



Delta II

Payload Planners Guide

December 2006

DELTA II

PAYLOAD PLANNERS GUIDE

The Delta II Payload Planners Guide has been cleared for public release by the Chief, Office of Security Review, Department of Defense, as stated in letter 06-S-1913, dated July 17, 2006.

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CHANGE RECORD

Revision Date	Version	Change Description
July 2006	2006	All <ul style="list-style-type: none"> • Minor corrections throughout • Replaced “Delta Launch Services” with “Delta Program Office” • Deleted Appendices A and B
		Introduction <ul style="list-style-type: none"> • Deleted Figure 1
		Section 1 <ul style="list-style-type: none"> • Updated Delta history graphic (Figure 1-1) • Updated Delta II configuration graphic (Figure 1-2)
		Section 2 <ul style="list-style-type: none"> • Updated discrete values in performance tables (Tables 2-3 and 2-4) • Reordered performance curves into different groupings (Figures 2-7 through 2-32)
		Section 3 <ul style="list-style-type: none"> • Updated usable envelope information for all fairings (Figures 3-3, 3-4, 3-5, 3-8, 3-9, 3-10, 3-11, 3-14, 3-15, and 3-16) • Added fairing envelope information for the reduced height dual-payload attach fitting (Figure 3-12) • Added payload fairing access door information (Section 3.5)
		Section 4 <ul style="list-style-type: none"> • Updated Eastern Range and Western Range facility and electromagnetic environments • Added GN₂ purge connector details (Section 4.1.1.2) • Updated fairing pressure envelope (Figure 4-7) • Updated payload environments: thermal, acoustic, vibration, and shock • Updated third-stage mass properties
		Section 5 <ul style="list-style-type: none"> • Added 3715 and 4717 PAFs • Added reduced-height dual payload attach fitting (RHDPAF) • Added information on customer-provided PAFs • Updated capabilities of PAFs • Updated figures for PAFs • Updated electrical design criteria
		Section 6 <ul style="list-style-type: none"> • Deleted in-depth information on Astrotech Space Operations facilities (Section 6.2.1) • Updated Bldg AE Mission Director Center floor plan (Figure 6-3) • Deleted weather constraint information (Section 6.5.2.3)

Revision Date	Version	Change Description
		Section 7 <ul style="list-style-type: none"> Deleted in-depth information on Astrotech Space Operations and Spaceport Systems International facilities (Sections 7.2.3 and 7.2.4) Updated launch operations floor plans (Figures 7-4, 7-7, 7-8, 7-9, 7-19, and 7-21)
		Section 8 <ul style="list-style-type: none"> Updated listing of customer and Boeing data requirements (Tables 8-1, 8-2, and 8-3) Revised entire Spacecraft Questionnaire (Table 8-4)
		Section 9 <ul style="list-style-type: none"> Updated safety document references to current versions
		Appendix A <ul style="list-style-type: none"> Deleted Appendix A
		Appendix B <ul style="list-style-type: none"> Deleted Appendix B

PREFACE

This Delta II Payload Planners Guide (PPG) is issued to the spacecraft user community to provide information about the Delta II family of launch vehicles and its related systems and launch services.

This document contains current Delta II information and includes United Launch Alliance plans and projections for Delta II launch services launch vehicle specifications. Included are Delta II family vehicle descriptions, target vehicle performance figures, payload envelopes, anticipated spacecraft environments, mechanical and electrical interfaces, payload processing, and other related information of interest to our potential customers.

As new development in the Delta II program progresses, United Launch Alliance will periodically update the information presented in the following pages. To this end, you are urged to visit our Web site so that you can download updates as they become available.

Recipients are also urged to contact United Launch Alliance with comments, requests for clarification, or requests for supplementary information to this document.

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GLOSSARY

°C	Celsius
°F.....	Fahrenheit
ε.....	emittance
μV	microvolt
σ	standard deviation
Ω.....	ohm
1 SLS OB	First Space Launch Squadron Operations Building
30 SW.....	30 th Space Wing
45 SW.....	45 th Space Wing
A-50	Aerozone 50
AASHTO	American Association of State Highway and Transportation Office
AC, A/C	air-conditioning
ACEB.....	air conditioning equipment building
ACS.....	attitude control system/auxiliary control system
ACSR	auxiliary control system rack
ADOTS	advanced Delta ordnance test set
ADS.....	analysis description sheet/automatic destruct system
AFB.....	Air Force Base
AFSPCMAN	Air Force Space Command Manual
AGE	aerospace ground equipment
AKM	apogee kick motor
ALCS	advanced launch vehicle control system
ANSI.....	American National Standards Institute
ATF	Bureau of Alcohol, Tobacco, Firearms and Explosives
AWG	American wire gage
B&W	black and white
B/H.....	blockhouse

BAS..... breathing-air supply

BET..... best estimate trajectory

BLS..... Boeing Launch Services

Btu..... British Thermal Unit

C3..... launch energy

CAD..... computer-aided drawing; computer-aided design

CCAFS..... Cape Canaveral Air Force Station

CCAM..... contamination and collision avoidance maneuver

CCTV..... closed-circuit television

CD..... countdown

CG..... center-of-gravity

CL..... centerline

CLA..... coupled loads analysis

cm..... centimeter

CRD..... command receiver decoder

CSR..... control system rack

CW..... clockwise

DAT..... digital audio tape

dB..... decibel

DCI..... document change instruction

deg..... degree

dia..... diameter

DIS..... Defense Investigative Service

DMA..... direct mate adapter

DMCO..... Delta mission checkout

DOD..... Department of Defense

DOP..... dioctyl phthalate

DOT..... Department of Transportation

DPAF	dual-payload attach fitting
dps	degrees per second
DTO	detailed test objective
DXF.....	data exchange format
ECS	environmental control system
EEB	electrical equipment building
EED.....	electro-explosive device
EIA.....	Electronic Industry Association/electronic initiator assembly
EIRP	effective isotropic radiated power
El	elevation
EMI	electromagnetic interference
EMT	electrical-mechanical testing
E-pack	electronics package
ER	Eastern Range
ESA.....	engineering support area
ETA.....	explosive transfer assembly
EWR.....	Eastern and Western Regulation
F/O; FO	fiber-optic
FAA.....	Federal Aviation Administration
fc	foot-candle
FCC.....	Federal Communications Commission
FED-STD	Federal Standard
FOTS.....	fiber-optic transmission system
FRR.....	flight readiness review
FSPO	Flight Safety Project Officer
ft	feet
FUT	fixed umbilical tower
g.....	gravity

GCR	ground control rack
GEM.....	graphite-epoxy motor
GHe	gaseous helium
GHz	gigahertz
GMT.....	Greenwich mean time
GN ₂	gaseous nitrogen
GPS	global positioning system
GSE	ground support equipment
GSFC.....	Goddard Space Flight Center
GTO	geosynchronous transfer orbit
HB	Huntington Beach
HEPA	high-efficiency particulate air
HPF	hazardous processing facilities
HTPB	hydroxyl terminated polybutadiene
Hz.....	hertz
i	inclination
ICD.....	interface control document
IGES.....	Initial Graphics Exchange Specification
IIP.....	instantaneous impact point
in	inch
IPF.....	integrated processing facility
ISDS	inadvertent separation destruct system
I _{sp}	specific impulse
J-box.....	junction box
kg.....	kilogram
KHz	kilohertz
km	kilometer
KNPR	Kennedy NASA Procedural Requirement

KSC.....	Kennedy Space Center
lb	pound
lbm	pound mass
LCC.....	launch control center
LCCD.....	line charge coupling device
LEO.....	low-Earth orbit
LH ₂	liquid hydrogen
LO ₂	liquid oxygen
LOCC.....	launch operations control center
LOP	launch operations plan
LPD.....	launch processing document
LRR.....	launch readiness review
LSIM.....	launch site integration manager
LSRR.....	launch site readiness review
LSTP.....	launch site test plan
lux	lumen per square meter
LV	launch vehicle
LVDC.....	Launch Vehicle Data Center
m	meter
MD	mission director
MDC	Mission Director Center
MECO.....	main engine cutoff
MEOP	maximum expected operating pressure
MHz	megahertz
MIC.....	meets intent certification
MIL	military
MIL-STD	military standard
MIM	mission integration manager

MLV	medium launch vehicle
mm	millimeter
MMS	multimission modular spacecraft
MOI	moments of inertia
MSPSP	missile systems prelaunch safety package
MSR	mission support request
MST	mobile service tower
mV	millivolt
N	newton
N/A	not applicable
N ₂ O ₄	nitrogen tetroxide
NASA	National Aeronautics and Space Administration
NCS	nutation control system
nmi	nautical mile
NOAA	National Oceanographic and Atmospheric Administration
NVR	nonvolatile residue
OASPL	overall sound pressure level
OB	operations building
OD	operations directive
OR	operations requirement
OSM	operations safety manager
OSMC	operations safety manager's console
P&C	power and control
P/N	part number
PAF	payload attach fitting
PAM	payload assist module
PCM	pulse code modulated
PCS	probability of command shutdown

PDS	propellant depletion shutdown
PHE	propellant handler's equipment
PIP	push-in-pull-out
PLF	payload fairing
PMA	preliminary mission analysis
PPF	payload processing facilities
PPG	payload planners guide
ppm	parts per million
PPRD.....	Payload Processing Requirements Document
PRD	Program Requirements Document
psi	pounds per square inch
psia	pounds per square inch absolute
PSM	program support manager
PWU	portable weight unit
Q	dynamic pressure
R	radius
RAAN	right ascension of ascending node
RACS	redundant attitude control system
RCO	Range Control Officer
RCS	reaction control system
RF	radio frequency
RFA	radio frequency application
RFI	radio frequency interference
RGA	rate gyro assembly
RGEA	rate gyro electronics assembly
RHDPAF	reduced-height dual-payload attach fitting
RIFCA	redundant inertial flight control assembly
RLC	remote launch center

RLCC	remote launch control center
ROC	Range Operations Center
RP-1	kerosene
rpm	revolutions per minute
S&A	safe and arm
SAEF 2.....	Spacecraft Assembly and Encapsulation Facility Number 2
SC, S/C.....	spacecraft
SCA.....	spring cartridge assembly
SCAPE	self-contained atmospheric protection ensemble
scfm.....	standard cubic feet per minute
sec	second
SECO	second-stage engine cutoff
SLC	Space Launch Complex
SLS.....	Space Launch Squadron
SMC	Space and Missile Systems Center
SMFCO	senior mission flight control officer
SOP	standard operating procedure
SRM	solid-rocket motor
SSI.....	Spaceport Systems International
ST.....	straight tip
STD.....	standard
STEP	Standard for the Exchange of Product
STP.....	special technical publication
SVAFB.....	South Vandenberg Air Force Base
SW.....	Space Wing
SW/CC	Space Wing Commander
TBD.....	to be determined
TECO	third-stage burnout

TIMtechnical interchange meeting

TLX.....thin-layer explosive

TM..... telemetry

TMS telemetry system

TOPS..... transistorized operations phone system

TT&C.....telemetry, tracking, and command

U.S. United States

UDS.....Universal Document System

UHF.....ultra-high frequency

UPS uninterruptible power supply

USAFUnited States Air Force

UV.....ultraviolet

V..... volt

VABvehicle assembly building

VAC volts alternating current

VAFB.....Vandenberg Air Force Base

VC..... visible cleanliness

VCRvehicle control rack

VDC volts direct current

VIM..... vehicle information memorandum

VM video monitor

VOS.....vehicle on stand

VRR vehicle-on-stand readiness review

W.....watt

WR Western Range

INTRODUCTION

This guide describes the Delta II launch system including its background, heritage, and performance capabilities. Additionally, launch facilities, operations, and mission integration are discussed, as is the payload environment during ascent. Documentation and procedural requirements associated with preparing and conducting the launch are also defined herein.

The Delta II design evolved from our reliable Delta launch vehicle, developed to provide the domestic and international user community with an efficient, low-cost launch system. In over four decades of service, Delta launch vehicle success stems from its evolutionary design, which has been steadily upgraded to meet the needs of the user community while maintaining a high reliability record of over 98%.

United Launch Alliance (ULA) operates two launch sites within the continental United States (U.S.)—Eastern Range (ER) in Florida and Western Range (WR) in California. The Space Launch Complex (SLC) of the ER is located at Cape Canaveral Air Force Station (CCAFS) and consists of two launch pads, designated SLC-17A and SLC-17B. Maintenance, mission modifications, and launch preparation may be conducted at one pad without impacting operations at the other. This arrangement enables ULA to provide launch-period flexibility, minimizing risk to customers' schedules. The SLC-2 of the WR is located at Vandenberg Air Force Base (VAFB) and is typically used for missions requiring high-inclination orbits, while SLC-17 is used for low- to medium-inclination orbits. Both launch complexes are open to commercial and government customers and have been regularly upgraded to meet the increasingly rigorous requirements of the space community.

When providing commercial launch services, ULA acts as the coordinating agent for the customer to interface with the United States Air Force (USAF), National Aeronautics and Space Administration (NASA), Federal Aviation Administration (FAA), and any other relevant agency when commercial or government facilities are engaged for payload processing. Commercial agreements with the USAF and NASA make available to ULA the use of the launch facilities and services in support of Delta II launch services.

United Launch Services (ULS) is the single point of contact for all U.S. government customer new-business activities. ULS offers full-service launch solutions using the Delta II and Delta IV family of launch vehicles. The customer is supported by an organization consisting of highly knowledgeable technical and managerial personnel who are dedicated to open communication and responsive to all customer needs.

United Launch Services has ultimate responsibility, authority, and accountability for all Delta U.S. government customer opportunities. This includes developing mission-unique launch

solutions to meet customer needs as well as providing customers with a launch service agreement for the selected launch services.

United Launch Services and the Delta II program work together to ensure that high-level technical customer requirements are fully coordinated. The Delta II program is responsible for the development, production, integration, test, mission integration, and launch of the Delta II system.

For contracted launch services, a dedicated mission integration manager is appointed from within the Delta II program to support the customer. The mission integration manager works with ULS early in the process to define customer mission requirements and the appropriate launch solution and then transitions to provide the day-to-day mission integration support necessary to successfully satisfy the customer's launch requirements. The mission integration manager supports the customer's mission from before contract award through launch and postflight analysis.

The Delta team addresses each customer's specific concerns and requirements, employing a meticulous, systematic, user-specific process that addresses advance mission planning and analysis of payload design; coordination of systems interface between payloads and Delta II; processing of all necessary documentation, including government requirements; prelaunch systems integration and checkout; launch-site operations dedicated exclusively to the user's schedule and needs; and postflight analysis.

The Delta team works closely with its customers to define optimum performance for mission payload(s). In many cases, we can provide innovative performance trades to augment the performance shown in Section 2. Our Delta team also has extensive experience in supporting customers around the world. This demonstrated capability to use the flexibility of the Delta launch vehicle and design team, together with our experience in supporting customers worldwide, makes Delta the ideal choice as a launch service provider.

Section 1 LAUNCH VEHICLE DESCRIPTIONS

This section provides an overall description of the Delta II launch vehicle and its major components. In addition, the Delta vehicle designations are explained in Table 1-1.

1.1 DELTA LAUNCH VEHICLES

The Delta launch vehicle program was initiated in the late 1950s by the National Aeronautics and Space Administration (NASA). The Boeing Company, then McDonnell Douglas (previously Douglas Aircraft Missiles and Space Systems), was the prime contractor. Boeing developed an interim space launch vehicle using a modified Thor as the first stage and Vanguard components as the second and third stages. The vehicle was capable of delivering a payload of 54 kg (120 lb) to geosynchronous transfer orbit (GTO) and 181 kg (400 lb) to low-Earth orbit (LEO). The Boeing commitment to vehicle improvement to meet customer needs led to the Delta family of launch vehicles, with a wide range of increasing capability to GTO (Figure 1-1).

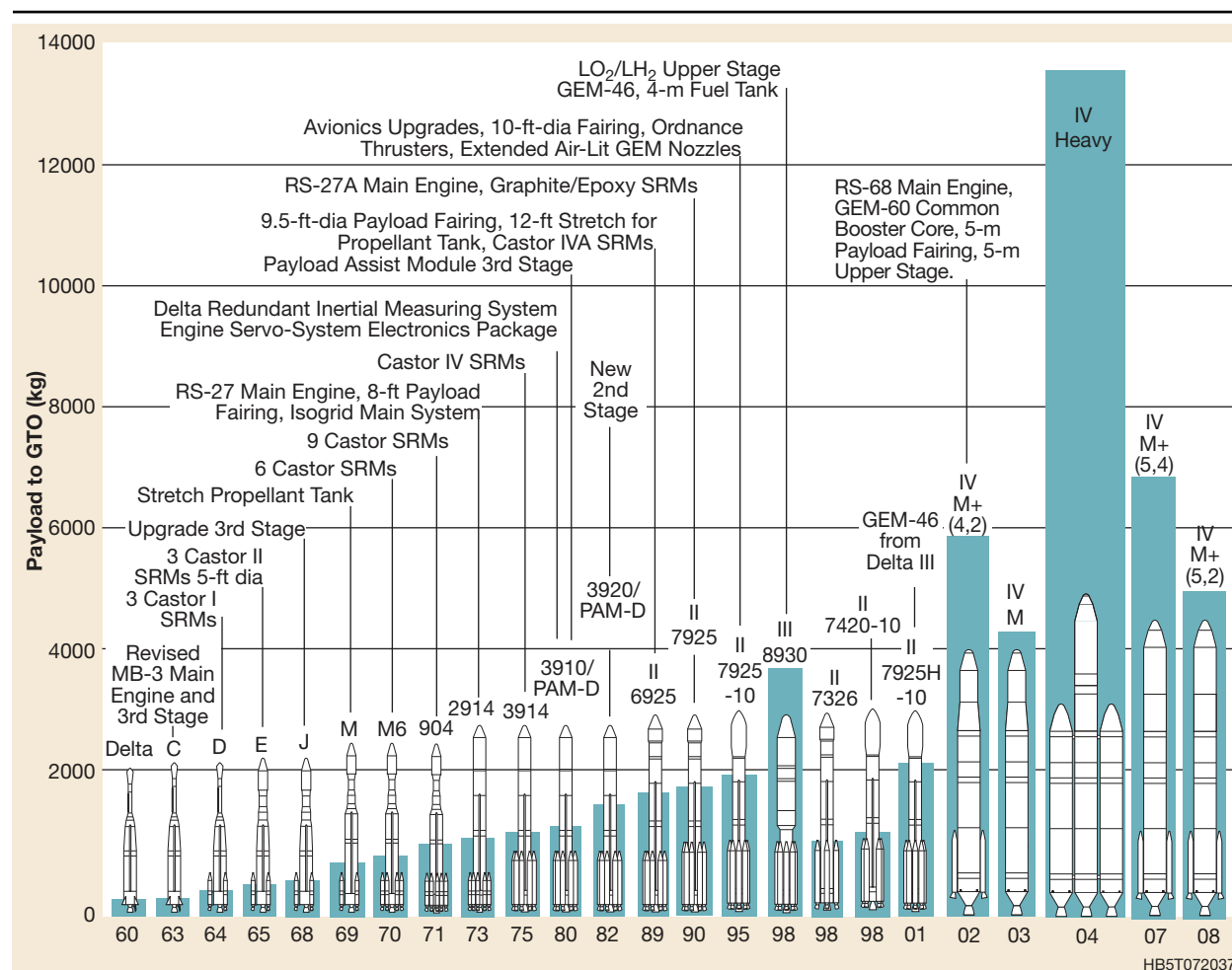


Figure 1-1. Heritage of Delta Family

The Boeing commitment to continuous improvement in meeting customer needs is evident in the many configurations developed to date. Delta II has provided customers with a demonstrated world-class success rate over 98%, and processing times on the launch pad have been reduced. The Delta IV launch system is a continuation of the 45-year evolution, with even more capability. By incorporating heritage hardware, proven processes, and lessons learned, Delta IV provides a broad spectrum of performance capabilities for Medium- to Heavy-class payloads. Boeing is committed to working with our customers to satisfy payload requirements while providing the best value for launch services across the entire Delta fleet.

1.2 DELTA II LAUNCH VEHICLE DESCRIPTION

The major elements of the Delta II launch vehicle are the first stage with its graphite-epoxy motor (GEM) solid strap-on rocket motors, the second stage, an optional third stage with spin table, and the payload fairing (PLF). The vehicle's design robustness has made available a number of configurations suiting customers' needs while optimizing performance (Figure 1-2).

The Delta II launch vehicle series are the 7300, 7400, and 7900; a four-digit system is used to identify various Delta II configurations (Table 1-1). The three-stage 7925-9.5 and the two-stage 7920-10 vehicles shown in Figures 1-3 and 1-4 are representatives of the Delta II family series. The Delta II also has a "Heavy" configuration that employs larger diameter GEM-46 solid strap-on rocket motors on the 7900-series vehicle to further improve the performance capability of Delta II. This new configuration is designated as 7920H for two-stage missions and 7925H for three-stage missions.

1.2.1 First Stage

The first-stage subassemblies include the RS-27A engine section, liquid oxygen (LO₂) tank, centerbody, fuel tank, and the interstage.

The Rocketdyne RS-27A main engine has a 12:1 expansion ratio and employs a turbine/turbopump and a regeneratively cooled thrust chamber and nozzle. The thrust chamber and nozzle are hydraulically gimballed to provide pitch and yaw control. Two Rocketdyne vernier engines provide roll control during main-engine burn and attitude control between main-engine cutoff (MECO) and second-stage separation.

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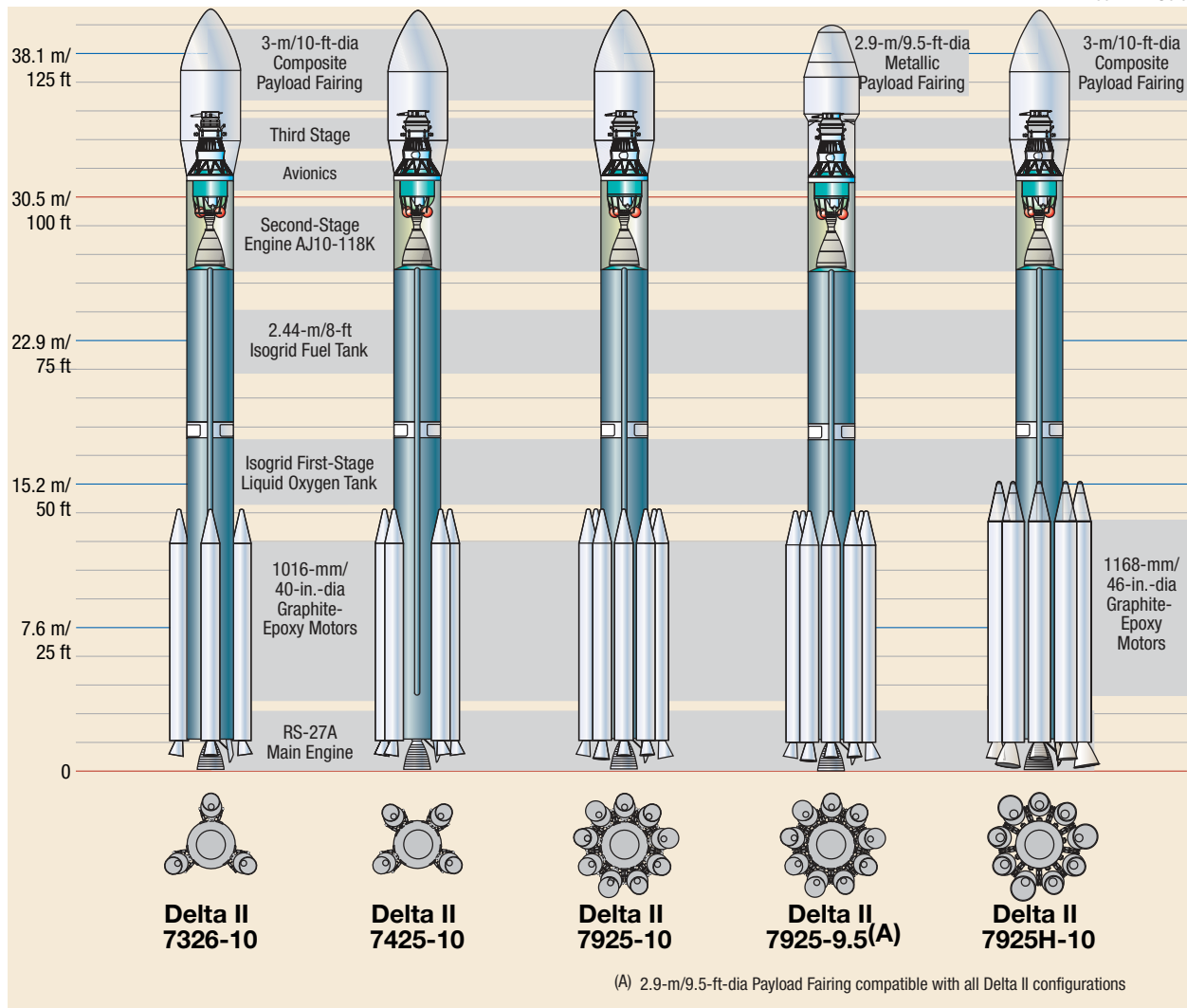


Figure 1-2. Some Typical Configurations of the Delta II Launch Vehicle with Optional Third Stage

The 792X vehicle configuration includes nine Alliant Techsystems' solid rocket GEMs to augment first-stage performance. Six of these GEMs are ignited at liftoff; the remaining three GEMs with extended nozzles are ignited in flight after burnout of the first six. Ordnance for the motor ignition and separation systems is fully redundant. The 732X and 742X vehicles include three or four GEMs respectively, all of which are ignited at liftoff.

In addition to the standard 40-in.-dia GEM that is flown on the Delta II 732X, 742X, and 792X vehicle configurations, the heavier GEM-46 previously flown on Delta III is made available in a Heavy configuration designated 792XH. The GEM-46 has a 46-in. core dia and burns approximately 14 sec longer than the standard GEM-40. Both types of GEMs are flown with a fixed nozzle that is canted outboard from the vehicle centerline at 10 deg.

Table 1-1. Delta II Four-Digit Designation

Digit	Indicates	Examples	
1st	Type of first stage engine and solid rocket motors	7	RS-27A engine (12:1 nozzle ratio); solid rocket GEM by Alliant Tech.
2nd	Number of solid rocket motors	9 4 3	Nine solid rocket motors Four solid rocket motors Three solid rocket motors
3rd	Type of second stage	2	Aerojet AJ10-118K engine
4th	Type of third stage	0 0H 5 5H 6	No third stage No third stage; Heavy configuration with GEM-46 solid rocket motor Star-48B solid motor Star-48B solid motor; Heavy configuration with GEM-46 solid rocket motor Star-37FM solid motor
Dash no.	Type of fairing	-9.5 -10 -10L	2.9-m (9.5-ft)-dia x 8.5-m (27.8-ft)-long fairing 3.0-m (10-ft)-dia x 8.9-m (29.1-ft)-long fairing 3.0-m (10-ft)-dia x 9.2-m (30.4-ft)-long fairing
Example: Delta 7925-10			
Digit	Indicates		
7	RS-27A engine (12:1 nozzle ratio) for first stage augmented by solid rocket GEM		
9	Nine GEM strap-on solid rocket motors		
2	Aerojet AJ10-118K engine for second stage		
5	Star-48B third stage		
-10	3.0-m (10-ft)-dia x 8.9-m (29.1-ft)-long fairing		

002167.5

The LO₂ oxidizer tank, RP-1 fuel tank, and interstage are constructed of aluminum isogrid shells and aluminum tank domes. The centerbody between the fuel tank and LO₂ tank houses the first-stage electronic components on hinged panels for easy checkout access and maintainability.

The interstage, located between the first stage and second stage, carries the loads from the second stage and fairing to the first stage. The interstage provides clearance for the second-stage engine nozzle and contains range safety antennas, exhaust vent for fairing cavity, and six guided-spring actuators to separate the second stage from the first stage.

1.2.2 Second Stage

The second stage is powered by the proven Aerojet AJ10-118K engine and includes fuel and oxidizer tanks that are separated by a common bulkhead. The simple, reliable start and restart operation requires only the actuation of a bipropellant valve to release the pressure-fed hypergolic propellants, with no need for a turbopump or an ignition system. Typical two- and three-stage missions use two second-stage starts, but the restart capability has been used as many as six times on a single mission, for a total of seven burns. During powered flight, the second-stage hydraulic system gimbals the engine for pitch and yaw control. A redundant attitude control system (RACS) using nitrogen gas provides roll control. The RACS also provides pitch, yaw, and roll control during unpowered flight. The guidance system is installed in the forward section of the second stage. The payload attach fitting (PAF) provides the interface between the second stage and the spacecraft for two-stage missions.

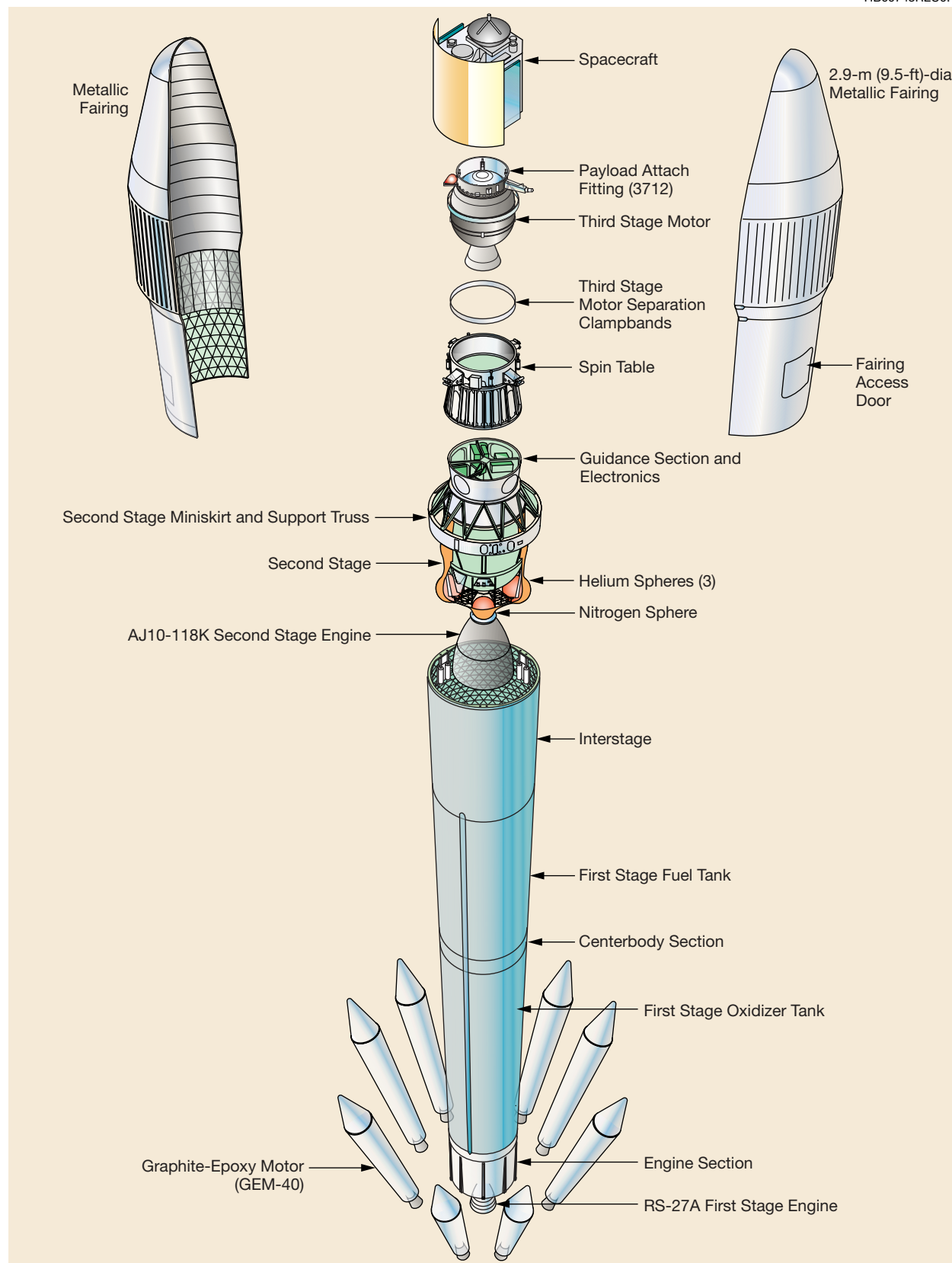


Figure 1-3. Delta 7925-9.5 Launch Vehicle

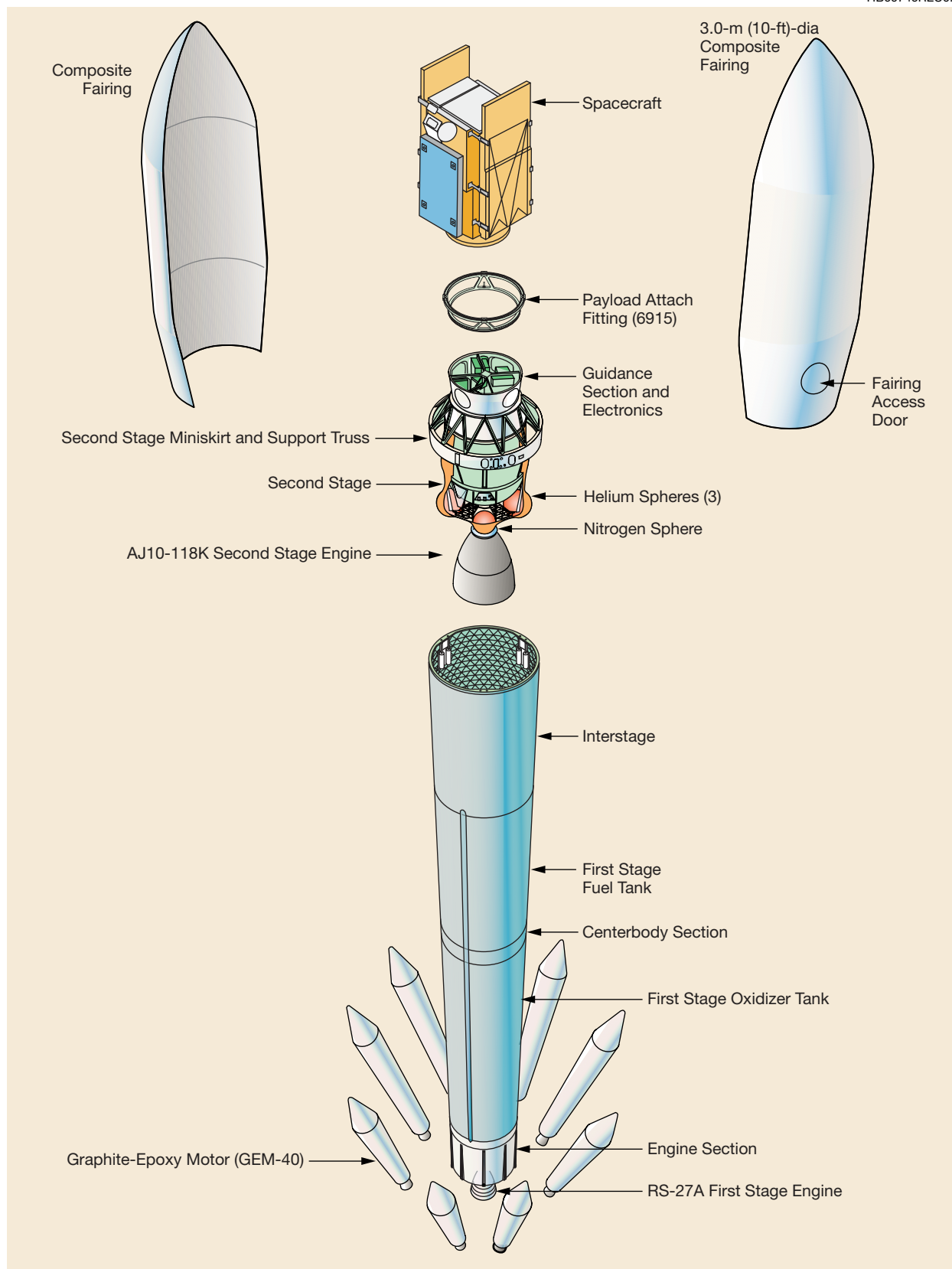


Figure 1-4. Delta 7920-10 Launch Vehicle

1.2.3 Third Stage

The Delta II series of launch vehicles offers two optional spin-stabilized third-stage motors. Depending on payload requirements, either a Star-37FM or Star-48B solid-rocket motor (SRM) can be used. These flight-proven motors are produced by Alliant Techsystems. A spin table, containing small rockets, mounts the third stage to the second stage and is used to spin up the third stage prior to separation. The third-stage payload attach fitting mates the third stage with the spacecraft; this stage can be flown with or without a nutation control system (NCS).

Our flight-proven NCS maintains orientation of the spin axis of the SRM/spacecraft during third-stage flight until just prior to spacecraft separation. The NCS uses monopropellant hydrazine that is prepressurized with helium. This simple system has inherent reliability with only one functioning component and a leak-free design.

An ordnance sequence system is used to release the third stage after spin-up, to fire the motor, and to separate the spacecraft following motor burn. To preclude recontact between the spacecraft and the third stage due to motor residual thrust, a yo-weight system is used to tumble the third stage after spacecraft separation. If a lower spin rate is desired, the third stage can be equipped with a yo-yo weight system to despin prior to spacecraft separation. In this case, recontact is prevented by increasing the ordnance sequence time between motor ignition and spacecraft separation, allowing for sufficient residual thrust decay.

Star-48B SRM. The long nozzle version of the Star-48B motor has a diameter of 1244.6 mm (49.0 in.) and an overall length of 2032.0 mm (80.0 in.). The motor has two integral flanges, the lower for attachment to the third-stage spin table and the upper for attachment to the 3712 PAF. The motor consists of a carbon-phenolic exit cone, 6AL-4V titanium high-strength motor case, silica-filled rubber insulation system, and a propellant system using high-energy TP-H-3340 ammonium perchlorate and aluminum with a hydroxyl terminated polybutadiene (HTPB) binder.

The Star-48B motor is available in propellant off-loaded configurations. The motor is currently qualified for propellant weights ranging from 2010 kg (4430 lb) to 1739 kg (3833 lb) in the maximum off-loaded condition. The amount of off-load is a function of spacecraft weight and the velocity requirements of the mission.

Star-37FM SRM. The Star-37FM motor has a diameter of 934.7 mm (36.8 in.) and an overall length of 1689.1 mm (66.5 in.). The motor has two integral flanges, the lower for attachment to the third-stage spin table conical motor adapter and the upper for attachment to the 3724C PAF. The motor consists of a carbon-phenolic exit cone, 6AL-4V titanium high-strength motor case, silica-filled rubber insulation system, and a propellant system using high-energy TP-H-3340 ammonium perchlorate and aluminum with an HTPB binder.

The Star-37FM motor is also available in propellant off-loaded configurations. The motor is currently qualified for propellant weights ranging from 1066 kg (2350 lb) to 1025 kg (2260 lb) in the maximum off-loaded condition. The amount of off-load is a function of spacecraft weight and the velocity requirements of the mission.

1.2.4 Payload Attach Fittings

The spacecraft interfaces with the launch vehicle by means of a payload attach fitting including the payload separation system. The Delta II launch system offers a wide selection of standard and modifiable PAFs to accommodate customer needs. Payload separation systems typically incorporated on the PAFs include clampband separation or explosive attach-bolt systems as required. PAFs and separation systems are discussed in greater detail in Section 5.

The customer has the option to provide their own PAF and separation system to interface directly with the Delta II second stage or modified Boeing PAF.

1.2.5 Dual- and Multiple-Manifest Capability

The Delta II dual-manifest system provides significant cost reduction with payload autonomy similar to a dedicated launch, via the use of a flight-proven dual-payload attach fitting (DPAF). There are two versions available, the standard-height DPAF and the reduced-height DPAF, which allow for a combination of payload sizes, and enables the launch of two spacecraft with a combined mass of up to 2268 kg (5000 lb) to LEO in a 7920-10 vehicle configuration. Both spacecraft are fully encapsulated on standard PAF separation interfaces within independent payload bays. Standard access doors are provided for each payload. The DPAF is discussed in more detail in Section 5.

Multiple-manifest is accommodated by using a dispenser that provides the interface between the launch vehicle and the payloads, while supporting spacecraft deployment in orbit as well. Depending on customer requirements, Boeing currently offers two designs, a platform dispenser and post dispenser, both of which have been flight proven with a 100% success rate. Contact BLS for additional information.

1.2.6 Payload Fairings (PLF)

The Delta II launch vehicle offers the user a choice of three fairings: a 2.9-m (9.5-ft)-dia skin-and-stringer center section fairing (bisector), and two versions of a 3-m (10-ft)-dia (bisector) composite fairing with two different lengths. Each of these fairings (Figure 1-5) can be used on either two- stage or three-stage missions. The stretched-length 3.0-m (10-ft) composite fairing, designated 10L, offers more payload volume. The stretched 3-m (10-ft)-dia composite fairing has a reshaped nose cone and a cylindrical section 0.91 m (3 ft) longer than the standard 3-m (10-ft) version.

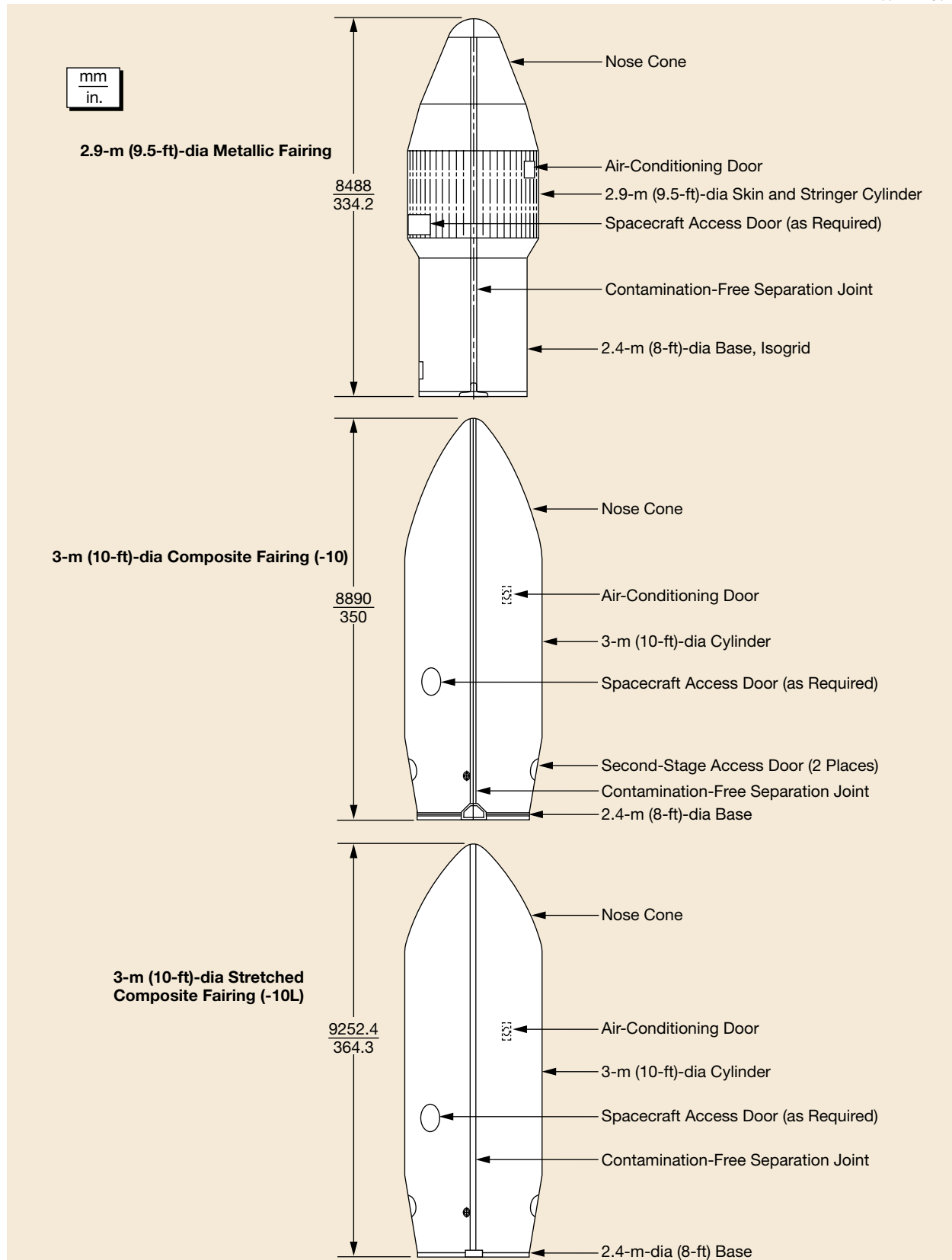


Figure 1-5. Delta II Payload Fairing Options

The fairings incorporate interior acoustic absorption blankets as well as flight-proven contamination-free separation joints. The Delta Program supplies mission-specific modifications to the fairings as required by the customer. These include access doors, additional acoustic blankets, and RF windows. Fairings are discussed in greater detail in Section 3.

1.2.7 Guidance, Control, and Navigation System

Since 1995, the Delta II launch system has used a modernized avionics suite with single-fault-tolerant guidance system, including the redundant inertial flight control assembly (RIFCA) with its integrated software design. RIFCA uses ring laser gyros and accelerometers to provide redundant three-axis rate and acceleration data. In addition to RIFCA, both the first- and second-stage avionics include a power and control (P&C) box to support power distribution, an ordnance box to issue ordnance commands, an electronics package (E-pack) that interfaces with RIFCA through the P&C box to control the vehicle attitude, and a pulse code modulated (PCM) telemetry system that provides vehicle system performance data.

The RIFCA contains the basic control logic that processes rate and accelerometer data to form the proportional and discrete control output commands needed to drive the control actuators and cold gas jet control thrusters; the RIFCA sequences the remainder of the vehicle commands using on-board timing.

Position and velocity data are explicitly computed to derive guidance steering commands. Early in flight, a load relief guidance mode turns the vehicle into the wind to reduce the angle of attack, thus relieving structural loads and increasing control ability. After dynamic pressure decay, the guidance system corrects trajectory dispersions caused by load relief and directs the vehicle to the nominal end-of-stage orbit. Space vehicle separation in the desired transfer orbit is accomplished by applying time adjustments to the nominal sequence.

1.3 VEHICLE AXES/ATTITUDE DEFINITIONS

The vehicle axes are defined in Figure 1-6. The vehicle centerline is the vehicle longitudinal axis. Axis II is on the downrange side of the vehicle, and axis IV is on the uprange side. The vehicle pitches about axes I/III. Positive pitch rotates the nose of the vehicle up, toward axis IV. The vehicle yaws about axes II/IV. Positive yaw rotates the vehicle's nose to the right, toward axis I. The vehicle rolls about the centerline. Positive roll is clockwise rotation, looking forward (i.e., from axis I toward II). The third-stage spin table also spins in the same direction (i.e., the positive roll direction).

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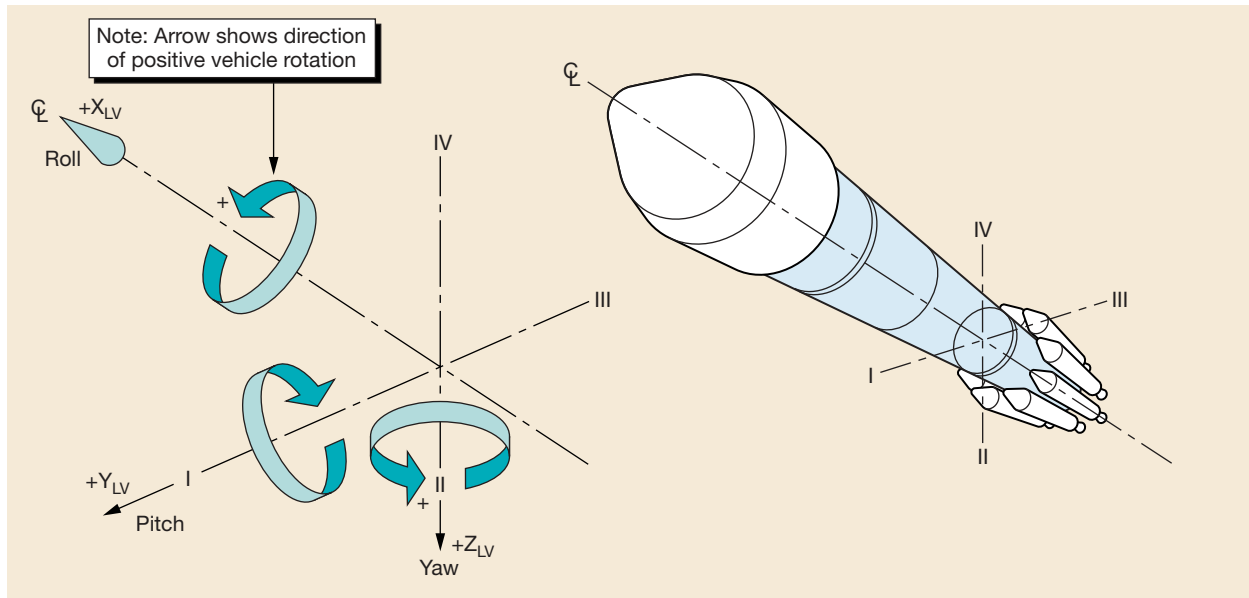


Figure 1-6. Vehicle Axes

1.4 LAUNCH VEHICLE INSIGNIA

Delta II users may request a mission-specific insignia to be placed on their launch vehicles. The user is invited to submit the proposed design to the Delta Program Office no later than 9 months prior to launch for review and approval. Maximum insignia size is 2.4 by 2.4 m (8 by 8 ft). Following approval, the Delta Program Office will have the flight insignia prepared and placed on the uprange side of the launch vehicle.

Section 2 GENERAL PERFORMANCE CAPABILITY

The Delta II can accommodate a wide range of spacecraft requirements. The following sections detail specific performance capabilities of Delta II launch vehicle configurations from the eastern and western ranges. In addition to the capabilities shown herein, our mission designers can provide innovative performance trades to meet the particular requirements of our customers.

2.1 LAUNCH SITES

Depending on the specific mission requirement and range safety restrictions, the Delta II 7300- 7400- and 7900-series vehicle can be launched from either the eastern range (ER) or western range (WR) launch site (7900H series can only use the ER SLC-17B launch pad at present).

- **Eastern Launch Site.** The ER launch site for Delta II is Space Launch Complex 17 (SLC-17), launch pads A and B, at the Cape Canaveral Air Force Station (CCAFS) in Florida. This site can accommodate flight azimuths in the range of 65 to 110 deg, with 95 deg being the most commonly flown.
- **Western Launch Site.** The WR launch site for Delta II is Space Launch Complex 2 (SLC-2) at Vandenberg Air Force Base (VAFB) in California. Flight azimuths in the range of 190 to 225 deg are currently approved by the 30th Space Wing, with 196 deg being the most commonly flown.

2.2 MISSION PROFILES

Typical profiles for both two- and three-stage missions are shown in Figures 2-1 and 2-2.

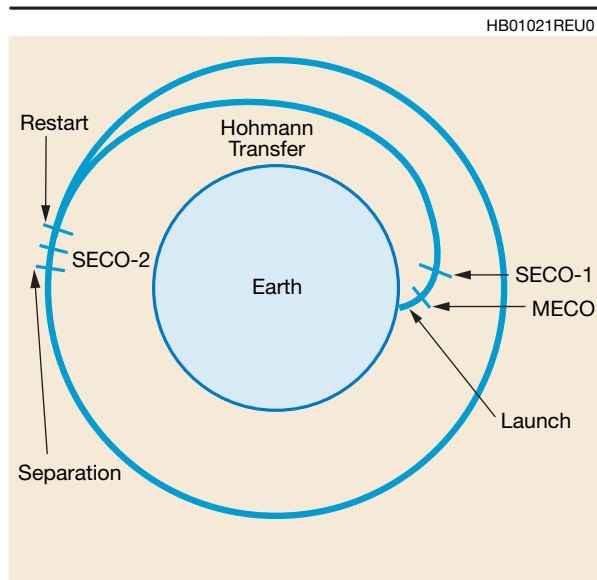


Figure 2-1. Typical Two-Stage Mission Profile

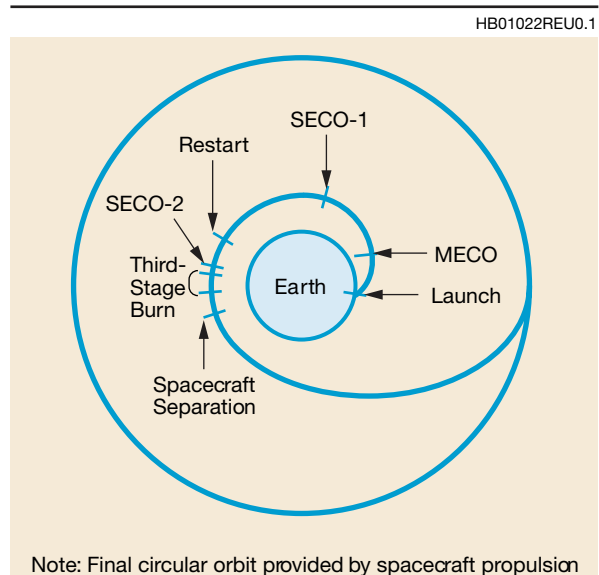


Figure 2-2. Typical Three-Stage Mission Profile

2.2.1 First-Stage Flight Profiles

- **7300-Series Vehicle.** In launches from both the ER and WR, the first-stage RS-27A engine and three strap-on solid-rocket motors (SRMs) are ignited on the ground at liftoff. The solids are then jettisoned following burnout. The main engine continues to burn until main engine cutoff (MECO) at propellant depletion.
- **7400-Series Vehicle.** For customers who require slightly more performance, the 7400-series vehicle provides approximately 15% greater performance than the 7300-series vehicle for a low-Earth orbit (LEO). The first-stage RS-27A engine and four strap-on solid-rocket motors are ignited on the ground at liftoff. The remaining vehicle sequence of events is approximately the same as with the 7300 series vehicle.
- **7900-Series Vehicle.** The 7900-series vehicle provides the customer with a payload capability of approximately 55% greater than the 7400-series vehicle to LEO. In launches from both the ER and WR, the first-stage RS-27A main engine and six of the nine strap-on solid-rocket motors are ignited on the ground at liftoff. Following burnout of these six SRMs, the remaining three are ignited. The six spent SRMs are then automatically jettisoned in sets of three after vehicle and range safety constraints have been satisfied. Jettisoning of the second set occurs 1 sec after the first set. The remaining three SRMs are jettisoned approximately 3 sec after burnout. The main engine then continues to burn until MECO.
- **7900H-Series Vehicle.** At present, the 7900H-series Delta II is available in both two- and three-stage configurations for launches from the ER launch site only. The Delta 7920H (with nine graphite epoxy (GEM-46) strap-on solid-rocket motors) provides approximately 20% greater performance than the 7900 series to LEO. With the exception of the solid-rocket motor burn durations (which are approximately 14 sec longer), the vehicle sequence of events is approximately the same as with the 7900-series vehicle.

2.2.2 Second-Stage and Third-Stage Flight Profiles

The remainder of the two- and three-stage mission profiles for the 7300-, 7400-, and 7900-series vehicles are almost identical. Eight seconds after MECO, the first stage separates and is expended; the second stage ignites five seconds later. Payload fairing (PLF) separation occurs early in the second-stage flight, after an acceptable free-molecular-heating rate has been reached.

In the typical two-stage mission (Figure 2-1), the second stage burns for approximately 340 to 420 sec, at which time second-stage engine cutoff (SECO 1) occurs. The vehicle then follows a Hohmann transfer trajectory to the desired LEO altitude. Near apogee of the transfer orbit, the second stage is restarted and completes its burn to inject the payload into the desired orbit. Separation takes place approximately 250 sec after second-stage engine cutoff (SECO 2) once the spacecraft's separation attitude requirements have been satisfied.

The typical three-stage mission to geosynchronous transfer orbit (GTO), shown in Figure 2-2, uses the first burn of the second stage to place the payload into a 185-km (100-nmi) circular parking orbit inclined at 28.7 deg. The vehicle then coasts to a position near the equator where the second stage is restarted. Following SECO-2, the third stage is spun up, separated, and burned to establish GTO. At apogee altitude, the spacecraft provides the final propulsion to circularize the orbit to GEO. Depending on mission requirements and spacecraft mass, some inclination may be removed or apogee altitude raised to optimize satellite lifetime.

After payload separation, the Delta second stage is restarted to deplete any remaining propellants (depletion burn) and/or to move the stage to a safe distance from the spacecraft (evasive burn).

If required, the multiple restart capability of the Delta II second stage provides the customer with a wide range of orbit flexibility and launch of multiple spacecraft.

Typical flight sequences using LEO missions for the 7320/7420 vehicles from eastern and western launch sites are shown in Figures 2-3 and 2-4, while sequences for a GTO mission using the 7925/7925H vehicles and a polar mission using the 7920 vehicle are shown in Figures 2-5 and 2-6. Typical event times for both two- and three-stage versions of the 7300-, 7400-, 7900-, and 7900H-series configurations from the eastern and western launch sites are presented in Tables 2-1 and 2-2.

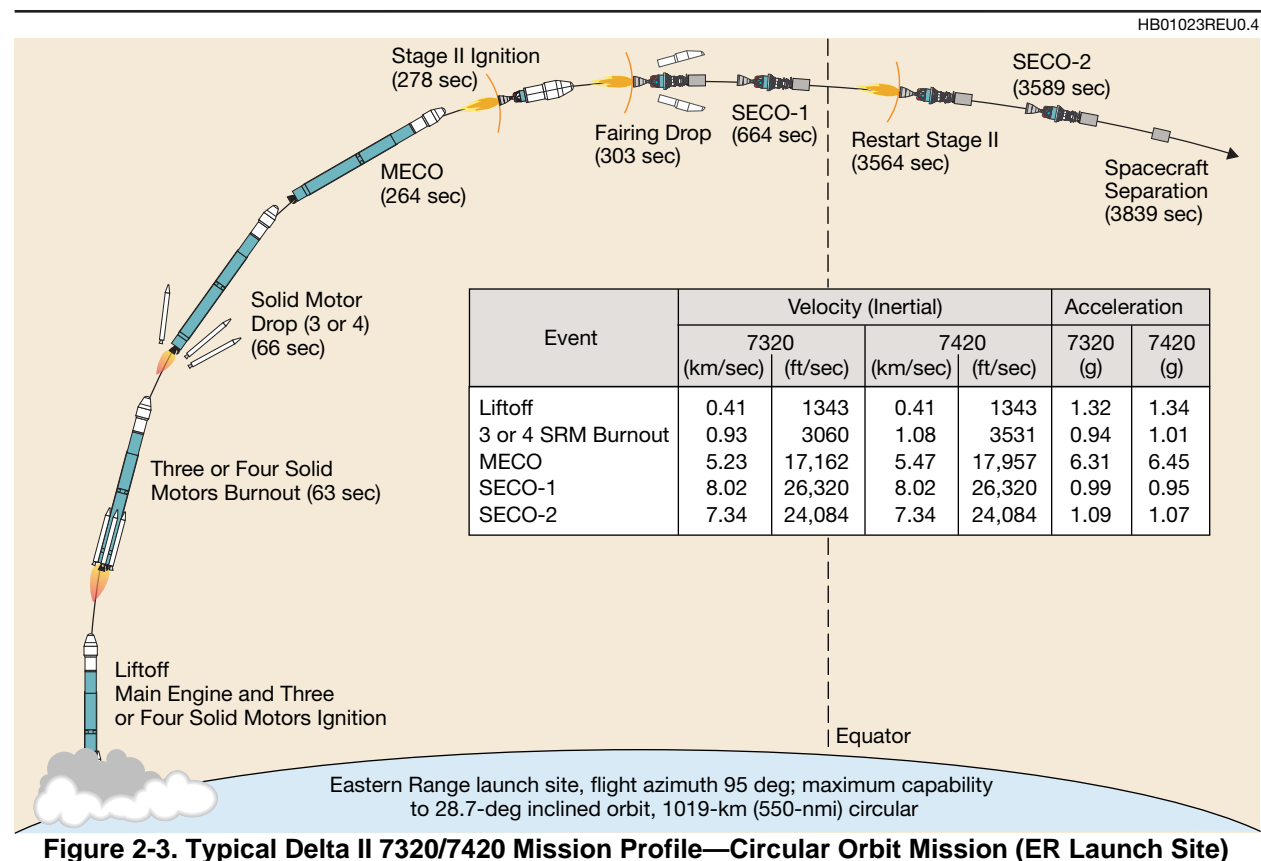


Figure 2-3. Typical Delta II 7320/7420 Mission Profile—Circular Orbit Mission (ER Launch Site)

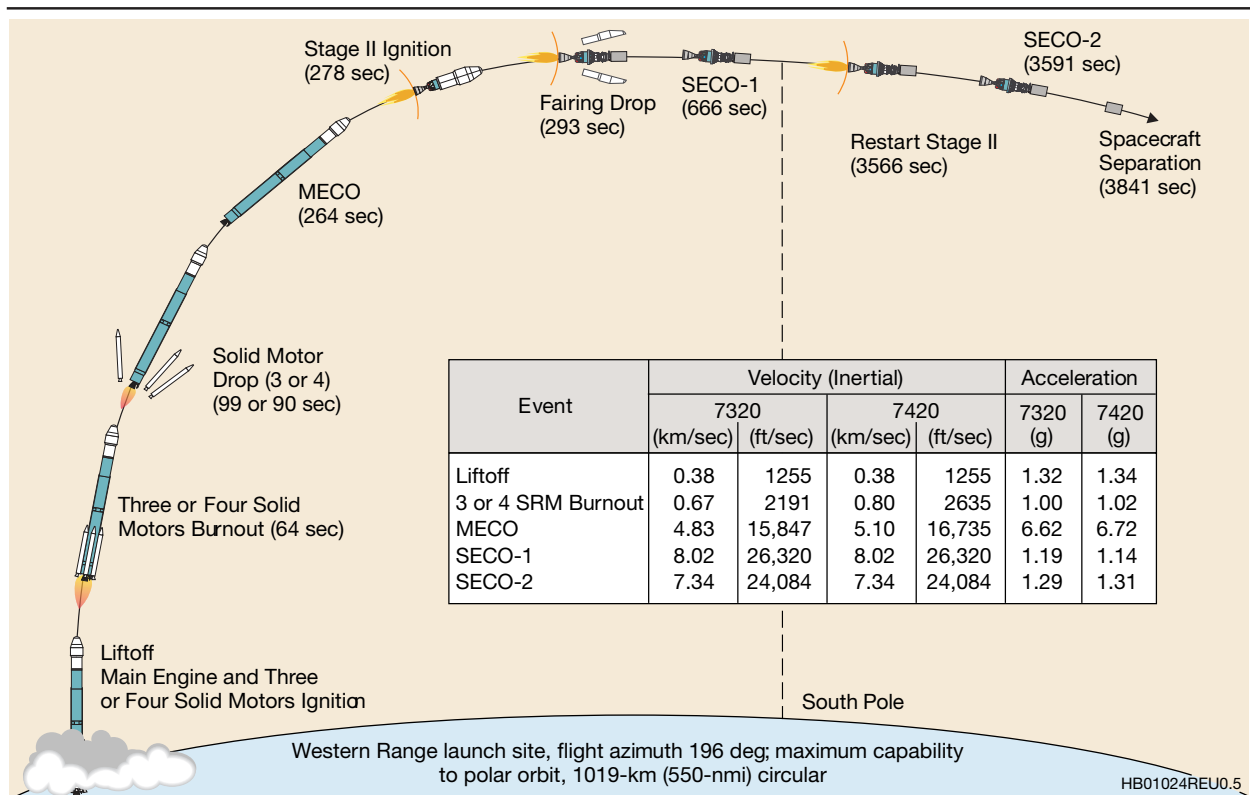


Figure 2-4. Typical Delta II 7320/7420 Mission Profile—Polar Orbit Mission (WR Launch Site)

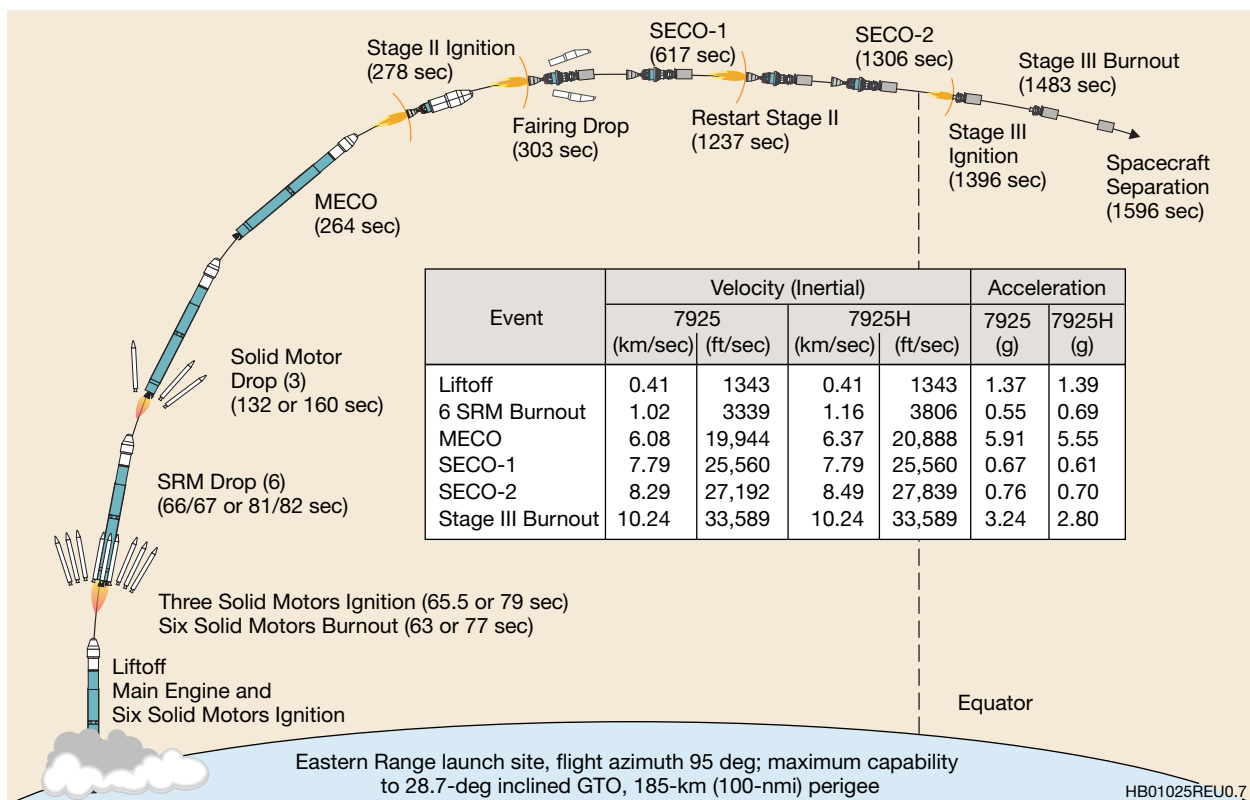


Figure 2-5. Typical Delta II 7925/7925H Mission Profile—GTO Mission (ER Launch Site)

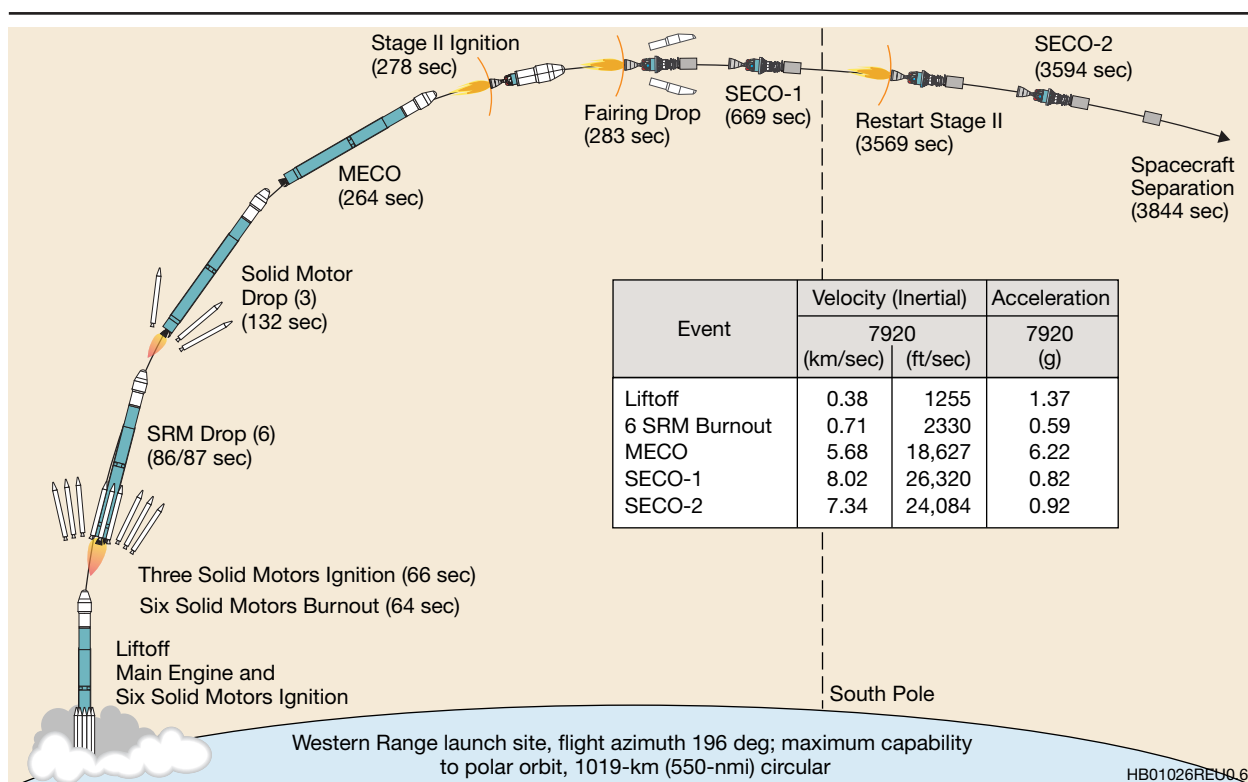


Figure 2-6. Typical Delta II 7920 Mission Profile—Polar Mission (WR Launch Site)

Table 2-1. Delta II Typical Eastern Launch Site Event Times*

Event	Vehicle Configuration					
	7320/7420	7920/7920H	7325/7425	7925/7925H	7326/7426	7926/7926H
First Stage						
Main engine ignition	T + 0	T + 0	T + 0	T + 0	T + 0	T + 0
Solid-motor ignition (3, 4, or 6)	T + 0	T + 0	T + 0	T + 0	T + 0	T + 0
Solid-motor burnout (3, 4, or 6)	T + 63	T + 63 or 77	T + 63	T + 63 or 77	T + 63	T + 63 or 77
Solid-motor ignition (3)	N/A	T + 66 or 79	N/A	T + 66 or 79	N/A	T + 66 or 79
Solid-motor separation (3, 4, or 3/3)	T + 66	T + 66/67 or 81/82	T + 66	T + 66/67 or 81/82	T + 66	T + 66/67 or 81/82
Solid-motor burnout (3)	N/A	T + 129 or 157	N/A	T + 129 or 157	N/A	T + 129 or 157
Solid-motor separation (3)	N/A	T + 132 or 160	N/A	T + 132 or 160	N/A	T + 132 or 160
MECO (M)	T + 264	T + 264	T + 264	T + 264	T + 264	T + 264
Second Stage						
Activate Stage I/II separation bolts	M + 8	M + 8	M + 8	M + 8	M + 8	M + 8
Stage II ignition	M + 13.5	M + 13.5	M + 13.5	M + 13.5	M + 13.5	M + 13.5
Fairing separation	M + 39	M + 39	M + 39	M + 39	M + 39	M + 39
SECO (S1)	M + 400	M + 408	M + 415	M + 353	M + 390	M + 340
Stage II engine restart	S1 + 2900	S1 + 2900	S1 + 610	S1 + 620	S1 + 610	S1 + 620
SECO (S2)	S1 + 2925	S1 + 2925	S1 + 631	S1 + 689	S1 + 650	S1 + 710
Third Stage						
Activate spin rockets, start Stage III sequencer	N/A	N/A	S2 + 50	S2 + 50	S2 + 50	S2 + 50
Separate Stage II	N/A	N/A	S2 + 53	S2 + 53	S2 + 53	S2 + 53
Stage III ignition	N/A	N/A	S2 + 90	S2 + 90	S2 + 90	S2 + 90
Stage III burnout	N/A	N/A	S2 + 177	S2 + 177	S2 + 155	S2 + 155
Spacecraft						
Spacecraft separation	S2 + 250	S2 + 250	S2 + 290	S2 + 290	S2 + 225	S2 + 225

*All times shown in seconds

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Table 2-2. Delta II Typical Western Launch Site Event Times*

Event	Vehicle Configuration					
	7320/7420	7920	7425	7925	7326/7426	7926
First Stage						
Main engine ignition	T + 0	T + 0	T + 0 sec	T + 0	T + 0 sec	T + 0
Solid-motor ignition (3, 4, or 6)	T + 0	T + 0	T + 0	T + 0	T + 0	T + 0
Solid-motor burnout (3, 4, or 6)	T + 64	T + 64	T + 64	T + 64	T + 64	T + 64
Solid-motor ignition (3)	N/A	T + 66	N/A	T + 66	N/A	T + 66
Solid-motor separation (3, 4, or 3/3)	T + 99 or 83	T + 86/87	T + 83	T + 86/87	T + 99 or 83	T + 86/87
Solid-motor burnout (3)	N/A	T + 129	N/A	T + 129	N/A	T + 129
Solid-motor separation (3)	N/A	T + 132	N/A	T + 132	N/A	T + 132
MECO (M)	T + 264	T + 264	T + 264	T + 264	T + 264	T + 264
Second Stage						
Activate Stage I/II separation bolts	M + 8	M + 8	M + 8	M + 8	M + 8	M + 8
Stage II ignition	M + 13.5	M + 13.5	M + 13.5	M + 13.5	M + 13.5	M + 13.5
Fairing separation	M + 29	M + 19	M + 29	M + 19	M + 29	M + 19
SECO (S1)	M + 402	M + 405	M + 415	M + 356	M + 390	M + 340
Stage II engine restart	S1 + 2900	S1 + 2900	S1 + 610	S1 + 620	S1 + 610	S1 + 620
SECO (S2)	S1 + 2925	S1 + 2925	S1 + 631	S1 + 689	S1 + 650	S1 + 710
Third Stage						
Activate spin rockets, start Stage III sequencer	N/A	N/A	S2 + 50	S2 + 50	S2 + 50	S2 + 50
Separate Stage II	N/A	N/A	S2 + 53	S2 + 53	S2 + 53	S2 + 53
Stage III ignition	N/A	N/A	S2 + 90	S2 + 90	S2 + 90	S2 + 90
Stage III burnout	N/A	N/A	S2 + 177	S2 + 177	S2 + 155	S2 + 155
Spacecraft						
Spacecraft separation	S2 + 250	S2 + 250	S2 + 290	S2 + 290	S2 + 225	S2 + 225

*All times shown in seconds

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2.3 PERFORMANCE CAPABILITY

This section presents a summary of the performance capabilities of the 7300, 7400, and 7900 launch vehicles, from the ER and WR launch sites, while that of the 7900H-series vehicle from the ER only.

The performance estimates that follow are computed based on the following assumptions:

- A. Nominal propulsion system and weight models were used on all stages.
- B. The first stage is burned to propellant depletion.
- C. Extended nozzle airlit GEMs are incorporated (only airlit GEMs have extended nozzles).
- D. Second-stage propellant reserve is sufficient to provide a 99.7% probability of command shutdown (PCS) by the guidance system.
- E. PLF separation occurs at a time when free-molecular heating rate is equal to or less than 1135 W/m² (0.1 Btu/ft²-sec).
- F. Perigee velocity is the vehicle burnout velocity at 185-km (100-nmi) altitude and zero-deg flight path angle.
- G. Initial flight azimuth is 95 deg from the eastern launch site and 196 deg from the western launch site.
- H. For two-stage missions, a 6306 payload attach fitting (PAF) is assumed for the 7300/7400-series, and a 6915 PAF is assumed for the 7900/7900H-series. It should be noted that alternate PAFs and the dual-payload attach fitting (DPAF) can be used but will affect the payload mass capability shown in the respective figures.

- I. For three-stage missions using a Star-48B third stage, a 3712A PAF with standard nutation control system (NCS) and yo-weight tumble system is assumed. It should be noted that other three-stage PAFs can be used but will affect the three-stage payload mass capability. If the spacecraft requires a lower spin rate, an NCS with a yo-yo-weight despin system would add approximately 4.5 kg (10 lbm) to the standard system.
- J. For three-stage missions using a Star-37FM third stage, a 3724C PAF with a yo-weight tumble system and without an NCS is assumed. If the spacecraft requires a lower spin rate, an NCS with a yo-yo-weight despin system would add approximately 23.1 kg (51 lbm).
- K. Capabilities are shown for standard 2.9-m (9.5-ft), 3.0-m (10-ft), and 3.0-m (10-ft) stretched (7900/7900H-series only) PLFs.

A summary of maximum performance for common two- and three-stage missions is presented in Tables 2-3 and 2-4.

Table 2-3. Two-Stage Mission Capabilities

	Vehicle Designation	Spacecraft mass capabilities					
		LEO ■ CCAFS, i = 28.7 deg ■ 185 km/100 nmi circular		LEO ■ VAFB, i = 90.0 deg ■ 185 km/100 nmi circular		Sun-Synchronous Orbit ■ VAFB, i = 98.7 deg ■ 833 km/450 nmi circular	
		(kg)	(lbm)	(kg)	(lbm)	(kg)	(lbm)
7300-Series Vehicle 2.9-m (9.5-ft) Fairing 3.0-m (10-ft) Fairing	7320-9.5 7320-10	2809 2703	6194 5958	2063 1982	4548 4370	1651 1579	3639 3481
7400-Series Vehicle 2.9-m (9.5-ft) Fairing 3.0-m (10-ft) Fairing	7420-9.5 7420-10	3185 3099	7022 6833	2436 2351	5370 5184	1966 1895	4334 4177
7900-Series Vehicle 2.90-m (9.5-ft) Fairing 3.0-m (10-ft) Fairing 3.0L-m (10L-ft) Fairing	7920-9.5 7920-10 7920-10L	5030 4844 4805	11089 10680 10593	3755 3639 3599	8277 8022 7934	3123 3017 2984	6886 6651 6578
7900H-Series Vehicle 2.9-m (9.5-ft) Fairing 3.0-m (10-ft) Fairing 3.0L-m (10L-ft) Fairing	7920H-9.5 7920H-10 7920H-10L	6097 5959 5899	13443 13137 13005	Currently Not Available From WR Launch Site			
Note: 7300/7400 baseline uses a 6306 payload attach fitting with a mass of 47.6 kg (105 lbm) 7900/7900H baseline uses a 6915 payload attach fitting with a mass of 93.0 kg (205 lbm)							

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Table 2-4. Three-Stage Mission Capabilities

Spacecraft mass capabilities							
		Geosynchronous Transfer Orbit (GTO) ■ CCAFS, i = 28.7 deg ■ 185 x 35,786 km/100 x 19,323 nmi		Interplanetary Transfer Orbit ■ CCAFS, i = 28.7 deg ■ C3 = 0.4 km2/sec2		Molniya Orbit ■ VAFB, i = 63.4 deg ■ 370 x 40,094 km/ 200 x 21,649 nmi	
		(kg)	(lbm)	(kg)	(lbm)	(kg)	(lbm)
7300-Series Vehicle							
■ Star-48B Third Stage							
– 2.9-m (9.5-ft) Fairing	7325-9.5	N/A*	N/A*	N/A*	N/A*	N/A*	N/A*
– 3.0-m (10-ft) Fairing	7325-10	N/A*	N/A*	N/A*	N/A*	N/A*	N/A*
■ Star-37FM Third Stage		934	2058				
– 2.9-m (9.5-ft) Fairing	7326-9.5	898	1979	629	1387	636	1402
– 3.0-m (10-ft) Fairing	7326-10			604	1331	611	1347
7400-Series Vehicle							
■ Star-48B Third Stage							
– 2.9-m (9.5-ft) Fairing	7425-9.5	1110	2446	804	1772	N/A*	N/A*
– 3.0-m (10-ft) Fairing	7425-10	1073	2366	779	1717	N/A*	N/A*
■ Star-37FM Third Stage							
– 2.9-m (9.5-ft) Fairing	7426-9.5	1058	2331	711	1568	734	1618
– 3.0-m (10-ft) Fairing	7426-10	1029	2269	692	1525	709	1564
7900-Series Vehicle							
■ Star-48B Third Stage							
– 2.9-m (9.5-ft) Fairing	7925-9.5	1819	4011	1265	2789	1177	2594
– 3.0-m (10-ft) Fairing	7925-10	1747	3852	1211	2670	1143	2520
– 3.0L-m (10L-ft) Fairing	7925-10L	1739	3833	1207	2660	1131	2493
■ Star-37FM Third Stage							
– 2.9-m (9.5-ft) Fairing	7926	1660	3659	1121	2471	1056	2328
– 3.0-m (10-ft) Fairing	7926-10	1581	3486	1065	2348	1022	2253
– 3.0L-m (10L-ft) Fairing	7926-10L	1578	3480	1064	2346	1012	2230
7900H-Series Vehicle						Currently Not Available From WR Launch Site	
■ Star-48B Third Stage							
– 2.9-m (9.5-ft) Fairing	7925H-9.5	2171	4787	1508	3325		
– 3.0-m (10-ft) Fairing	7925H-10	2123	4680	1474	3249		
– 3.0L-m (10L-ft) Fairing	7925H-10L	2102	4635	1460	3219		
■ Star-37FM Third Stage							
– 2.9-m (9.5-ft) Fairing	7926H	1981	4368	1333	2939		
– 3.0-m (10-ft) Fairing	7926H-10	1934	4264	1302	2870		
– 3.0L-m (10L-ft) Fairing	7926H-10L	1916	4224	1290	2844		
Note: Star-48B uses a 3712A payload attach fitting with a mass of 45.4 kg (100 lbm) Star-37FM uses a 3724C payload attach fitting with a mass of 56.7 kg (125 lbm) *Not available, exceeds maximum allowable Star-48B motor offload capability.							

002202.6

The second stage can be flown to propellant depletion shutdown (PDS) if the mission desires a slightly higher performance capability. Depending on the launch vehicle configuration, performance increases from 2% to 4% can be achieved.

The performance capability for any given mission depends upon quantitative analysis of all known mission requirements and range safety restrictions. The allowable payload mass should be coordinated with the Delta Program Office as early as possible in the basic mission planning. Preliminary error analysis, performance optimization, and trade-off studies will be performed, as required, to arrive at an early commitment of allowable payload mass for each specific mission.

