading Edge and Tip Assemblies

TRAILING EDGE ASSEMBLY

The trailing edge assembly is a build-up of the following components:

- Elevator hinge ribs
- Elevator power control unit (PCU) fittings
- Carbon fiber reinforced polymer (CFRP) support beams
- Shroud ribs
- Trailing edge panels

The elevator hinge ribs, elevator PCU fittings, and shroud ribs are machined from aluminum alloy plate. The shroud ribs provide support for the upper and lower trailing edge panels.

All ribs are connected to the rear spar box with CFRP rib angles. Trailing edge panels are manufactured from CFRP fabric with honeycomb cores. The PCU access panels are solid CFRP laminate panels.

The forward edges of the upper and lower trailing edge panels are attached to the rear spar of the multispar box with CFRP straps. All the trailing edge lower panels are removable for maintenance or servicing. The upper trailing edge panel outboard of the outboard elevator PCU, is provided with a cutout for a logo light fixture. The upper outboard elevator PCU access panel is removable.

Two CFRP beams provide support for the trailing edge panels at their rear edges. A P-seal on the aft upper and lower edges of the beams provide aerodynamic sealing for the elevator.

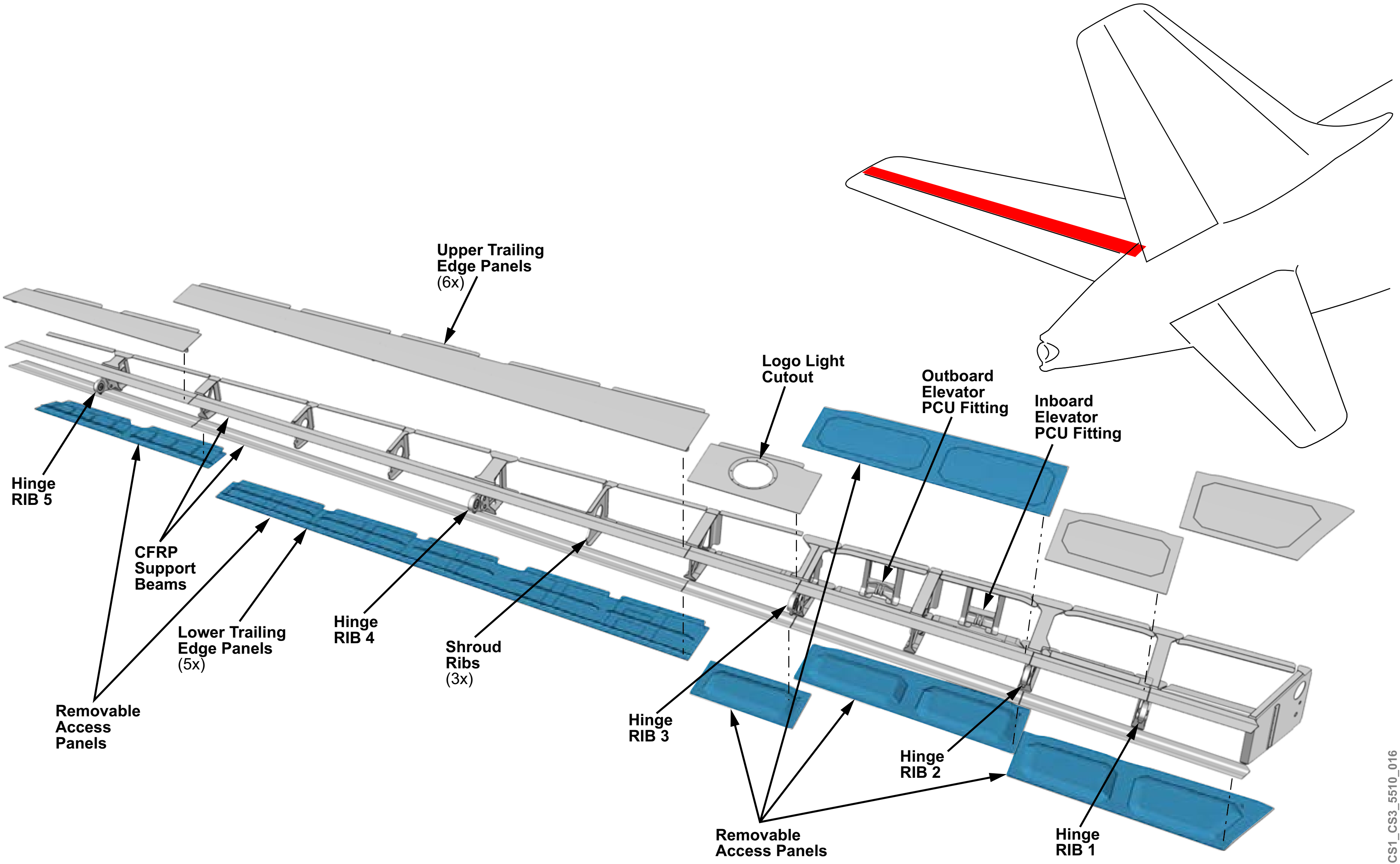


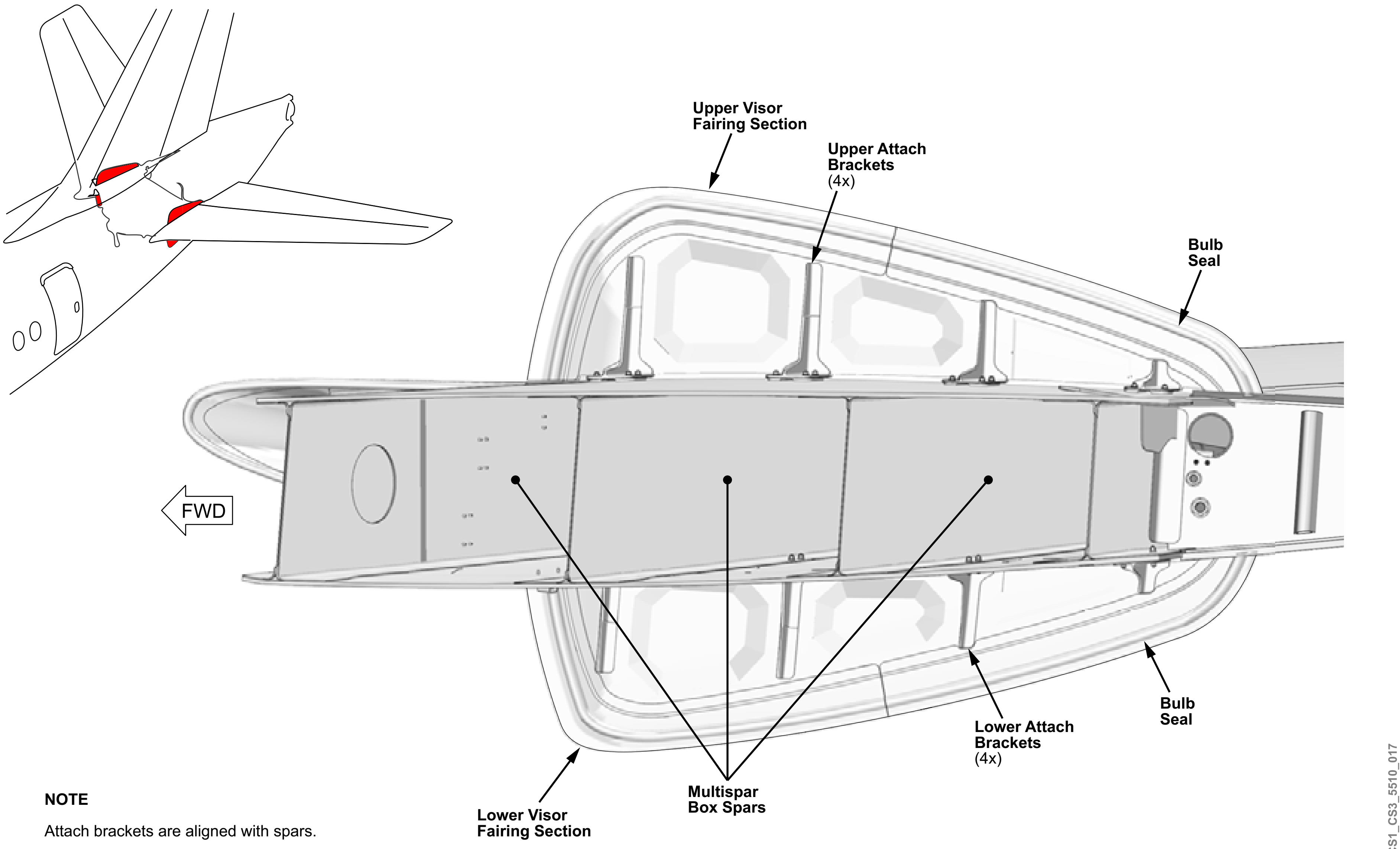
Figure 4: Trailing Edge Assembly

VISOR FAIRING

The visor fairing provides an aerodynamic seal over the fuselage cutout, which provides horizontal stabilizer movement. The visor fairings are removable, and are fastened to machined aluminum alloy brackets and plates located on the spars of the stabilizer center box.

The CFRP visor fairing contains a honeycomb core. A bulb seal is installed along its edges that are in contact with the aft fuselage.

The visor fairing follows the movement of the attached horizontal stabilizer. To prevent binding, installation procedures must be followed in accordance with the Aircraft Maintenance Publication (AMP).



NOTE

Attach brackets are aligned with spars.

Figure 5: Visor Fairing

HORIZONTAL STABILIZER CENTER BOX

The horizontal stabilizer center box is a major load bearing structural assembly, and serves as the attachment point for the left and right horizontal stabilizers.

The horizontal stabilizer center box consists of the following structural members:

- Upper and lower skins
- Mid spars
- Rear spar
- Splice ribs
- Trim actuator fitting assembly
- Pivot fittings
- Access panels

The upper and lower skin panels are CFRP solid laminate, with honeycomb cores.

The two CFRP mid spars and rear spar are fastened to the upper and lower skin panels. The spars are attached to the splice ribs by two machined titanium attach angles. The splice rib is an assembly consisting of upper and lower machined titanium T-chords, which are mechanically attached to a carbon fiber laminate shear web. Two CFRP attach angles, located on the shear web, mechanically attach to the mid spars of the left and right horizontal stabilizer. The splice ribs form the angled sides of the horizontal stabilizer center box.

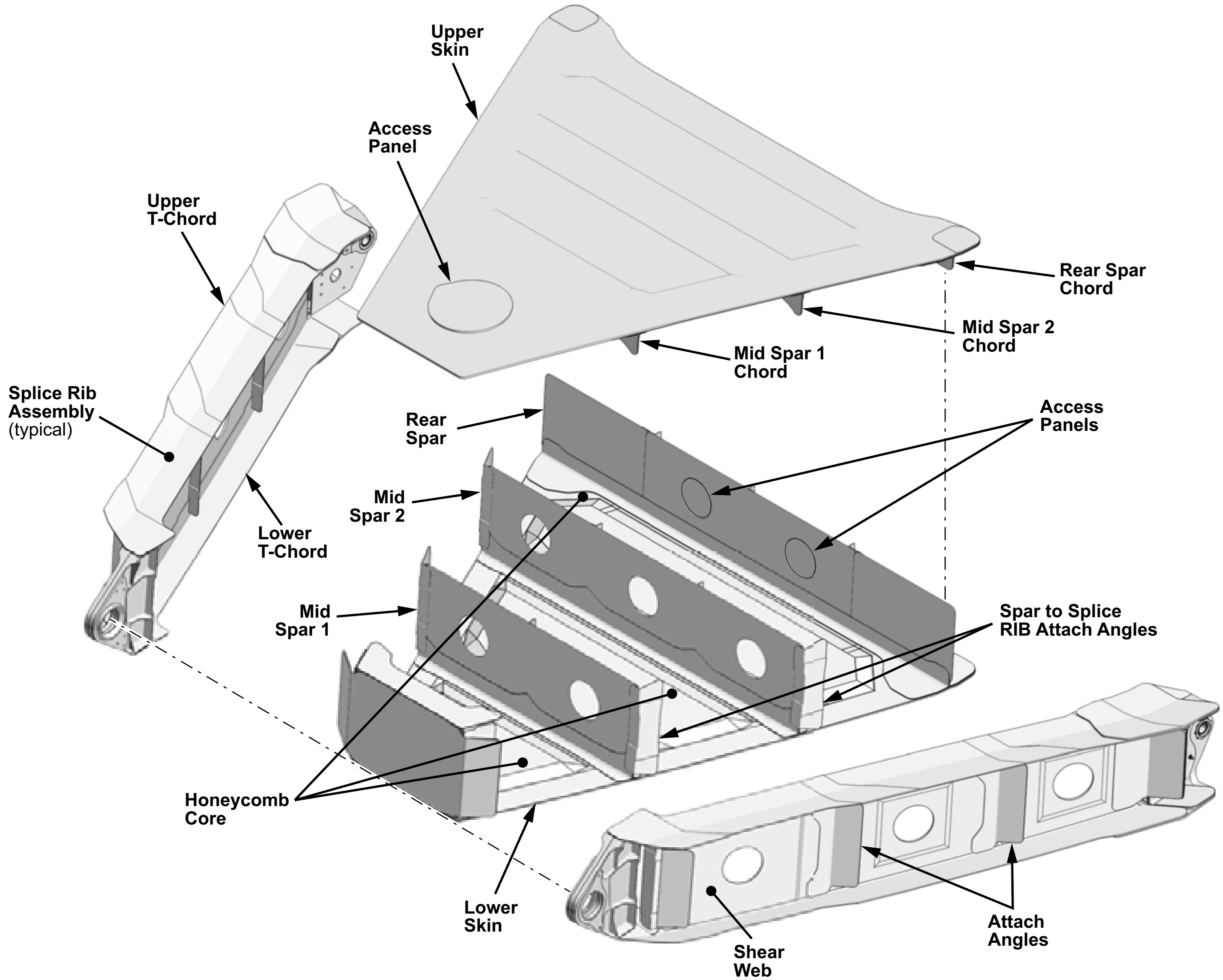


Figure 6: Center Box Skins, Spars, and Ribs

Horizontal Stabilizer Trim Actuator Fitting

The trim actuator attachment is located at the forward end of the center box assembly. It consists mainly of an attachment splice for mounting to the center box assembly, and two actuator fitting assemblies.

Each actuator attach fitting assembly consists of a primary and secondary load path fitting connected to form a single unit. The dual-load path concept provides a failsafe attachment to the actuator. The fittings are made from machined aluminum alloy plate.

Pivot Fittings

To provide aircraft trim adjustment, the stabilizer rotates about an axis through the pivot fittings.

The two-piece pivot fittings are manufactured from machined titanium plate. Each fitting is comprised of two halves secured back-to-back for failsafe purposes. Each fitting assembly is attached to a backup fitting for added strength. The pivot fittings interface with the aircraft structure through pivot bolts on the left and right sides of the center box aft spar.

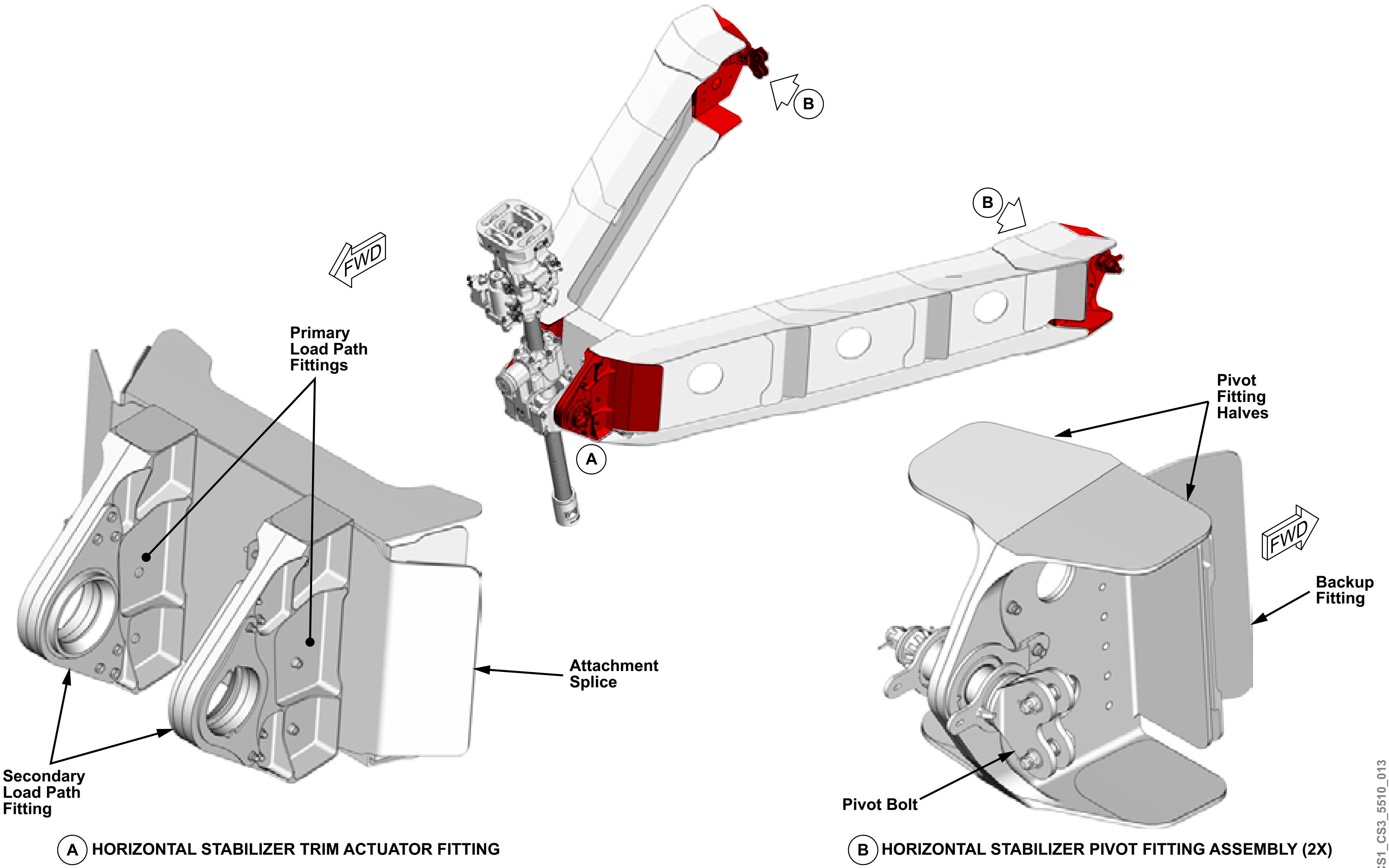


Figure 7: Horizontal Stabilizer Trim Actuator Fittings and Pivot Fittings

55-20 ELEVATORS

GENERAL DESCRIPTION

Two one-piece elevator assemblies are mounted on the rear spars of the horizontal stabilizers. The elevators are attached by five hinge fittings, and two elevator power control unit (PCU) attach fittings, mounted on the elevator front spar.

DETAILED COMPONENT INFORMATION

The left and right elevators are built-up structures. They consist of the following structural members:

- Carbon fiber reinforced polymer (CFRP) upper and lower skin panels
- Front spar
- Intermediate ribs
- Inboard closure rib
- Tip

The upper and lower skins are comprised of a honeycomb core sandwiched between carbon fiber reinforced polymer (CFRP) fabric panels. The spars, intermediate and inboard closure ribs are manufactured from CFRP solid laminate, while the tip is made of aluminum alloy.

The front spars have five machined aluminum alloy hinge fittings, which interface with the horizontal stabilizer aft spar fittings. Two machined aluminum alloy elevator PCU attach fittings are also installed on the front spars.

The elevators have upper and lower lightning protection frames, and four static dischargers, which are mounted to the outboard trailing edge structure.

Four hoist fittings are provided on the front spar for elevator removal, and installation purposes using a lifting fixture that is attached to the elevator.

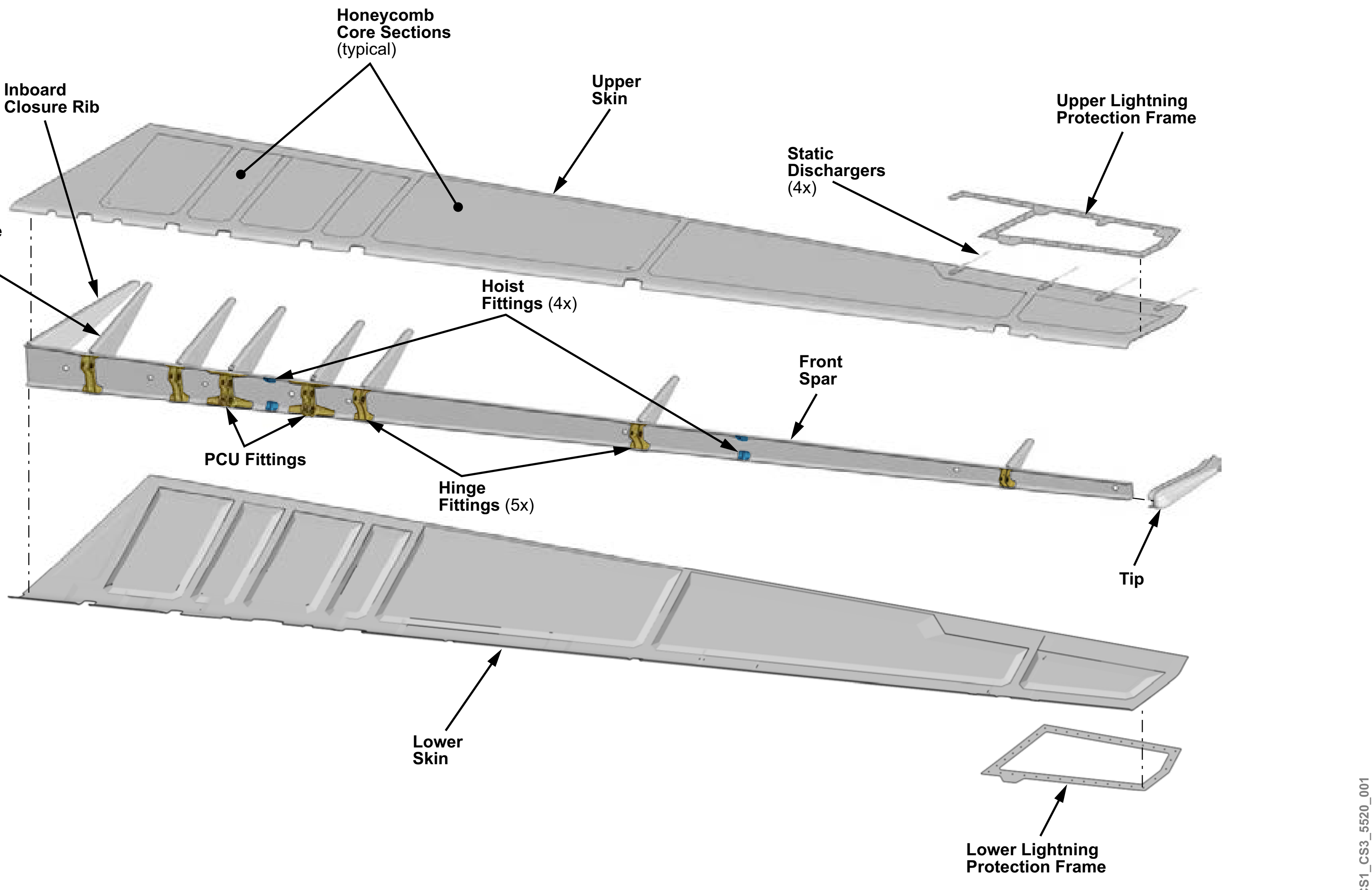


Figure 8: Elevator Assembly

55-30 VERTICAL STABILIZER

GENERAL DESCRIPTION

The vertical stabilizer is mounted on the upper aft fuselage. It is mainly a carbon fiber reinforced polymer (CFRP) torque box assembly. A CFRP rudder is hinged to the rear spar of the vertical stabilizer.