EASTERN RANGE LAUNCH SITE

■ Two-Stage Performance

- Two-stage circular orbit altitude, 7320/7420 Vehicle (Figure 2-7).
- Two-stage circular orbit altitude, 7920/7920H Vehicle (Figure 2-8).
- Two-stage apogee altitude, 7320/7420 Vehicle (Figure 2-9).
- Two-stage apogee altitude, 7920/7920H Vehicle (Figure 2-10).
- Two-stage perigee velocity, 7320/7420 Vehicle (Figure 2-11).
- Two-stage perigee velocity, 7920/7920H Vehicle (Figure 2-12).

■ Three-Stage Performance

- Three-stage GTO inclination, 732X/742X Vehicle (Figure 2-13).
- Three-stage GTO inclination, 792X/792XH Vehicle (Figure 2-14).
- Three-stage launch energy capability, 732X/742X Vehicle (Figure 2-15).
- Three-stage launch energy capability, 792X/792XH Vehicle (Figure 2-16).
- Three-stage apogee altitude, 732X/742X Vehicle (Figure 2-17).
- Three-stage apogee altitude, 792X/792XH Vehicle (Figure 2-18).
- Three-stage perigee velocity, 732X/742X Vehicle (Figure 2-19).
- Three-stage perigee velocity, 792X/792XH Vehicle (Figure 2-20).

WESTERN RANGE LAUNCH SITE

■ Two-Stage Performance.

- Two-stage circular orbit altitude, 7320/7420 Vehicle (Figure 2-21).
- Two-stage circular orbit altitude, 7920 Vehicle (Figure 2-22).
- Two-stage sun-synchronous orbit, 7320/7420 Vehicle (Figure 2-23).
- Two-stage sun-synchronous orbit, 7920 Vehicle (Figure 2-24).
- Two-stage apogee altitude, 7320/7420 Vehicle (Figure 2-25).
- Two-stage apogee altitude, 7920 Vehicle (Figure 2-26).
- Two-stage perigee velocity, 7320/7420 Vehicle (Figure 2-27).
- Two-stage perigee velocity, 7920 Vehicle (Figure 2-28).

■ Three-Stage Performance.

- Three-stage apogee altitude, 7326/7426 Vehicle (Figure 2-29).
- Three-stage apogee altitude, 792X Vehicle (Figure 2-30).
- Three-stage perigee velocity, 732X/742X Vehicle (Figure 2-31).
- Three-stage perigee velocity, 792X Vehicle (Figure 2-32).

HB00961REU0.3

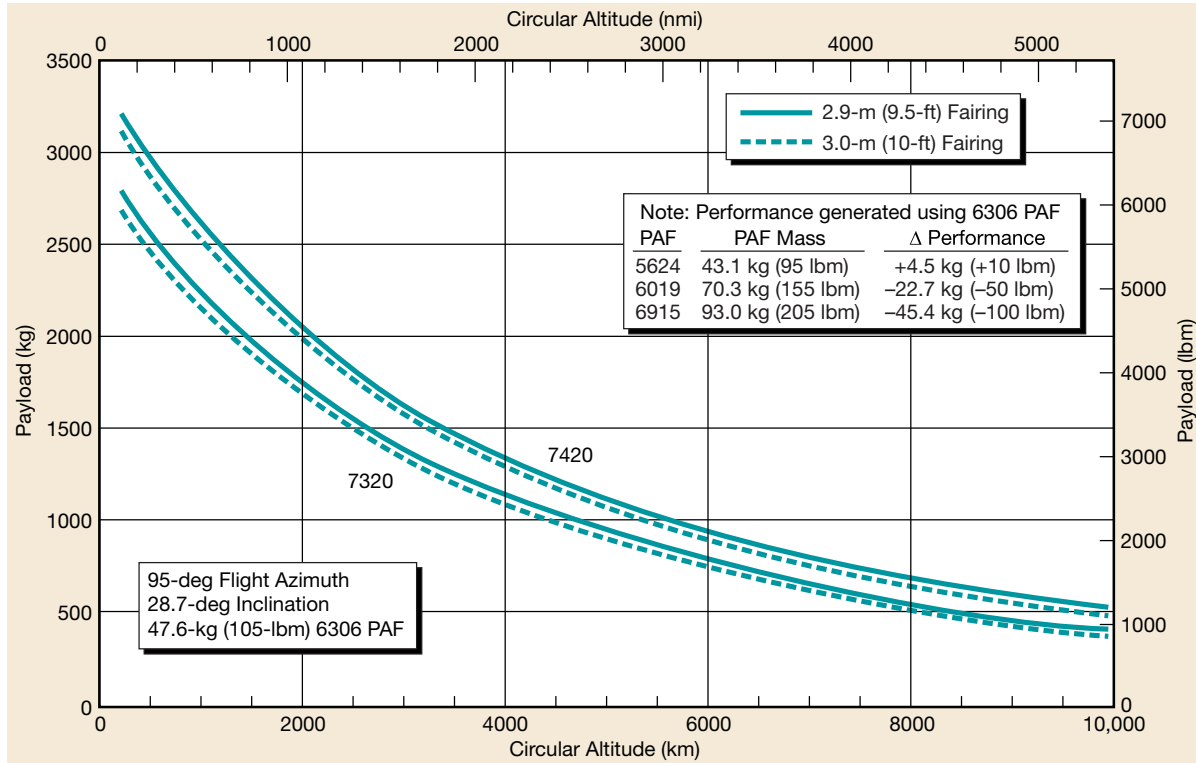


Figure 2-7. Delta II 7320/7420 Vehicle, Two-Stage Circular Orbit Altitude Capability—Eastern Launch Site

HB01010REU0.4

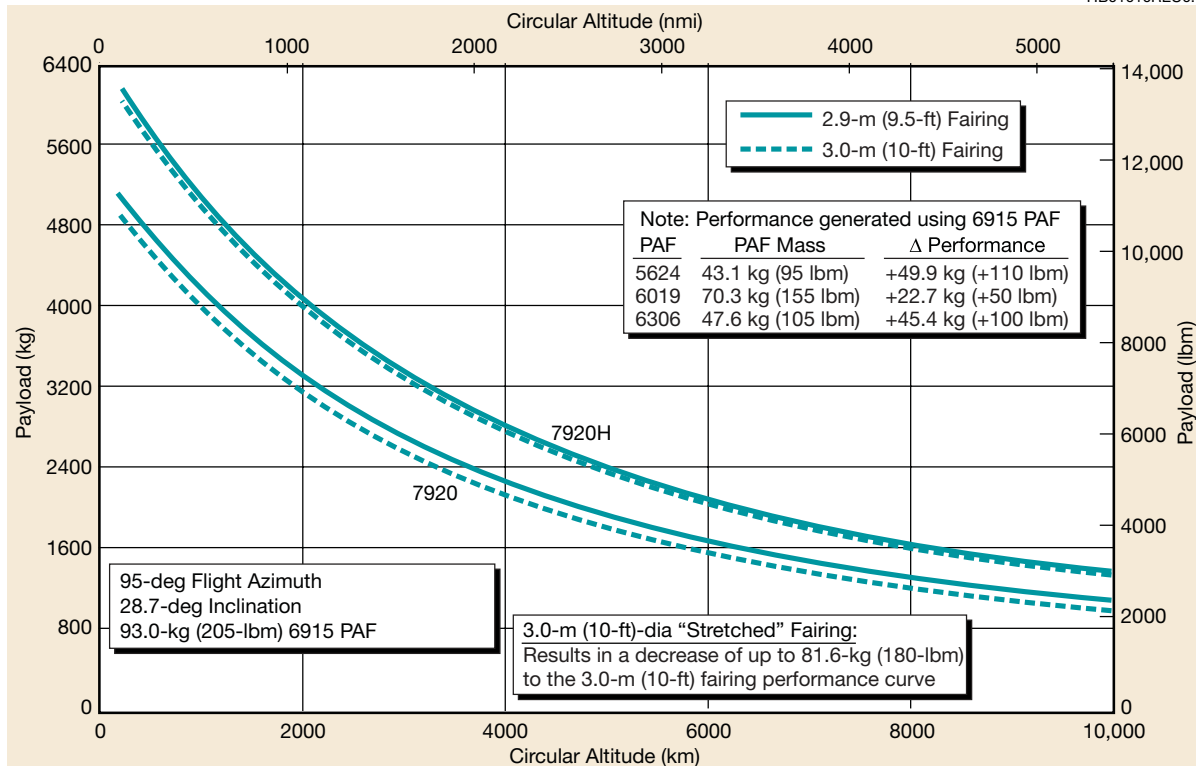


Figure 2-8. Delta II 7920/7920H Vehicle, Two-Stage Circular Orbit Altitude Capability—Eastern Launch Site

HB00960REU0.2

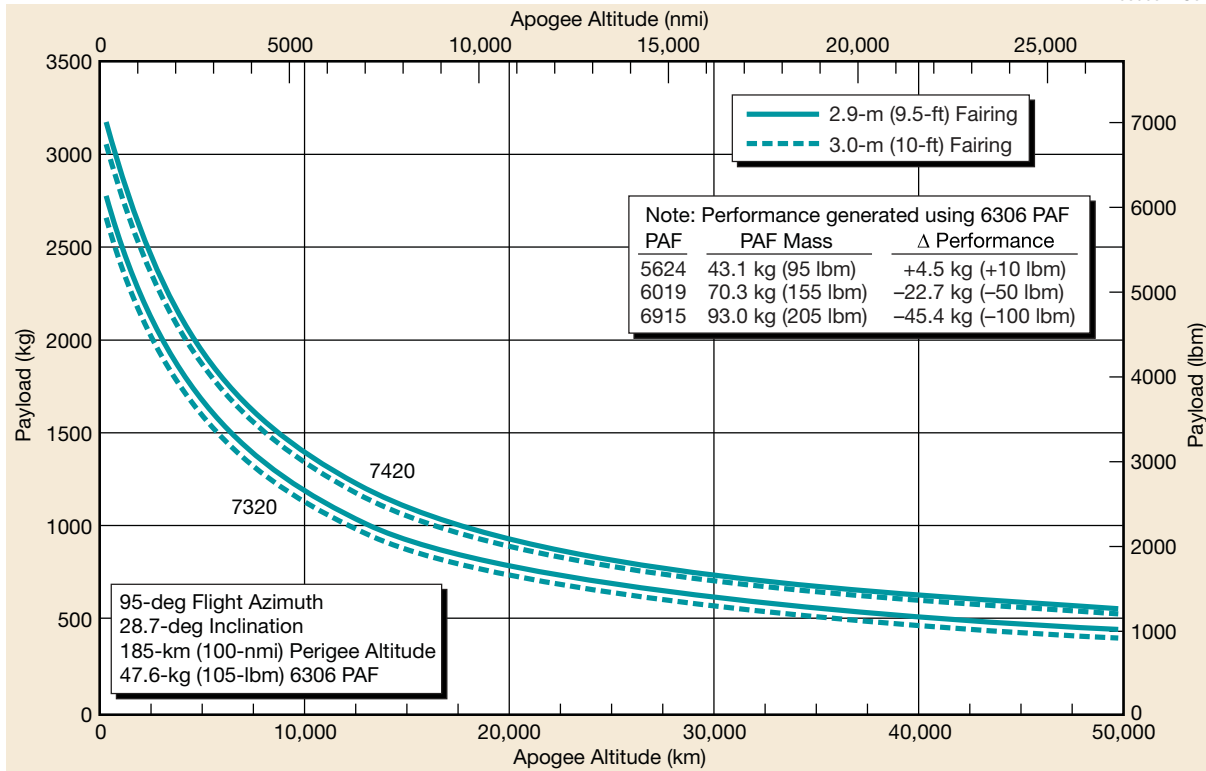


Figure 2-9. Delta II 7320/7420 Vehicle, Two-Stage Apogee Altitude Capability—Eastern Launch Site

HB01009REU0.3

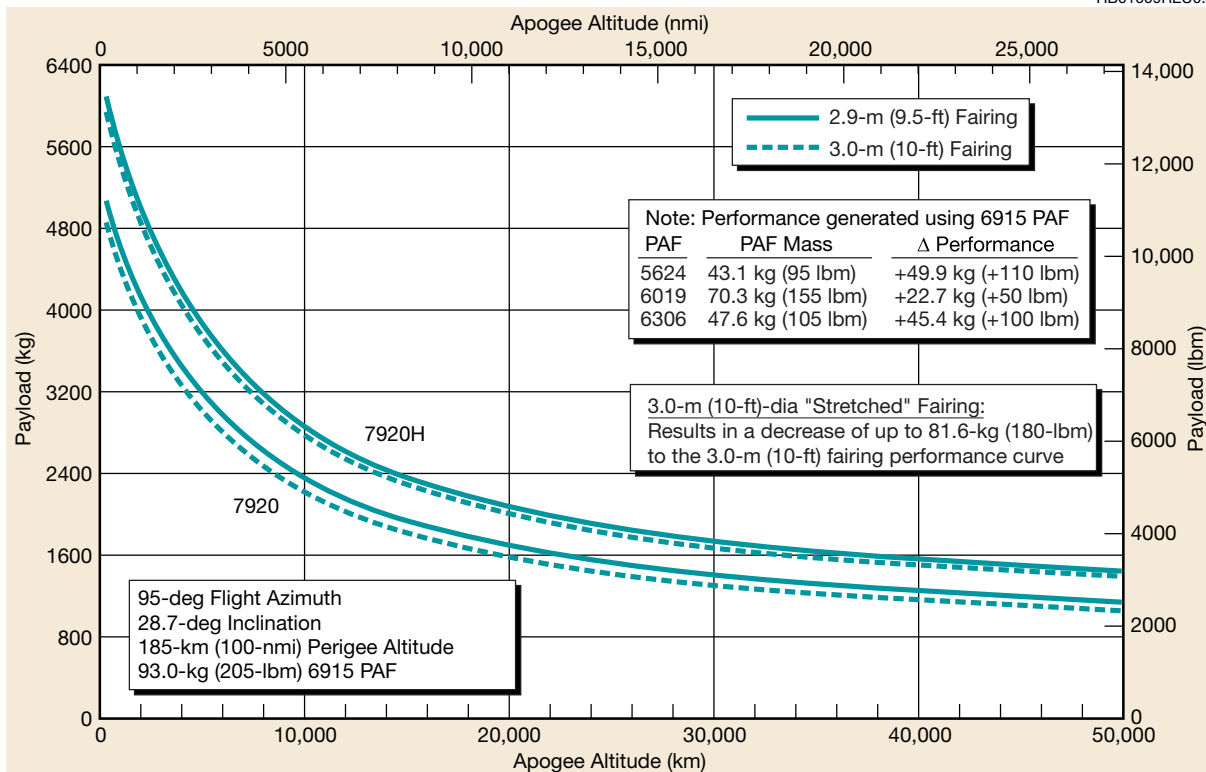


Figure 2-10. Delta II 7920/7920H Vehicle, Two-Stage Apogee Altitude Capability—Eastern Launch Site

HB00959REU0.2

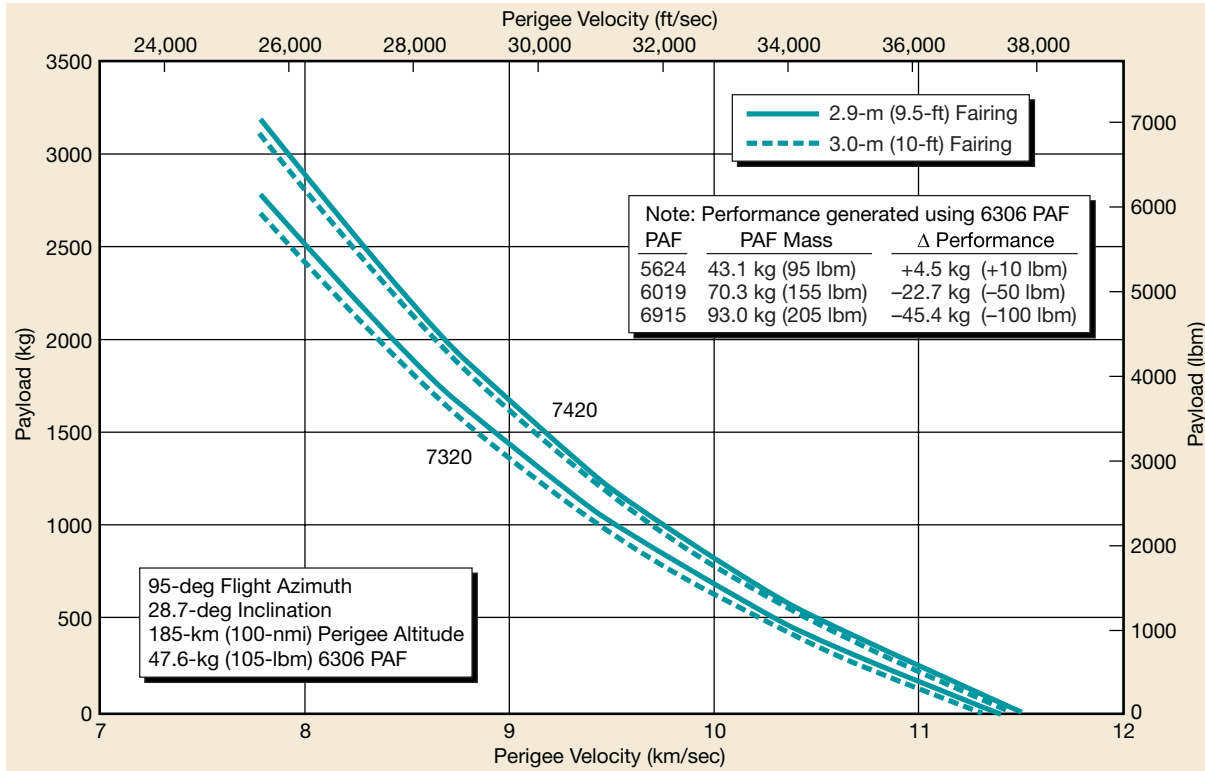


Figure 2-11. Delta II 7320/7420 Vehicle, Two-Stage Perigee Velocity Capability—Eastern Launch Site

HB01008REU0.4

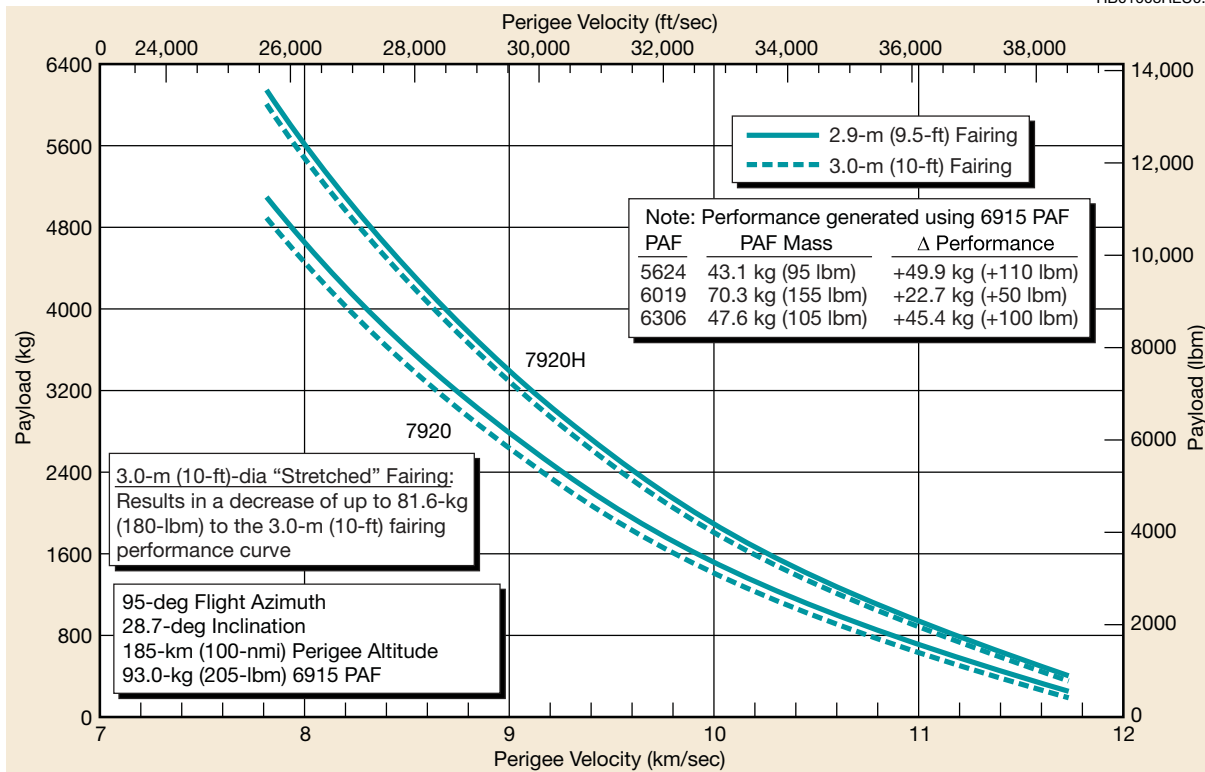


Figure 2-12. Delta II 7920/7920H Vehicle, Two-Stage Perigee Velocity Capability—Eastern Launch Site

HB00964REU0.3

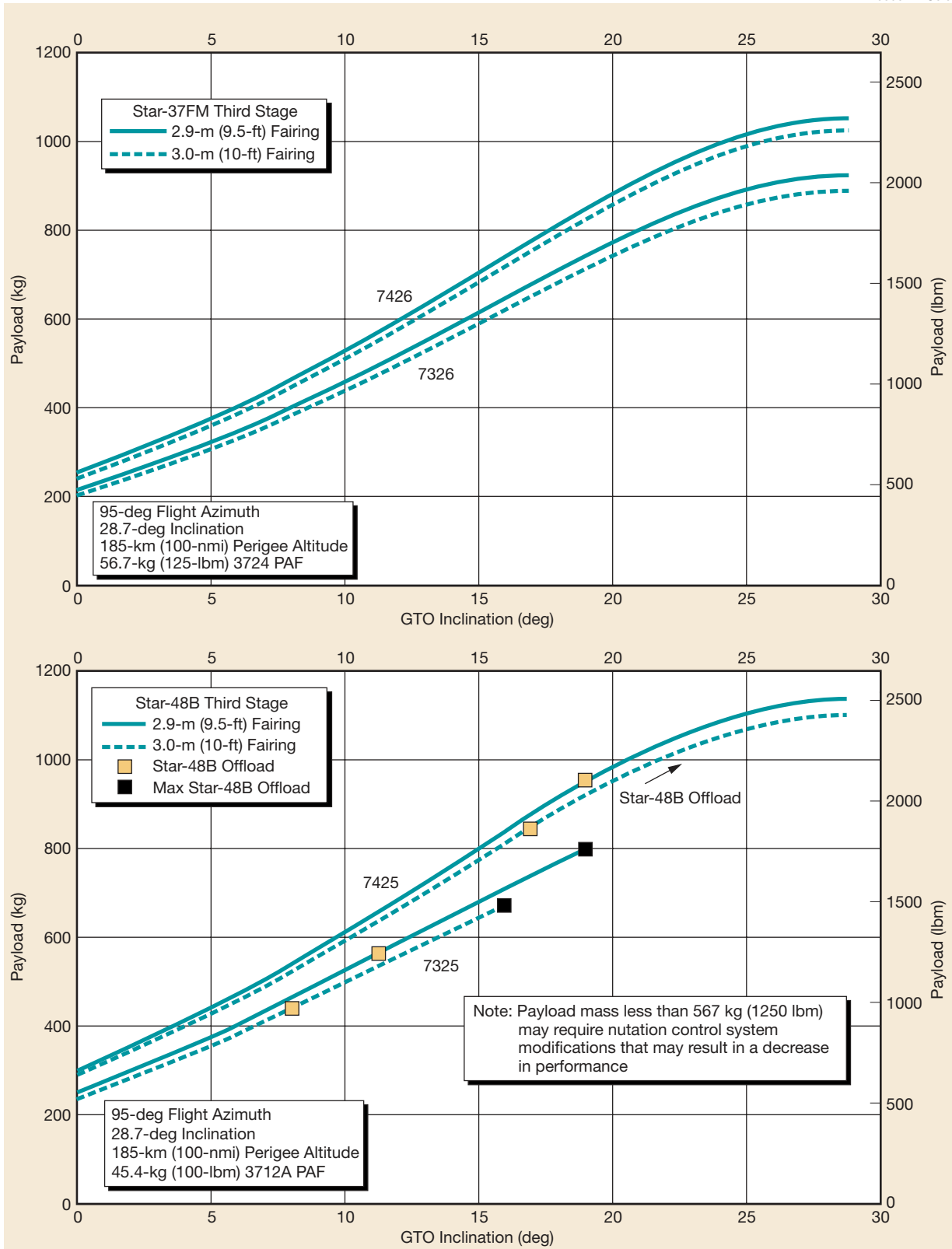


Figure 2-13. Delta II 732X/742X Vehicle, Three-Stage GTO Inclination Capability—Eastern Launch Site

HB01013REU0.5

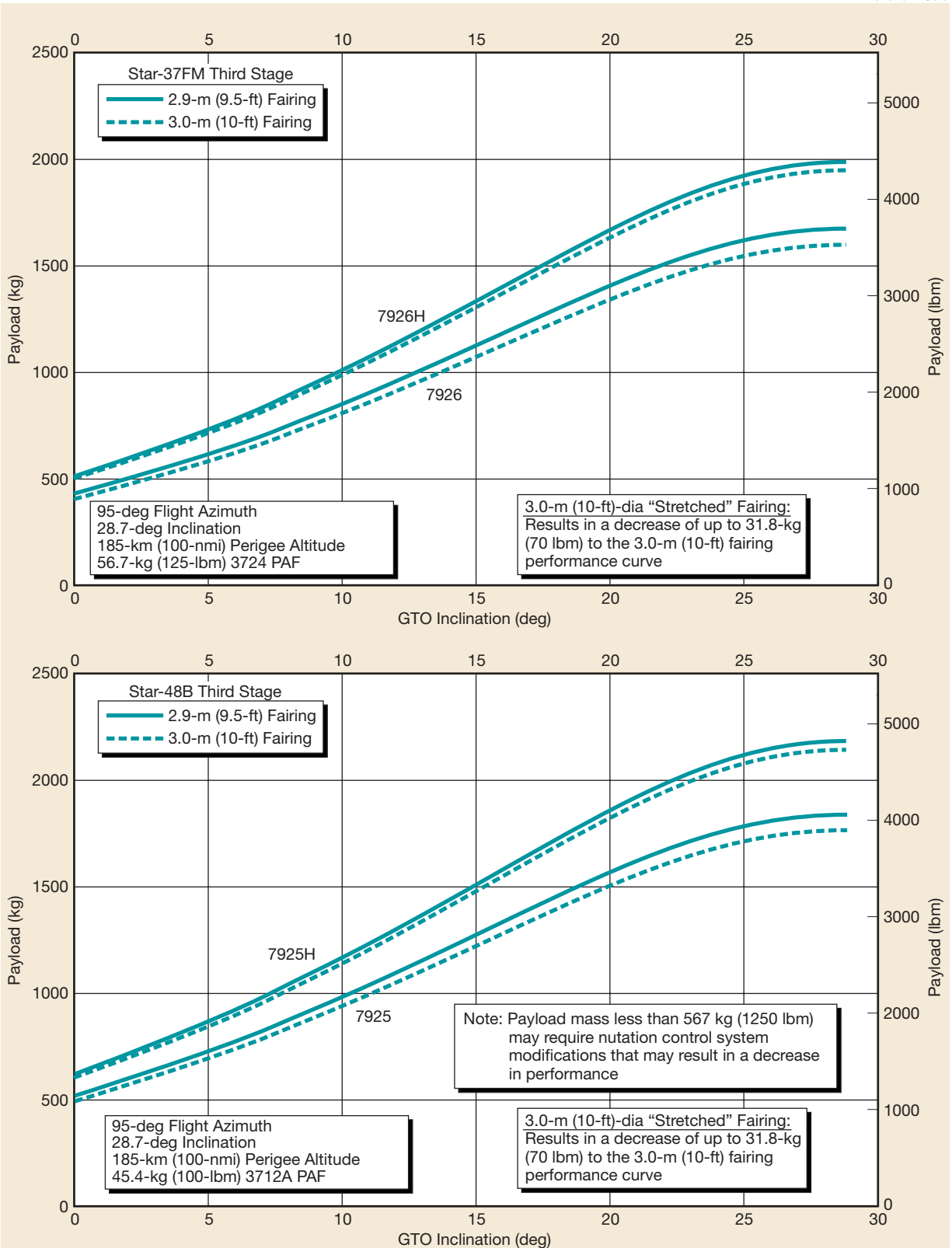


Figure 2-14. Delta II 792X/792XH Vehicle, Three-Stage GTO Inclination Capability—Eastern Launch Site

HB00965REU0.4

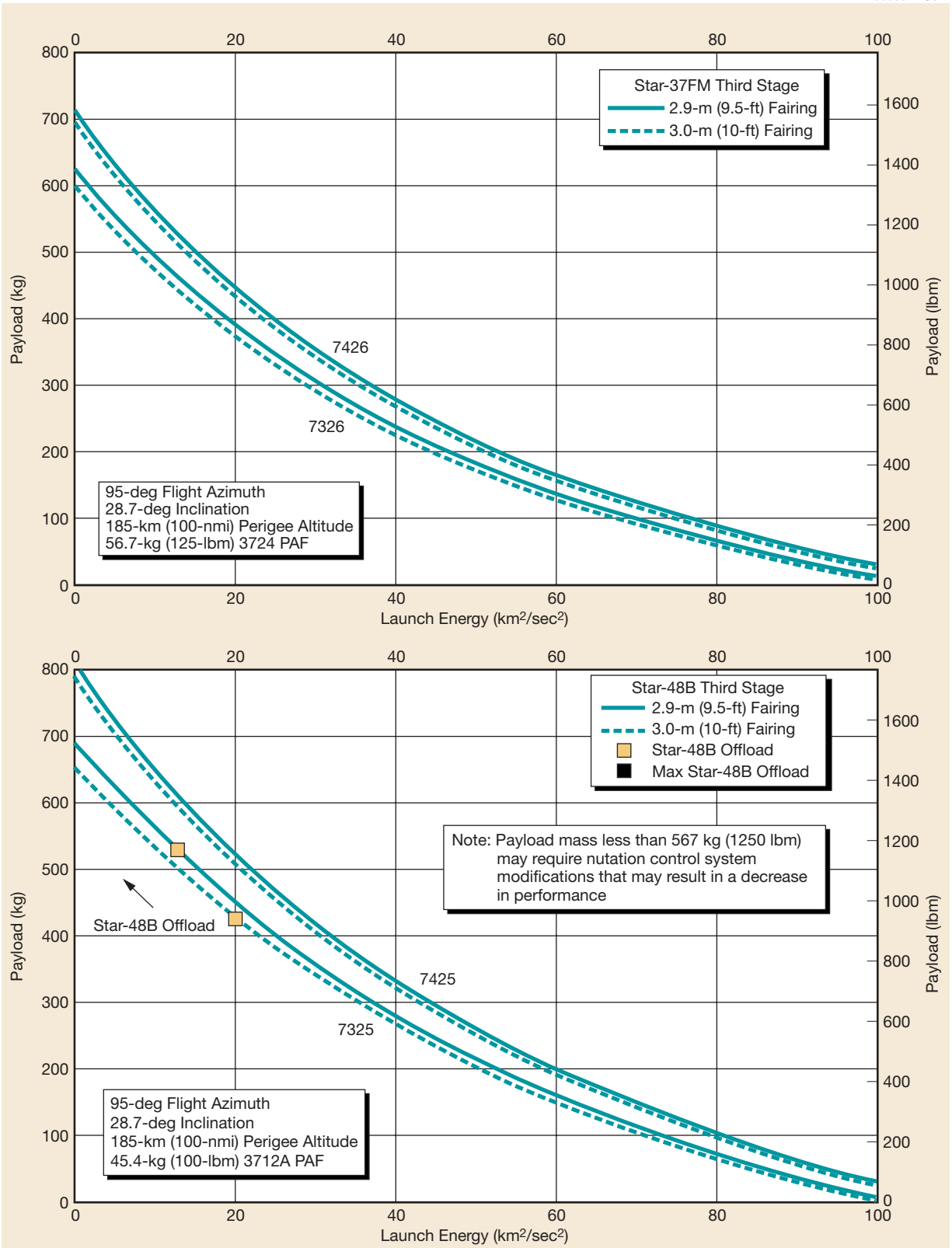


Figure 2-15. Delta II 732X/742X Vehicle, Three-Stage Launch Energy Capability—Eastern Launch Site

HB01014REU0.5

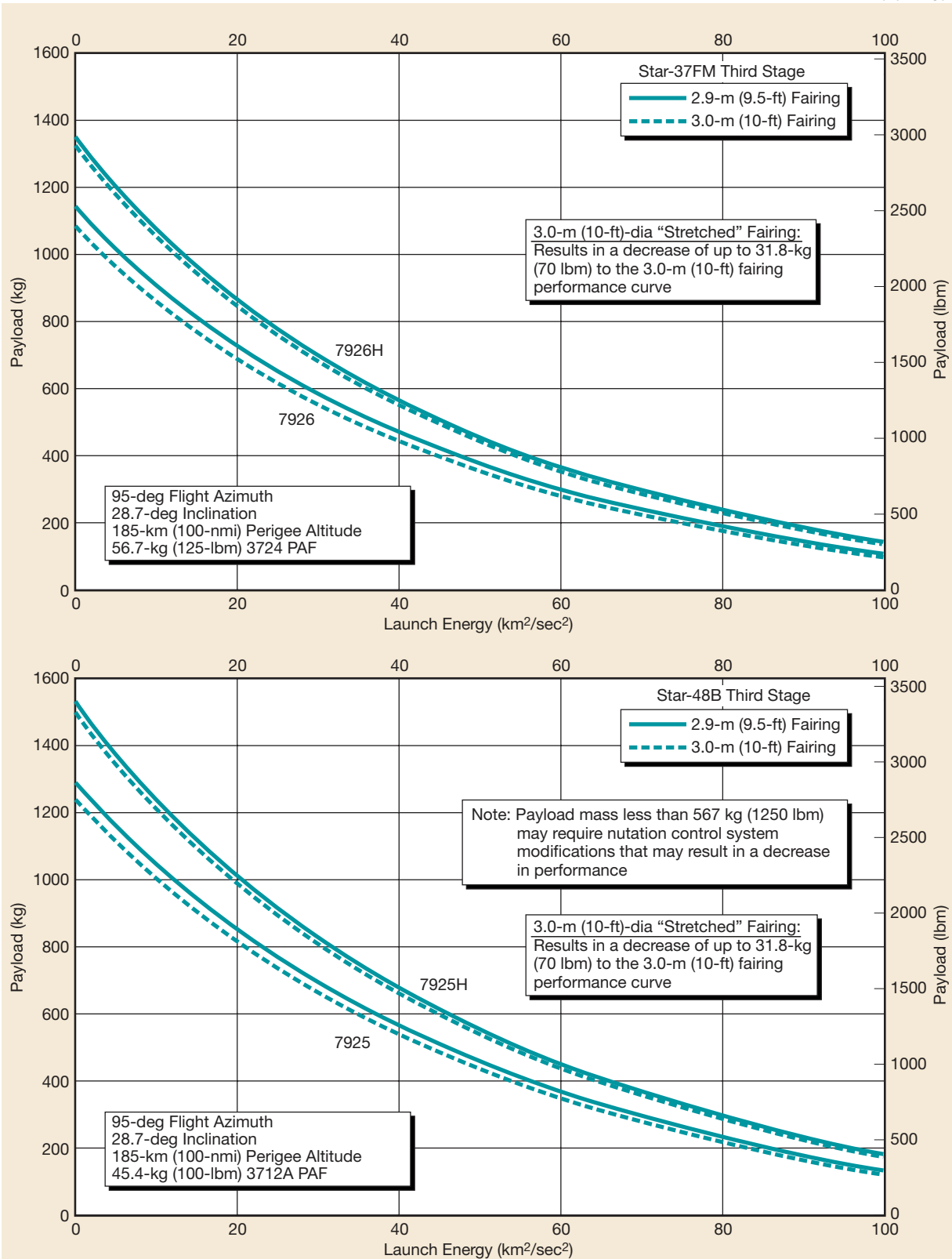
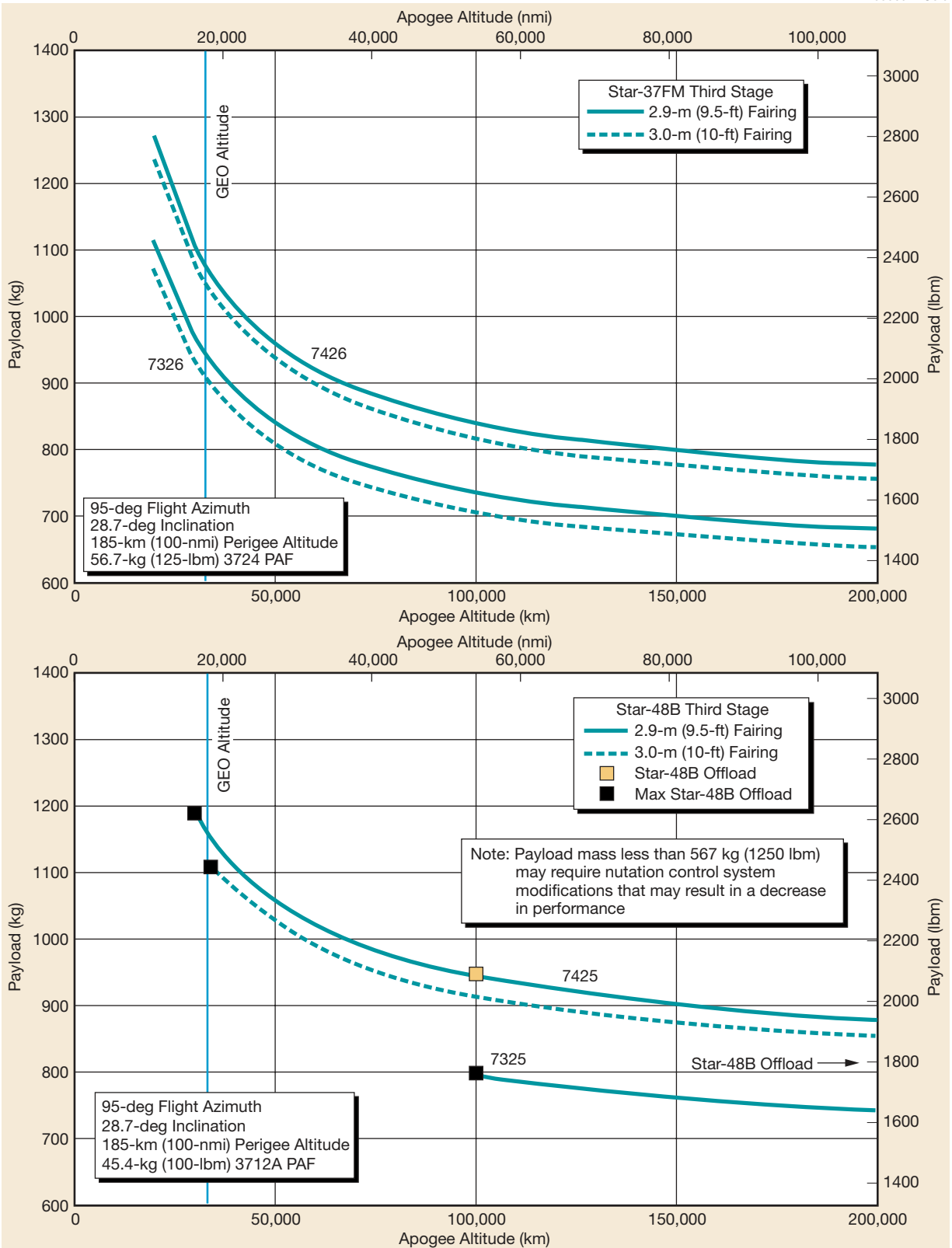


Figure 2-16. Delta II 792X/792XH Vehicle, Three-Stage Launch Energy Capability—Eastern Launch Site

HB00963REU0.3



HB01012REU0.5

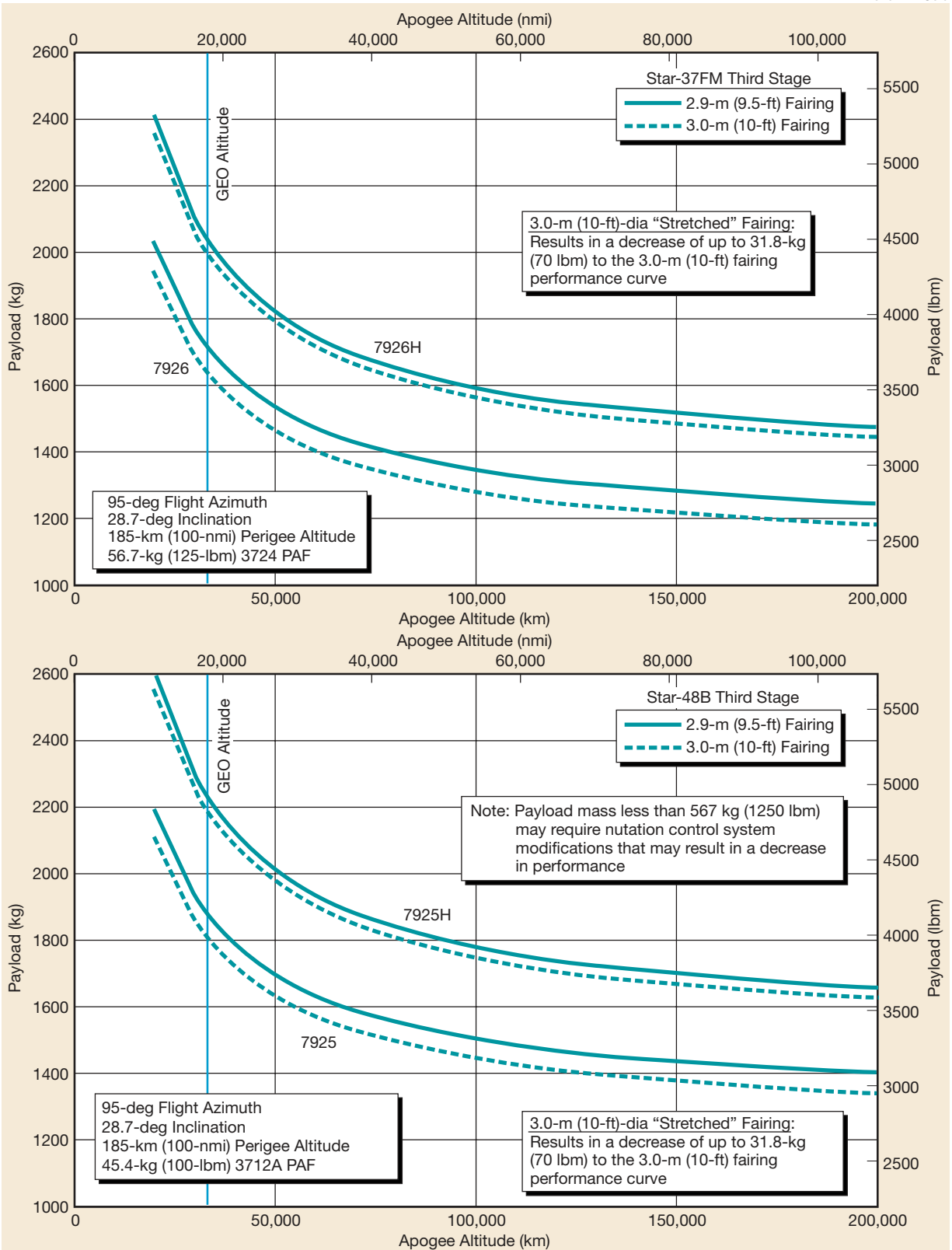


Figure 2-18. Delta II 792X/792XH Vehicle, Three-Stage Apogee Altitude Capability—Eastern Launch Site

HB00962REU0.4

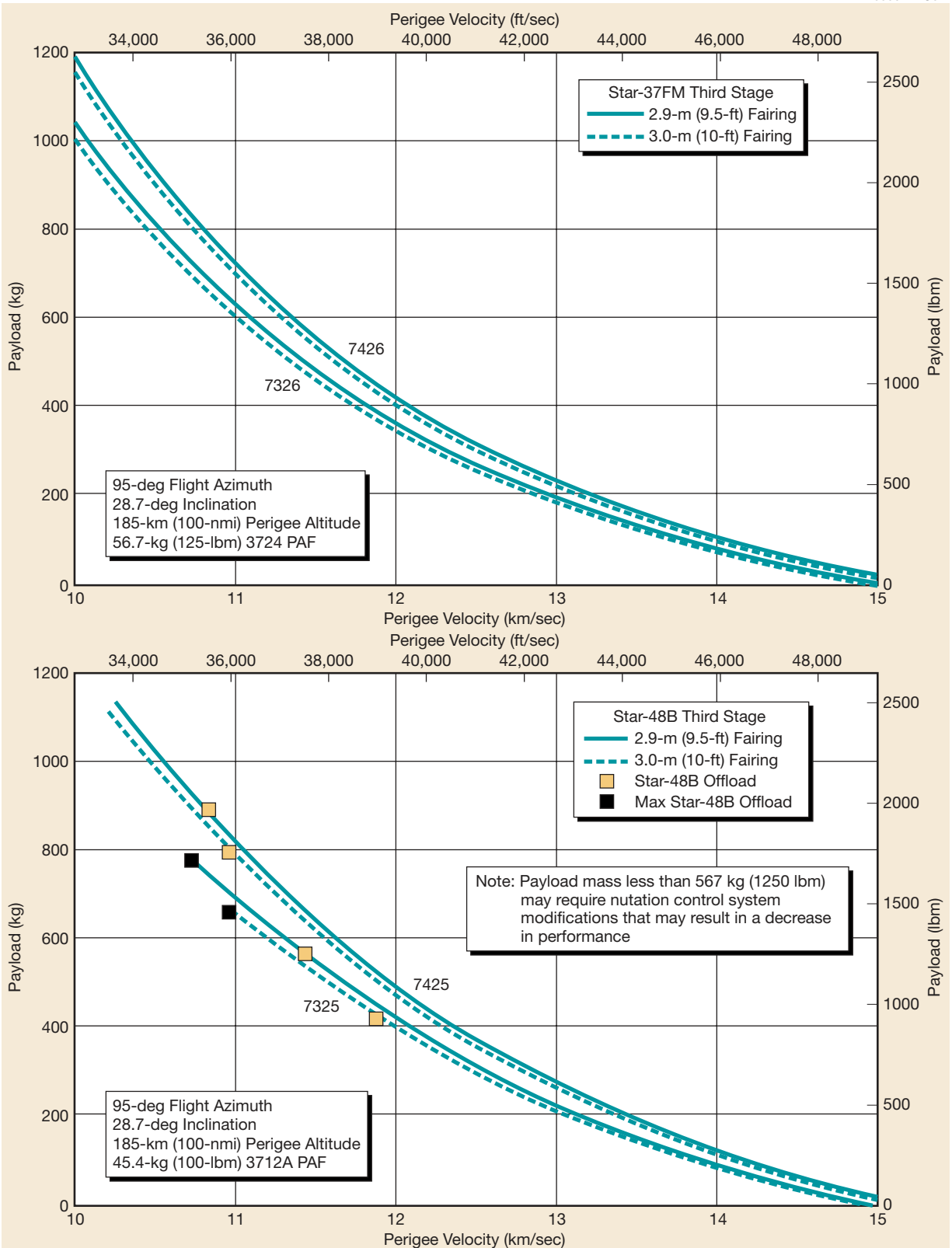


Figure 2-19. Delta II 732X/742X Vehicle, Three-Stage Perigee Velocity Capability—Eastern Launch Range

HB01011REU0.5

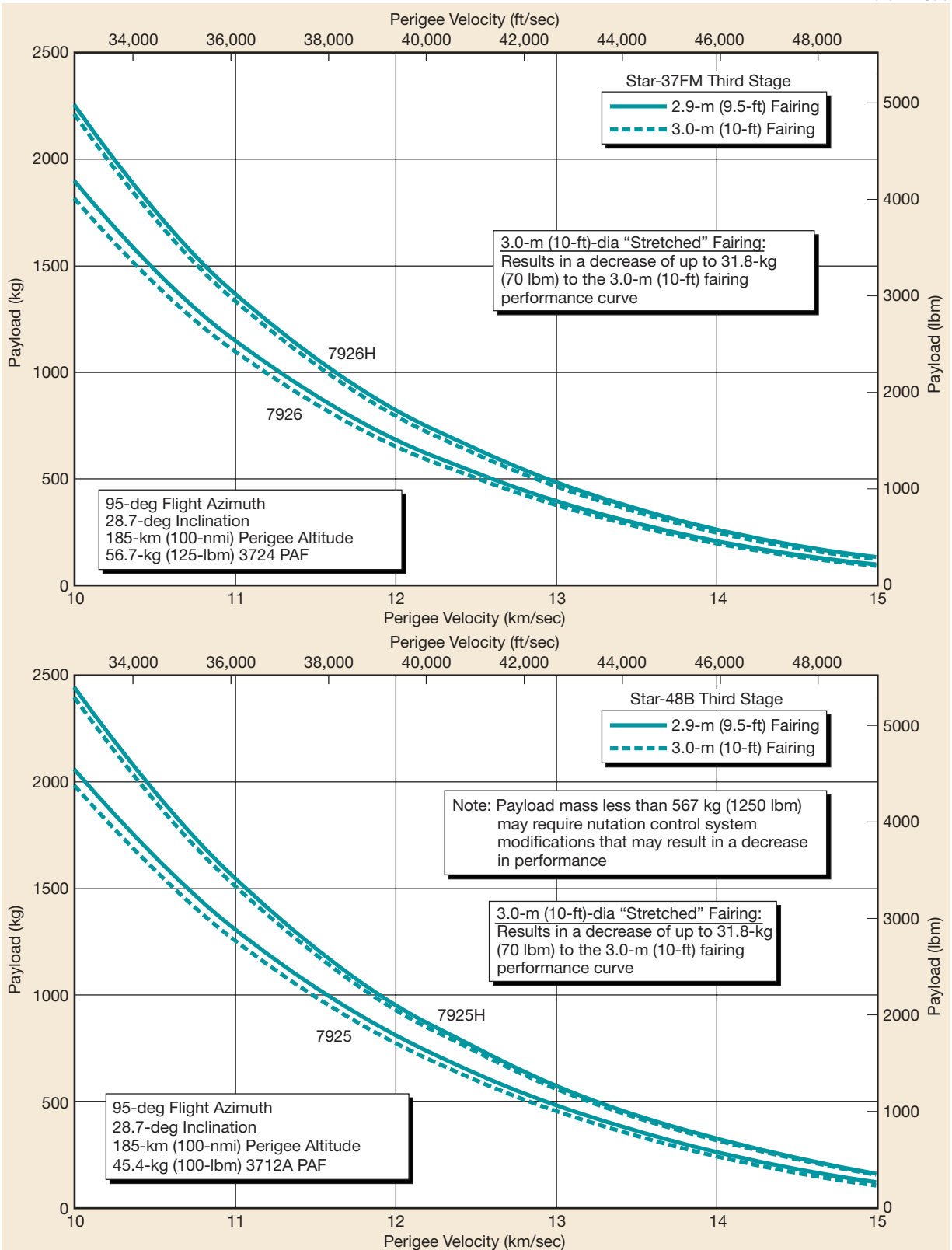


Figure 2-20. Delta II 792X/792XH Vehicle, Three-Stage Perigee Velocity Capability—Eastern Launch Site

HB00968REU0.3

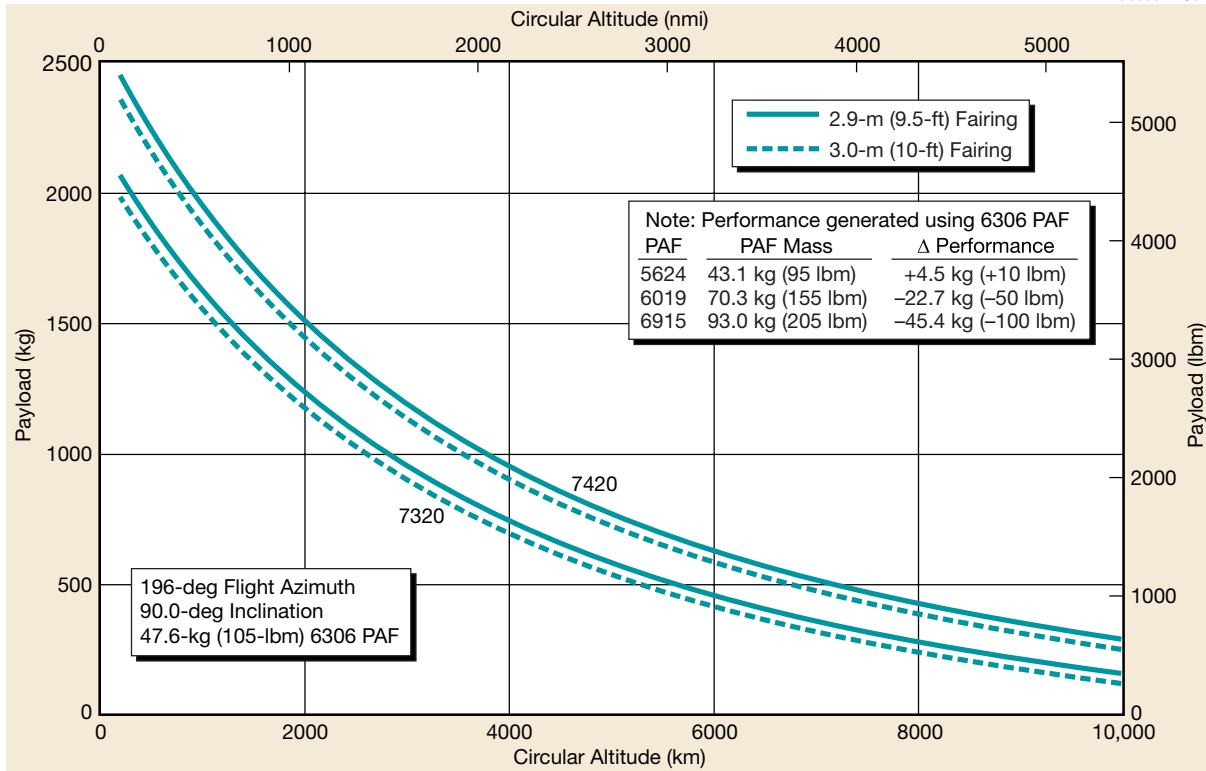


Figure 2-21. Delta II 7320/7420 Vehicle, Two-Stage Circular Orbit Altitude Capability—Western Launch Site

HB01017REU0.4

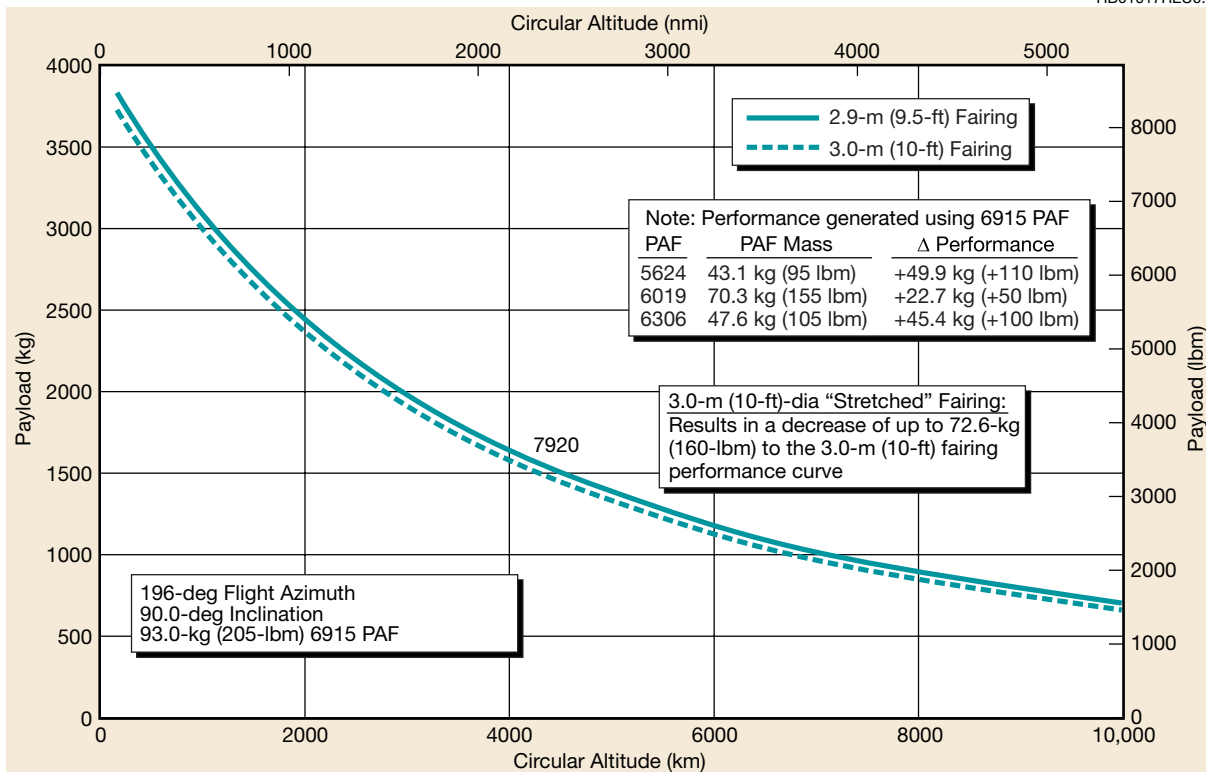


Figure 2-22. Delta II 7920 Vehicle, Two-Stage Circular Orbit Altitude Capability—Western Launch Site

HB00969REU0.3

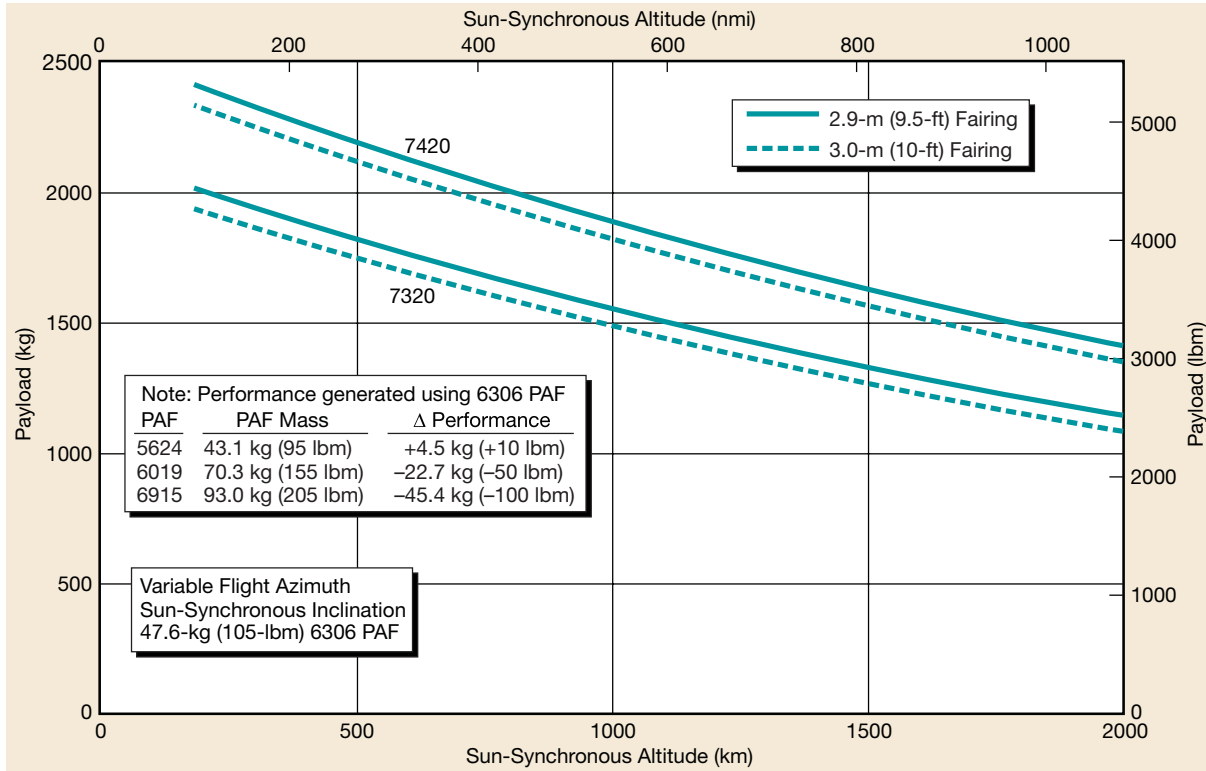


Figure 2-23. Delta II 7320/7420 Vehicle, Two-Stage Sun-Synchronous Capability—Western Launch Site

HB01018REU0.4

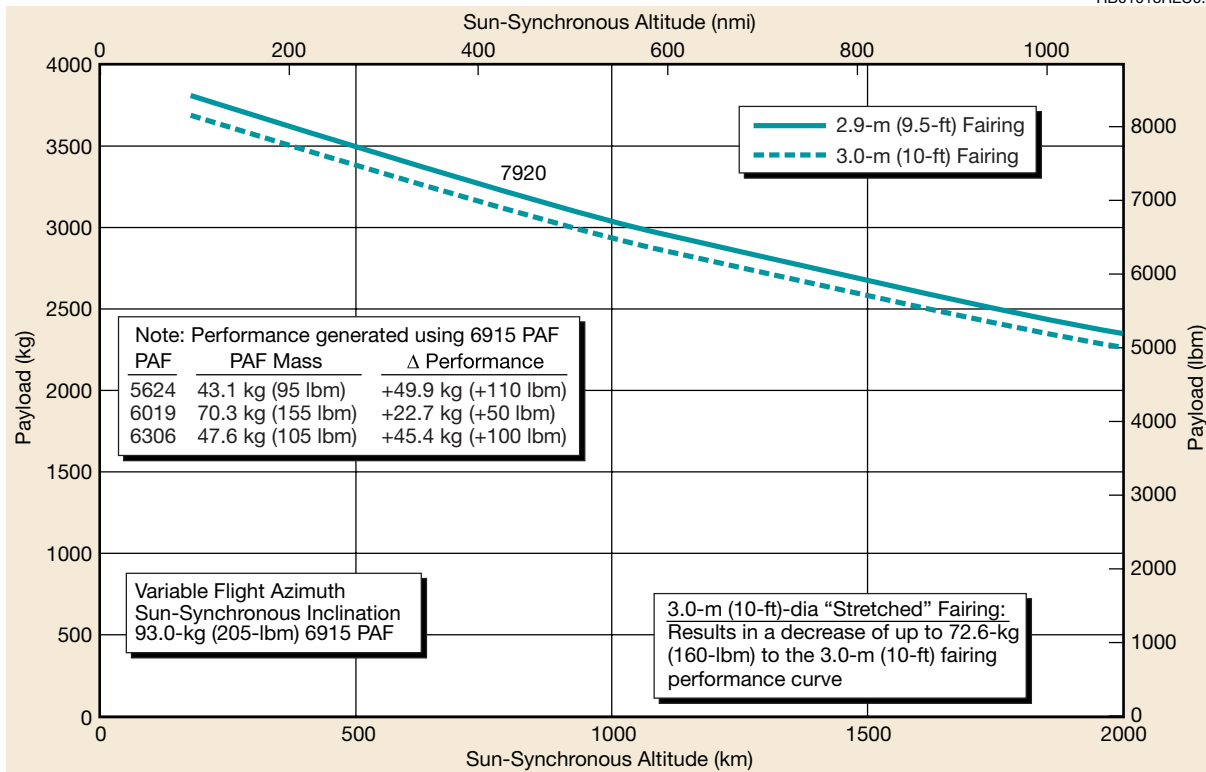


Figure 2-24. Delta II 7920 Vehicle, Two-Stage Sun-Synchronous Capability—Western Launch Site

HB00967REU0.2

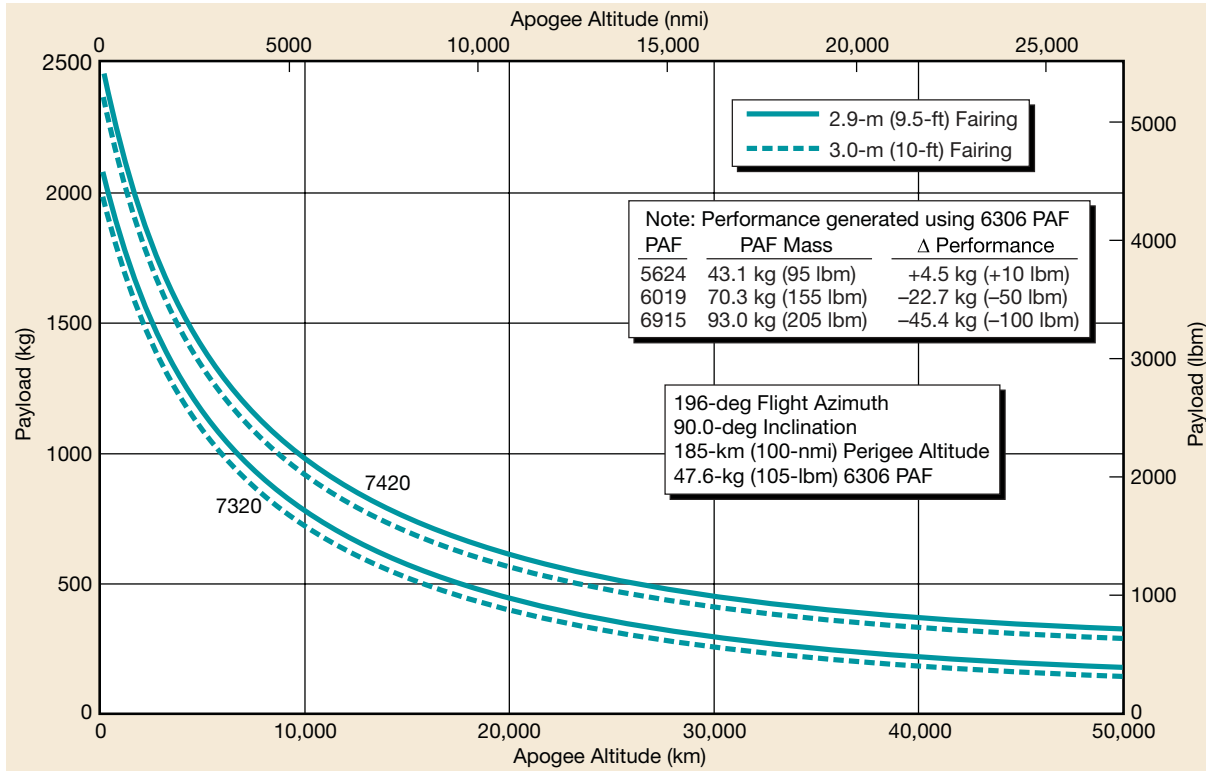


Figure 2-25. Delta II 7320/7420 Vehicle, Two-Stage Apogee Altitude Capability—Western Launch Site

HB01016REU0.3

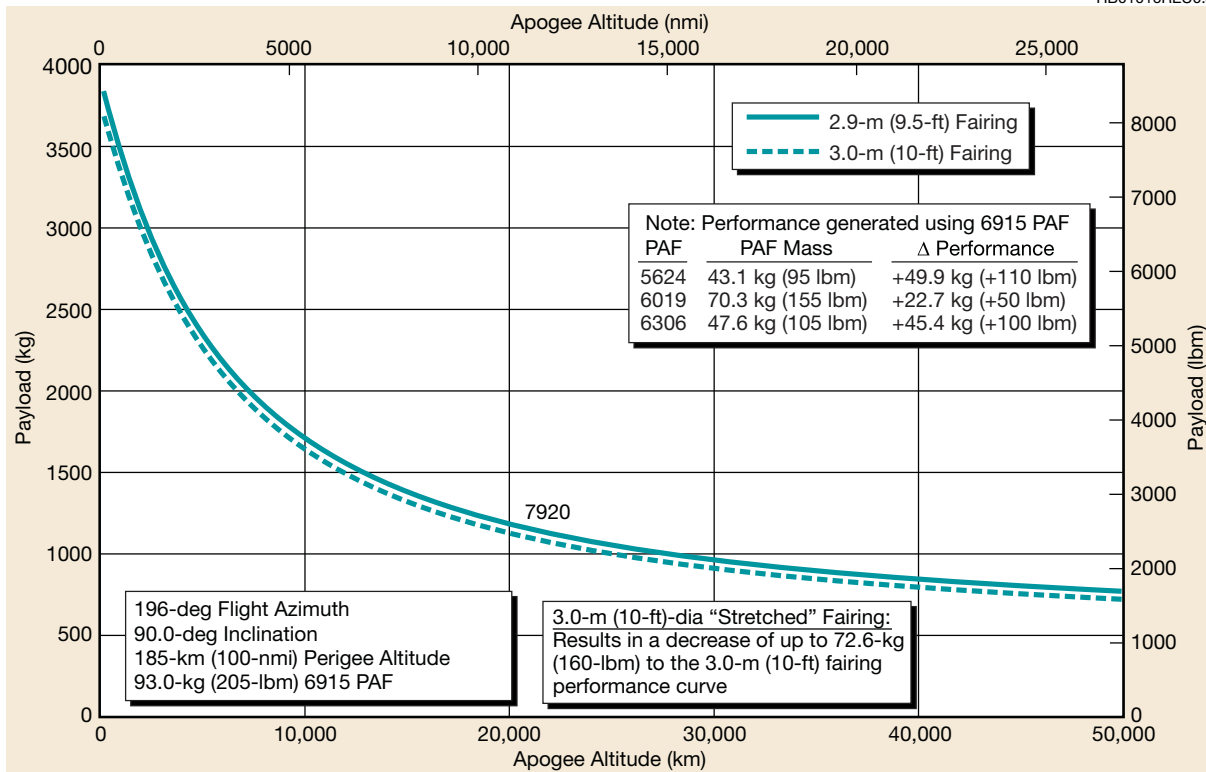


Figure 2-26. Delta II 7920 Vehicle, Two-Stage Apogee Altitude Capability—Western Launch Site

HB00966REU0.2

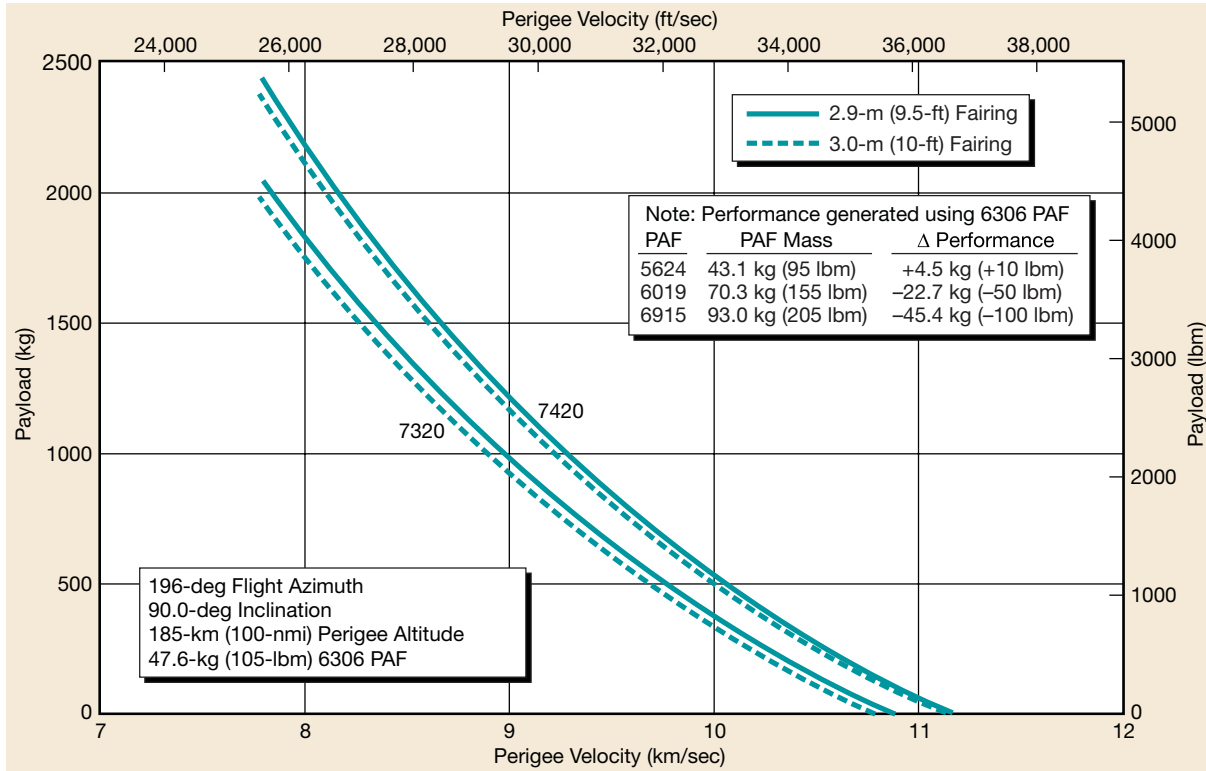


Figure 2-27. Delta II 7320/7420 Vehicle, Two-Stage Perigee Velocity Capability—Western Launch Site

HB01015REU0.3

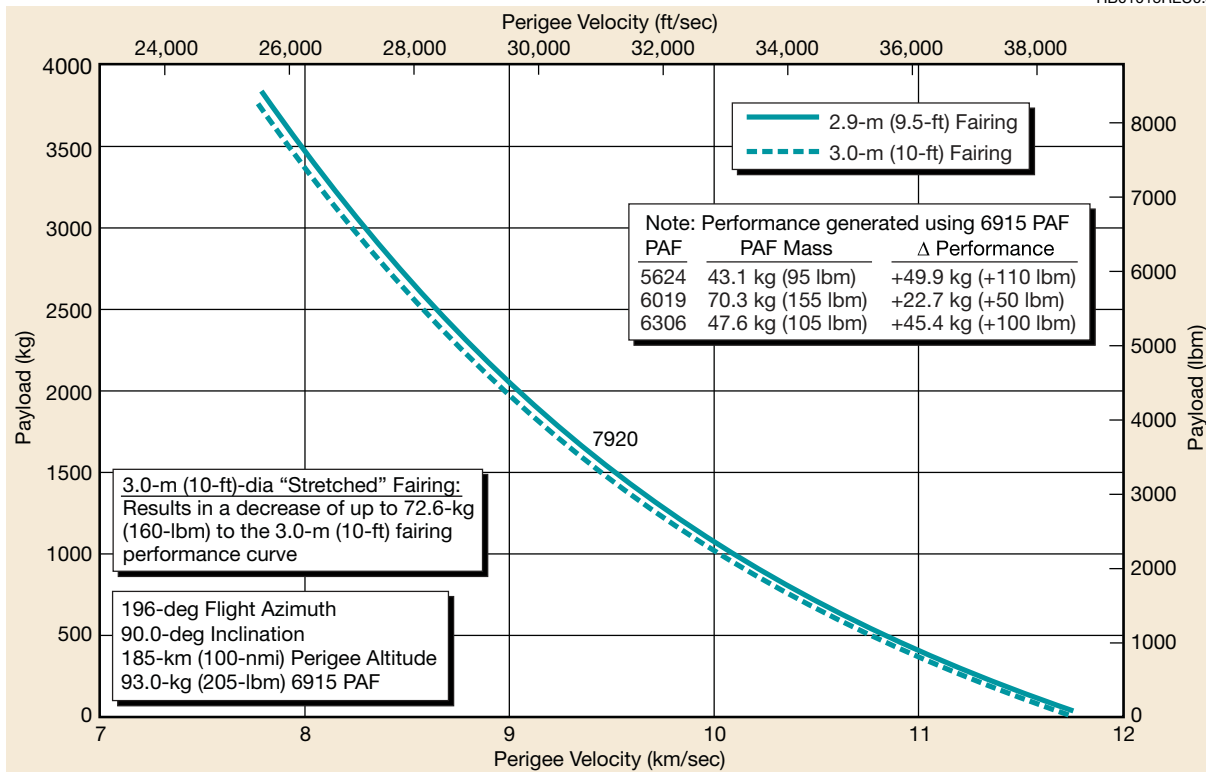


Figure 2-28. Delta II 7920 Vehicle, Two-Stage Perigee Velocity Capability—Western Launch Site

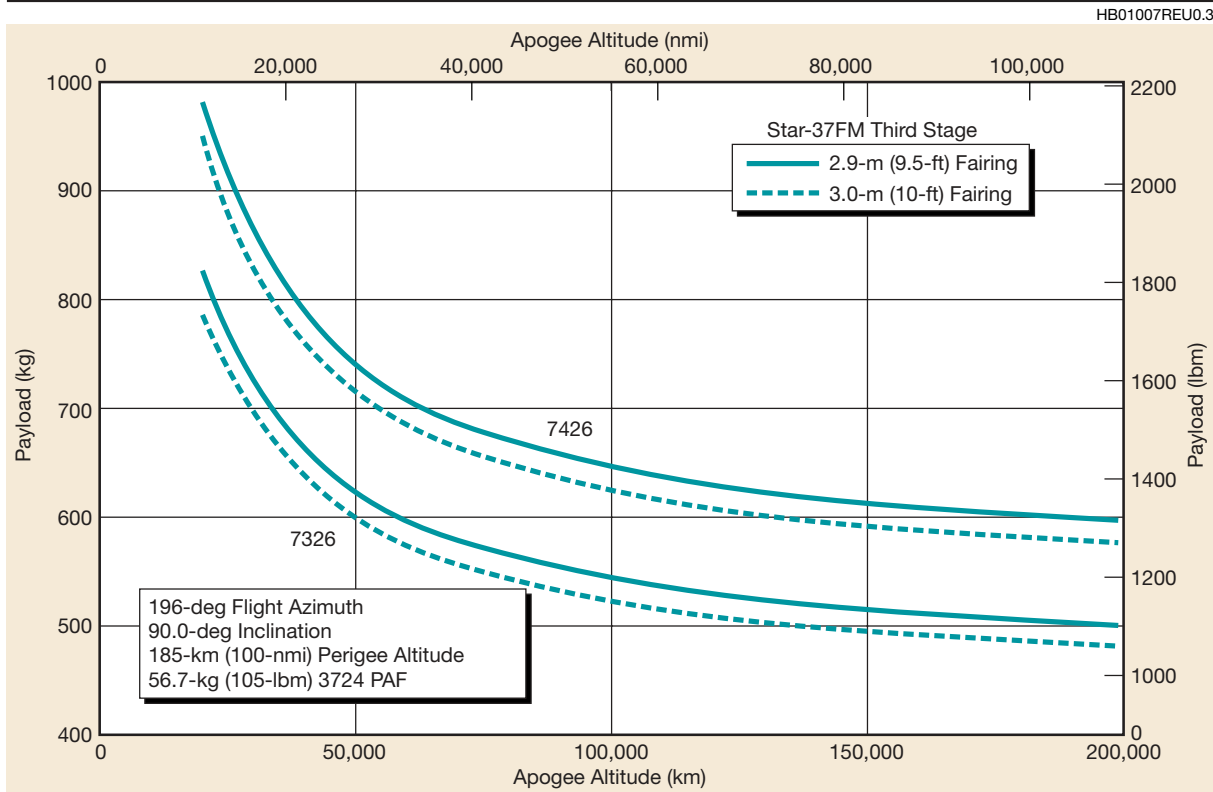


Figure 2-29. Delta II 7326/7426 Vehicle, Three-Stage Apogee Altitude Capability—Western Launch Site

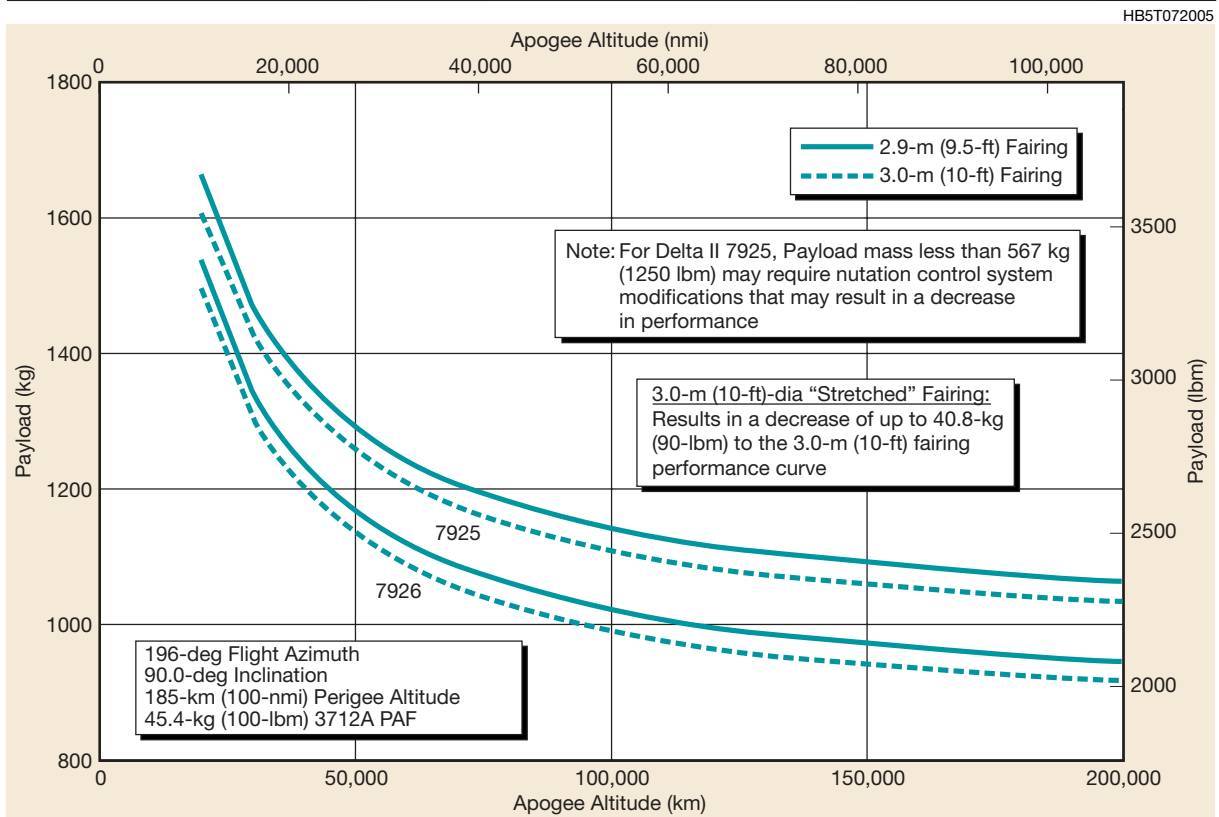


Figure 2-30. Delta II 792X Vehicle, Three-Stage Apogee Altitude Capability—Western Launch Site

HB00970REU0.3

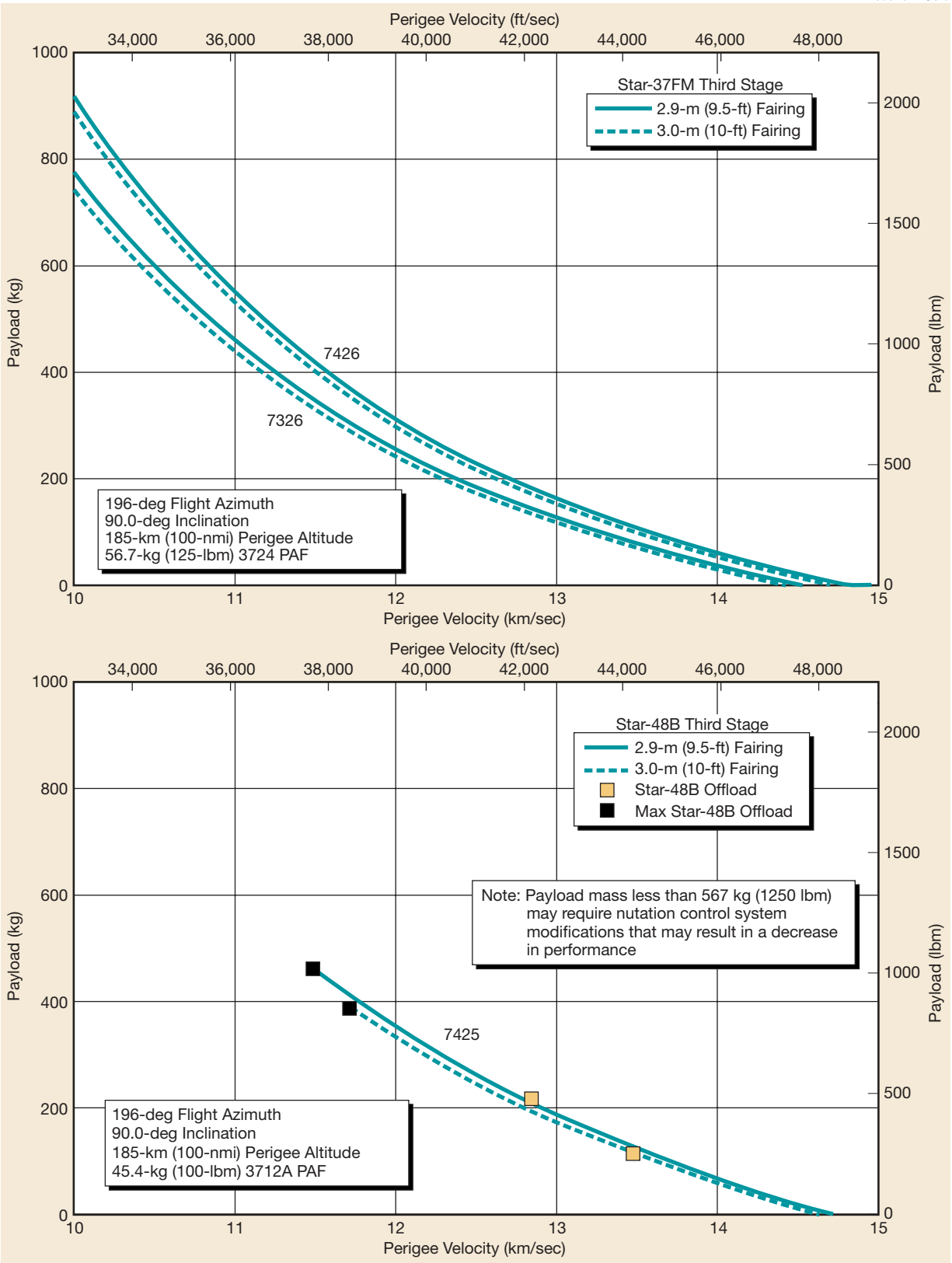
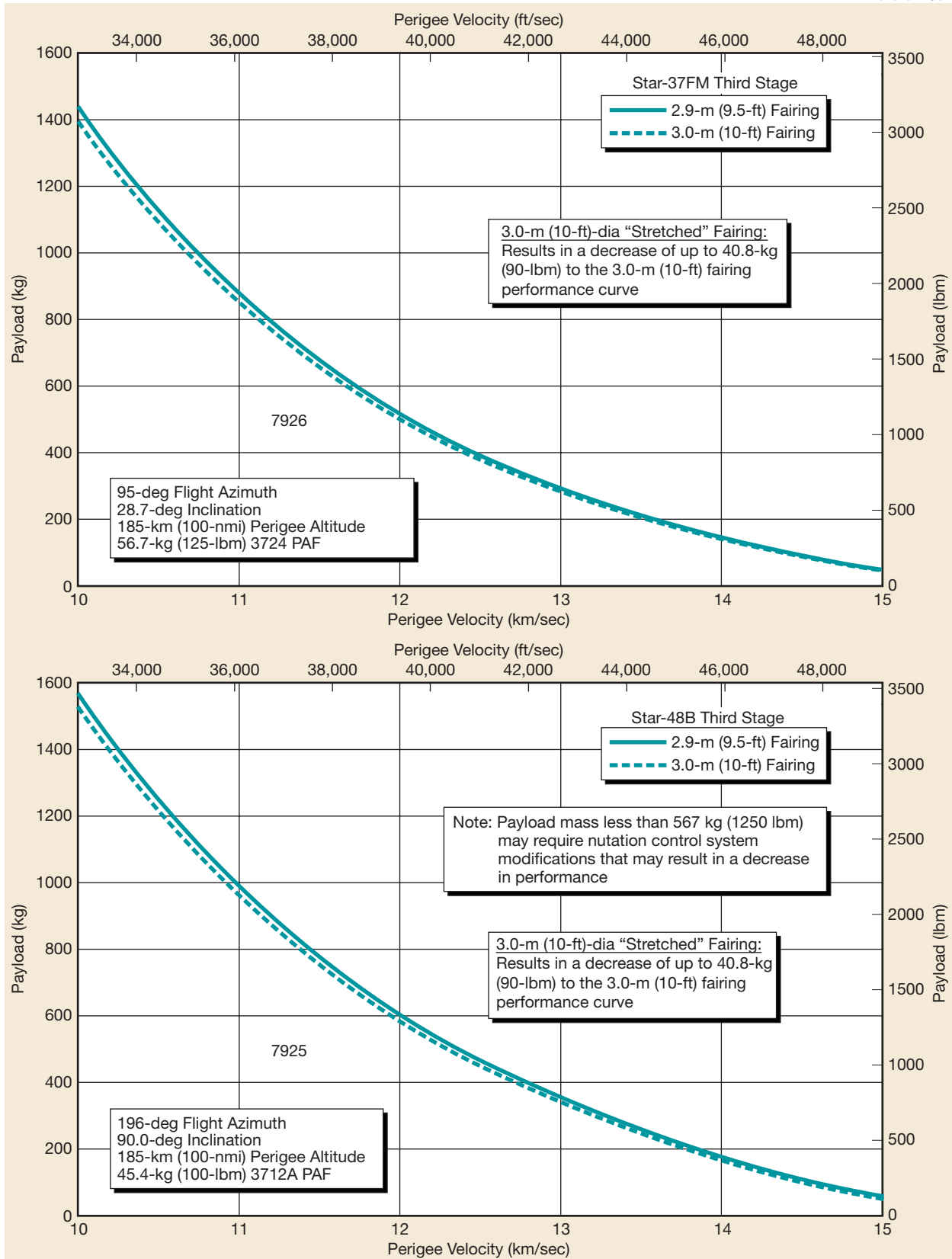


Figure 2-31. Delta II 732X/742X Vehicle, Three-Stage Perigee Velocity Capability—Western Launch Site

HB01019REU0.4



2.4 MISSION ACCURACY DATA

All Delta II configurations employ the RIFCA mounted in the second-stage guidance compartment. This system provides precise pointing and orbit accuracy for both two- and three- stage missions.

For a second-stage PCS of 99.7%, the typical three-sigma (3σ) dispersions for a two-stage mission to low-earth orbit are:

- Perigee altitude: -25.0 km (-13.5 nmi)/+9.3 km (+5.0 nmi).
- Apogee altitude: -9.3 km (-5.0 nmi)/+9.3 km (+5.0 nmi).
- Orbit inclination: ± 0.05 deg.

In a three-stage mission, the parking orbit parameters achieved are quite accurate. The final orbit (e.g., GTO) is primarily affected by the third-stage pointing and the velocity errors from the third-stage solid-motor burn. The pointing error for a given mission depends on the third-stage/spacecraft mass properties and the spin rate. The typical pointing error at third-stage ignition is approximately 1.5 deg for the Star-48B and 2.0 deg for the Star-37FM motor based on past Delta experience. Deviations from nominal apogee altitude using the 7300, 7400, 7900, and 7900H launch vehicles for GTO mission from ER launch site are shown in Figure 2-33. The transfer orbit inclination error is typically from ± 0.2 to ± 0.6 deg over the range shown, while the perigee altitude variation is typically about ± 9.3 km (± 5 nmi). All errors are $3\text{-}\sigma$ values.

These data are presented as general indicators only. Individual mission requirements and specifications will be used as the basis for detailed analyses for specific missions. The customer is invited to contact the Delta Program Office for further information.

HB01033REU0.3

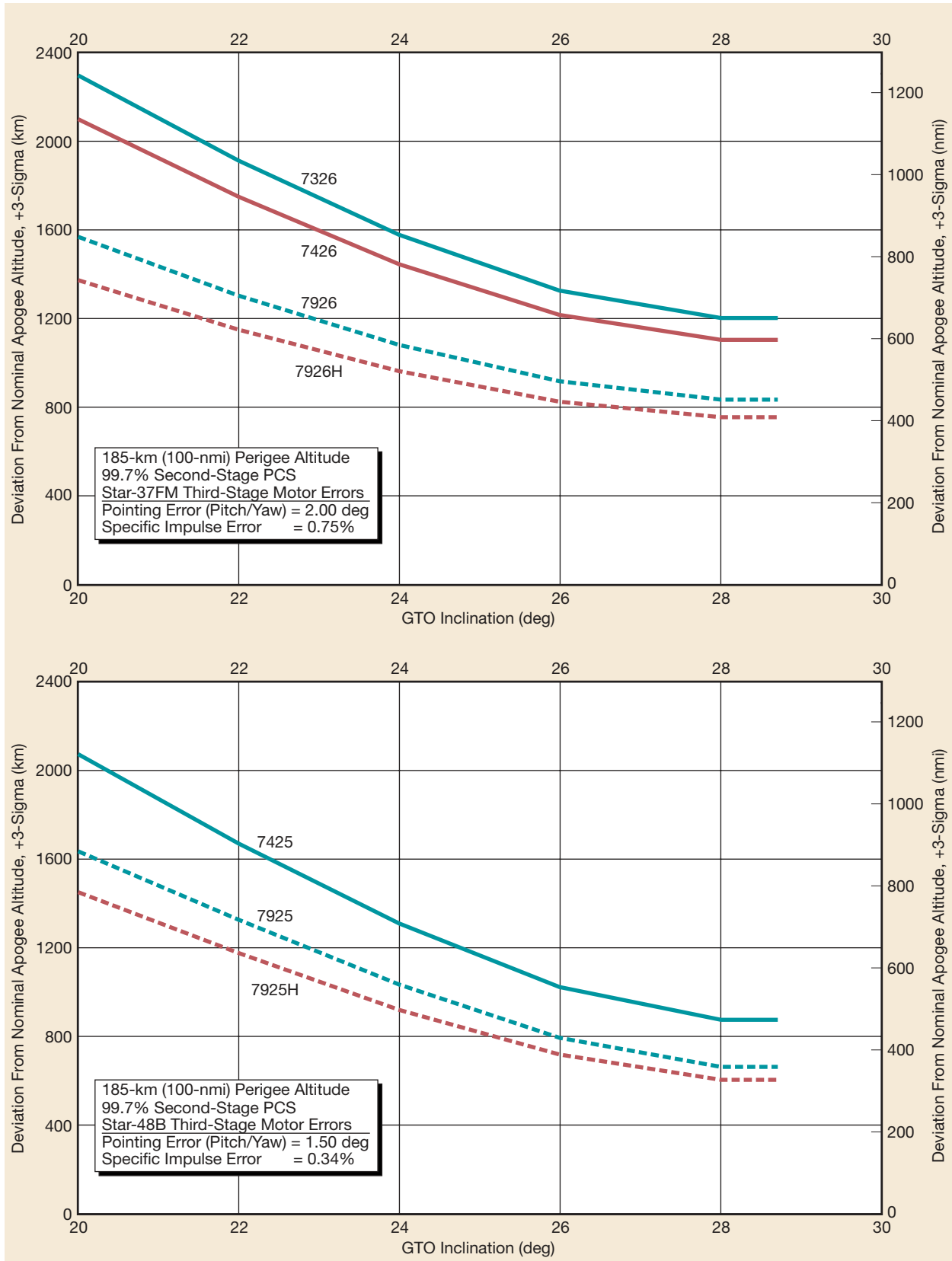


Figure 2-33. Delta II Vehicle, GTO Deviations Capability—Eastern Launch Site

Section 3 PAYLOAD FAIRINGS

The payload is protected by a fairing that shields it from aerodynamic buffeting and heating while in the lower atmosphere. The Delta II launch vehicle currently offers three fairings: a 2.9-m (9.5-ft)-dia metallic fairing and a 3.0-m (10-ft)-dia composite fairing that comes in two different lengths. A general discussion of the available fairings is presented below, while detailed descriptions and payload static envelopes for the fairings are presented in following sections.

3.1 GENERAL DESCRIPTION

The payload envelopes presented in the following sections define the maximum allowable static dimensions of the spacecraft (including manufacturing tolerances) for the spacecraft/payload attach fitting (PAF) interface. If the spacecraft dimensions are maintained within these envelopes, there will be no contact of the spacecraft with the fairing during flight, provided that the frequency and structural stiffness characteristics of the spacecraft are in accordance with the dynamic environmental limits specified in Section 4. The envelopes include allowances for relative static/dynamic deflections between the launch vehicle and spacecraft. Also included are the manufacturing tolerances of the launch vehicle as well as the thickness of the acoustic blanket installed on the fairing interior with billowing effect accounted for. Available blanket configurations are described in Table 3-1.

Table 3-1. Typical Acoustic Blanket Configurations

Fairing	Location
2.9-m (9.5-ft)-dia by 8.5 m (27.8 ft) long	Blankets extend from the nose cap to approximately Station 491. The blanket thicknesses are as follows: 38.1 mm (1.5 in.) in the nose section, 76.2 mm (3.0 in.) in the 2896-mm (114-in.)-dia section, and 38.1 mm (1.5 in.) in the upper portion of the 2438-mm (96-in.)-dia section.
3-m (10-ft)-dia by 8.9 m (29.1 ft) long	The baseline configuration for acoustic blankets extends from the aft end of the boattail to station 213.42 in the nose section. These blankets are 76.2 mm (3 in.) thick throughout this region.
3-m (10-ft)-dia by 9.2 m (30.3 ft) long	The baseline configuration for acoustic blankets extends from the aft end of the boattail to station 201.04 in the nose section. These blankets are 76.2 mm (3 in.) thick throughout this region.
<ul style="list-style-type: none"> ■ These configurations may be modified to meet mission-specific requirements. ■ Blankets for the 2.9-m (9.5-ft) Delta fairing are constructed of silicone-bonded heat-treated glass-fiber batt enclosed between two 0.076-mm (0.003-in.) conductive Teflon-impregnated fiberglass facesheets. Blankets for the 3.0-m (10-ft)-dia Delta composite fairings are constructed of melamine foam covered with reinforced carbon-loaded kapton facesheets. The blankets are vented through a 5-μm stainless steel mesh filter, which controls particulate contamination to levels better than a class 10,000 cleanroom environment. ■ Outgassing of the acoustic blankets meets the criteria of 1.0% maximum total weight loss and 0.10% maximum volatile condensable material with line-of-sight to payloads for the 2.9-m (9.5-ft) and 3.0-m (10-ft) fairings. 	

0024.8

Clearance layouts and analyses are performed and, if necessary, critical clearances are measured after the fairing is installed to ensure positive clearance during flight. To accomplish this, it is important that the spacecraft description (refer to Section 8) include an accurate definition of the physical location of all points on the spacecraft that are within 51 mm (2 in.) of the allowable envelope. The dimensions must include the maximum manufacturing tolerances.

An air-conditioning inlet umbilical door on the fairing provides a controlled environment to the spacecraft and launch vehicle second stage while on the launch stand. A gaseous nitrogen (GN₂) purge system can be incorporated to provide continuous dry nitrogen to the spacecraft until liftoff.

Contamination is minimized by cleaning the payload fairing at the factory prior to shipment to the launch site. Special cleaning in a cleanroom environment using black light is available upon request at the launch site.

3.2 THE 2.9-M (9.5-FT)-DIAMETER PAYLOAD FAIRING

The 2.9-m (9.5-ft)-dia fairing (Figures 3-1 and 3-2) is an aluminum skin-and-stringer structure fabricated in two half-shells. These shells consist of a hemispherical nose cap, a biconic section, a cylindrical 2896-mm (114-in.)-dia center section (the maximum diameter of the fairing), a 30-deg conical transition, and a cylindrical base section having the 2438-mm (96-in.) core vehicle diameter. The biconic section is a ring-stiffened monocoque structure; one-half of which is fiberglass covered with a removable aluminum foil lining to create a radio frequency (RF) window. The cylindrical base section is an integrally stiffened isogrid structure, and the cylindrical center section has a skin-and-stringer construction. The fairing has an overall length of 8488 mm (334.2 in.).

The half-shells are joined by a contamination-free linear piston/cylinder thrusting separation system that runs longitudinally the full length of the fairing. Two functionally redundant explosive bolt assemblies provide structural continuity at the fairing base ring. Four functionally redundant explosive bolt assemblies (two each) provide circumferential structural continuity at the 30-deg transition section between the 2896-mm (114-in.)-dia section and the 2438-mm (96-in.)-dia section.



Figure 3-1. Delta 2.9-m (9.5-ft)-dia Payload Fairing

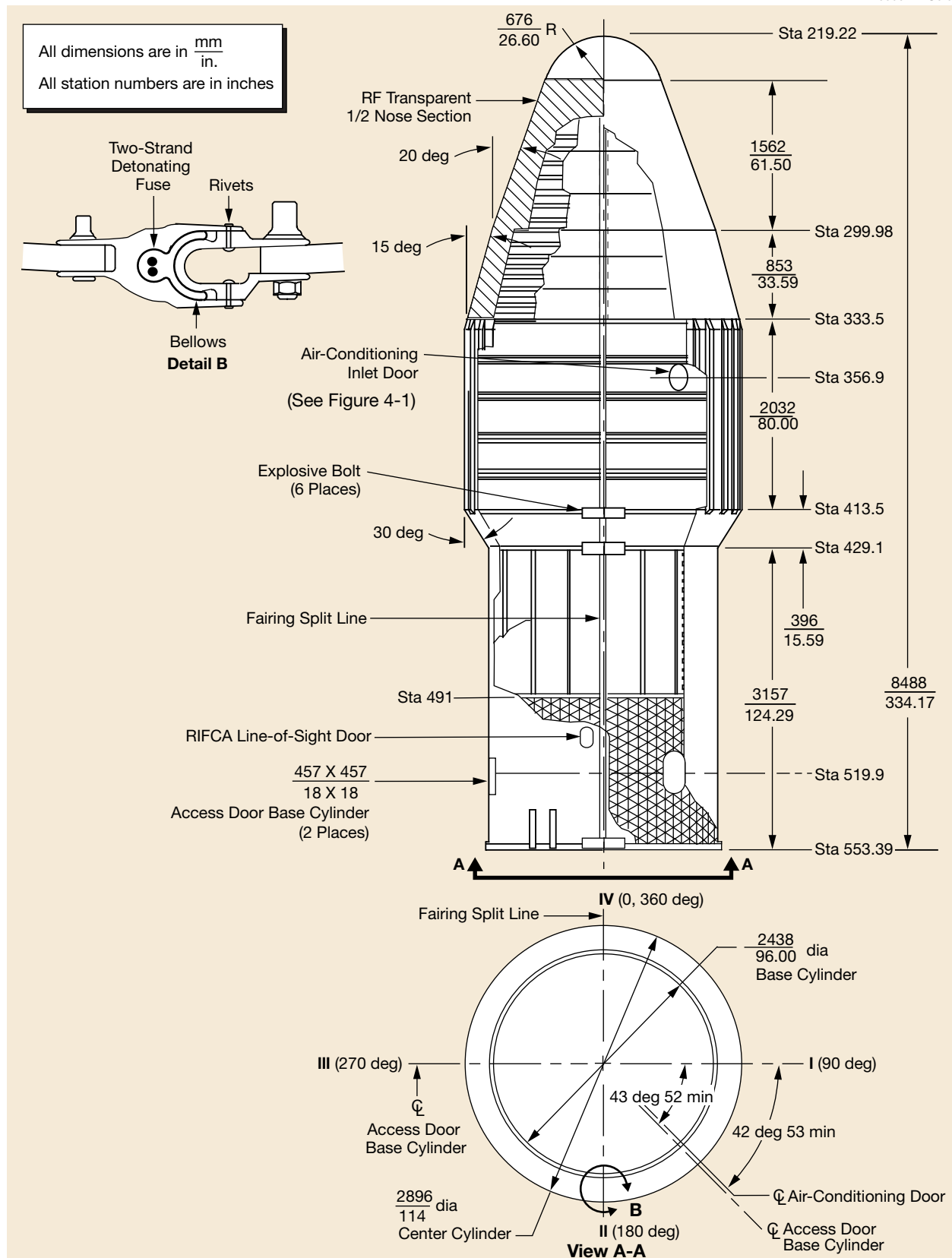


Figure 3-2. Profile, 2.9-m (9.5-ft)-dia Payload Fairing

The fairing half-shells are jettisoned by actuation of the base and transition separation nuts and by the detonating fuse in the thrusting joint cylinder rail cavity. A bellows assembly within each cylinder rail retains the detonating-fuse gases to prevent contamination of the spacecraft during the fairing separation event.

Two 457-mm by 457-mm (18-in. by 18-in.) access doors for second-stage access are part of the baseline fairing configuration (Figure 3-2). To satisfy spacecraft requirements, additional removable doors of various sizes and locations can be provided to permit access to the spacecraft following fairing installation. See Section 3.5 for specific information. It should be noted that the large access doors will have acoustic blankets. The quantity and location of access doors must also be coordinated with the Delta Program Office.

The fiberglass biconic section can be made RF transparent by removal of its aluminum foil lining. Location and size of the RF panels must be coordinated with the Delta Program Office.

Acoustic absorption blankets are provided within the fairing interior. The typical blanket configuration is described in Table 3-1. Blanket thermal characteristics are discussed in Section 4.2.2.

The allowable static spacecraft envelopes for existing PAFs within the fairing are shown in Figures 3-3 through 3-5 and assume that the spacecraft stiffness recommended in Section 4 is maintained. Usable envelopes below the separation plane and local protuberances outside the envelopes presented require coordination and approval of the Delta Program Office.

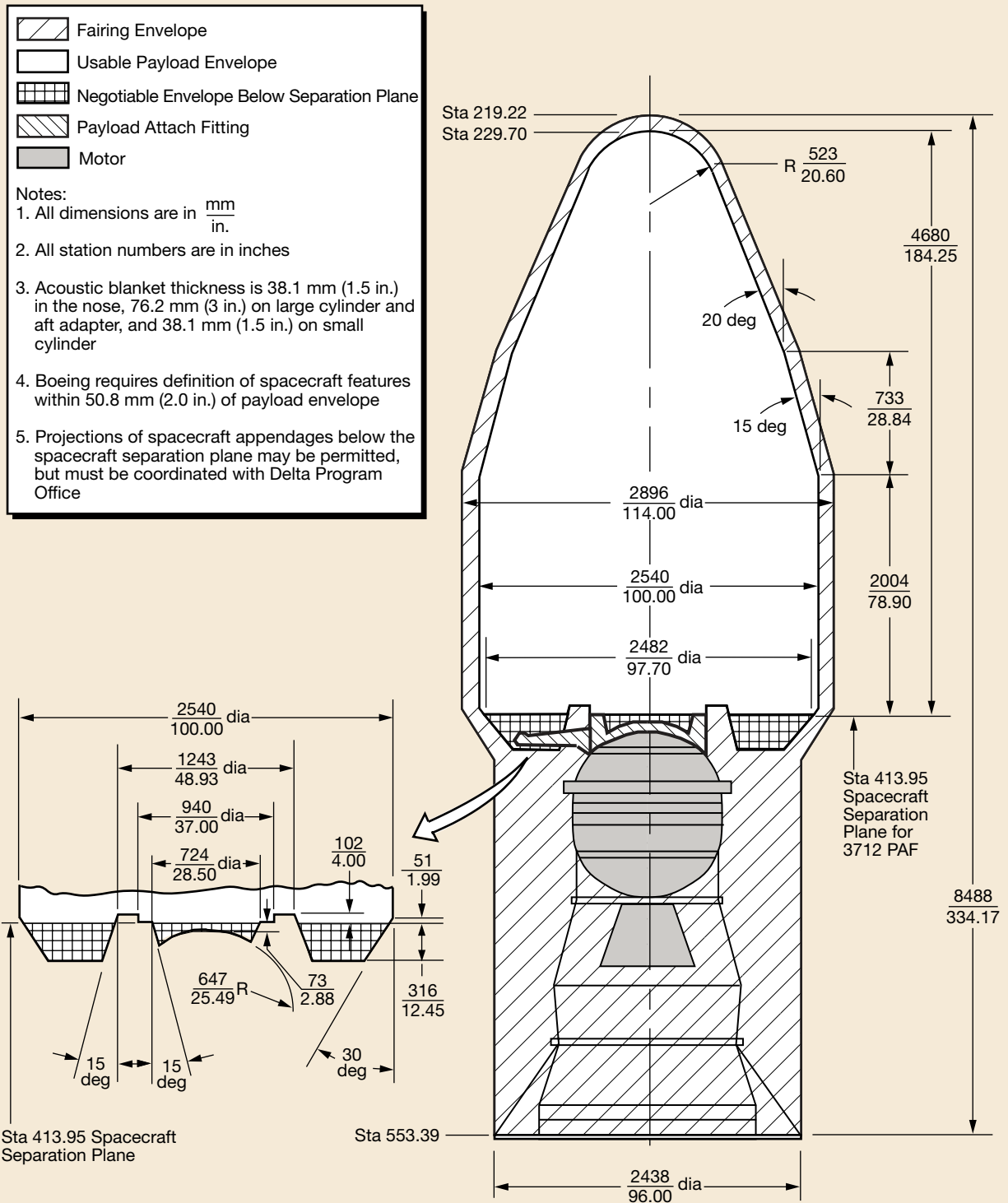


Figure 3-3. Payload Static Envelope, 2.9-m (9.5-ft)-dia Fairing, Three-Stage Configuration (3712 PAF)

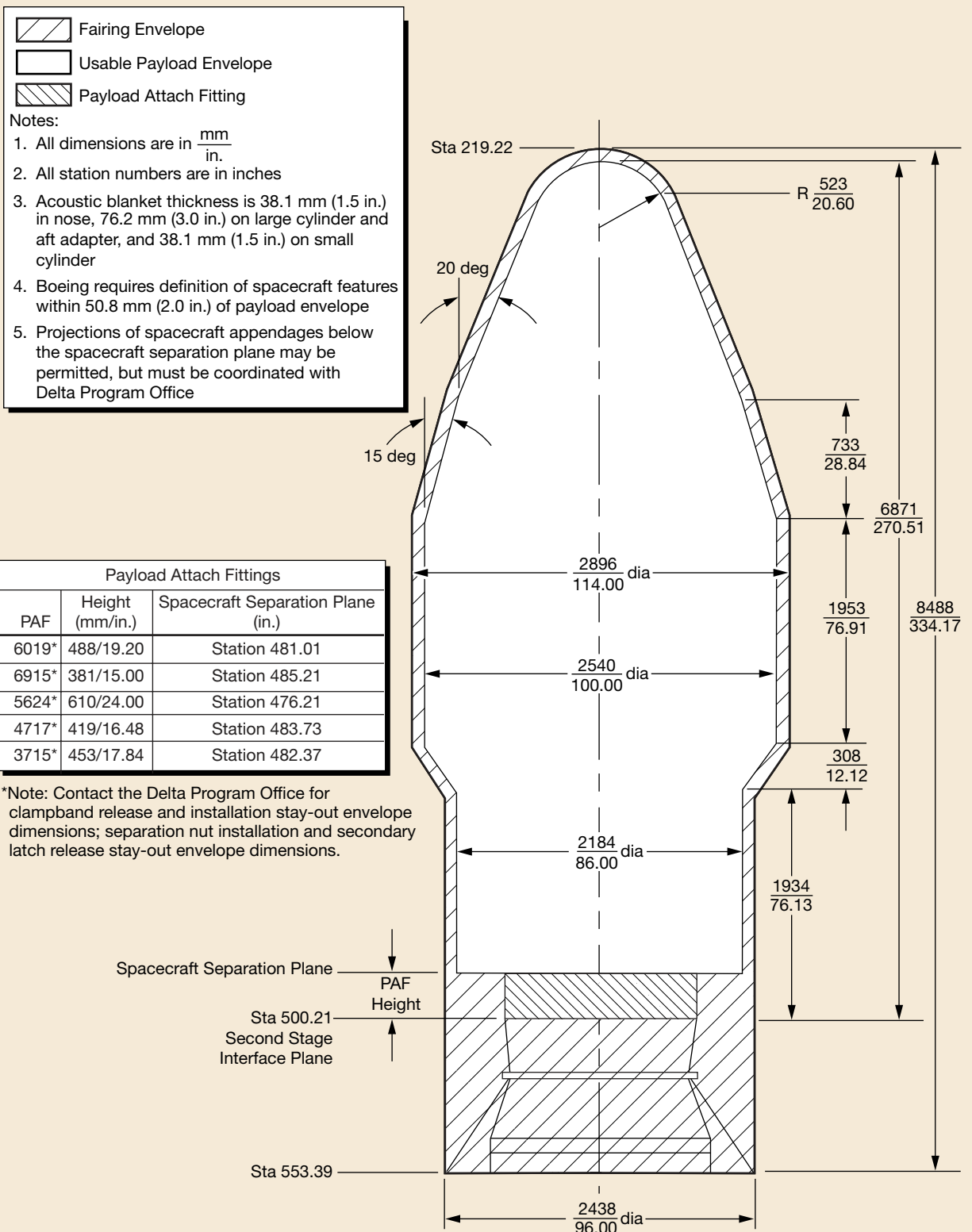


Figure 3-4. Payload Static Envelope, 2.9-m (9.5-ft)-dia Fairing, Two-Stage Configuration (Various PAFs)

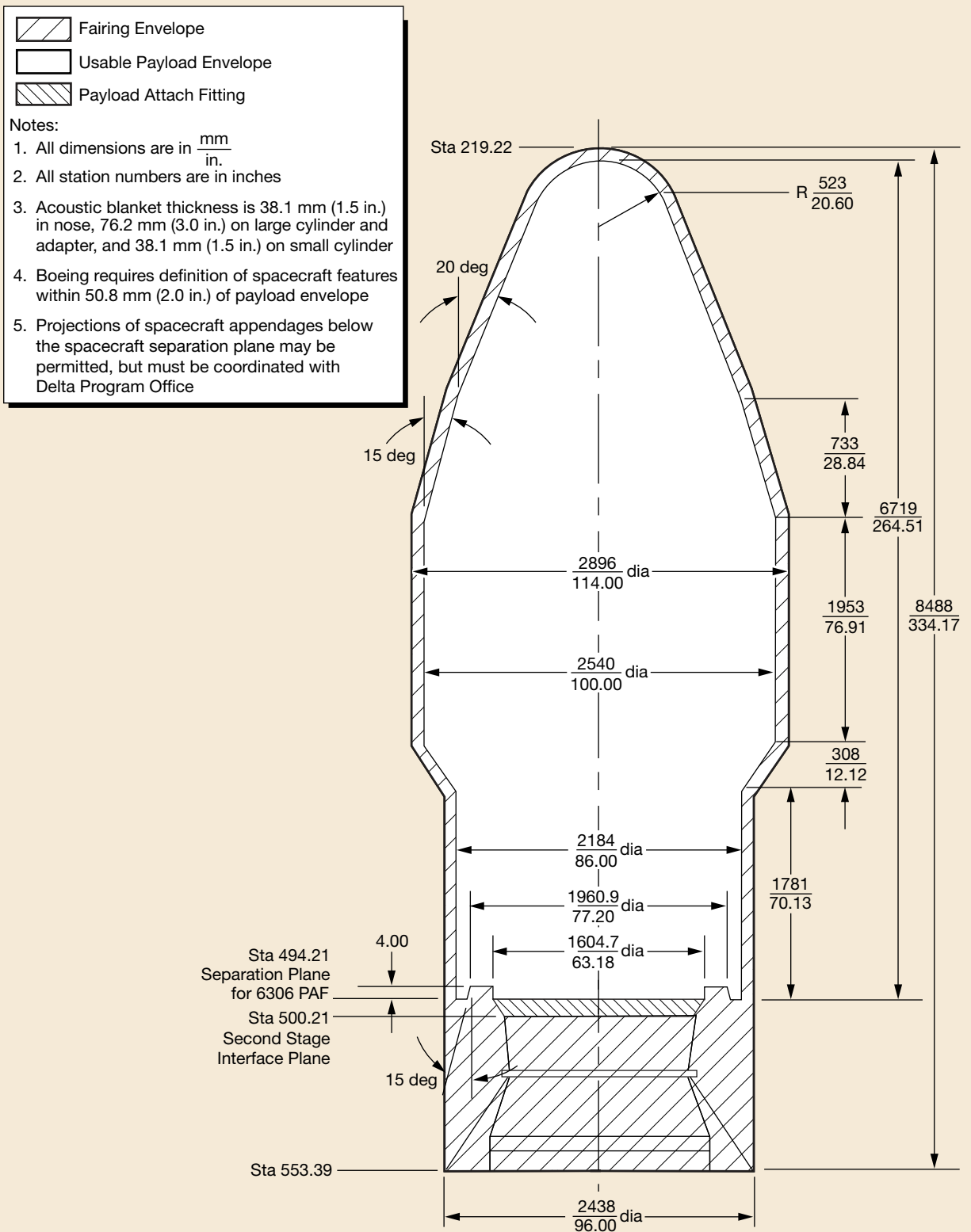


Figure 3-5. Payload Static Envelope, 2.9-m (9.5-ft)-dia Fairing, Two-Stage Configuration (6306 PAF)

3.3 THE 3-M (10-FT)-DIAMETER PAYLOAD FAIRING

The 3-m (10-ft)-dia fairing is available for spacecraft requiring a larger envelope. The fairing (Figures 3-6 and 3-7) is a composite sandwich structure that separates into bisectors. Each bisector is constructed in a single co-cured layup, eliminating the need for module-to-module manufacturing joints and intermediate ring stiffeners. The resulting smooth inside skin enables the flexibility to install mission-unique access doors almost anywhere in the cylindrical portion of the fairing. An RF window can be accommodated, similar to mission-unique access doors. All these requirements must be coordinated with the Delta Program office.

The bisectors are joined by a contamination-free linear piston/cylinder thrusting separation system that runs longitudinally the full length of the fairing. Two functionally redundant explosive bolt assemblies provide the structural continuity at the fairing base ring. The fairing bisectors are jettisoned by actuation of the base separation nuts, and by the detonating fuse in the thrusting joint cylinder rail cavity. A bellows assembly within each cylinder rail retains the detonating-fuse gases to prevent spacecraft contamination during the fairing separation event.

Two standard 457-mm (18-in.)-dia access doors are part of the baseline fairing configuration for second-stage access (Figure 3-7). To further meet customer needs, additional 610-mm (24-in.)-dia doors can be provided in the fairing cylindrical section for spacecraft access after encapsulation. See Section 3.5 for specific information. The quantities and locations of additional access doors must be coordinated with the Delta Program Office.



Figure 3-6. 3-m (10-ft)-dia Composite Fairing

Acoustic absorption blankets are provided on the fairing interior. Typical blanket configurations are described in Table 3-1.

The allowable static spacecraft envelopes within the fairing are shown in Figures 3-8 through 3-10 for the three- and two-stage configurations. For dual-payload missions, two configurations of the dual-payload attach fitting (DPAF) are used for spacecraft interfaces to the launch vehicle. The allowable static envelope for lower and upper spacecraft is shown in Figures 3-11 through 3-13. The prescribed static envelopes are valid provided that the spacecraft stiffness recommended in Section 4 is maintained. Any protuberance outside the envelopes requires coordination with and approval of Delta Program Office.

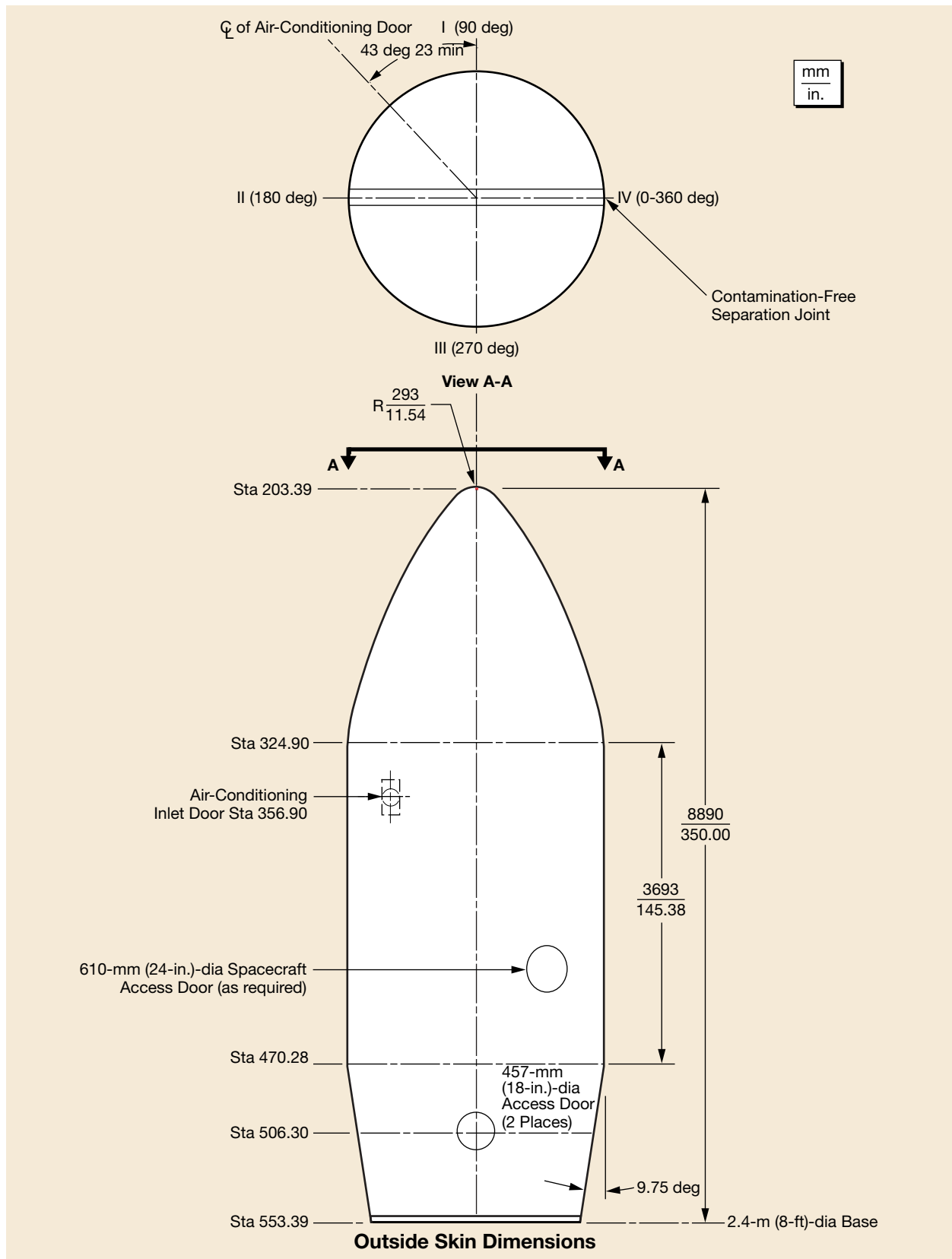


Figure 3-7. Profile, 3-m (10-ft)-dia Composite Fairing

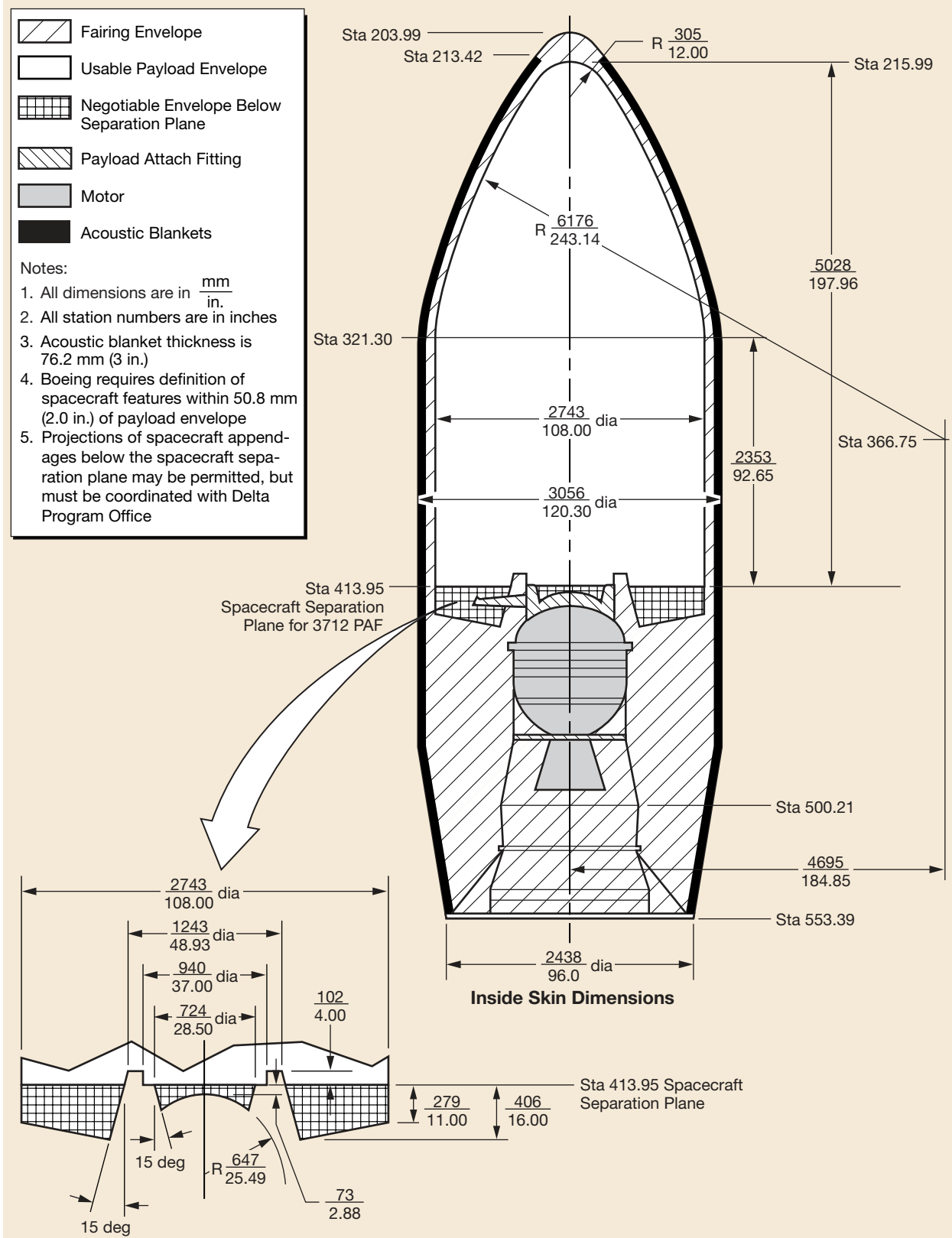






Figure 3-8. Payload Static Envelope, 3-m (10-ft)-dia Fairing, Three-Stage Configuration (3712 PAF)

 Fairing Envelope
 Usable Payload Envelope
 Payload Attach Fitting
 Acoustic Blankets

Notes:

1. All dimensions are in $\frac{\text{mm}}{\text{in.}}$
2. All station numbers are in inches
3. Acoustic blanket thickness is 76.2 mm (3 in.)
4. Boeing requires definition of spacecraft features within 50.8 mm (2.0 in.) of payload envelope
5. Projections of spacecraft appendages below the spacecraft separation plane may be permitted, but must be coordinated with Delta Program Office

Payload Attach Fittings		
PAF	Height (mm/in.)	Spacecraft Separation Plane (in.)
6019*	488/19.20	Station 481.01
6915*	381/15.00	Station 485.21
5624*	610/24.00	Station 476.21
4717*	419/16.48	Station 483.73
3715*	453/17.84	Station 482.37

*Note: For clamband interfaces, contact the Delta Program Office for clamband release and installation stay-out envelope dimensions; separation nut installation and secondary latch release stay-out envelope dimensions.

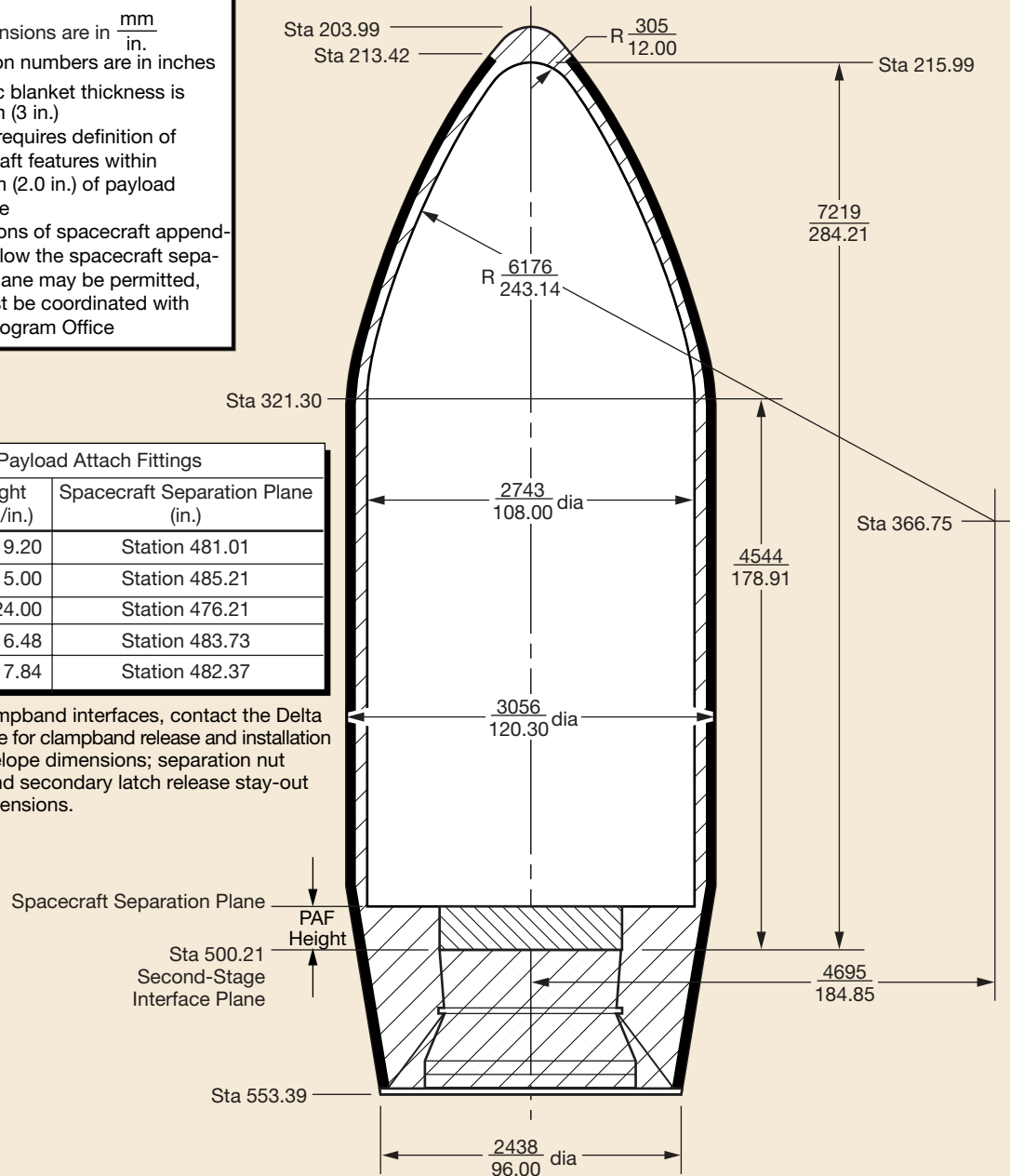


Figure 3-9. Payload Static Envelope, 3-m (10-ft)-dia Fairing, Two-Stage Configuration (Various PAFs)

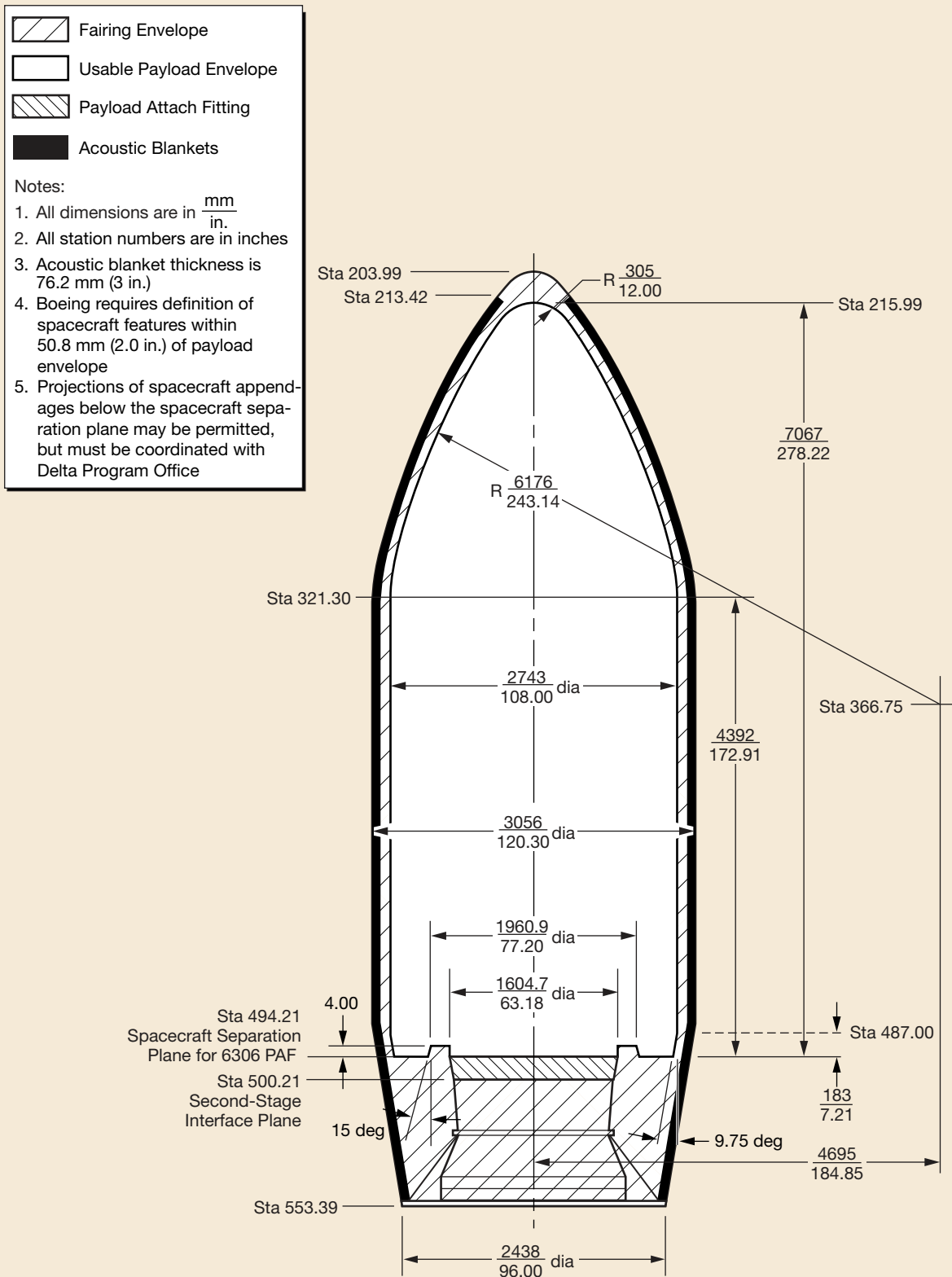


Figure 3-10. Payload Static Envelope, 3-m (10-ft)-dia. Fairing Two-Stage Configuration (6306 PAF)

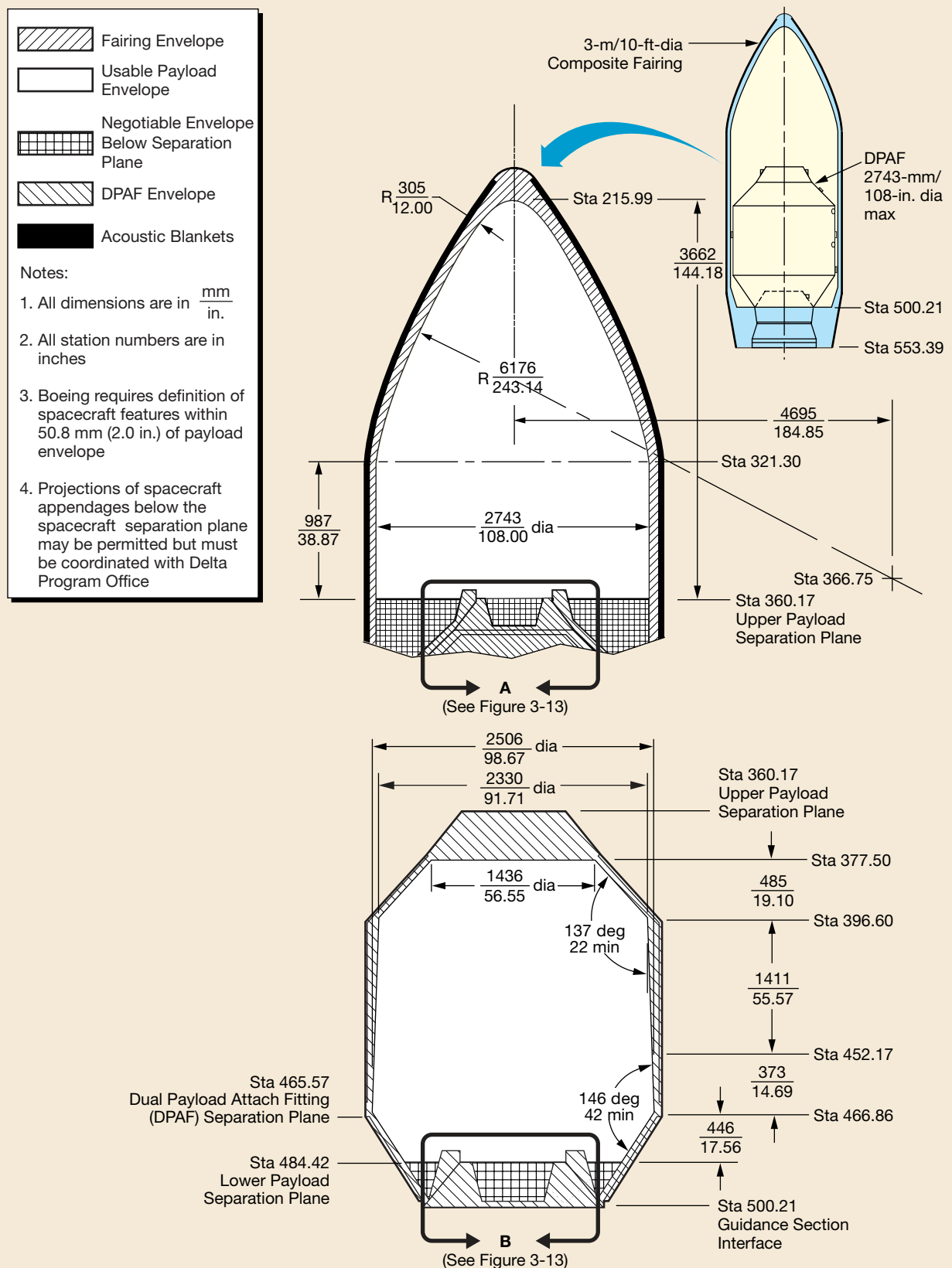


Figure 3-11. Maximum Payload Envelope for 3.0-m (10-ft)-dia Fairing, Dual-Payload Attach Fitting

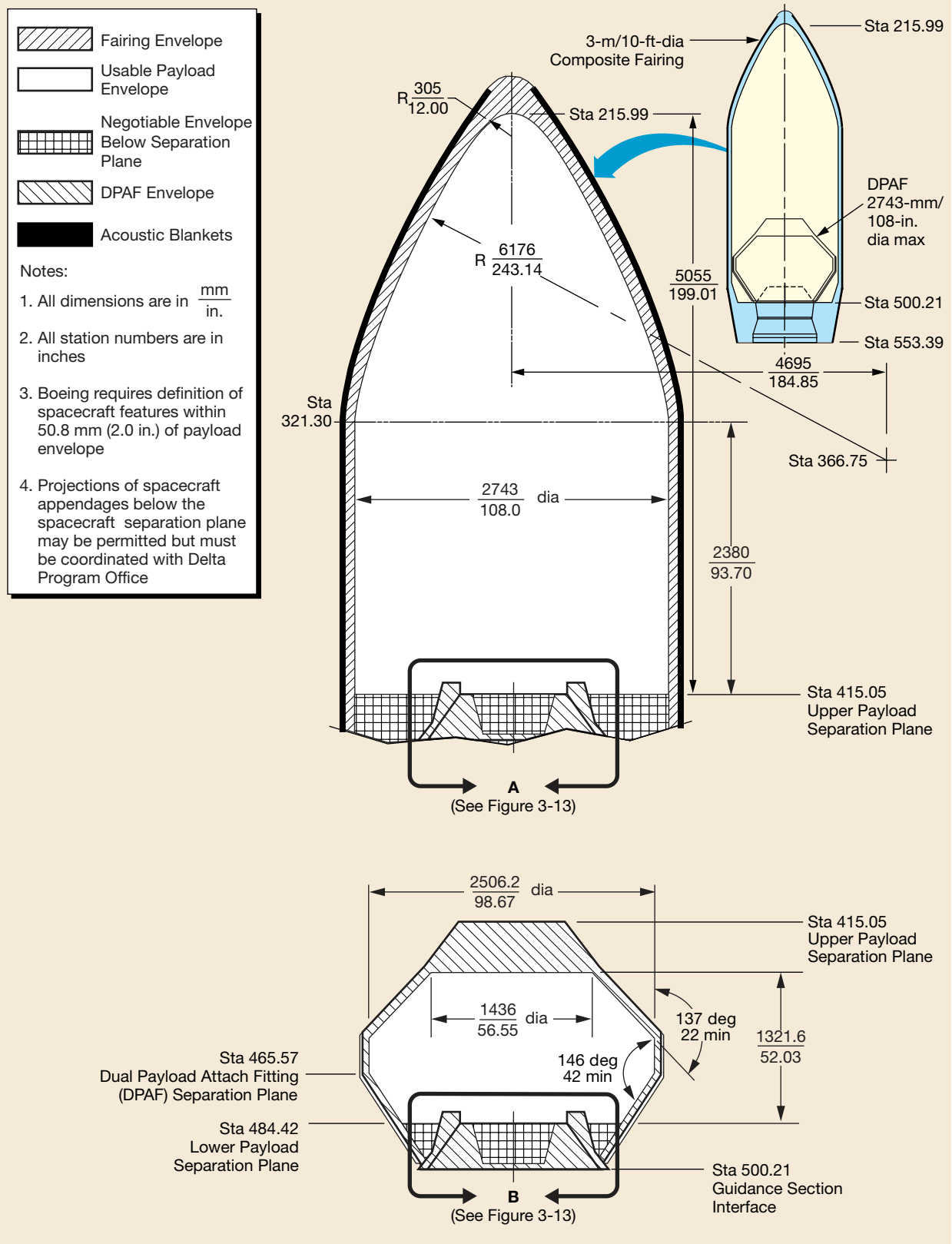


Figure 3-12. Maximum Payload Envelope for 3.0-m (10-ft)-dia Fairing, Reduced Height Dual-Payload Attach Fitting

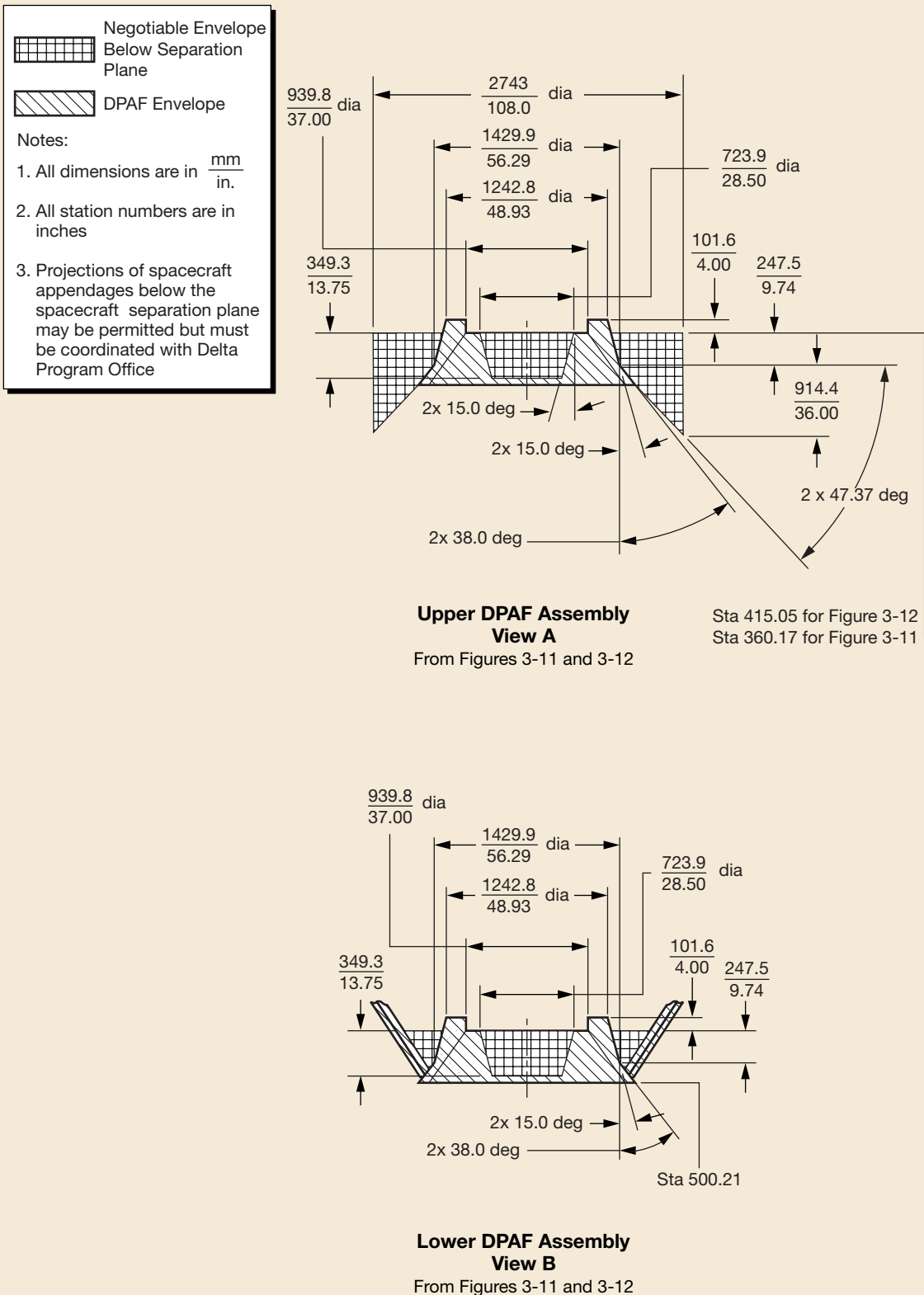


Figure 3-13. Detailed Payload Envelope for 3.0-m (10-ft) dia Fairing, Dual-Payload Attach Fitting and Reduced-Height Dual-Payload Attach Fitting

3.4 THE STRETCHED 3-M (10-FT)-DIAMETER PAYLOAD FAIRING -10L

The stretched 3-m (10-ft)-dia fairing, designated -10L, is available for payloads requiring a longer envelope than the 3-m (10-ft)-dia fairing described in Section 3.3. The -10L fairing (Figures 3-14 and 3-15) is also a composite sandwich structure that separates into bisectors. The cylindrical section is lengthened by 0.979 m (3.21 ft), making the overall length 0.36 m (1.19 ft) longer than the 3-m (10-ft)-dia fairing.

Other than the difference in length, the discussion in Section 3.3 also applies to the stretched 3-m (10-ft)-dia fairing. The dual-payload attach fitting (DPAF) is also available for the stretched 3-m (10-ft)-dia (-10L) fairing. Contact the Delta Program Office for envelope definition.

The allowable static spacecraft envelopes are shown in Figures 3-16 through 18 for the three- and two-stage configurations, assuming that the spacecraft stiffness recommended in Section 4 is maintained. Any protuberance outside the envelopes requires coordination with and approval of the Delta Program Office.

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Figure 3-14. 3-m (10-ft) Stretched Composite Fairing (-10L)

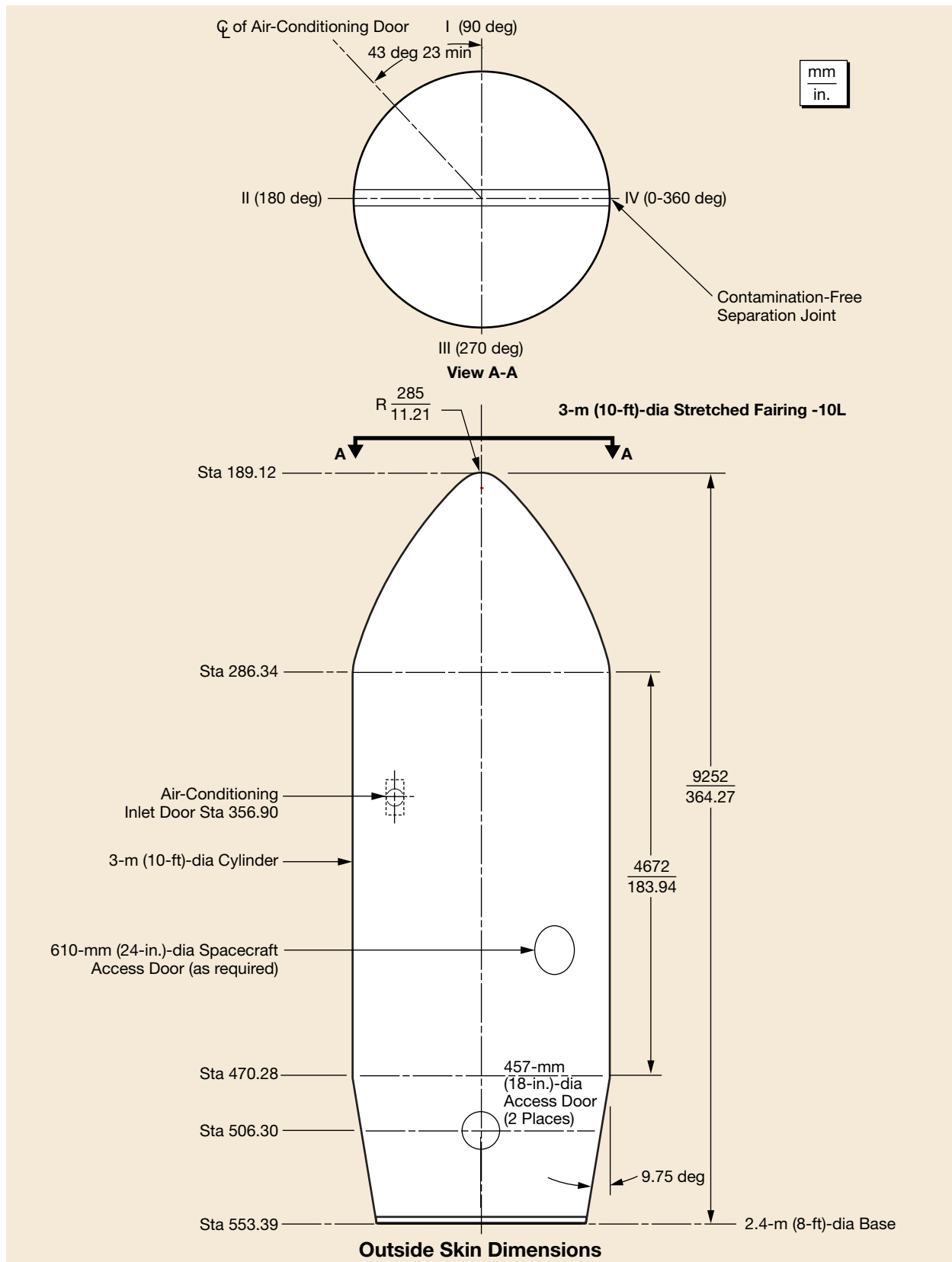


Figure 3-15. Profile, 3-m (10-ft)-dia Stretched Composite Fairing (-10L)

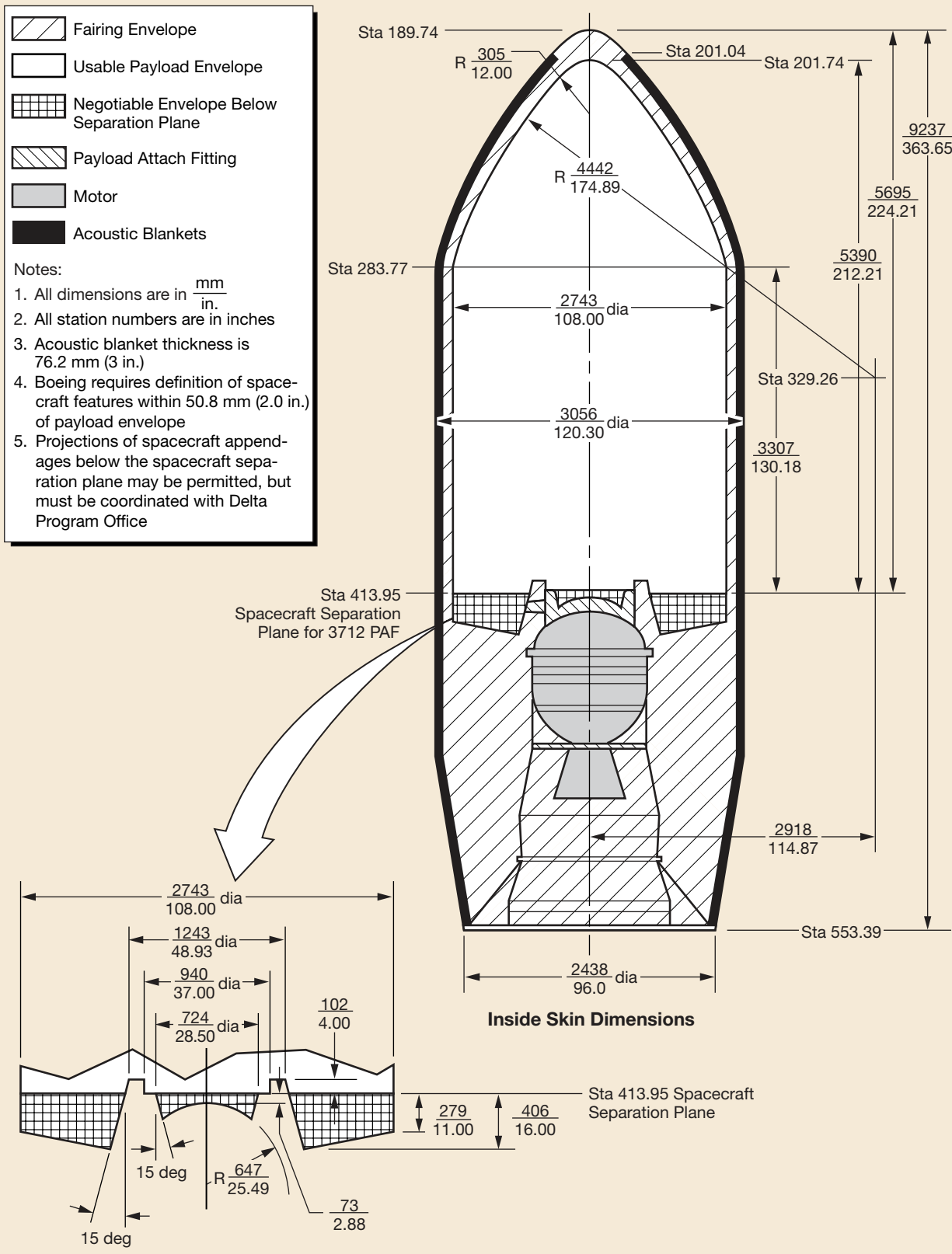


Figure 3-16. Payload Static Envelope, 3-m (10-ft)-dia Stretched Composite Fairing (-10L), Three-Stage Configuration (3712 PAF)

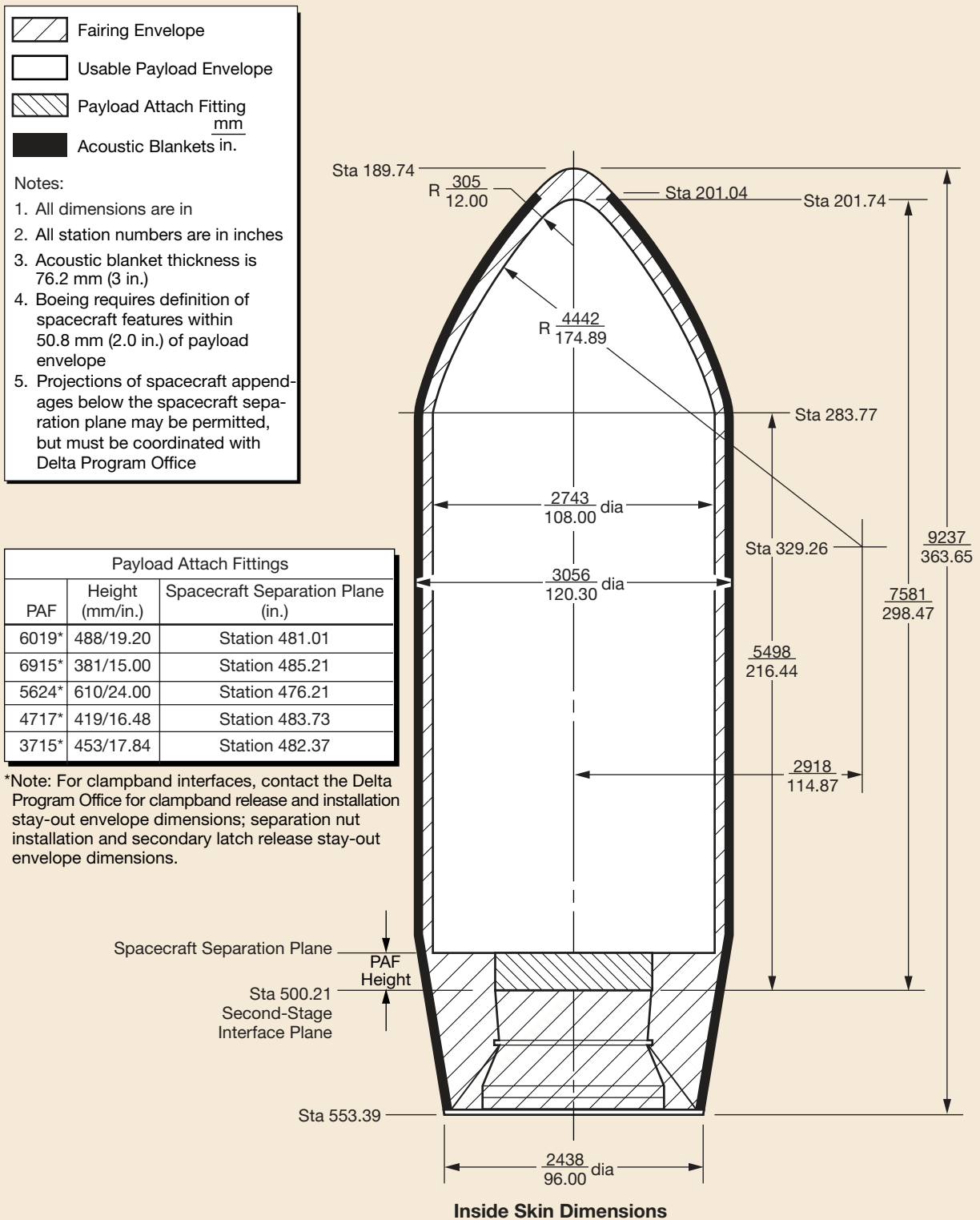


Figure 3-17. Payload Static Envelope, 3-m (10-ft)-dia Stretched Composite Fairing (-10L), Two-Stage Configuration (Various PAFs)

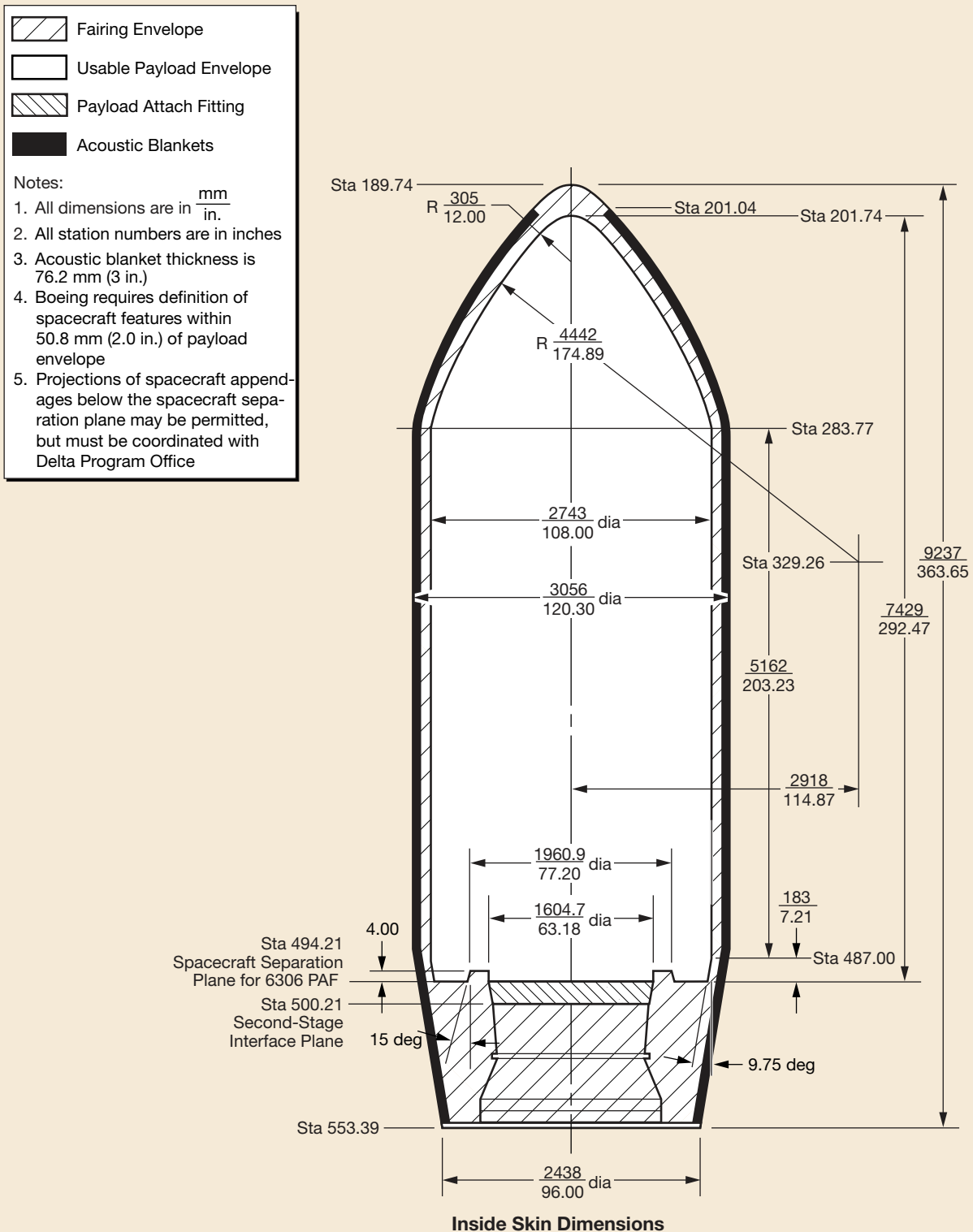


Figure 3-18. Payload Static Envelope, 3-m (10-ft) -dia Stretched Fairing (-10L), Two-Stage Configuration (6306 PAF)

3.5 PAYLOAD FAIRING DOOR LOCATIONS

Each Delta II payload fairing can accommodate multiple access doors to support spacecraft servicing while on the pad. Because it is understood that customers may need access to items such as payload ordnance devices, electrical connectors, and fill-and-drain valves for payloads using liquid propellants, additional access doors can be installed on a mission-unique basis. Also, differing diameters or shapes for the access doors can be accommodated on a mission-unique basis. Access doors typically do not have acoustic blankets attached to their inboard surfaces, but can have them, on a mission-unique basis, to provide additional acoustic attenuation. Access door locations and sizes should be coordinated with Boeing Launch Services.

3.5.1 Delta II Metallic Fairing Door Locations

For the Delta II 9.5-ft-dia metallic fairing (-9.5), three door size types can be installed: 21.48 by 25 in., 21.86 by 25 in., and 25 by 25 in. These doors can be installed in the payload fairing as follows (Figure 3-19):

- Axial location
 - Circumferential centerline for the 21.48-by-25-in. door must be located at Delta II Station 400.48.
 - Circumferential centerline for the 21.86-by-25-in. door must be located at Delta II Station 346.42.
 - Circumferential centerline for the 25-by-25-in. doors must be located at Delta II Station 373.22 or 389.22.