COMPONENT LOCATION

The vertical stabilizer consists of the following principal components:

- CFRP torque box structure
- Aluminum leading edge
- Aluminum leading edge D-nose
- Glass fiber reinforced polymer (GFRP) leading edge panels
- CFRP trailing edge assembly
- Aluminum dorsal fairing
- GFRP tip assembly
- Forward and aft skirt fairings
- Removable access panels

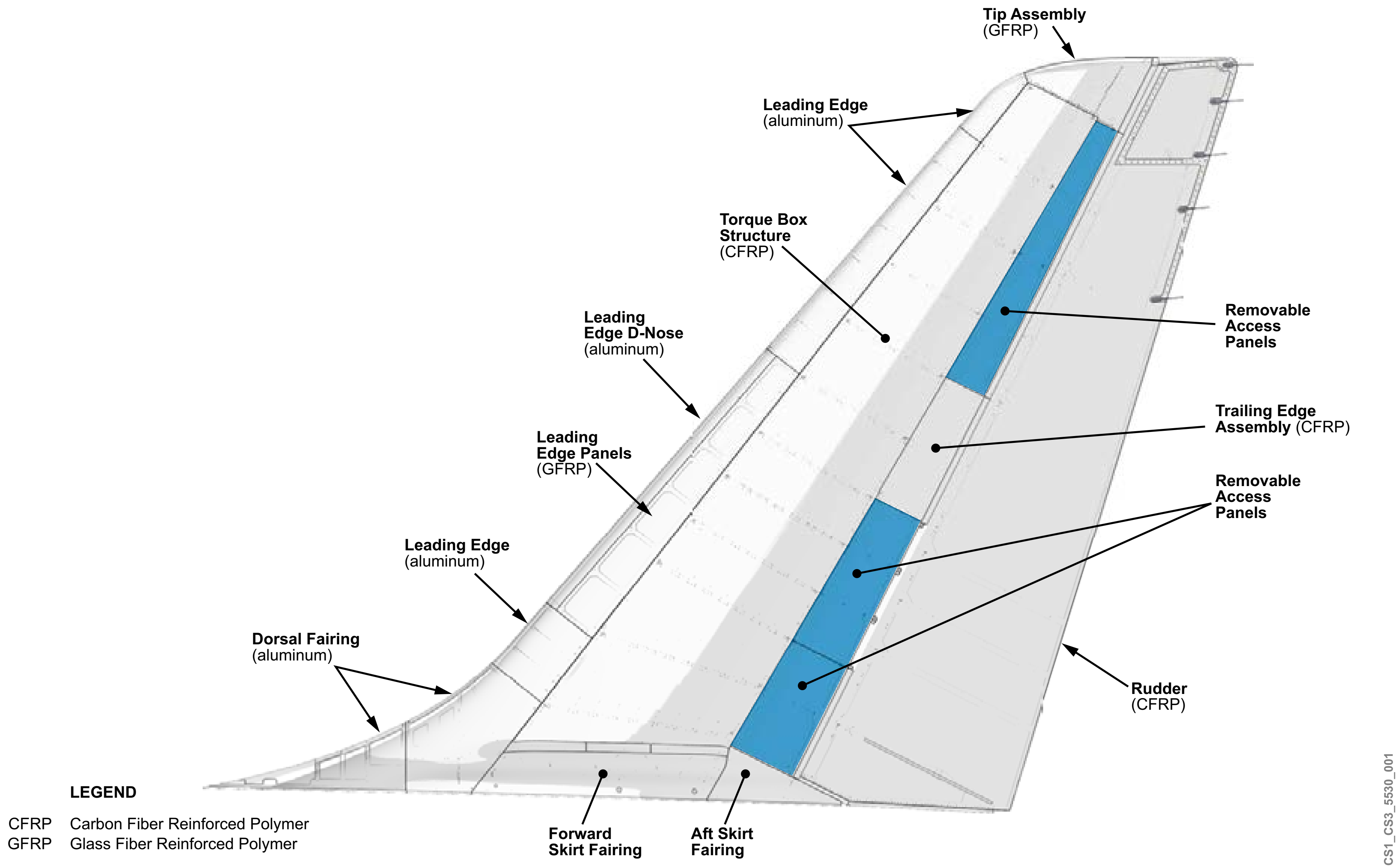


Figure 9: Vertical Stabilizer Assembly

DETAILED COMPONENT INFORMATION

TORQUE BOX STRUCTURE

Spars and Fuselage Attach Fittings

The front and rear spars are made of CFRP formed into a C-section, the front spar is oriented backward, and the rear spar is oriented forward. Inspection holes are provided for inspection of the areas between the ribs. Fuselage attachment fittings are secured to the forward and aft spars, as well as to a mid spar, which provides a backup element for the mid attachment fittings.

The vertical stabilizer torque box assembly is attached to the aft fuselage through six machined aluminum fittings. Front and rear spar lug fittings, and diagonally mounted links, are used at the front and rear attachment points to transfer side loads to the aircraft structure.

Four hoist fittings are provided on the front spar for vertical stabilizer removal, and installation purposes using a lifting fixture that is attached to the stabilizer.

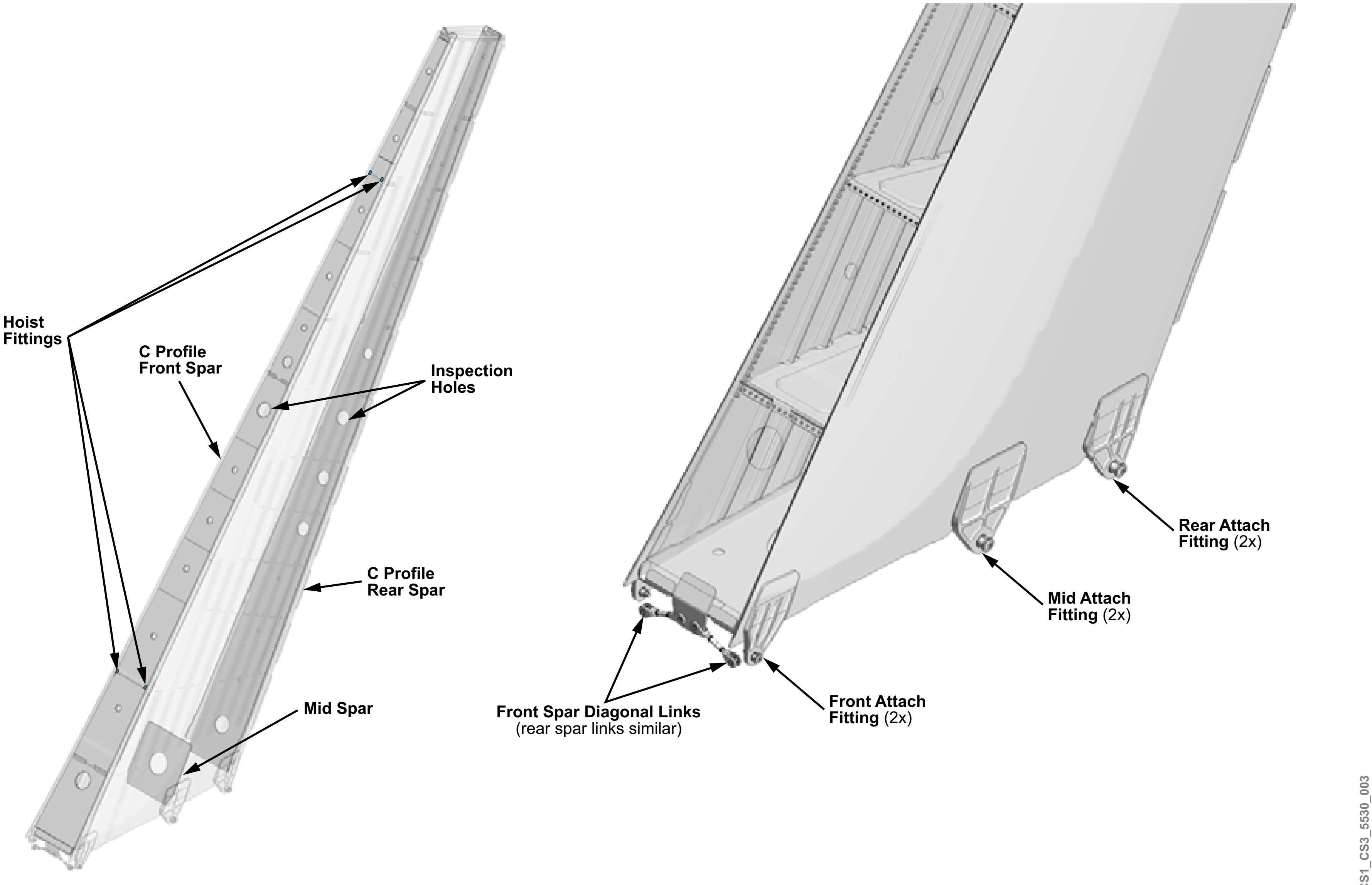


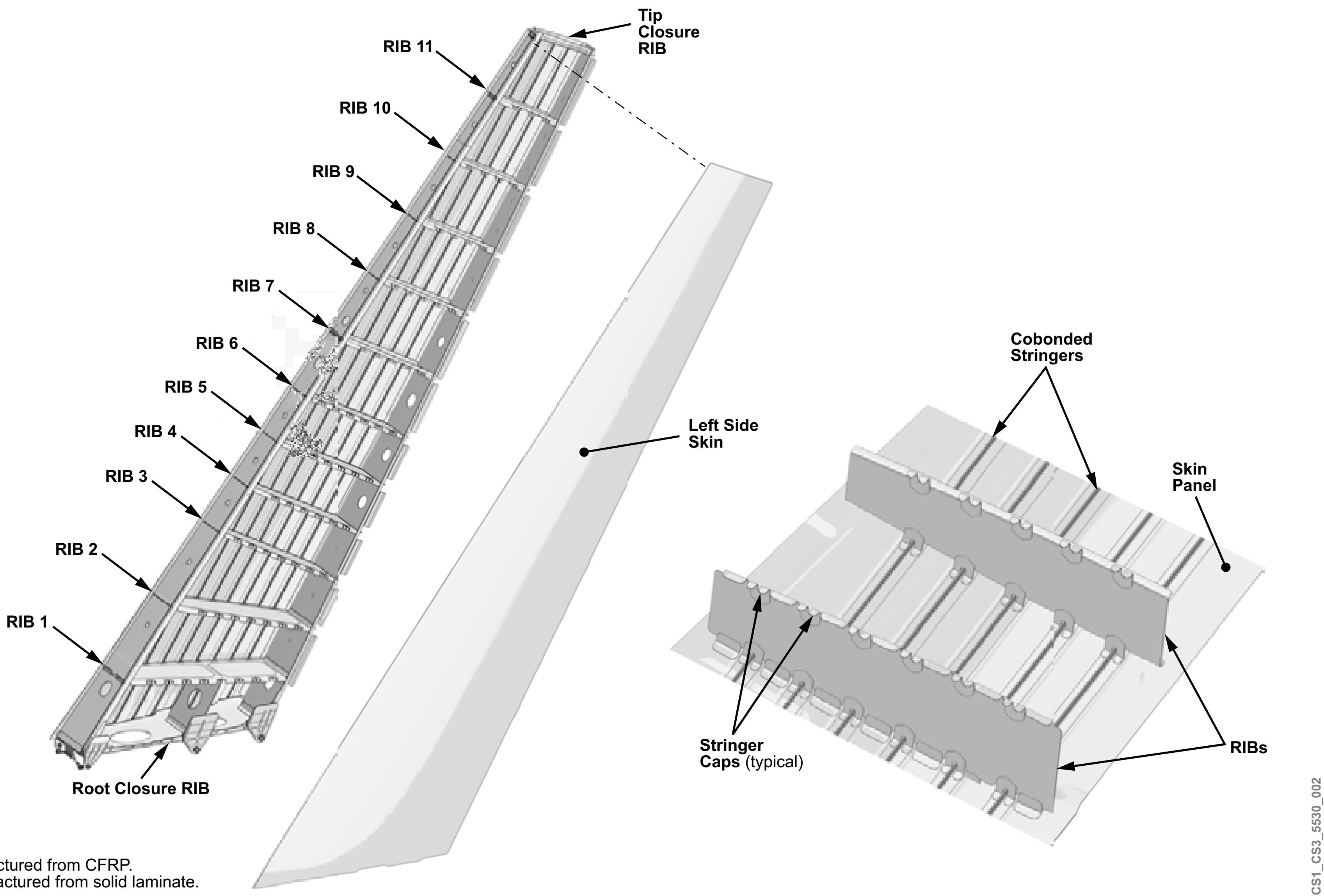
Figure 10: Spars and Fuselage Attach Fittings

Skin Panels and Ribs

The torque box skins are comprised of CFRP panels with eight co-bonded T-shaped stringers spaced approximately 16.25 cm (6 in) apart. The ribs are mechanically attached to the stringers with stringer caps and four bolts. The stringer caps also provide an antipeeling barrier.

The torque box consists of 11 ribs connected to front and rear spars, enclosed by left and right skin panels. Closure ribs are installed at the root and tip of the box assembly. Additional strength is provided by T-profile stringers bonded to the skin panels.

Ribs one to six are manufactured from laminated CFRP fabric, while ribs seven to eleven are solid laminate.



NOTE

RIBs 1-6 manufactured from CFRP.
RIBs 7-11 manufactured from solid laminate.

Figure 11: Skin Panels and Ribs

Vertical Stabilizer Leading Edges

The leading edges are comprised of an upper and lower segment, with a dorsal fairing at the base of the assembly. The upper segment is made from aluminum, with formed ribs, an upper and lower splice rib, reinforcing straps at the aft edges, and a formed aluminum skin panel.

The lower leading edge segment is made from glass fiber reinforced polymer (GFRP) and honeycomb, with an aluminum coupler panel at the lower end of the assembly, and an aluminum D-nose panel. The D-nose panel acts as a high frequency (HF) antenna. The assembly has GFRP ribs and an aluminum trailing edge L-channel for increased strength.

The dorsal fairing is comprised of a forward and aft fairing. Both fairings are made of aluminum, with formed ribs and a silicon bulb seal at the fuselage contact areas.

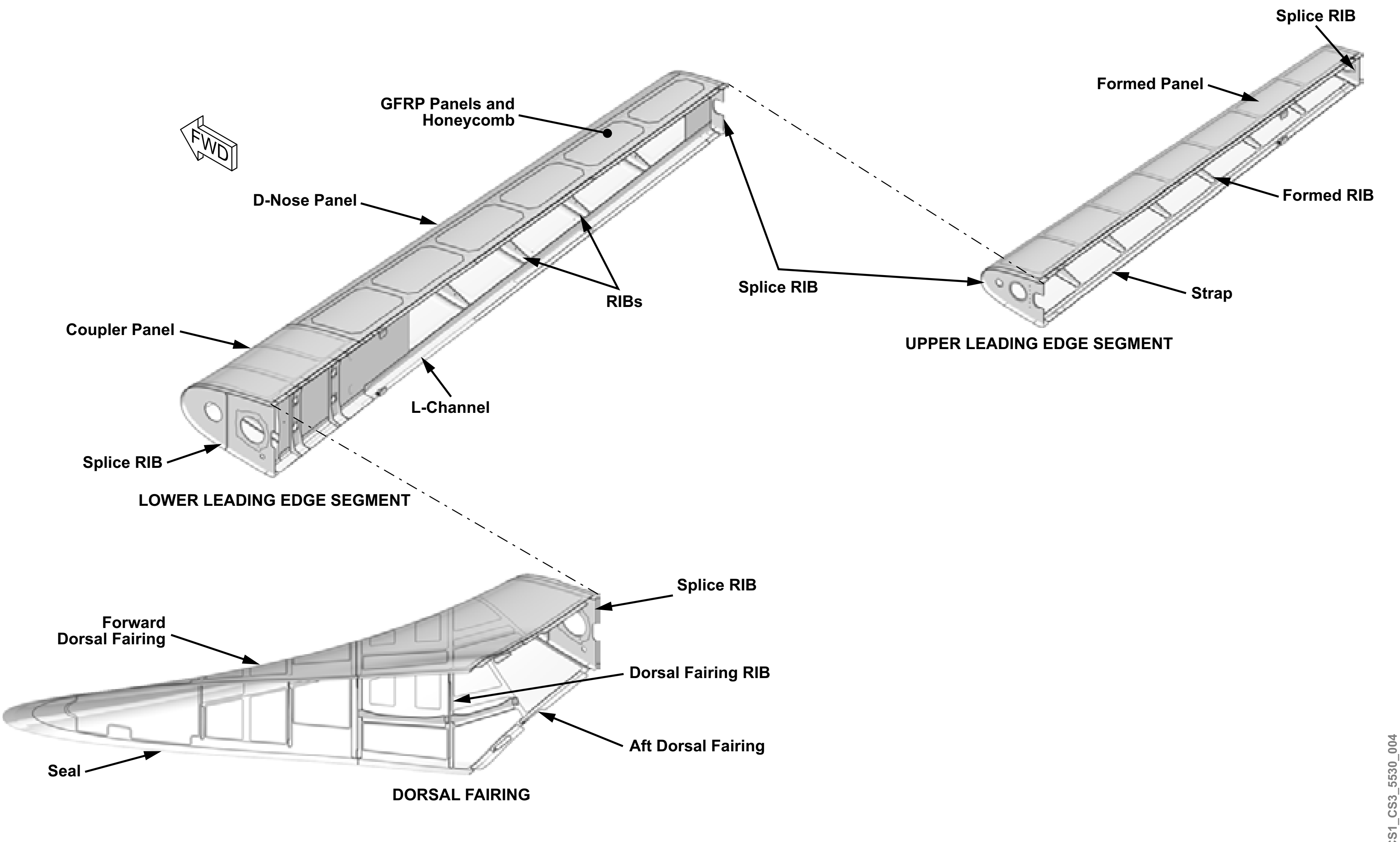


Figure 12: Vertical Stabilizer Leading Edges

Vertical Stabilizer Trailing Edges

The trailing edge assembly consists of a combination of carbon fiber reinforced polymer (CFRP) and aluminum structural members. The assembly is mechanically attached to the rear spar of the vertical stabilizer.

Shroud ribs (3), and hinge ribs (8), are located along the rear spar and provide support for the CFRP trailing edge panels. The hinge ribs include machined aluminum rudder hinge fittings.

The trailing edge panels allow access to trailing edge components, and include silicone bulb seals for weather and aerodynamic sealing between the panels and the rudder assembly.

Three machined aluminum rudder PCU fittings are located at the lower portion of the trailing edge area.

A closure rib is installed at the bottom end of the vertical stabilizer trailing edge.

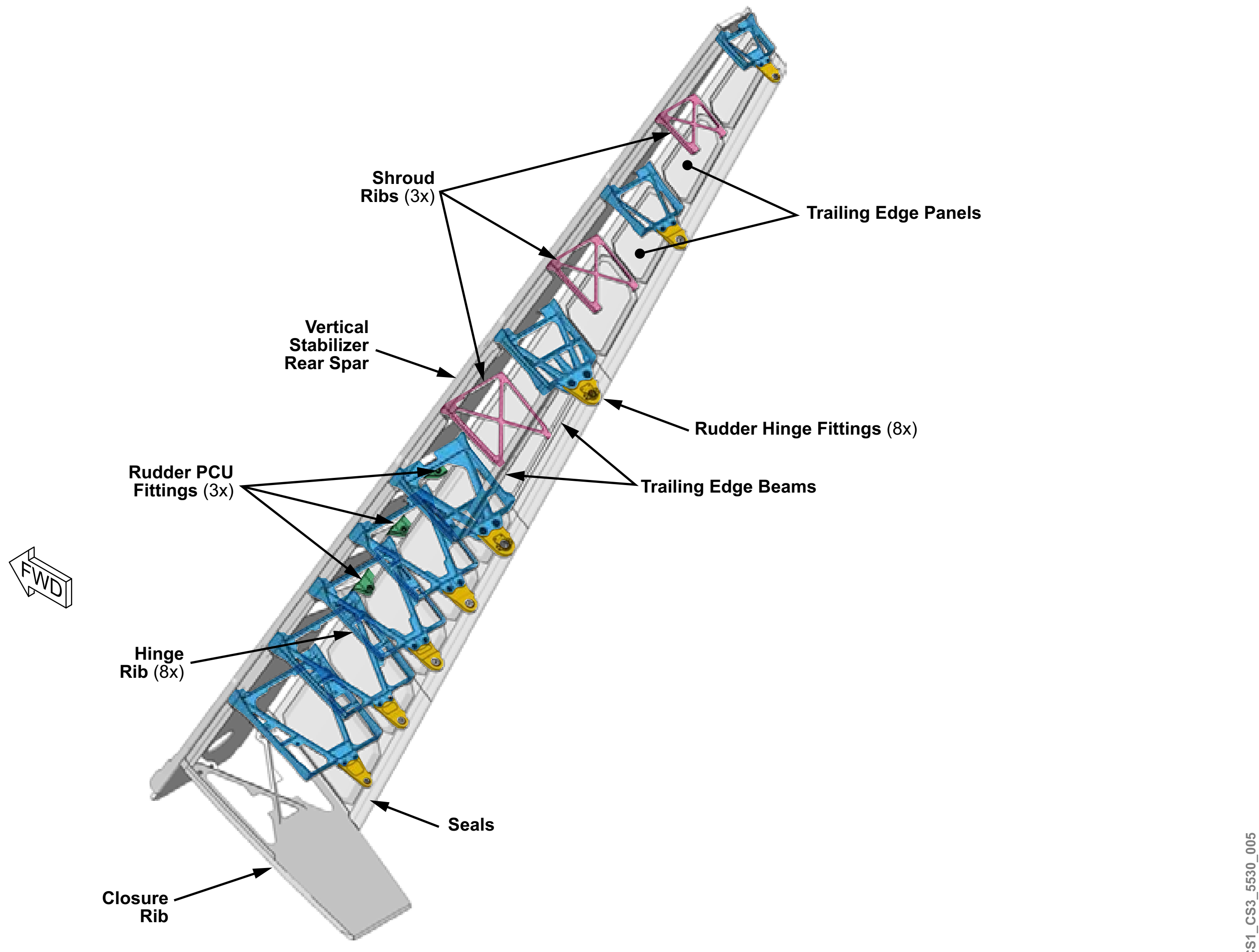


Figure 13: Vertical Stabilizer Trailing Edge Assembly

Tip Assembly

The stabilizer tip assembly is composed of a two-piece GFRP tip panel, joined by a strap assembly. The tip assembly contains a VOR antenna. The tip is mounted to an aluminum rib assembly, as well as a GFRP rib, which supports the VOR antenna mounting bracket.

The tip assembly is installed using standard aircraft screws and nut plates. The tip has a silicone seal around its perimeter to prevent moisture ingress.

The upper trailing edge section of the rudder is fitted with static dischargers to dissipate accumulated static electricity.

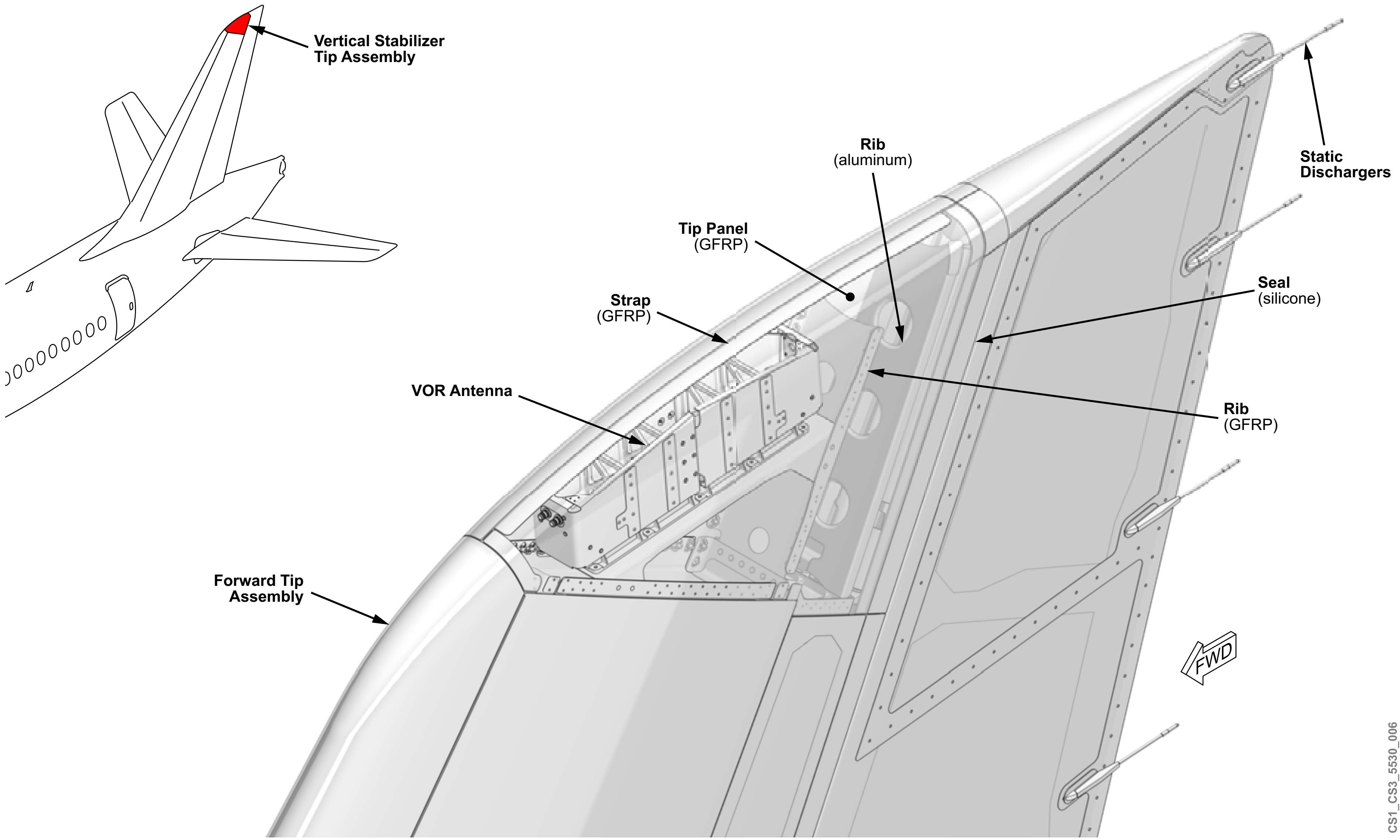


Figure 14: Tip Assembly

55-40 RUDDER ASSEMBLY

GENERAL DESCRIPTION

The rudder is attached to the aft spar of the vertical stabilizer with machined hinge fittings, power control unit (PCU) fittings, and associated hardware.

DETAILED COMPONENT INFORMATION

The rudder is a built-up structure, consisting of the following structural members:

- Left and right skin panels
- Front spar
- Intermediate ribs
- Lower closure rib
- Cast tip assembly

The left and right skins are comprised of a honeycomb core sandwiched between carbon fiber reinforced polymer (CFRP) fabric panels. The spar and ribs are manufactured from CFRP solid laminate.

The front spar has eight machined aluminum alloy hinge fittings, which interface with the vertical stabilizer aft spar fittings. Three machined aluminum alloy rudder PCU attach fittings are also installed on the rudder front spar. The spar also has shroud panel ribs to support the leading edge aerodynamic shroud panels.

The upper rudder portion has a cast aluminum tip, with dual-aluminum lightning protection frames, and five static dischargers.

A rudder seal is installed on the lower closure rib providing aerodynamic sealing between the rudder and tailcone.

Six rudder hoist fittings are also installed on each sides of the control surface for removal and installation purposes using a lifting fixture that is attached to the rudder surface.

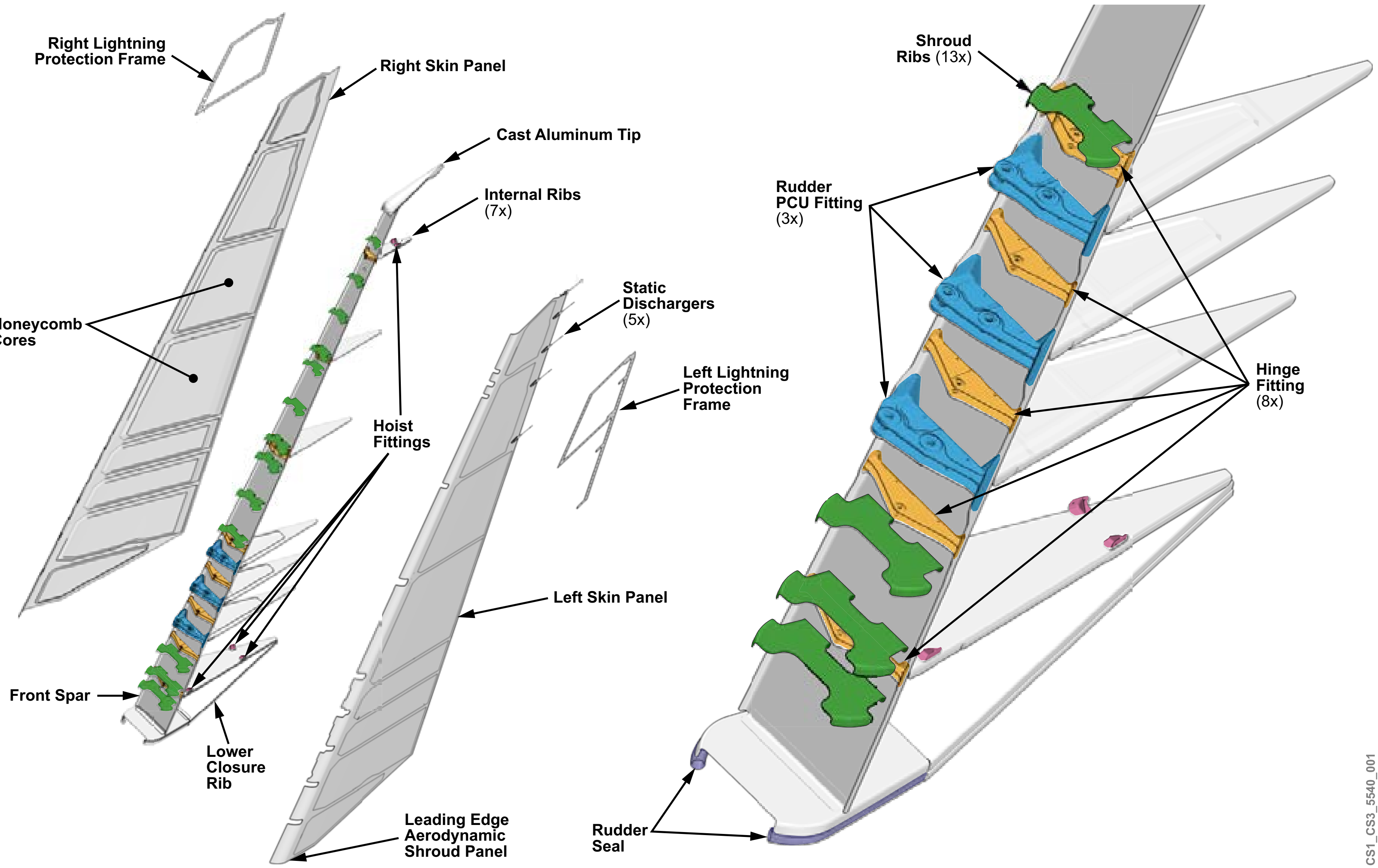


Figure 15: Rudder Assembly

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ATA 56 - Windows

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WINDOWS - CHAPTER BREAKDOWN

56-10 FLIGHT DECK WINDSHIELD

DETAILED COMPONENT INFORMATION

FLIGHT DECK WINDSHIELD

Each windshield is composed of three layers of chemically strengthened glass. An antistatic coating is applied on the outer surface of the windshield. An electric heating film, located between the outer and middle glass layers provides window anti-icing.

The urethane adhesive interlayer between the three glass sections provides maximum adhesion, and increases resistance to delamination. It is strengthened by a fiberglass insert around the perimeter. Moisture ingress is prevented by a polysulfide moisture seal that extends around the outer glass layer, strengthened by a phenolic insert. The seal interfaces with the retainers and the window frame structure.

Both the middle and inner glass plies are structural.

A four-piece silicone gasket is installed between the retainers and the windshield outer perimeter. The gasket seal is completed by four splice retainers at the gasket joints. A four-piece retainer is installed over the gasket and splice retainers. The retainer is secured to the window surround structure by bolts, washers, and nuts. Each windshield also has an anti-icing electrical connector.

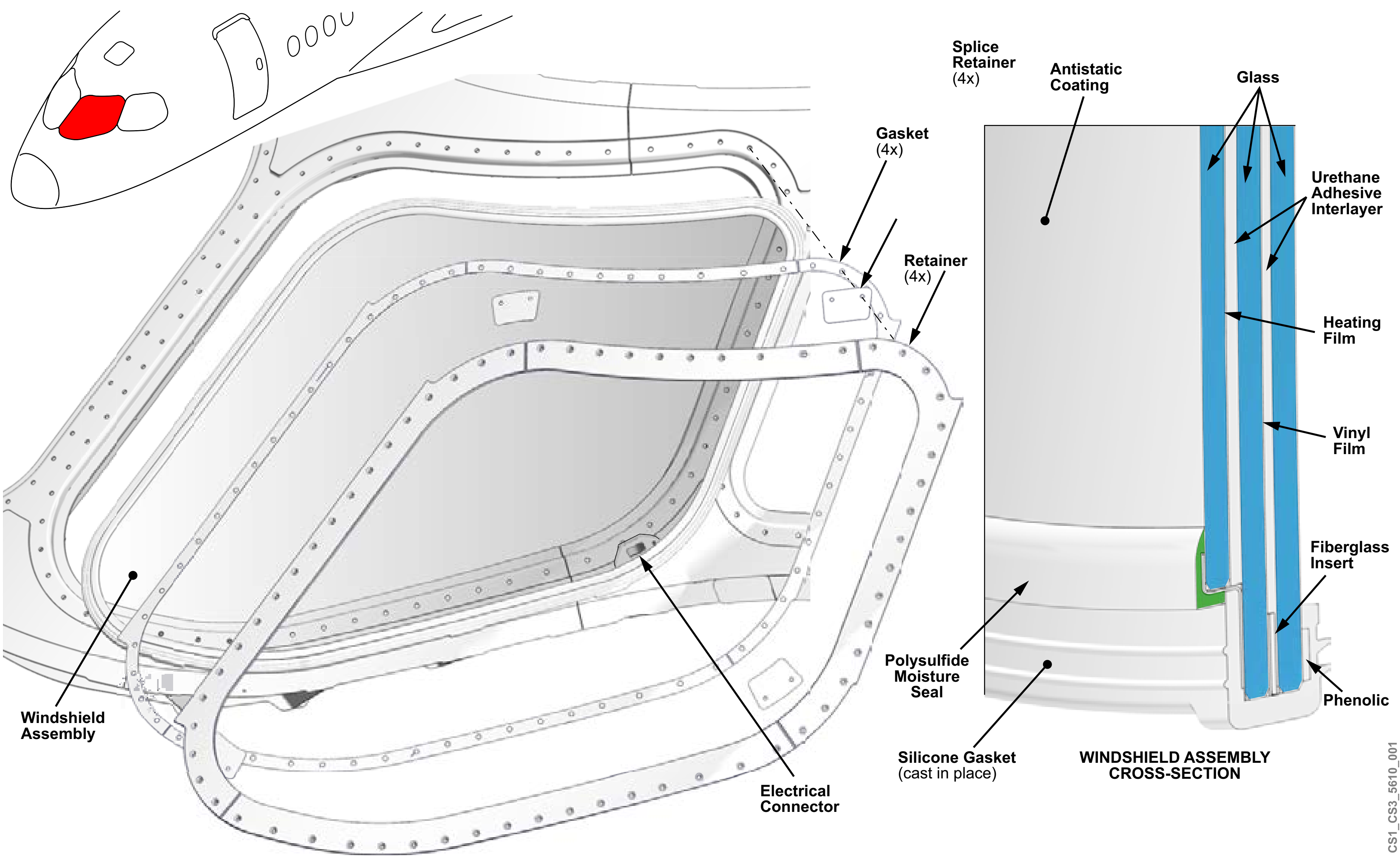


Figure 1: Flight Deck Windshield

56-10 FLIGHT DECK SIDE WINDOW

DETAILED COMPONENT INFORMATION

FLIGHT DECK SIDE WINDOW

Each side window is composed of three layers of chemically strengthened glass. An antistatic coating is applied on the outer surface of the side window. An electric heating film, located between the middle and inner glass layers provides de-fogging for the window.

The side window heating system has a defogging function only.

The urethane adhesive interlayer, between the three glass sections, provides maximum adhesion, and increases resistance to delamination. The inner interlayer is strengthened by a vinyl interlayer around its perimeter. Moisture ingress is prevented by a polysulfide moisture seal that spreads all around the outer glass layer, strengthened by a phenolic insert around its perimeter. The seal interfaces with the retainers and the window frame structure.

The side windows are installed with similar gaskets, splice retainers, retainers, and attaching hardware, as those used for the windshield assemblies.

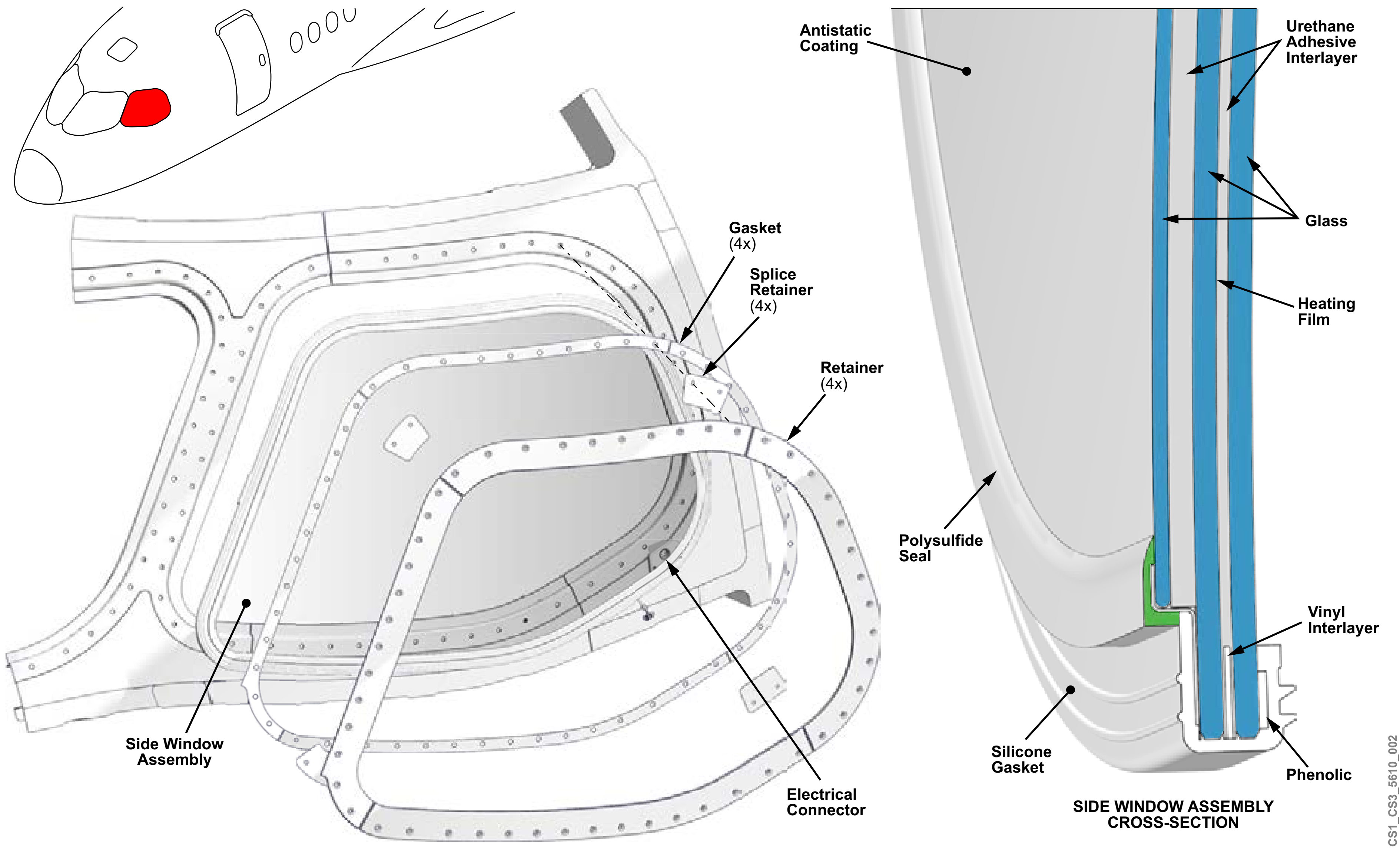


Figure 2: Flight Deck Side Window

56-20 CABIN WINDOWS

DETAILED COMPONENT INFORMATION

CABIN WINDOWS

The window assemblies consist of an inner pane, an outer pane, and a seal. The inner and outer panes are manufactured from acrylic plastic.

The single-molded seal assembly holds the two panes apart, and becomes a pressure seal when the window is installed on the aircraft fuselage window frame.

The outer pane has a beveled edge to fit the window frame. The inner pane has non-beveled edges with a small vent hole near the bottom of the pane.

Each cabin window is installed with eight spring type retaining clips, washers, bolts, and captive nuts. The clips provide retention of the window assembly, and provide a centering function to assist in properly mounting the window.

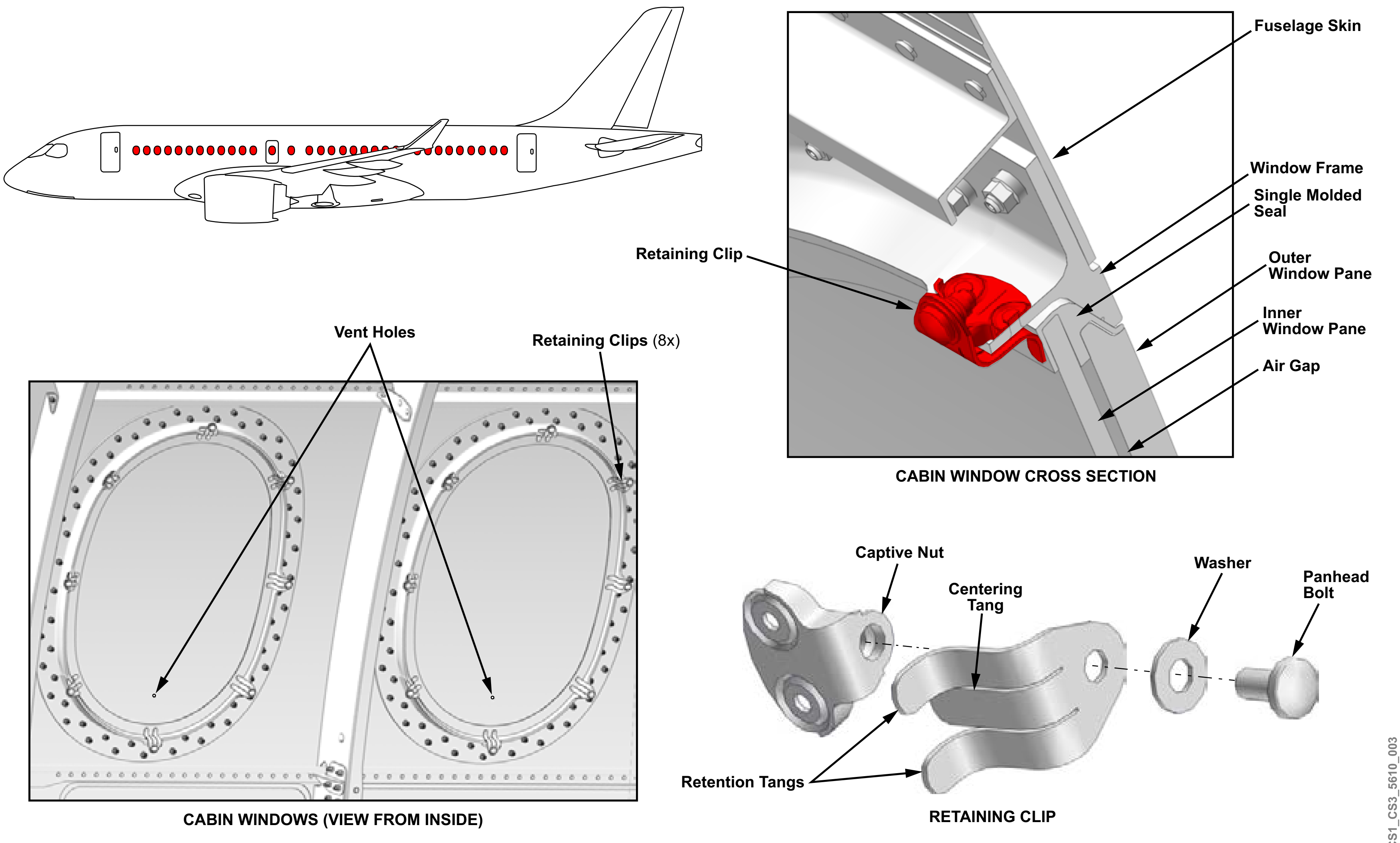


Figure 3: Cabin Windows

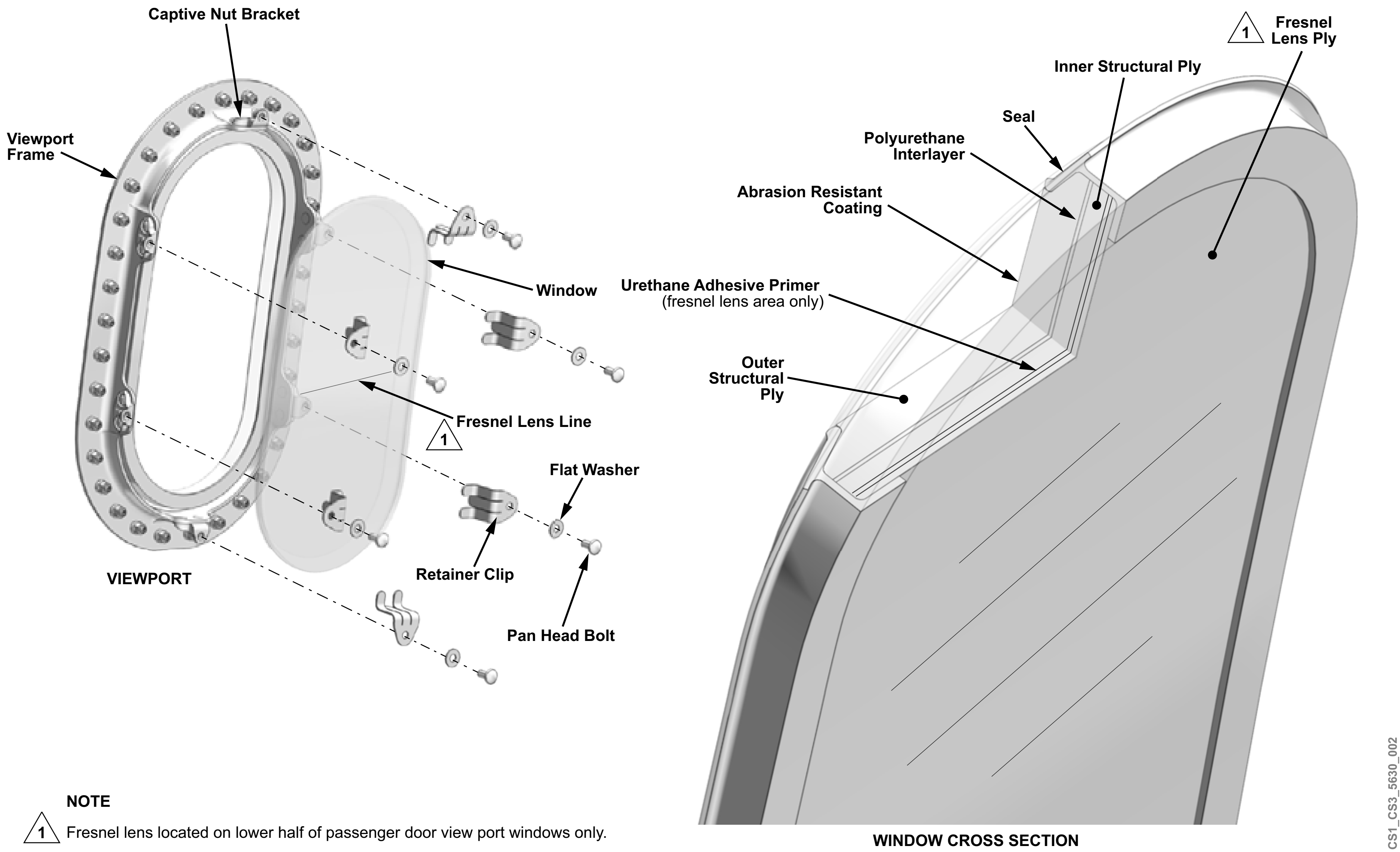
56-30 PASSENGER AND SERVICE DOOR VIEWPORTS