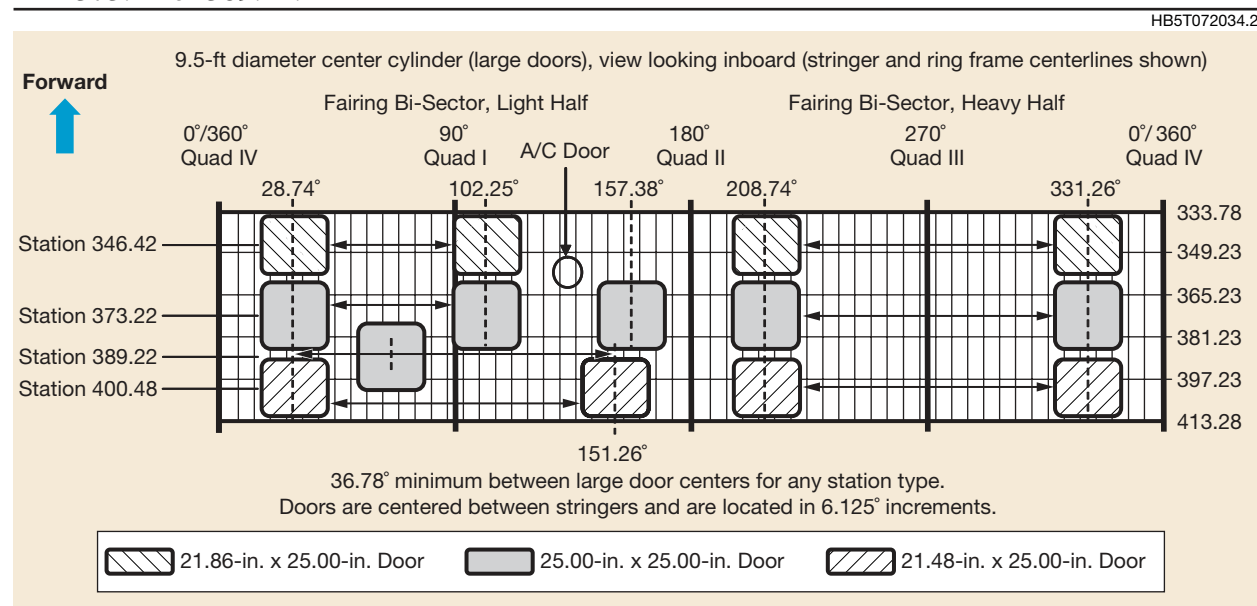


Figure 3-19. Allowable Access Door Locations for 9.5-ft-dia Metallic Fairing

- Circumferential location
 - 21.48-by-25-in. door and 25-by-25-in. door (located at Station 389.22) axial centerlines must be 28.74 deg to 151.26 deg for the light half and between 208.74 deg and 331.26 deg for the heavy half (Quad IV = 0 deg, Quad I = 90 deg) in increments of 6.13 deg (6.1 in.).
 - 21.86-by-25-in. door and 25-by-25-in door (located at Station 373.22) axial centerlines must be 28.74 deg to 102.25 deg and 157.38 deg (located at Station 373.22) for the light half and between 208.74 deg and 331.26 deg for the heavy half (Quad IV = 0 deg, Quad I = 90 deg) in increments of 6.13 deg (6.1 in.). The 25-by-25-in. door (located at Station 373.22) may also be located at 157.38 deg.
- Door-to-door spacing: Circumferential center-to-center spacing between doors must be a minimum of 36.8 deg (36.6 in.).
- Door orientation: Rectangular doors are oriented with their 21.48/21.86-long sides parallel to the axial direction.

3.5.2 Delta II Composite Fairing Door Locations

For the Delta II 10-ft-dia composite payload fairings (-10 and -10L), standard 24-in. diameter access doors can be installed, and can be located only in the cylindrical section of the fairing as follows (Figures 3-20 and 3-21):

- Axial location: Door circumferential centerlines must be located at least 45.72 cm (18.0 in.) above the boattail transition and at least 45.72 cm (18.0 in.) below the nose cone transition.
- Circumferential location: Door axial centerlines must be at least 59.31 cm (23.35 in.) from Quad II and IV axial centerlines, with the exception (for -10 only) that doors with circumferential centerlines between 45.72 cm (18.0 in.) and 143.51 cm (56.5 in.) aft of the nose cone transition must have axial centerlines at least 107.32 cm (42.25 in.) from the Quad II and IV axial centerlines. Door axial centerlines must be at least 51.44 cm (20.25 in.) from the air-conditioning door doubler edge.
- Door-to-door spacing: Center-to-center spacing between doors (axial and/or circumferential) must be at least 121.92 cm (48 in.).

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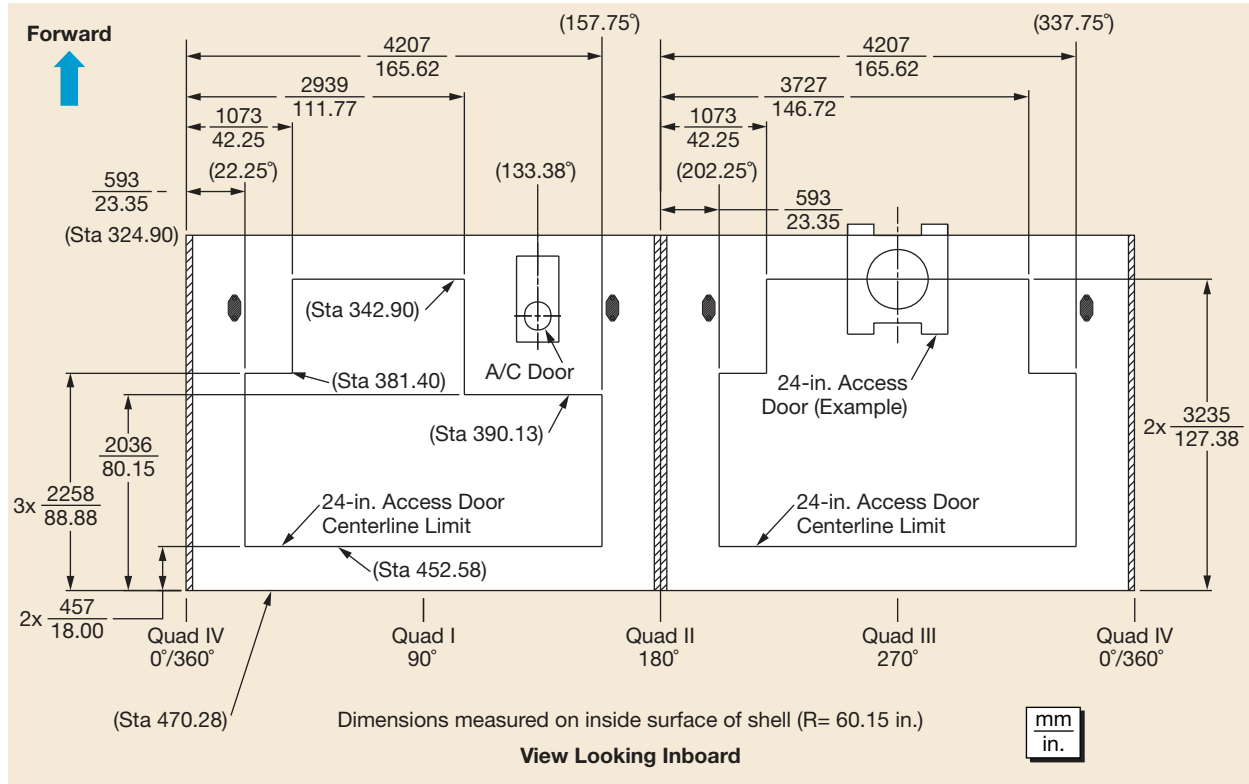


Figure 3-20. Allowable Access Door Locations for 10-ft-dia Composite Fairing

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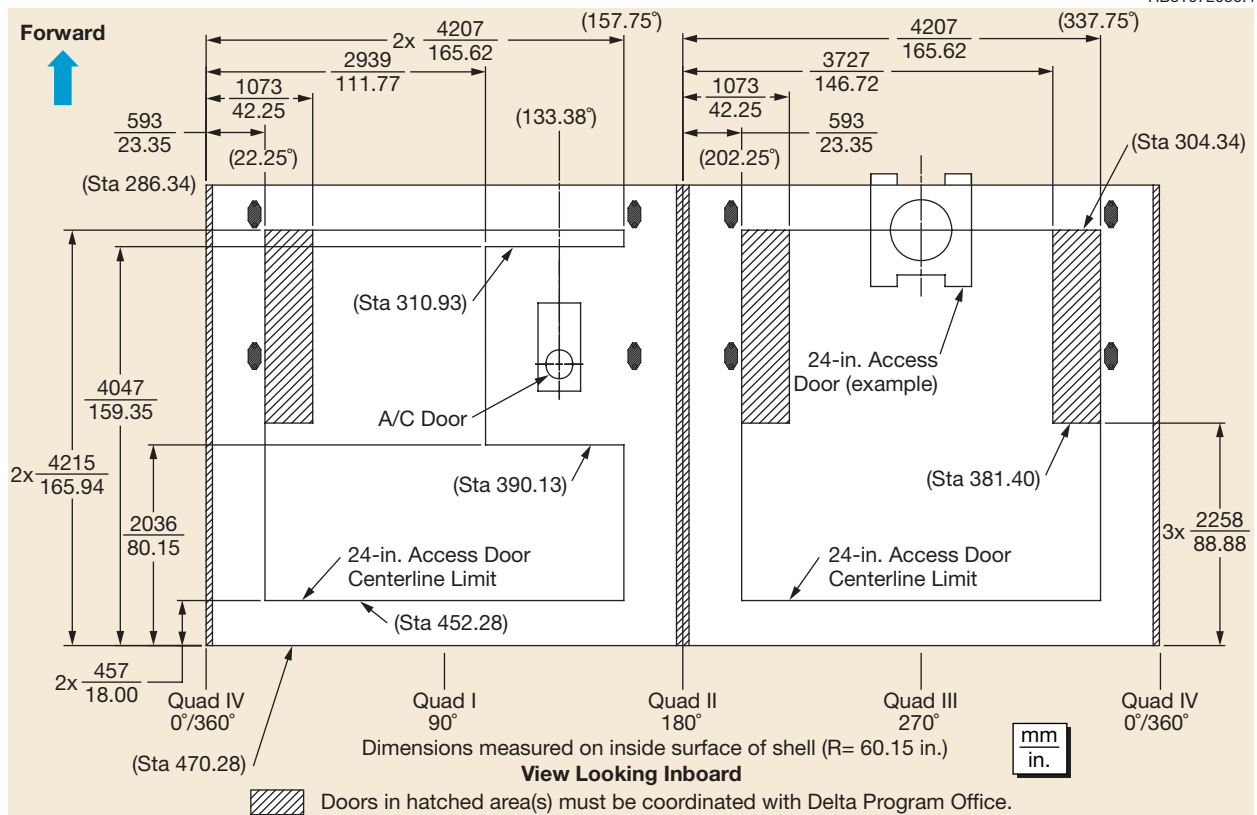


Figure 3-21. Allowable Access Door Locations for 10-ft-dia Stretched Composite Fairing

Section 4 PAYLOAD ENVIRONMENTS

This section describes the launch vehicle environments to which the spacecraft is exposed during prelaunch activities and launch. Section 4.1 discusses prelaunch environments for processing facilities at both eastern and western ranges. Section 4.2 presents the Delta II launch and flight environments for the spacecraft.

4.1 PRELAUNCH ENVIRONMENTS

4.1.1 Payload Air Conditioning and Gaseous Nitrogen (GN₂) Purge

The environment experienced by the payload during its launch site processing is carefully controlled for temperature, relative humidity, and cleanliness. This includes the payload processing conducted before it is installed in the ground handling can (see Figures 6-5 and 7-14). The ground handling can, with the payload inside, is subsequently transferred to the launch pad and hoisted into the mobile service tower (MST) white room. Before the spacecraft is mounted on to the launch vehicle, the MST white room is closed and the white room air-conditioning is stabilized. Mating to the second stage is completed, and the ground handling can is disassembled in sections.

4.1.1.1 Payload Air Conditioning

Air-conditioning is supplied to the spacecraft via an umbilical after the payload fairing is mated to the launch vehicle. The payload air-distribution system (Figure 4-1) provides air at the required temperature, relative humidity, and flow rate as measured at the end of the fairing duct hardline in the fixed umbilical tower (FUT). The air-distribution system uses a diffuser on the inlet air-conditioning duct at the fairing interface. The air-conditioning inlet is in the Quad I half of the fairing. Unique mission requirements or equipment should be coordinated with the Delta Program Office. If required, a deflector can be installed on the inlet to direct the airflow away from sensitive spacecraft components. The air can be supplied to the payload between a rate of 1300 to 1700 scfm. The air flows downward around the spacecraft and is discharged below the second stage through vents in the interstage. The air-conditioning umbilical is pulled away at liftoff by lanyard disconnects, and the access door on the fairing automatically closes.

If an environmental shroud is required around the spacecraft prior to fairing installation, it receives the same fairing air. The environmental

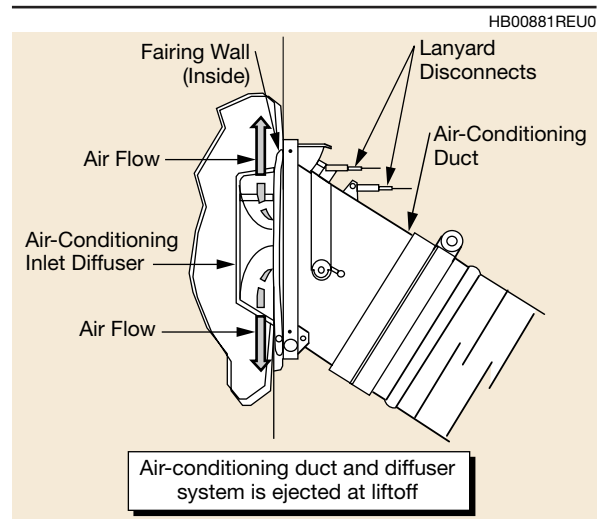


Figure 4-1. Payload Air Distribution System

shroud and payload work stand for Space Launch Complex-2 (SLC-2) are shown in Figure 4-2. A similar system for SLC-17 is shown in Figure 4-3.

The fairing air hardline downstream of the high-efficiency particulate air (HEPA) filter contains an inline particle counter for continuous particle count sampling. A separate backup environmental control unit is also provided for fairing air-conditioning redundancy. This unit is operated in a hot standby mode for automatic transfer during launch day. Both fairing air environmental control units are backed up by diesel generator power. If auxiliary air-conditioning (drag on) is required in addition to the fairing air, a battery cooling unit is available for supplemental cooling during pad processing prior to fairing closeout.

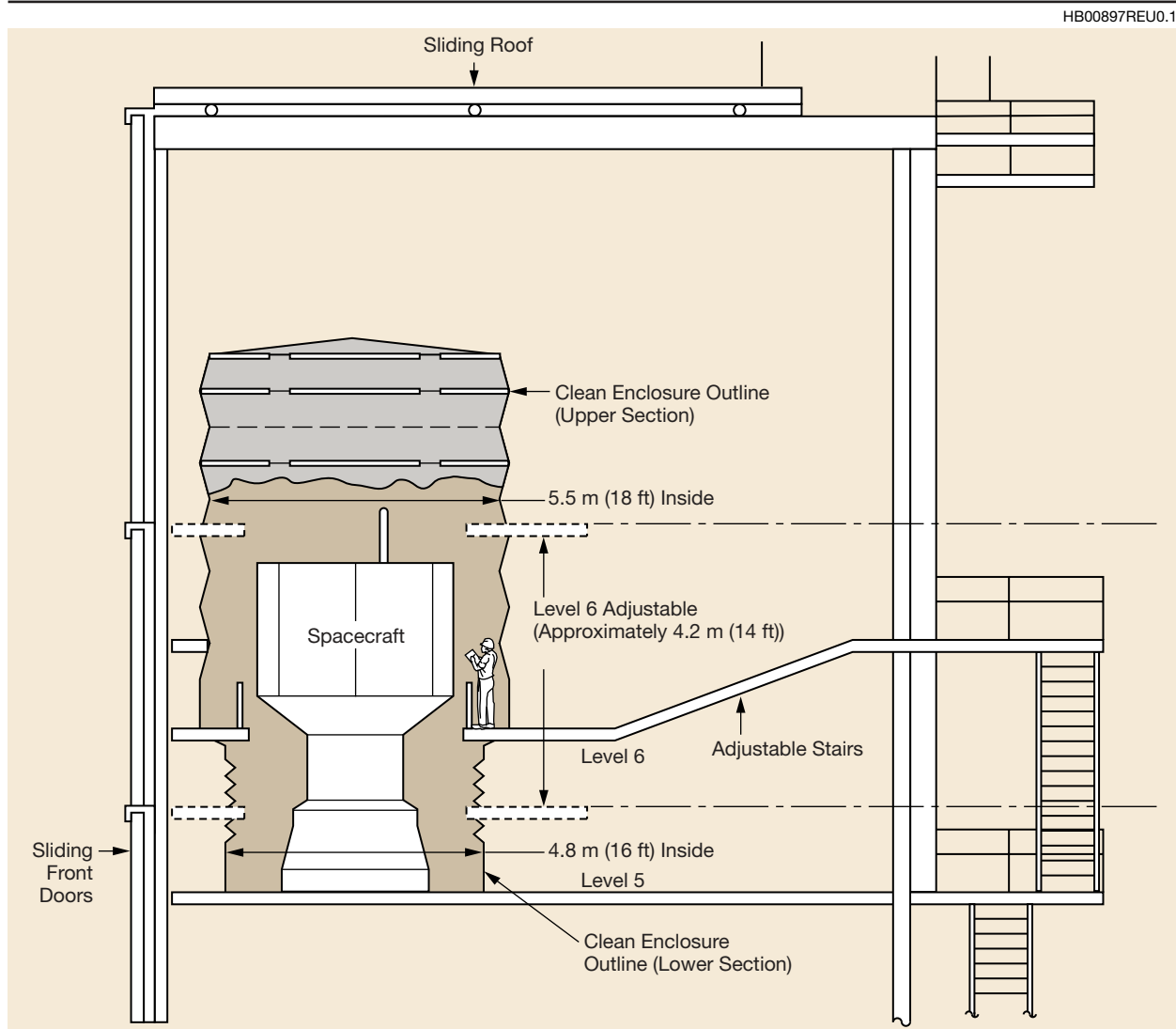


Figure 4-2. Environmental Shroud and Payload Workstand (SLC-2)

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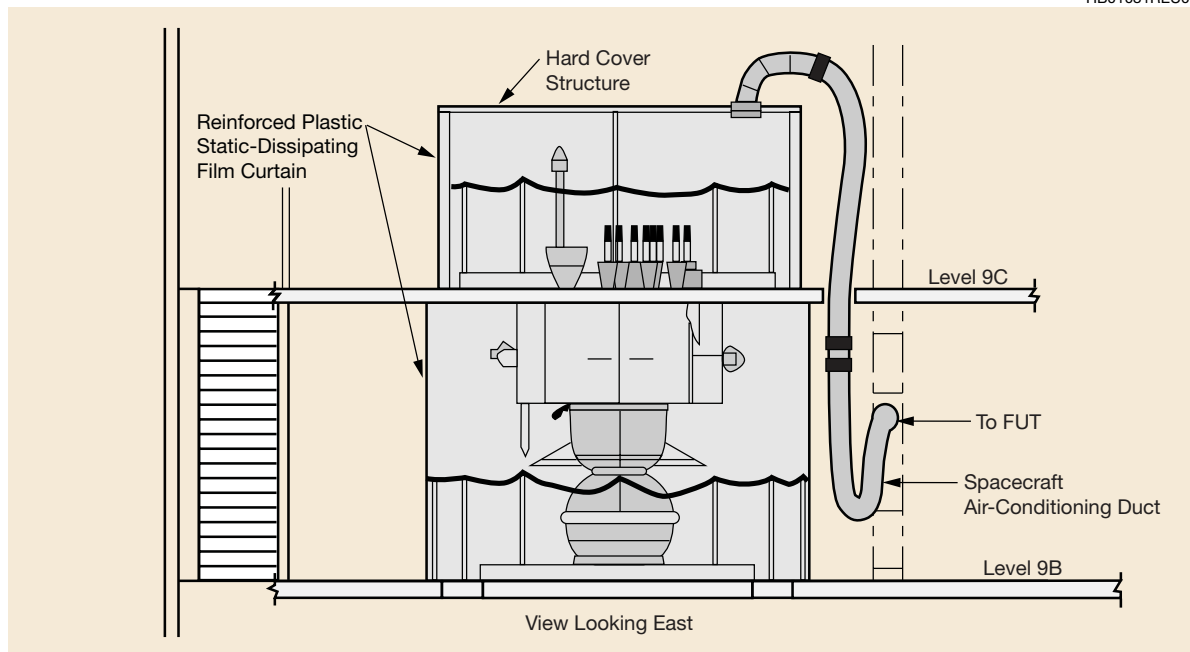


Figure 4-3. Environmental Shroud and Payload Workstand (SLC-17A and SLC-17B)

At SLC-17, the battery cooling unit is located on the MST and provides low-temperature air with limited humidity control through a 6-in. interface at level 9B. The system capabilities are detailed in Table 4-1. At SLC-2, a battery cooling system is available that can provide a maximum of 240 scfm through the T-0 umbilical on the second stage. System capabilities are detailed in Table 4-2.

Table 4-1. Eastern Range Facility Environments

SLC-17 Facility Environments				
Payload Environment	Temperature ⁽¹⁾	Relative Humidity ⁽¹⁾	Flow Rate	Cleanliness ⁽²⁾
Handling Can	Not controlled	Not controlled ⁽³⁾	N/A	Not controlled
MST White Room (all doors closed)	65°F to 75°F (18.3°C to 23.9°C)	30% to 50%	N/A	Class 100,000
Environmental shroud	45°F to 80°F ⁽⁴⁾ (7.2°C to 26.7°C) Controlled within ±2°F (±1°C)	30% to 50%, Controlled within ±5%	1000 to 1700 scfm, Controlled within ±100 scfm	Class 10,000 ⁽⁵⁾
Payload fairing interior prior to 2nd Stage Propellant Loading ⁽⁶⁾	45°F to 80°F ⁽⁴⁾ (7.2°C to 26.7°C) Controlled within ±2°F (±1°C)	30% to 50%, Controlled within ±5%	1300 to 1700 scfm, Controlled within ±100 scfm	Class 10,000 ⁽⁵⁾
Payload fairing interior during and after 2nd Stage Propellant Loading ⁽⁷⁾	53°F to 73°F ⁽⁴⁾ (11.7°C to 22.8°C) Controlled within ±2°F (±1°C)	30% to 50%, Controlled within ±5%	1300 to 1700 scfm, Controlled within ±100 scfm	Class 10,000 ⁽⁵⁾
Battery cooling air ⁽⁶⁾	50°F to 80°F (10.0°C to 26.7°C) Controlled within ±5°F (±2.8°C)	90% max (not controlled)	0 to 600 scfm	Class 10,000 ⁽⁵⁾

Notes:
⁽¹⁾ Temperature and relative humidity requirements can be accommodated between the ranges stated for each location.
⁽²⁾ Reference FED-STD-209E, Airborne Particulate Cleanliness Classes in Cleanrooms and Clean Zones, except as noted.
⁽³⁾ Dry nitrogen gas purge per MIL-P-27401C, Type 1, Grade B, during transport.
⁽⁴⁾ Fairing air temperatures outside of the specified ranges must be coordinated with the Delta Program Office.
⁽⁵⁾ Fairing interior cleanliness levels cleaner than class 10,000 must be coordinated with the Delta Program Office.
⁽⁶⁾ All conditions specified are inlet conditions. Temperatures and relative humidity are measured in the FUT at the flexible air conditioning duct inlet.
⁽⁷⁾ Measured inside the payload fairing.

Table 4-2. Western Range Facility and Transportation Environments

SLC-2 Facility Environments				
Location	Temperature ⁽¹⁾	Relative Humidity ⁽¹⁾	Flow Rate	Cleanliness ⁽²⁾
Handling Can	Not controlled	Not controlled ⁽³⁾	N/A	Not controlled
MST White Room (all doors closed)	65°F to 75°F (18.3°C to 23.9°C)	30% to 60%, Controlled within ±5%	N/A	Class 100,000
Environmental shroud	55°F to 70°F ⁽⁴⁾ (12.8°C to 21.1°C) Controlled within ±5°F (±2.8°C)	30% to 50%, Controlled within ±5%	1000 to 1700 scfm, Controlled within ±100 scfm	Class 10,000 ⁽⁵⁾
Payload fairing interior prior to 2 nd Stage Propellant Loading ⁽⁶⁾	55°F to 70°F ⁽⁴⁾ (12.8°C to 21.1°C) Controlled within ±5°F (±2.8°C)	30% to 50%, Controlled within ±5%	1300 to 1700 scfm, Controlled within ± 100 scfm	Class 10,000 ⁽⁵⁾
Payload fairing interior during and after 2 nd Stage Propellant Loading ⁽⁷⁾	55°F to 70°F ⁽⁴⁾ (12.8°C to 21.1°C) Controlled within ±5°F (±2.8°C)	30% to 50%, Controlled within ±5% ⁽⁸⁾	1300 to 1700 scfm, Controlled within ±100 scfm	Class 10,000 ⁽⁵⁾
Battery cooling air ⁽⁶⁾	45°F to 70°F (7.2°C to 21.1°C) Controlled within ±5°F (±2.8°C)	<80%, Non-condensing	≤ 240 scfm	Class 10,000 ⁽⁵⁾
Notes: ⁽¹⁾ Temperature and relative humidity requirements can be accommodated between the ranges stated for each location. ⁽²⁾ Reference FED-STD-209E, Airborne Particulate Cleanliness Classes in Cleanrooms and Clean Zones, except as noted. ⁽³⁾ Dry nitrogen gas purge per MIL-P-27401C, Type 1, Grade B, during transport. ⁽⁴⁾ Fairing air temperatures outside of the specified ranges must be coordinated with the Delta Program Office. ⁽⁵⁾ Fairing interior cleanliness levels cleaner than class 10,000 must be coordinated with the Delta Program Office. ⁽⁶⁾ All conditions specified are inlet conditions. Temperatures and relative humidity are measured in the FUT at the flexible air conditioning duct inlet. ⁽⁷⁾ Measured inside the payload fairing. ⁽⁸⁾ Humidity levels after tower rollback may be lower than 30%.				

4.1.1.2 Gaseous Nitrogen Purge

At SLC-17, GN₂ purge can be accommodated during hoist into the white room and/or through the air-conditioning duct after fairing installation. The GN₂ for the purge can be supplied from facility MIL-P-27401C, Type 1, Grade B nitrogen boil-off or customer-supplied k-bottles or dewars normally located at the base of the fixed umbilical tower (FUT). Purge gas control panel(s) are normally furnished by the customer.

At SLC-2, GN₂ purge can be accommodated during hoist into the white room and/or through the fairing AC Adapter (Figure 4-1) after fairing installation. GN₂ purge can also be accommodated through the T-0 umbilical on the second-stage miniskirt from spacecraft erection through liftoff. The GN₂ purge gas and regulator panel are normally provided by the customer. Typical spacecraft gas purge accommodations are detailed in Figure 4-4.

Various payload processing facilities are available at the launch site for use by the customer. The facilities used depend on spacecraft program requirements. See Section 6 for descriptions of Eastern Range and Section 7 for Western Range facilities.

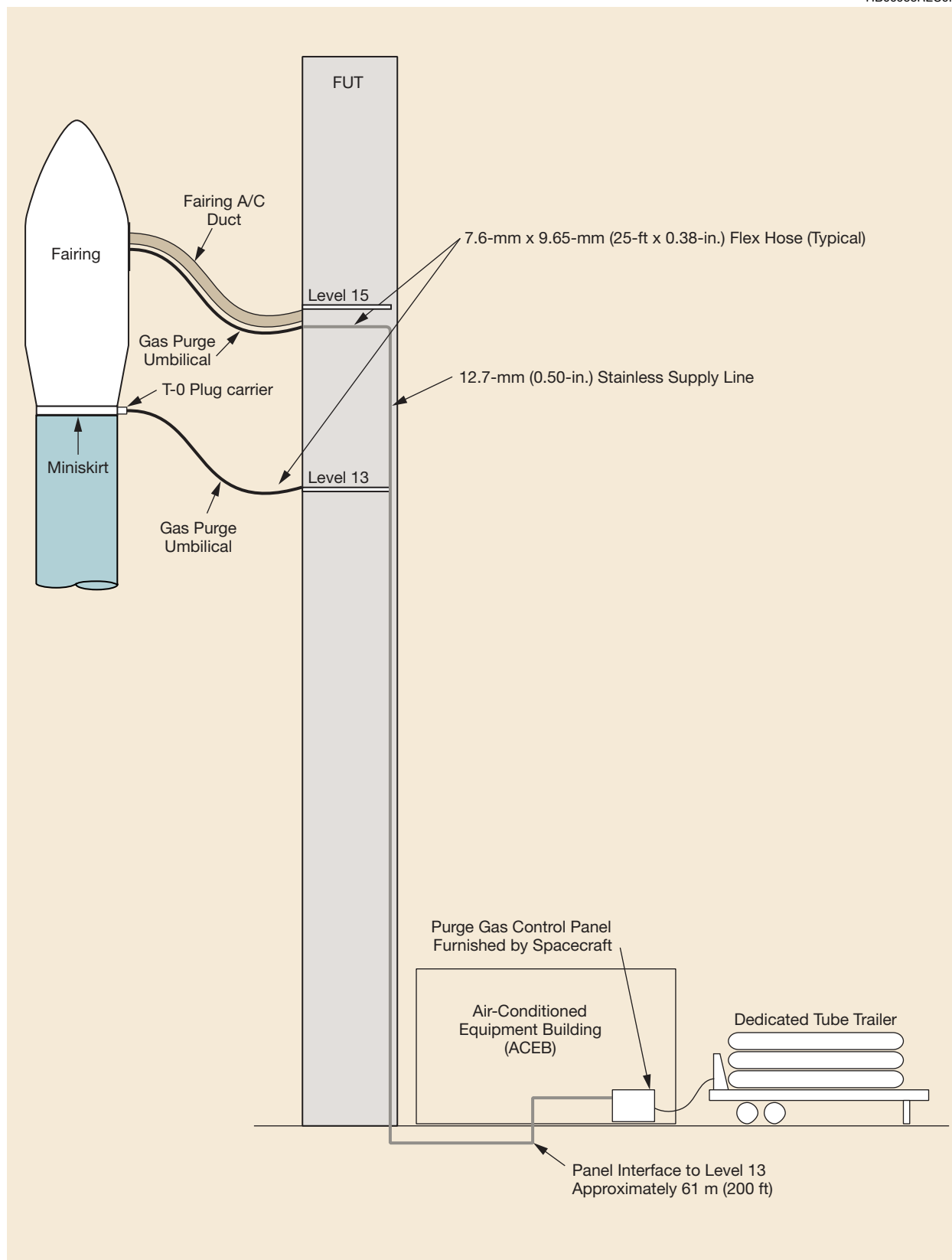


Figure 4-4. Payload Gas Purge Accommodations (Typical at SLC-2 Shown)

The spacecraft GN₂ purge connection must be located in the Quad I half of the fairing so that the tubing can be routed through the A/C inlet door. The purge connection must be within 5 deg of the Quad I fairing centerline and parallel to Quad I. An access door is required for mating the purge tube to the connector. No surrounding spacecraft intrusions are allowed within a 30-deg half-cone angle separation clearance envelope at the mated interface. The location of the spacecraft purge connector interface as measured radially from vehicle centerline is typically 32-in. to 46-in. for the 9.5-ft fairing and 42-in. to 51-in. for the 10-ft composite fairing. Details of a typical purge system interface is shown in Figure 4-5.

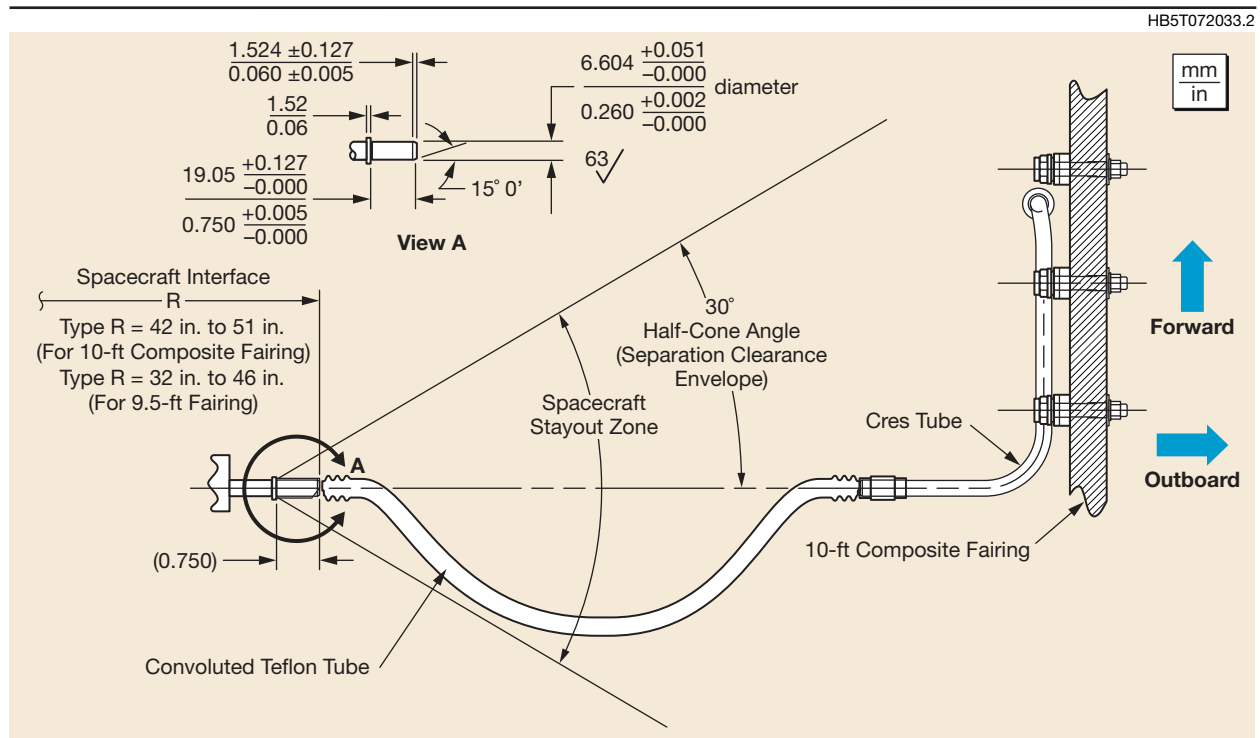


Figure 4-5. GN₂ Purge System—Typical Interface Details

4.1.2 MST White Room

Located at the upper levels within the MST, the environmentally controlled white room has provisions for maintaining spacecraft cleanliness. White room environments are listed in Table 4-1 for pads A and B at SLC-17 and in Table 4-2 for SLC-2 at Vandenberg Air Force Base (VAFB).

4.1.3 Radiation and Electromagnetic Environments

The Delta II transmits launch vehicle telemetry and beacon signals on several frequencies to the appropriate range tracking stations. It also has uplink capability to onboard command receiver decoders (CRDs) for command destruct capability. Two S-band telemetry systems are provided (one each on the second and third stages), as well as two CRD systems on the second

stage and a C-band transponder (beacon) on the second stage. The radiation characteristics of these systems are shown in Table 4-3. The RF systems are switched on prior to launch and remain on until stage separation and battery depletion. Payload launch environment data, such as low- and high-frequency vibration, acceleration transients, shock velocity increments, and health status, may also be obtained from the launch vehicle telemetry system.

Table 4-3. Delta II Transmitter Characteristics

	Second-stage T/M Radiation Characteristics	Third-stage T/M Radiation Characteristics	Second-stage C-band Beacon Characteristics
Transmitter			
Nominal frequency	2241.5 MHz	2252.5 MHz	5765 MHz (transmit) 5690 MHz (receive)
Power output	2.0 W min	5.0 W min	400 W min
Modulation bandwidth	±160 kHz at 20 dB ±650 kHz at 60 dB	±70 kHz at 20 dB ±250 kHz at 60 dB	6 MHz at 6 dB
Stability	+67 kHz max	+68 kHz max	3 MHz max
Antenna			
Type	Cavitybacked slot	Circumferential belt	Transverse slot, dipole loaded
Polarization	Essentially linear parallel to booster roll axis	Essentially linear parallel to booster roll axis	Left-hand circular
Pattern	Nearly omnidirectional	Nearly omnidirectional	Nearly omnidirectional
Gain	+2.35 dB max	+3 dB max	+6 dB max

At both the eastern and western ranges, the electromagnetic environment that the payload is exposed to includes emissions resulting from general purpose RF emitters such as broadcast services, cellular phones and facilities 2-way communications radios; emissions derived from the launch vehicle transmitter systems and their associated antennas; and emissions due to the operation of on-site range radars.

The general purpose on-site RF emitters are generally controllable, however, off-site emitters such as maritime emitters, broadcast emitters and overhead flying aircraft emitters generally are not. The launch pads are typically exposed to a controllable on-site RF emitter environment of 20 V/m maximum at frequencies from 14 KHz to 40 GHz from general purpose emitters. From onboard launch vehicle S-band and C-band emitters, the RF field value is typically 40 V/m maximum at the payload separation plane.

Onsite S-band and C-band range radars are generally controlled to a maximum of 40 V/m at the launch vehicle including during ascent. The maximum radar derived RF environment at the launch site is controlled through coordination with the range and with protective masking of range radars. If reduced levels from the on-site range radars are desired, they should be identified early in the integration process and coordinated with the range.

The maximum allowable spacecraft radiated emissions at the spacecraft/vehicle separation plane are provided in Figure 4-6. Spacecraft are permitted to radiate inside the fairing provided that the emissions do not exceed the maximum level deemed safe for launch vehicle avionics and ordnance circuits. Operation times during launch processing must be coordinated with the Delta Program Office to evaluate noninterference/safety concerns. The RF field strength inside the

fairing is a function of the antenna's gain, location, and other physical characteristics of the spacecraft; and the RF properties of the fairing with the acoustic blanket accounted for. Upon request, the Delta Program Office will calculate these levels as early as possible in the integration process using spacecraft-supplied data, empirical and analytic formulas that account for cavity resonances and other influencing factors if applicable. Analysis is also required if the spacecraft does not intend to radiate within the fairing, but cannot verify a dual-inhibit design preventing inadvertent radiation. An RF compatibility analysis is also performed to verify that the vehicle and satellite transmitter frequencies do not have interfering intermodulation products or image rejection problems.

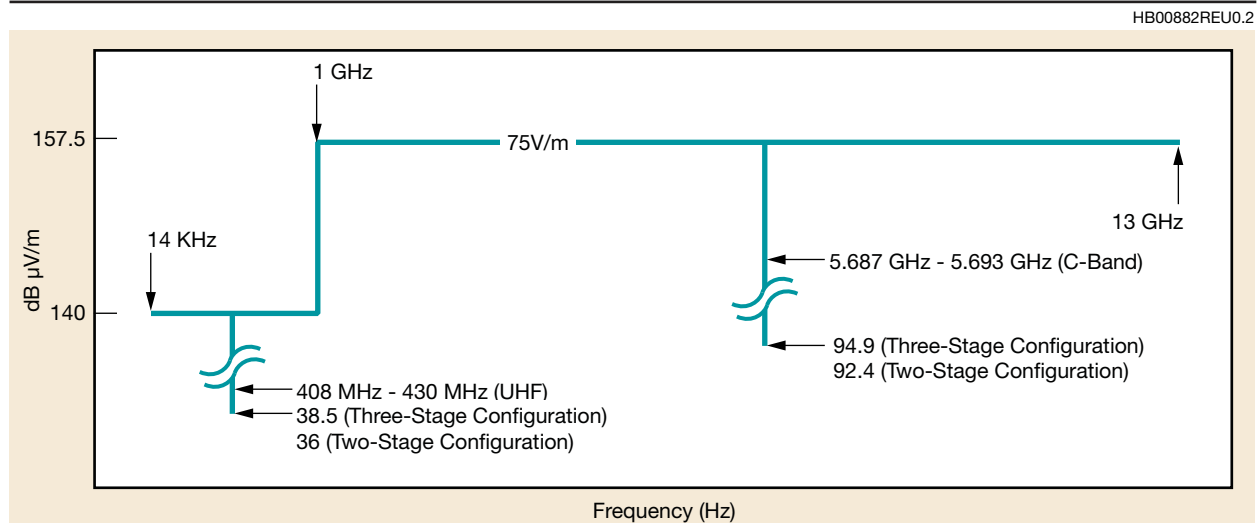


Figure 4-6. Maximum Allowable Payload Radiated Emissions at the Payload/Launch Vehicle Separation Plane

4.1.4 Electrostatic Potential

During ground processing, the spacecraft must be equipped with an accessible ground attachment point to which a conventional alligator-clip ground strap can be attached. Preferably, the ground attachment point is located on or near the base of the spacecraft, at least 31.8 mm (1.25 in.) above the separation plane. The vehicle/spacecraft interface provides the conductive path for grounding the spacecraft to the launch vehicle. Therefore, dielectric coating should not be applied to the spacecraft interface. The electrical resistance of the spacecraft to the payload attach fitting (PAF) interface surfaces must be 0.0025 ohm or less and is verified during spacecraft-to-PAF mating. (Reference MIL-B-5087B, Class R.)

4.1.5 Contamination and Cleanliness

Delta II payloads cleanliness conditions represent the minimum available. The following guidelines and practices from prelaunch through spacecraft separation provide the minimum class 100,000 cleanliness conditions (per Federal Standard 209E):

- A. Precautions are taken during manufacture, assembly, test, and shipment to prevent contaminant accumulations in the Delta II upper-stage area, fairing, and PAF.
- B. Encapsulation of the payload into the handling can is performed at the payload processing facility that is environmentally controlled to class 100,000 conditions. All handling equipment is cleanroom compatible and is cleaned and inspected before it enters the facility. These environmentally controlled conditions are available for all remote encapsulation facilities and include SLC-17 and SLC-2. The handling can that is used to transport the payload to the white room provides environmental protection for the payload.
- C. The fairing is cleaned using alcohol and then inspected for cleanliness prior to spacecraft encapsulation. Six levels of cleanliness are defined below. The standard level for a typical mission using the 9.5-ft fairing is VC2, and the standard level for a typical mission using the 10-ft fairing is VC3. Other cleanliness levels are available but need to be coordinated with the Delta Program Office. Table 4-4 provides Boeing Cleanliness Specification STP0407 visible cleanliness (VC) levels with their NASA SN-C-0005 equivalency.
- D. The payload attach fitting and second-stage guidance section are cleaned to VC2.

Table 4-4. Cleanliness Level Definitions

Boeing STP0407-0X	NASA SN-C-0005
VC 1	None
VC 2	VC Standard (9.5-ft fairing)
VC 3	VC Highly Sensitive, Standard Level (10-ft fairing)
VC 4	VC Sensitive + UV (Closest equivalent, Boeing is more critical)
VC 5	VC Highly Sensitive
VC 6	VC Highly Sensitive + UV
VC 7	VC Highly Sensitive + NVR Level A

Cleanliness Level Definitions

VC 1—All surfaces shall be visibly free of all particulates and nonparticulates visible to the normal unaided/corrected-vision eye. Particulates are defined as matter of miniature size with observable length, width, and thickness. Nonparticulates are film matter without definite dimension. Inspection operations shall be performed under normal shop lighting conditions at a maximum distance of 0.915 m (3 ft).

VC 2—All surfaces shall be visibly free of all particulates and nonparticulates visible to the normal unaided/corrected-vision eye. Particulates are defined as matter of miniature size with observable length, width, and thickness. Nonparticulates are film matter without definite dimension. Inspection operations shall be performed at incident light levels of 538.2 lux (50 foot-candles [fc]) and observation distances of 1.52 m to 3.05 m (5 ft to 10 ft).

VC 3—All surfaces shall be visibly free of all particulates and nonparticulates visible to the normal unaided/corrected-vision eye. Particulates are identified as matter of miniature size with observable length, width, and thickness. Nonparticulates are film matter without definite

dimension. Incident light levels shall be 1076.4 lux to 2152.8 lux (100 fc to 200 fc) at an observation distance of 45.2 cm (18 in.) or less.

VC 4—All surfaces shall be visibly free of all particulates and nonparticulates visible to the normal unaided/corrected-vision eye. Particulates are identified as matter of miniature size with observable length, width, and thickness. Nonparticulates are film matter without definite dimension. This level requires no particulate count. The source of incident light shall be a 300-W explosion-proof droplight held at distance of 1.52 m (5 ft), maximum, from the local area of inspection. There shall be no hydrocarbon contamination on surfaces specifying VC 4 cleanliness.

VC 5—All surfaces shall be visibly free of all particulates and nonparticulates visible to the normal unaided/corrected-vision eye. Particulates are identified as matter of miniature size with observable length, width, and thickness. Nonparticulates are film matter without definite dimension. This level requires no particulate count. Incident light levels shall be 1076.4 lux to 2152.8 lux (100 fc to 200 fc) at an observation distance of 15.2 cm to 45.7 cm (6 in. to 18 in.). Cleaning must be done in a class 100,000 or better cleanroom.

VC 6—All surfaces shall be visibly free of all particulates and nonparticulates visible to the normal unaided/corrected-vision eye. Particulates are identified as matter of miniature size with observable length, width, and thickness. Nonparticulates are film matter without definite dimension. This level requires no particulate count. Incident light levels shall be 1076.4 lux to 2152.8 lux (100 fc to 200 fc) at an observation distance of 15.2 cm to 45.7 cm (6 in. to 18 in.). Additional incident light requirements are 8 W minimum of long-wave ultraviolet (UV) light at 15.2-cm to 45.7-cm (6-in. to 18-in.) observation distance in a darkened work area. Protective eyewear may be used as required with UV lamps. Cleaning must be done in a class 100,000 or better cleanroom.

VC 7—All surfaces shall be visibly free of all particulates and nonparticulates visible to the normal unaided/corrected-vision eye. Particulates are identified as matter of miniature size with observable length, width, and thickness. Nonparticulates are film matter without definite dimension. This level requires no particulate count. Incident light levels shall be 1076.4 lux to 2152.8 lux (100 fc to 200 fc) at an observation distance of 15.2 cm to 45.7 cm (6 in. to 18 in.). Cleaning must be done in a class 100,000 or better cleanroom. The nonvolatile residue (NVR) is to be one microgram or less per square centimeter (one milligram or less per square foot) of surface area as determined by the laboratory using a minimum of two random NVR samples per quadrant per bisector or trisector.

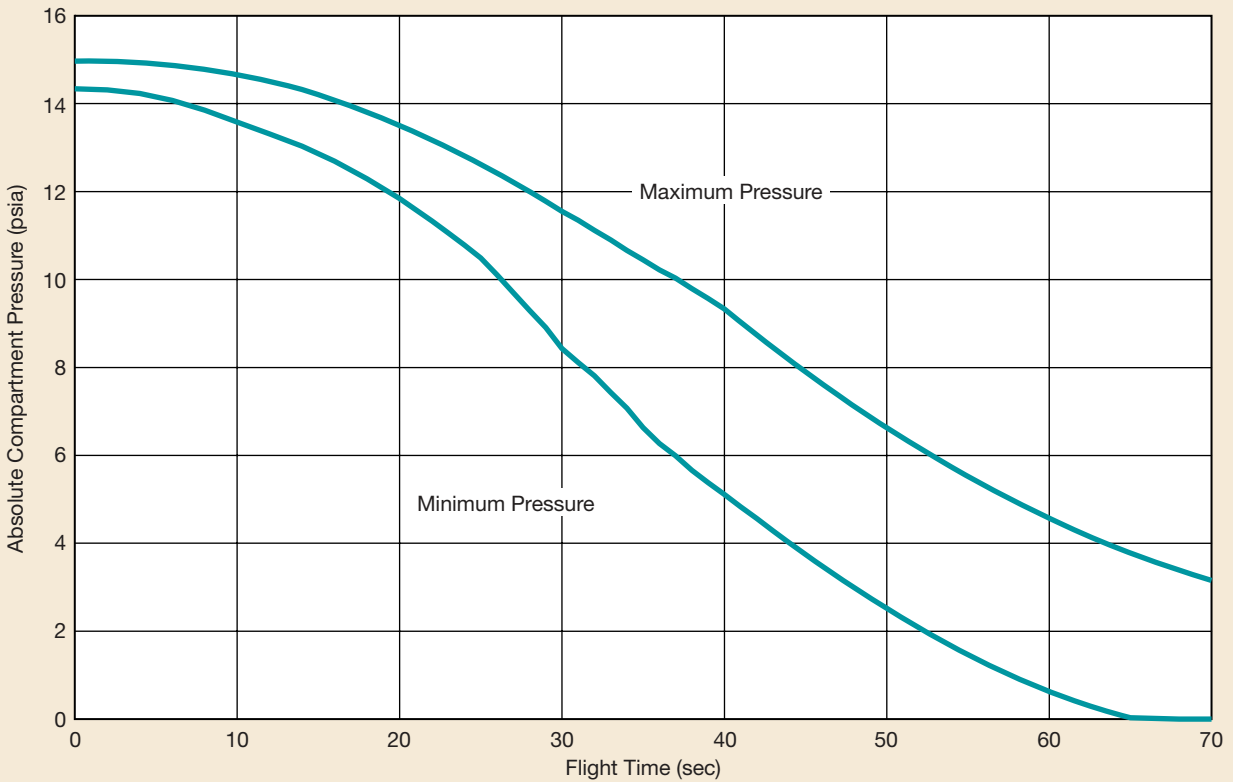
- E. Personnel and operational controls are employed during spacecraft encapsulation to maintain spacecraft cleanliness.
- F. The customer may place a protective barrier (bag) over the spacecraft prior to encapsulation in the handling can.
- G. A contamination barrier (bag) is installed around the handling can immediately following encapsulation operations. An outer bag is installed for transportation. A nitrogen purge is provided to the handling can during transport.
- H. A payload environmental shroud can be provided in the white room for the spacecraft prior to fairing installation. This shroud enables the spacecraft to be showered with class 10,000 fairing air at the Western Range and class 5,000 at the Eastern Range.

4.2 LAUNCH AND FLIGHT ENVIRONMENTS

4.2.1 Fairing Internal Pressure Environment

As the Delta II vehicle ascends through the atmosphere, the fairing is vented through leak paths in the vehicle and a dedicated vent opening on the interstage. The extremes of internal pressure during ascent are presented in Figure 4-7 for all Delta II vehicles (732X, 742X, 792XH, and 792X), including any dual-payload mission where a dual-payload attach fitting (DPAF) is utilized. The maximum expected pressure decay rate inside the compartment is -0.6 psi/sec.

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Flight Time (sec)	Minimum Pressure Pmin (psi)	Maximum Pressure Pmax (psi)	Flight Time (sec)	Minimum Pressure Pmin (psi)	Maximum Pressure Pmax (psi)	Flight Time (sec)	Minimum Pressure Pmin (psi)	Maximum Pressure Pmax (psi)
0	14.34	14.97	31	8.11	11.35	51	2.29	6.39
2	14.31	14.97	32	7.81	11.12	52	2.07	6.17
4	14.23	14.93	33	7.43	10.90	53	1.87	5.95
6	14.07	14.87	34	7.07	10.66	54	1.66	5.73
8	13.85	14.78	35	6.63	10.45	55	1.46	5.52
10	13.58	14.66	36	6.27	10.22	56	1.28	5.32
12	13.31	14.51	37	5.99	10.03	57	1.10	5.12
14	13.03	14.32	38	5.66	9.79	58	0.94	4.93
16	12.69	14.08	39	5.38	9.57	59	0.78	4.75
18	12.29	13.81	40	5.11	9.33	60	0.63	4.57
20	11.84	13.51	41	4.83	9.03	61	0.49	4.40
22	11.33	13.17	42	4.56	8.74	62	0.36	4.23
23	11.06	12.99	43	4.28	8.45	63	0.24	4.07
24	10.78	12.81	44	4.01	8.17	64	0.13	3.92
25	10.49	12.62	45	3.74	7.90	65	0.03	3.78
26	10.10	12.42	46	3.49	7.63	66	0.02	3.64
27	9.70	12.22	47	3.23	7.37	67	0.01	3.51
28	9.30	12.00	48	2.99	7.11	68	0.00	3.39
29	8.91	11.78	49	2.75	6.87	70	0.00	3.15
30	8.43	11.55	50	2.52	6.63			

Figure 4-7. Delta II Payload Fairing Compartment Absolute Pressure Envelope

4.2.2 Thermal Environment

Prior to and during launch, the Delta II payload fairing and upper stages contribute to the thermal environment of the spacecraft.

4.2.2.1 Payload Fairing Thermal Environment. Upon payload fairing (PLF) installation, air-conditioning is provided at a typical temperature range as stated in Tables 4-1 and 4-2, depending on mission requirements. Variations in temperature range can be accommodated and should be coordinated with the Delta Program Office.

The ascent thermal environments of the Delta II fairing surfaces facing the payload, based on historical flight data, are shown in Figures 4-8 and 4-9. Temperatures are provided for both the PLF conical section and the cylindrical section. PLF inboard-facing surface emissivity values are also provided. All temperature histories presented are based on a worst-case trajectory, ignoring expansion cooling effects of ascent.

The acoustic blankets provide a relatively cool radiation environment by effectively shielding the spacecraft from ascent heating in blanketed areas. Figures 4-8 and 4-9 depict the areas of the various Delta II fairings that are typically blanketed. There may be slight variations in blanket coverage areas based on mission-unique requirements. Inclusion of an RF window in the 2.9-m (9.5-ft) PLF conical section results in a local increase in acoustic blanket temperature inboard of the RF window, as shown in Figure 4-8.

The fairing skin temperature is representative of the radiation environment to the spacecraft in unblanketed areas such as the air-conditioning inlet door, unblanketed access doors, and blanket cutout regions. Maximum skin temperatures are shown in Figures 4-8 and 4-9.

The 2.9-m (9.5-ft) fairing frame temperatures are somewhat less severe than skin temperatures. Information regarding frame locations, exposure, and temperature history is available on request.

Unless otherwise requested, fairing jettison will occur shortly after the theoretical free molecular heating for a flat plate normal to the free stream drops below $0.1 \text{ Btu/ft}^2\text{-sec}$ (1135 W/m^2) based on the 1962 U.S. standard atmosphere.

4.2.2.2 On-Orbit Thermal Environment. During coast periods, the launch vehicle can be oriented to meet specific sun angle requirements. A slow roll during a long coast period can also be used to moderate orbital heating and cooling. The roll rate for thermal control is typically between 1 and 3 deg/sec.

4.2.2.3 Payload/Launch Vehicle Interface. The customer is required to provide interface geometry, thermal properties, and temperatures for the injection period assuming an adiabatic interface. The Delta Program Office will provide launch vehicle interface temperatures based on payload interface and preliminary mission analysis (PMA) or detailed test objective (DTO) sun-angle data.

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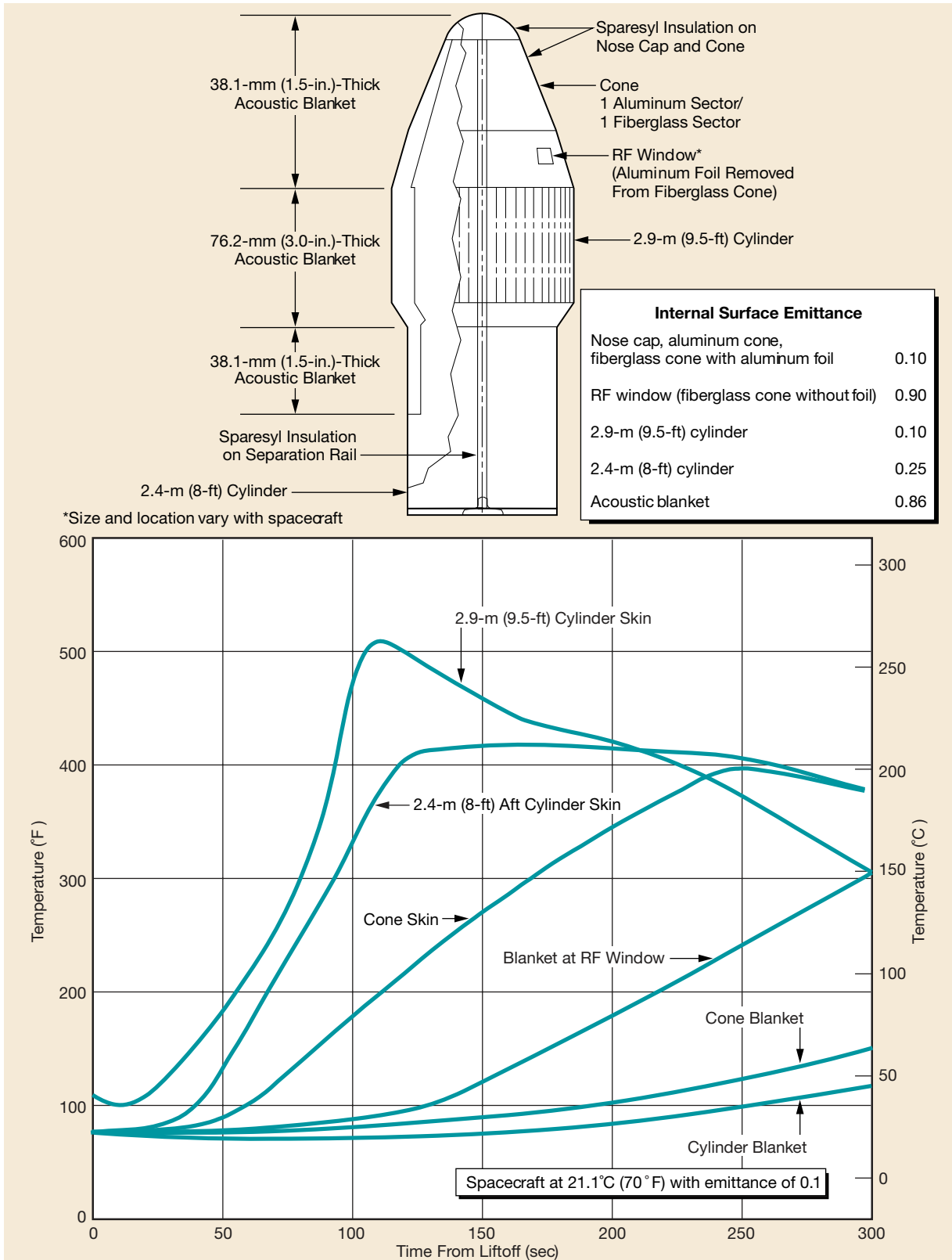


Figure 4-8. Predicted Maximum Internal Wall Temperature and Internal Surface Emittance (9.5-ft Fairing)

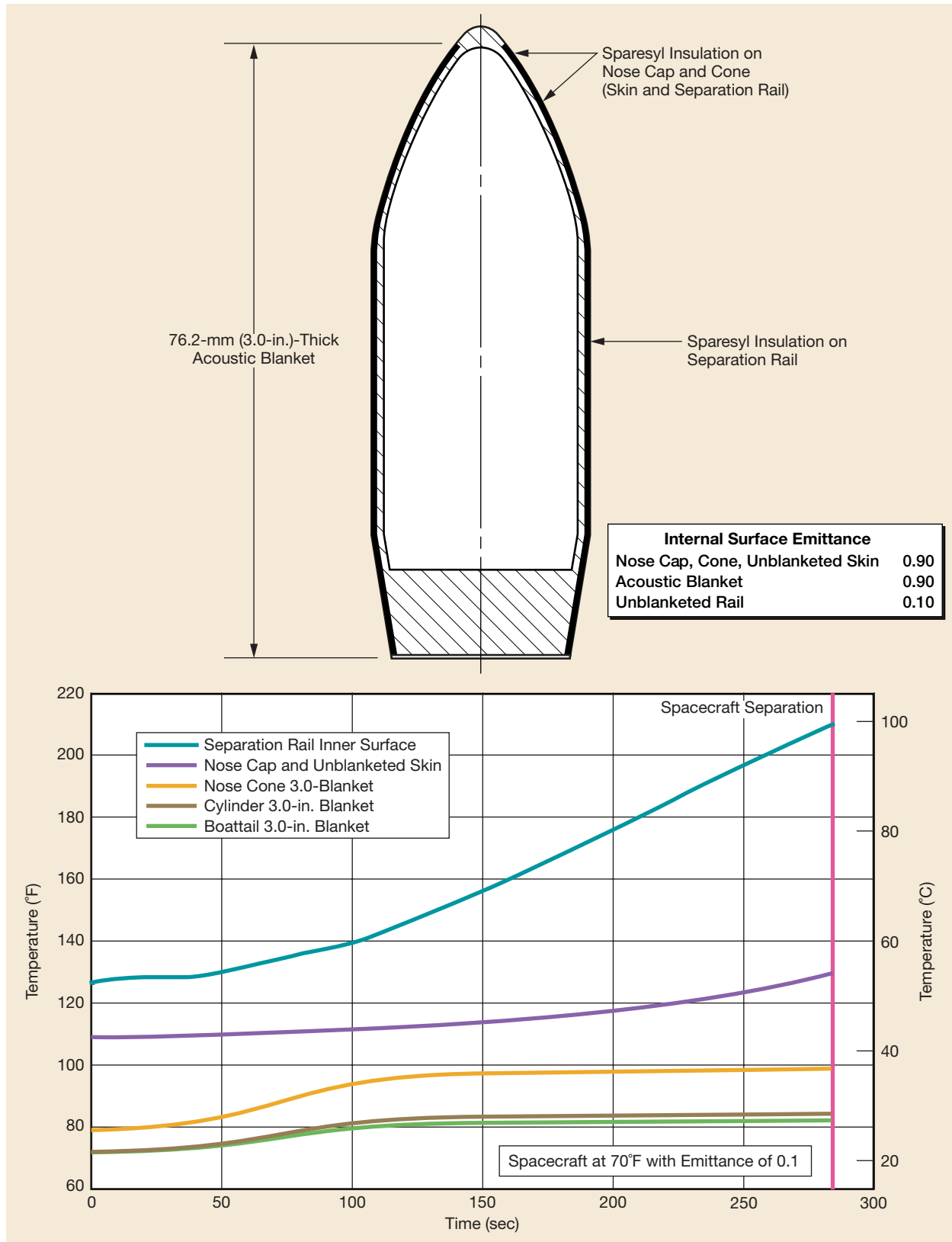
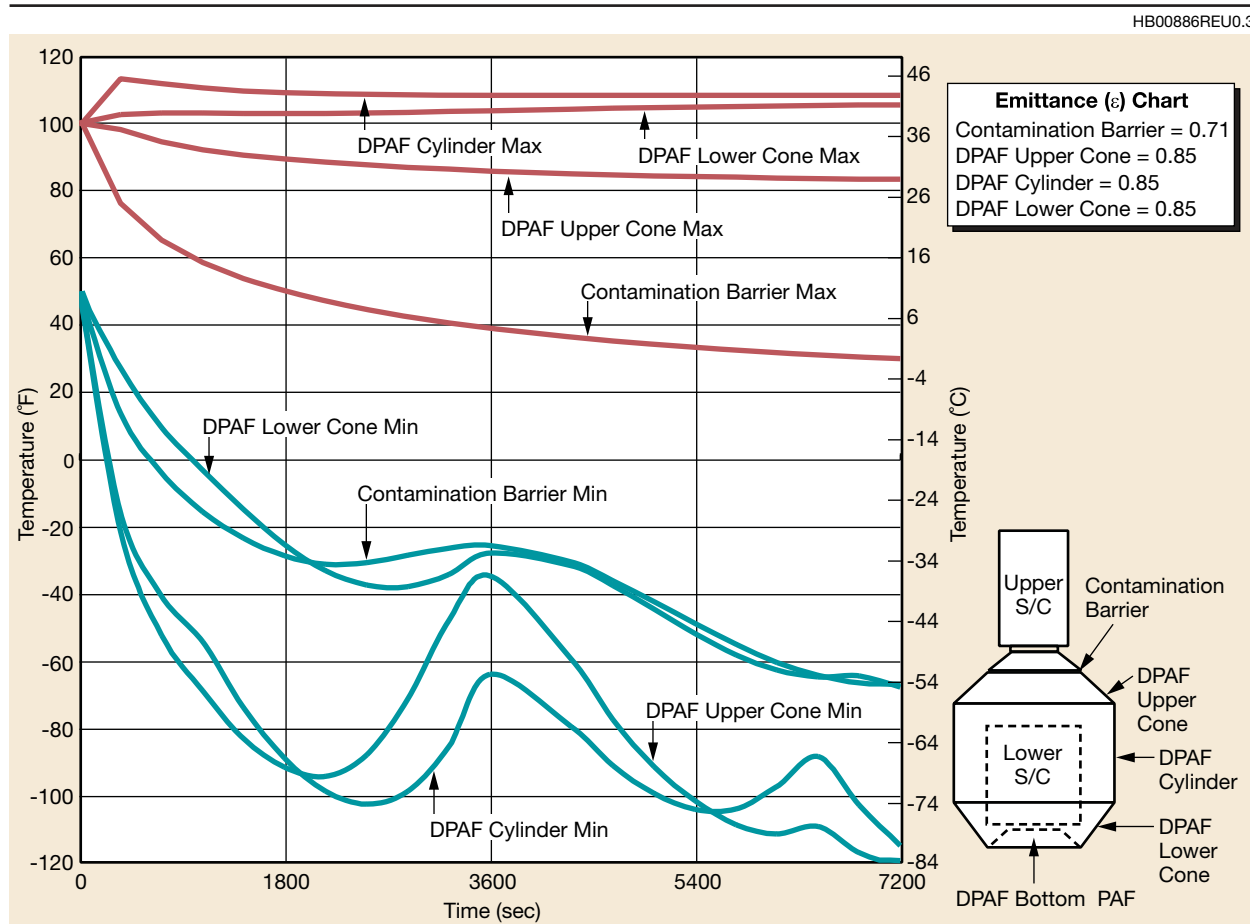


Figure 4-9. Predicted Maximum Internal Wall Temperature and Internal Surface Emittance (10-ft Fairing, Standard or Stretched)

4.2.2.4 Dual Payload Attach Fitting (DPAF) Thermal Environment. The DPAF is encompassed by the 3-m (10-ft) composite fairing, and the initial internal DPAF thermal environment (until fairing separation) is based on the fairing environment as detailed in Section 4.2.2.1.

The transfer orbit thermal environments of the Delta II internal DPAF surfaces are shown in Figure 4-10. Maximum and minimum temperatures for the internal surface, based on worst-case sun angles, are predicted for the time of fairing separation until DPAF separation. Mission-specific temperatures will be determined based on PMA or DTO sun-angle data.

From the time of fairing separation to DPAF separation, the lower spacecraft will experience a thermal radiation environment represented by the internal DPAF temperatures shown in Figure 4-10.



4.2.2.5 Third-Stage Induced Thermal Environments. The payload receives convective heat energy from the third-stage spin rocket plumes during burn and radiant heat energy from the third-stage motor plume during burn. The third-stage spin rocket plumes subject the spacecraft to a maximum heat flux of 2840 W/m^2 ($0.25 \text{ Btu/ft}^2\text{-sec}$) at the payload/third stage

separation plane for the Star-48B motor and 4771 W/m^2 ($0.42 \text{ Btu/ft}^2\text{-sec}$) for the Star-37FM using 1KS190 spin rockets. For the 1KS210 spin rockets, maximum heat flux is 6248 W/m^2 ($0.55 \text{ Btu/ft}^2\text{-sec}$) at the payload/third-stage separation plane for the Star-48B motor and 10451 W/m^2 ($0.92 \text{ Btu/ft}^2\text{-sec}$) for the Star-37FM. This heat flux is a pulse of 1-sec duration.

The Star-48B third-stage motor plume subjects the payload to a maximum heat flux of 2044 W/m^2 ($0.18 \text{ Btu/ft}^2\text{-sec}$) during the 87-sec burn. Plume heat flux is plotted versus radial distance in Figure 4-11. The variation of the heat flux with time during third stage burn is shown in Figure 4-12. The Star-37FM third-stage motor plume subjects the payload to a maximum heat flux of 3634 W/m^2 ($0.32 \text{ Btu/ft}^2\text{-sec}$) during the 65-sec burn. Plume heat flux is plotted versus radial distance in Figure 4-13. The variation of the heat flux with time during third-stage burn is shown in Figure 4-14.

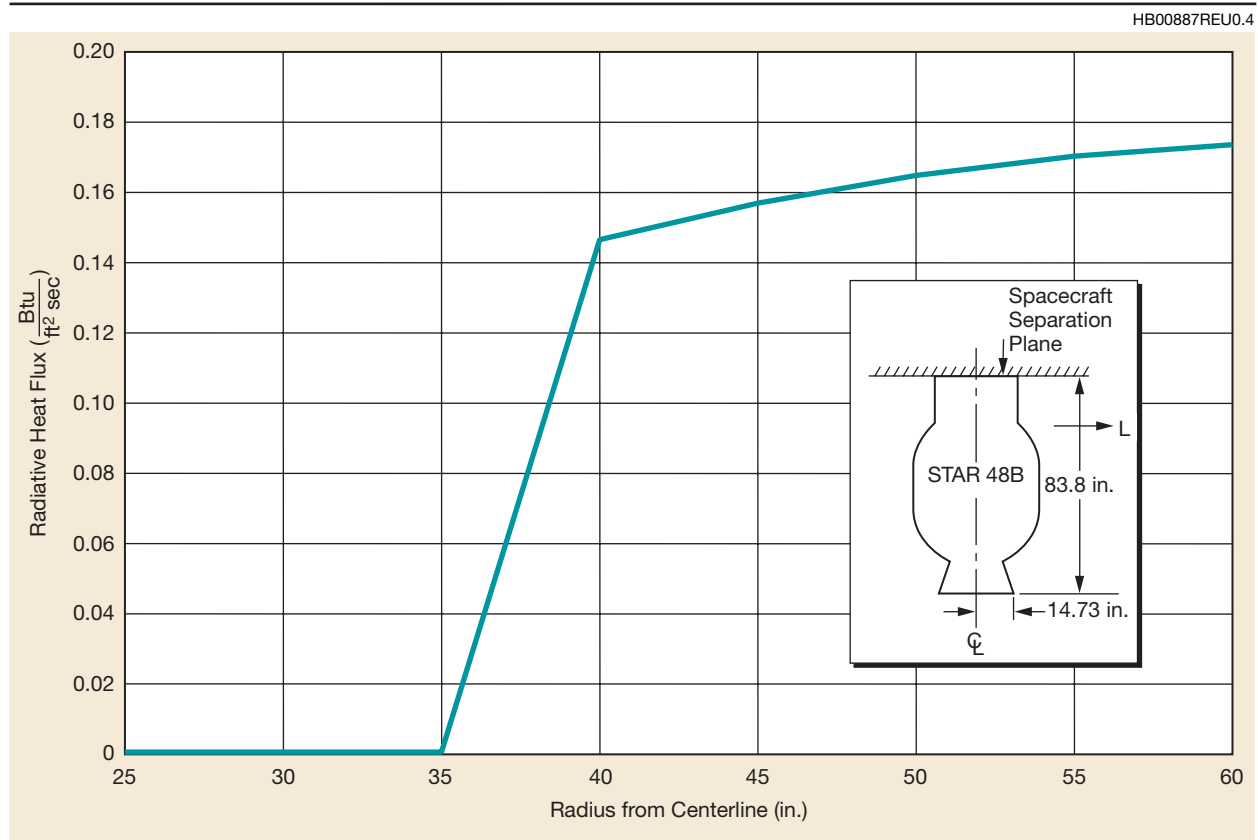


Figure 4-11. Predicted Star-48B Plume Radiation at the Spacecraft Separation Plane vs. Radial Distance

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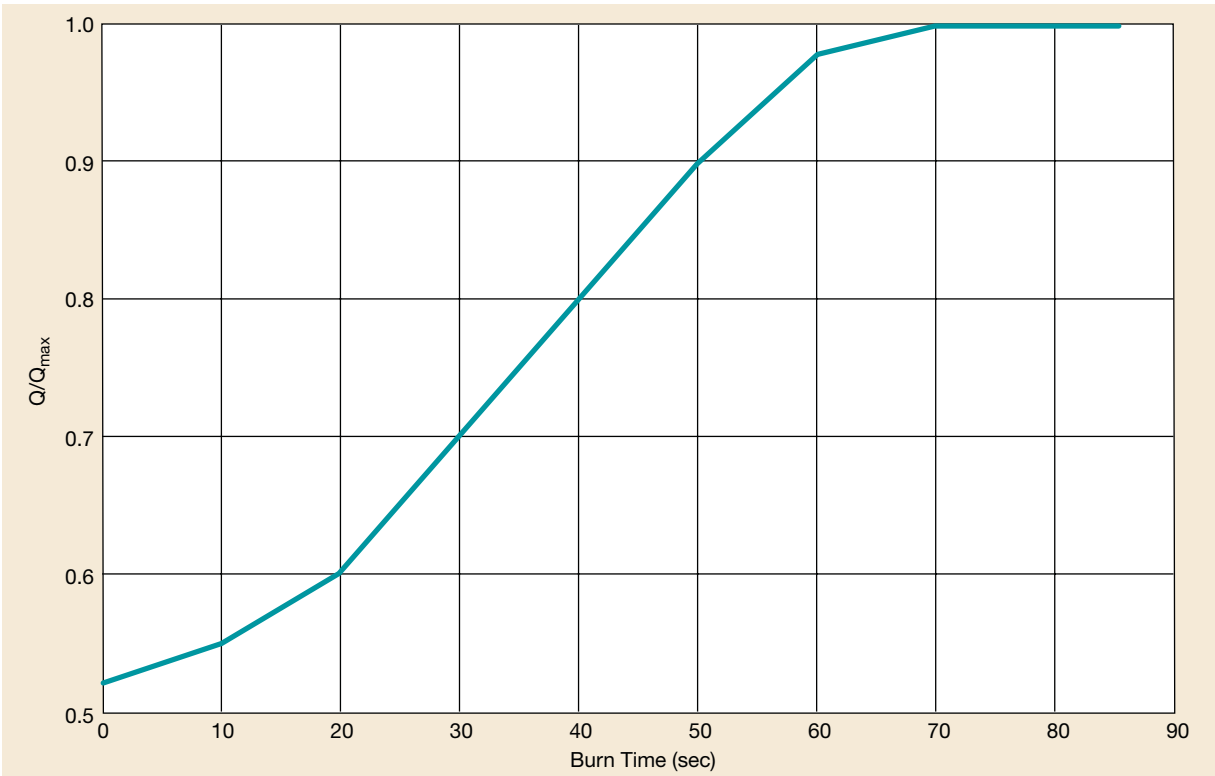


Figure 4-12. Predicted Star-48B Plume Radiation at the Spacecraft Separation Plane vs. Burn Time

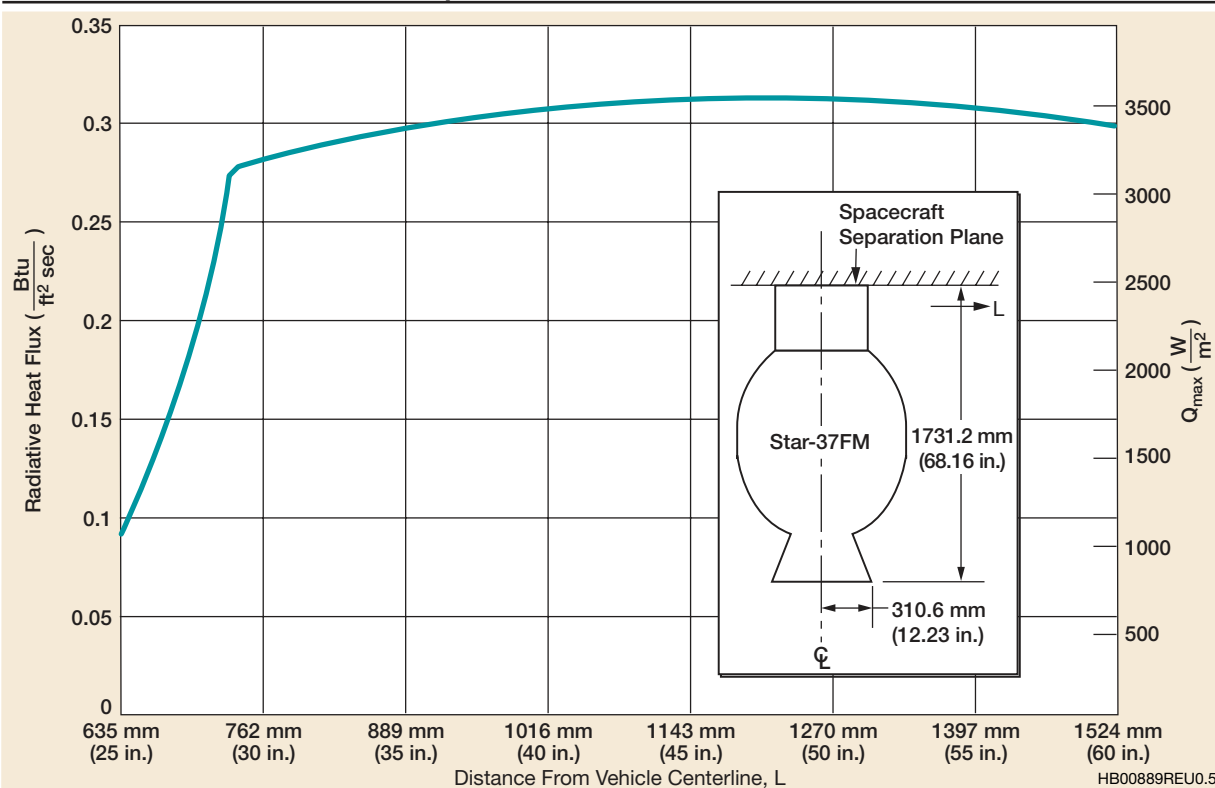


Figure 4-13. Predicted Star-37FM Plume Radiation at the Spacecraft Separation Plane vs. Radial Distance

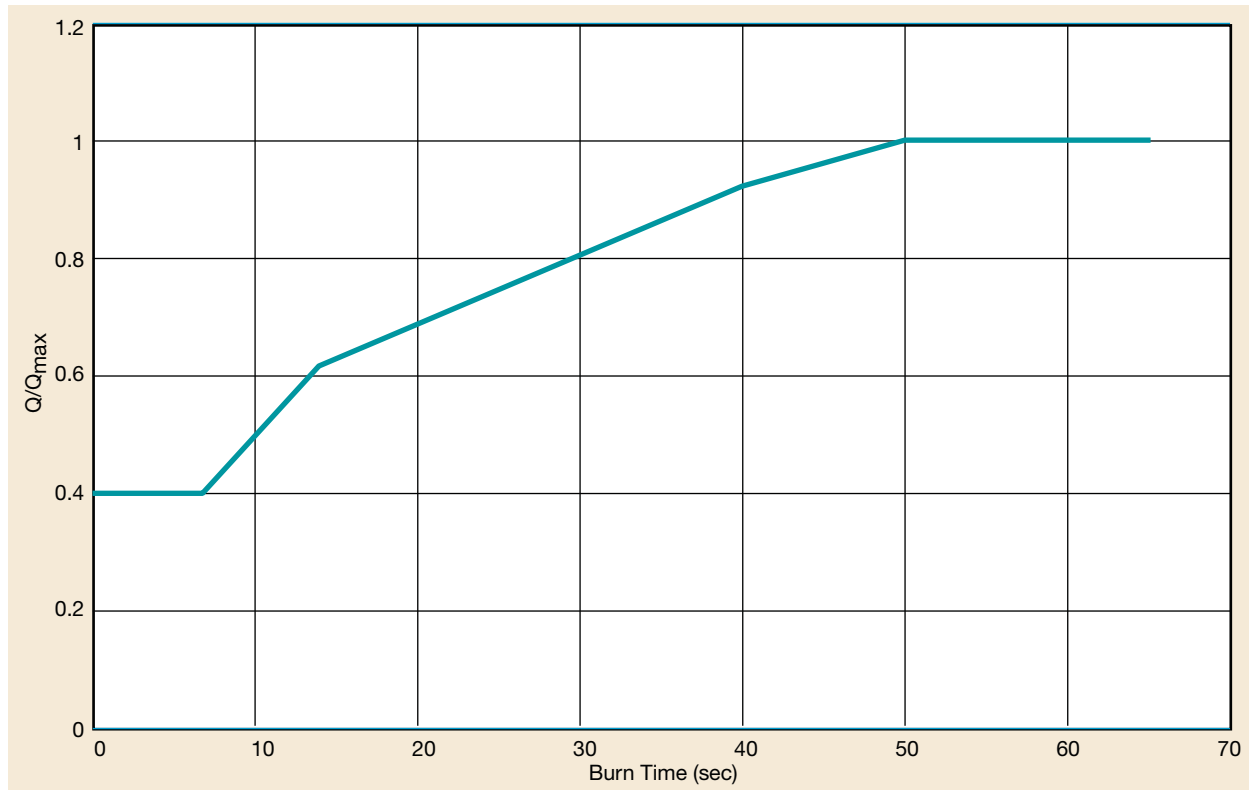


Figure 4-14. Predicted Star-37FM Plume Radiation at the Spacecraft Separation Plane vs. Burn Time

After third-stage motor burnout, the titanium motor case temperature rises rapidly, as shown in Figures 4-15 through 4-18. The temperature history shown is the maximum expected along the forward dome of the motor case and corresponds to both the Star-48B and Star-37FM motors. Figure 4-15 corresponds to a 7925 Delta II-class payload weight of 910 kg (2006 lb) and greater. Figures 4-16 and 4-17 correspond to lighter payloads that produce a greater amount of slag and result in greater titanium dome temperatures. Figure 4-18 corresponds to the Star-37FM, and titanium dome temperature is not dependent on spacecraft weight. The external surface emissivity for the Star-48B and Star-37FM motors is 0.34 and 0.2, respectively. Mission users should contact the Delta Program Office for more details.

The hydrazine thruster plume of the third-stage nutation control system (NCS) does not introduce significant heating to the payload interface plane. Any appendages that protrude below the interface plane should be evaluated for proximity to the NCS thruster. Information regarding this plume can be provided upon request.