DETAILED COMPONENT INFORMATION

PASSENGER AND SERVICE DOOR VIEWPORTS

Each door viewport window consists of two plies of stretched acrylic structural laminate. The passenger door viewports have a Fresnel lens bonded to the inner surface of the inner structural ply. The Fresnel lens covers the lower half of the inner surface of the viewport, and provides a good view of the lower areas outside of the aircraft.

The outer surface of the structural ply is coated with an abrasion resistant coating. The door viewport window is sealed against the door structure with a rubber seal, and is held in place by six spring-type retainer clips, pan head screws, washers, and captive nuts.



NOTE

⚠1 Fresnel lens located on lower half of passenger door view port windows only.

Figure 4: Passenger and Service Door Viewports

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ATA 57 - Wings

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WINGS - CHAPTER BREAKDOWN

57-10 CENTER WING BOX

GENERAL DESCRIPTION

The carbon fiber reinforced polymer (CFRP) center wing box (CWB) is attached to fuselage FR 40 at the front spar, and at fuselage FR 48 at its rear spar. The CWB is connected to the left and right wings at the left and right RIB 3 locations. These ribs form the two side sections of the center fuel tank.

The upper and lower CFRP skins complete the center wing box enclosure. The center wing is completely sealed for fuel storage. Four fuel tank access panels are installed in the lower skin panel.

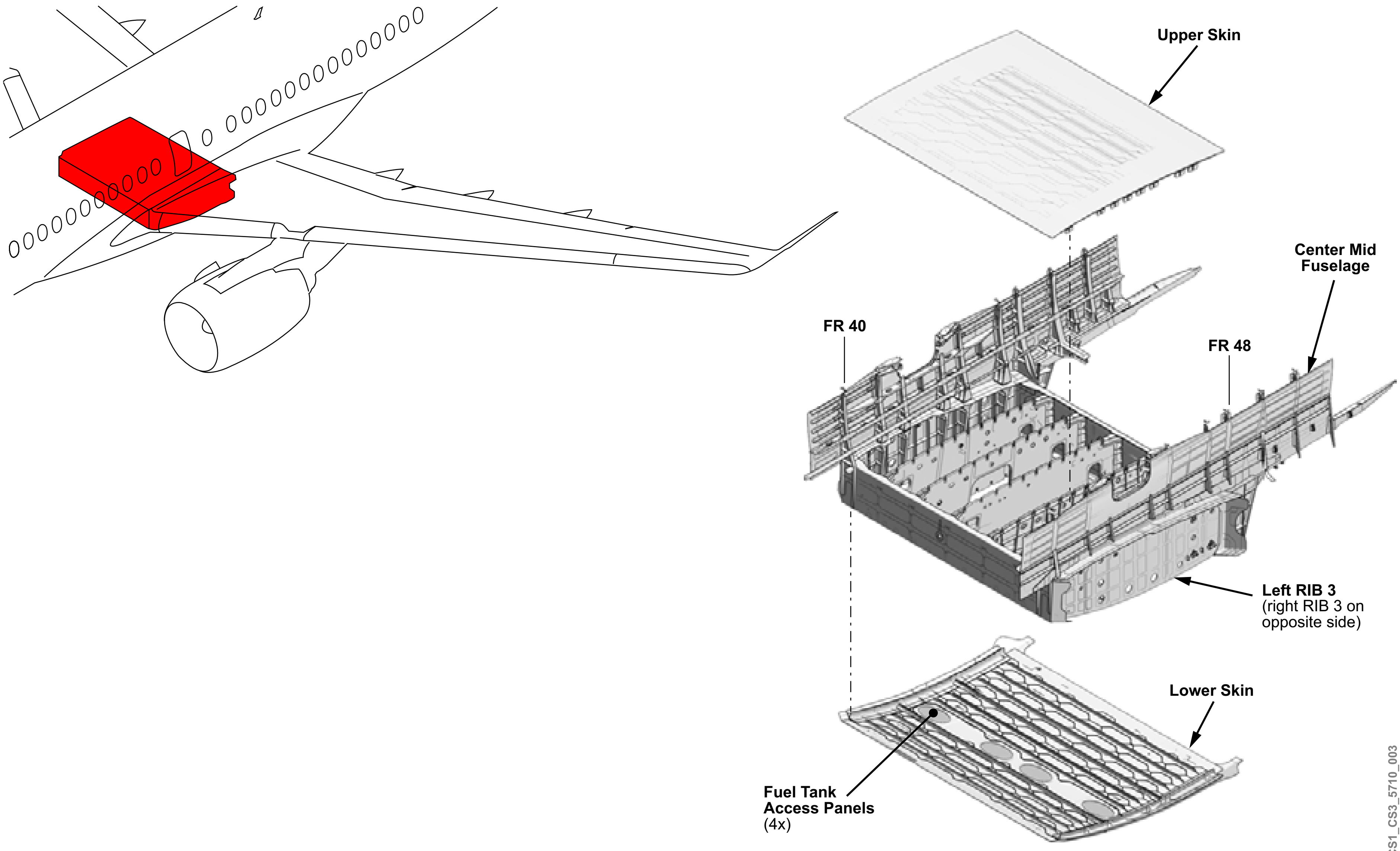


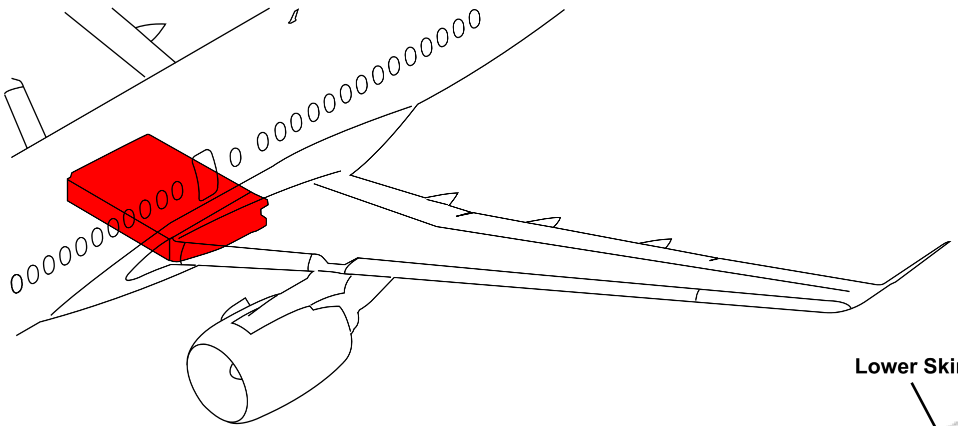
Figure 1: Center Wing Box

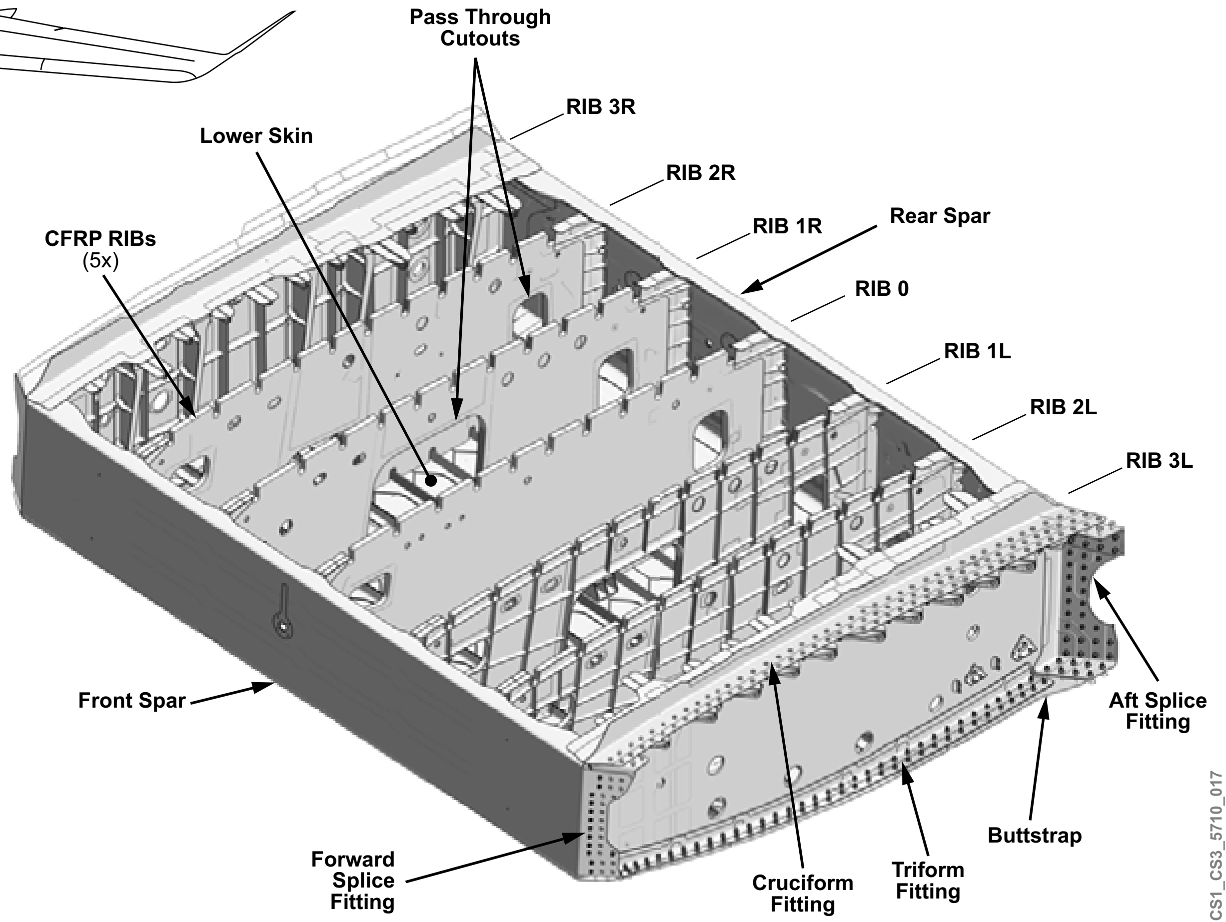
DETAILED COMPONENT INFORMATION

CENTER WING BOX

The center wing box is the interface structure between the left and right outer wings. It consists of upper and lower CFRP skins, five CFRP ribs, and front and rear spars. The ribs and spars have various cutouts for aircraft system plumbing and wiring pass through.

The center box has upper cruciform fittings, lower triform fittings, and buttstraps at RIB 3 left and right, for attachment of the outer wings. The upper cruciform flanges attach to the fuselage structure. The front and rear spars have splice fittings, which attach to the outer wing front and rear spars.





NOTE

Upper skin removed for clarity.

Figure 2: Center Wing Box

CENTER WING BOX TO FUSELAGE ATTACHMENTS

The center wing attaches to the fuselage center mid section, between fuselage frames 40 and 48.

Frame extension fittings are used to attach the center wing box. The fittings at fuselage FR 40 attach to the center wing front spar, and fittings at FR 48 attach to the center wing rear spar.

Additional fuselage frame attachments are located between frames 40 and 48 to provide complete additional strength to the wing/fuselage interface.

Cruciform fittings are attached to the center wing at RIB 3 left and right. The upper vertical segment of the cruciform attaches the center wing box to the fuselage structure.

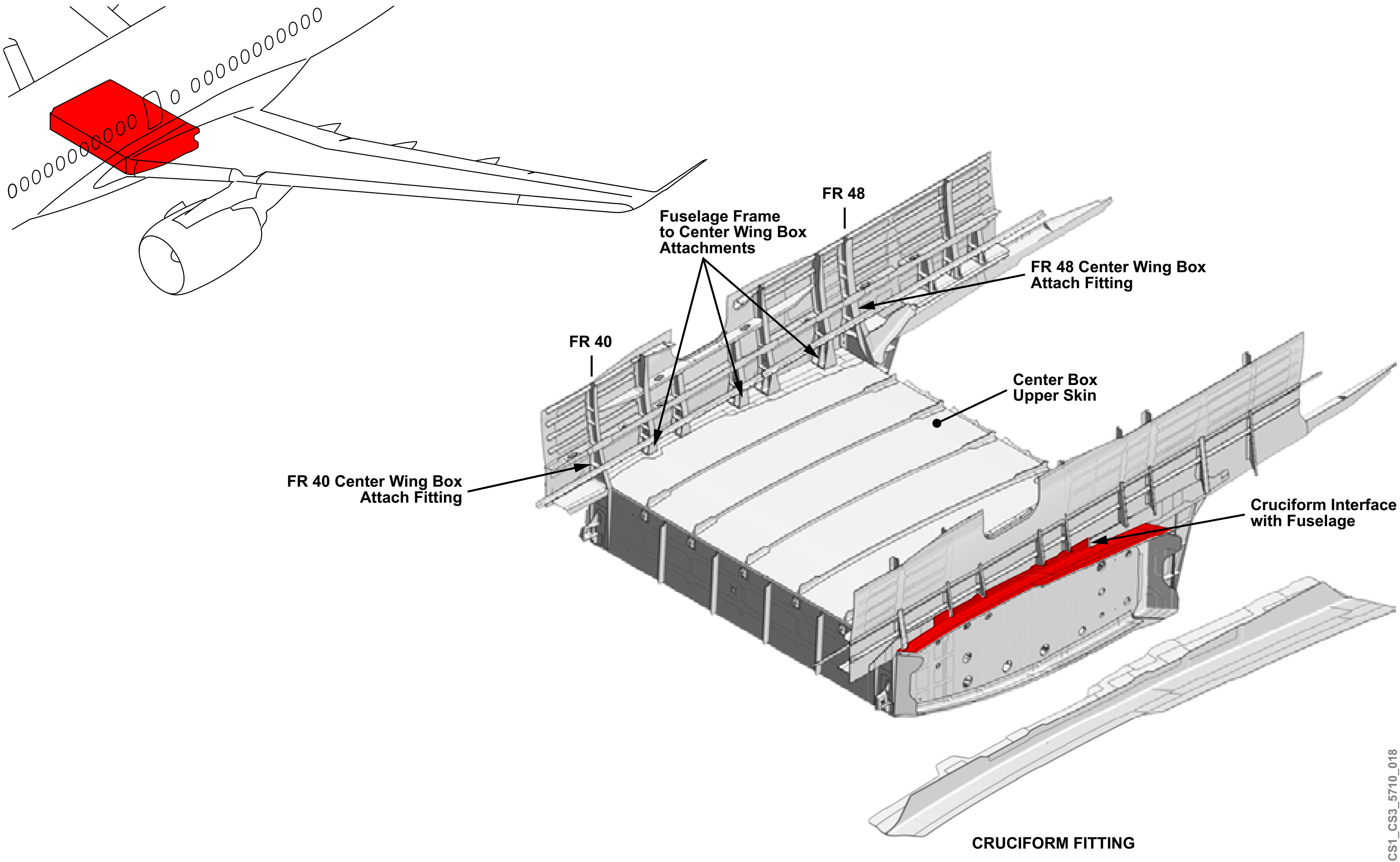


Figure 3: Center Wing Box to Fuselage Attachments

CENTER WING BOX TO OUTER WING ATTACHMENTS

The outer wings are attached to the center wing box at RIB 3, left and right.

Numerous types of fittings are used for the attachment. Large splice fittings are installed at the front and rear spars of the center box. The fittings connect directly to the outer wing spars.

Between the front and rear splices, paddle fittings are installed inside the outer wing structure at each outer wing stringer, and attach directly to RIB 3 of the center box.

The lower skins of the center and outer wings are connected using an internal triform fitting, and an external butt strap for added strength.

The upper skins are connected using a cruciform fitting that interfaces with the center and outer wing, as well as the fuselage structure.

The upper and lower center wing box skins have support brackets and fittings for the environmental control system ducting, piping, wiring, and components.

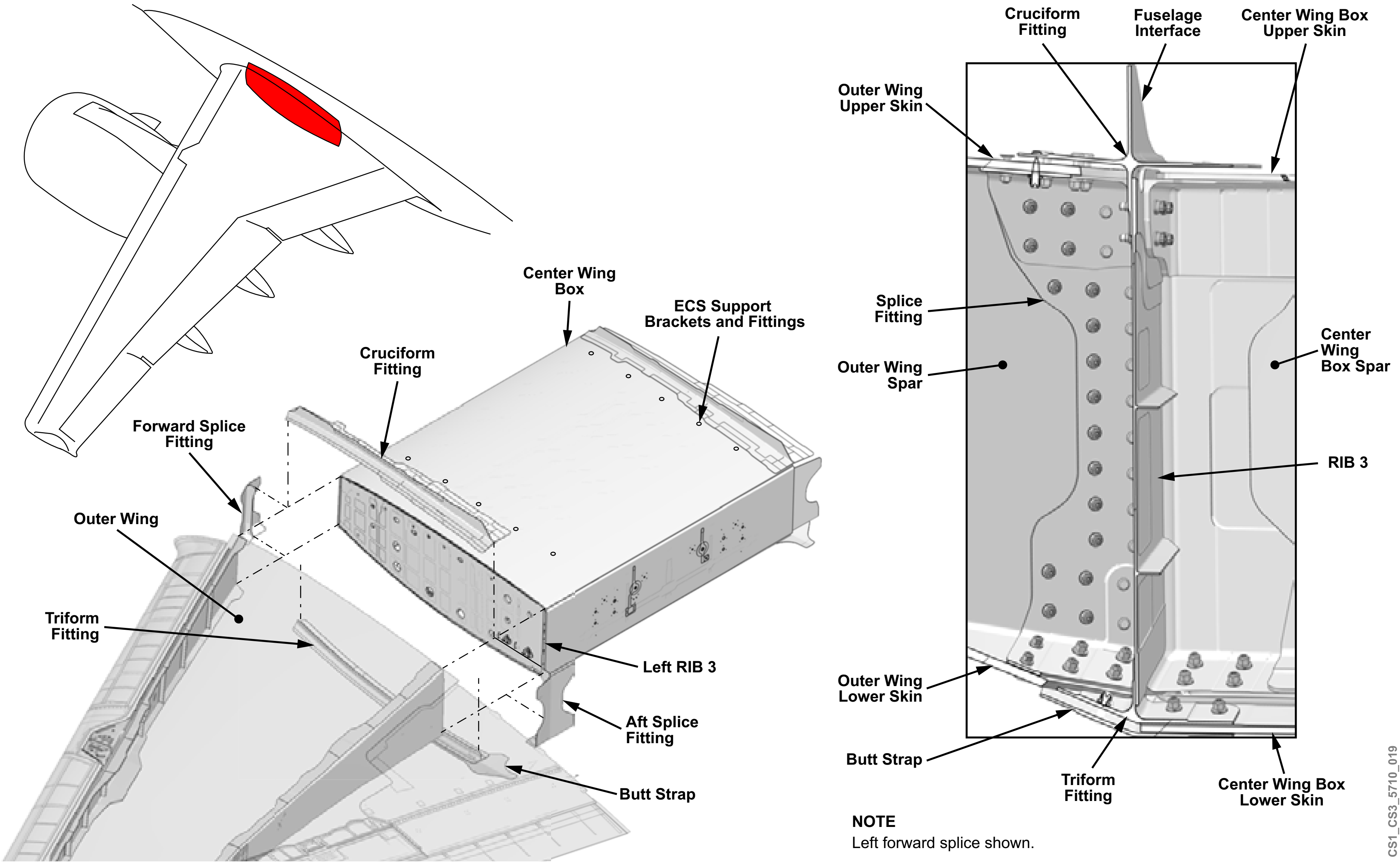


Figure 4: Center Wing Box to Outer Wing Attachments

57-20 OUTER WINGS

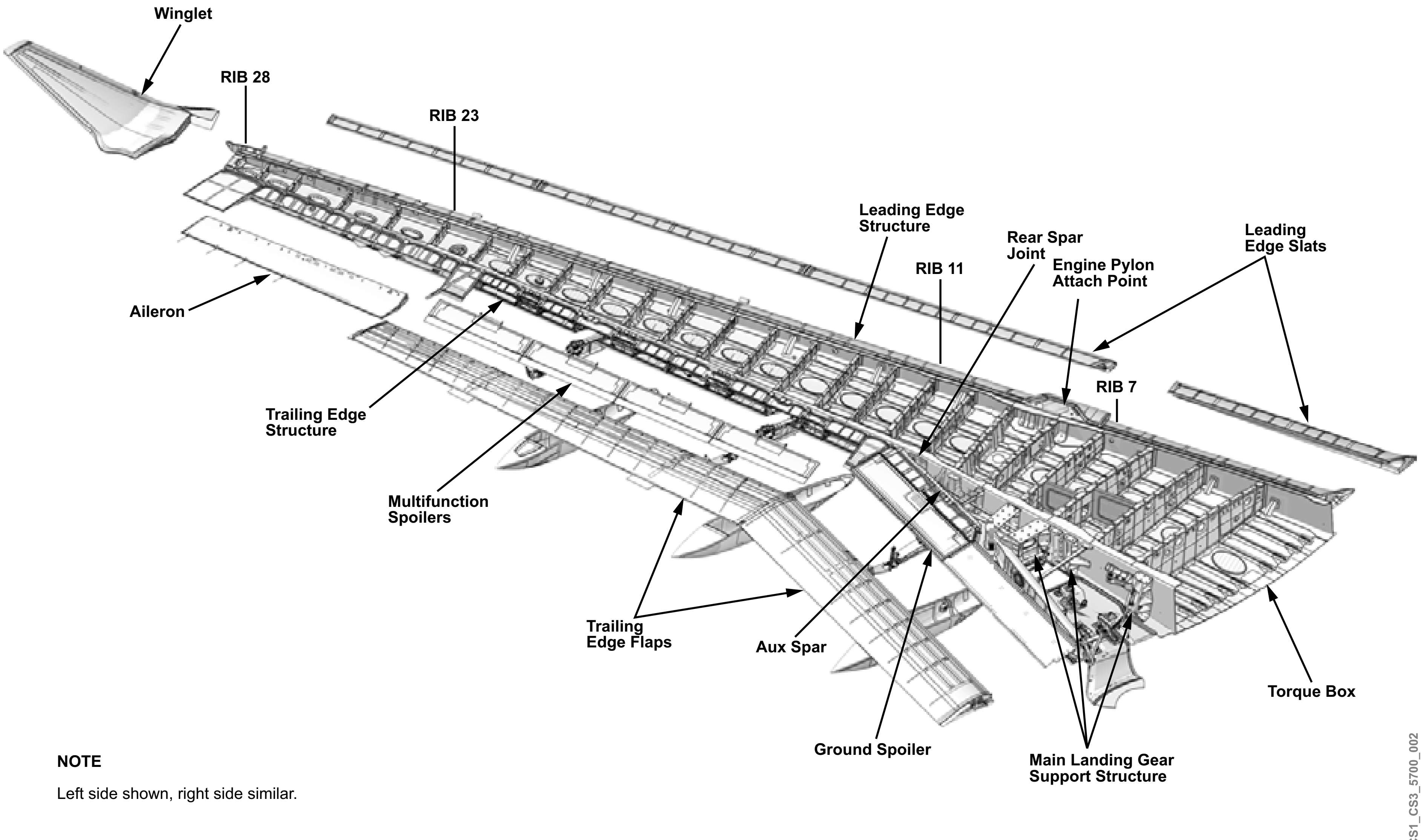
GENERAL DESCRIPTION

Both wings extend from their joints at the center wing box RIB 3 to their respective wing RIB 28 at the winglets. An auxiliary spar is located between RIB 7 and the rear spar joint area.