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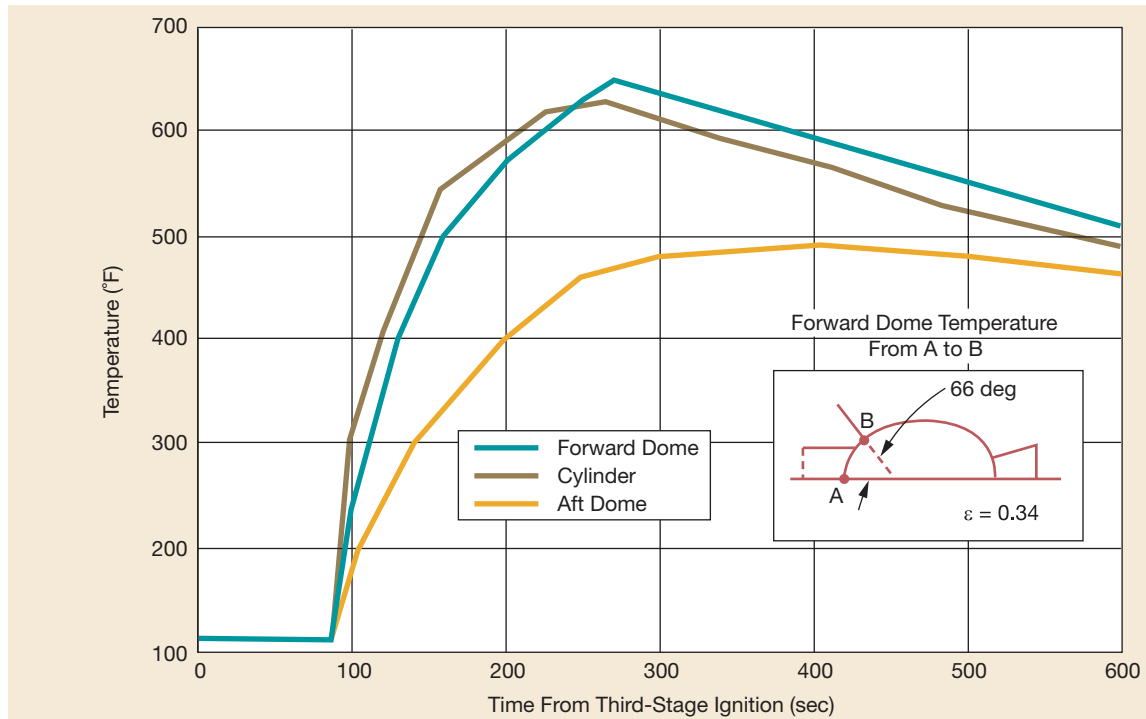


Figure 4-15. Star-48B Motor Case Soakback Temperature for Payload Mass Greater Than 910 kg (2006 lb)

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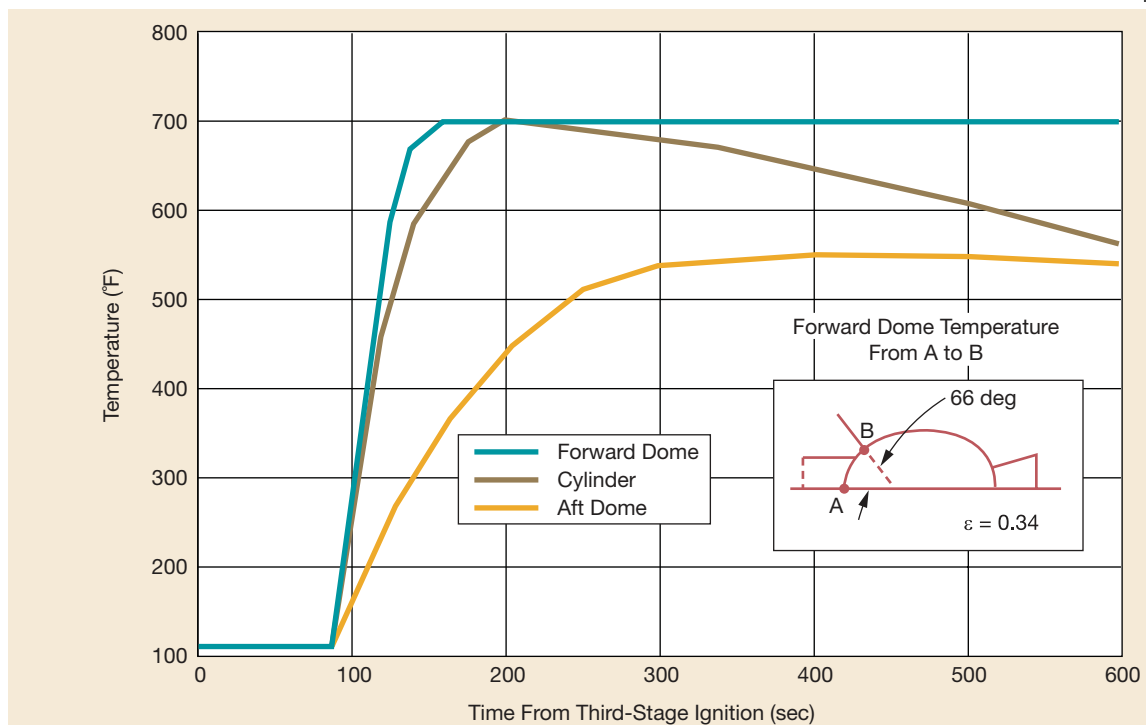


Figure 4-16. Star-48B Motor Case Soakback Temperature for Payload Mass Between 460 kg (1014 lb) and 910 kg (2006 lb)

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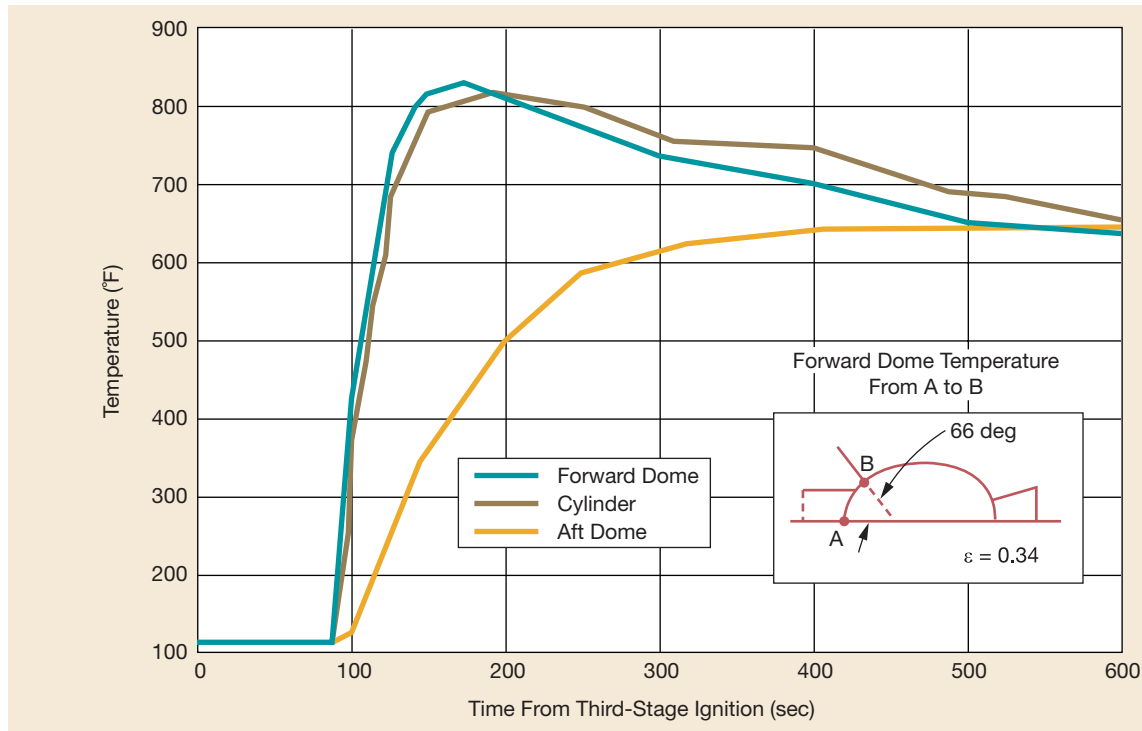


Figure 4-17. Star-48B Motor Case Soakback Temperature for Payload Mass Between 300 kg (661 lb) and 460 kg (1014 lb)

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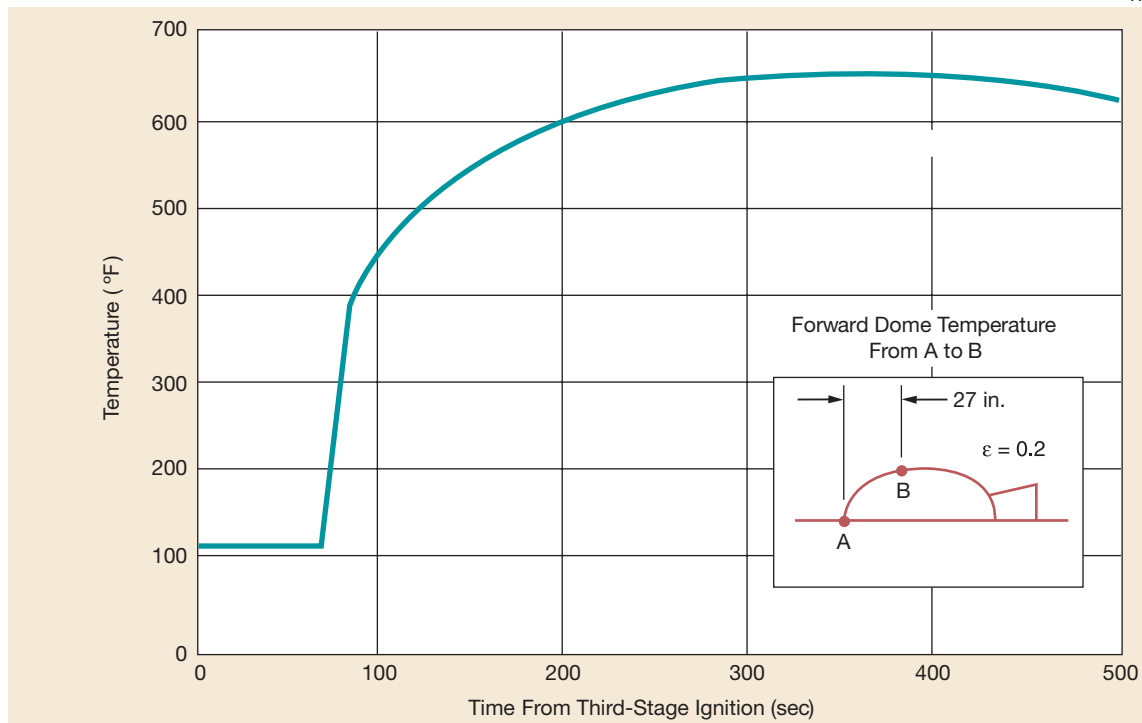


Figure 4-18. Star-37FM Motor Case Soak Back Temperature

4.2.3 Flight Dynamic Environment

The acoustic, sinusoidal, and shock environments provided in Sections 4.2.3.3, 4.2.3.4, and 4.2.3.5 are based on maximum flight levels for a 95th percentile statistical estimate.

4.2.3.1 Steady-State Acceleration. For the two-stage Delta II vehicle, the maximum axial acceleration occurs at the end of the first-stage burn main engine cutoff (MECO). For the three-stage Delta II vehicle, the maximum steady-state acceleration occurs at the end of third-stage flight for payloads up to 890.6 kg (1963 lb) for the Star-48B and 610.0 kg (1345 lb) for the Star-37FM. Above this weight, the maximum acceleration occurs at MECO. A plot of steady-state axial acceleration at MECO versus payload weight is shown in Figure 4-19 and is representative for the acceleration at MECO for the 2.9-m (9.5-ft) fairing as well as the standard and stretched 3-m (10-ft) fairings. Steady-state axial acceleration versus payload weight at third-stage motor burnout is shown in Figure 4-20.

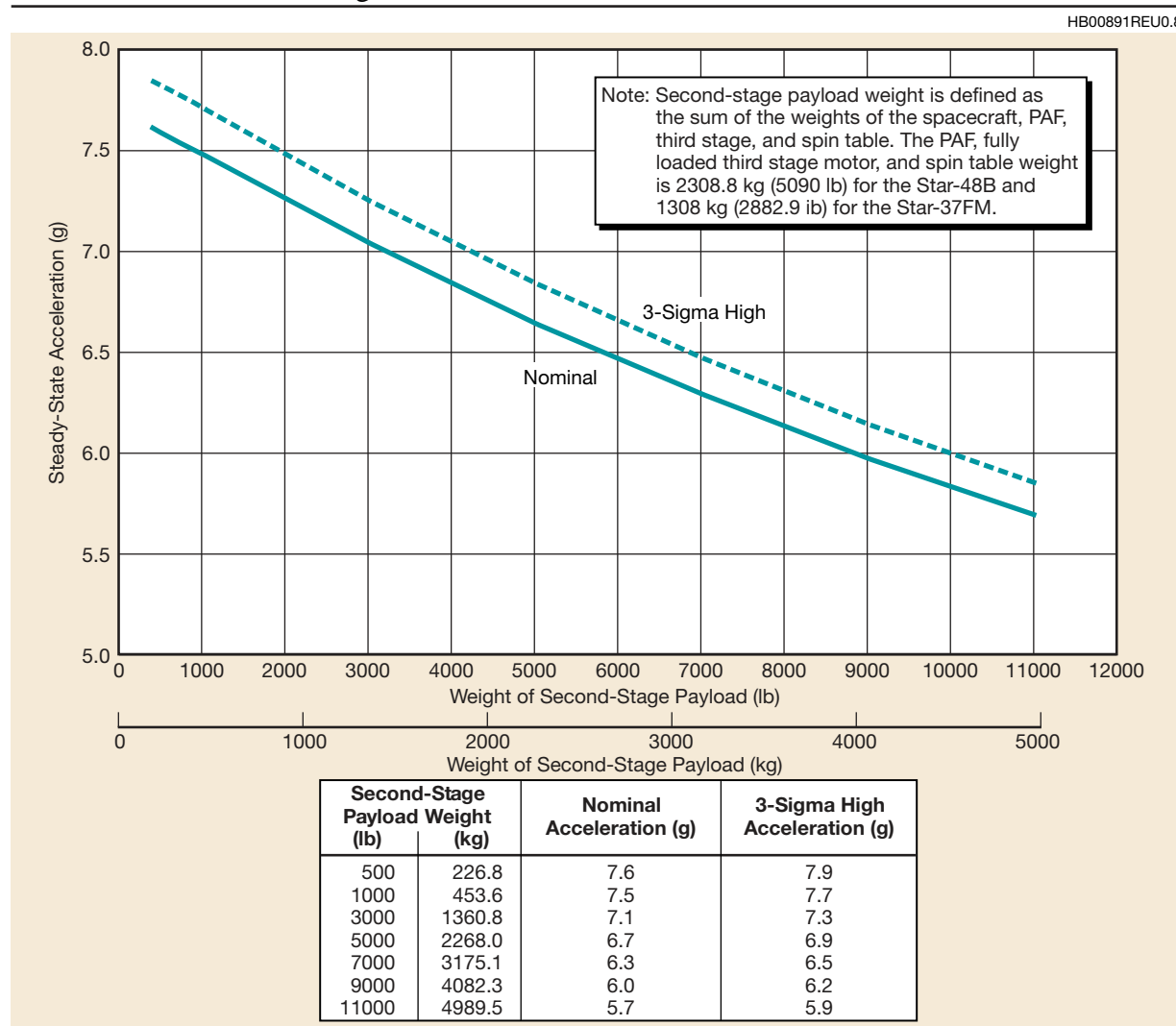


Figure 4-19. Axial Steady-State Acceleration at MECO vs. Payload Weight, Two-Stage and Three-Stage Missions

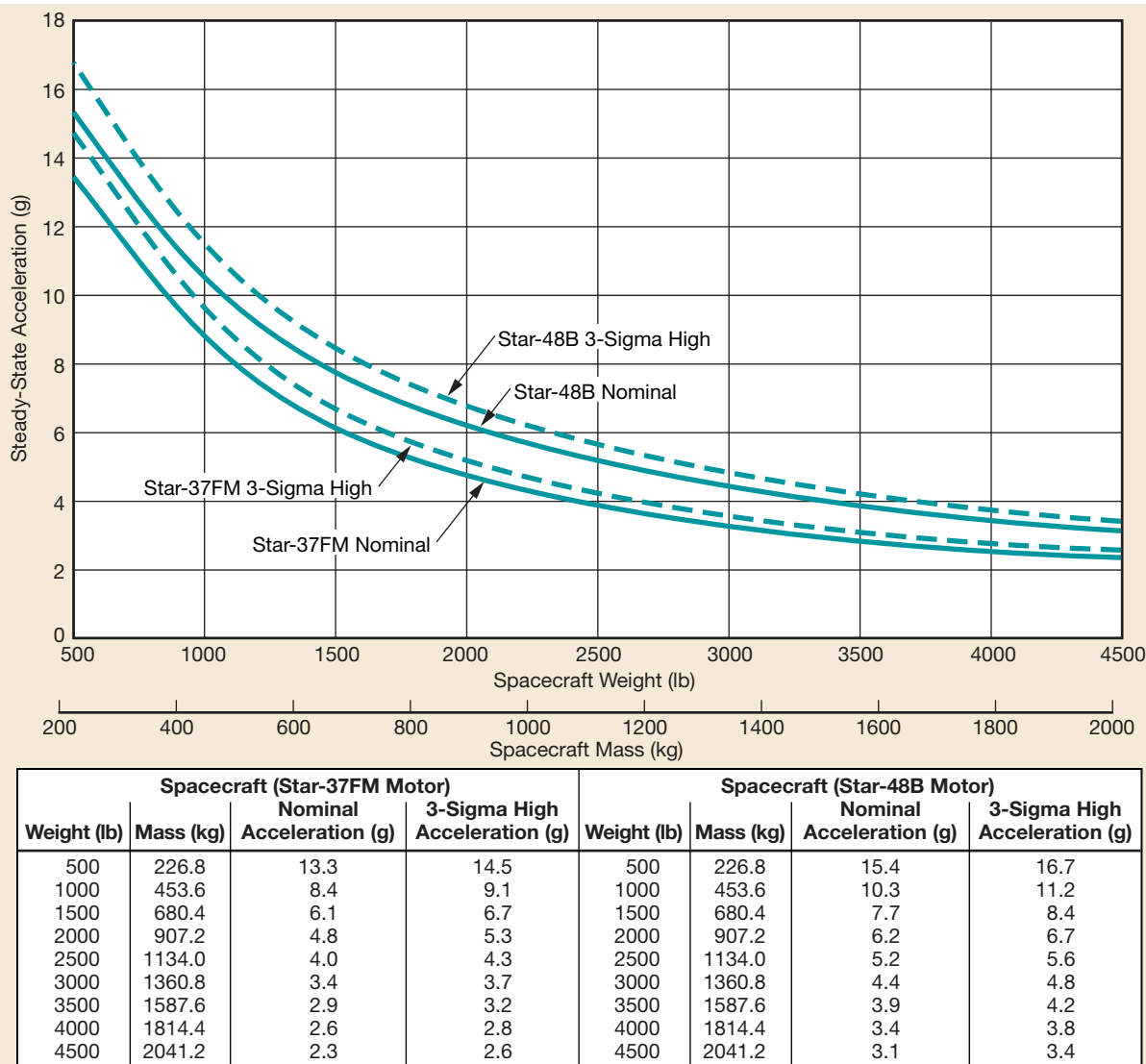


Figure 4-20. Axial Steady-State Acceleration vs. Spacecraft Weight at Third-Stage Burnout (TECO)

4.2.3.2 Combined Loads. Dynamic excitations, which occur predominantly during liftoff and transonic periods of flight, are superimposed on steady-state accelerations to produce combined accelerations that must be used in the spacecraft structural design. The combined spacecraft accelerations are a function of launch vehicle characteristics as well as spacecraft dynamic characteristics and mass properties. To prevent dynamic coupling between the launch vehicle and the spacecraft in the low-frequency range for the three-stage Delta 792X and 792XH configurations, the spacecraft structural stiffness should produce fundamental frequencies above 35 Hz in the thrust axis and 15 Hz in the lateral axes. For three-stage Delta II 732X or 742X configurations, the spacecraft structural stiffness should produce fundamental frequencies above 35 Hz in the thrust axis and 20 Hz in the lateral axes of the spacecraft. For all two-stage Delta II configurations, the spacecraft structural stiffness should produce fundamental frequencies above 35 Hz in the thrust

axis and 12 Hz in the lateral axes. The spacecraft should meet these criteria, while being hard-mounted at the separation plane (without compliance from the PAF and separation clampband). In addition, secondary structure mode frequencies should be above 35 Hz to prevent undesirable coupling with launch vehicle modes and/or large fairing-to-spacecraft relative dynamic deflections. The spacecraft design-limit load factors presented in Table 4-5 are applicable for spacecraft meeting the above fundamental frequency criteria. For very flexible, lighter weight, or dual-manifested spacecraft, the combined accelerations and subsequent design-limit load factors could be higher than shown. The customer should consult the Delta Program Office so that appropriate analyses can be performed to better define loading conditions.

Table 4-5. Payload Center-of-Gravity Limit Load Factors (g)

	Payload weight											
	362.8–680.3 kg (800–1500 lb)		680.3–907.2 kg (1500–2000 lb)		907.2–1134.0 kg (2000–2500 lb)		1134.0–2268.0 kg (2500–5000 lb)		2268.0–2812.2 kg (5000–6200 lb)		2812.2 kg (6200 lb)	
	Axial	Lateral	Axial	Lateral	Axial	Lateral	Axial	Lateral	Axial	Lateral	Axial	Lateral
Liftoff/Aero	+2.8/ -0.2	±4.5	+2.8/ -0.2	±4.0	+2.8/ -0.2	±3.5	+2.8/ -0.2	±3.0	+2.8/ -0.2	±2.5	+2.8/ -0.2	±2.0
MECO	X±0.6	±0.2	X±0.6	±0.2	X±0.6	±0.2	X±0.6	±0.2	X±0.6	±0.2	X±0.6	±0.2
TECO	Y	±0.1	Y	±0.1	Y	±0.1	Y	±0.1	Y	±0.1	Y	±0.1
Notes: 1. Positive axial denotes compression. 2. Lateral load factor provides proper bending moment at the spacecraft-to-launch-vehicle interface. 3. Refer to Figures 4-19 and 4-20 for 3-sigma steady-state axial accelerations for MECO and TECO. 4. Assumes that spacecraft meets minimum frequency guidelines specified in paragraph 4.2.3.2 and spacecraft center-of-gravity (CG) offset from the vehicle centerline is less than 20.3 mm (0.8 in.) 5. TECO: Third-stage burn-out.												

4.2.3.3 Acoustic Environment. The maximum acoustic environment for the payload occurs during liftoff and transonic flight. The duration of the maximum environment is less than 10 sec. The payload acoustic environment is a function of the configuration of the launch vehicle, the fairing, and the fairing acoustic blankets. Section 3 defines the fairing blanket configurations. Table 4-6 identifies figures that define the payload acoustic environment for several versions of the Delta II. The maximum flight level payload acoustic environments for the blanketed region for different Delta II launch vehicle configurations are defined in Figures 4-21 and 4-22 based on typical spacecraft with payload bay fills up to 60%. Launch vehicles with payload bay fills above 80% will experience approximately 1-1/2 dB higher levels. The overall sound pressure level (OASPL) for each acoustic environment is also shown in the figures.

Table 4-6. Spacecraft Acoustic Environment Figure References

Delta II launch vehicle configuration	Mission type	Fairing configuration	Fairing acoustic blanket configuration	Spacecraft acoustic environment
7320 7325, 7326 7425, 7426 7420 7920 7925, 7926	Two-stage and three-stage	2.9-m dia (9.5-ft) dia	76.2-mm (3-in.) configuration	See Figure 4-21
7320-10, -10L 7325-10, -10L 7326-10, -10L 7420-10, -10L 7425-10, -10L 7426-10, -10L 7920-10, -10L 7925-10, -10L 7926-10, -10L	Two-stage and three-stage	3.0-m (10-ft) dia and 3.0-m (-10L) stretched fairings	76.2-mm (3-in.) configuration	See Figure 4-22

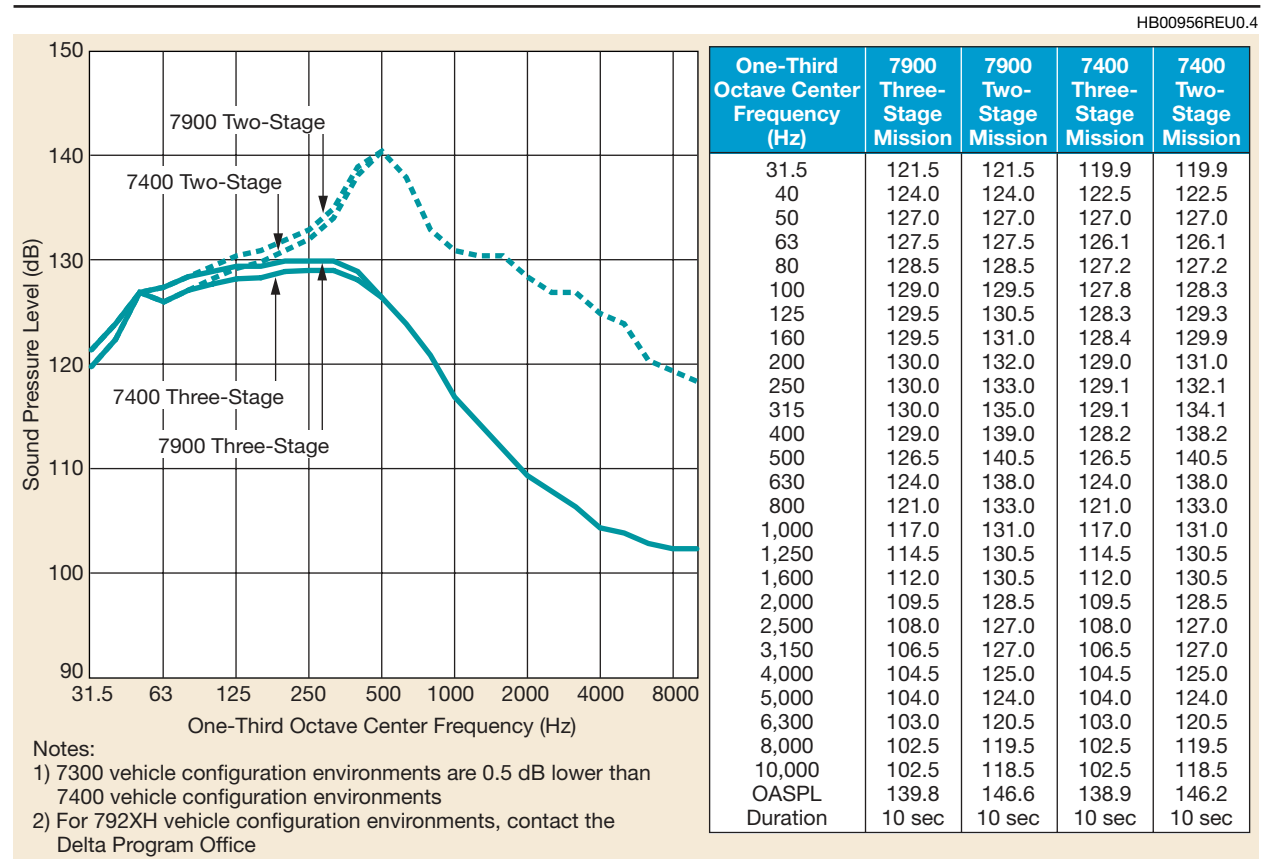


Figure 4-21. Predicted Delta II Acoustic Environments for 9.5-ft Fairing Missions

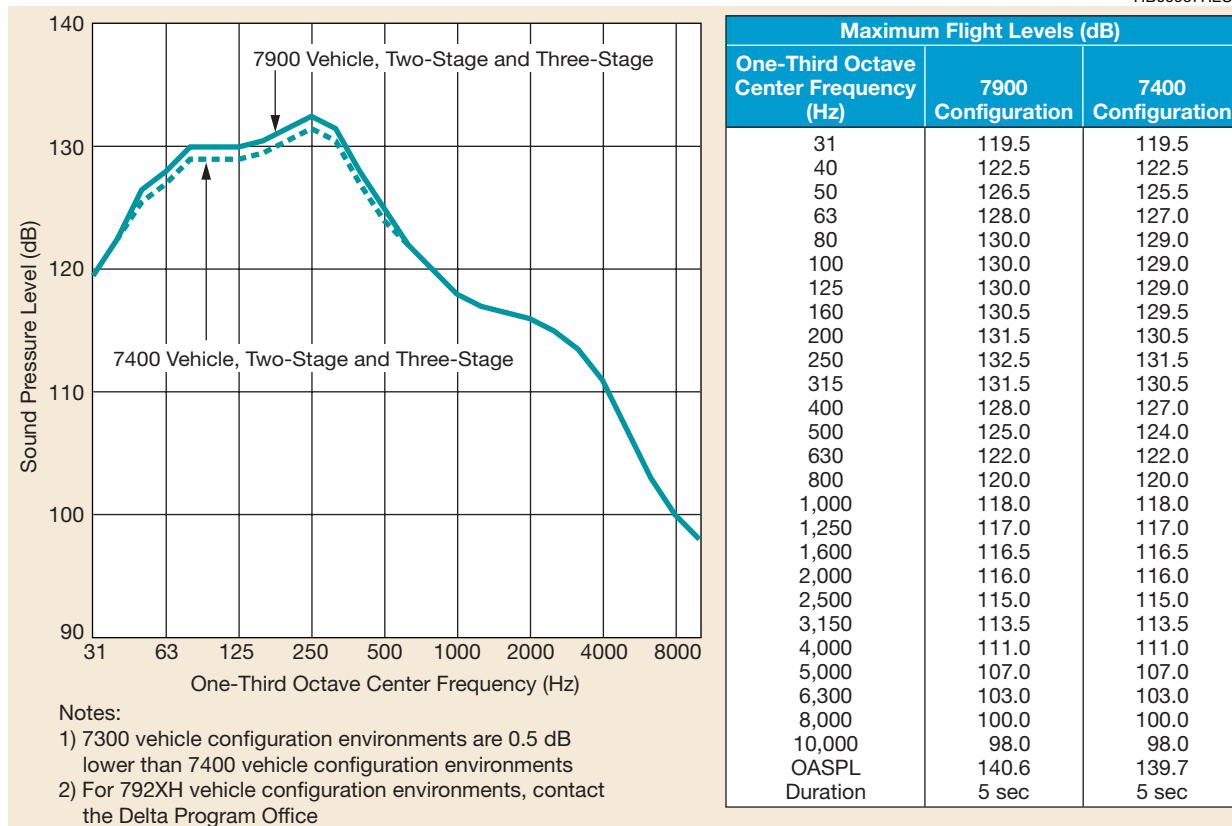
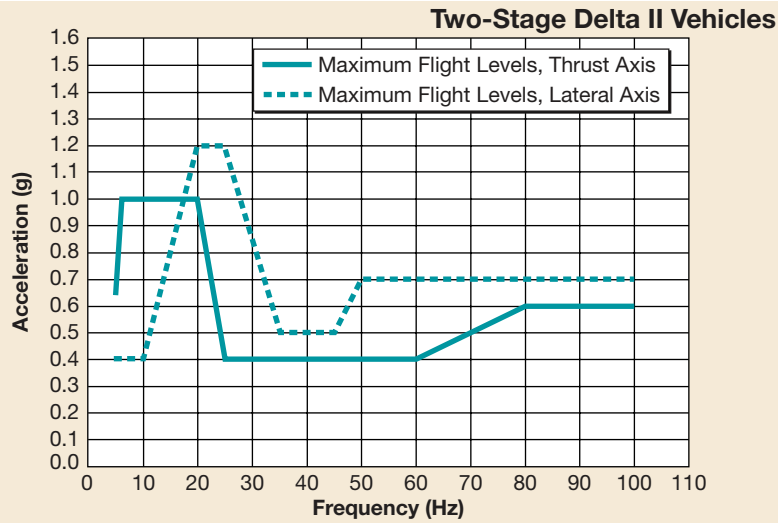


Figure 4-22. Predicted Delta II Acoustic Environments for 10-ft and -10L Fairing Missions

The acoustic environments shown here for missions with a 10-ft fairing also envelop those for missions with a 10-ft-long (-10L) fairing or with a DPAF. The acoustic environment produces the dominant high-frequency random vibration responses in the payload. A properly performed acoustic test offers the best simulation of the acoustically-induced random vibration environment. (See Section 4.2.4.2.) No significant high-frequency random vibration inputs at the PAF/spacecraft interface are generated by the Delta II launch vehicle; consequently, a random vibration environment is not specified at this interface.

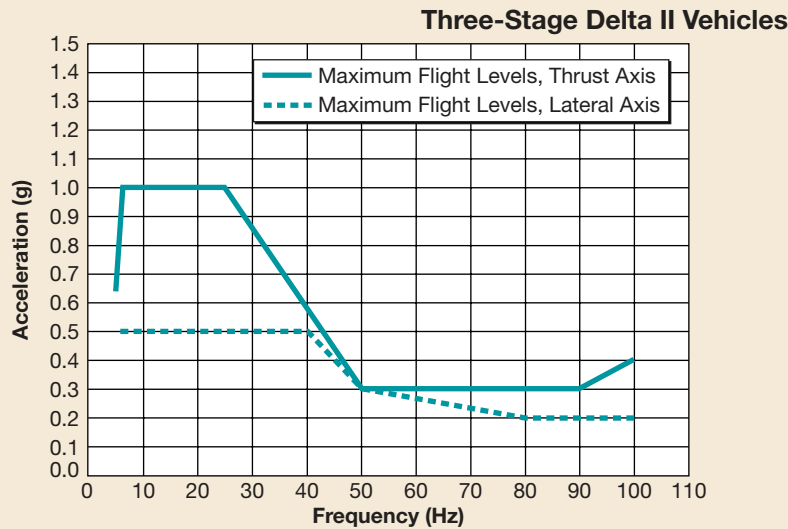
4.2.3.4 Sinusoidal Vibration Environment. The payload will experience sinusoidal vibration inputs during flight as a result of launch, ascent transients, and oscillatory flight events. The maximum flight level sinusoidal vibration inputs are defined in Figures 4-23 and 4-24. These sinusoidal vibration levels envelope low-frequency flight dynamic events such as liftoff transients, transonic/maximum Q oscillations, pre-MECO sinusoidal oscillations, MECO transients, and second/third-stage events. The levels provided in Figures 4-23 and 4-24 are limit-level acceleration and should be multiplied by the appropriate qualification factor when used for spacecraft qualification.



Axis	Frequency (Hz)	Maximum Flight Levels (g)
Thrust	5	0.64
	6.2	1.00
	20	1.00
	25	0.40
	60	0.40
	80	0.60
	100	0.60
Lateral	5	0.40
	10	0.40
	20	1.20
	25	1.20
	35	0.50
	45	0.50
	50	0.70
	100	0.70

Notes:

- 1) Lateral accelerations to be applied in any spacecraft lateral direction
- 2) Vibration inputs are defined at the base of the payload attach fitting



Note: For DPAF missions contact the Delta Program Office.

Axis	Frequency (Hz)	Maximum Flight Levels (g)
Thrust	5	0.64
	6.2	1.00
	25	1.00
	50	0.30
	90	0.30
	100	0.40
Lateral	5	0.50
	40	0.50
	50	0.30
	80	0.20
	100	0.20

Notes:

- 1) Lateral accelerations to be applied in any spacecraft lateral direction
- 2) Vibration inputs are defined at the top of the payload attach fitting

Figure 4-23. Delta II Sinusoidal Vibration Levels (Q=10) Except MECO for all Delta II Vehicles

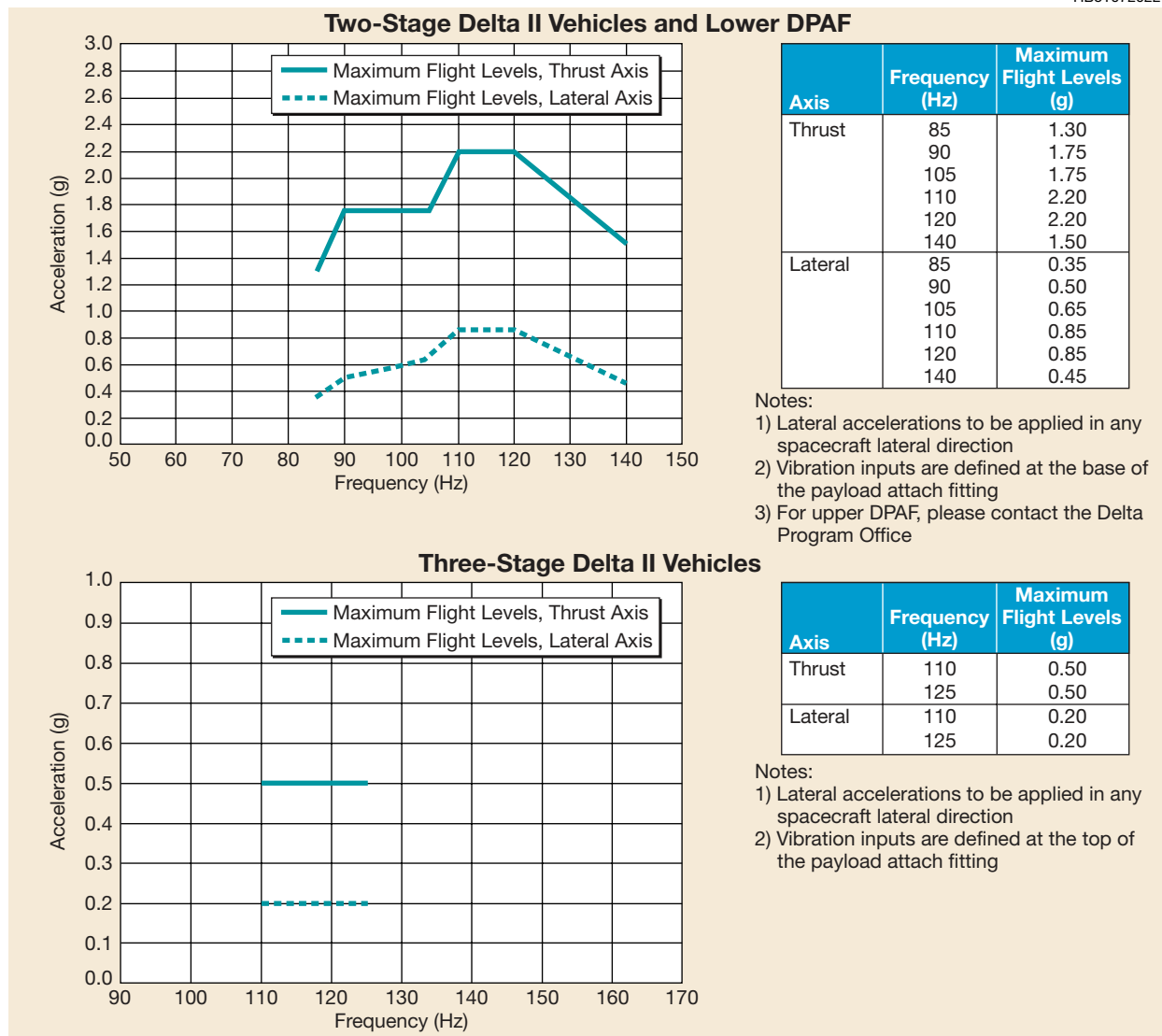


Figure 4-24. Delta II Recommended MECO Sinusoidal Vibration Levels (Q=10)

The sinusoidal vibration levels in Figures 4-23 and 4-24 are not intended for use in the design of spacecraft primary structure; limit load factors for spacecraft primary structure design are specified in Table 4-5.

The sinusoidal vibration levels should be used in conjunction with the results of the coupled dynamic loads analysis to aid in the design of secondary structure (e.g., solar arrays, antennae, appendages) that may experience dynamic loading due to coupling with the launch vehicle low-frequency dynamic oscillations. Notching of the sinusoidal vibration input levels at spacecraft fundamental frequencies may be required during testing and should be based on the results of the vehicle coupled dynamic loads analysis. (See Section 4.2.4.3.)

4.2.3.5 Shock Environment. The maximum shock environment at the PAF/spacecraft interface occurs during spacecraft separation from the launch vehicle and is a function of the PAF/

spacecraft separation system configuration. Table 4-7 lists the figures that define the shock environment at the spacecraft interface for various missions, PAF configurations, and types of separation systems. Shock levels at the PAF/spacecraft interface due to other flight shock events, such as stage separation, fairing separation, and engine ignition/shutdown, are not significant compared to the spacecraft separation shock environment.

Table 4-7. Spacecraft Interface Shock Environment Figure References

Mission Type	PAF Configuration	Spacecraft Separation System Type	Spacecraft Interface Shock Environment
Three-stage	3712A 3712B 3712C 3724C	939.8-mm (37-in.)-dia V-block clamp	See Figure 4-25
Two-stage	3715	939.8-mm (37-in.)-dia V-block clamp	See Figure 4-25
Two-stage	6306	1600-mm (63-in.)-dia V-block clamp	See Figure 4-26
Two-stage	6019	1524-mm (60-in.) dia Three explosive separation nuts	See Figure 4-27
Two-stage	6915	1752.6-mm (69-in.) dia Four explosive separation nuts	See Figure 4-27
Two-stage	5624	1422.4-mm (56-in.)-dia V-block clamp	See Figure 4-28

The maximum flight level shock environments at the PAF/spacecraft interface defined in Figures 4-25 through 4-28 are intended to aid in the design of spacecraft components and secondary structure that may be sensitive to high-frequency pyrotechnic-shock. As is typical for this type of shock, the level dissipates rapidly with distance and the number of joints between the shock source and the component of interest. A properly performed system-level shock test offers the best simulation of the high-frequency pyrotechnic shock environment. (See Section 4.2.4.4.)

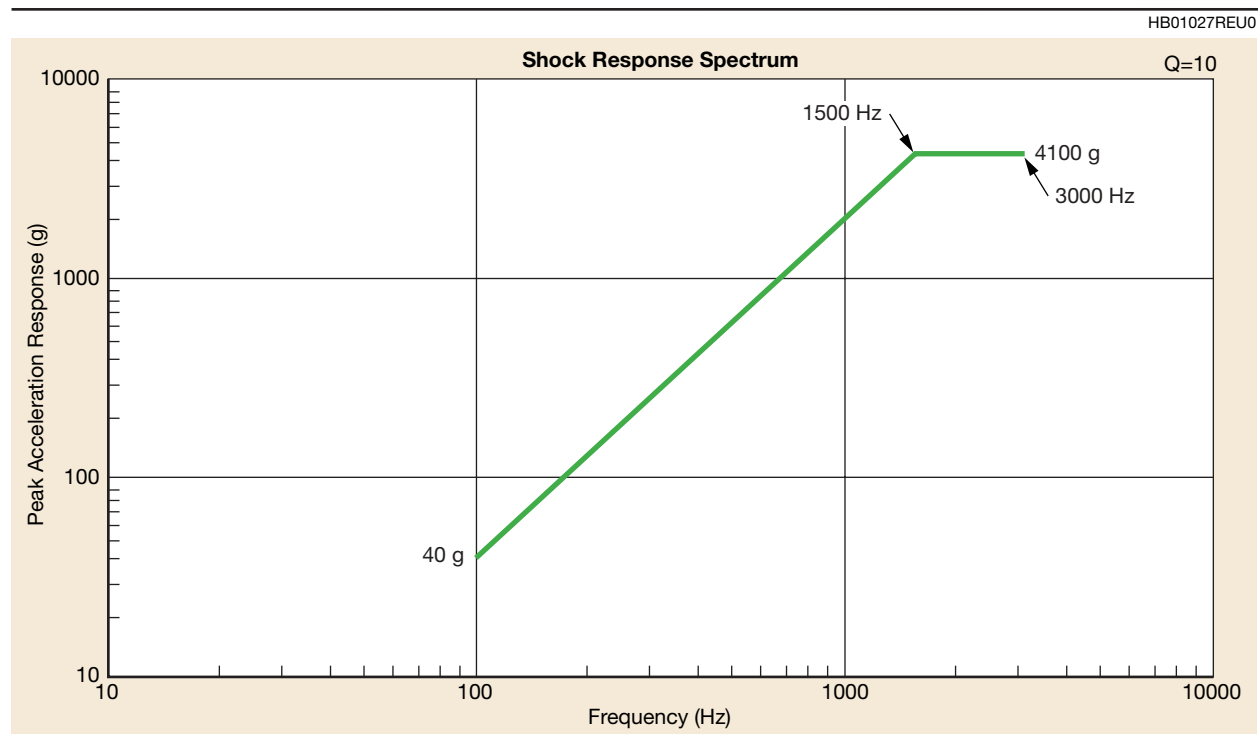


Figure 4-25. Maximum Flight Spacecraft Interface Shock Environment 3712A, 3712B, 3712C, 3715, 3724C Payload Attach Fitting

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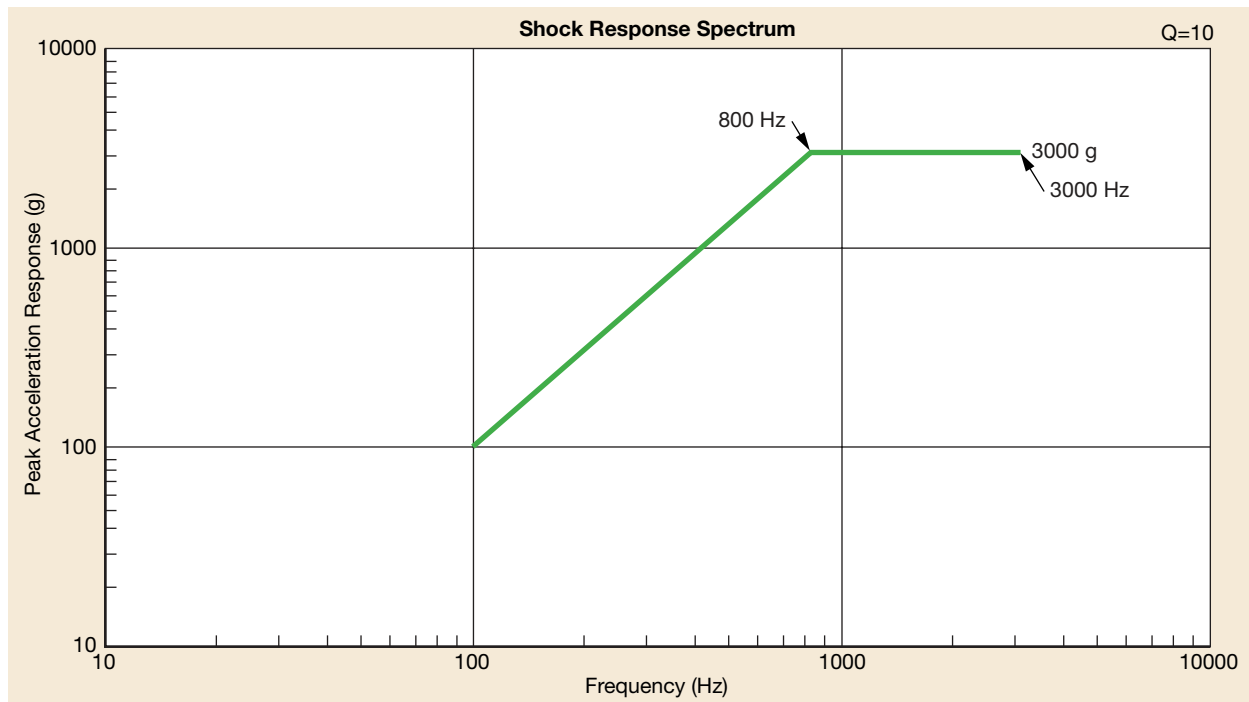


Figure 4-26. Maximum Flight Spacecraft Interface Shock Environment 6306 Payload Attach Fitting

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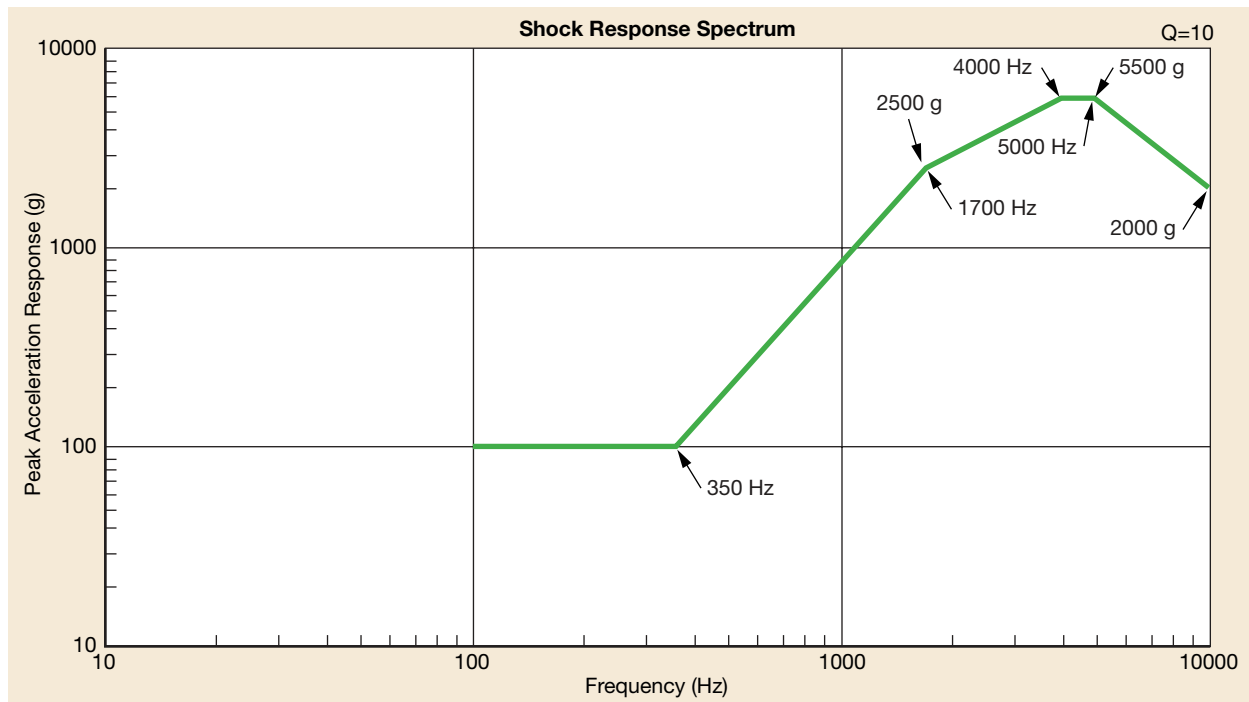


Figure 4-27. Maximum Flight Spacecraft Interface Shock Environment 6019 and 6915 Payload Attach Fitting

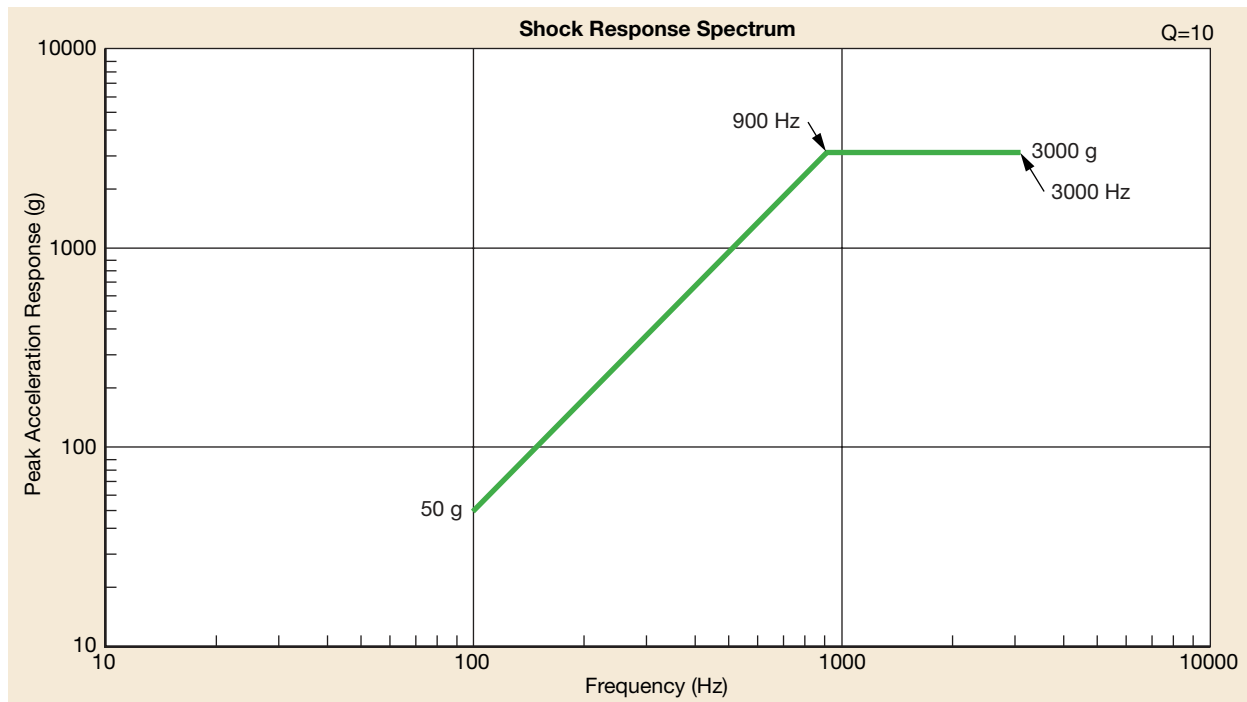


Figure 4-28. Maximum Flight Spacecraft Interface Shock Environment 5624 Payload Attach Fitting

4.2.4 Payload Qualification and Acceptance Testing

This section outlines a series of environmental system-level qualification, acceptance, and protoflight tests for payloads launched on Delta II vehicles. The tests presented here are, by necessity, generalized so as to encompass numerous payload configurations. For this reason, each payload should be critically evaluated for its own specific requirements and detailed test specifications developed and tailored to its particular requirements. Coordination with the Delta Program Office during the development of test specifications is encouraged to ensure the adequacy of the payload test approach.

The qualification test levels presented in this section are intended to ensure that the payload possesses adequate design margin to withstand the maximum expected Delta II dynamic environmental loads, even with minor weight and design variations. The acceptance test levels are intended to verify adequate spacecraft manufacture and workmanship by subjecting the flight spacecraft to maximum expected flight environments. The protoflight test approach is intended to combine verification of adequate design margin and adequacy of spacecraft manufacture and workmanship by subjecting the flight spacecraft to protoflight test levels, which are equal to qualification test levels with reduced durations.

4.2.4.1 Structural Load Testing. Structural load testing is performed by the user to demonstrate the design integrity of the primary structural elements of the spacecraft. These loads are

based on worst-case conditions as defined in Sections 4.2.3.1 and 4.2.3.2. Maximum flight loads will be increased by a factor of 1.25 to determine qualification test loads.

A test PAF is required to provide proper load distribution at the spacecraft interface. The customer shall consult the Delta Program Office before developing the structural load test plan and shall obtain concurrence for the test load magnitude to ensure that the PAF will not be stressed beyond its load-carrying capability.

When the maximum axial load is controlled by the third stage, radial accelerations due to spin must be included. Spacecraft combined-loading qualification testing is accomplished by a static load test or on a centrifuge. Generally, static load tests can be readily performed on structures with easily defined load paths, whereas for complex spacecraft assemblies, centrifuge testing may be the most economical.

4.2.4.2 Acoustic Testing. The maximum flight level acoustic environments defined in Section 4.2.3.3 are increased by 3.0 dB for spacecraft acoustic qualification and protoflight testing. The acoustic test duration is 120 sec for qualification testing and 60 sec for protoflight testing. For spacecraft acoustic acceptance testing, the acoustic test levels are equal to the maximum flight level acoustic environments defined in Section 4.2.3.3. The acoustic acceptance test duration is 60 sec. The acoustic qualification, acceptance, and protoflight test levels for several of the Delta II launch vehicle configurations are defined in Tables 4-8, 4-9, and 4-10.

The acoustic test tolerances are +4 dB and -2 dB from 50 Hz to 2000 Hz. Above and below these frequencies, the acoustic test levels should be maintained as close to the nominal test levels as possible within the limitations of the test facility. The OASPL should be maintained within +3 dB and -1 dB of the nominal overall test level.

**Table 4-8. Acoustic Test Levels, Delta II, 2.9-m (9.5-ft)-dia Faring,
Three-Stage Mission, 3-in. Blanket Configuration**

One-third octave center frequency (Hz)	7900 configuration**			7400 configuration*		
	Acceptance test levels (dB)	Qualification test levels (dB)	Protoflight test levels (dB)	Acceptance test levels (dB)	Qualification test levels (dB)	Protoflight test levels (dB)
31.5	121.5	124.5	124.5	119.9	122.9	122.9
40	124.0	127.0	127.0	122.5	125.5	125.5
50	127.0	130.0	130.0	127.0	130.0	130.0
63	127.5	130.5	130.5	126.1	129.1	129.1
80	128.5	131.5	131.5	127.2	130.2	130.2
100	129.0	132.0	132.0	127.8	130.8	130.8
125	129.5	132.5	132.5	128.3	131.3	131.3
160	129.5	132.5	132.5	128.4	131.4	131.4
200	130.0	133.0	133.0	129.0	132.0	132.0
250	130.0	133.0	133.0	129.1	132.1	132.1
315	130.0	133.0	133.0	129.1	132.1	132.1
400	129.0	132.0	132.0	128.2	131.2	131.2
500	126.5	129.5	129.5	126.5	129.5	129.5
630	124.0	127.0	127.0	124.0	127.0	127.0
800	121.0	124.0	124.0	121.0	124.0	124.0
1000	117.0	120.0	120.0	117.0	120.0	120.0
1250	114.5	117.5	117.5	114.5	117.5	117.5
1600	112.0	115.0	115.0	112.0	115.0	115.0
2000	109.5	112.5	112.5	109.5	112.5	112.5
2500	108.0	111.0	111.0	108.0	111.0	111.0
3150	106.5	109.5	109.5	106.5	109.5	109.5
4000	104.5	107.5	107.5	104.5	107.5	107.5
5000	104.0	107.0	107.0	104.0	107.0	107.0
6300	103.0	106.0	106.0	103.0	106.0	106.0
8000	102.5	105.5	105.5	102.5	105.5	105.5
10000	102.5	105.5	105.5	102.5	105.5	105.5
OASPL	139.8	142.8	142.8	138.9	141.9	141.9
Duration	60 sec	120 sec	60 sec	60 sec	120 sec	60 sec
*Note: 7300 vehicle configuration environments are 0.5 dB below 7400 configuration vehicle environments.						
**For 792XH vehicle configuration environments, contact the Delta Program Office.						

**Table 4-9. Acoustic Test levels, Delta II, 2.9-m (9.5-ft)-dia Faring,
Two-Stage Mission, 3-in. Blanket Configuration**

One-third octave center frequency (Hz)	7900 configuration**			7400 configuration*		
	Acceptance test levels (dB)	Qualification test levels (dB)	Protoflight test levels (dB)	Acceptance test levels (dB)	Qualification test levels (dB)	Protoflight test levels (dB)
31.5	121.5	124.5	124.5	119.9	122.9	122.9
40	124.0	127.0	127.0	122.5	125.5	125.5
50	127.0	130.0	130.0	127.0	130.0	130.0
63	127.5	130.5	130.5	126.1	129.1	129.1
80	128.5	131.5	131.5	127.2	130.2	130.2
100	129.5	132.5	132.5	128.3	131.3	131.3
125	130.5	133.5	133.5	129.3	132.3	132.3
160	131.0	134.0	134.0	129.9	132.9	132.9
200	132.0	135.0	135.0	131.0	134.0	134.0
250	133.0	136.0	136.0	132.1	135.1	135.1
315	135.0	138.0	138.0	134.1	137.1	137.1
400	139.0	142.0	142.0	138.2	141.2	141.2
500	140.5	143.5	143.5	140.5	143.5	143.5
630	138.0	141.0	141.0	138.0	141.0	141.0
800	133.0	136.0	136.0	133.0	136.0	136.0
1000	131.0	134.0	134.0	131.0	134.0	134.0
1250	130.5	133.5	133.5	130.5	133.5	133.5
1600	130.5	133.5	133.5	130.5	133.5	133.5
2000	128.5	131.5	131.5	128.5	131.5	131.5
2500	127.0	130.0	130.0	127.0	130.0	130.0
3150	127.0	130.0	130.0	127.0	130.0	130.0
4000	125.0	128.0	128.0	125.0	128.0	128.0
5000	124.0	127.0	127.0	124.0	127.0	127.0
6300	120.5	123.5	123.5	120.5	123.5	123.5
8000	119.5	122.5	122.5	119.5	122.5	122.5
10000	118.5	121.5	121.5	118.5	121.5	121.5
OASPL	146.6	149.6	149.6	146.2	149.2	149.2
Duration	60 sec	120 sec	60 sec	60 sec	120 sec	60 sec
*Note: 7300 vehicle configuration environments are 0.5 dB below 7400 configuration vehicle environments.						
**Note: For 792XH vehicle configuration environments, contact the Delta Program Office.						

**Table 4-10. Acoustic Test Levels, Delta II, 3.0-m (10-ft)-dia Fairing,
Two- and Three-Stage Missions, 3-in. Blanket Configuration**

One-third octave center frequency (Hz)	7900 Configuration**			7400 Configuration*		
	Acceptance test levels (dB)	Qualification test levels (dB)	Protoflight test levels (dB)	Acceptance test levels (dB)	Qualification test levels (dB)	Protoflight test levels (dB)
31.5	119.5	122.5	122.5	119.5	122.5	122.5
40	122.5	125.5	125.5	122.5	125.5	125.5
50	126.5	129.5	129.5	125.5	128.5	128.5
63	128.0	131.0	131.0	127.0	130.0	130.0
80	130.0	133.0	133.0	129.0	132.0	132.0
100	130.0	133.0	133.0	129.0	132.0	132.0
125	130.0	133.0	133.0	129.0	132.0	132.0
160	130.5	133.5	133.5	129.5	132.5	132.5
200	131.5	134.5	134.5	130.5	133.5	133.5
250	132.5	135.5	135.5	131.5	134.5	134.5
315	131.5	134.5	134.5	130.5	133.5	133.5
400	128.0	131.0	131.0	127.0	130.0	130.0
500	125.0	128.0	128.0	124.0	127.0	127.0
630	122.0	125.0	125.0	122.0	125.0	125.0
800	120.0	123.0	123.0	120.0	123.0	123.0
1000	118.0	121.0	121.0	118.0	121.0	121.0
1250	117.0	120.0	120.0	117.0	120.0	120.0
1600	116.5	119.5	119.5	116.5	119.5	119.5
2000	116.0	119.0	119.0	116.0	119.0	119.0
2500	115.0	118.0	118.0	115.0	118.0	118.0
3150	113.5	116.5	116.5	113.5	116.5	116.5
4000	111.0	114.0	114.0	111.0	114.0	114.0
5000	107.0	110.0	110.0	107.0	110.0	110.0
6300	103.0	106.0	106.0	103.0	106.0	106.0
8000	100.0	103.0	103.0	100.0	103.0	103.0
10000	98.0	101.0	101.0	98.0	101.0	101.0
OASPL	140.6	143.6	143.6	139.7	142.7	142.7
Duration	60 sec	120 sec	60 sec	60 sec	120 sec	60 sec
*Note: 7300 vehicle configuration acoustic environments are 0.5 dB below 7400 configuration environments.						
**Note: For 792XH vehicle configuration environments, contact the Delta Program Office.						

4.2.4.3 Sinusoidal Vibration Testing. The maximum flight-level sinusoidal vibration environments defined in Section 4.2.3.4 are typically increased by 3.0 dB (a factor of 1.4) for spacecraft qualification and protoflight testing. For spacecraft acceptance testing, the sinusoidal vibration test levels are equal to the maximum flight level sinusoidal vibration environments defined in Section 4.2.3.4. The sinusoidal vibration acceptance, qualification, and protoflight test levels for all Delta II launch vehicle configurations are defined in Table 4-11. Contact the Delta Program Office for MECO environment testing guidelines.

Table 4-11. Delta II Sinusoidal Vibration Test Levels

Two-Stage Delta II Vehicles				
Sine Test	Axis	Frequency (Hz)	Test Levels	Sweep Rate
Acceptance	Thrust/Lateral	5 to 100	Figure 4-23	4 octaves/min
Protoflight	Thrust	5 7.4 7.4 to 100	0.64 g 1.40 g Figure 4-23 + 3.0 dB	4 octaves/min
	Lateral	5 to 100	Figure 4-23 + 3.0 dB	4 octaves/min
Qualification	Thrust	5 7.4 7.4 to 100	0.64 g 1.40 g Figure 4-23 + 3.0 dB	2 octaves/min
	Lateral	5 to 100	Figure 4-23 + 3.0 dB	2 octaves/min
Notes: (1) Lateral accelerations to be applied in any spacecraft lateral direction. (2) Vibration inputs are defined at the base of the payload attach fitting.				
Three-Stage Delta II Vehicles				
Sine Test	Axis	Frequency (Hz)	Test Levels	Sweep Rate
Acceptance	Thrust/Lateral	5 to 100	Table 4-7	4 octaves/min
Protoflight	Thrust	5 7.4 7.4 to 100	0.64 g 1.40 g Figure 4-23 + 3.0 dB	4 octaves/min
	Lateral	5 5.2 5.2 to 100	0.64 g 0.70 g Figure 4-23 + 3.0 dB	4 octaves/min
Qualification	Thrust	5 7.4 7.4 to 100	0.64 g 1.40 g Figure 4-23 + 3.0 dB	2 octaves/min
	Lateral	5 5.2 5.2 to 100	0.64 g 0.70 g Figure 4-23 + 3.0 dB	2 octaves/min
Notes: (1) Lateral accelerations to be applied in any spacecraft lateral direction. (2) Vibration inputs are defined at the top of the payload attach fitting.				

The spacecraft sinusoidal vibration qualification test consists of one sweep through the specified frequency range using a logarithmic sweep rate of 2 octaves per minute. For spacecraft acceptance and protoflight testing, the test consists of one sweep through the specified frequency range using a logarithmic sweep rate of 4 octaves per minute. The sinusoidal vibration test input levels should be maintained within $\pm 10\%$ of the nominal test levels throughout the test frequency range.

When testing a spacecraft with a laboratory shaker, it is not within the current state of the art to duplicate at the shaker input the boundary conditions that actually occur in flight. This is notably evident in the spacecraft lateral axis during test, when the shaker applies large vibratory forces to maintain a constant acceleration input level at the spacecraft fundamental lateral test frequencies. The response levels experienced by the spacecraft at these fundamental frequencies during test are usually much more severe than those experienced in flight. The significant lateral loading to the spacecraft during flight is usually governed by the effects of spacecraft/launch vehicle dynamic coupling.

Where it can be shown by a spacecraft launch vehicle coupled-dynamic-loads analysis that the spacecraft or PAF/spacecraft assembly would experience unrealistic response levels during test, the sinusoidal vibration input level can be reduced (notched) at the fundamental resonances of the hardmounted spacecraft or PAF/spacecraft assembly to more realistically simulate flight loading conditions. This has been accomplished on many previous spacecraft in the lateral axis by correlating one or several accelerometers mounted on the spacecraft to the bending moment at

the PAF/spacecraft separation plane. The bending moment is then limited by (1) introducing a narrow-band notch into the sinusoidal vibration input program or (2) controlling the input by a servo system using a selected accelerometer on the spacecraft as the limiting monitor. A redundant accelerometer is usually used as a backup monitor to prevent shaker runaway.

When developing the sinusoidal vibration test plan, the customer should coordinate with the Delta Program Office.

4.2.4.4 Shock Testing. High-frequency pyrotechnic shock levels are very difficult to simulate mechanically on a shaker at the spacecraft-system level. The most direct method for this testing is to use a Delta II flight configuration PAF/spacecraft separation system and PAF structure with functional ordnance devices. Spacecraft qualification and protoflight shock testing are performed by installing the in-flight configuration of the PAF/spacecraft separation system and activating the system twice. Spacecraft shock acceptance testing is performed in a similar manner by activating the PAF/spacecraft separation system once.

4.2.5 Dynamic Analysis Criteria and Balance Requirements

Standard payload separation attitude and rate dispersions are shown in Table 4-12. Dispersions are defined for each vehicle configuration and consist of all known error sources. Dispersions are affected by spacecraft mass properties and center of gravity (CG) offsets. Mission-specific attitude and rate dispersions are defined in the payload/expended stage separation analysis.

Table 4-12. Standard Payload Separation Attitudes/Rates

Configuration	Spinning	PAF	Payload separation attitude and rate dispersions (3- σ values)			
			Attitude (deg)	Rate (dps)	Momentum vector	Cone angle
Two Stage	No	6306, 6019, 6915 ⁽¹⁾ , 4717 ⁽¹⁾	<3.0	<0.25 (/axis)	—	—
		5624, 6915 ⁽²⁾ , 4717 ⁽²⁾ , DPAF, 3715	<0.70	<3.0 (trans), <1.0 (roll)	—	—
	Up to 5 rpm (± 1 deg/sec)	5624, DPAF, 3715, 4717	—	—	<5.0 deg	<5.0 deg
Three Stage	Up to 100 rpm ($\pm 15\%$)	3712, 3724	—	—	<10.0 deg	<6.0 deg
	Despun (0 ± 5 rpm)	3712, 3724	<10.0	<7.0 (trans)	—	—
Note: Attitude/momentum vector pointing dispersions for two-stage missions are defined with respect to the customer-specified separation attitude. Attitude/momentum vector pointing dispersions for three-stage missions are defined with respect to the orientation of the third-stage centerline prior to spin-up/separation from the second stage. (1) With secondary latch system (2) Without secondary latch system						

4.2.5.1 Two-Stage Missions. Two-stage missions utilize the capability of the second stage to provide terminal velocity, roll, final spacecraft orientation, and separation.

Balance Requirements. The spacecraft lateral CG offset must be limited to provide acceptable loading, adequate control system performance, and acceptable tip-off angular rates imparted to the spacecraft. For missions that use a two-step (secondary latch) system, the spacecraft lateral CG offset is required to be within 2 in. from the vehicle centerline (3-sigma value, including measurement uncertainties). Larger values may be acceptable based on mission-specific analysis

and must be coordinated with the Delta Program Office. For missions using all other separation systems, i.e., springs, the 2-in. lateral CG offset requirement does not apply and mission-specific analysis is required to quantify the maximum allowable spacecraft lateral CG offset.

Two-Step Separation System. For missions in which there is a critical constraint on separation tipoff angular rate, a two-step (secondary latch) separation system can be employed. The 6306, 6019, 6915, and 4717 PAFs support secondary latch systems. The second stage and spacecraft are held together by loose-fitting latches following primary separation of the nuts and bolts or clampbands. After a sufficient time (30 sec) for the angular rates to dissipate, the latches are released and the second-stage retro thrust provides the required relative separation velocity from the spacecraft.

Second-Stage Roll Rate Capability. For some two-stage missions, the spacecraft may require a low roll rate at separation. The Delta II second stage can command roll rates up to 5 rpm (30 deg/sec) using control jets. Higher roll rates are also possible; however, accuracy is degraded as the rate increases. Roll rates higher than 5 rpm (30 deg/sec) must be assessed relative to specific spacecraft requirements. Significantly higher roll rates may require the use of a spin-table assembly.

4.2.5.2 Three-Stage Missions. Three-stage missions employ a spin-stabilized upper stage. The spin table, third-stage motor, PAF, and spacecraft combination are accelerated to the initial spin rate prior to third-stage ignition by the activation of two to eight spin rockets mounted on the spin table. Two rocket sizes are available to achieve the desired spin rate.

Spin Balance Requirements. To minimize the cone angle and momentum vector pointing error of the spacecraft/third-stage combination after second-stage separation, it is necessary that the imbalance of the spacecraft alone be within specified values. The spacecraft should be balanced to produce a $3\text{-}\sigma$ maximum CG within 1.3 mm (0.05 in.) of the centerline, and a $3\text{-}\sigma$ maximum principal axis misalignment of less than 0.25 deg with respect to the centerline. The spacecraft centerline is defined as a line perpendicular to the separation plane of the spacecraft that passes through the center of the theoretical spacecraft/PAF diameter (refer to Section 5).

A composite balance of the entire third-stage/spacecraft assembly is not required. It has been shown analytically that the improvements derived from a composite balance were generally small and do not justify the handling risk associated with spacecraft spin balance on a live motor.

For most spinning spacecraft, it has been demonstrated that the static and dynamic balance limits defined herein can be satisfied. For missions where such a constraint may be difficult to satisfy, the effects of broadened tolerances are analyzed on a per-case basis.

The angular momentum/velocity pointing errors and cone angle are highly dependent upon the spacecraft spin rate, CG location, moments and products of inertia, NCS operation during

upper-stage motor burn and coast periods, and the spacecraft energy dissipation sources. The Delta Program Office, therefore, should be consulted if the above constraints cannot be met. Pointing errors and cone angles are estimated as required for the mission-specific spacecraft characteristics.

Spin Rate Capability. Spin-up of the third stage/spacecraft combination is accomplished by activating small rocket motors mounted on the spin table that supports the payload. Spin direction is clockwise, looking forward. Spin rates from 30 to 110 rpm are attainable for a large range of spacecraft roll moments of inertia (MOI) as shown in Figure 4-29 for the Star-48B third stage motor and 30 to 60 rpm as shown in Figure 4-30 for the Star-37FM third-stage motor. Nominal spin rates can be provided within ± 5 rpm for any value specified in the region of spin rate capability. Once a nominal spin rate has been determined, 3- σ variations in relevant parameters will cause a spin rate prediction uncertainty of $\pm 15\%$ about that nominal value at spacecraft separation.

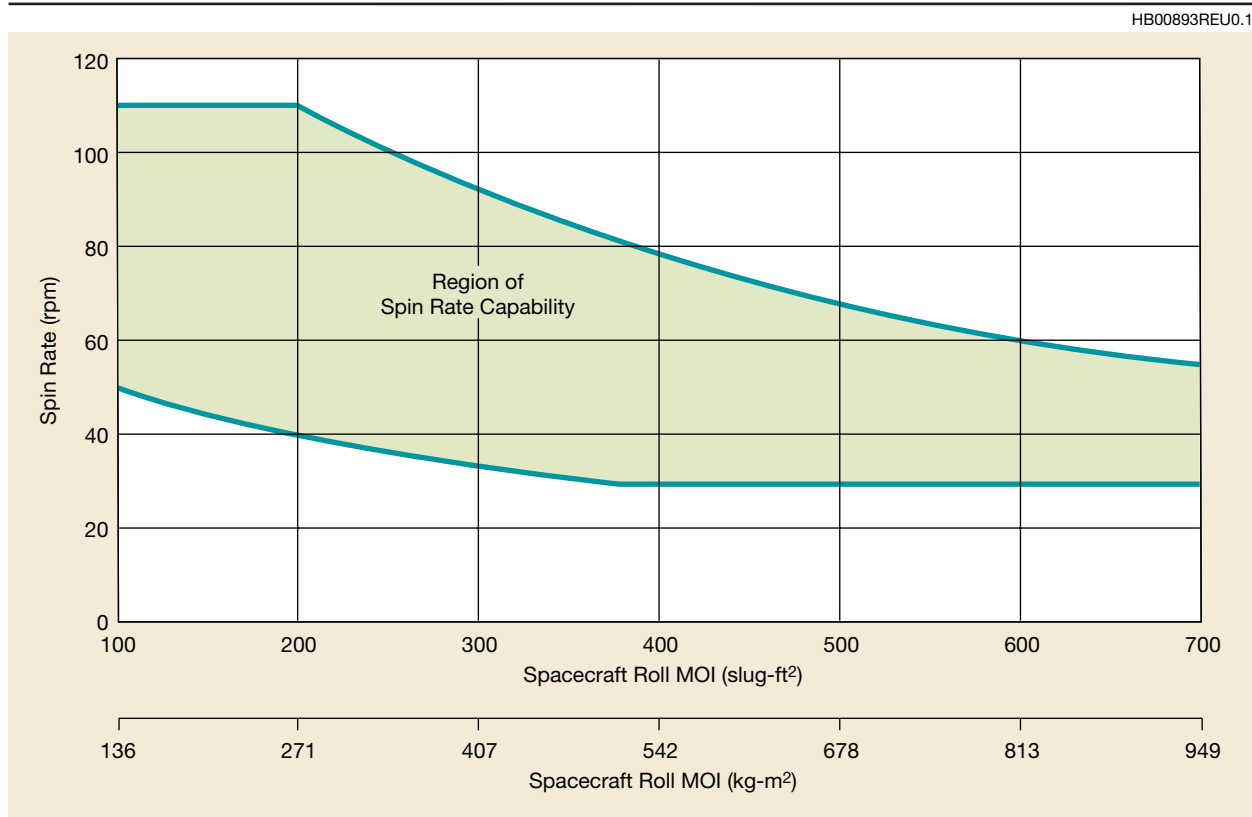


Figure 4-29. Delta II Star-48B Spin Rate Capability

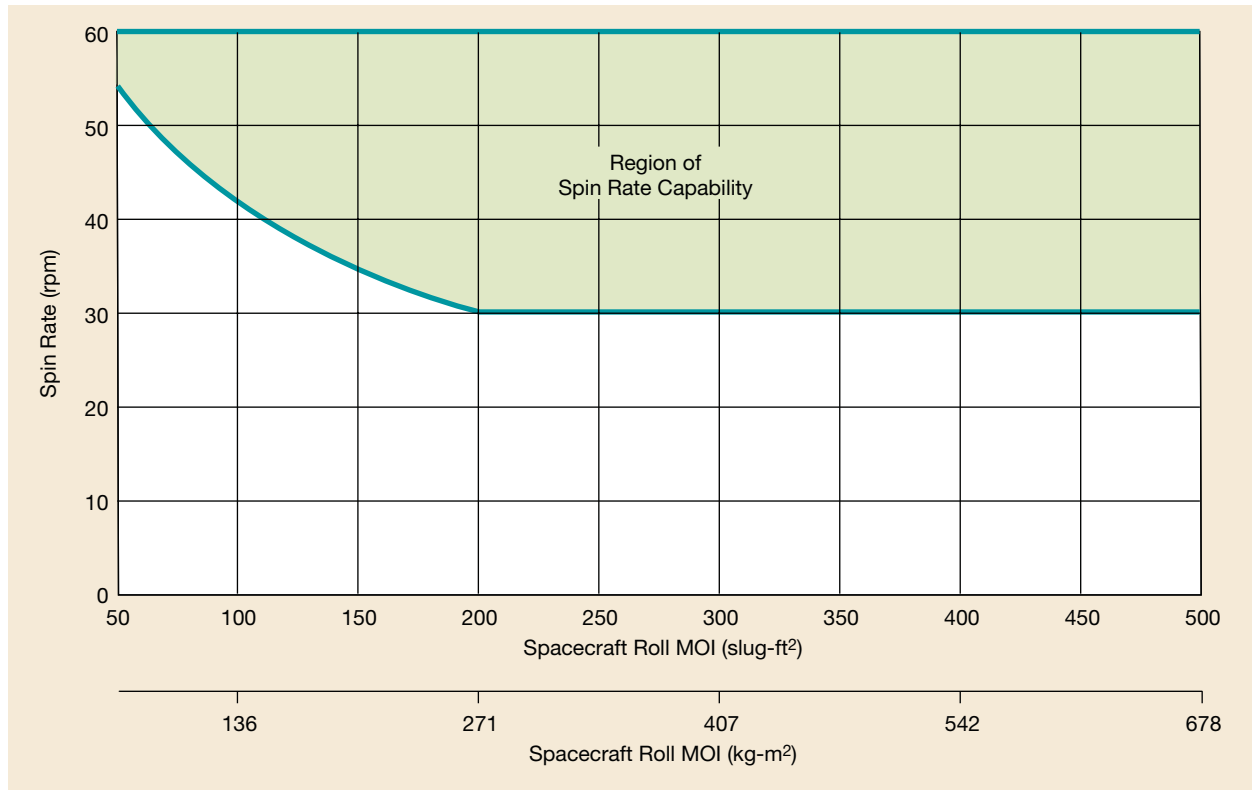


Figure 4-30. Delta II Star-37FM Spin Rate Capability

Because orbit errors are dependent upon spin rate, the magnitude of the orbit errors must be assessed relative to the mission requirements and spacecraft mass properties before final resolution of the spin rate for a specific spacecraft mission is accomplished.

For three-stage missions requiring low to zero spin rate at spacecraft separation, a yo-yo despin system can be employed to reduce the spin rate prior to spacecraft separation. Negative spin rates can be targeted with the despin system to compensate for the effects of residual spinning of propellants in the spacecraft tanks. The uncertainty in the spin rate after despin is a function of the uncertainty in the spacecraft spin MOI. Three-sigma spin rate uncertainties of ± 5 rpm can be achieved for spacecraft spin MOI uncertainties of $\pm 5\%$. If a tighter spin rate tolerance is required, measurement of the spacecraft spin MOI may be required.

Angular Acceleration. The maximum angular acceleration loads imparted to the spacecraft occur during spin-up. The maximum angular acceleration that will occur while attaining a desired spin rate is fixed by spin motor thrust characteristics.

The Delta II spin system uses two different spin motors in various combinations to attain specified spin rates. Figures 4-31 and 4-32 show the maximum angular acceleration that could be incurred by the system for the Star-48B and Star-37FM motors, respectively. Two curves are shown on each figure, one for a nominal propellant temperature condition of 70°F (21.1°C) and the other for a maximum spin rocket allowable temperature of 130°F (54.4°C) and $+3\text{-}\sigma$ burn rate.