The integral left and right fuel tanks are located at the boundary of the center fuel tank, and extend to wing RIB 23.

The outer wing structure includes leading and trailing edge structures. Various aircraft system ducting, plumbing, components, and wiring are located at the leading edge and trailing edge of each wing.

Secondary structures on the wing provide support for the slats, flaps, spoilers, and ailerons. The outer wings have attachment points for the engine pylons and the main landing gear (MLG). A winglet is installed at RIB 28.



NOTE

Left side shown, right side similar.

Figure 5: Outer Wings

DETAILED COMPONENT INFORMATION

OUTER WING TORQUE BOX

The primary structure of the outer wing is the torque box assembly. The torque box is the main load bearing member for flight loads on the aircraft.

The torque box consists of upper and lower carbon fiber reinforced polymer (CFRP) skins, CFRP front and rear spars, and aluminum alloy ribs.

Aluminum alloy fittings are installed on the rear spar to support the ailerons, and multifunction spoilers. A wing jack fitting is located on the rear spar between the aux spar attach ribs.

Titanium fittings are mounted on the lower surface of the wing to provide attachment of the engine pylon. Aluminum alloy fittings are mounted on the lower surface to provide attachment and support for the three flap track beam assemblies.

An auxiliary spar is attached to the rear spar through two fixed trailing edge ribs and spans from wing RIB 7 area to the outboard of rear spar joint. Aluminum alloy fittings are mounted on the auxiliary spar for the attachment of the ground spoiler. Support structures for the main landing gear (MLG) are located between the auxiliary and rear spars.

The torque box structure is completely sealed for fuel storage. The integral fuel tank extends from the rib 3 attachment point to RIB 23. The area between RIB 23 and RIB 28 is a dry area. The forward area between RIB 6 and RIB 7 is a dry bay. CFRP access panels are located along the wing lower surface, to allow for fuel tank and inner wing access.

Access to area between wing RIB 3 and RIB 4 is provided through an access panel on rib 4 forward end.

Two collector tank access panels are located between RIB 5 and 6, and RIB 6 and 7 allowing access to the inside of the collector tank.

Wing RIB 28 is aluminum alloy machined and is located between the upper and lower wing skin panels, the front and rear spars, and is the attachment for the winglet assembly.

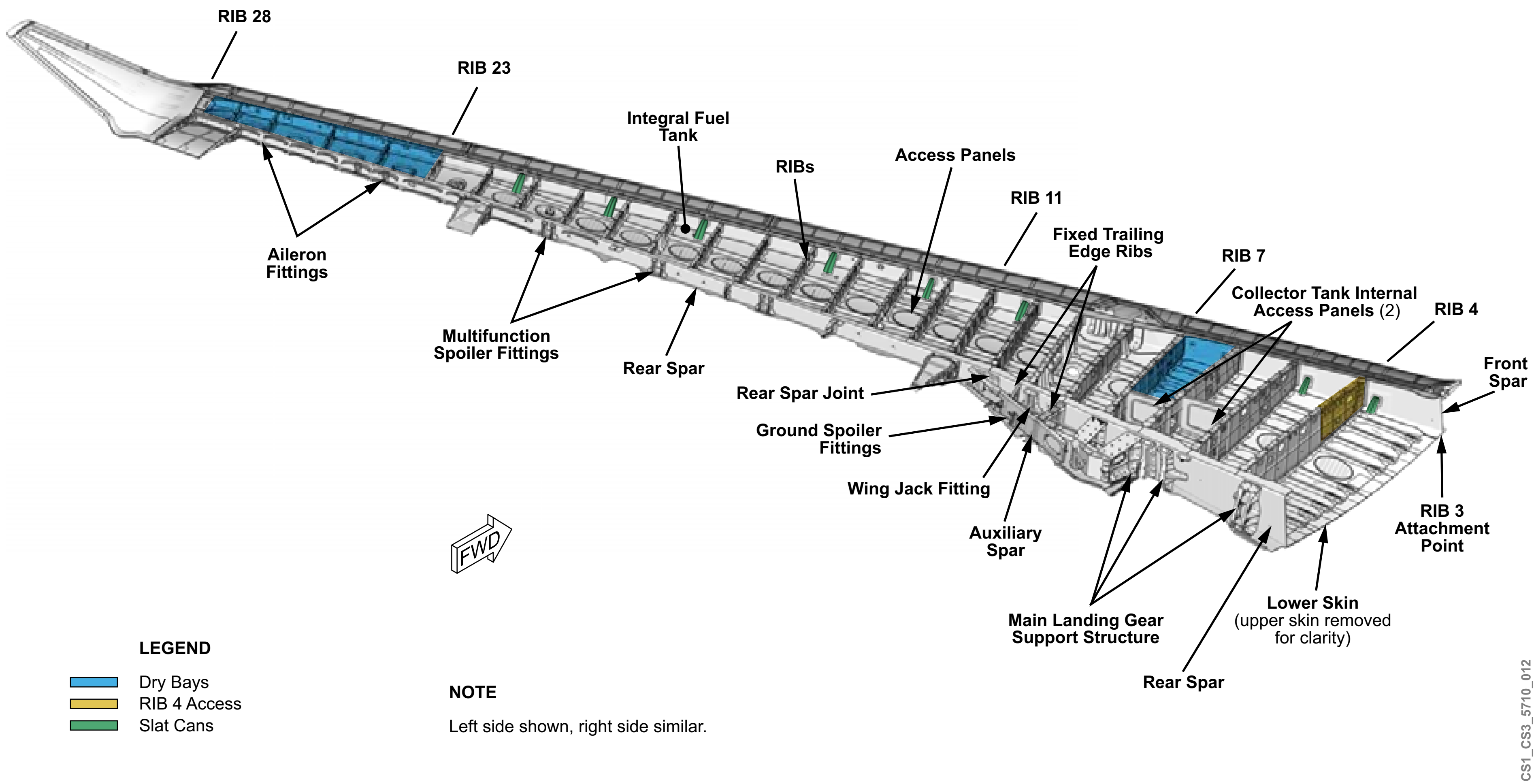


Figure 6: Outer Wing Torque Box

LEADING EDGE STRUCTURE

The fixed leading edge is installed on the wing front spar. The external aerodynamic profile is given by the assembly of metallic aluminum alloy D-nose skins.

The bottom of the fixed leading edge has aluminum alloy lower access panels to allow maintenance access to systems and components, (electrical, hydraulic, and anti-icing systems) installed in the leading edge structure. Dedicated upper access panels give maintenance access to specific systems.

The internal metallic structure consists of the following chord wise, and span wise structural elements:

- Machined ribs, and sheet metal ribs (aluminum alloy)
- Upper and lower butt straps (aluminum alloy), which allow connection between D-nose skins, wing upper panels, and upper and lower access panels
- U-channels (aluminum alloy), which allow connection between D-nose skins and access panels

The fixed leading edge also contain many of the leading edge slat transmission components. Some of the machined ribs support the slat drive rotary geared actuators, the slat drive tracks, and associated components.

The forward face of the front spar is protected from the hot wing anti-ice and the ECS bleed air ducts by thermal insulation blankets. The blankets protect the area from wing RIB 3 to 12. Blankets are held in place using Velcro strips and retainer arrangements.

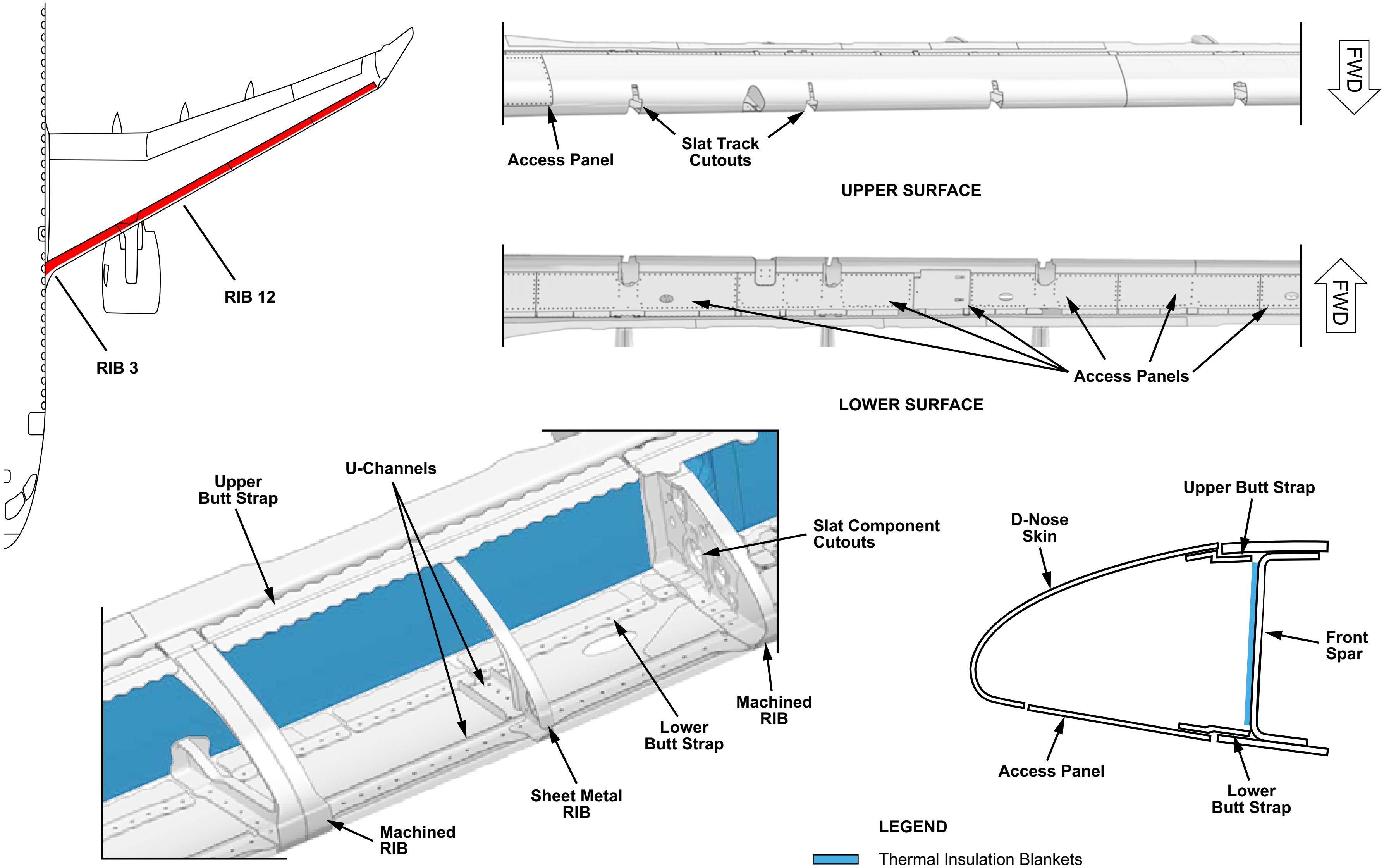


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Figure 7: Leading Edge Structure

TRAILING EDGE STRUCTURE

The fixed trailing edge extends from RIB 3 (where the outer wing joins to the aircraft center wing section), to the machined RIB 28 at the winglet.

The fixed trailing edge structure consists of the following primary structural members:

- The winglet, outboard, and mid false work assemblies
- The aileron, multifunction spoiler (MFS), ground spoiler, and MLG shroud panels
- Inboard and outboard flap lower shroud panels
- The auxiliary spar

All of the false work structures, shrouds, and support fittings are aluminum alloy, with the exception of the three MLG shroud panels. The three MLG shroud panels are CFRP, honeycomb reinforced sandwich structures.

All shroud panels are removable to permit access for maintenance. They are also fitted with a structural seal at their aft edge, preventing wear and tear to the structure caused by the movement of the flaps and ailerons.

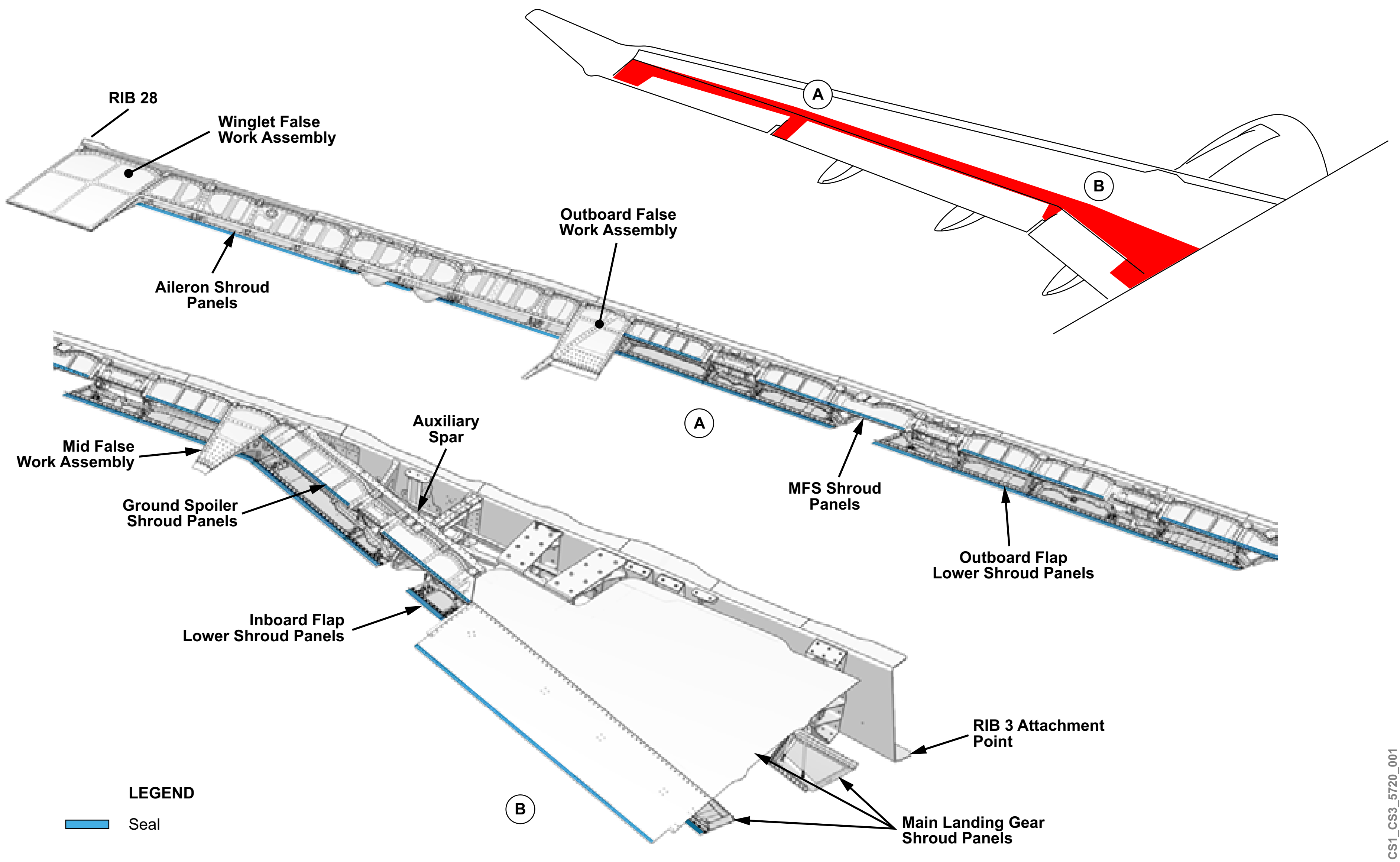


Figure 8: Trailing Edge Structure

PYLON ATTACHMENT FITTINGS

The pylon attachment fittings are located between the left and right wing front and rear spars, at RIBS 8 and 9, on the wing lower surface. All of the fittings are made from titanium plate.

The forward thrust link fitting is installed on the front spar within the wing leading edge skin. This fitting interfaces with the pylon torque box upper thrust link. Two additional thrust link fittings interface with the inboard and outboard pylon torque box thrust fittings.

Between these two fittings, a lateral load fitting is installed, which interfaces with the pylon torque box inboard and outboard lateral links.

A drag brace fitting is located between RIBS 8 and 9, aft of the thrust link fittings. This fitting interfaces with the pylon torque box diagonal brace.

Pylon attachment fittings are bolted to the wing lower surface, and structurally supported from the inside of the wing box using backup fittings. Access to these internal fittings is through two access panels on wing RIB 8.

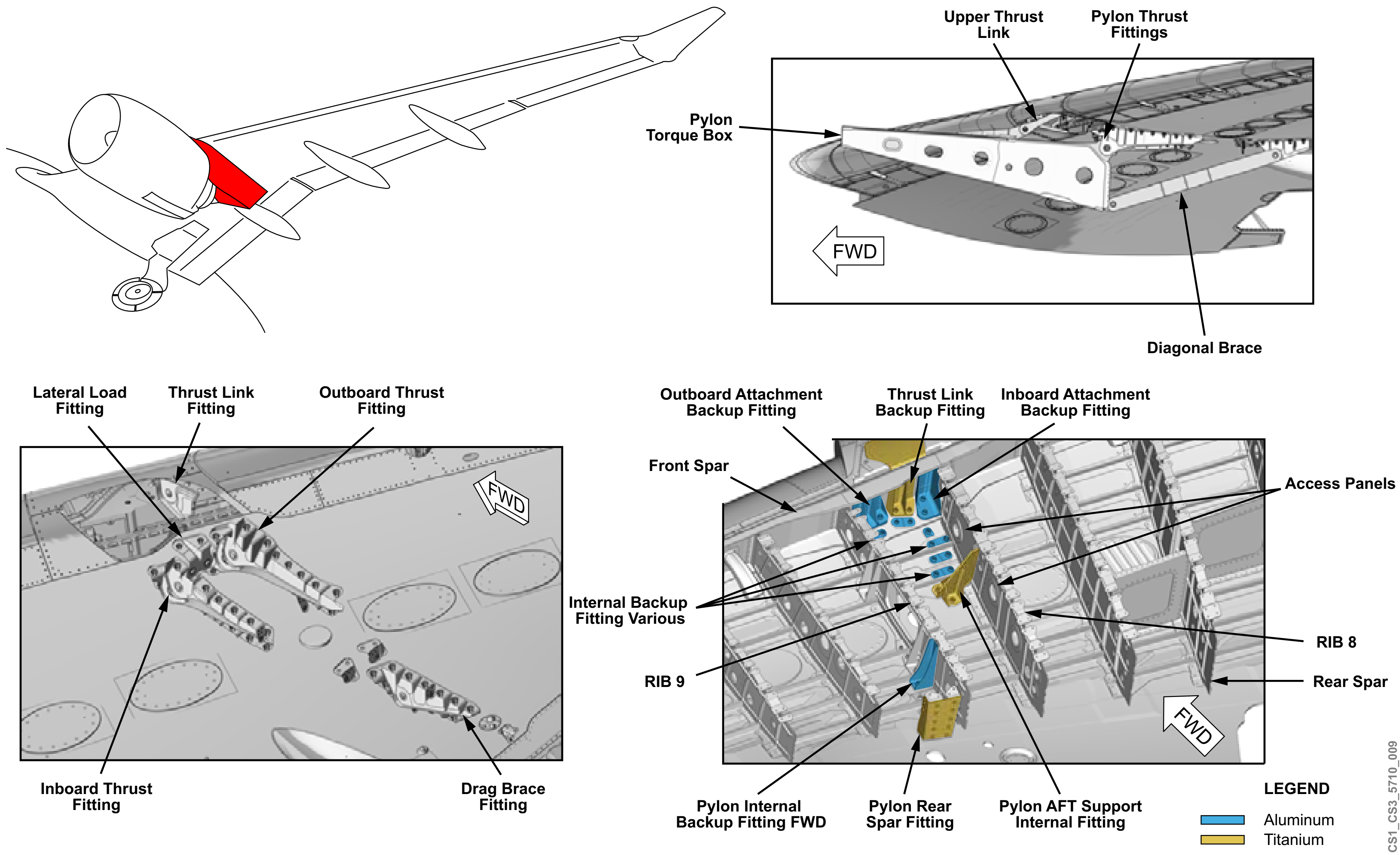


Figure 9: Pylon Attachment Fittings

MAIN LANDING GEAR SUPPORT ATTACHMENTS

The main landing gear (MLG) support structures are located between wing RIB 3 and RIB 7 on each outer wing. The support components are installed between the wing auxiliary spar, and the rear spar.