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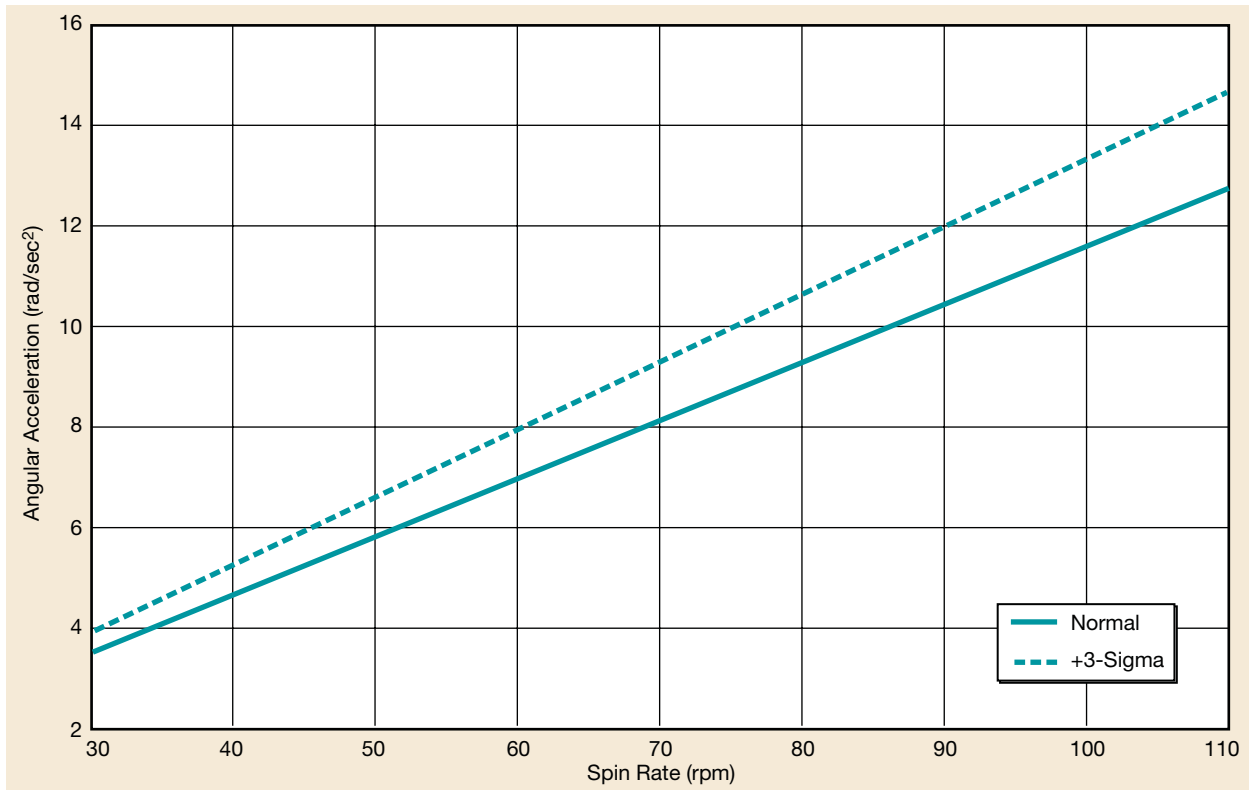


Figure 4-31. Maximum Expected Angular Acceleration vs. Spin Rate—Star-48B

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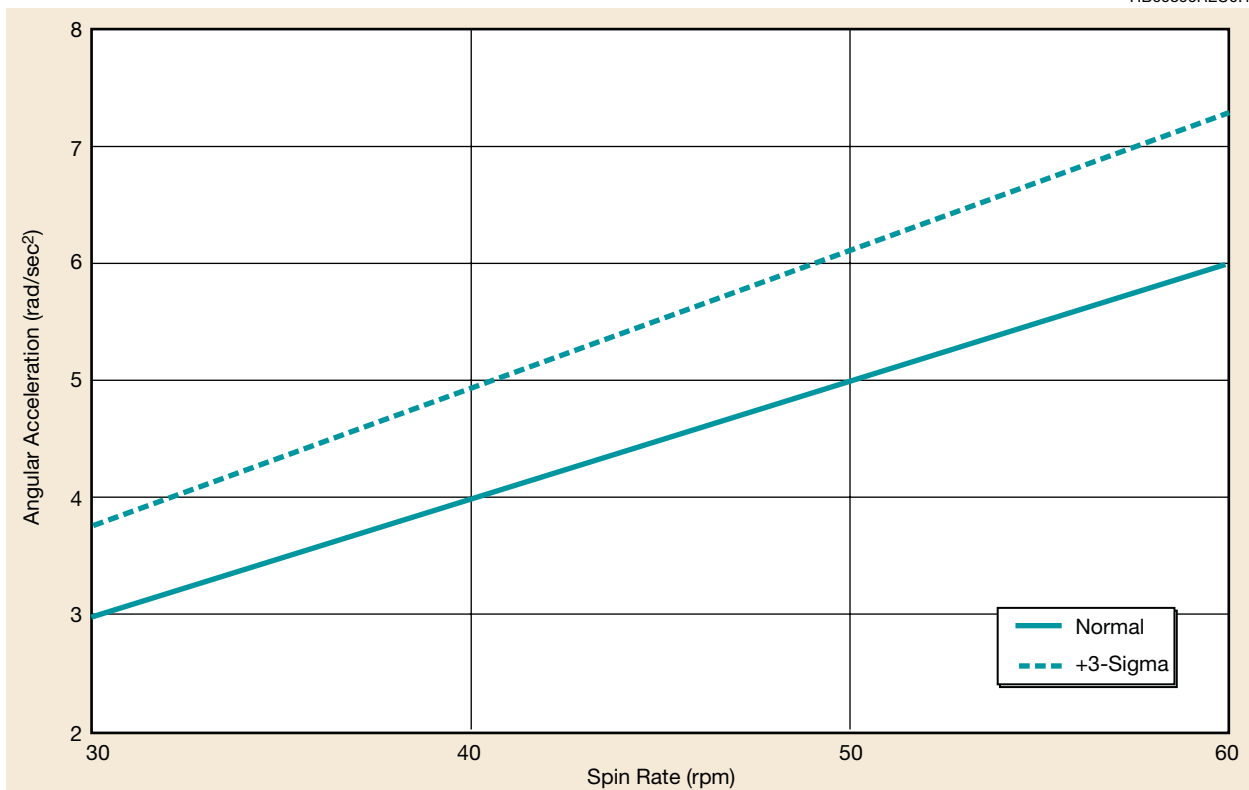


Figure 4-32. Maximum Expected Angular Acceleration vs. Spin Rate—Star-37FM

Figures 4-28 and 4-29 are based on the maximum motor thrust, which occurs for a duration of approximately 30 msec during ignition. If the maximum acceleration is excessive, a detailed angular acceleration history can be provided for customer evaluation. If not tolerable, special provisions such as sequential firing of spin motors can be considered.

Spacecraft Energy Dissipation During Coast Periods. Dissipation of energy caused by spacecraft nutation dampers, fuel slosh in the propellant tanks, inertial propellant waves, flexible antennas, etc., can cause divergence in the cone angle between the spin axis of the spacecraft/third-stage combination and its angular momentum vector when the spin MOI is less than the transverse MOI, affecting orbit accuracy, clearance between the spacecraft and the PAF during separation, and spacecraft coning/momentum pointing after separation.

The effect of energy dissipation is highly dependent on the mass properties and spin rate of the spacecraft/third stage combination. In order for Boeing to evaluate the effect on a particular mission, the customer must provide a worst-case energy dissipation time constant for the combined third stage and spacecraft for conditions before and after third-stage burn. Time constants of 150 sec (pre-burn) and 50 sec (post-burn) are the design goal, but additional analysis would be required for values below 150 sec and 50 sec. Mass properties for the Star-48B and the Star-37FM third stages are shown in Table 4-13.

Table 4-13. Third-Stage Mass Properties

	Star-48B		Star-37FM	
	Before motor ignition	After motor burnout	Before motor ignition	After motor burnout
Weight (kg/lb)	2227/4910	206/454	1245/2745	172/380
CG aft of spacecraft separation plane (mm/in.)	780/30.7	815/32.1	709/27.9	676/26.6
Spin MOI (kg-m ² /slug-ft ²)	390/288	50/37	141/104	34/25
Transverse MOI (kg-m ² /slug-ft ²)	457/337	95/70	199/147	58/43

NCS nominal characteristics are listed in Table 4-14. For Star-48B missions, spacecraft weights less than 1250 lb may require additional NCS modifications due to the high third-stage burnout acceleration.

Table 4-14. Nutation Control System Nominal Characteristics

Propellant weight	2.72 kg/6.00 lb
Helium prepressure	2.26 $\cdot 10^5$ N/m ² /400 psia
Thrust	164.6 N/37 lb
Minimum Isp (pulsing mode)	202.5 sec
Pressure at end of blowdown	9.7 $\times 10^5$ N/m ² /141 psia
Transverse rate threshold	2 deg/sec

Nutation Control System. The NCS is designed to maintain small cone angles of the combined upper stage and spacecraft and operates during the motor burn and post-burn coast phase. The NCS is required for missions using the yo-yo despin system.

The NCS design concept uses a single-axis rate gyro assembly (RGA) to sense coning and a monopropellant (hydrazine) propulsion module to provide control thrust. The RGA angular rate signal is processed by circuitry that generates thruster on/off commands.

Section 5 *PAYLOAD INTERFACES*

This section presents the detailed descriptions and requirements of the mechanical and electrical interfaces between the payload and the Delta II family of launch vehicles for two- and three-stage missions. Boeing uses a heritage design approach for its payload attach fittings (PAFs); hence, unique interface requirements can be accommodated through natural extension of proven designs.

5.1 DELTA II PAYLOAD ATTACH FITTINGS

The Delta II vehicle offers several PAFs for use with three available payload fairings (Figures 5-1 and 5-2). The first two digits of each PAF's designation indicate its payload interface diameter in inches, and the last two digits indicate the PAF's height in inches. All PAFs are designed such that payload electrical interfaces and separation springs can be located to accommodate specific customer requirements. Because of the development time and cost associated with a custom PAF, it is advantageous to use existing PAF designs. Selection of an appropriate PAF should be coordinated with the Delta Program Office as early as possible.

5.1.1 Customer-Provided Payload Attach Fittings

Spacecraft customers may use their own PAF instead of using a Delta-provided one. If the customer prefers to use its own PAF, rather than selecting a Delta-provided PAF, special interface requirements must be coordinated with the Delta Program Office. In addition to the typical launch vehicle flight interface requirements, there are special hardware provisions required for operations, including spacecraft transport to the pad and mating to the launch vehicle. Requirements include but are not limited to index holes and specific holes with attached nuts or inserts on the spacecraft side of the interface. These features facilitate proper clocking during the mating operation with the launch vehicle. It is strongly recommended that the customer coordinate with the Delta Program Office early in the mission-design process.

5.2 PAYLOAD ATTACH FITTINGS FOR THREE-STAGE MISSIONS

There are four standard PAFs available for three-stage missions. The 3712 PAF (Figure 5-3) comes in three forward flange configurations, designated 3712A, 3712B, and 3712C. The 3724 PAF is available with one forward flange configuration, designated 3724C. The maximum clampband flight preload for the 3712 and 3724 configurations is given in Table 5-1.

The Delta II vehicle third stage consists of either an Alliant Techsystems Star-48B or Star-37FM solid rocket motor, a cylindrical PAF with a clamp assembly and four separation spring actuators, a nutation control system (NCS) that is standard with the Star-48B and optional for the Star-37FM, an ordnance sequencing system, and a yo-weight system for tumbling the stage after spacecraft separation. If required, a yo-yo weight despin system can be incorporated into the

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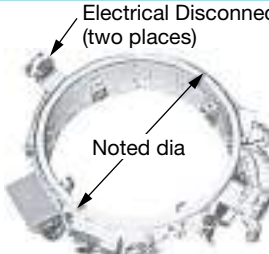
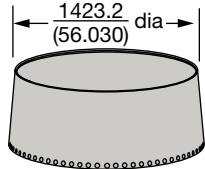
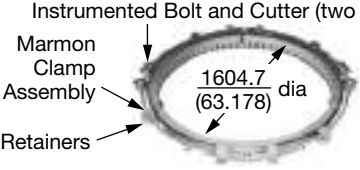
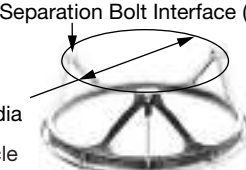

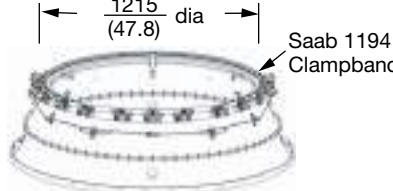
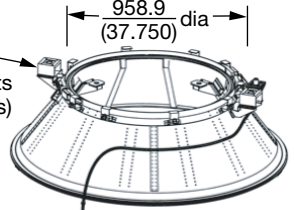
Model/ Mass	Note: All dimensions are in mm (in.)	Separation Mechanism	Features
3712A 3712B 3712C 45.4 kg/ 100 lb 3724C 56.7 kg/ 125 lb		Noted dia Clampband, Springs	Three-Stage Missions: Two instrumented studs verify clampband preload. Retention system prevents clampband recontact. Four matched spring actuators reduce separation-induced tipoff rates. Two 37-pin spacecraft interface electrical connectors across the separation plane. Note: <u>945.3</u> dia for 3712A (37.215) <u>958.9</u> dia for 3712B, 3712C, and 3724C (37.750)
5624 43.1 kg/ 95 lb		<u>1423.2</u> dia (56.030) Clampband, Springs	Two-Stage Missions: Two instrumented studs verify clampband preload. Retention system prevents clampband recontact. Four matched spring actuators reduce tipoff rates. Two 37-pin spacecraft interface electrical connectors across the separation plane.
6306 47.6 kg/ 105 lb		<u>1604.7</u> dia (63.178) Clampband and Secondary Latch System	Two-Stage Missions: Two instrumented studs verify clampband preload. Retention system prevents clampband recontact. Secondary latch system minimizes tipoff rates. Second stage backs away using helium retro system to prevent recontact after spacecraft separation. Up to two 37-pin spacecraft interface electrical connectors from the PLF to the spacecraft.
6019 70.3 kg/ 155 lb		Three Separation Bolts and Secondary Latch System	Two-Stage Missions: Three hard-point attachments released by redundantly initiated explosive nuts. Secondary latch system minimizes tipoff rates. Second stage backs away using helium retro system to prevent recontact after spacecraft separation. Up to two 37-pin spacecraft interface electrical connectors from the PLF to the spacecraft.
6915 93.0 kg/ 205 lb		Four Separation Bolts and Secondary Latch System or Springs	Two-Stage Missions: Four hard-point attachments released by redundantly initiated explosive nuts. Secondary latch system or four matched spring actuators may be used based on tipoff rate requirements. Second stage backs away using helium retro system to prevent recontact after spacecraft separation if secondary latch system used. Up to two 37-pin spacecraft interface electrical connectors from the PLF to the spacecraft.
4717 81.6 kg/ 180 lb		<u>1215</u> dia (47.8) Clampband & Secondary Latch System or Springs	Two-Stage Missions: SAAB 1194 separation system. Two instrumented spacers verify clampband preload. Secondary latch system or four matched spring actuators may be used based on tipoff rate requirements. Second stage backs away using helium retro system to prevent recontact after spacecraft separation if secondary latch system used. Up to two 37-pin spacecraft interface electrical connectors from the PLF to the spacecraft for secondary latch use, or two 37-pin spacecraft interface electrical connectors across separation plane.
3715 86.2 kg/ 190 lb		<u>958.3</u> dia (37.750) Clampband, Springs	Two-Stage Missions: Two instrumented studs verify clampband preload. Retention system prevents clampband recontact after spacecraft separation. Four matched spring actuators reduce separation-induced tip off rates. Two 37-pin spacecraft interface electrical connectors across the separation plane.

Figure 5-1. Delta II Payload Adapters and Interfaces

HB5T072015.3

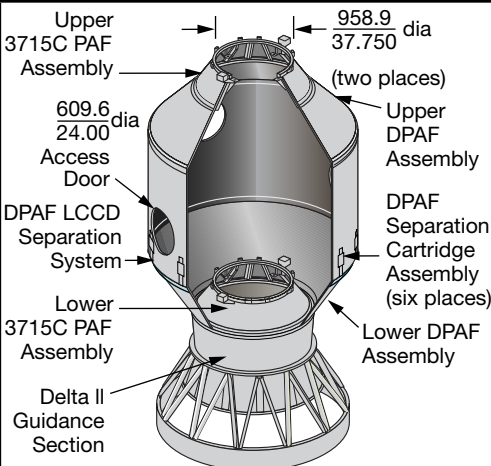
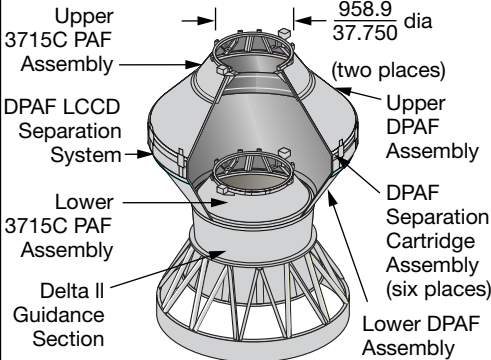
Model/ Mass	Note: All dimensions are in mm (in.)	Separation Mechanism	Features
Dual- Payload Attach Fitting (DPAF) 362.9 kg/ 800 lb		958.9 (37.750) dia Clampband, Springs	Dual-Manifest Missions: Common spacecraft interface on both upper and lower PAF assemblies. Two instrumented studs verify clampband preload. Retention system prevents clampband recontact. Four matched spring actuators reduce separation-induced tipoff rates. Line charge coupling device (LCCD) separates the DPAF structure circumferentially. DPAF structure pushed away using six matched spring cartridge assemblies. Two 37-pin spacecraft interface connectors across the separation plane.
Reduced Height Dual- Payload Attach Fitting (RHDPAF) 308.4 kg/ 680 lb		958.9 (37.750) dia Clampband, Springs	Dual-Manifest Missions: Common spacecraft interface on both upper and lower PAF assemblies. Two instrumented studs verify clampband preload. Retention system prevents clampband recontact. Four matched spring actuators reduce separation-induced tipoff rates. Line charge coupling device (LCCD) separates the DPAF structure circumferentially. DPAF structure pushed away using six matched spring cartridge assemblies. Two 37-pin spacecraft interface connectors across the separation plane.

Figure 5-2. Delta II Dual Payload Attach Fittings

HB00773REU0.2



Figure 5-3. 3712 Payload Attach Fitting (PAF)

stack as a nonstandard option in place of the yo-weight system to despin the spacecraft prior to separation. The pre- and post-burn mass properties of the stage are summarized in Table 4-17, Section 4.

In general, the component, sequencing, and separation system designs are the same for all three-stage applications. The spacecraft is fastened to the PAF by a two-piece V-block-type clamp assembly, that is secured by two instrumented studs for clampband tensioning. Spacecraft separation is initiated by actuation of ordnance cutters that sever the two studs.

Table 5-1. Maximum Clampband Assembly Preload

PAF	Max flight preload (N/lb)	NCS Blowdown	Spacecraft PAF flange angle (deg)
3712A	30,248/6800	Yes	15
3712B/3712C	17,348/3900	Yes	20
3724C	14,679/3300	No	20

002250.3

Clampband assembly design is such that cutting either stud will permit spacecraft separation. Springs assist in retracting the clampband assembly into retainers after release. A relative separation velocity ranging from 0.6 to 2.4 m/s (2 to 8 ft/sec) is imparted to the spacecraft by four spring actuators. Specific mission-oriented pads may be provided on the PAF at the separation plane to interface with spacecraft separation switches (Figure 5-4). A yo-weight tumble system imparts a coning motion to the expended third-stage motor 2 sec after spacecraft separation to prevent recontact with the spacecraft.

All hardware necessary for mating and separation (e.g., PAF, clampband assembly, studs, explosives, and timers) remains with the PAF upon spacecraft separation. Table 5-2 applies to the various PAF configuration drawing notes that accompany this section.

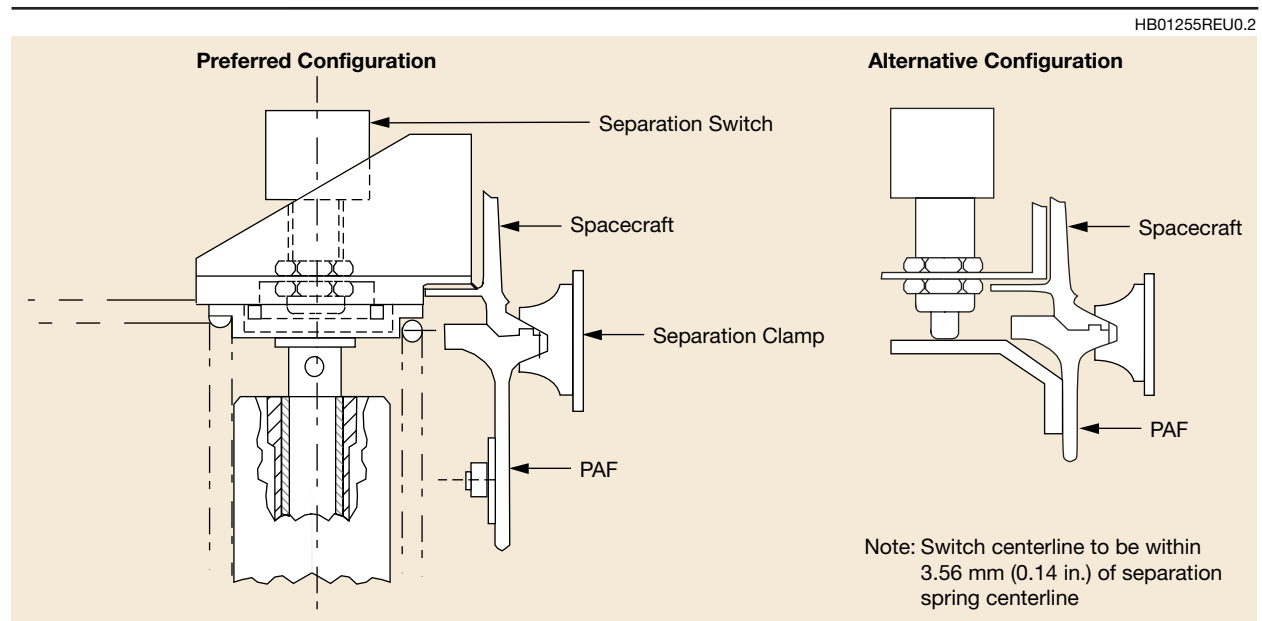


Table 5-2. Notes Used in Configuration Drawings

1. Interpret dimensional tolerance symbols in accordance with American National Standards Institute (ANSI) Y14.5M-1982. The symbols used in this section are as follows:		
Flatness		
Circularity		
Parallelism		
Perpendicularity (squareness)		
Angularity		
Circular runout		
Total runout		
True position		
Concentricity		
Profile of a surface		
Diameter		
2. Unless otherwise specified, tolerances are as follows:		
	Decimal	
mm		0.X = ± 0.7 0.XX = ± 0.38
in.		0.XX = ± 0.03 0.XXX = ± 0.015
Angles		= ± 0 deg. 30 min
3. Dimensions apply at 69°F (20°C) with interface in unrestrained condition.		
4. All machine surface roughness is $\sqrt{125}$ per ANSI B46.1, 1985.		
5. The V-block/PAF mating surface is chemically conversion-coated per MIL-C-5541, Class 3.		

002249.3

Figures 5-5 and 5-6 show the capabilities of the 3712 and 3724 PAFs in terms of spacecraft weight and CG location above the separation plane. The capability of a specific spacecraft (with its own unique mass, size, and flexibility) may vary from that presented; therefore, as the spacecraft configurations finalized, the Delta Program will initiate a coupled-loads analysis to verify that launch vehicle structural capability is not exceeded. The flange configurations and their associated spacecraft interface requirements are shown in Figures 5-7 through 5-19.

For spacecraft that require a longer PAF to eliminate interference with the third stage, a cylindrical extension adapter with customized length can be inserted between the PAF and the third stage. The extension adapter reduces the spacecraft allowable CG capability by approximately the length of the adapter.

Note that the discussion herein provides only a guideline for PAF selection, the actual PAF used for the mission is selected after detailed discussions with the customer since other requirements involving separation such as tip-off rates, spring forces, etc. are also considered.

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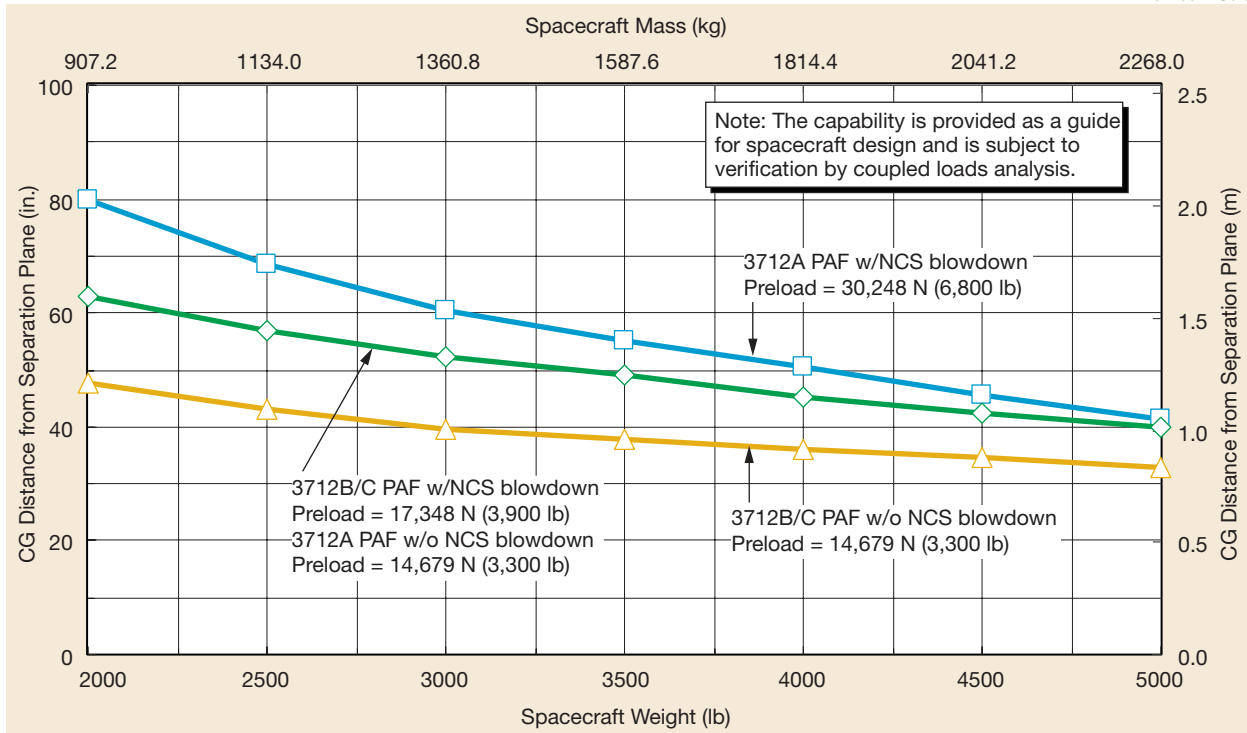


Figure 5-5. Capability of 3712 PAF

HB01257REU0.4

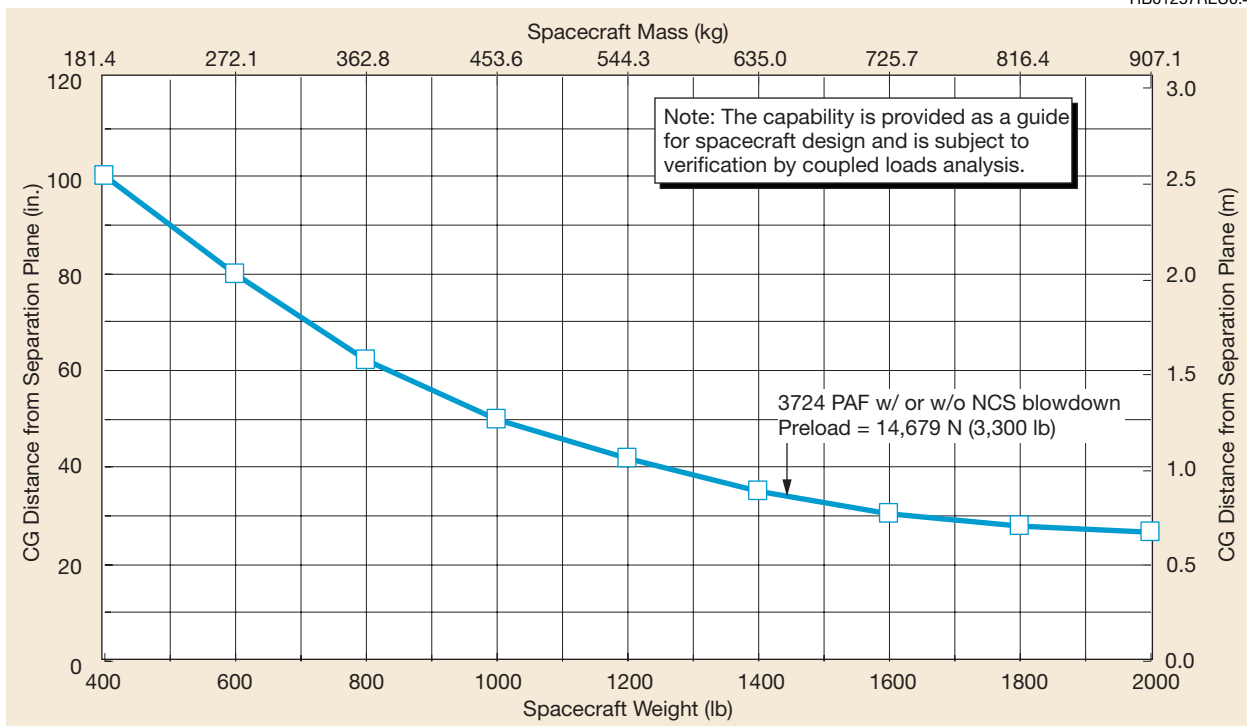


Figure 5-6. Capability of 3724 PAF

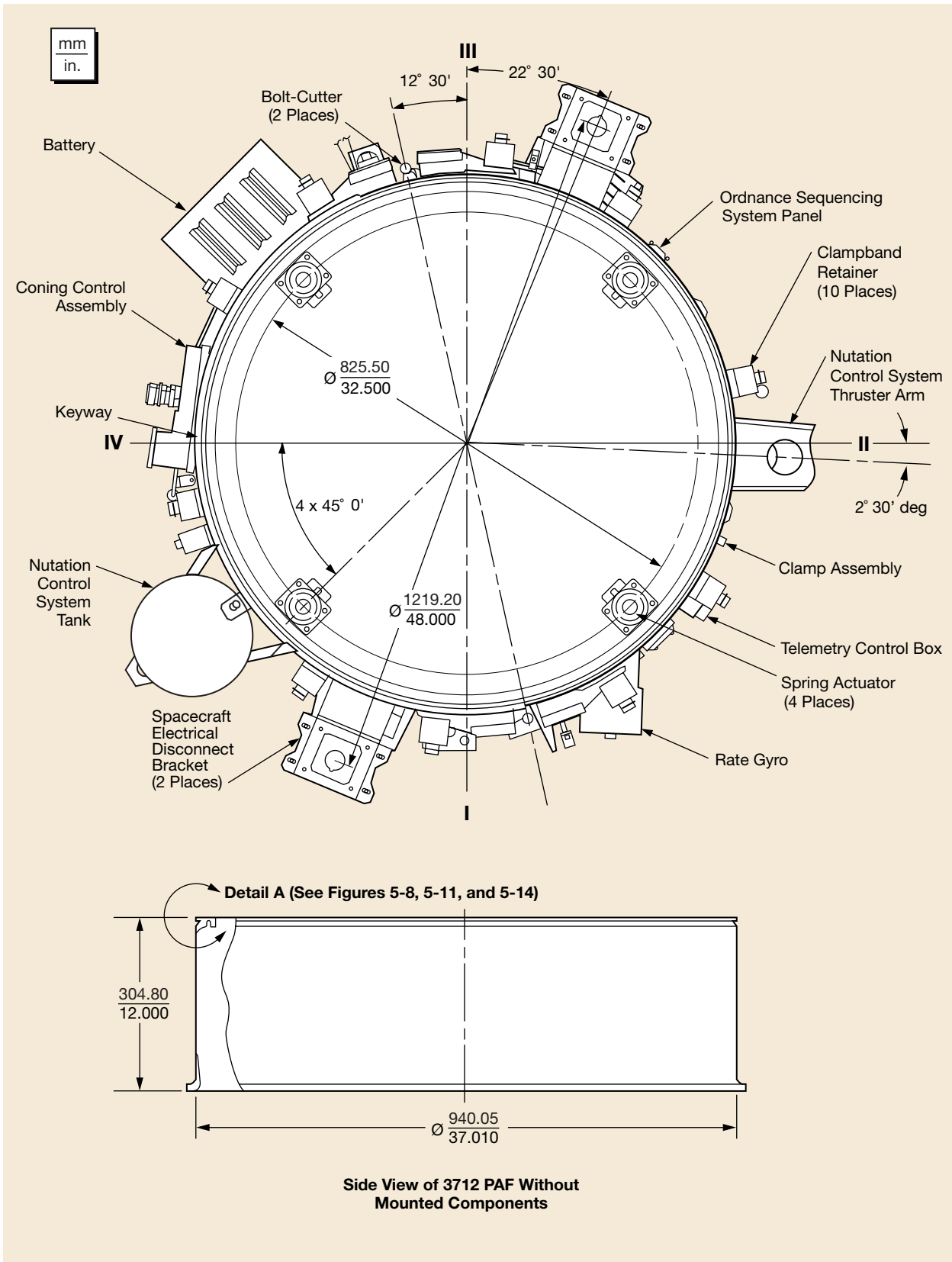


Figure 5-7. 3712 PAF Detailed Assembly

HB00865REU0.5

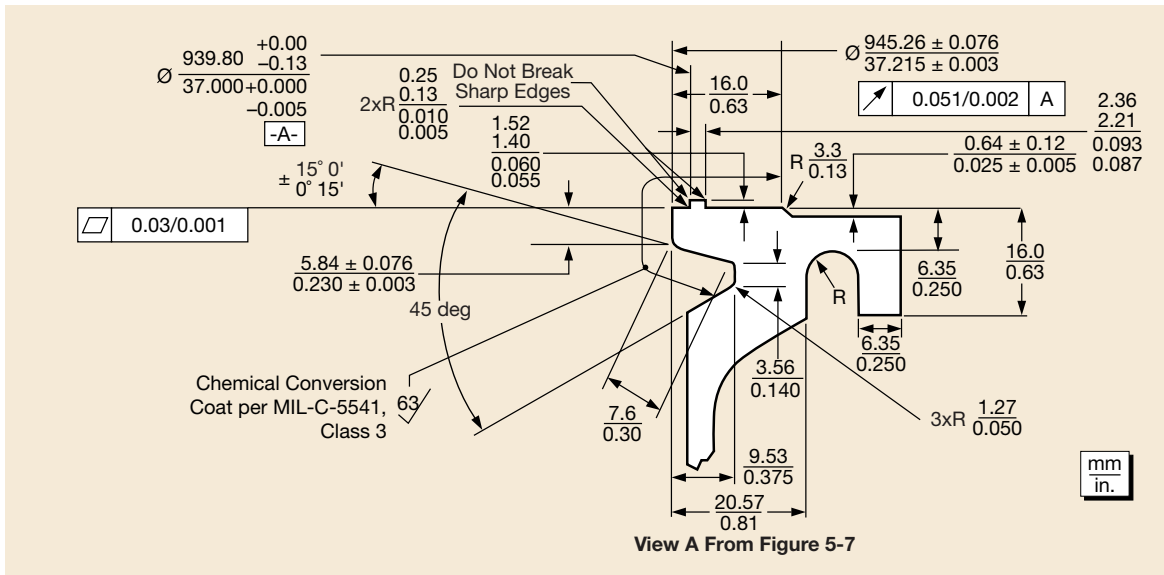


Figure 5-8. 3712A PAF Detailed Dimensions

HB00866REU0.5

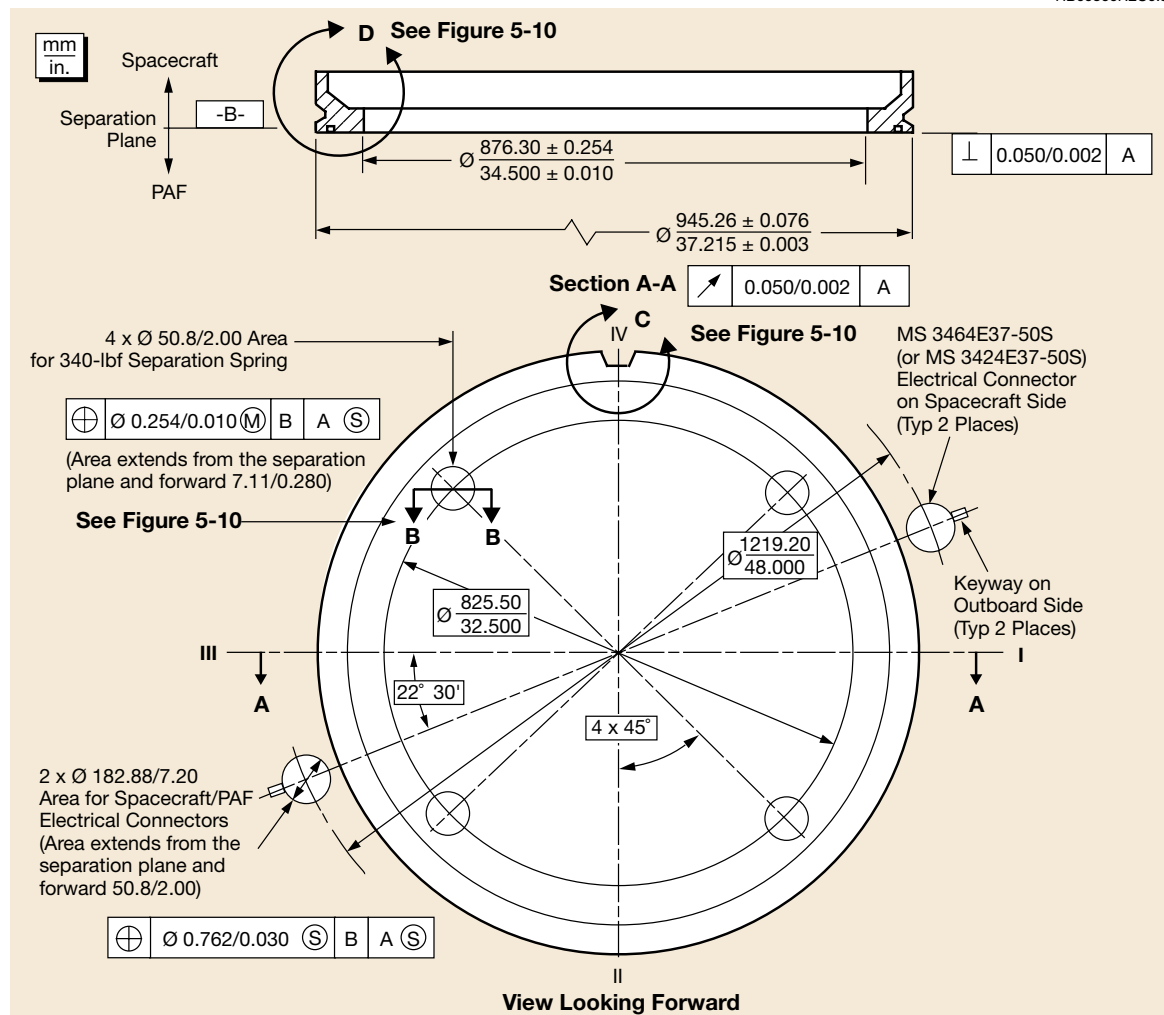


Figure 5-9. Dimensional Constraints on Spacecraft Interface to 3712A PAF

HB00867REU0.9

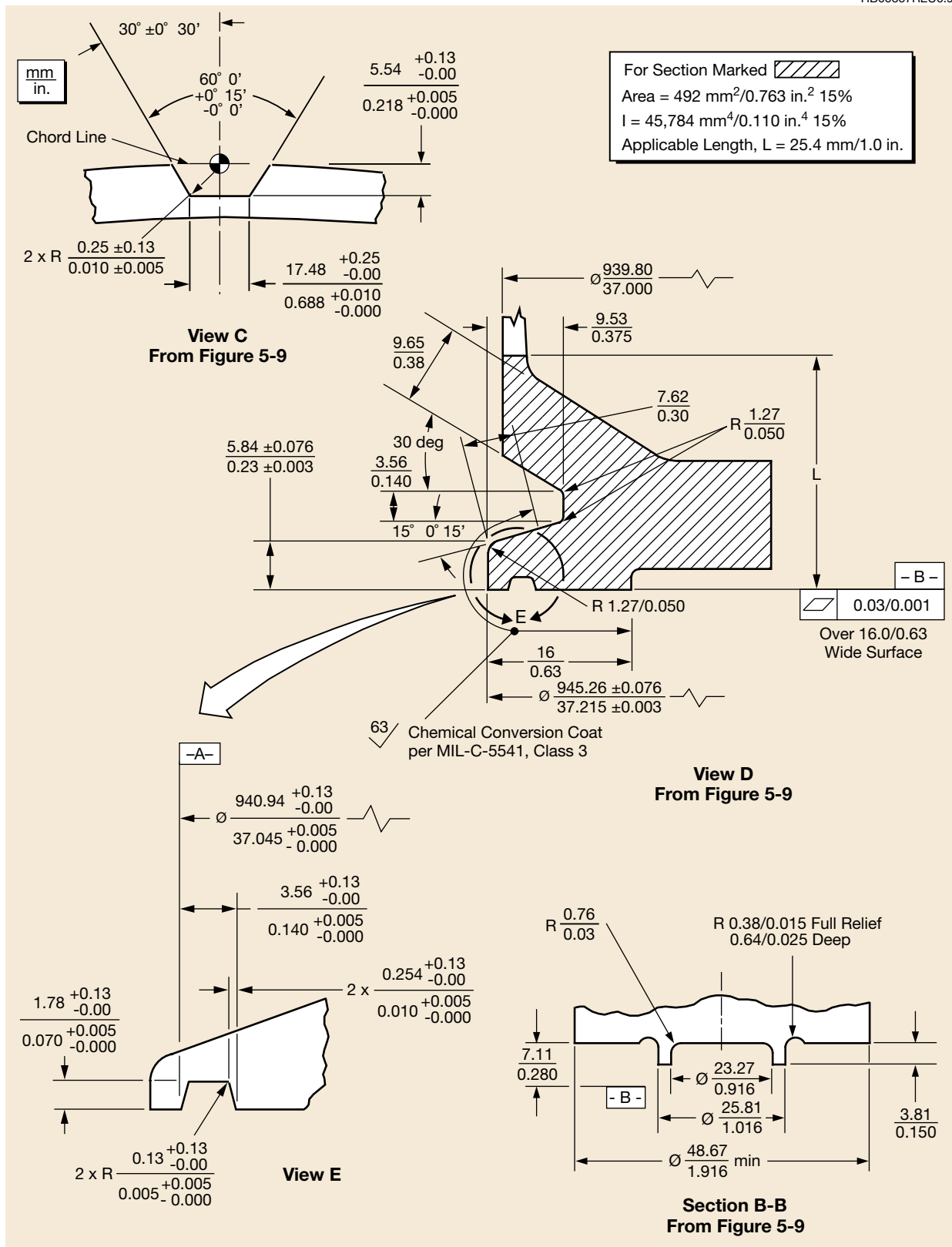


Figure 5-10. Dimension Constraints on Spacecraft Interface to 3712A PAF
(Views C, D, E, and Section B-B)

HB00868REU0.6

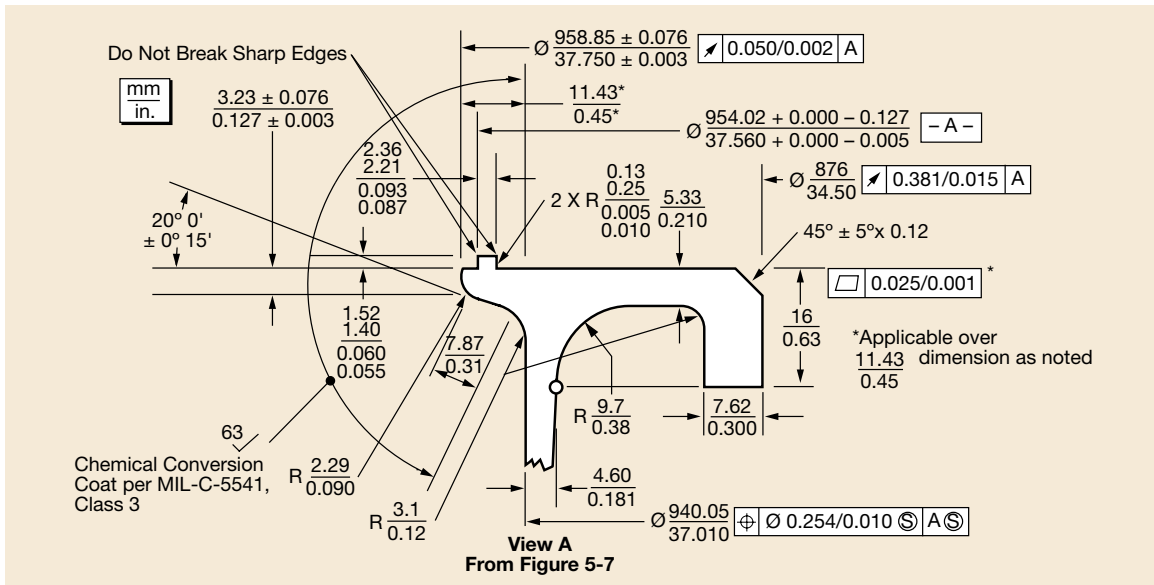


Figure 5-11. 3712B PAF Detailed Dimensions

HB00869REU0.6

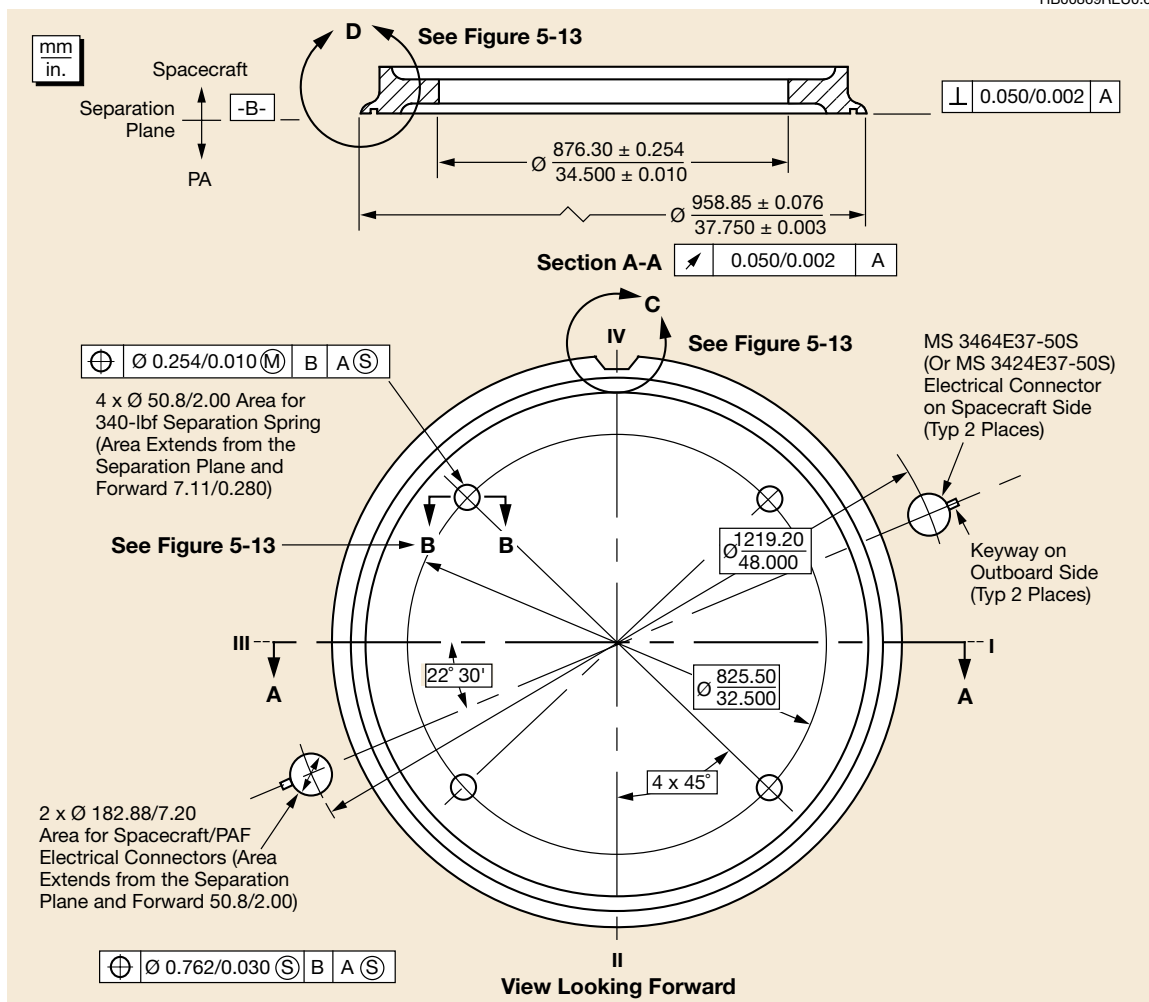


Figure 5-12. Dimensional Constraints on Spacecraft Interface to 3712B PAF

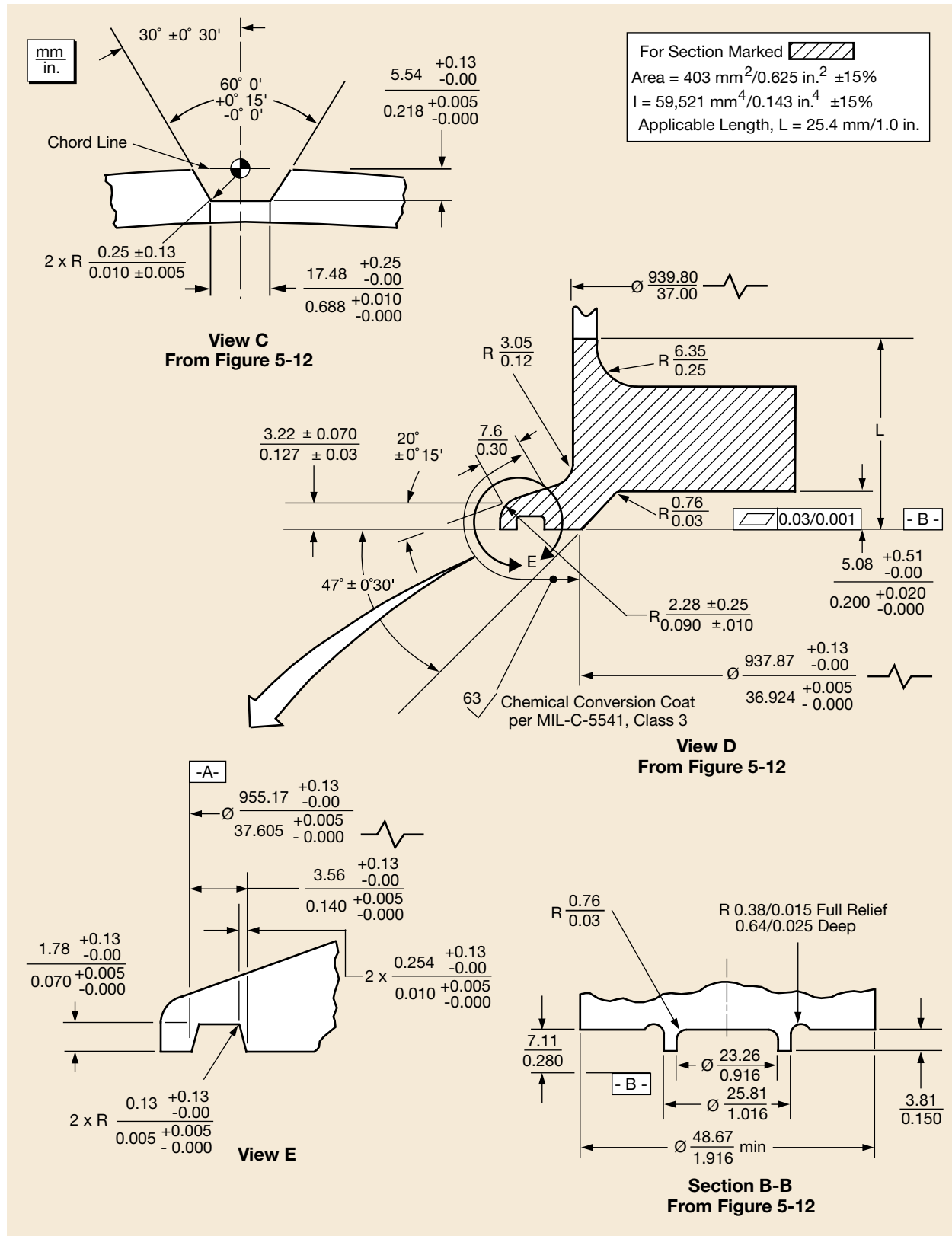


Figure 5-13. Dimensional Constraints on Spacecraft Interface to 3712B PAF (Views C, D, and E and Section B-B)



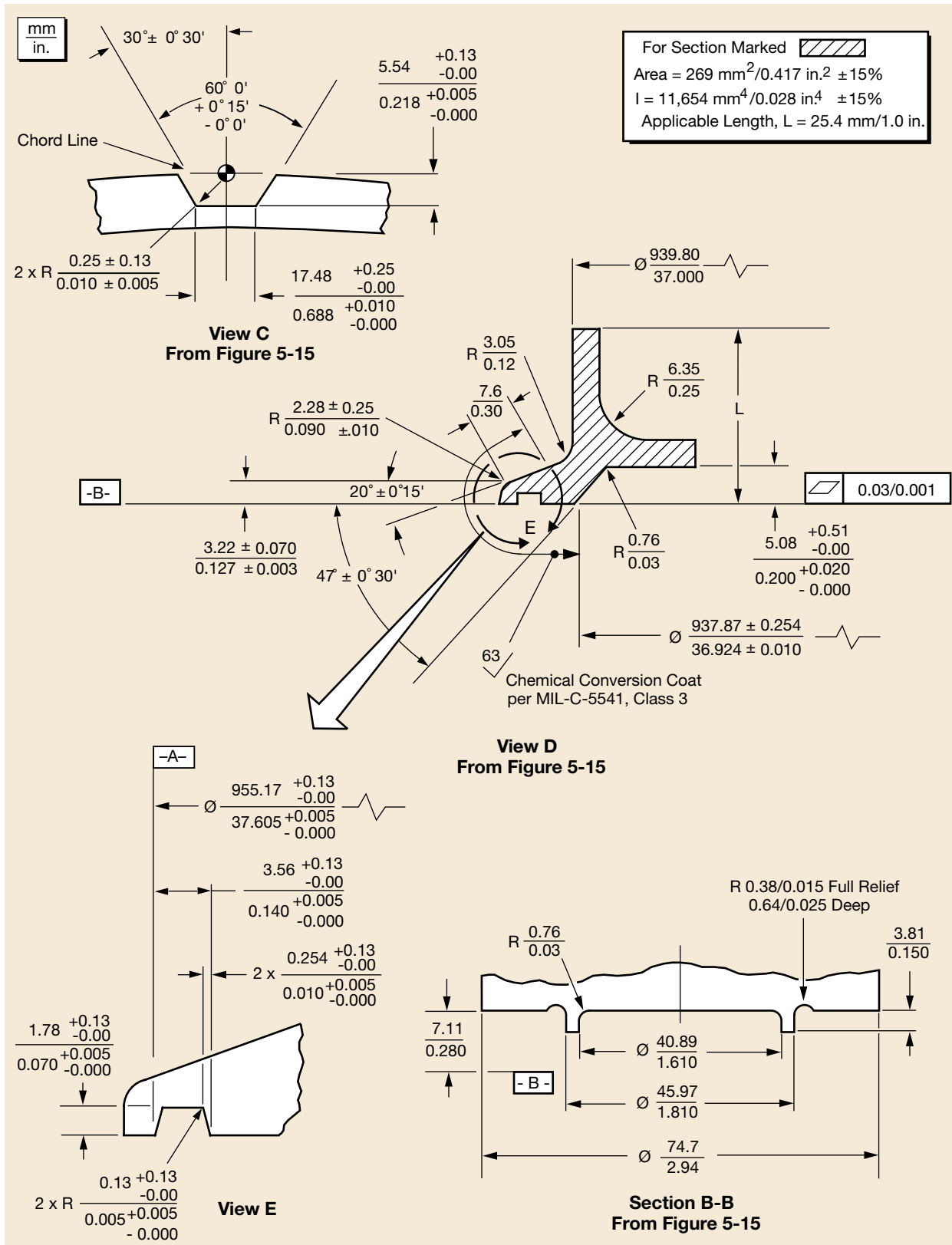


Figure 5-16. Dimensional Constraints on Spacecraft Interface to 3712C and 3724C PAFs (View C, D, E and Section B-B)

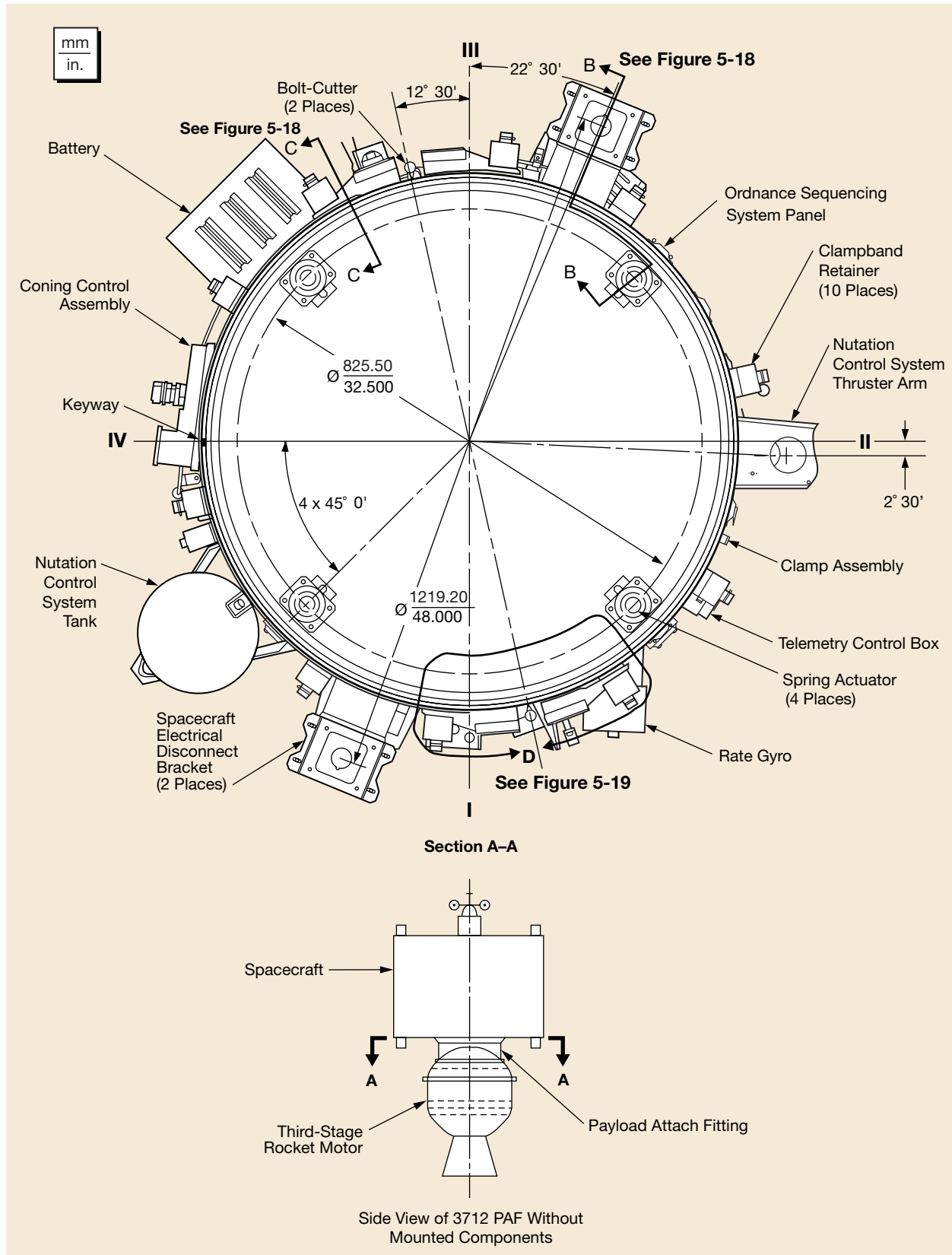


Figure 5-17. 3712 PAF Interface

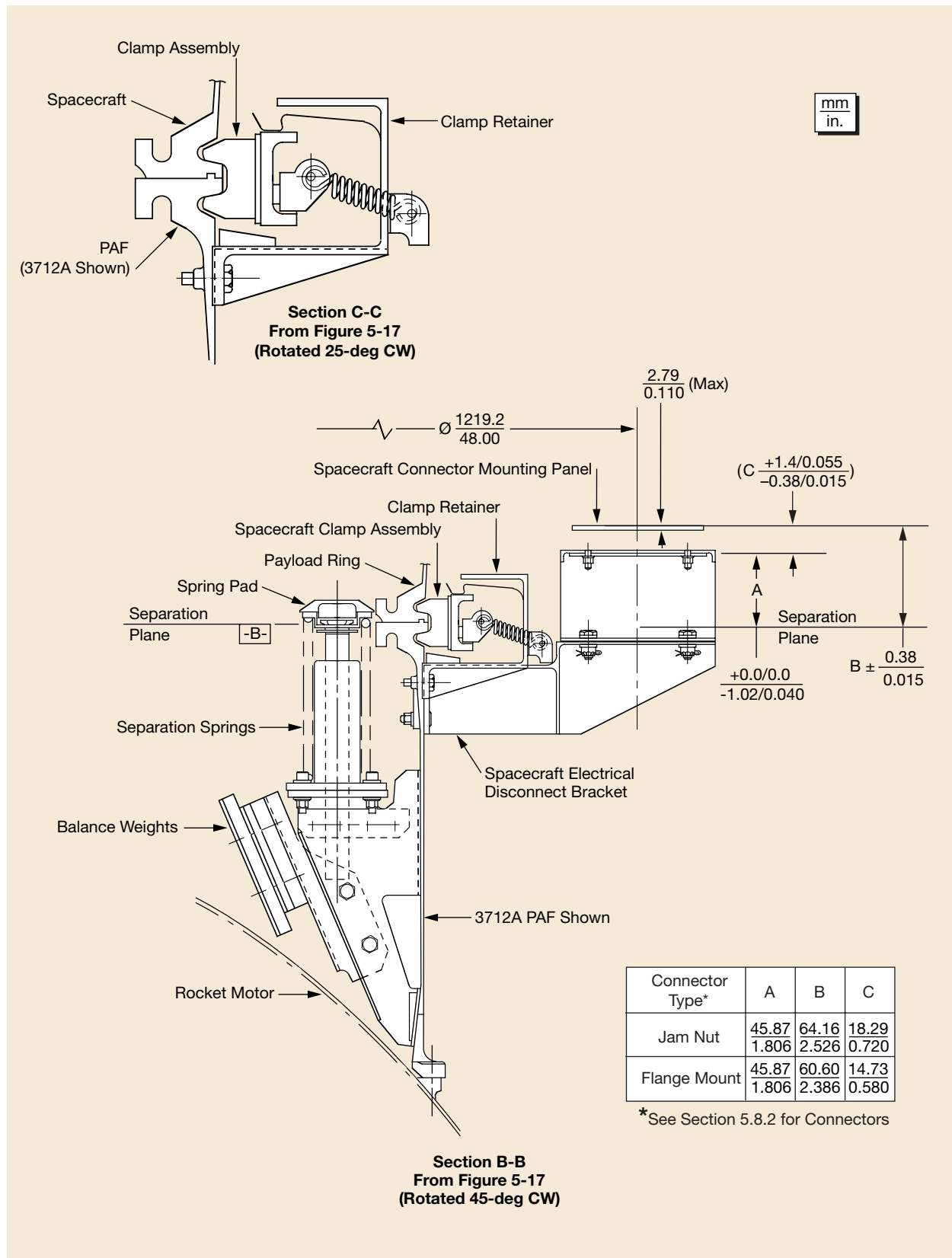


Figure 5-18. 3712A Clamp Assembly and Spring Actuator

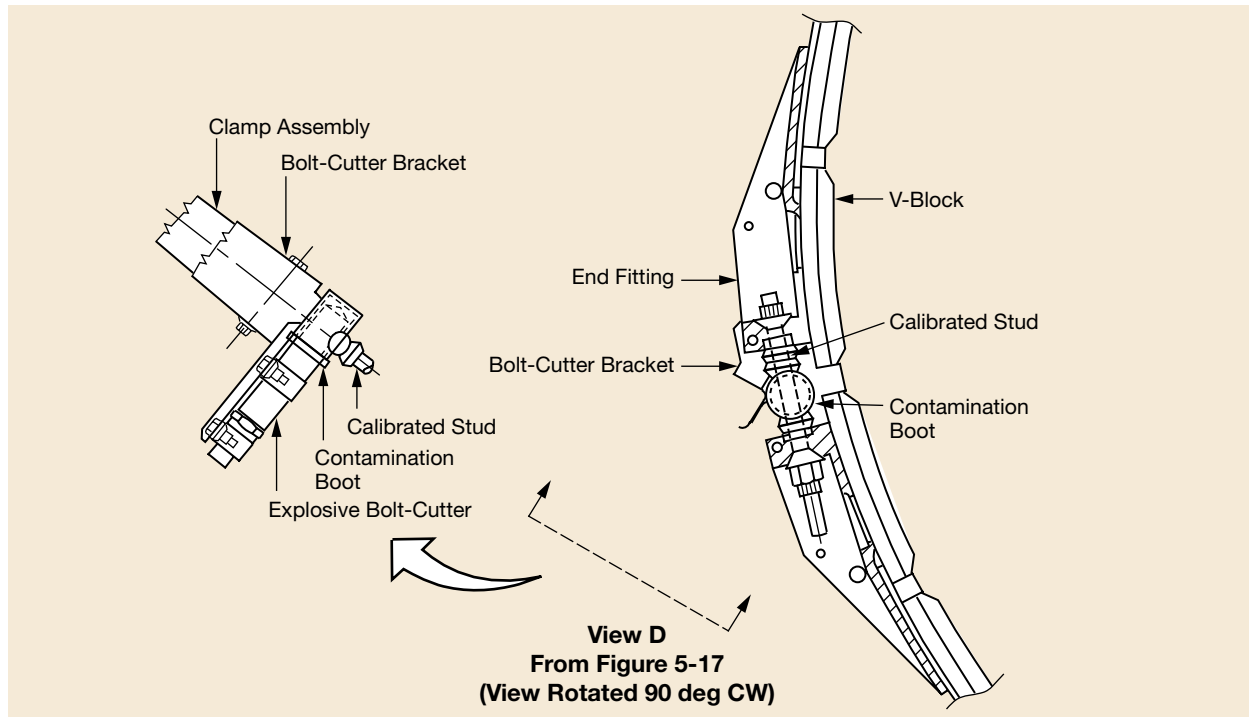


Figure 5-19. 3712 PAF Bolt-Cutter Detailed Assembly

5.3 PAYLOAD ATTACH FITTINGS FOR TWO-STAGE MISSIONS

Delta offers several PAF configurations for use on two-stage missions. The PAF for two-stage missions has a separation system that is activated by power signal from the second stage, rather than by a self-contained component, as on the three-stage PAF.

On two-stage configurations, the spacecraft is separated by the activation of separation nuts (for the 6019 and 6915 PAFs) or by the release of a V-band clamp (for the 6306, 5624, 3715, and 4717 PAFs) followed by the action of four separation spring actuators or the second-stage helium-gas retro system. A secondary latch system comes standard with the 6019, 6306, 6915, and 4717 PAFs. The secondary latch system, employed to minimize spacecraft tip-off rates, retains the spacecraft and second stage for a 30-sec period between activation of the separation nuts (or release of the V-band clamp) and activation of the helium-gas retro.

5.3.1 The 6019 PAF Assembly

The one-piece machined-aluminum 6019 PAF assembly (Figure 5-20) is approximately 483 mm (19 in.) high and 1524 mm (60 in.) in diameter. This fitting was designed specifically to interface with the NASA Multimission Modular Spacecraft (MMS); hence, customers should consult with the Delta Program Office to ensure that the interface stiffness is adequate.

HB01259REU.0



Figure 5-20. 6019 PAF Assembly

The PAF base is attached to the forward ring of the second stage. The spacecraft is fastened to the 1524-mm (60-in.)-dia bolt-circle at three equally spaced hard points using 15.9-mm (0.625-in.)-dia bolts that are preloaded to 53,378 N (12,000 lb). Figure 5-21 shows the capability of the 6019 PAF in terms of spacecraft weight and CG location above the separation plane. The capability for a specific payload with its own unique mass, size, flexibility, etc. might vary from that presented; therefore, as the spacecraft configuration is finalized, the Delta Program will initiate a coupled-loads analysis to verify that the structural capability of the launch vehicle is not exceeded. The spacecraft interface is shown in Figures 5-22 and 5-23. Matched tooling for the spacecraft-to-PAF interface is provided upon request.

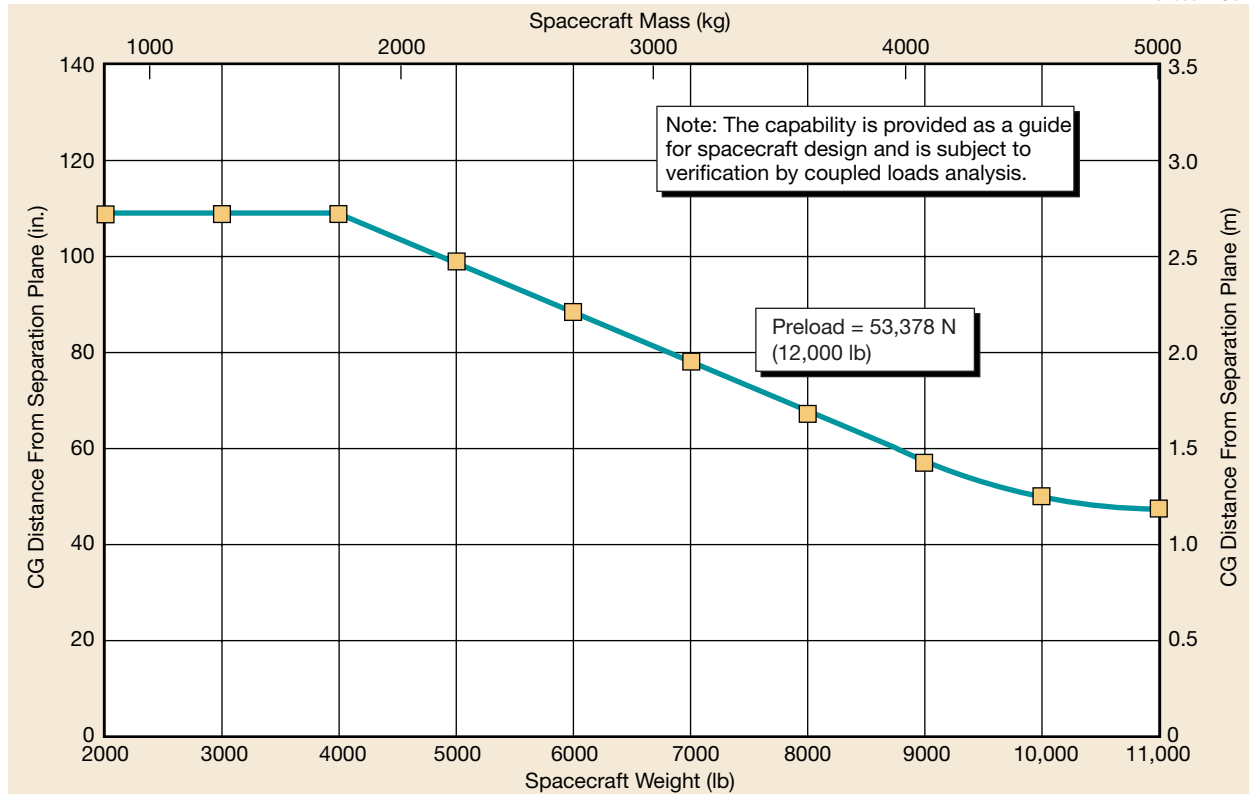


Figure 5-21. Capability of the 6019 PAF

Separation of the spacecraft from the launch vehicle begins when the separation nuts are activated. The secondary latch system then loosely holds the spacecraft to the second stage for a period of 30 sec. During this period, the spacecraft and second stage undergo many damped cycles of small amplitude rattling back and forth, reducing the angular rates to small values in comparison to that which would exist without the secondary latch system. At the end of the 30-sec rate-damping period, the secondary latches are released and the second stage is backed away from the spacecraft by activating the helium retro system. The second stage then performs a contamination and collision avoidance maneuver (CCAM) to remove the second stage from the vicinity of the spacecraft. Note that Boeing requires access on the spacecraft side of the separation plane for installation of the separation bolts, bolt-catcher assemblies, and latch clip brackets, which are retained on the spacecraft after separation. The secondary latch system also requires a small latch clip bracket provided by the Delta Program to be installed on the spacecraft at each separation bolt location (Figures 5-23, 5-24, and 5-25).

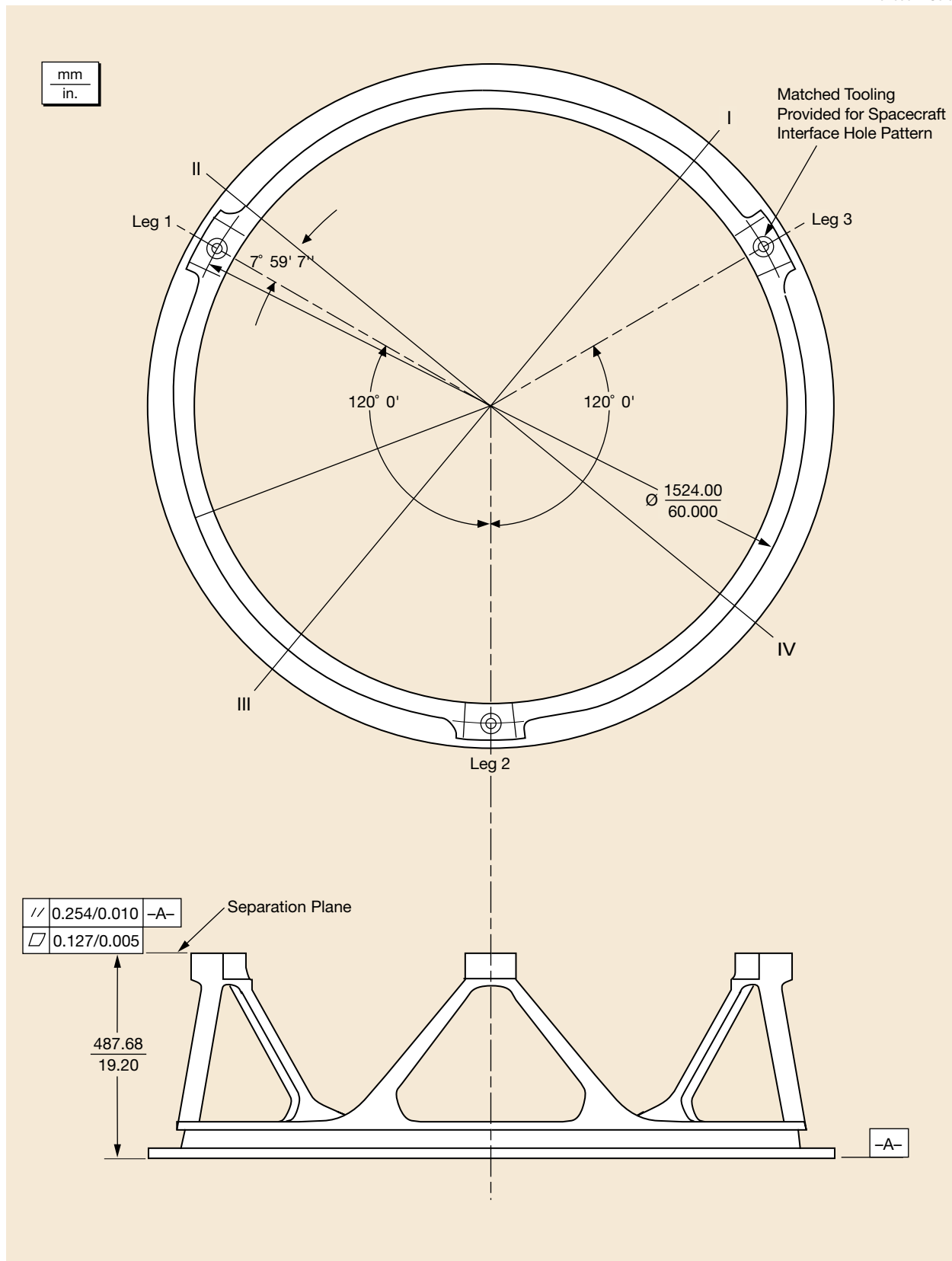


Figure 5-22. 6019 PAF Detailed Assembly

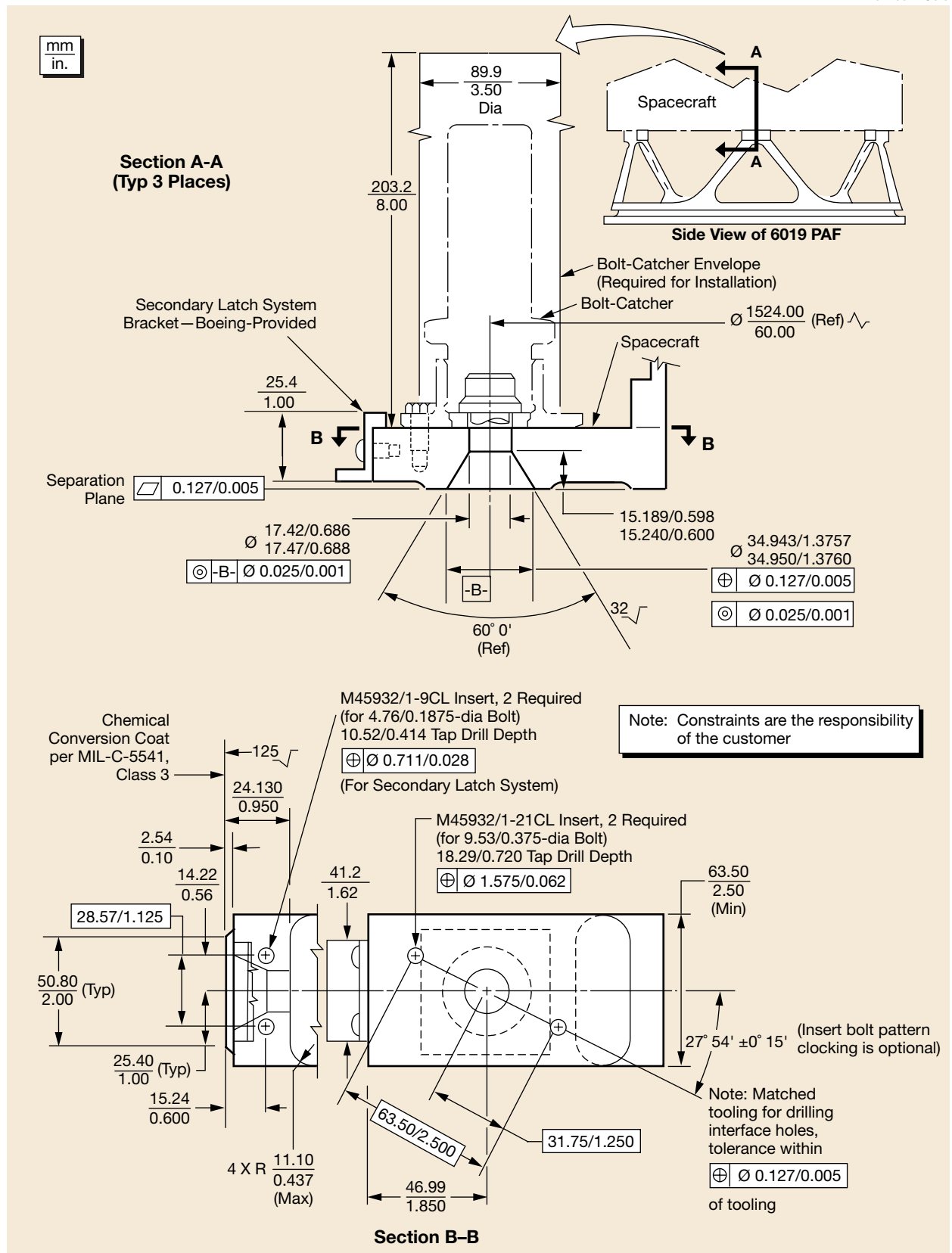


Figure 5-23. Dimensional Constraints on Spacecraft Interface to 6019 PAF

HB01151REU0.5

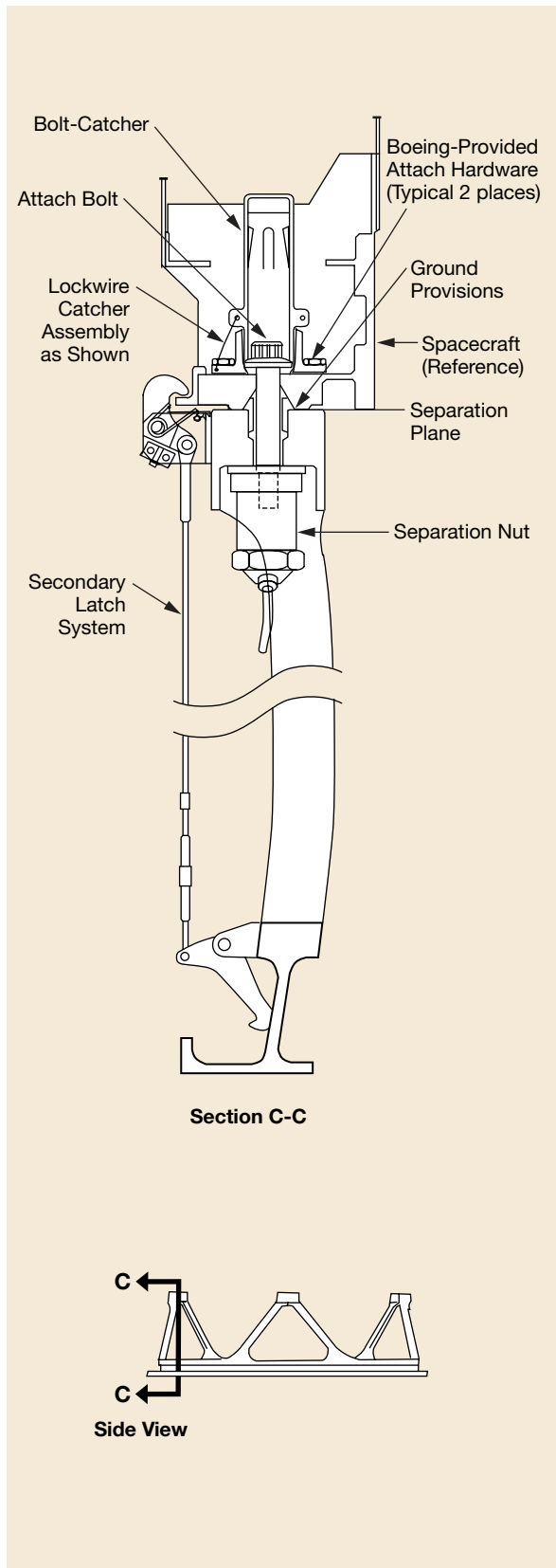


Figure 5-24. 6019 PAF Spacecraft Assembly

HB01152REU0.6

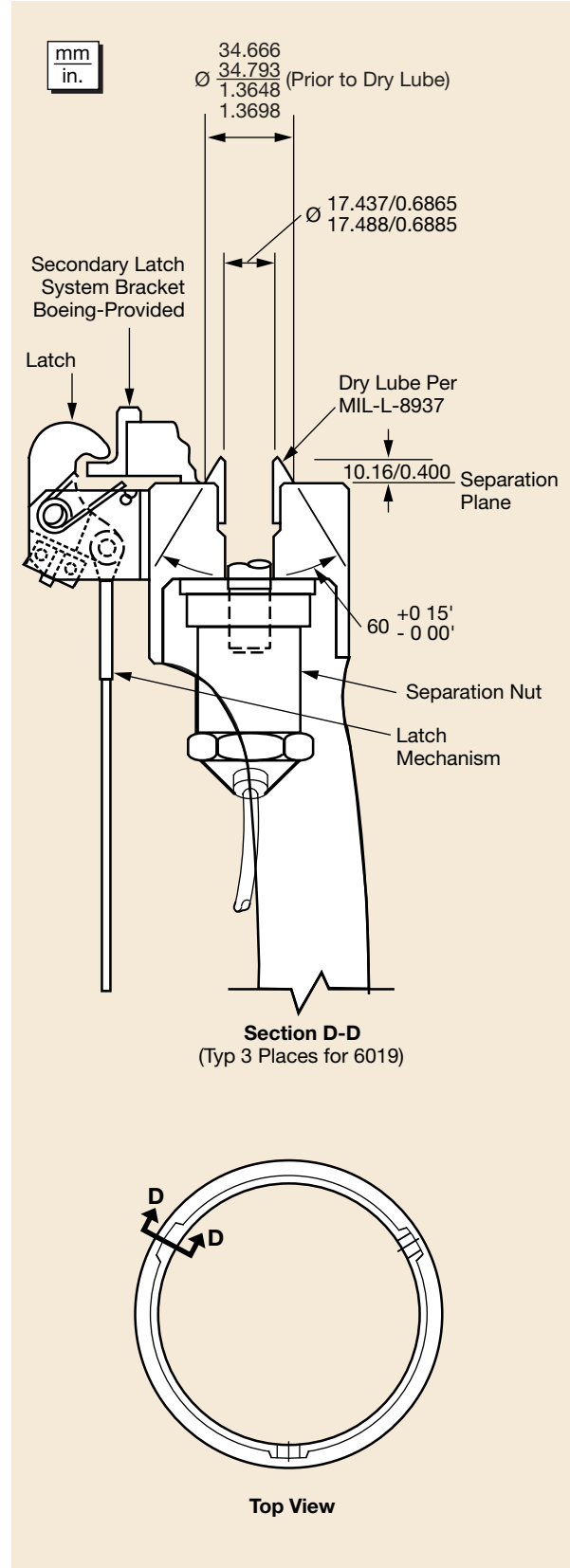


Figure 5-25. 6019 PAF Detailed Dimensions

5.3.2 The 6915 PAF Assembly

The one-piece machined-aluminum 6915 PAF assembly (Figure 5-26) is approximately 381 mm (15 in.) high and 1743 mm (68.6 in.) in diameter.