Two MLG support ribs are installed on the rear spar at RIBS 6 and 7. A lower stabilizer is located inboard of the RIB 6 support rib, and an upper stabilizer is installed at RIB 5.

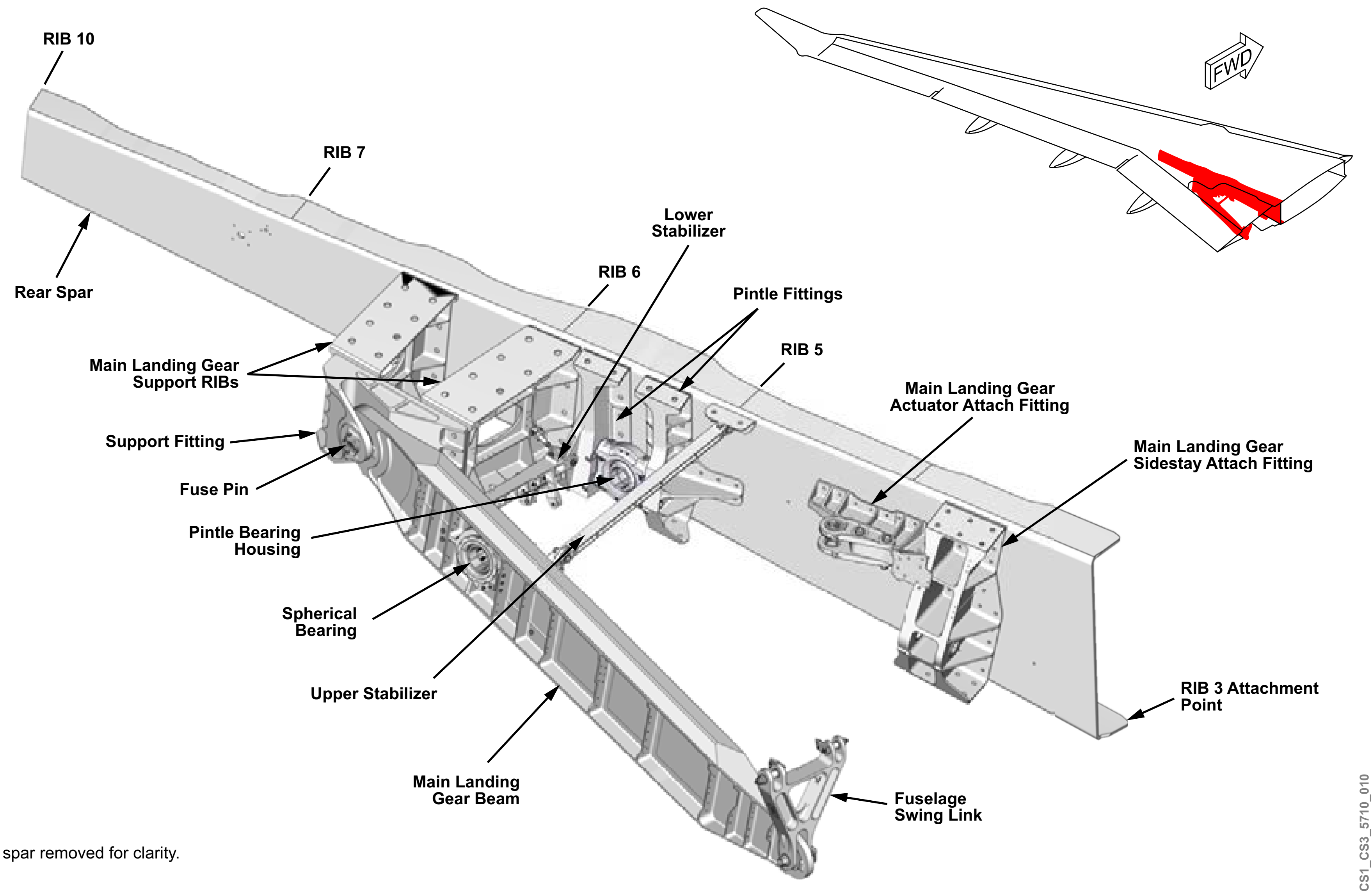
A support fitting is attached to the rear face of the MLG support ribs. This fitting is the outboard attachment point for the MLG beam. A fuse pin is used to attach the MLG beam to its support fitting. Its purpose is to shear in the event of severe overstressing of the MLG, allowing it to shear before extensive wing damage occurs.

The MLG beam is attached to a fuselage mounted swing link at the inboard end. The swing link allows limited lateral movement of the MLG beam to absorb takeoff, landing, and taxi shocks.

The MLG beam has a spherical bearing installed, which is the connection point for the MLG shock strut aft trunnion.

Two pintle fittings are installed on the rear spar, at RIBS 5 and 6, to which a pintle bearing housing is bolted to support the forward shock strut trunnion fitting.

The MLG actuator fittings, and sidestay fitting are installed inboard of the pintle fittings on the rear spar.



NOTE

Auxiliary spar removed for clarity.

Figure 10: Main Landing Gear Support Attachments

FLAP TRACK BEAM ATTACHMENTS

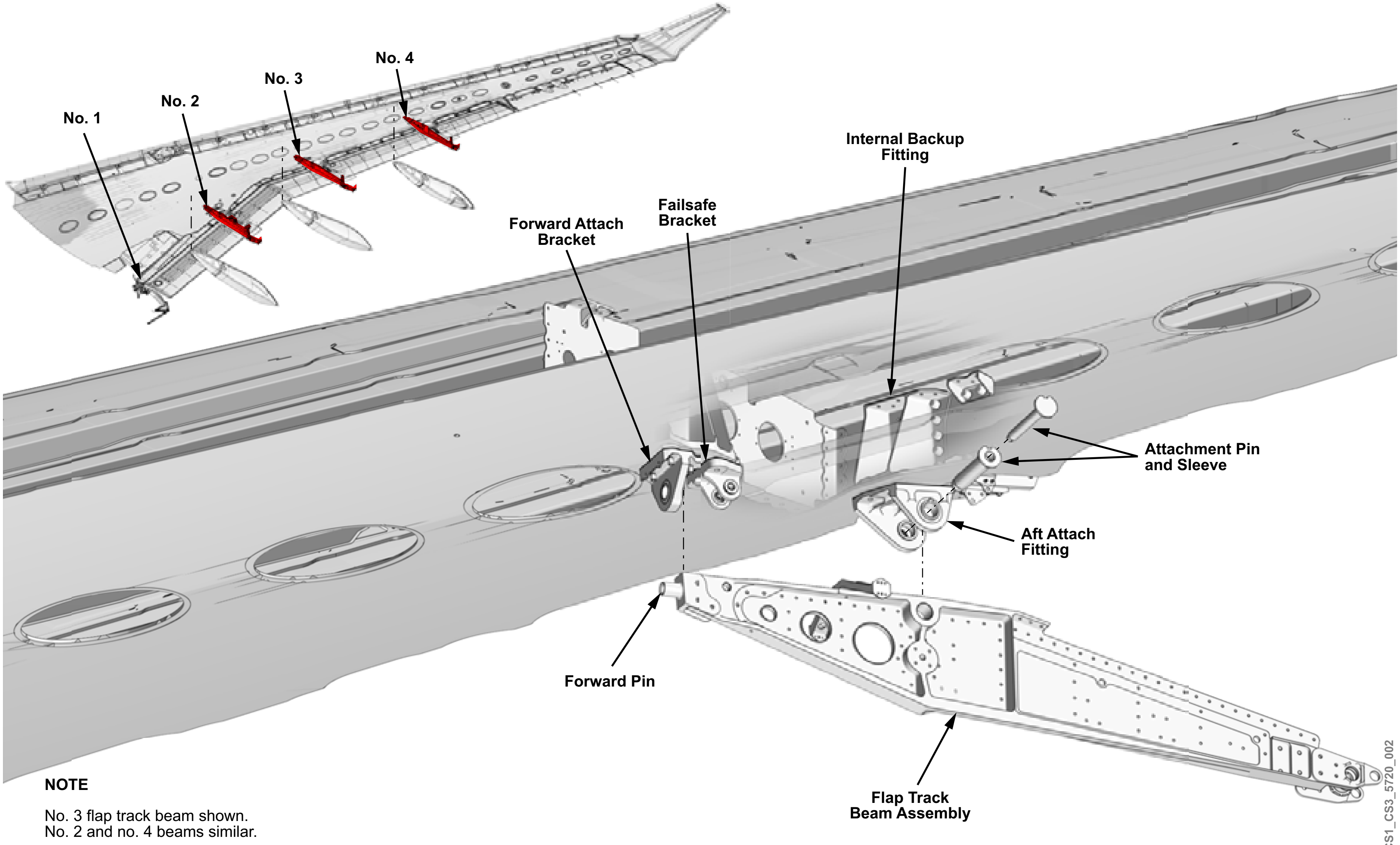
Each flap is supported by one master flap track beam, and one slave flap track beams. Flap track beams no. 2 and no. 3 are the master flap track beams. Beams no. 1 and no. 4 are the slave flap track beams.

Inboard flap track beam no. 1 is mounted to the mid fuselage structure.

Flap track beams no. 2, no. 3, and no. 4 lower wing attachment brackets and fittings are made of aluminum alloy.

Flap track beam no. 2 is attached to the wing by a forward attach bracket, which is installed on the wing lower surface. The forward attach bracket has a spherical bearing, which supports the forward pin on the track beam. Behind the forward attach bracket is a failsafe bracket which straddles the track beam, and accepts a bolt which passes through the track beam. The track beam aft attach fitting is installed on the auxiliary spar, it straddles the track beam, and accepts a bolt. The aft attach fitting is designed to limit lateral movement. Flap track beam no. 3 is attached in a similar manner, however the aft attach fitting is mounted directly to the wing rear spar.

Flap track beam no. 4 mounts in a similar manner, however as no. 4 is a slave track beam, the fittings are not as robust as those used for no. 3 and no. 2 track beams. The aft attach fitting for no. 4 track beam is mounted directly to the rear spar. In normal operating conditions, slave flap track beams do not take lateral loads.



NOTE

No. 3 flap track beam shown.
No. 2 and no. 4 beams similar.

Figure 11: Flap Track Beam Attachments

57-30 WINGLETS

GENERAL DESCRIPTION

A carbon fiber reinforced polymer (CFRP) blended winglet assembly is attached to each outer wing at RIB 28. The purpose of the winglet is to decrease drag by partially recovering the energy from the wing tip vortices.

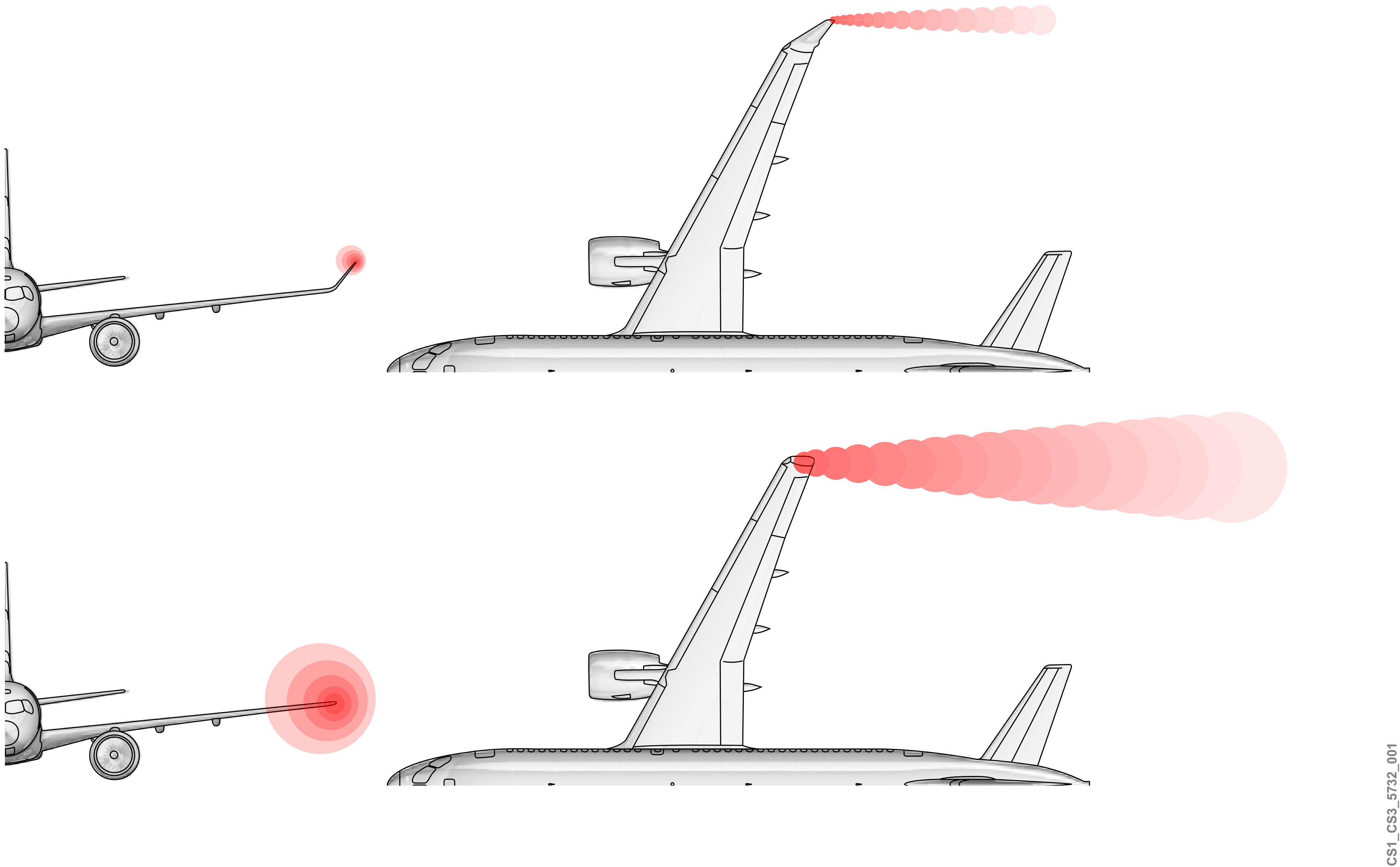


Figure 12: Winglets

DETAILED COMPONENT INFORMATION

The winglet consists of front, mid, and rear CFRP spars as well as two aluminum alloy ribs, located between the upper and the lower honeycomb reinforced CFRP skin panels.

The lower aluminum rib is an integral part of the root fitting, which is the interface with the wing outer RIB 28. A two-piece aluminum removable leading edge, an aluminum removable tip cap, and an aluminum trailing edge complete the winglet structure.

Four static dischargers are mounted to the upper trailing edge.

On the wing tip leading edge, a wing tip nav light cuff fairing overlaps wing RIB 28 and winglet. The cuff is screwed on the leading edge and has a quick removable lens. Attached to the winglets are two seals that assure aerodynamic airflow with the mating surface and the winglet false work assembly.

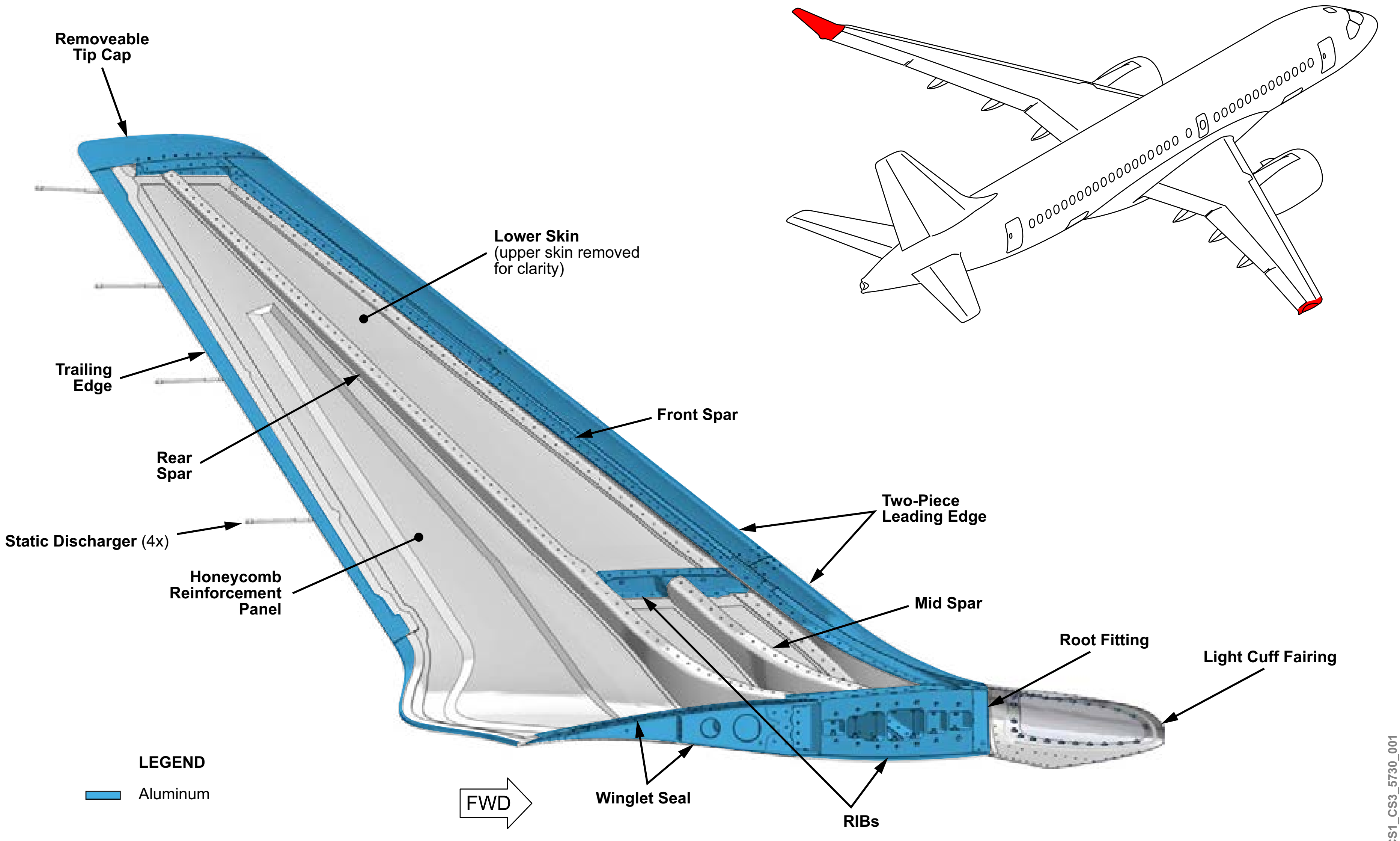


Figure 13: Winglet Construction

57-40 LEADING EDGES DEVICES - SLATS

GENERAL DESCRIPTION

The slats are the moveable parts of the left and right wing leading edge. Four slats are mounted on each leading edge. Slat 1 is mounted inboard of the engine pylon, slats 2, 3, and 4 are mounted outboard of the pylon. The slat structures are very similar, except for minor differences in attachment fittings, and dimensions.

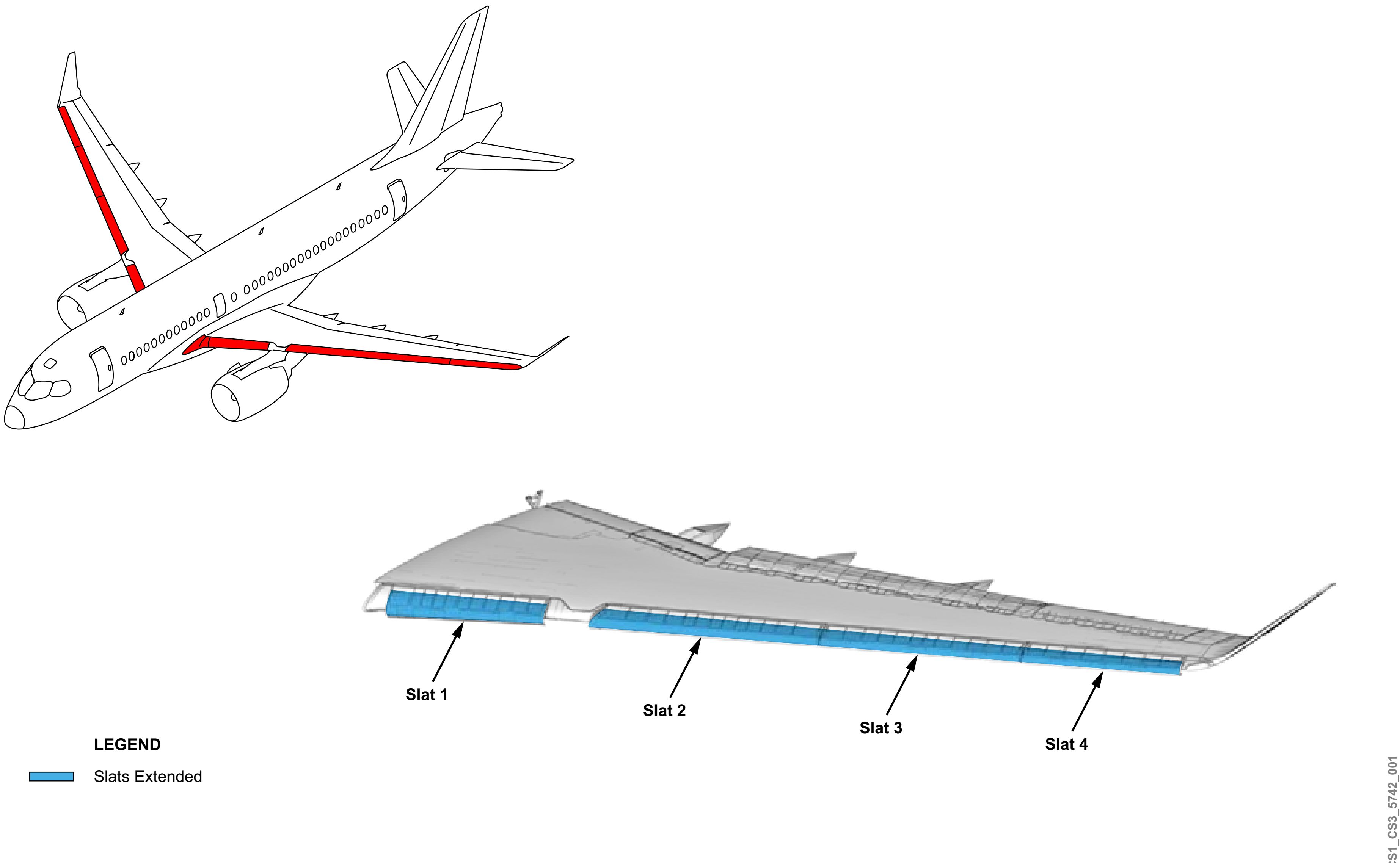


Figure 14: Slats

DETAILED COMPONENT INFORMATION

The external aerodynamic profile is provided by the assembly of aluminum alloy skins in the nose and bottom areas and an aluminum alloy trailing edge. The metallic structure is composed of chord wise ribs and span wise girders.

The end closing ribs and track ribs are machined, and the intermediate ribs are extruded sheet metal.

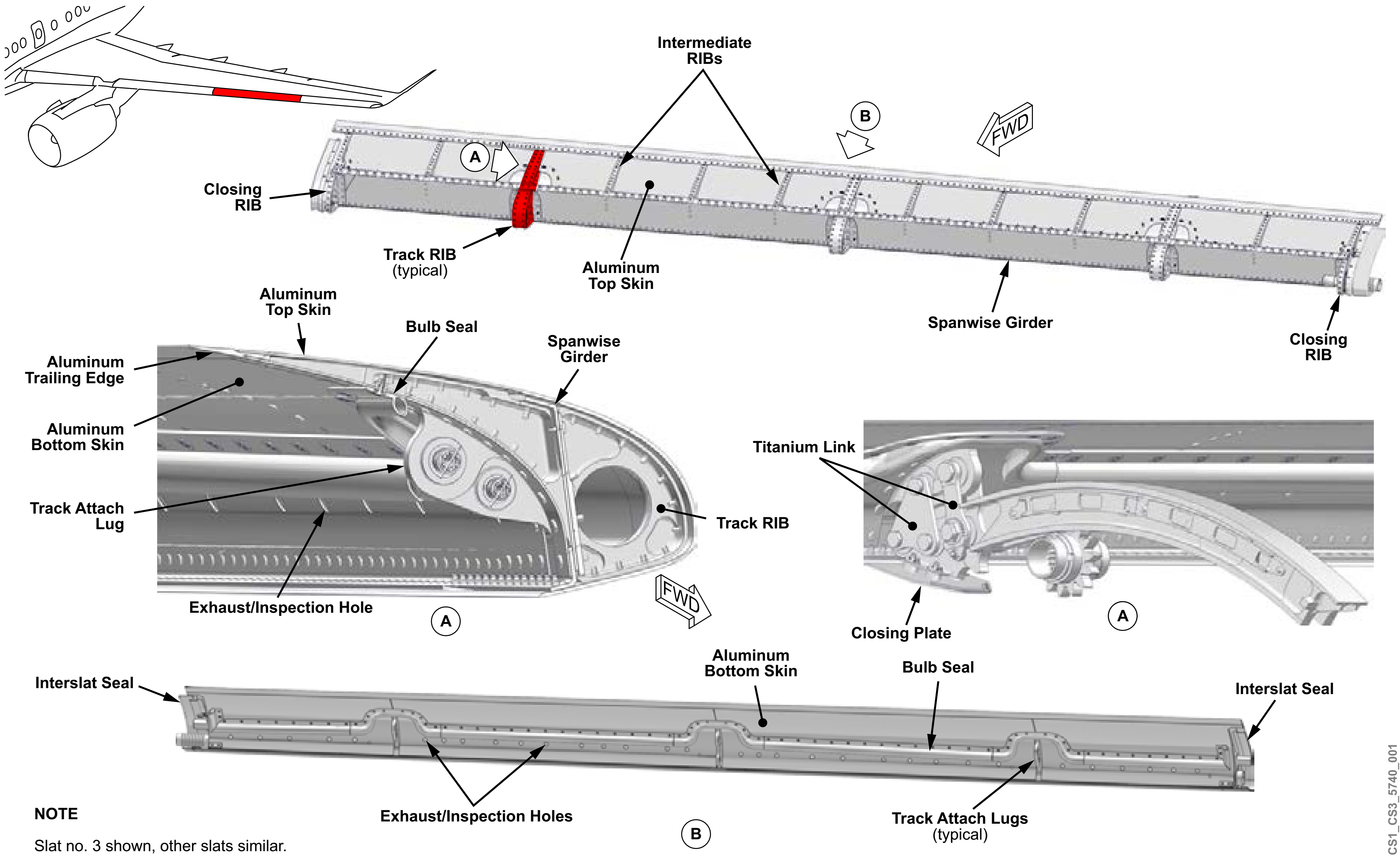
Silicon bulb seals are used at each slat end to provide an interslat air tight gap. A fixed slat seal that runs the length of the lower surface, providing an air tight gap when slats are retracted and protecting against rub wear of the fixed leading edge surface.

Each slat, with the exception of slat 4, has three slat track attachment lugs, located at each of the machined track ribs.

The bottom skin has exhaust/inspection holes that exhaust anti-ice air. The exhaust/inspection holes also provide borescope inspection access to the interior of the slat.

The inboard lower end of each slat has a drain hole to permit moisture out of the slat.

Titanium links attach the slat to the slat track through the slats attach ribs and hardware. Closing plates are installed using screws to cover the slats tracks attachment cutouts on the lower fixed leading edge.



NOTE

Slat no. 3 shown, other slats similar.

Figure 15: Slat Construction

SLAT 4 MAJOR DIFFERENCES

Slat 4 has two attachment lugs. It has a hold down bracket/bolt assembly at the lower mid span, between the two slat tracks. The bolt/bracket assembly contacts a replaceable hold down pad on the fixed leading edge, when the slat is retracted. This permits slat 4 flexing along the wing bending area.

Slat 4 outboard end has an access panel on its lower surface for accessing the wing anti-ice system pressure sensor.

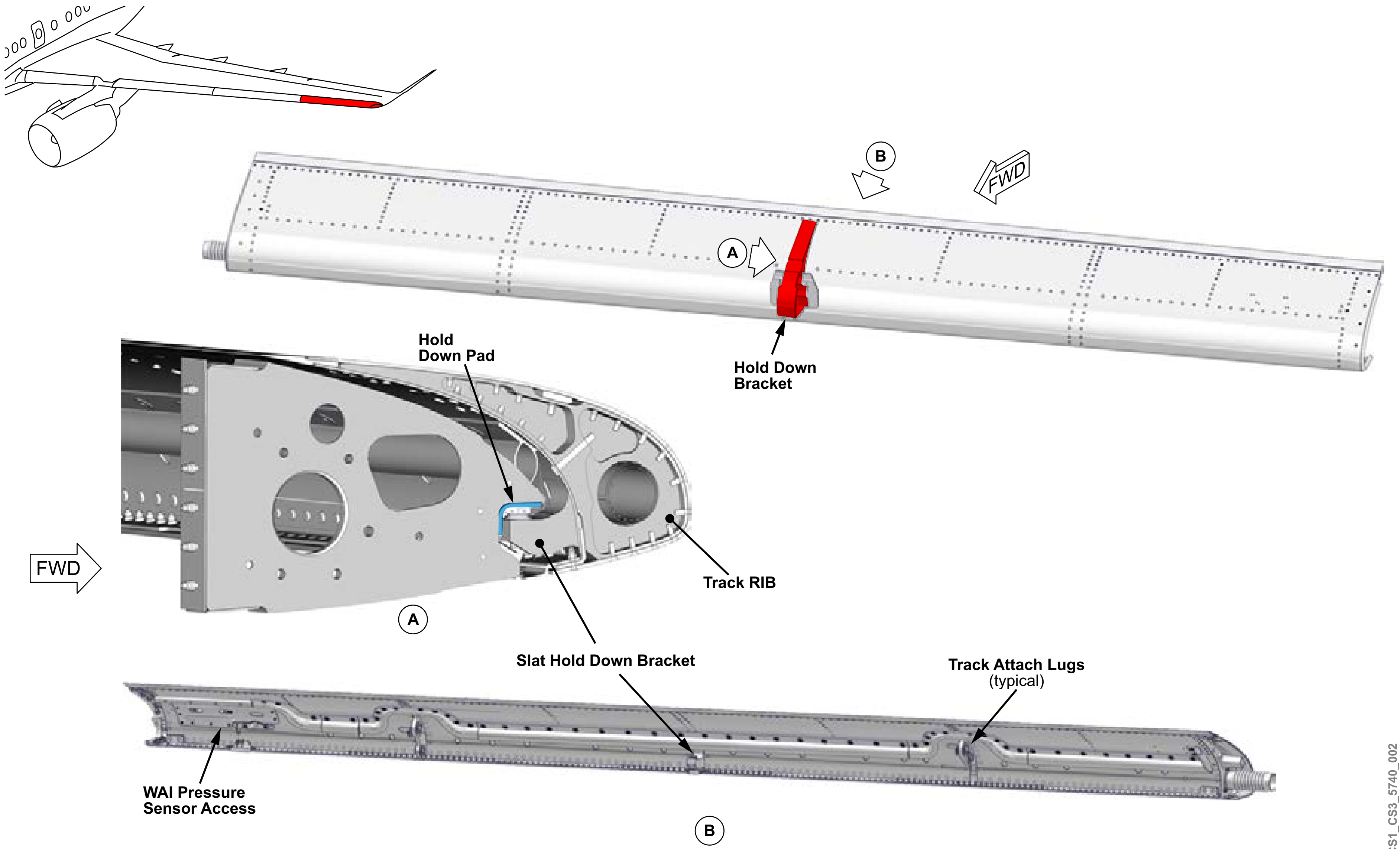


Figure 16: Slat 4

57-50 TRAILING EDGE DEVICES - FLAPS

GENERAL DESCRIPTION

There are four trailing edge flaps installed on the wing trailing edges, an inboard and an outboard on the left and right wings. The inboard and outboard flaps are essentially of the same construction. With the exception of dimensions, flap track beams, and transmission mounting fittings, the two flaps share similar structural components.

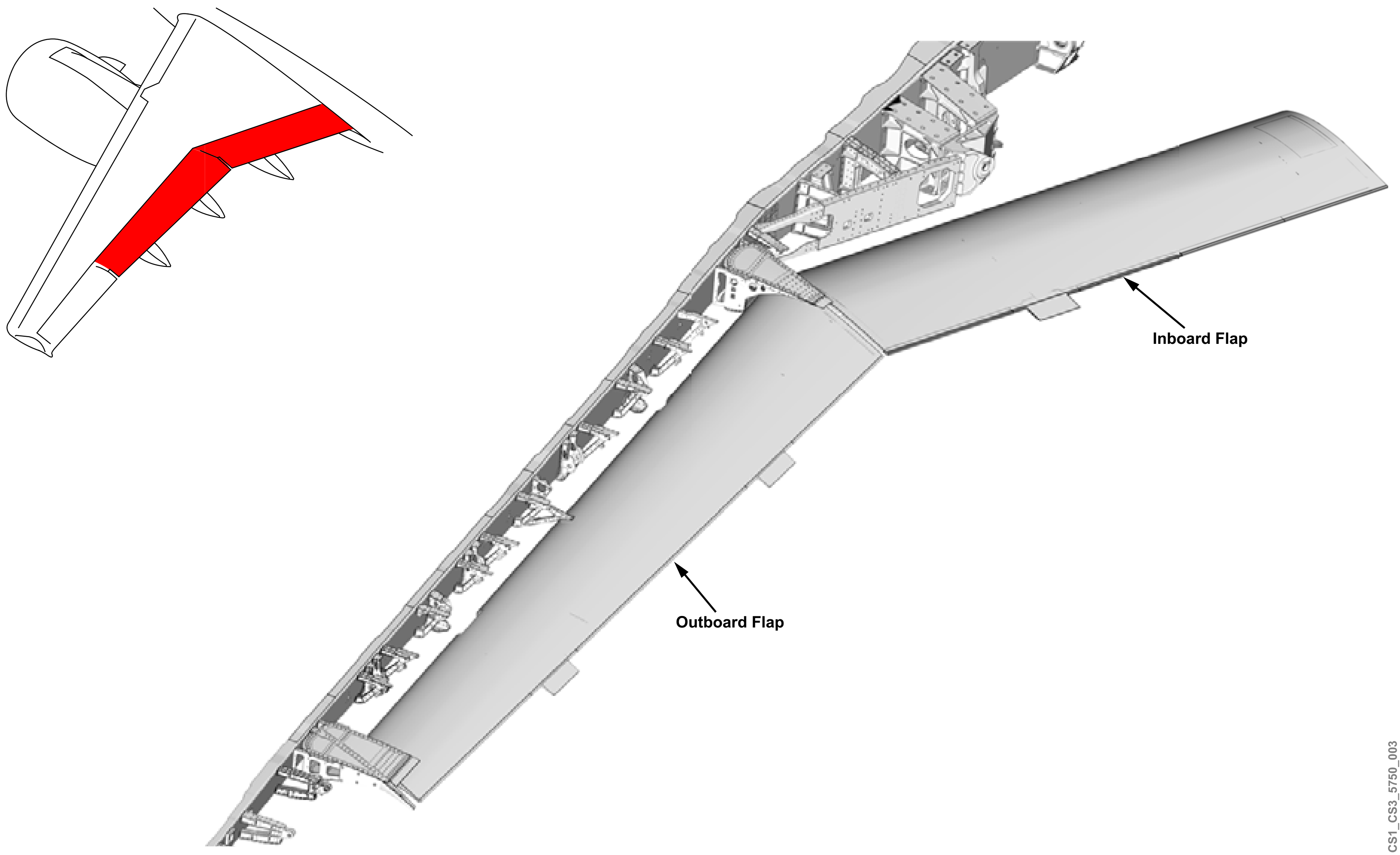


Figure 17: Flaps

DETAILED COMPONENT INFORMATION

INBOARD FLAPS

The inboard flap is installed on the wing trailing edge and extends from RIB 3 (where the outer wing joins to the aircraft center wing section), to approximately RIB 11, left and right.

The inboard flaps are also fitted with upper surface antifriction strips to prevent upper skin wearing against the trailing edge structure.

Borescope access holes are provided in the ribs to permit inspection of the internal structure of the inboard flaps. Access holes are located on the flap lower surface at both track ribs to provide access to the installation hardware inside the flap.

The flap assembly consists of the following components:

- CFRP upper and lower skins
- Integral co-cured front spar, rear spar, and stringers
- CFRP support ribs
- Machined aluminum alloy track ribs
- Aluminum alloy inboard closure rib
- CFRP outboard closure rib
- Aluminum alloy flap carriage fittings
- Two corrosion resistant steel torque tubes, which connect to the inner flap track carriage assembly
- Trailing edge aluminum alloy wedge sections for lightning strike protection
- Seals

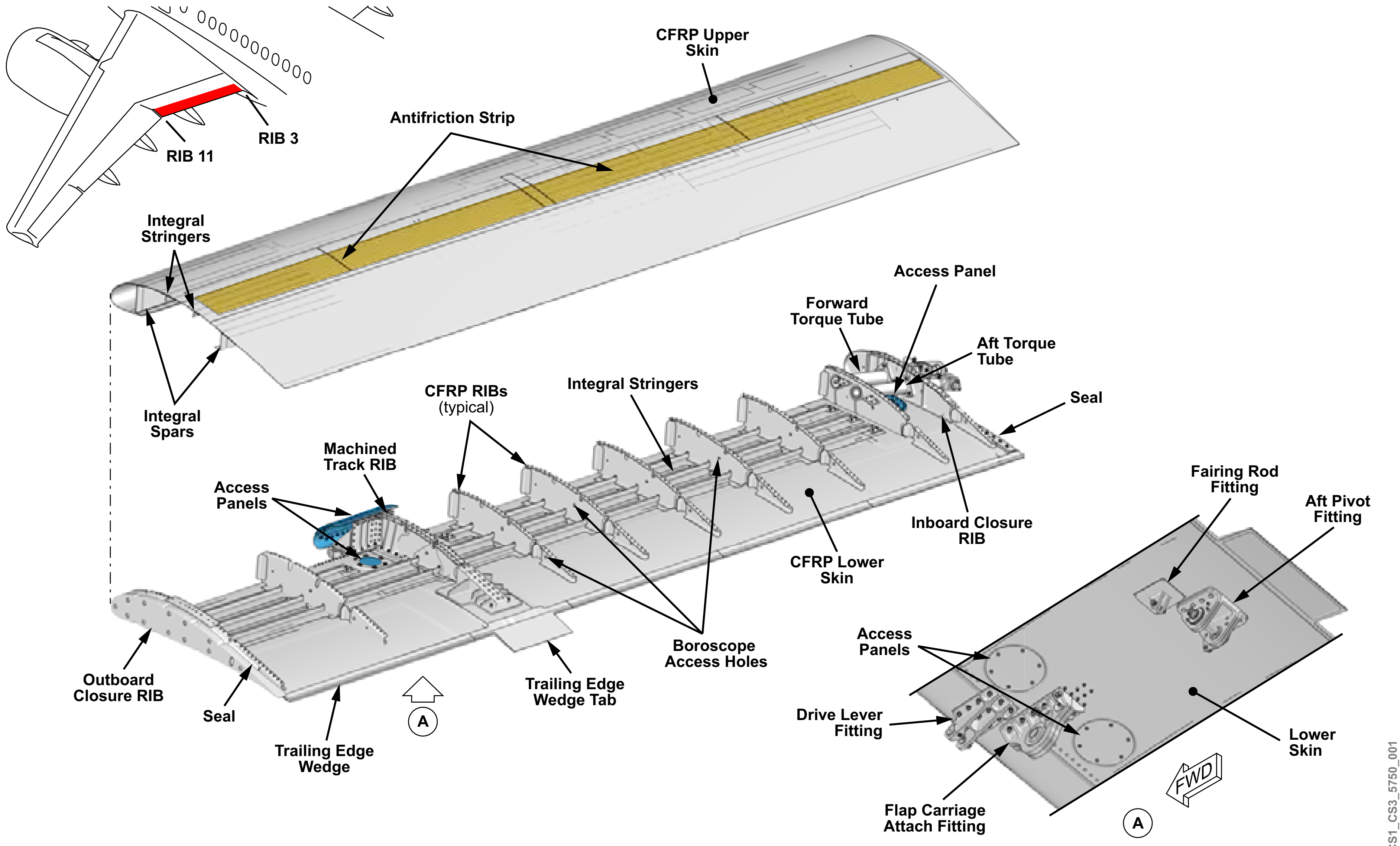


Figure 18: Inboard Flaps

OUTBOARD FLAPS

The outboard flap is installed on the wing trailing edge and extends from wing RIB 11 to approximately RIB 22, left and right.

Access panels are installed on the lower skin and the leading edge of the outboard flap.

Outboard flap end closure ribs have seals that provide aerodynamic airflow when flaps are fully retracted.

The outboard flap is fitted with an antifriction strip to prevent premature wearing of the upper skin against the trailing edge surface of the multifunction spoilers.

The flap assembly consists of the following components:

- CFRP upper and lower skins
- Integral co-cured front spar, rear spar, and stringers
- CFRP support ribs
- Machined aluminum alloy track ribs
- Aluminum alloy inboard closure rib
- CFRP outboard closure rib
- Aluminum alloy flap carriage fittings
- Trailing edge aluminum alloy wedge sections for lightning strike protection

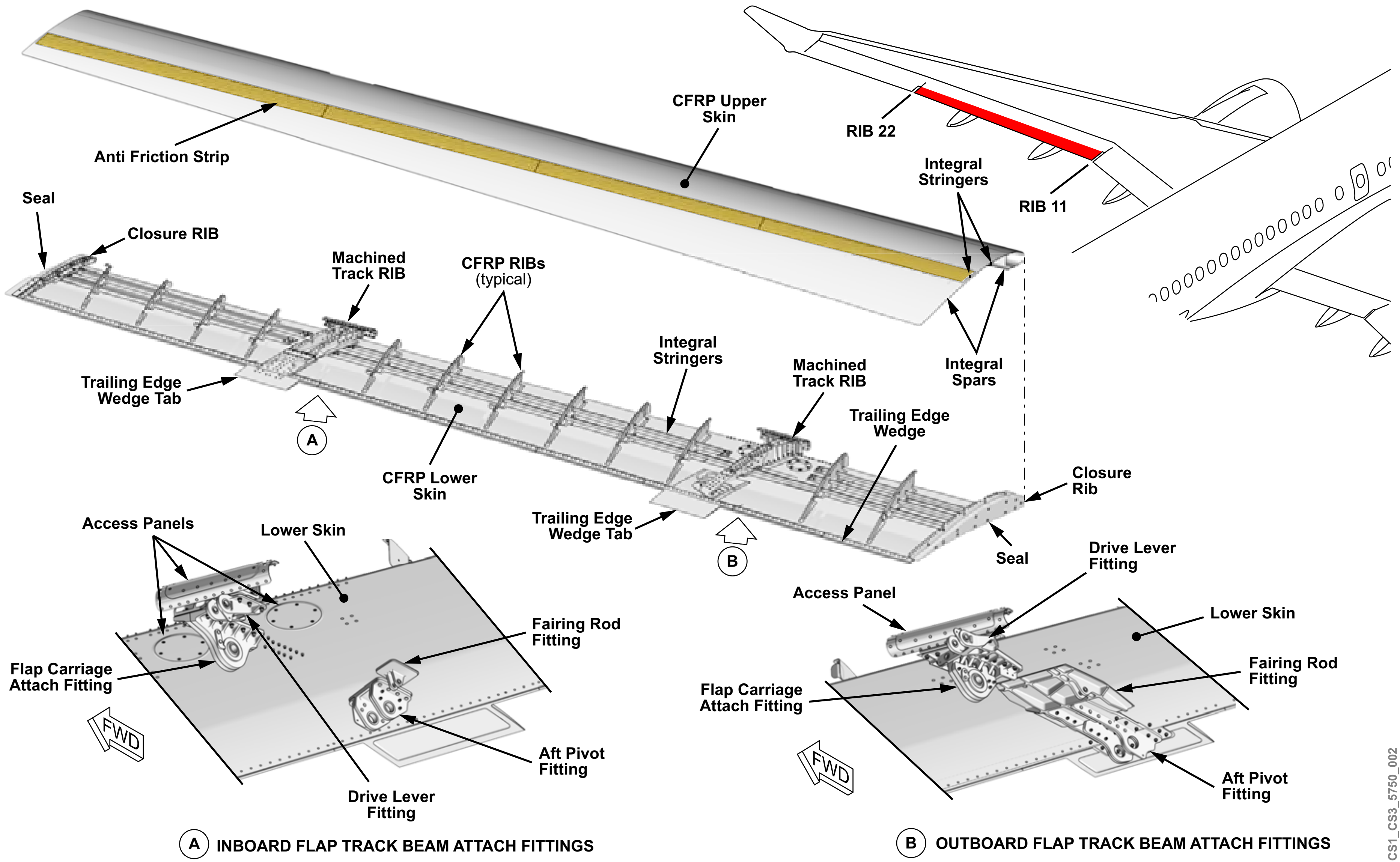
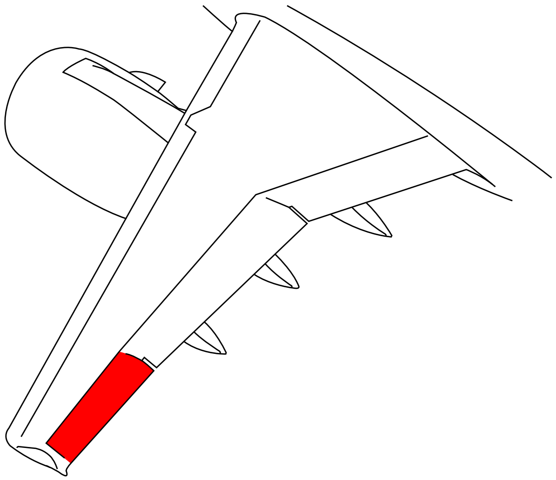