The PAF base is attached to the forward ring of the second stage. The spacecraft is fastened to the 1742.6-mm (68.6-in.)-dia PAF at four equally spaced hard-points using 15.9-mm (0.625 in.)-dia bolts that are preloaded to 53,378 N (12,000 lb). Figure 5-27 shows the capability of the PAF in terms of spacecraft weight and CG location above the separation plane. The capability for a specific spacecraft (with its own unique mass, size, flexibility, etc.) might vary from that presented; therefore, as the spacecraft configuration is finalized, Boeing will initiate a coupled-loads analysis to verify that the structural capability of the launch vehicle is not exceeded. The spacecraft interface is shown in Figures 5-28 through 5-32. Matched tooling for spacecraft interface to PAF is provided upon request.

Separation of the spacecraft from the launch vehicle occurs when the explosive nuts are activated, allowing the four guided separation spring actuators to push the second stage away from the spacecraft. The second stage then performs a CCAM to ensure a safe distance to the spacecraft.

For missions where a low tip-off rate is required, the four spring actuators are removed and replaced with a secondary latch system. A small latch clip bracket, required for the latch system and provided by the Delta Program, is installed on the spacecraft at each separation bolt location, as shown in Figures 5-30, 5-31, and 5-32. Following activation of the separation nuts, the secondary latch system loosely holds the spacecraft to the second stage for a period of 30 sec. During this period, the spacecraft and second stage undergo many damped cycles of small amplitude rattling back and forth, reducing the angular rates to small values in comparison to that which would exist without the secondary latch system. At the end of the 30-sec rate-damping period, the secondary latches are released and the second stage is backed away from the spacecraft by activating the helium retro system. Then a CCAM is performed to remove the second stage from the vicinity of the spacecraft. Note that the Delta Program requires access on the spacecraft side of the separation plane for installation of the separation bolts, bolt-catcher assemblies, and latch clip brackets which are retained on the spacecraft after separation.

HB01250REU0.1



Figure 5-26. 6915 PAF

HB01059REU0.2

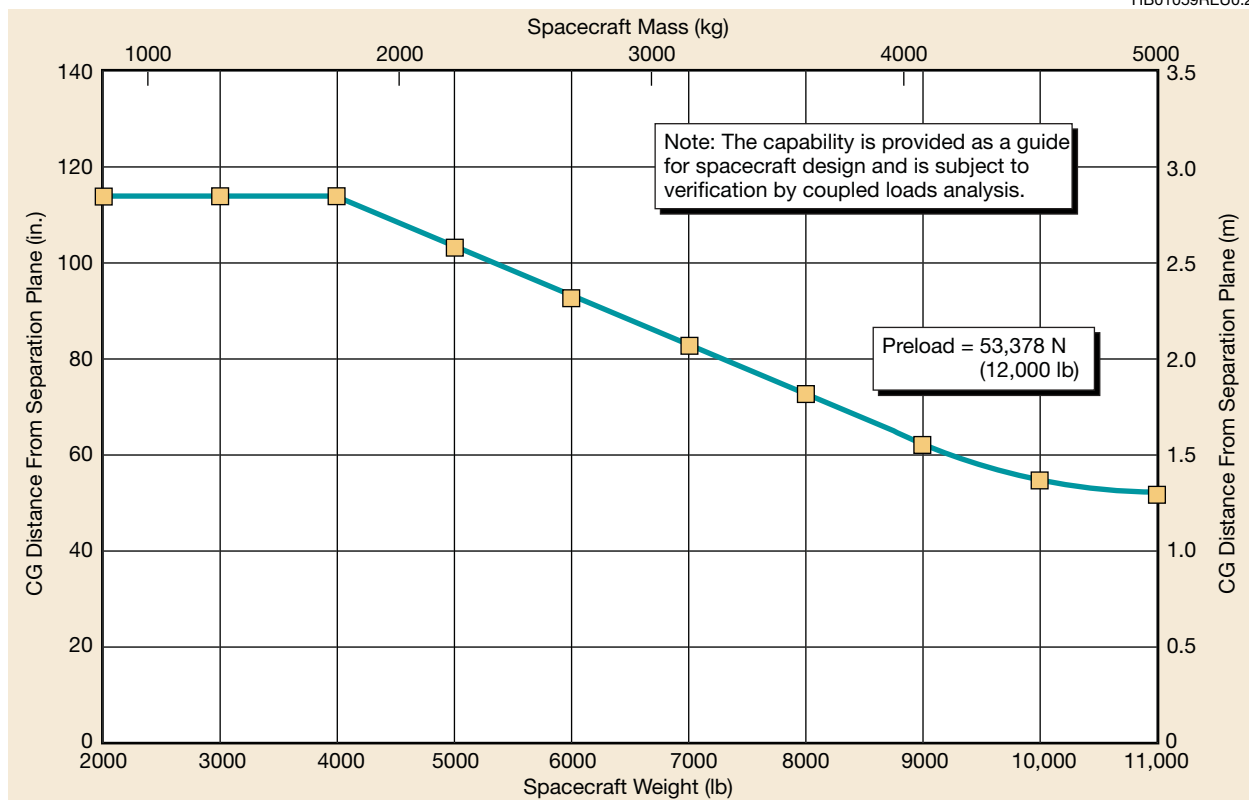


Figure 5-27. Capability of the 6915 PAF

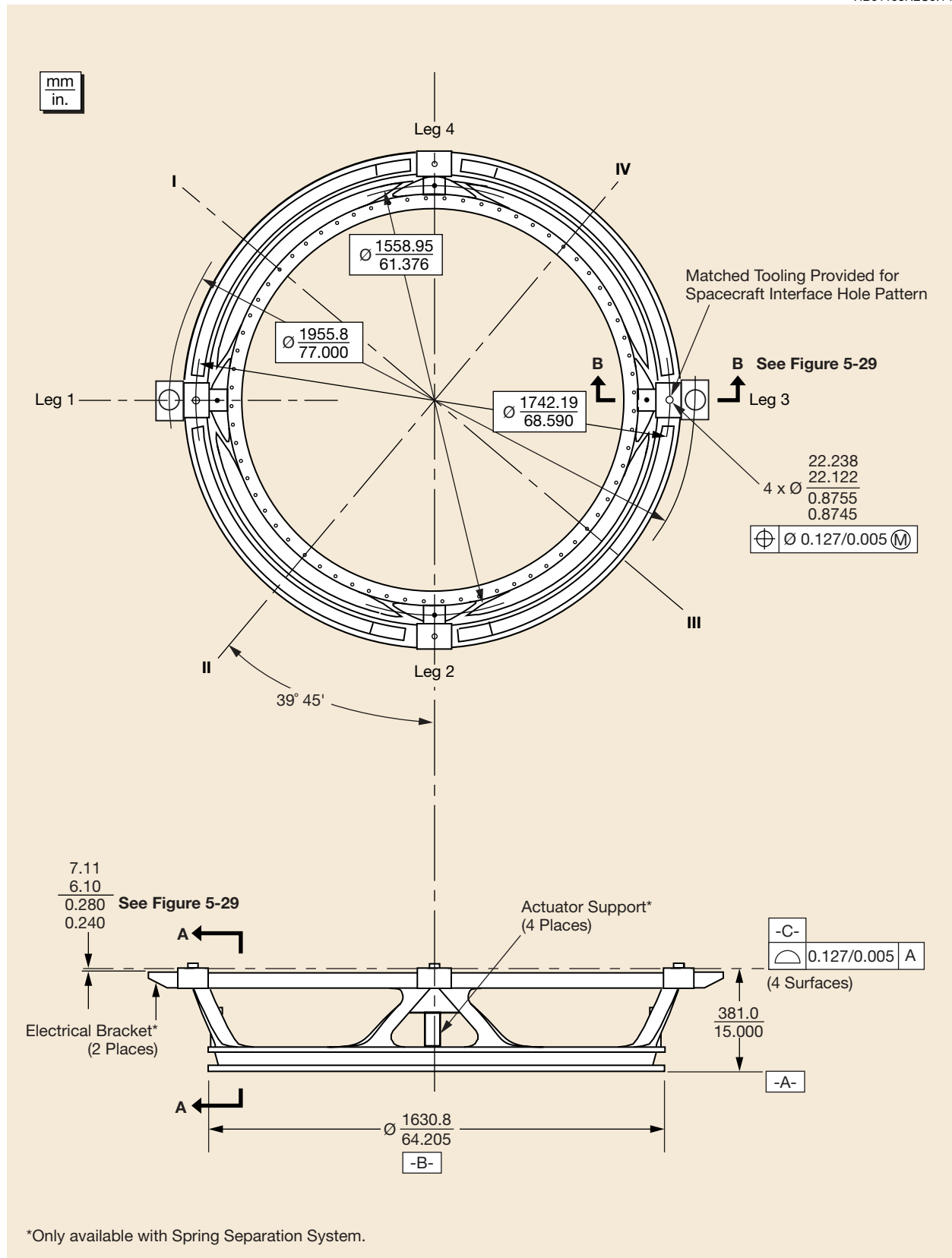


Figure 5-28. 6915 PAF Detailed Assembly

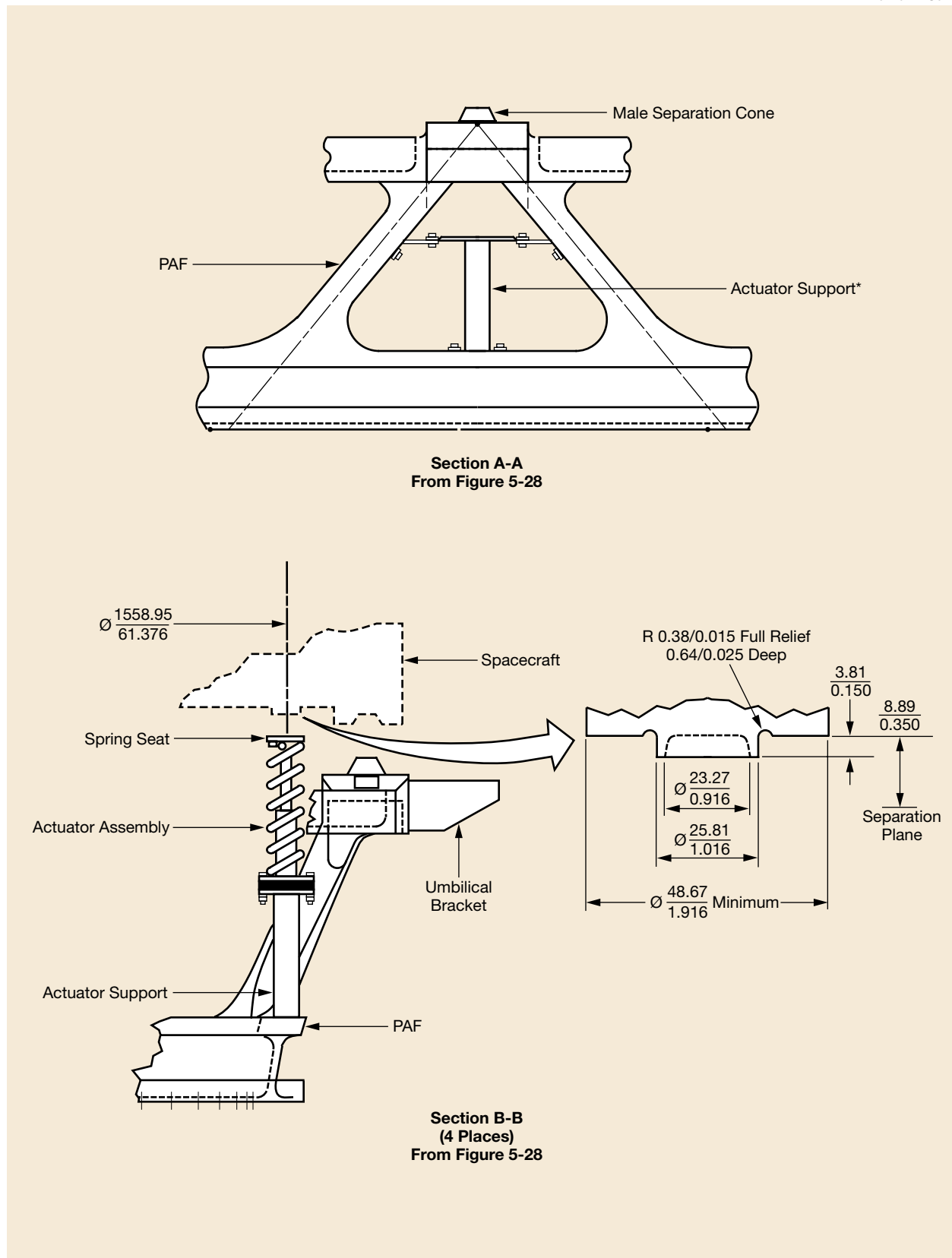


Figure 5-29. Actuator Assembly Installation—6915 PAF

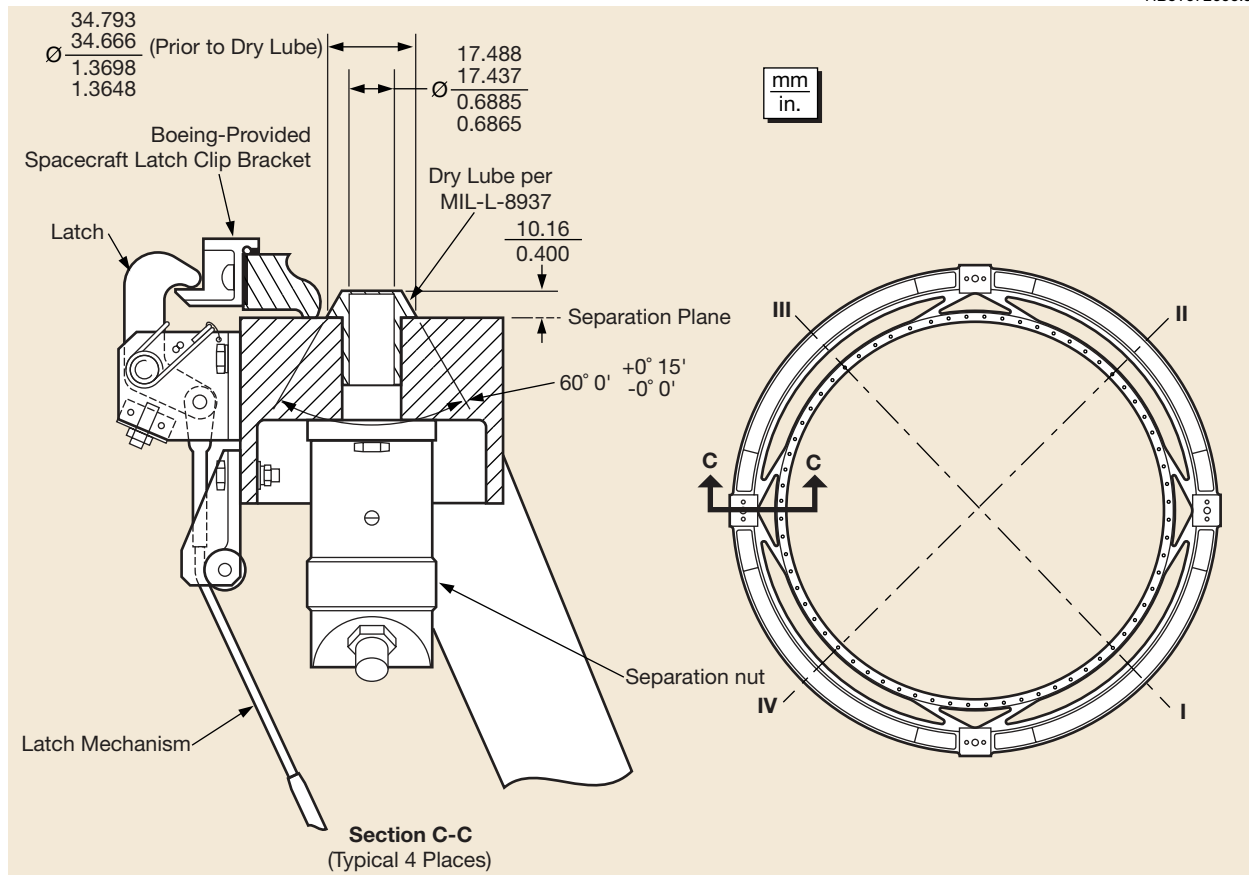


Figure 5-30. 6915 PAF Detailed Dimensions

HB01155REU0.5

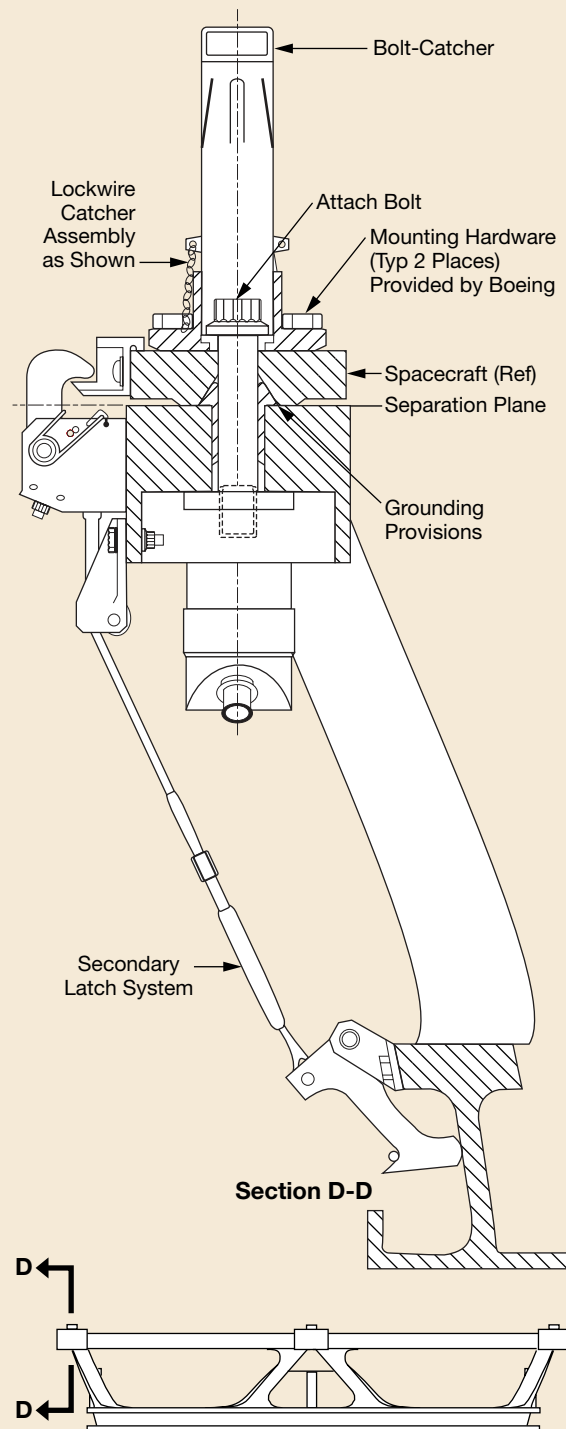
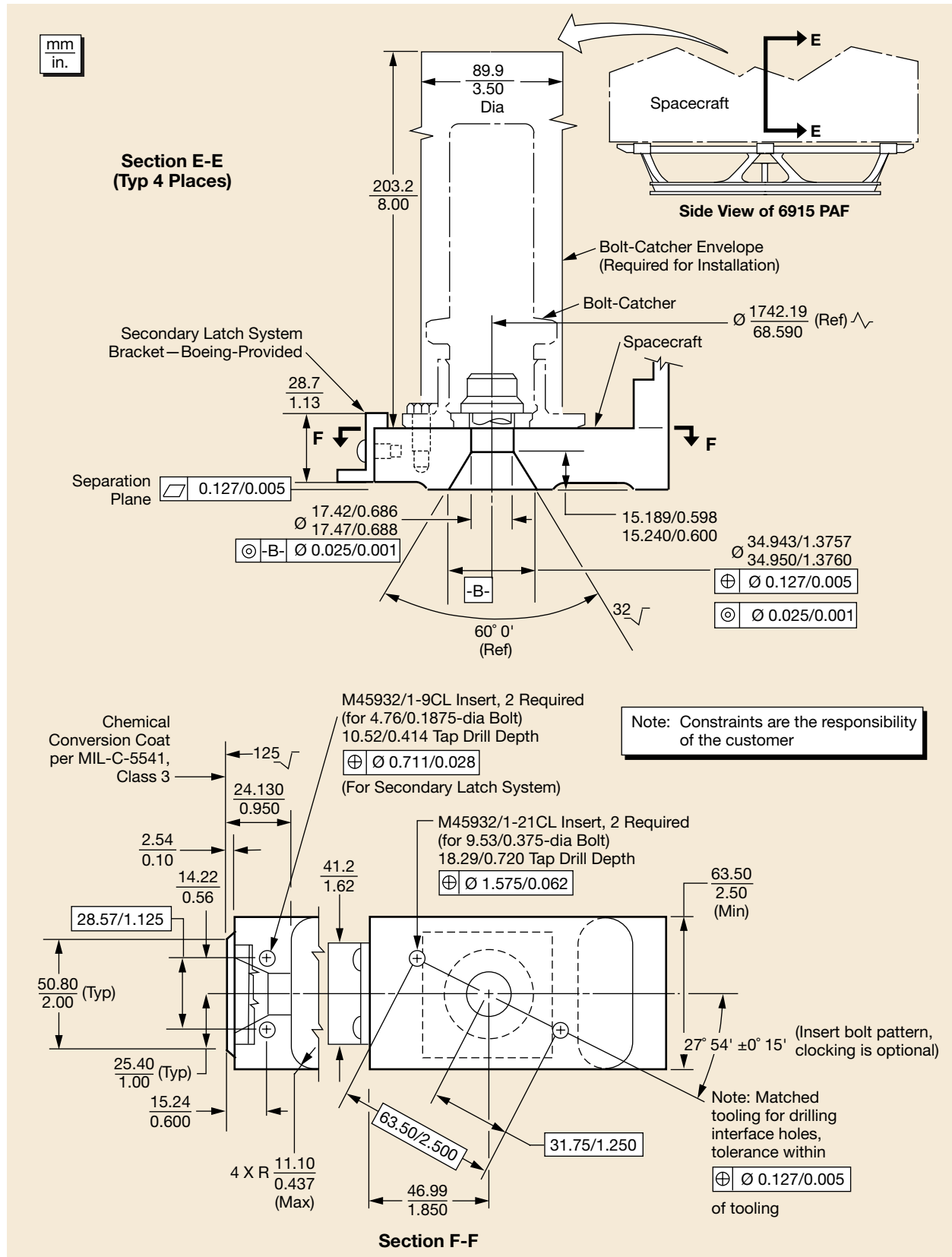


Figure 5-31. 6915 PAF Spacecraft Assembly



5.3.3 The 6306 PAF Assembly

The one-piece machined-aluminum 6306 PAF assembly (Figure 5-33) is approximately 152.4 mm (6 in.) high and 1600 mm (63 in.) in diameter.

The PAF base is attached to the forward ring of the second stage. The spacecraft is fastened to the 1600-mm (63-in.) PAF mating diameter with a V-band clamp assembly that is preloaded to 34,250 N (7,700 lb). Figure 5-34 shows the capability of the PAF in terms of spacecraft weight and CG location above the separation plane. The capability for a specific spacecraft (with its own unique mass, size, flexibility, etc.) might vary from that presented; therefore, as the spacecraft configuration is finalized, the Delta Program Office will initiate a coupled-loads analysis to verify that the structural capability of the launch vehicle is not exceeded. The spacecraft interface is shown in Figures 5-35 through 5-40.

Separation of the spacecraft from the launch vehicle begins when the V-band clamp assembly is released. The secondary latch system loosely holds the spacecraft for a period of 30 sec, during which the spacecraft and second stage undergo many damped cycles of small amplitude rattling back and forth, resulting in low angular rates in comparison to that would exist without the secondary latch system. At the end of the damping period, the secondary latches are released and the second stage is backed away from the spacecraft by activating the helium retro system. The second stage then performs a CCAM to remove itself from the vicinity of the spacecraft. Note that the secondary latch system requires the addition of four holes in the spacecraft interface ring (see Figures 5-39 and 5-40) to mate with the PAF-mounted lateral restraints. These holes also serve as the interface for spacecraft-provided separation switches. When the spacecraft does not require separation switches, Delta Program-provided damping devices, which interface directly with the aft side of the spacecraft interface ring, are mounted on the PAF to assist in damping the angular rates.

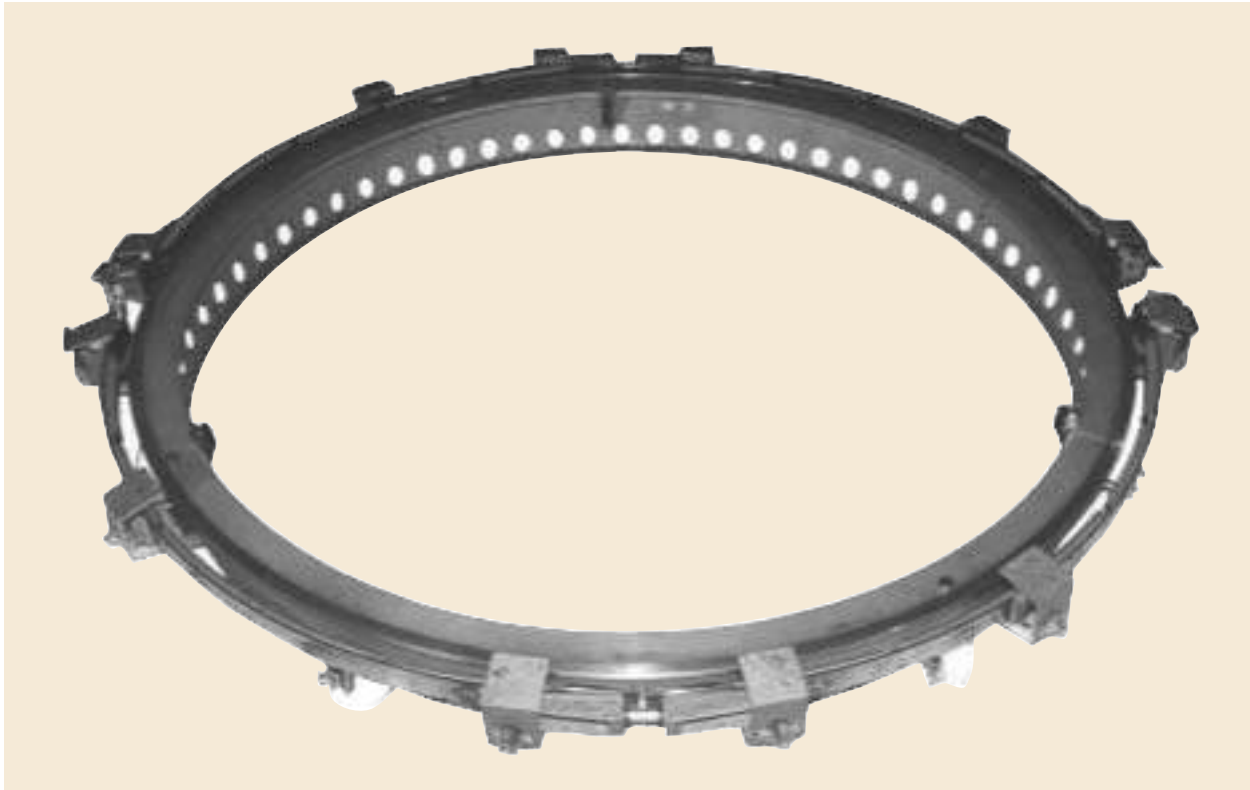


Figure 5-33. 6306 PAF Assembly

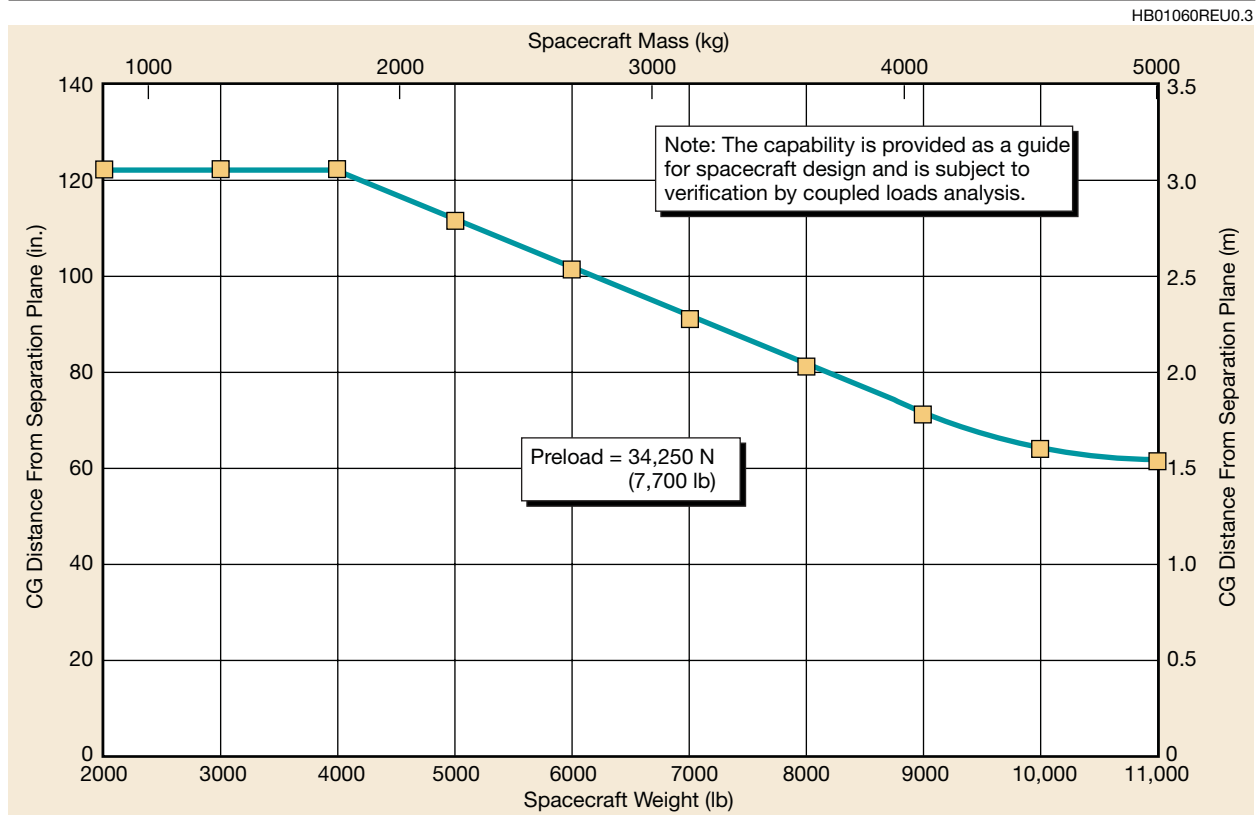


Figure 5-34. Capability of the 6306 PAF

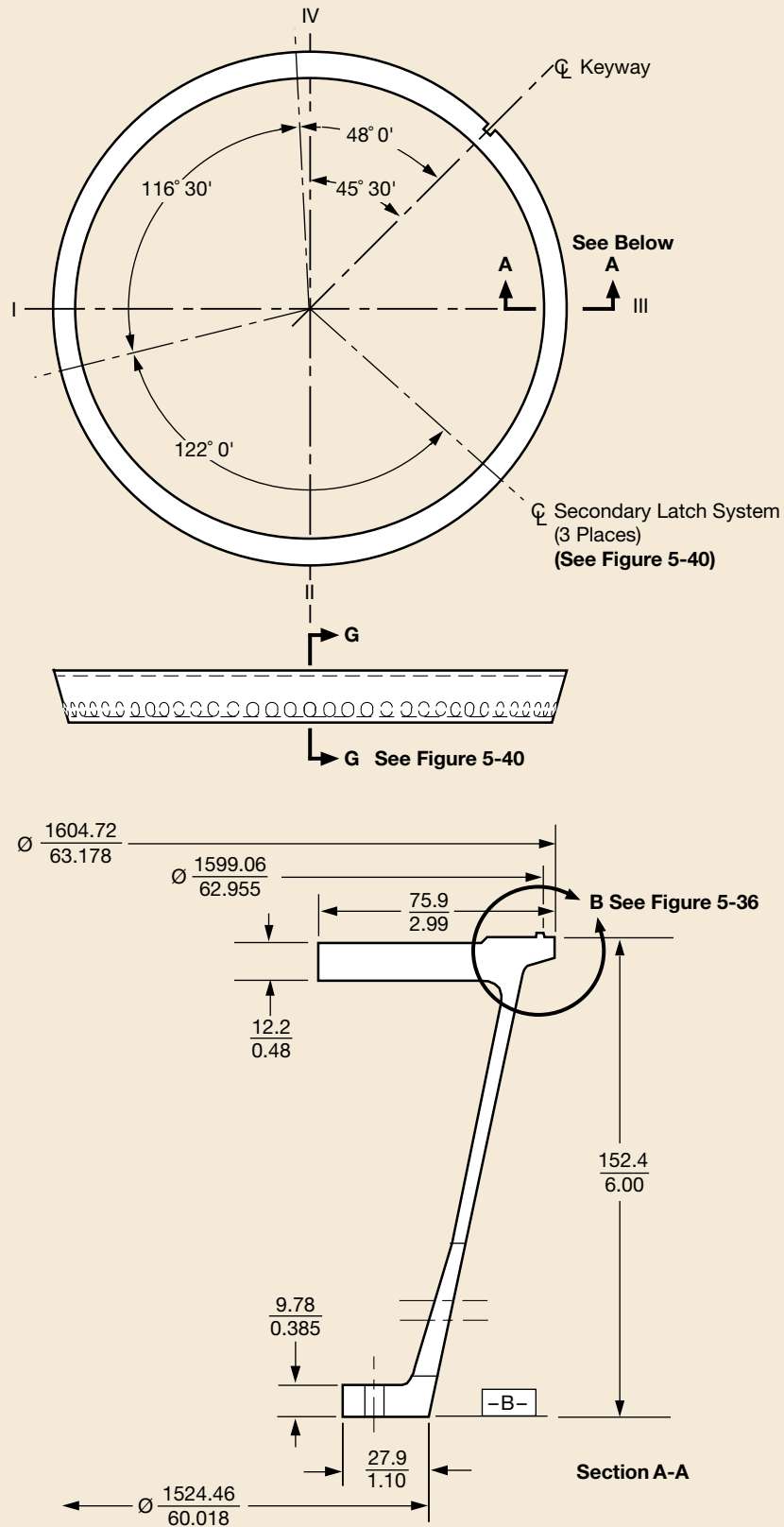


Figure 5-35. 6306 PAF Detailed Dimensions

HB01037REU0.7

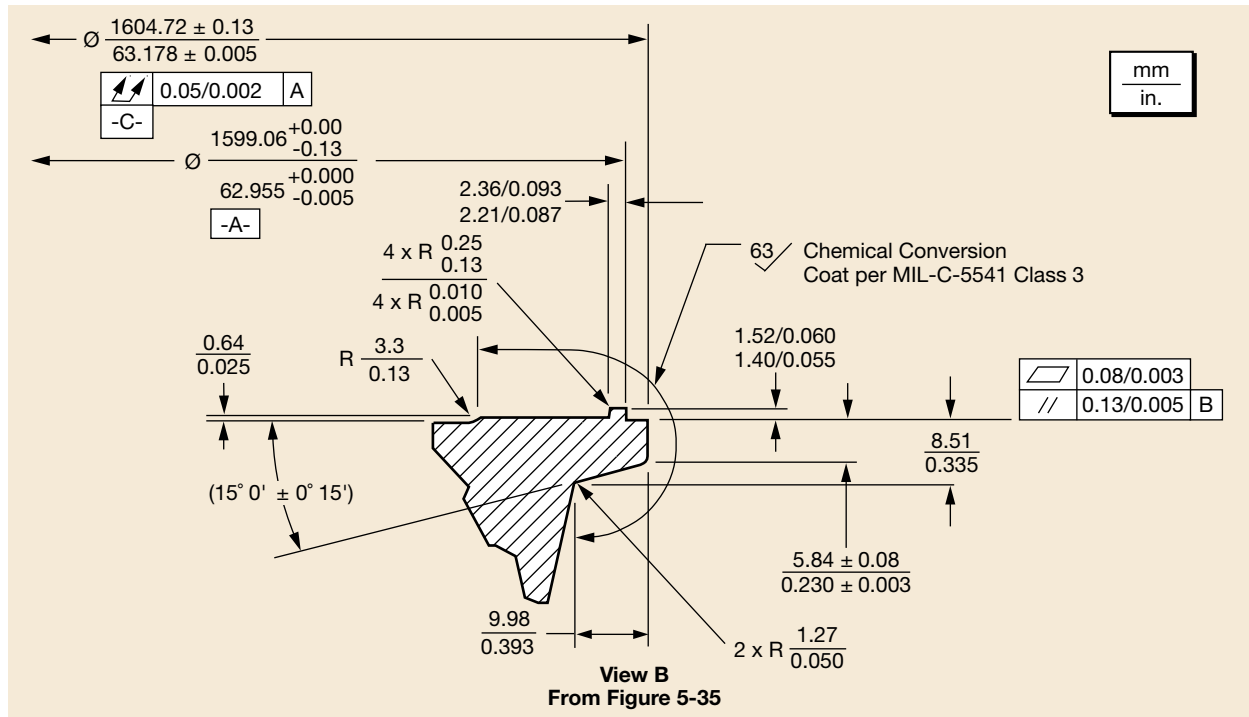


Figure 5-36. 6306 PAF Detailed Dimensions

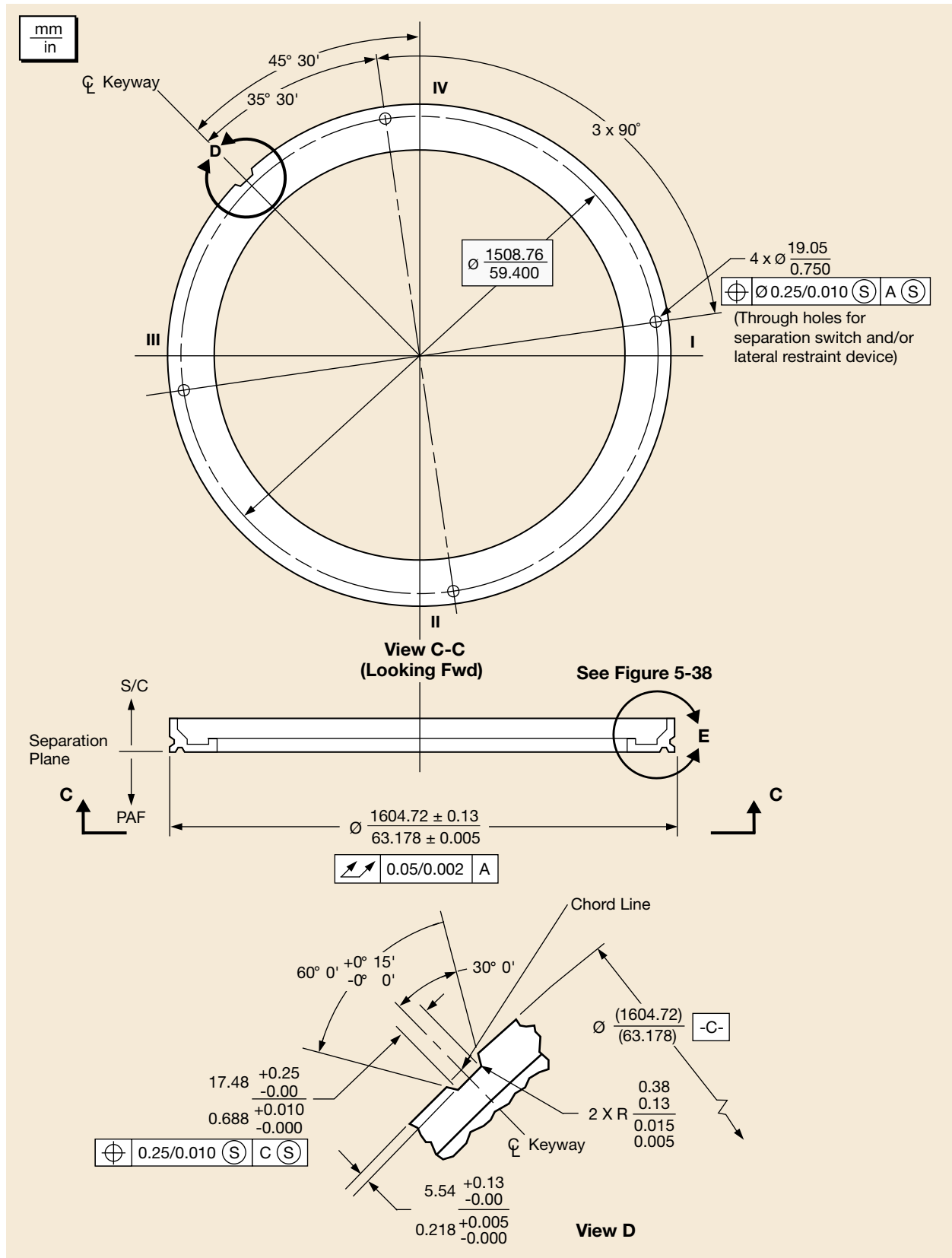


Figure 5-37. Dimensional Constraints on Spacecraft Interface to 6306 PAF

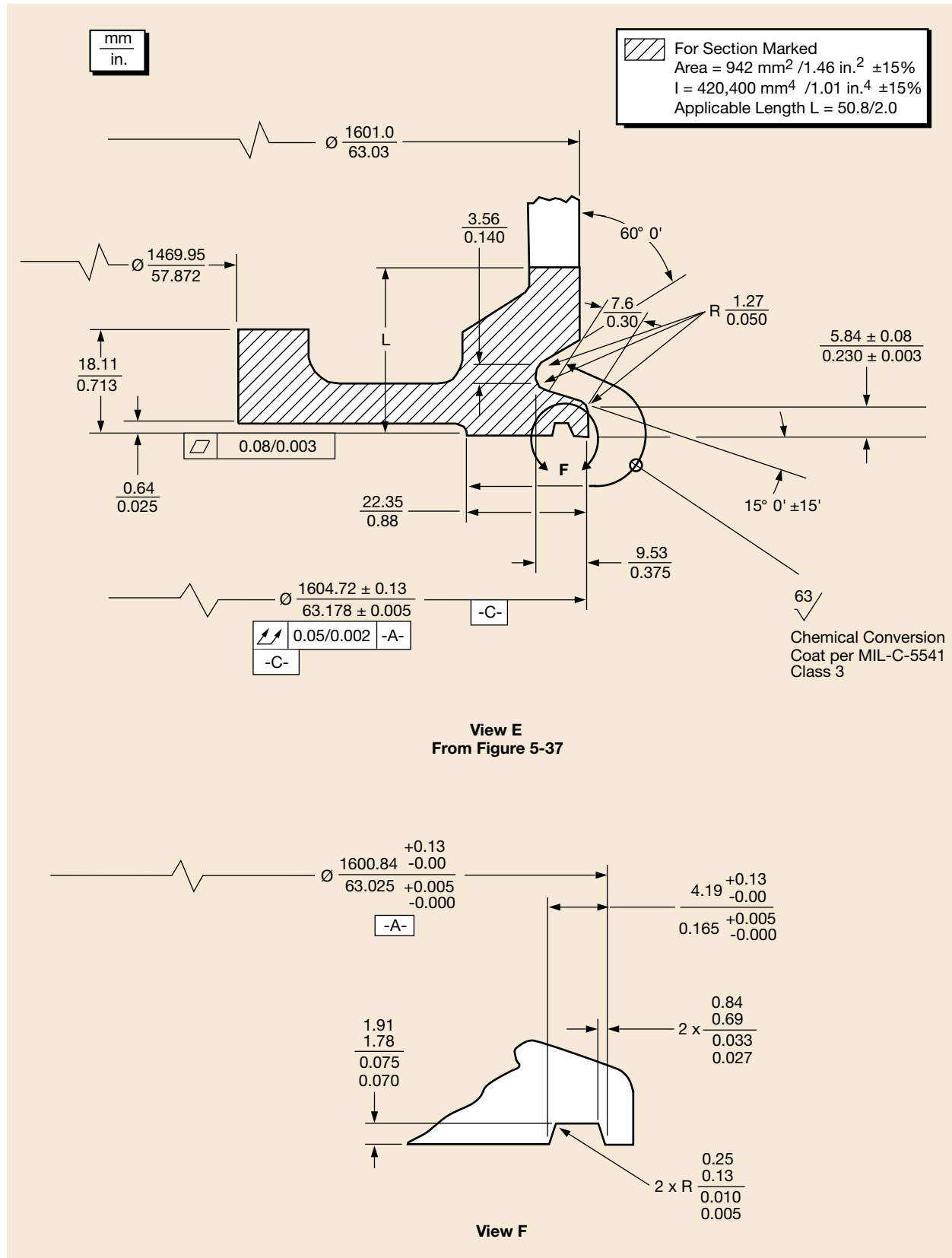


Figure 5-38. Dimensional Constraints on Spacecraft Interface to 6306 PAF

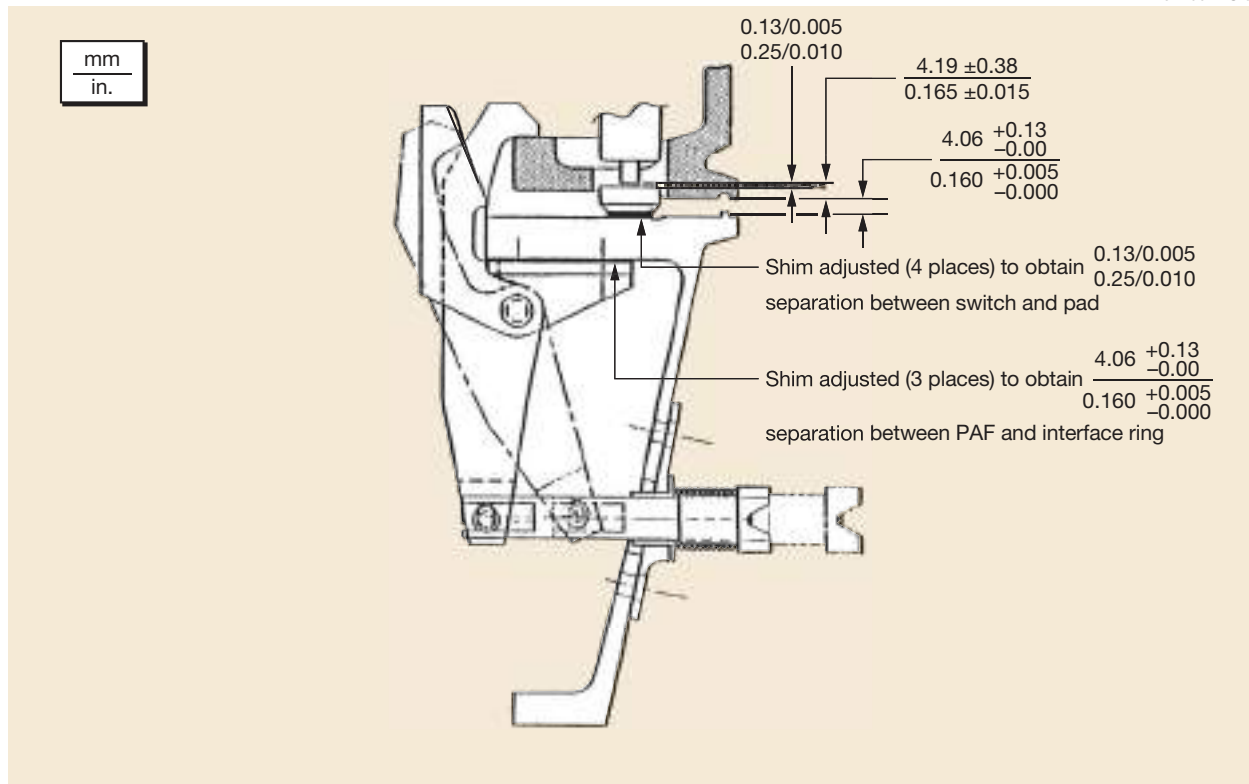


Figure 5-39. 6306 PAF Separation Switch Pad Interface

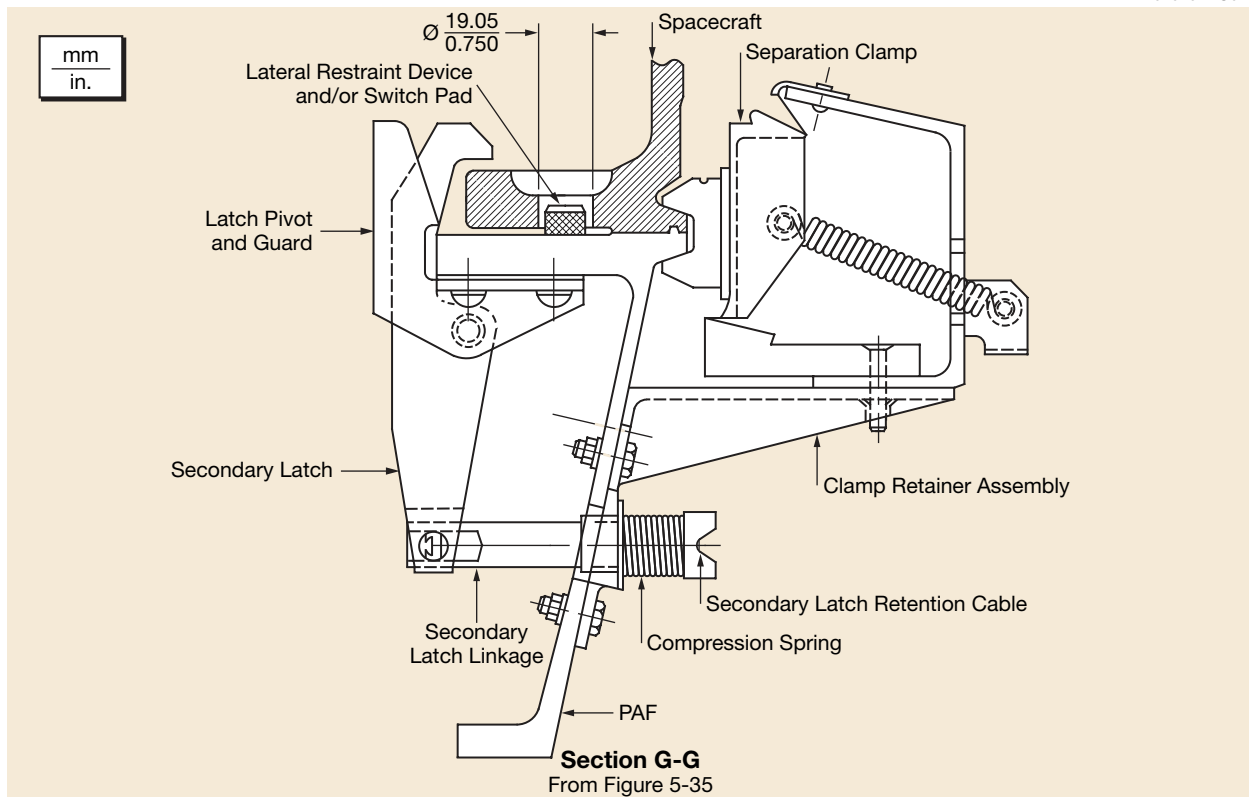


Figure 5-40. 6306 PAF Secondary Latch

5.3.4 The 5624 PAF Assembly

The one-piece machined-aluminum 5624 PAF assembly is approximately 609.6 mm (24 in.) high and 1422.4 mm (56 in.) in diameter.

The PAF base is attached to the forward ring of the second stage. The spacecraft is fastened to the 1422.4-mm (56-in.) PAF mating diameter with a V-band clamp assembly that is preloaded to 17350 N (3900 lb). Figure 5-41 shows the capability of the PAF in terms of spacecraft weight and CG location above the separation plane. The capability for a specific spacecraft (with its own unique mass, size, flexibility, etc.) might vary from that presented; therefore, as the spacecraft configuration is finalized, the Delta Program will initiate a coupled-loads analysis to verify that the structural capability of the launch vehicle is not exceeded. The spacecraft interface is shown in Figures 5-42 through 5-46.

This PAF design does not accommodate a secondary latch separation system. Spacecraft separation occurs when the V-band clamp is released and four spring actuators impart a relative separation velocity between the spacecraft and the second stage.

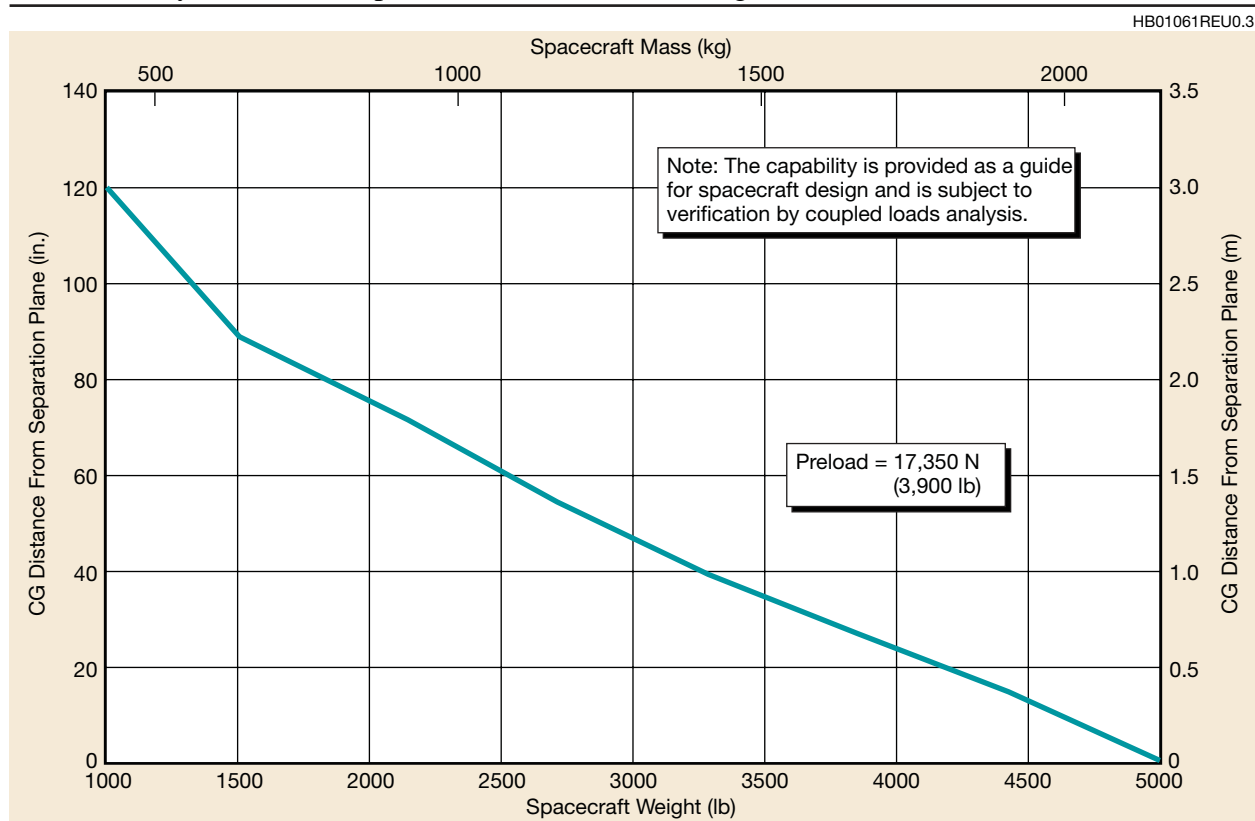


Figure 5-41. Capability of the 5624 PAF

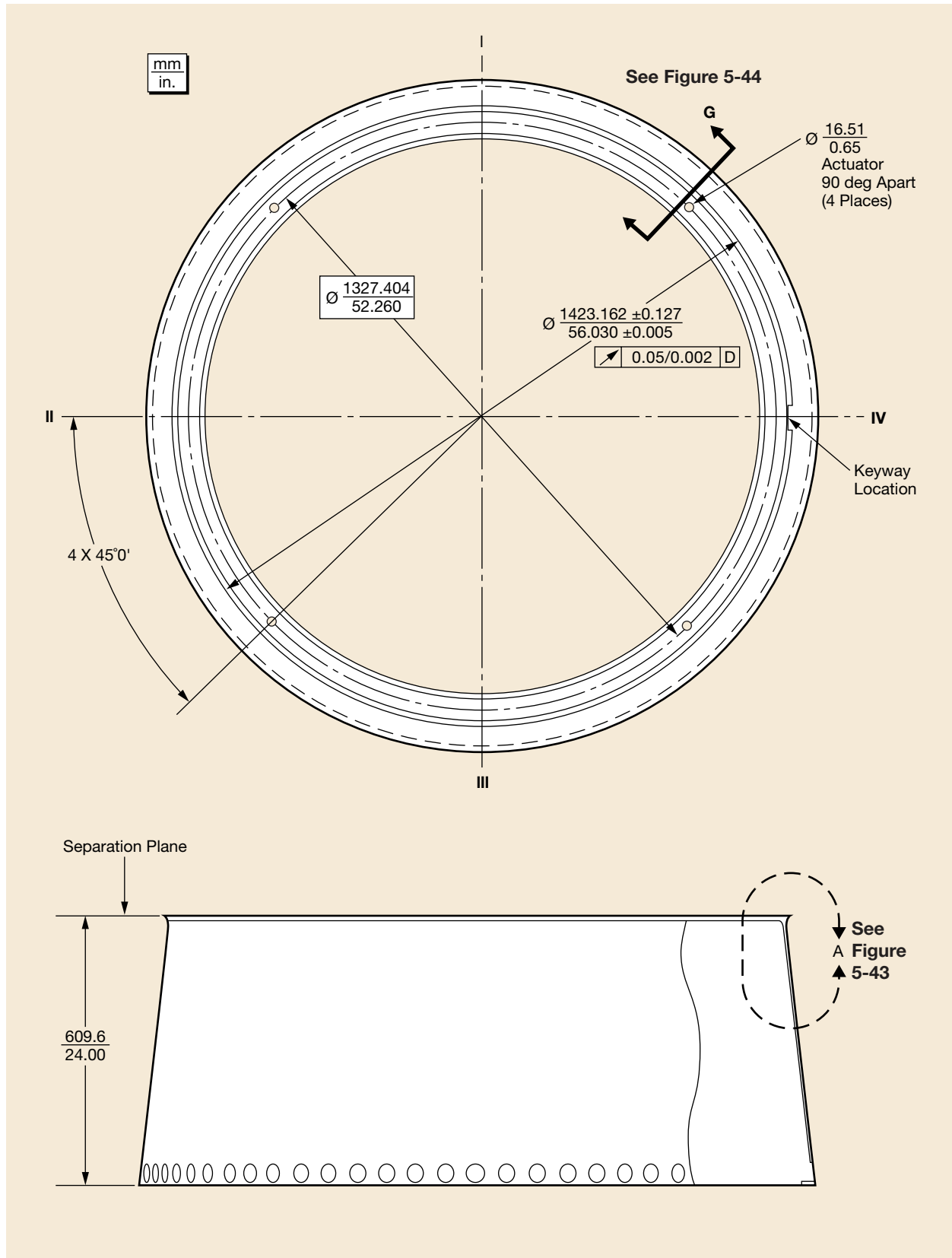


Figure 5-42. 5624 PAF Detailed Assembly

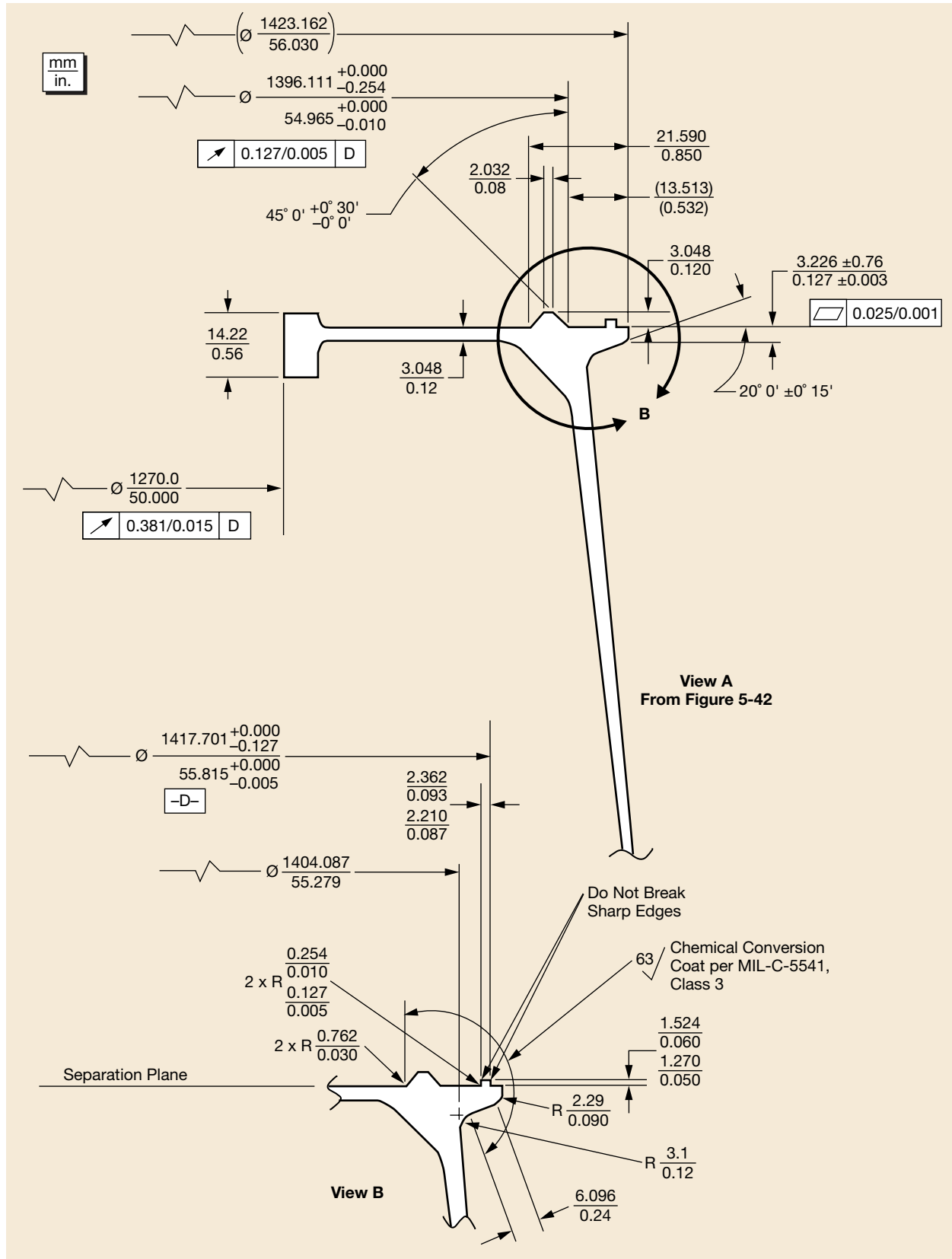


Figure 5-43. 5624 PAF Detailed Dimensions

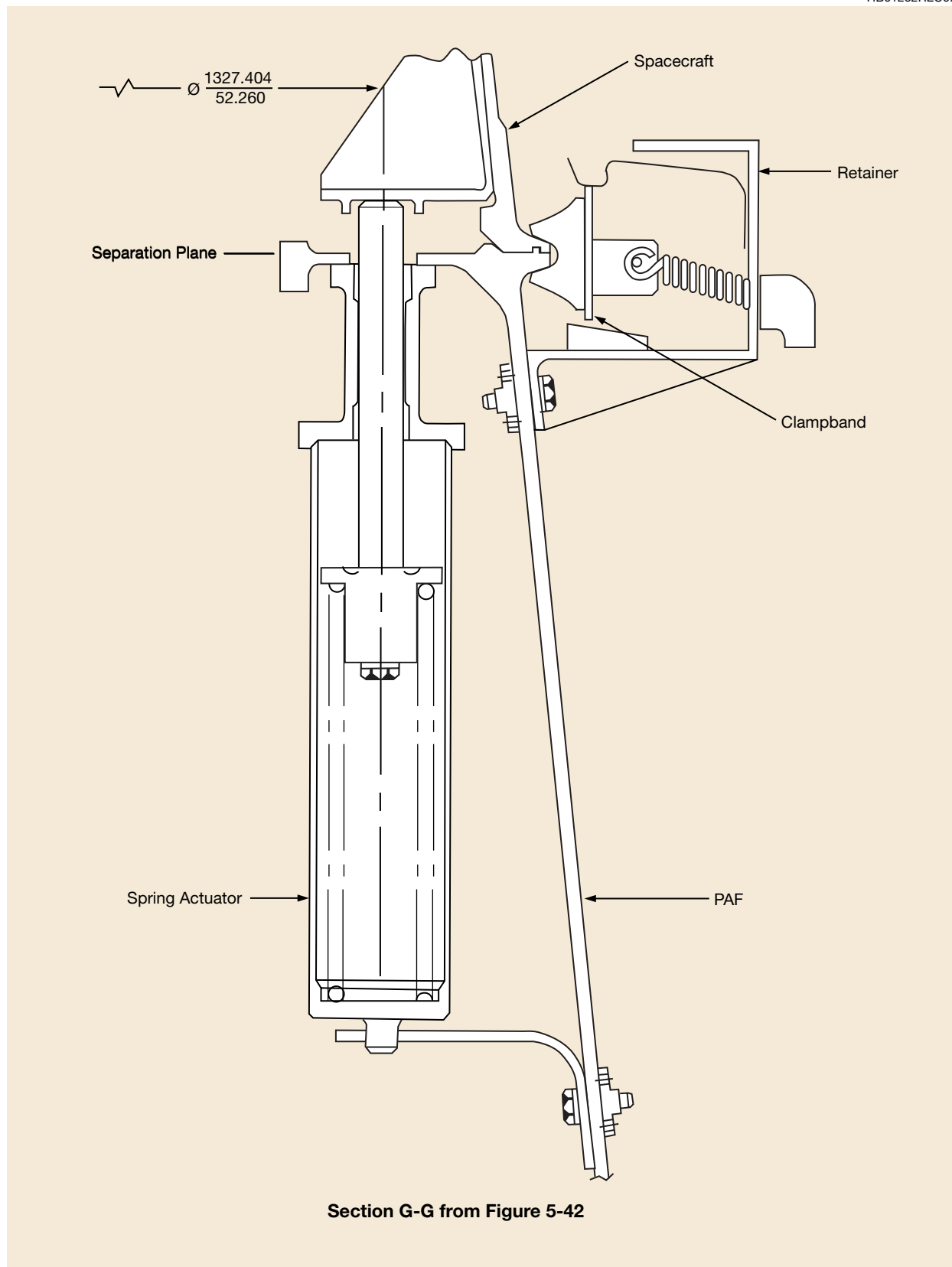


Figure 5-44. 5624 PAF Clamp Assembly and Spring Actuator

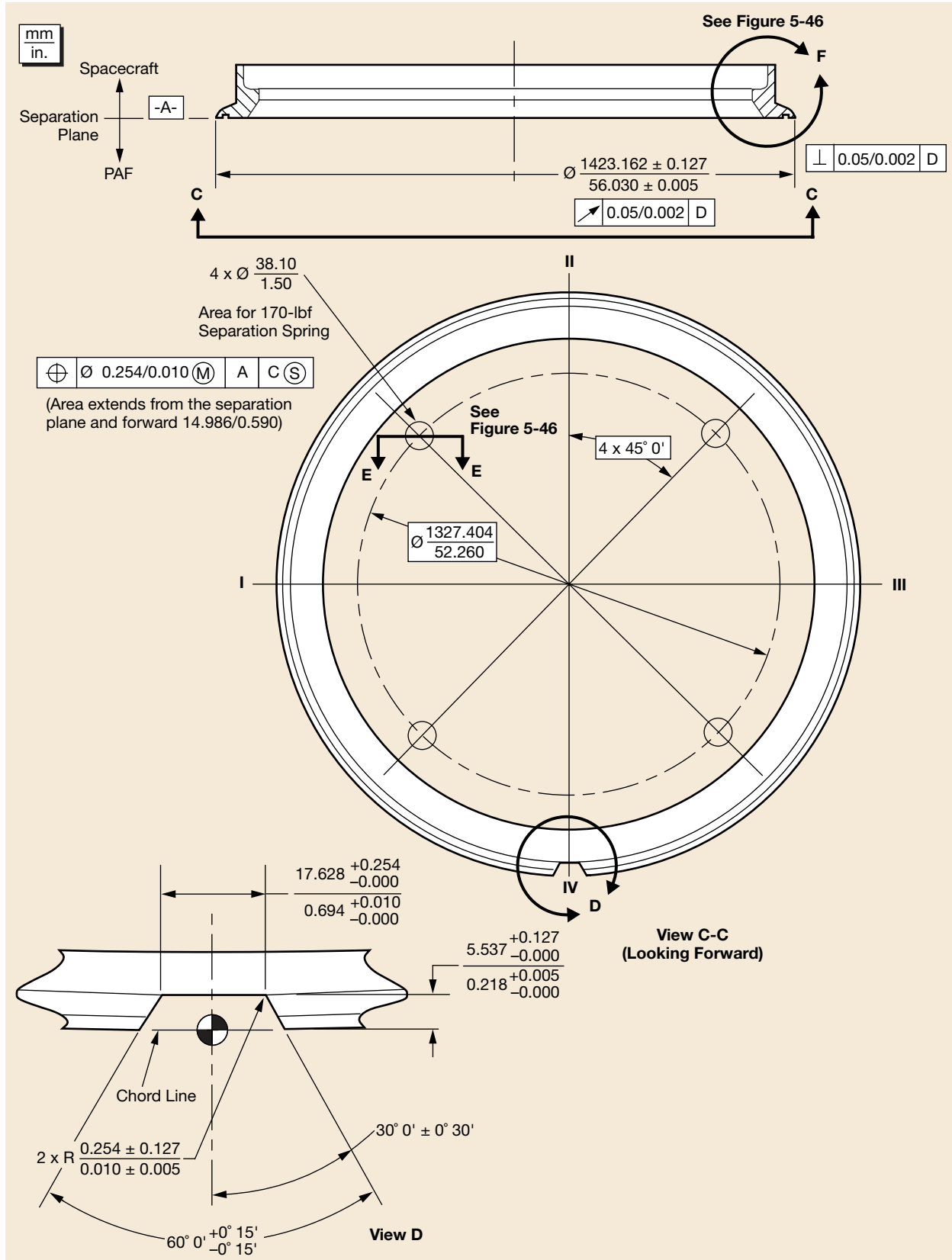


Figure 5-45. Dimensional Constraints on Spacecraft Interface to 5624 PAF

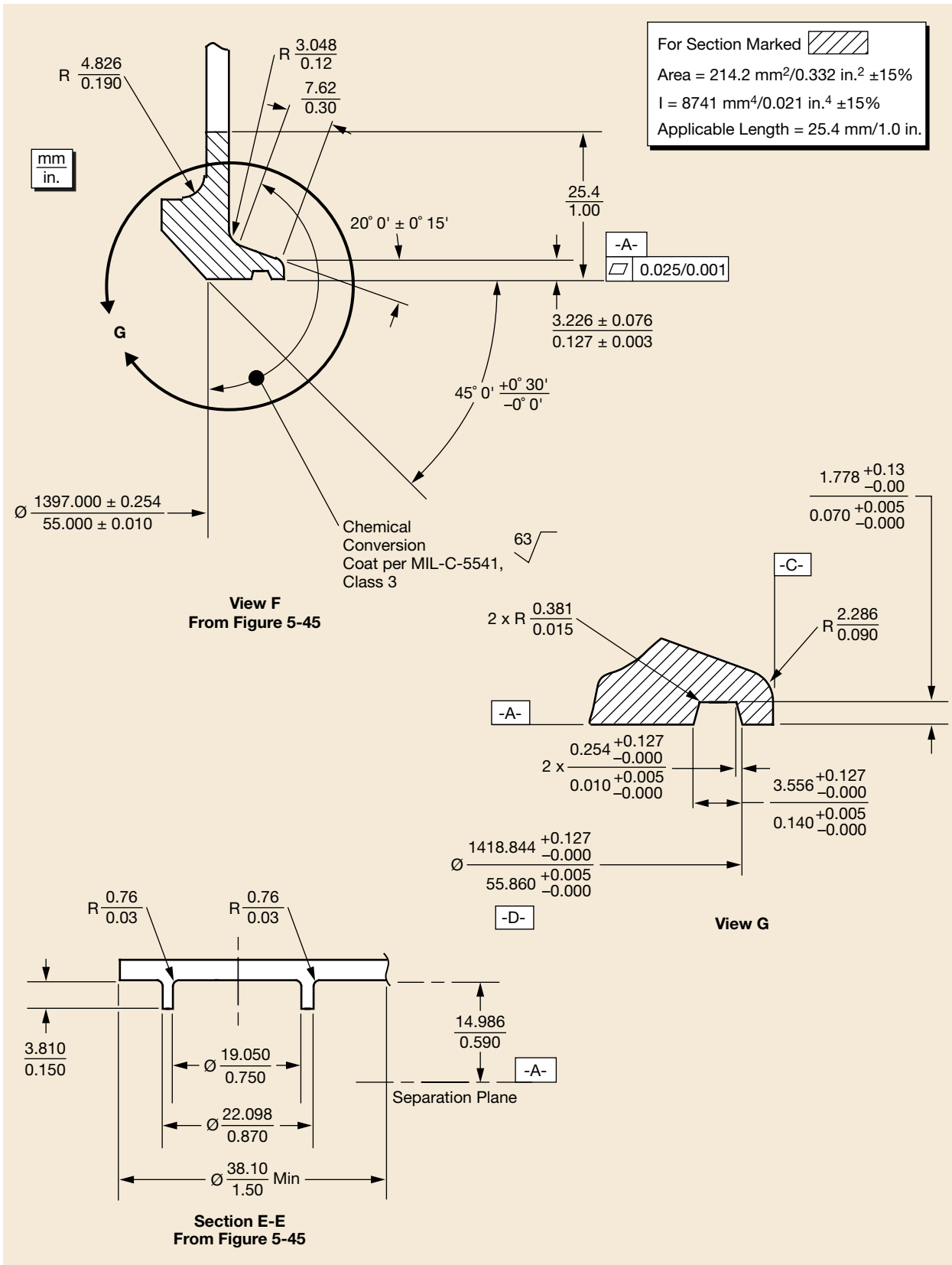


Figure 5-46. Dimensional Constraints on Spacecraft Interface to 5624 PAF

5.3.5 The 4717 PAF Assembly

The two-piece machined aluminum 4717 PAF assembly (Figure 5-47) is approximately 418.55 mm (16.478 in.) high and 1215.01 mm (47.835 in.) in diameter.

The PAF base is attached to the forward ring of the second stage. The spacecraft is fastened to the 1215.01 mm (47.835 in.) PAF mating diameter with a SAAB 1194 V-band clamp assembly that is preloaded to 30,000 N (6,744 lb). Figure 5-48 shows the capability of the PAF in terms of spacecraft weight and CG location above the separation plane. The capability for a specific spacecraft (with its own unique mass, size, flexibility, etc.) may vary from that presented; therefore, as the spacecraft configuration is finalized, the Delta Program will initiate a coupled-loads analysis to verify that the structural capability of the launch vehicle is not exceeded. The spacecraft interface is shown in Figures 5-49 through 5-54.

Separation of the spacecraft from the launch vehicle begins when the V-band clamp assembly is released. The secondary latch system loosely holds the spacecraft for a period of 30 sec, during which the spacecraft and second stage undergo many damped cycles of small amplitude rattling back and forth, resulting in low angular rates in comparison to what would exist without the secondary latch system. At the end of the 30-sec rate-damping period, the secondary latches are released and the second stage is backed away from the spacecraft by activating the helium retro system. The second stage then performs a CCAM to remove itself from the vicinity of the spacecraft. Note that the secondary latch system requires the addition of four holes in the spacecraft interface ring (Figures 5-53 and 5-54) to mate with the PAF mounted lateral restraint devices and spacecraft provided separation switches to interface with PAF mounted separation switch pads to assist in damping the angular rates. When the spacecraft does not require separation switches, Delta Program-provided damping devices, which interface directly with the aft side of the spacecraft interface ring, are mounted on the PAF to assist in damping the angular rates.

For missions where a low tip-off rate is not required, four guided separation spring actuators can be installed in place of the secondary latch system. Spacecraft separation occurs when the V-band clamp assembly is released and four spring actuators impart a relative separation velocity between the spacecraft and second stage. The second stage will then perform a CCAM to ensure a safe distance to the spacecraft.