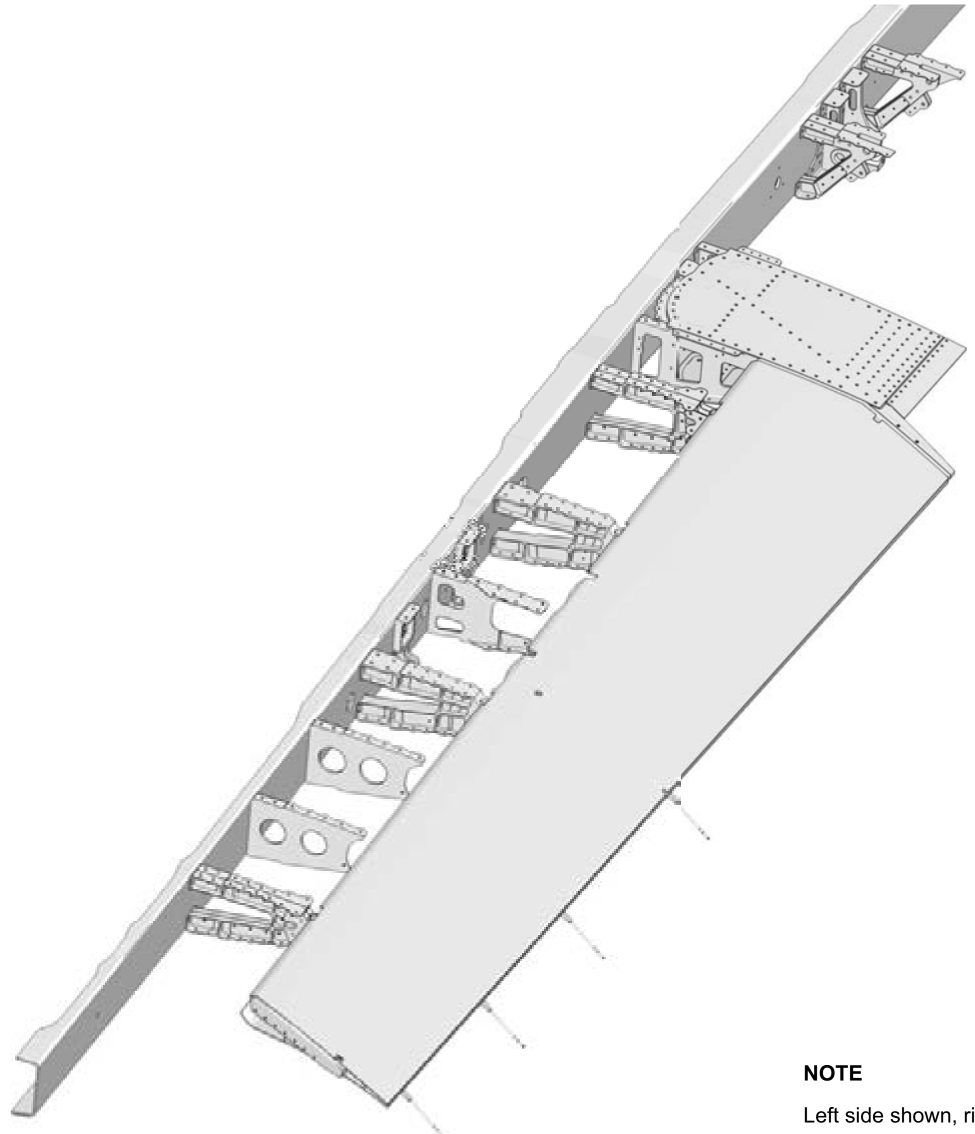
Figure 19: Outboard Flaps

57-60 AILERONS

GENERAL DESCRIPTION

The two ailerons are located outboard of the outboard flaps. They are attached to the wing rear spar by four hinge fittings. Two additional fittings are used to attach the aileron power control units.





NOTE

Left side shown, right side similar.

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Figure 20: Ailerons

DETAILED COMPONENT INFORMATION

The ailerons attach to the wing rear spar and spans from wing RIB 22 to approximately RIB 27.

The ailerons consist of upper and lower carbon fiber reinforced polymer (CFRP) skins with three integral co-cured stringers, and an integral co-cured front spar. The inboard and outboard closure ribs, and the trailing edge insert, are made from aluminum alloy.

Four machined aluminum alloy hinge fittings, and two machined aluminum alloy aileron power control unit attach fittings are attached to the front aileron integral spar.

Access panels are provided on the aileron lower surface at each of the six fittings and at both PCU fittings on the upper surface. To make aileron interior inspection possible, nine boroscope inspection holes are provided on the lower skin surface.

Four static dischargers are attached to the outboard trailing edge.

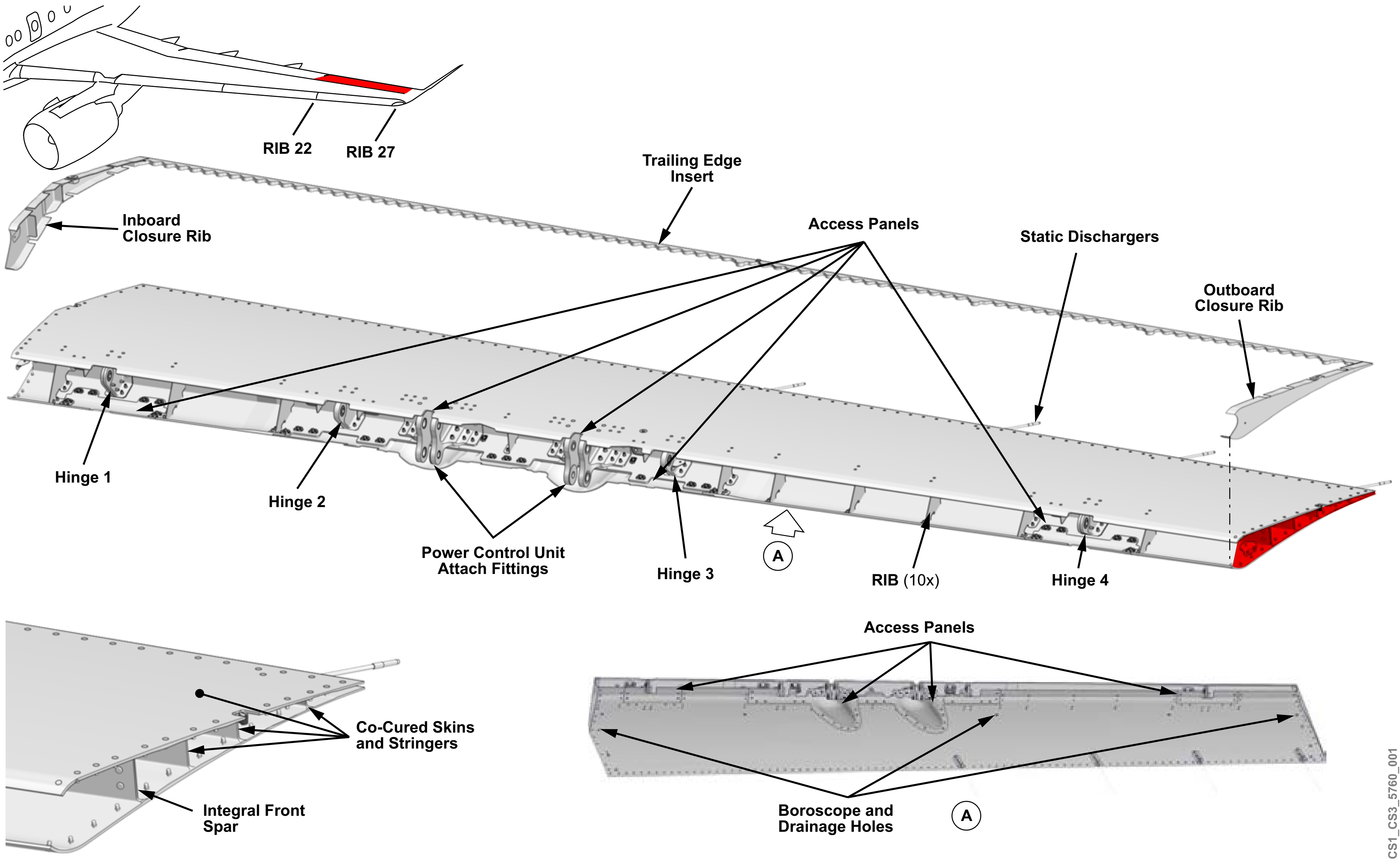


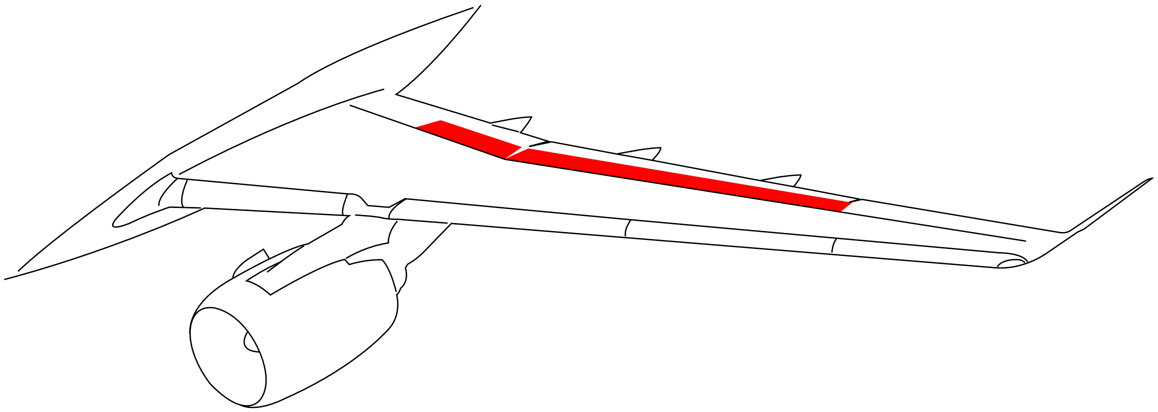
Figure 21: Ailerons Construction

57-70 SPOILERS

GENERAL DESCRIPTION

There are a total of five spoilers per wing, consisting of a single ground spoiler, and four multifunction spoilers (MFSs). The four MFSs are hinged to the wing rear spar, and stow flush to the wing upper surface, directly above the flaps. The ground spoiler is attached to the auxiliary spar. The ground spoiler is the innermost spoiler, and the MFSs are numbered inboard to outboard, from 1 to 4.

The ground spoiler panels and the MFSs have inboard and outboard hinges with drop links, and center fittings with integral hinge fittings, for attachment to the actuators (ground spoilers), or power control units (PCUs).



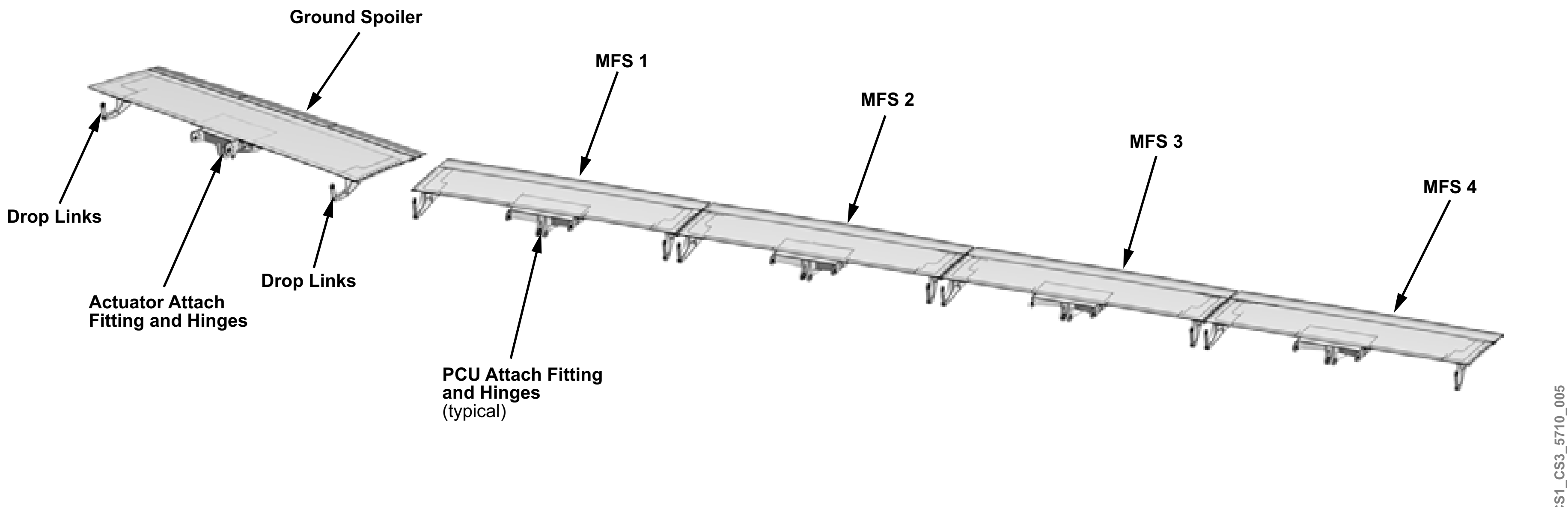


Figure 22: Spoilers

DETAILED COMPONENT INFORMATION

GROUND SPOILERS

The ground spoilers span from wing RIB 7 to RIB 11.

The ground spoiler hinge and actuator fittings are comprised of:

- A center hinge
- Inboard and outboard hinges
- Drop link assemblies

The hinges and drop links are manufactured from aluminum alloy.

Each ground spoiler assembly is made from honeycomb construction with CFRP laminate skins, and a Nomex core.

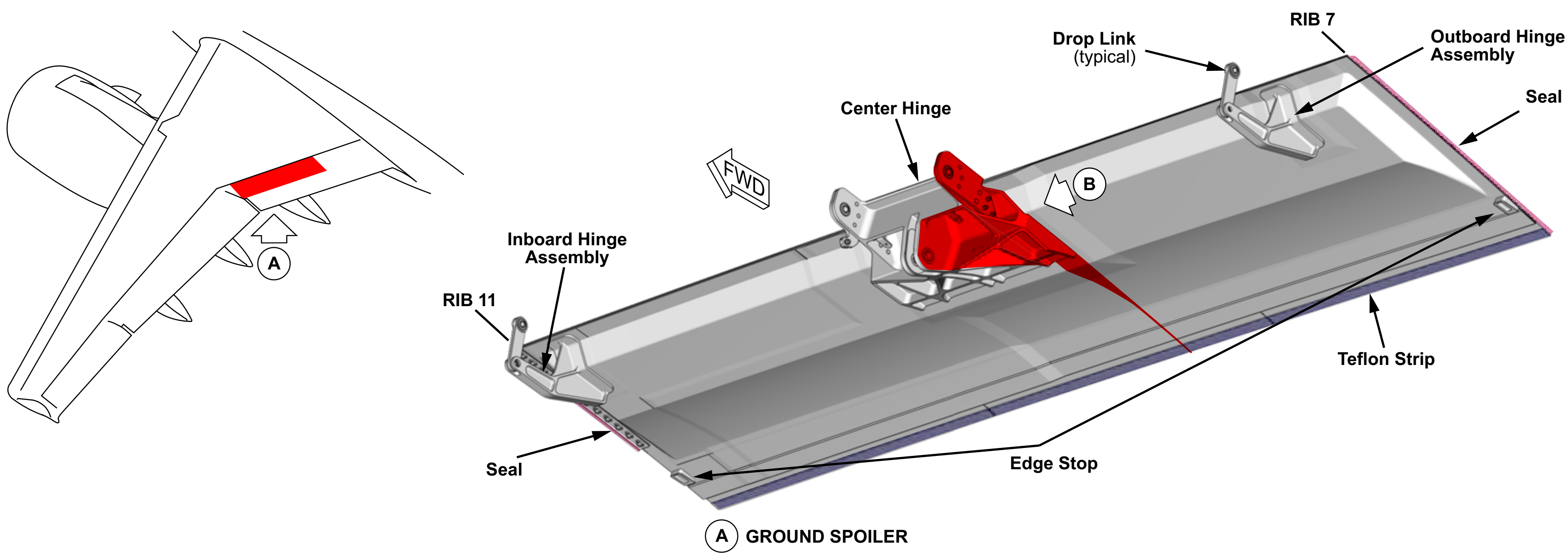
Glass fiber reinforced polymer cloth is incorporated in the composite where metallic components are installed. A high-density aluminum alloy core is installed in the spoiler honeycomb where the hinges are fitted.

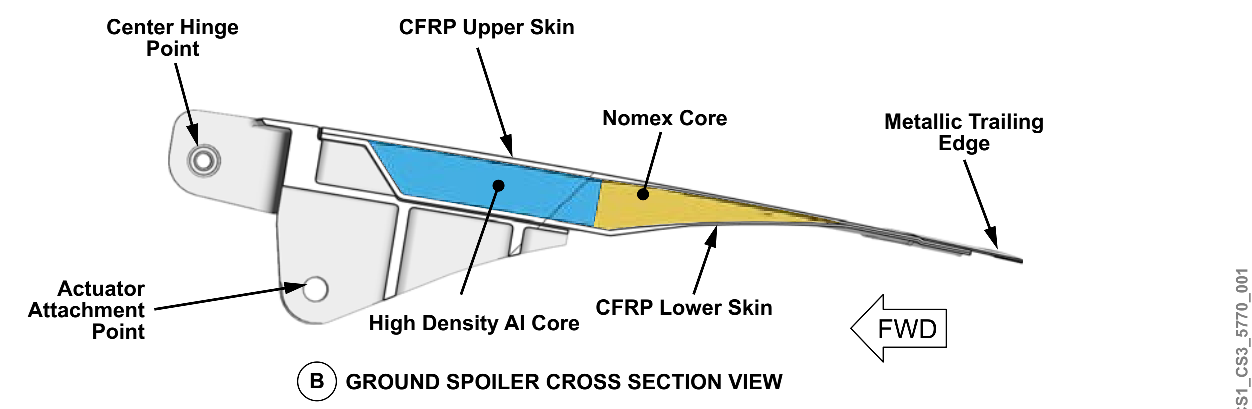
All of the components, and fasteners are wet. They are assembled to provide complete environment sealing.

The ground spoiler lower aft surface has a Teflon wear strip that protects both the spoiler and inboard flap upper surfaces against premature fretting wear.

Both inboard and outboard edges are fitted with a seal to provide aerodynamic sealing of the ground spoiler to trailing edge of the wing.

Both edges are also fitted with an edge stop that rests on the trailing edge structure, when the ground spoiler is stowed, to prevent surface vibration.





NOTE

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Figure 23: Ground Spoilers

MULTIFUNCTION SPOILERS

The four multifunction spoilers (MFSs) are installed on the wing trailing edge and extend from approximately wing RIB 11 to RIB 22. The MFS, when stowed, provides the aerodynamic outer mold line definition between the trailing edge shroud panels and the flaps.

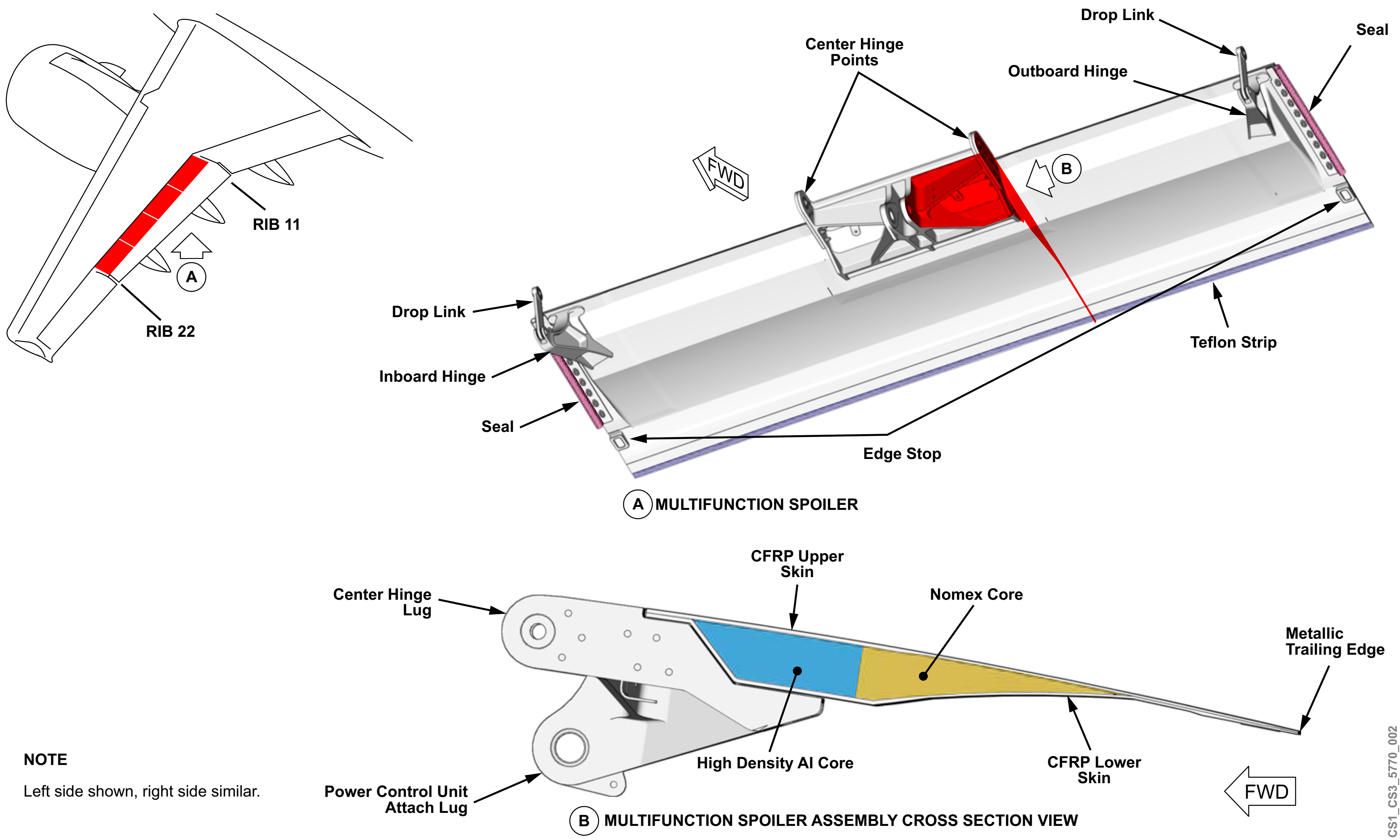
Each MFS assembly is made from honeycomb construction with CFRP laminate skins, and a Nomex core. A high-density aluminum alloy core is used in the area of the hinge and power control unit fittings. Glass fiber reinforced polymer cloth is co-cured into the CFRP assemblies for galvanic protection where there is contact with aluminum parts.

A ply of expanded copper foil is applied to the external surface for lightning protection.

The MFS lower aft surface has a Teflon wear strip that protects both the spoiler and outboard flap upper surface against premature fretting wear.

Both inboard and outboard edges of the MFS are fitted with a seal to provide aerodynamic sealing with the trailing edge of the wing.

Both edges are also fitted with an edge stop that rests on the trailing edge structure, when the MFS is stowed, to prevent surface vibration.



NOTE

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Figure 24: Multifunction Spoilers

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