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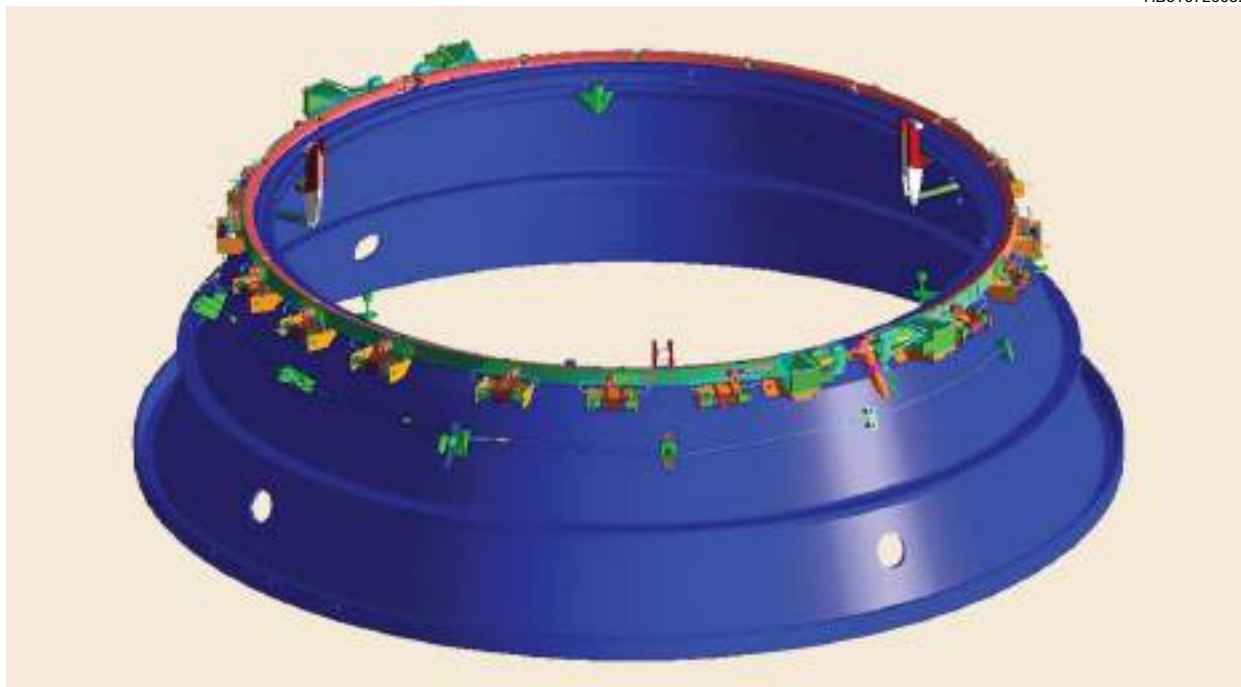


Figure 5-47. 4717 PAF

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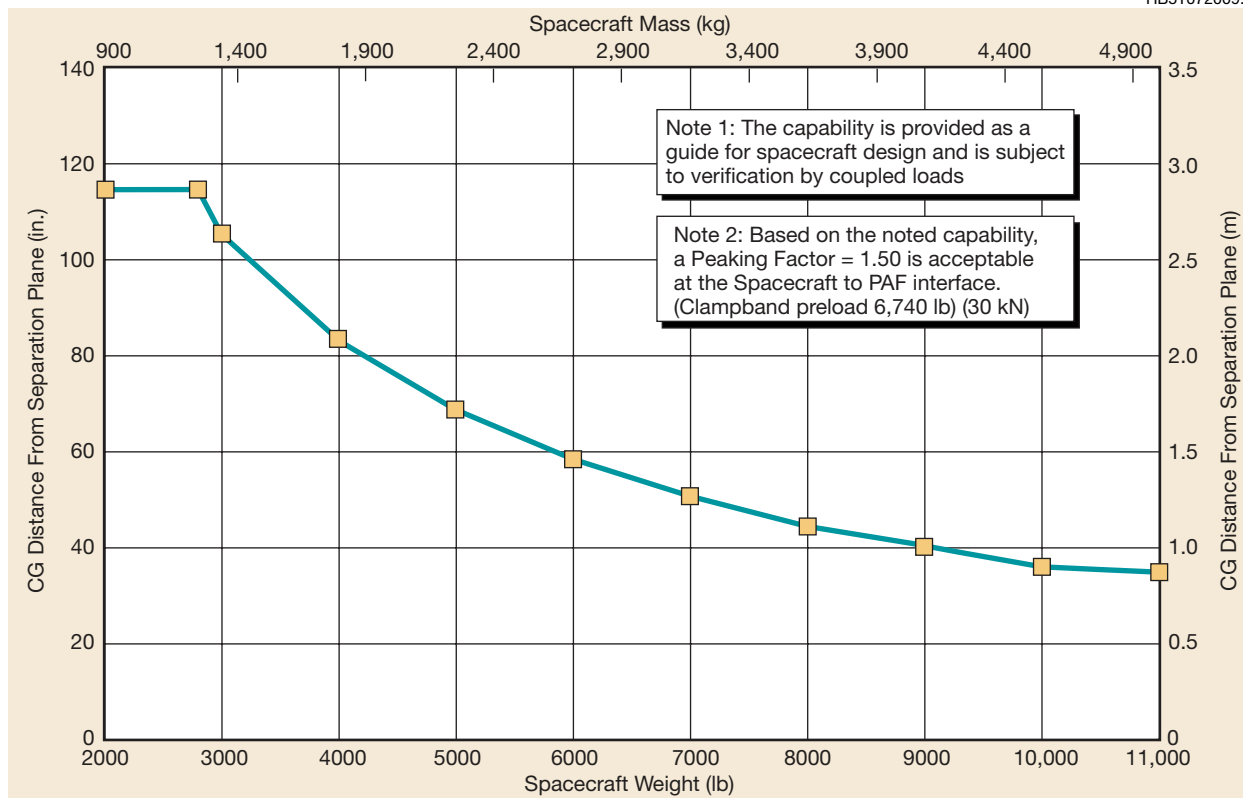


Figure 5-48. Capability of 4717 PAF

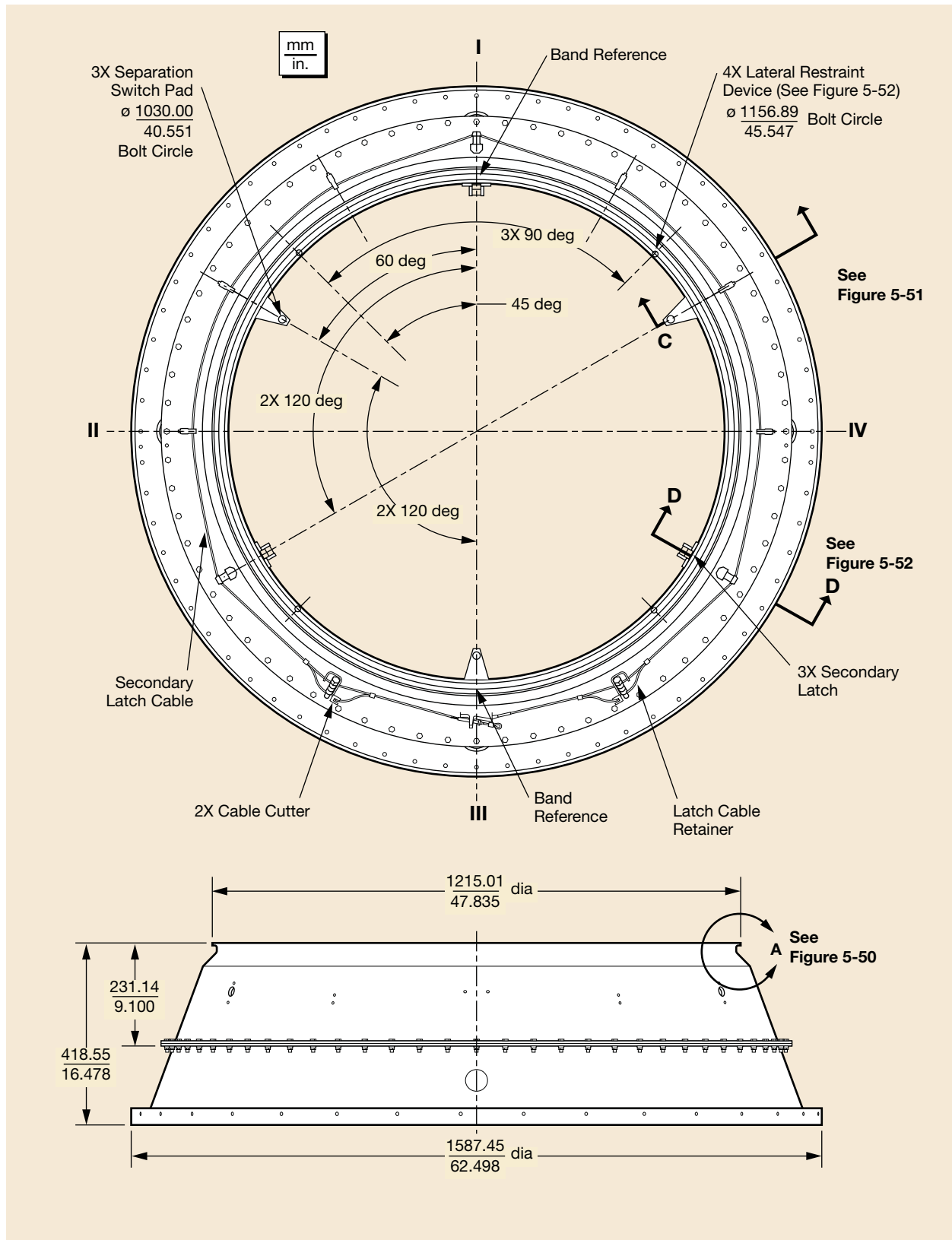
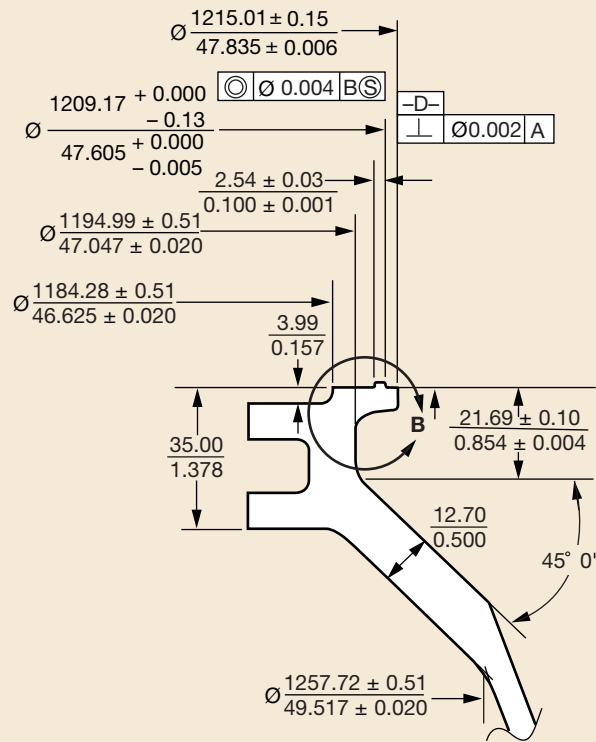


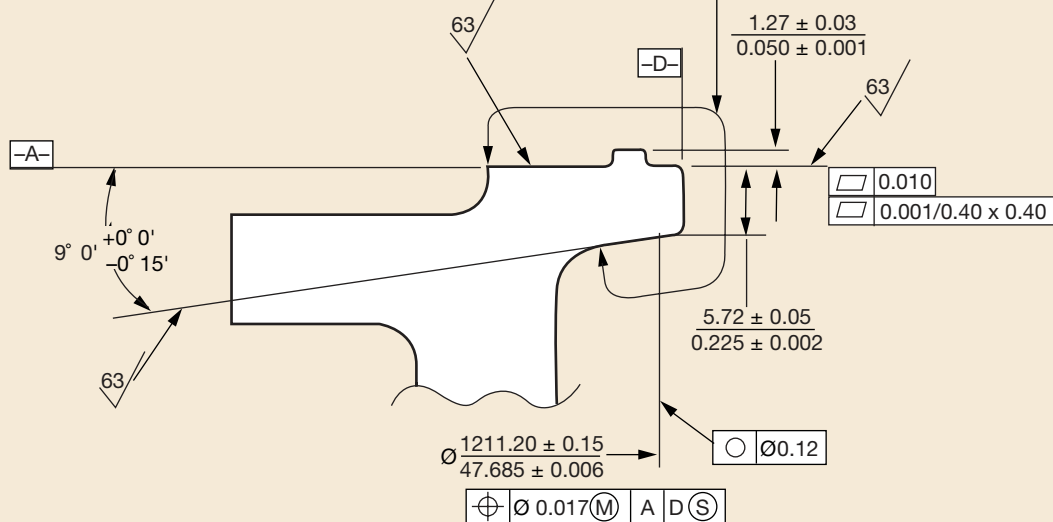
Figure 5-49. 4717 PAF Detailed Assembly

mm
in.



View A
From Figure 5-49

Chemical Conversion Coat per MIL-C-5541, Class 3



View B

Figure 5-50. 4717 PAF Detailed Dimension

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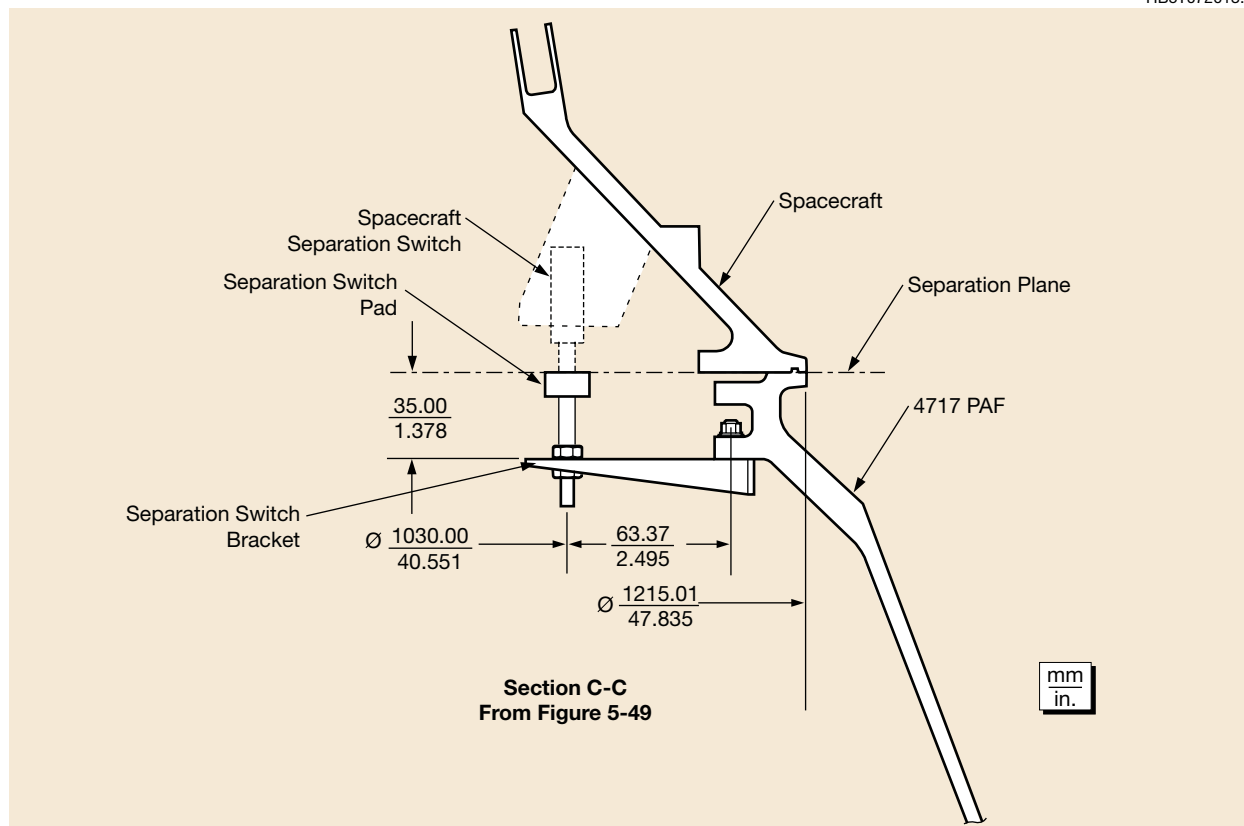


Figure 5-51. Spacecraft Separation Switch Interface—4717 PAF

HB5T072012.4

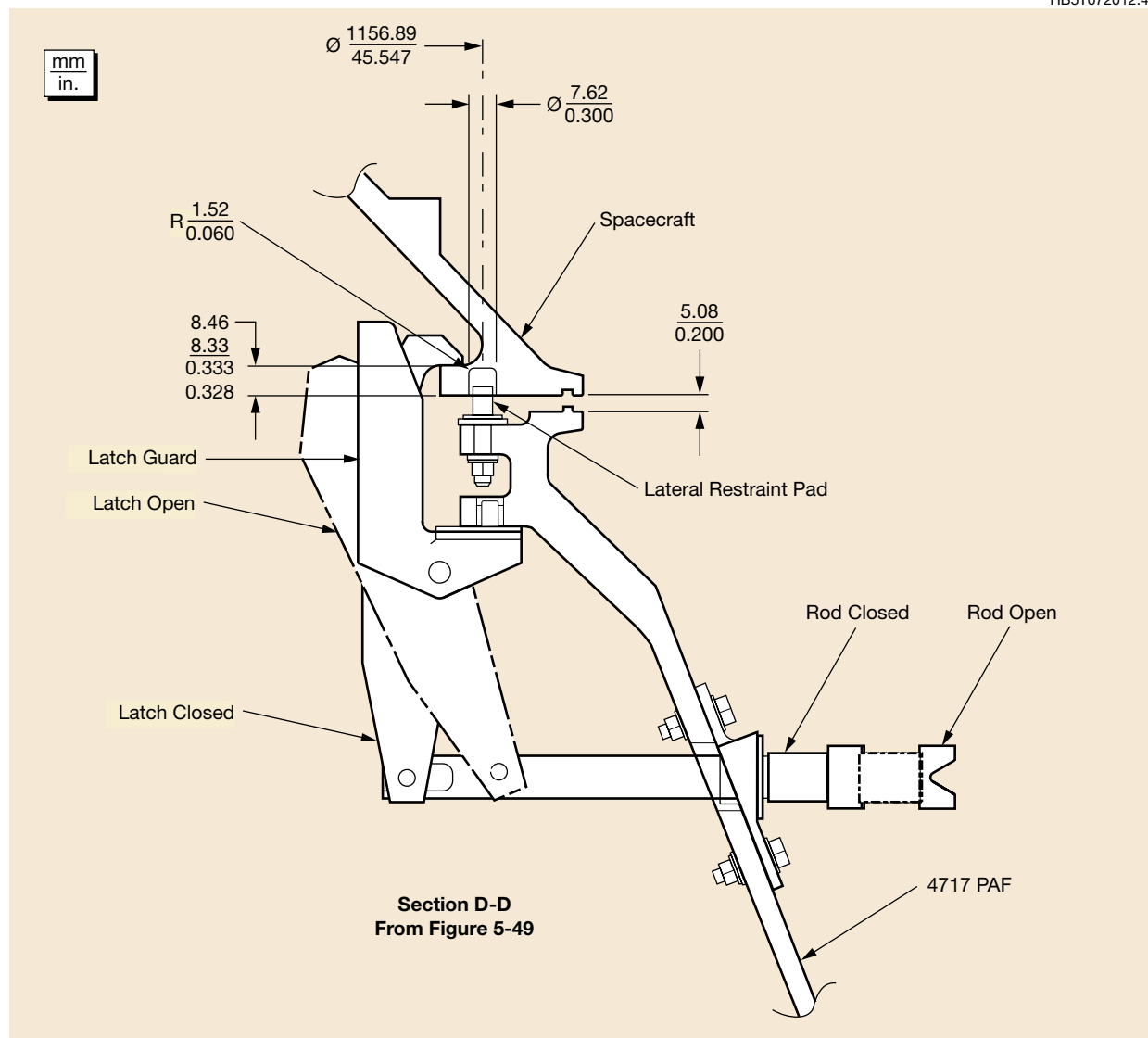


Figure 5-52. Latch Engagement Post-Clampband Separation—4717 PAF

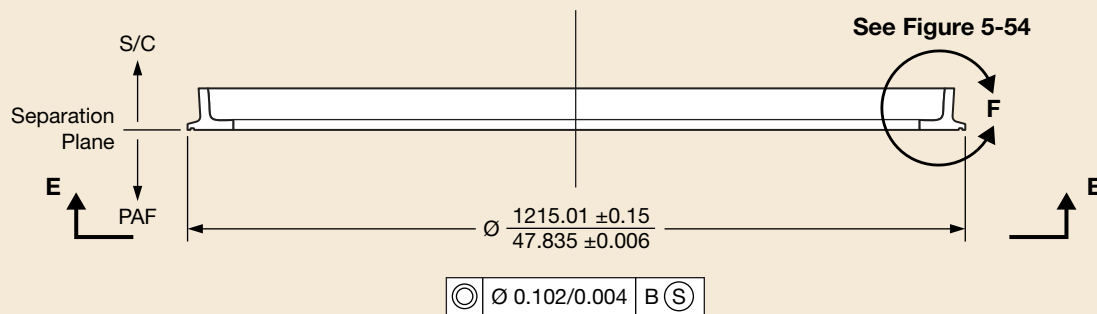
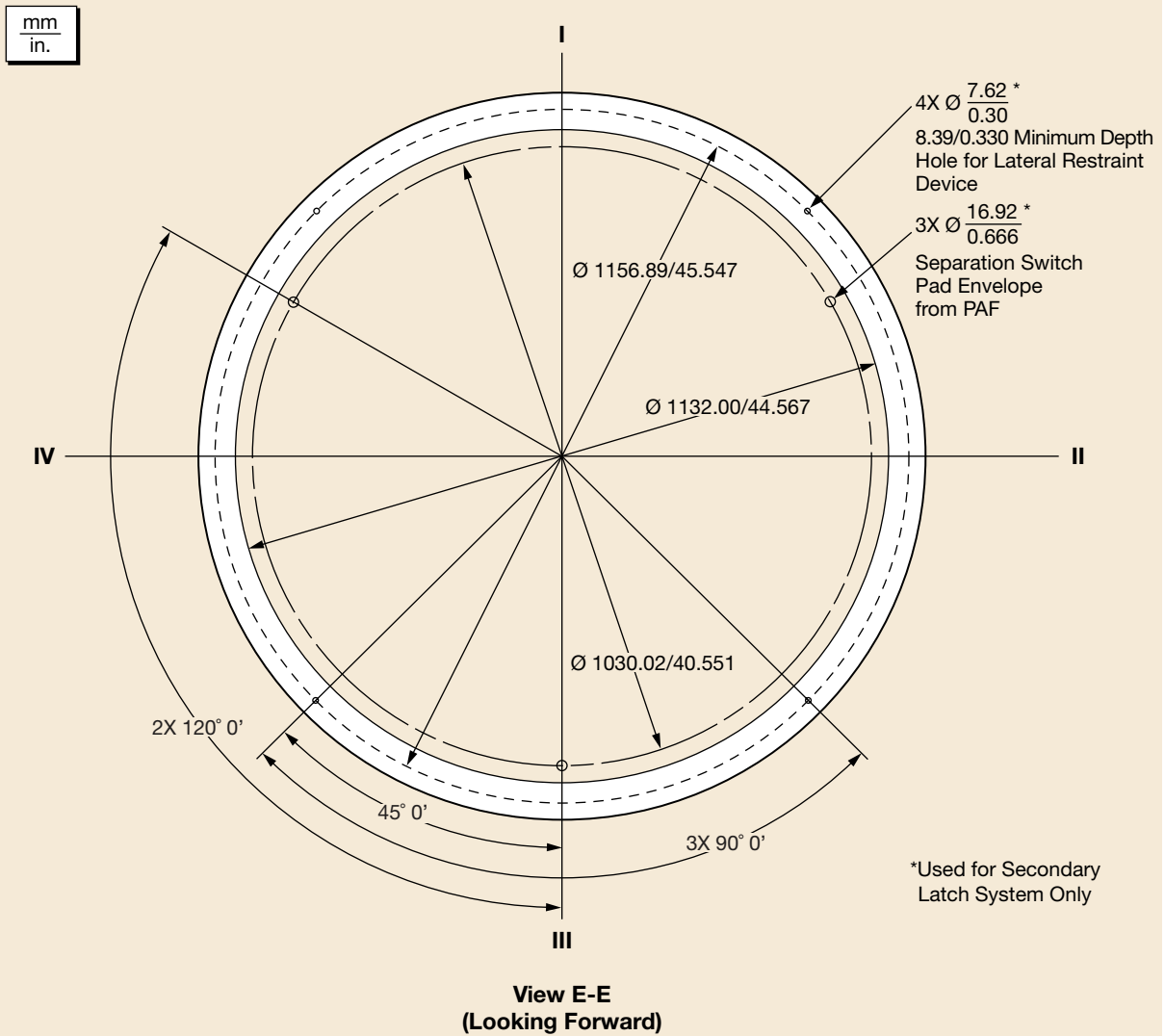


Figure 5-53. Dimensional Constraints on Spacecraft Interface to 4717 PAF

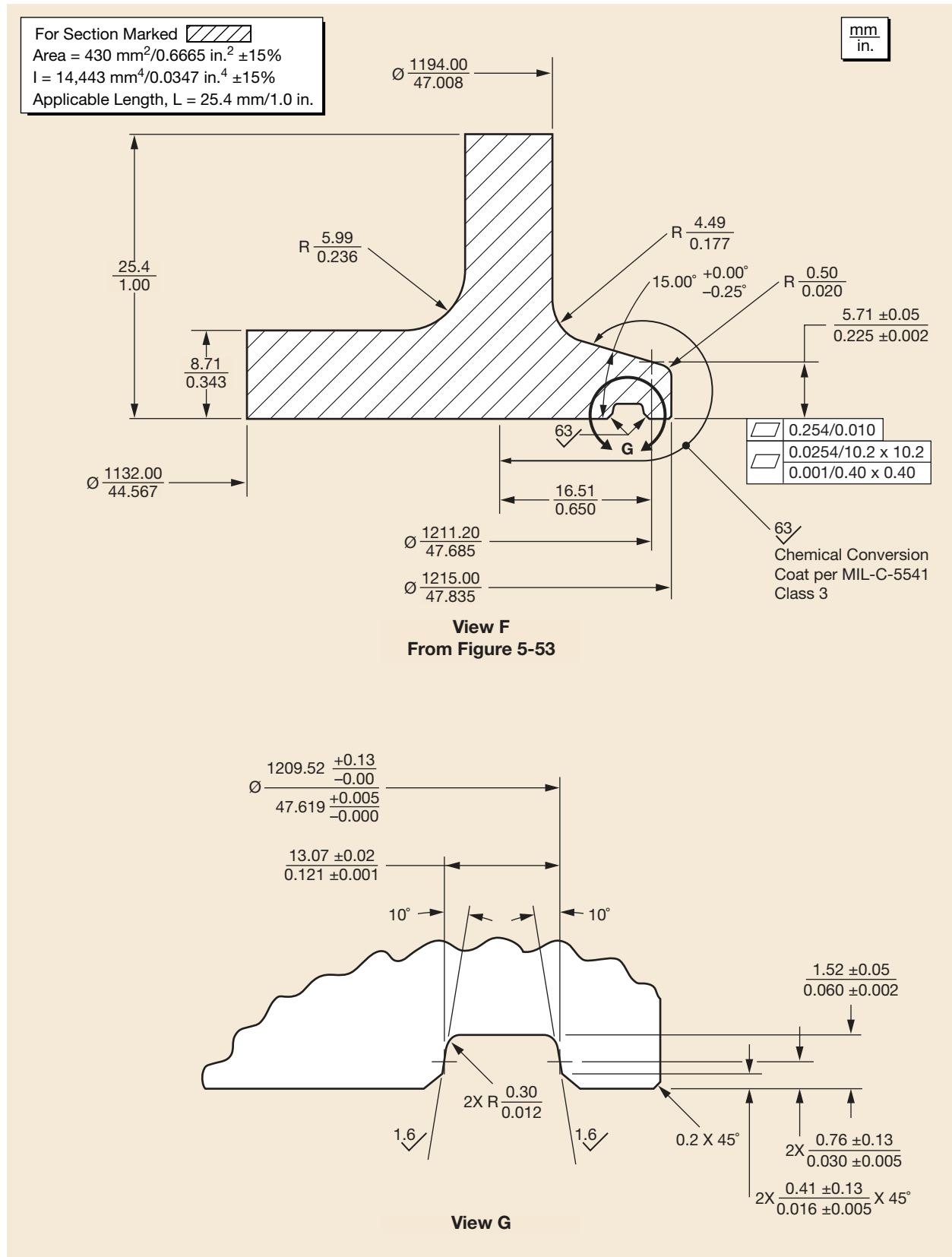


Figure 5-54. Dimensional Constraints on Spacecraft Interface to 4717 PAF

5.3.6 The 3715C PAF Assembly

The aluminum skin and stringer 3715C PAF assembly is approximately 389.64 mm (15.340 in.) high and 958.85 mm (37.750 in.) in diameter. The 3715C PAF configuration is shown in Figures 5-59 through 5-62.

The PAF base is attached on top of a 63.50 mm (2.500 in.) high direct mate adapter (DMA) interface ring to the forward ring located on top of the second stage for a standard two stage mission (Figure 5-55) or attached to a dual-payload attach fitting (DPAF) for a two stage dual payload mission (see section 5.4). A DMA interface ring is needed to mate with the GSE DMA structure for launch site payload processing of the combined spacecraft and PAF stack assembly.

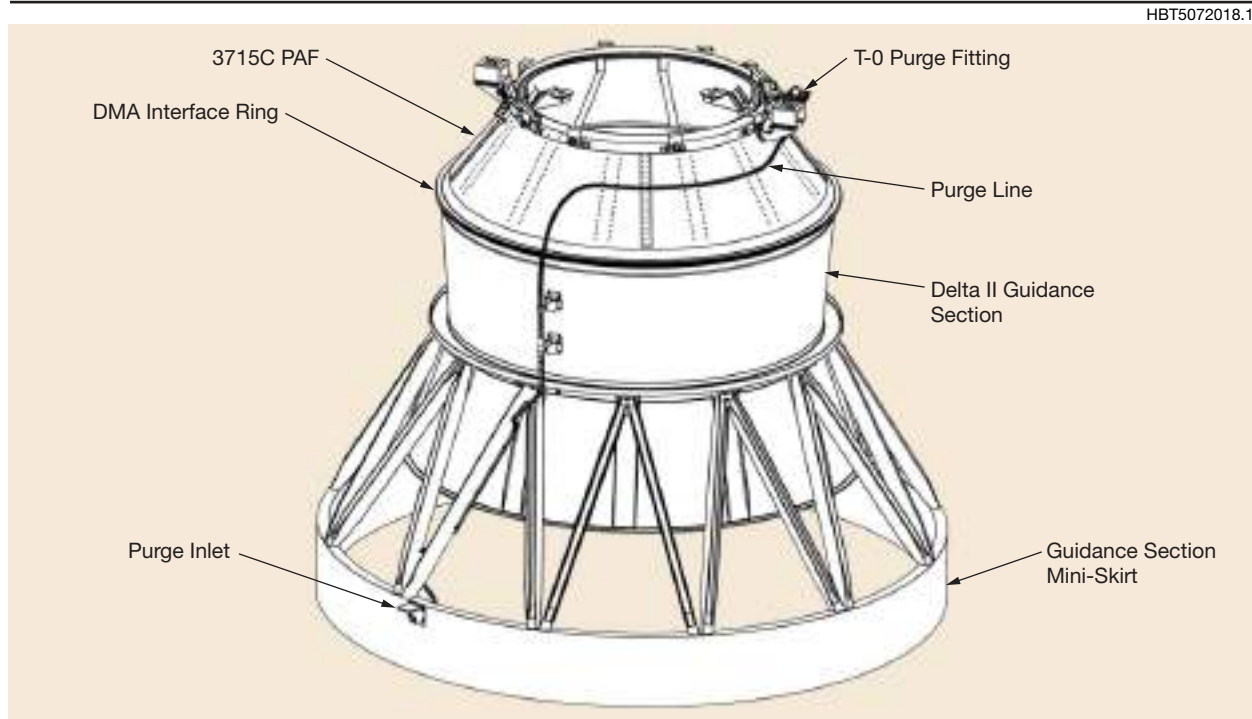


Figure 5-55. 3715C Payload Attach Fitting

The spacecraft is fastened to the PAF mating diameter with V-block type clamp assembly that is secured by two instrumented studs for clampband tensioning. Spacecraft separation is initiated by actuation of electrically initiated ordnance cutters that sever the two studs. Clamp assembly design is such that cutting either stud will permit spacecraft separation. Springs assist in retracting the clamp assembly into retainers after release to prevent recontact with spacecraft. A relative separation velocity is imparted to the spacecraft by four spring actuators. The second stage will then perform a CCAM to ensure a safe distance to the spacecraft. The associated spacecraft interface requirements are shown in Figures 5-63 and 5-64.

5.4 DUAL-PAYLOAD ATTACH FITTING (DPAF)

The Delta II dual-payload attach fitting (DPAF) (Figures 5-56 and 5-57) enables Boeing to offer alternate launch solutions by combining two payloads having similar orbit requirements onto a single launch vehicle. The DPAF is designed for use with the 3.0-m (10-ft)-dia and the stretched -10L composite fairing. The DPAF has an overall diameter of 2641.6 mm (104 in.) and an overall height to 3556.0 mm (140 in.) The PAFs for individual payloads are separate from the DPAF's shell structure to allow for streamlined independent payload processing.

Figure 5-58 shows PAF capability in terms of spacecraft weight and CG location above the separation planes. The maximum combined mass of both spacecraft cannot exceed 5000 lb. The capability for a specific spacecraft (with its own unique mass, size, and flexibility) might vary from that presented; therefore, when the spacecraft configurations determined, Boeing will initiate a coupled-loads analysis to verify that launch vehicle structural capability is not exceeded.

The payload attach fitting with associated separation mechanism for the upper and lower payloads are derived from the flight-proven 3712 PAF and designated as the 3715C PAF configuration, shown in Figures 5-59 through 5-64.

Each spacecraft is fastened to the PAF by a two-piece V-block type clamp assembly, which is secured by two instrumented studs. Spacecraft separation is initiated by actuation of electrically initiated ordnance cutters that sever the two studs. Clamp assembly design is such that cutting either stud will permit the spacecraft separation. Springs assist in retracting the clamp assembly into retainers after release to prevent recontact with the spacecraft. A relative separation velocity is imparted to the spacecraft by four spring actuators.

The DPAF separation system splits the shell structure circumferentially at a structural joint, allowing ejection of the upper portion of the DPAF using six matched spring cartridge assemblies. Access to the interior payload is through 0.61-m (24-in.)-dia access holes that are restricted to locations as defined in Figure 5-65. Two spacecraft access holes are provided as standard and must maintain a minimum center-to-center separation distance of 1 m (39.37 in.).

The DPAF is available with the following optional services for the internal payload: T-0 GN₂ purge across the separation plane, T-0 battery air-conditioning, contamination barrier, additional spacecraft access holes, and mission-specific instrumentation.



Figure 5-56. Dual-Payload Attach Fitting (DPAF)

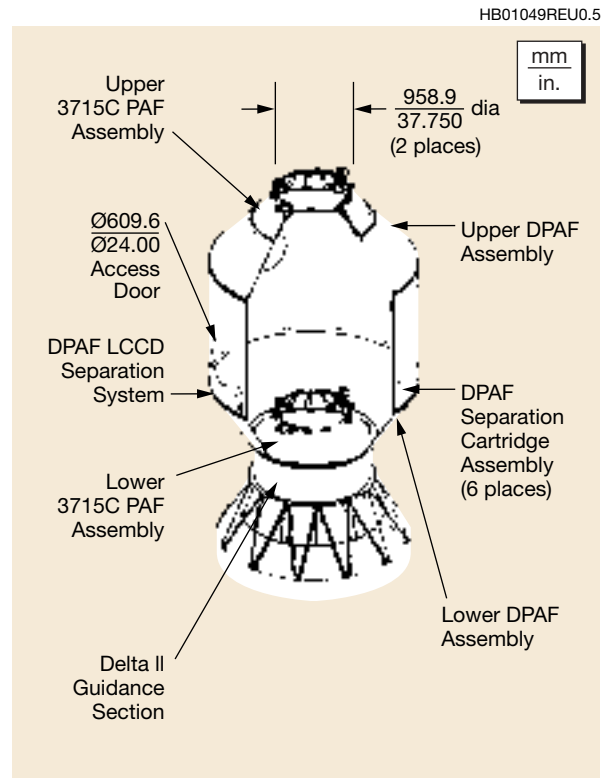


Figure 5-57. PAFs for Lower and Upper Payloads in Dual-Manifest

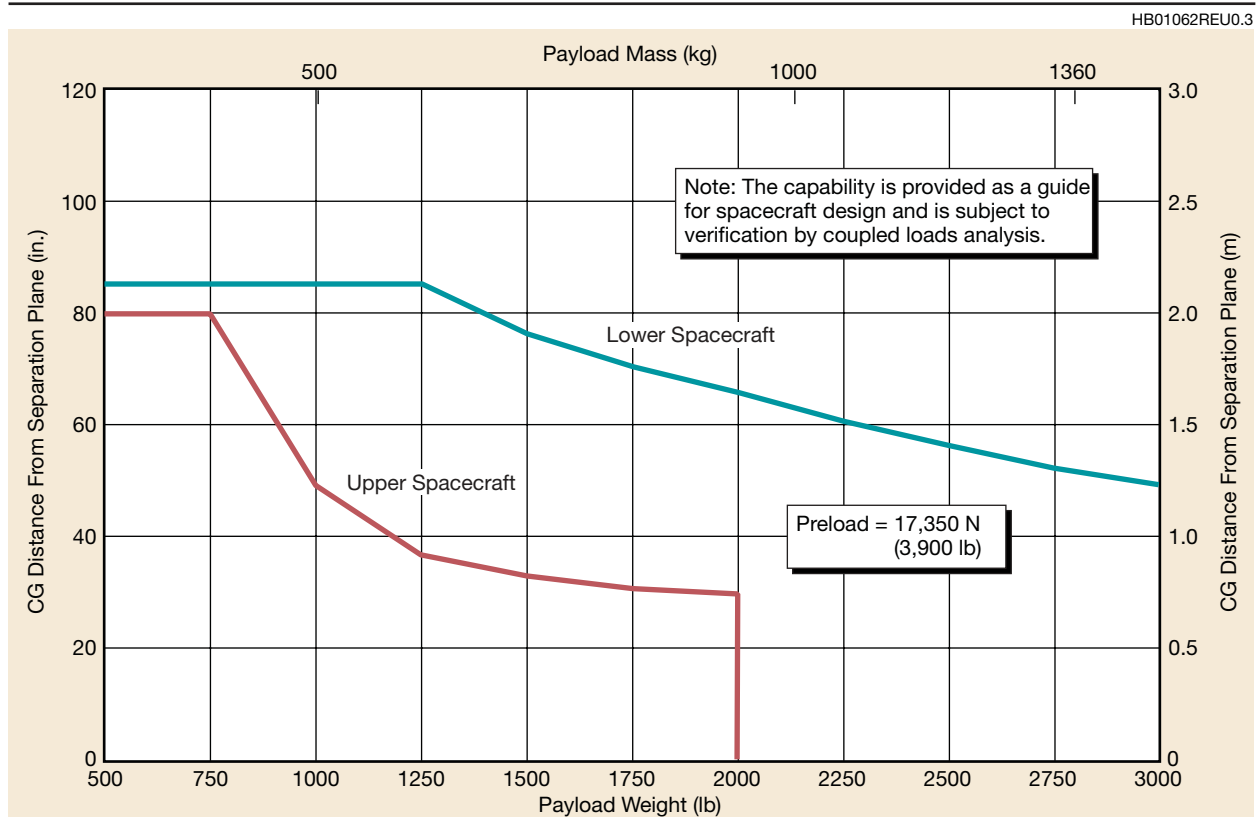


Figure 5-58. Capability of Dual-Payload Attach Fitting (DPAF)

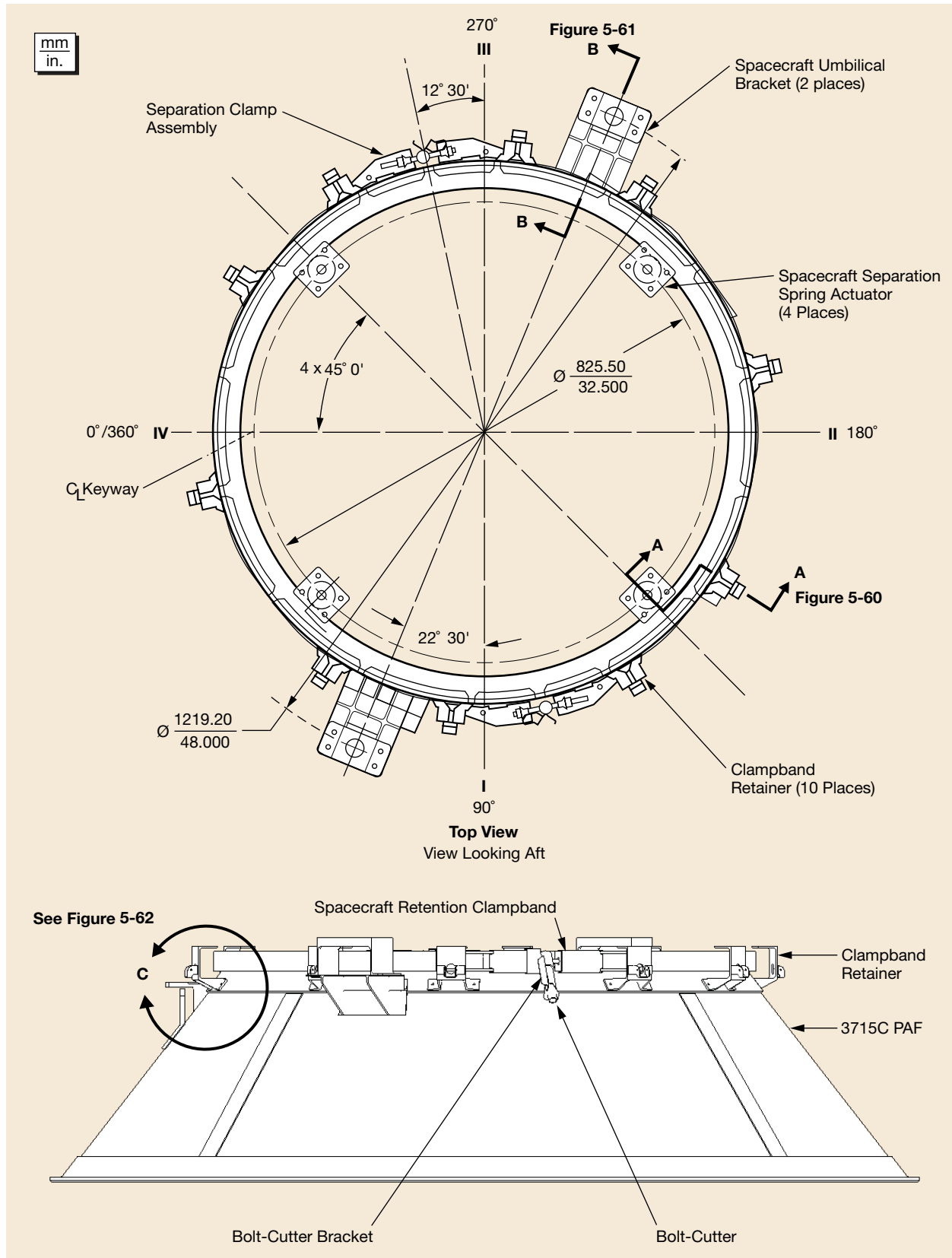


Figure 5-59. Dual-Payload Attach Fitting 3715C PAF Interface

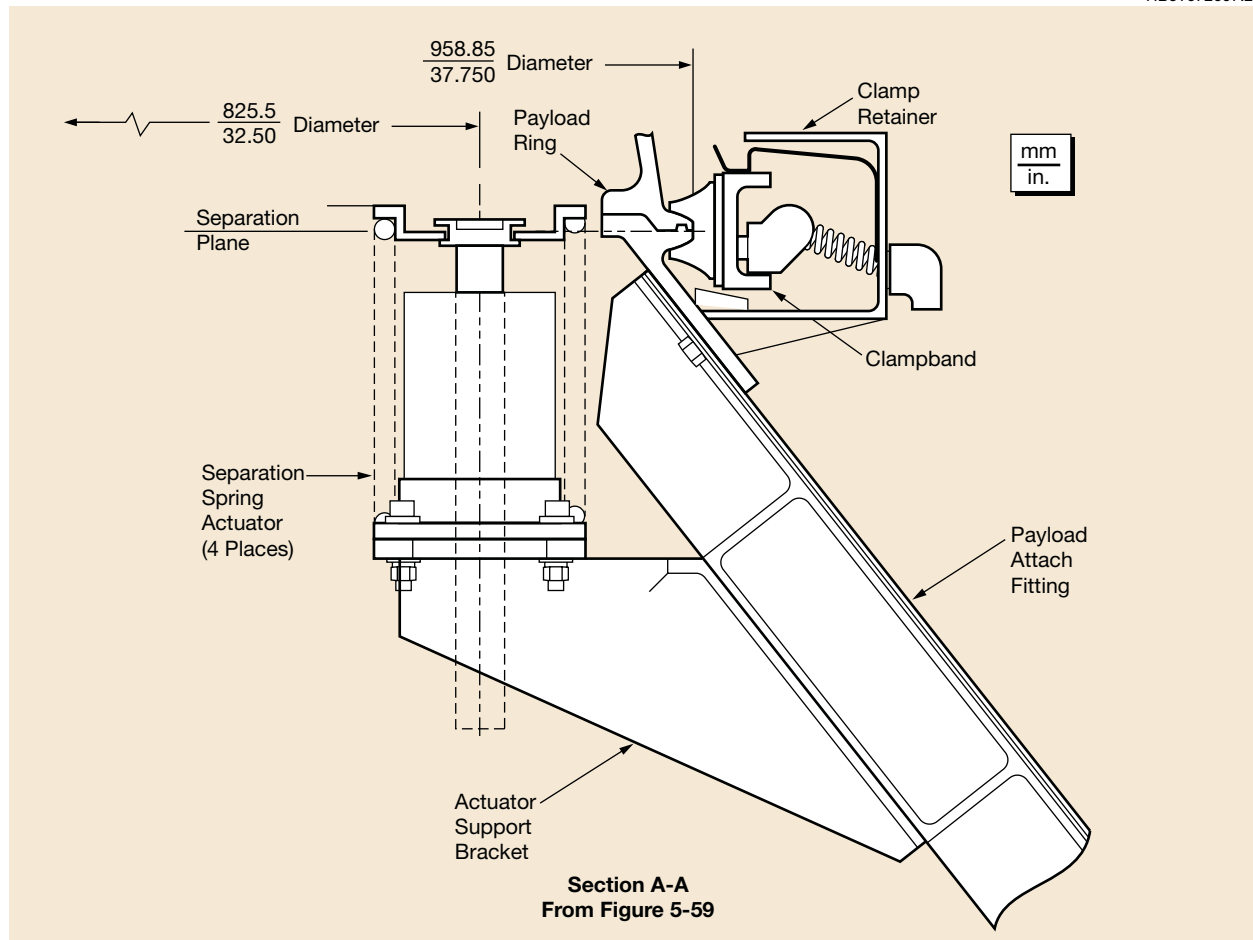


Figure 5-60. Dual-Payload Attach Fitting 3715C PAF Separation System Interfaces

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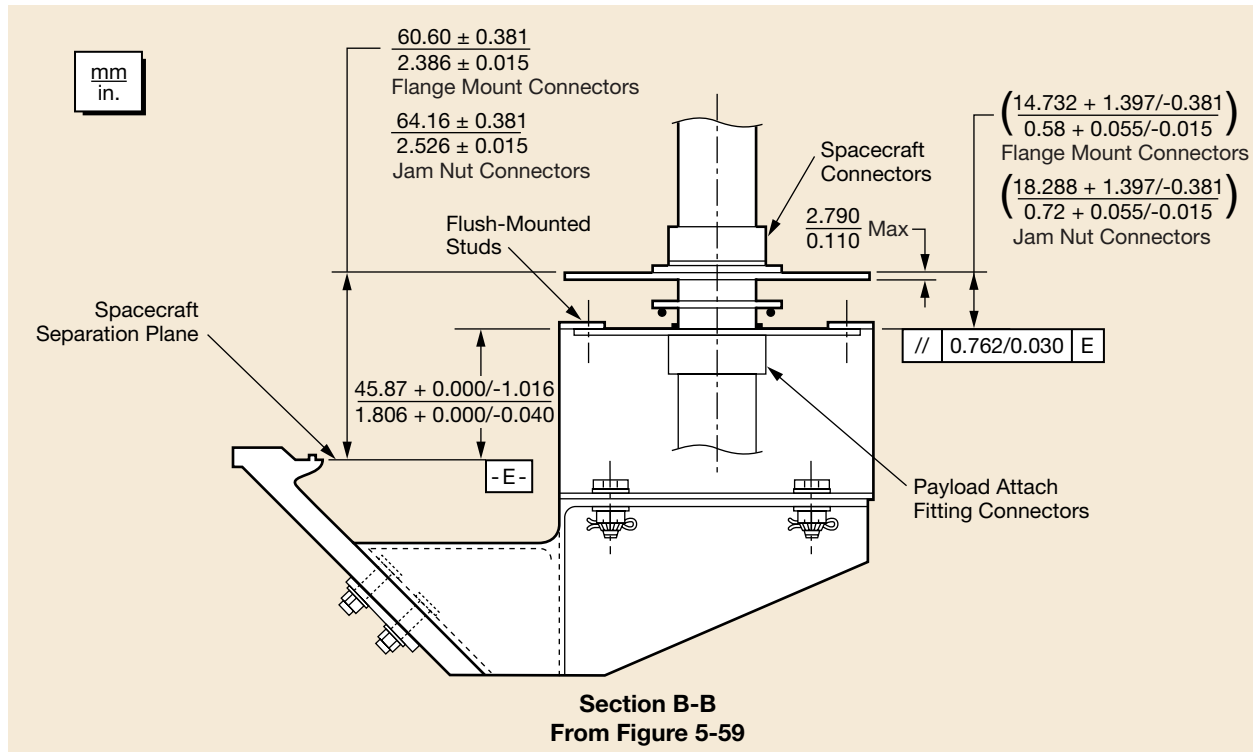


Figure 5-61. Dual-Payload Attach Fitting 3715C PAF Spacecraft Separation Interface—Electrical Connector Bracket

HB01053REU0.6

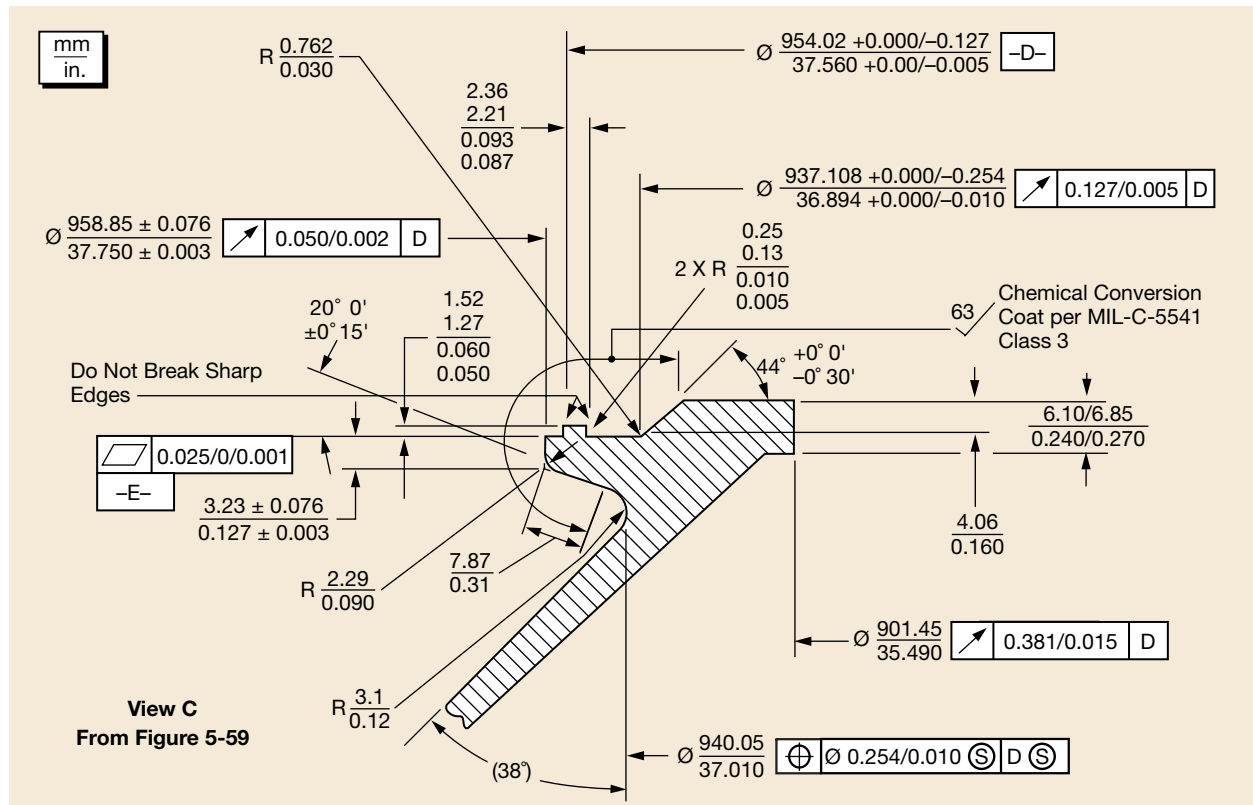


Figure 5-62. Dual-Payload Attach Fitting 3715C PAF Detailed Dimensions

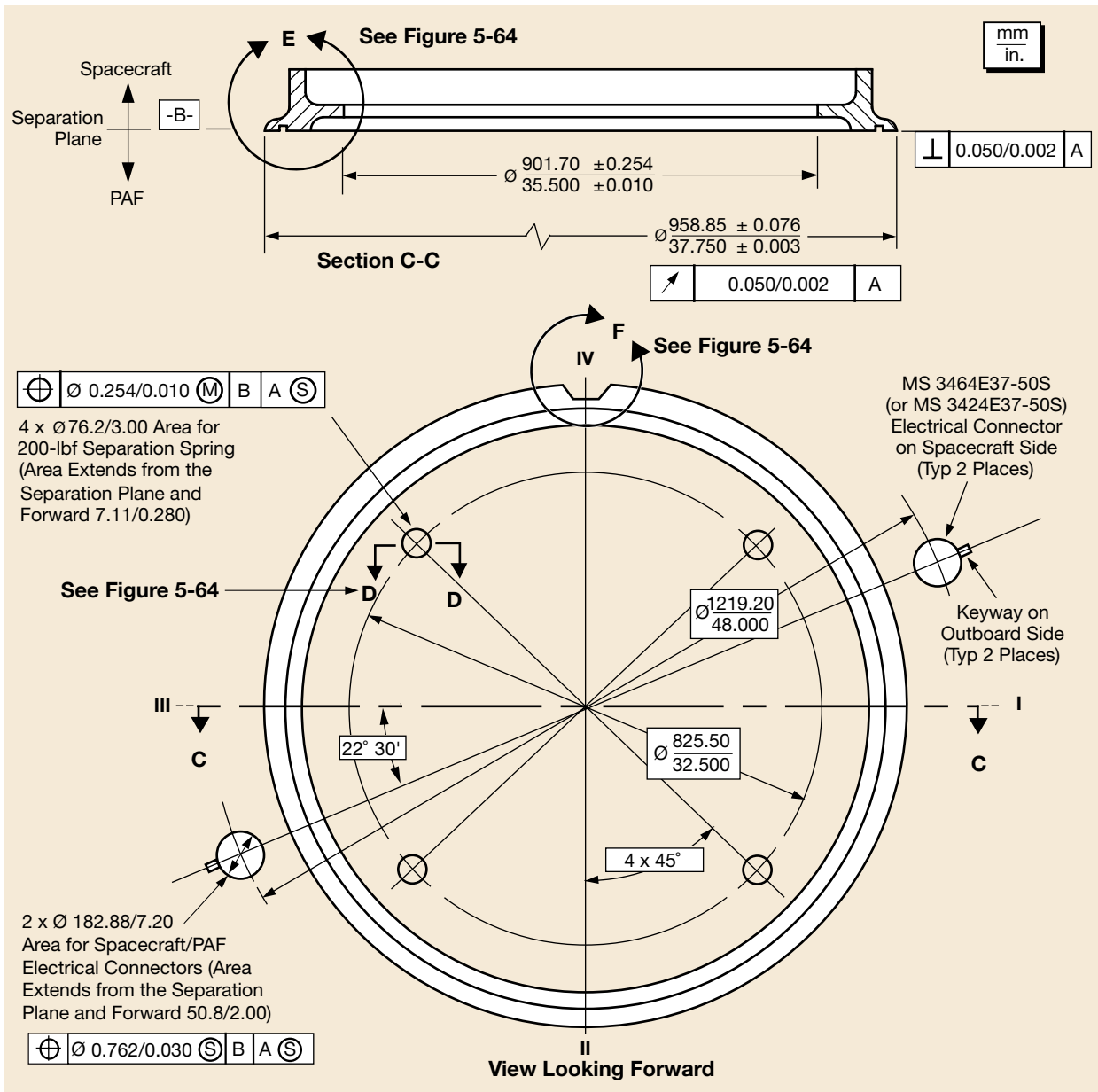
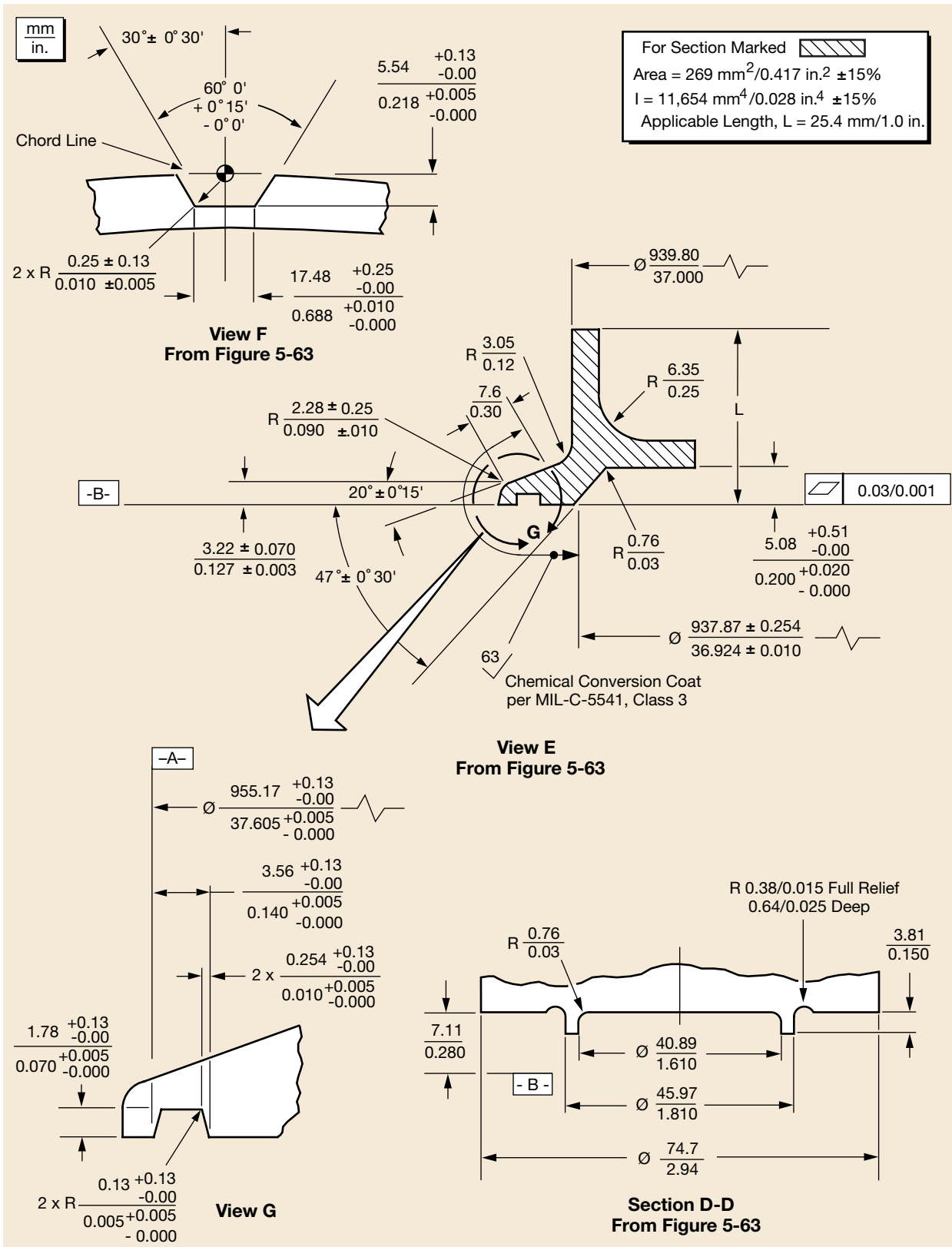
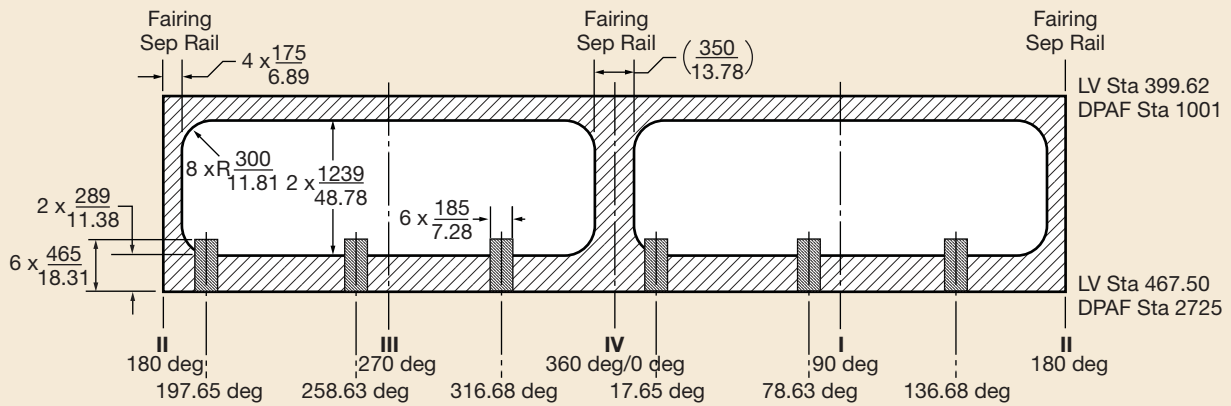


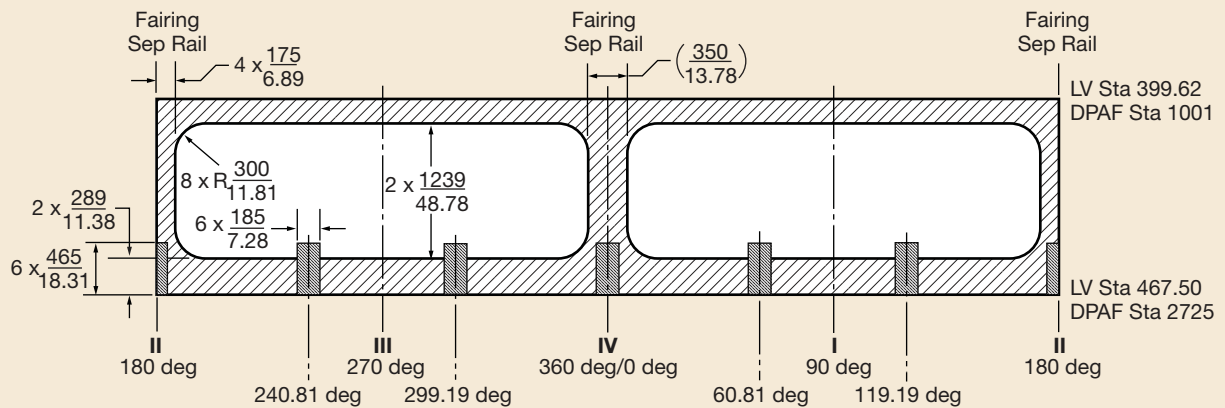
Figure 5-63. Dimensional Constraints on Spacecraft Interface to 3715C PAF



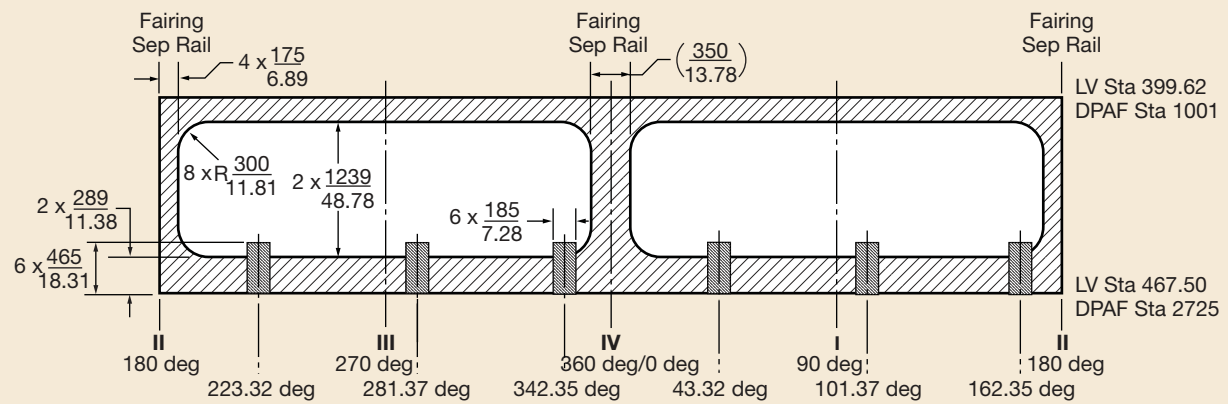
**Figure 5-64. Dimensional Constraints on Spacecraft Interface to 3715C PAF
(Views C, D, E, and Section B-B)**



Clocking (A)



Clocking (B)



Clocking (C)

Note: All Dimensions are in $\frac{\text{mm}}{\text{in.}}$ Allowable access hole area DPAF stayout area Spring cartridge assembly (SCA) stayout area

All views from outside DPAF

Figure 5-65. Dual-Payload Attach Fitting (DPAF) Allowable Access Hole Locations

5.5 SECONDARY PAYLOAD CHARACTERISTICS/INTERFACE

Where volume permits, provisions to accommodate two types of secondary payloads—separating and nonseparating—may be provided. The allowable characteristics of generic secondary payloads are specified in Table 5-3.

Table 5-3. Characteristics of Generic Separating and Nonseparating Secondary Payloads

Characteristic	Separating	Nonseparating
Weight/CG distance from separation plane (not to exceed)	45.4 kg (100 lb)/12.7 cm (5.0 in.)	69.8 kg (154 lb)/17.8 cm (7.0 in)
Volume (not to exceed)	47.8 by 34.8 by 29.3 cm (18.82 by 13.68 by 11.54 in.)	47.5 by 33.6 by 35.5 cm (18.71 by 13.23 by 11.96 in.)
Electrical interface	None	None
Attachment	24.1-cm (9.5-in.)-dia clampband (See Figure 5-66)	Bolted (see Figure 5-67)
Coupled frequency (coupled to Delta II second stage)	>35 Hz	>35 Hz

002248

The standard separation interface available for separating secondary payloads is shown in Figure 5-66. Each spacecraft is fastened to the PAF by a two-piece V-block type clamp assembly, which is secured by two instrumented studs. Spacecraft separation is initiated by actuation of electrically initiated ordnance cutters that sever the two studs. Clamp assembly design is such that cutting either stud will permit the spacecraft separation. The separation event is sequenced and controlled by the launch vehicle. The interface for nonseparating payloads is shown in Figure 5-67. Contact the Delta Program Office for spacecraft interface definition.

Figure 5-68 shows the capability of the secondary payload interface for separating payloads in terms of spacecraft weight and CG location above the separation plane. The capability for a specific spacecraft (with its own unique mass, size, and flexibility) may vary from that presented in Figure 5-68. Therefore, when the spacecraft configuration is determined, the Delta Program will initiate a coupled-loads analysis to verify that the launch vehicle structural capability is not exceeded.

No electrical interface is available between the launch vehicle and the secondary payload. Secondary payloads may require a battery trickle charge through the existing fairing access door that will be available until fairing close-out. Charging equipment and cabling are the responsibilities of the secondary payload customer. The secondary payload flight mechanical interfaces will be verified at the factory during fitcheck prior to shipping to the launch site. The fitcheck verification will also include access verification for connectors and payload installation clearance and interference.

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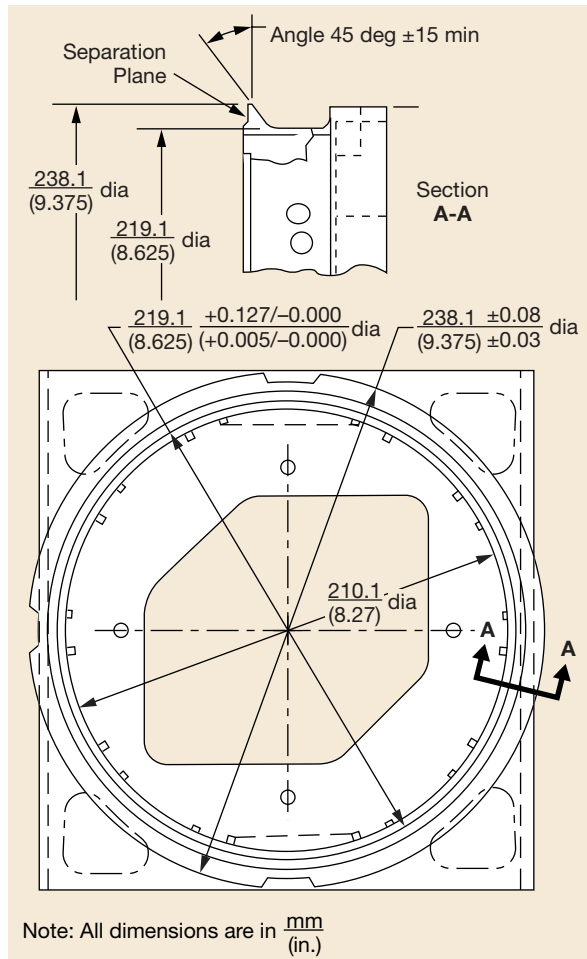


Figure 5-66. Separating Secondary Payload Standard Launch Vehicle Interface

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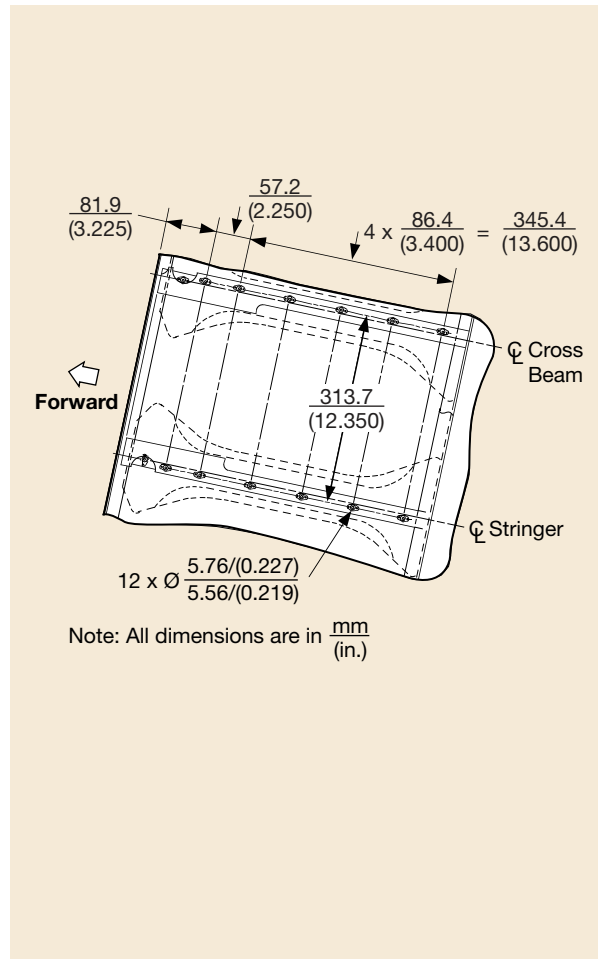


Figure 5-67. Nonseparating Secondary Payload Standard Mounting Interface

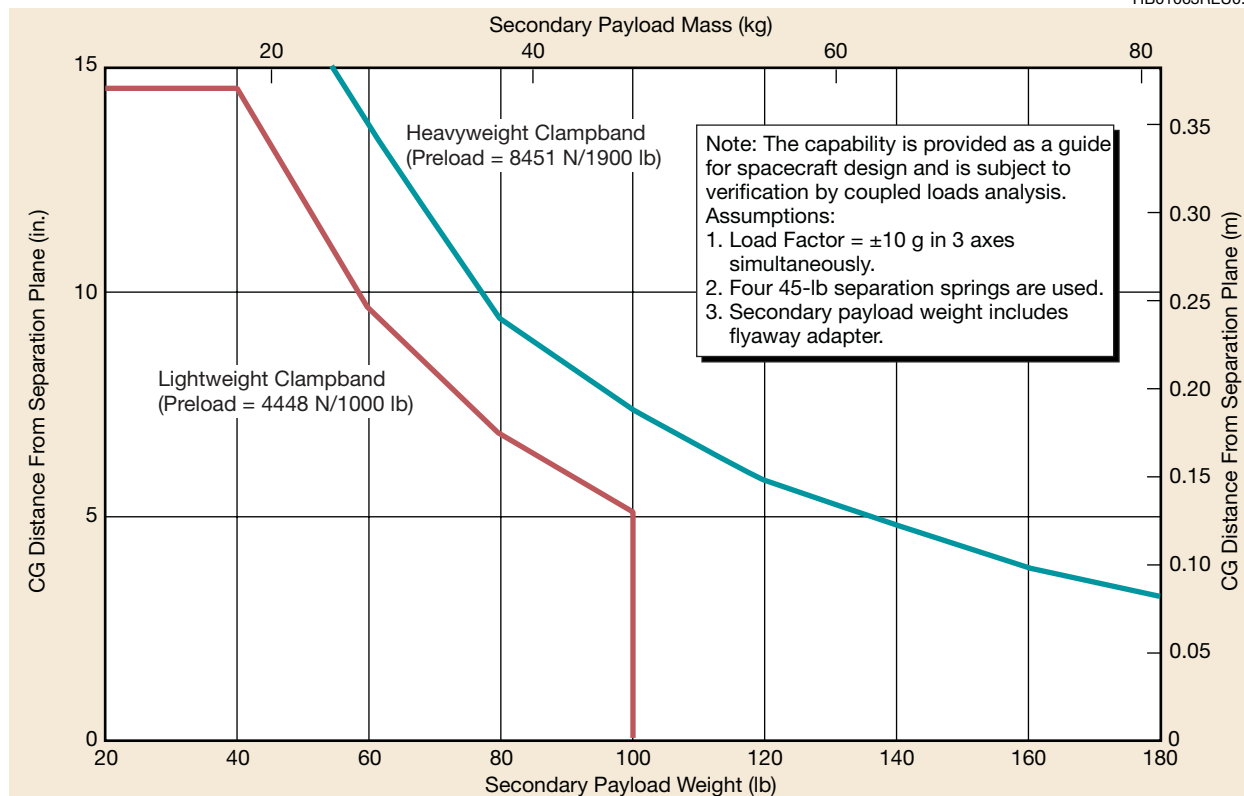


Figure 5-68. Capability of Separating Secondary Payloads

5.6 PAYLOAD ATTACH FITTING (PAF) DEVELOPMENT

Boeing continuously undertakes study of PAFs of differing interface diameters in supporting our customers' needs. The design of these PAFs takes into account the use of the separation clamp assembly interfaces that have been qualified for the Delta II launch vehicle. These clamp assemblies are listed in Table 5-4. For interfaces different than those listed, please consult the Delta Program Office.

Table 5-4. Separation Clamp Assemblies

Approximate diameter (mm/in.)	Max flight preload (N/lb)	Spacecraft PAF flange angle (deg)
1143/45	30,248/6800	15

002251.2

5.7 TEST FITTINGS AND FITCHECK POLICY

A PAF test fitting can be provided to the customer to assist in conducting environmental tests that are needed to ensure spacecraft flight readiness except for the 4717 PAF. The test fitting is returned after testing is completed. In addition, a fitcheck can be conducted with the spacecraft using the flight PAF. This is typically done prior to shipment of the spacecraft to the launch site. Boeing personnel will be available to conduct this activity. The fitcheck verifies the flight interfaces (mechanical and electrical) and the clearances of any attached hardware. The spacecraft must include all flight hardware so that adequate access and clearance can be demonstrated. The

customer will provide a support stand for the PAF and the bolts needed to secure the PAF to it. Specific detail requirements for the fitcheck will be provided by the Delta Program.

5.8 ELECTRICAL DESIGN CRITERIA

Presented in the following paragraphs is a description of the spacecraft/vehicle electrical interface design constraints. The discussion includes remote-launch-center-to-blockhouse, blockhouse-to-spacecraft wiring, spacecraft umbilical connectors, aerospace ground equipment (AGE), the grounding system, and separation switches. The remote launch center (RLC) for CCAFS is the 1st Space Launch Squadron (1SLS) Operations Building (OB), and the remote launch control center (RLCC) for VAFB is in building 8510.

5.8.1 Remote Launch Centers, Blockhouse-to-Spacecraft Wiring

Provisions are made for controlling and monitoring the spacecraft from the blockhouse or RLC. Spacecraft operations in the blockhouse are allowed after mating until second-stage propellant loading occurs, at which time all operations have to be conducted from the RLC until liftoff. Wiring is routed from a payload console in the blockhouse through a second-stage umbilical connector, through fairing wire harnesses (typically), and to the spacecraft or PAF by lanyard-operated quick-disconnect connectors (typically). Remote control of spacecraft functions is provided through fiber optic cables during testing and launch from the RLC.

For a typical vehicle, a second-stage umbilical connector (JU2) is provided for payload servicing wiring. A typical baseline wiring configuration provides up to 37 wires through each of two fairing sectors. The fairing wire harnesses terminate in lanyard disconnect connectors that mate to the PAF or directly to the spacecraft. Additional wiring can be provided by special modification. Available wire types are twisted/shielded (up to 4 conductors), single-shielded, or unshielded (up to 4 conductors). A typical vehicle wire harness configuration is shown in Figure 5-69. Other configurations can be accommodated.

The baseline wiring configuration between the fixed umbilical tower (FUT) and the blockhouse consists of the following. At Cape Canaveral Air Force Station (CCAFS), the configuration at Space Launch Complex (SLC)-17A and SLC-17B consists of 60 twisted and shielded pairs (120 wires), 12 twisted and shielded pairs (24 wires), and 14 twisted pairs (28 wires). At Vandenberg Air Force Base (VAFB), the configuration at SLC-2 consists of 30 twisted and shielded pairs (60 wires), 20 twisted and shielded pairs (40 wires), two twisted and shielded triplets (6 wires), eight 50-ohm coax cables, and six fiber-optic cables to blockhouse; or 60 twisted shielded pairs (120 wires), 28 twisted pairs (56 wires), and 8 TWINAX twisted shielded pairs 78Ω controlled impedance (16 wires), to electrical equipment building (EEB).

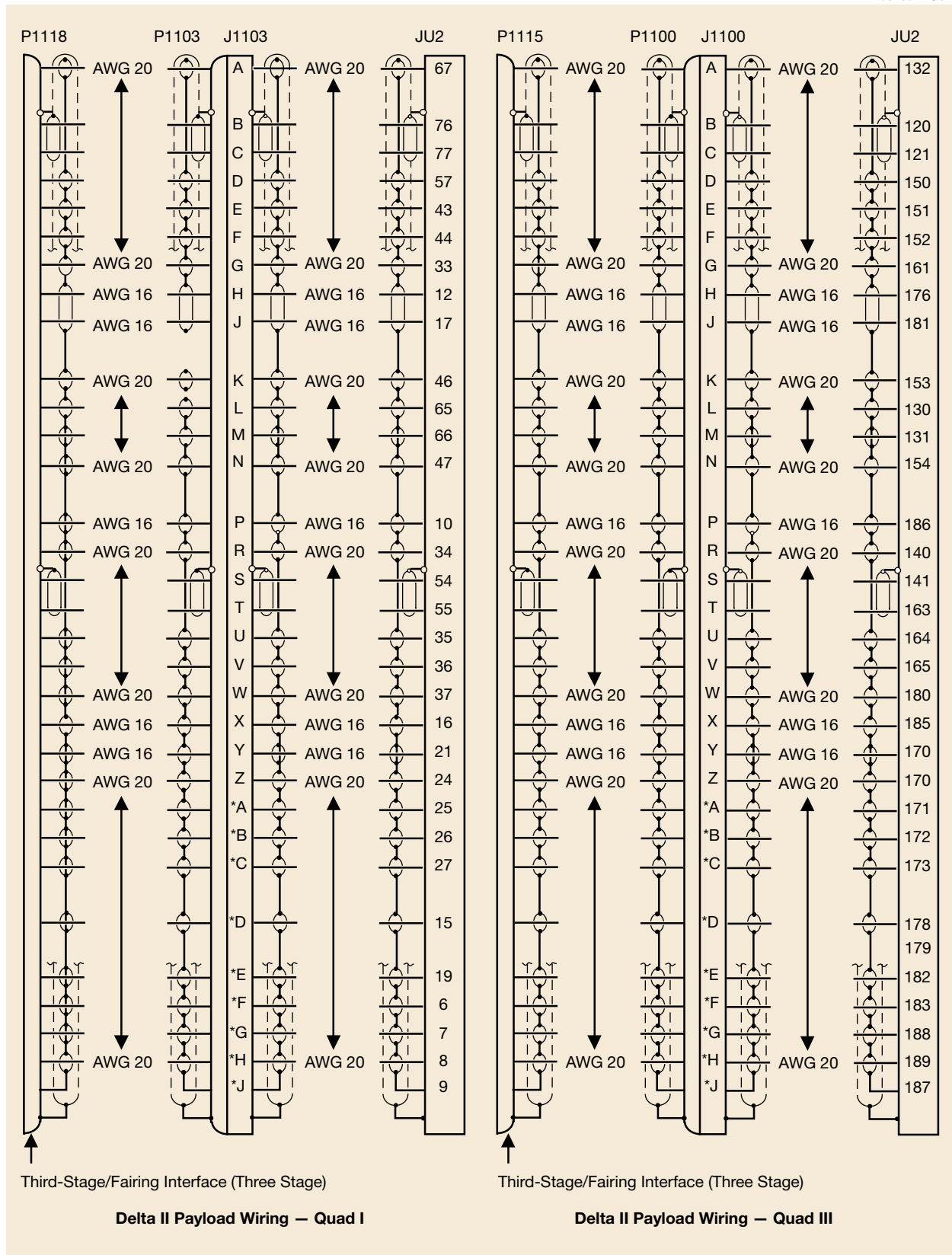


Figure 5-69. Typical Three-Stage Delta II Wiring Configuration

Space is available in the blockhouse for installation of the ground support equipment (GSE) required for spacecraft checkout. The space allocated for the spacecraft GSE is described in Section 6 for SLC-17 and Section 7 for SLC-2. There is also limited space in the umbilical J-box for a buffer amplifier or other data line conditioning modules required for data transfer to the blockhouse. The space allocated in the junction box (J-box) for this equipment has dimensions of approximately 303 by 305 by 203 mm (12 by 12 by 8 in.) at SLC-17A and B and 381 by 330 by 229 mm (15 by 13 by 9 in.) at SLC-2.

The standard interface method is as follows:

- A. The customer normally provides a console and a 12.2-m (40-ft) cable to interface with the spacecraft rack box in the blockhouse or EEB. Boeing will provide the interfacing cable if requested by the customer. Interface cable lengths and assignment of remote assists will be determined depending on customer needs.
- B. The spacecraft apogee motor safe and arm (S&A) circuit (if applicable) must interconnect with the operations safety manager's console (CCAFS only). The Delta Program provides a spacecraft remote control and monitoring interface between the blockhouse and remote launch centers (ISLS Operations Building, Eastern Range, and Remote Launch Control Center Bldg. 8510, Western Range).

The spacecraft remote capability listed below is the same at both ranges except as noted.

1. Discrete

Remote Launch Center	Blockhouse
28 inputs (CCAFS)	28 contact closures (CCAFS)
20 inputs (VAFB)	20 contact closures (VAFB)
18 contact closures	18 inputs

Note: A customer-provided high (28 VDC) at the Boeing discrete interface will result in a dedicated relay contact closure at the remote location (10-amp load capability).

2. Analog

Remote Launch Center	Blockhouse
48 analog outputs range ± 10 V	12 inputs ± 100 mV
	24 inputs ± 10 V
	12 inputs ± 100 V

3. Data Bus Communication between Remote Launch Centers and Blockhouse

- a. Fiber-optic RS232 modem/multiplexer card

Type:	4 each (CCAFS)
	1 each (VAFB)

 - Full duplex RS232 modem (13 wire) or
 - 6-channel multiplexer mode modem (2 wires each)

- b. Fiber-optic RS422 modem/multiplexer card 1 each
Type:
 - Full duplex RS422 modem (21 wire) or
 - 6-channel multiplexer mode modem (4 wires each)
- c. Fiber-optic RS232/RS422 dual-modem card 2 each
Type:
 - Up to 4 each RS232 modems (2 wire) or
 - Up to 4 each RS422 modems (4 wire) or
 - 2 each RS232 and 2 each RS422 modems
- d. Fiber-optic RS48 modem
Type:
 - Full duplex RS485 modem (4 wire) or
 - Full duplex RS485 modem (2 wire)
- 4. Fiber-optic ethernet campus bridge (CCAFS only) 2 each
- 5. Fiber-optic cable between remote launch center and blockhouse
Single-mode fiber optic cable interface with up to 24 fibers
Note: The number of available fibers depends on the number of fiber optic transceivers being used. Maximum number is 24, all terminated with ST connectors.
- C. A spacecraft-to-blockhouse-to RLC wiring schematic is prepared for each mission from requirements provided by the customer.
- D. To ensure proper design of the spacecraft-to-blockhouse wiring, the following information, which must comply with the above requirements, shall be furnished by the customer:
 - Number of wires required.
 - Pin assignments in the spacecraft umbilical connector(s).
 - Shield requirements for RF protection or signal noise rejection.
 - Function of each wire, including voltage, current, frequency, load type, magnitude, polarity, and maximum resistance or voltage-drop requirements. Note: There is a maximum allowable current of 10 mA per pin across the JU2 interface at liftoff.
 - Voltage of the spacecraft battery and polarity of the battery ground.
 - Part number and item number of the spacecraft umbilical connector(s) (compliance required with the standardized spacecraft umbilical connectors listed in Section 5.8.2).
 - Physical location of the spacecraft umbilical connector including (1) angular location in relation to the quadrant system, (2) station location, and (3) radial distance of the out-board face of the connector from the vehicle centerline for a fairing disconnect or connector centerline for PAF disconnect.
 - Periods (checkout or countdown) during which hard-line-controlled/monitored systems will be operated.

During on-pad checkout, the spacecraft can be operated with the fairing installed or stored. Typical harness arrangements for both configurations are shown in Figure 5-70 for the ER and Figure 5-71 for the WR.

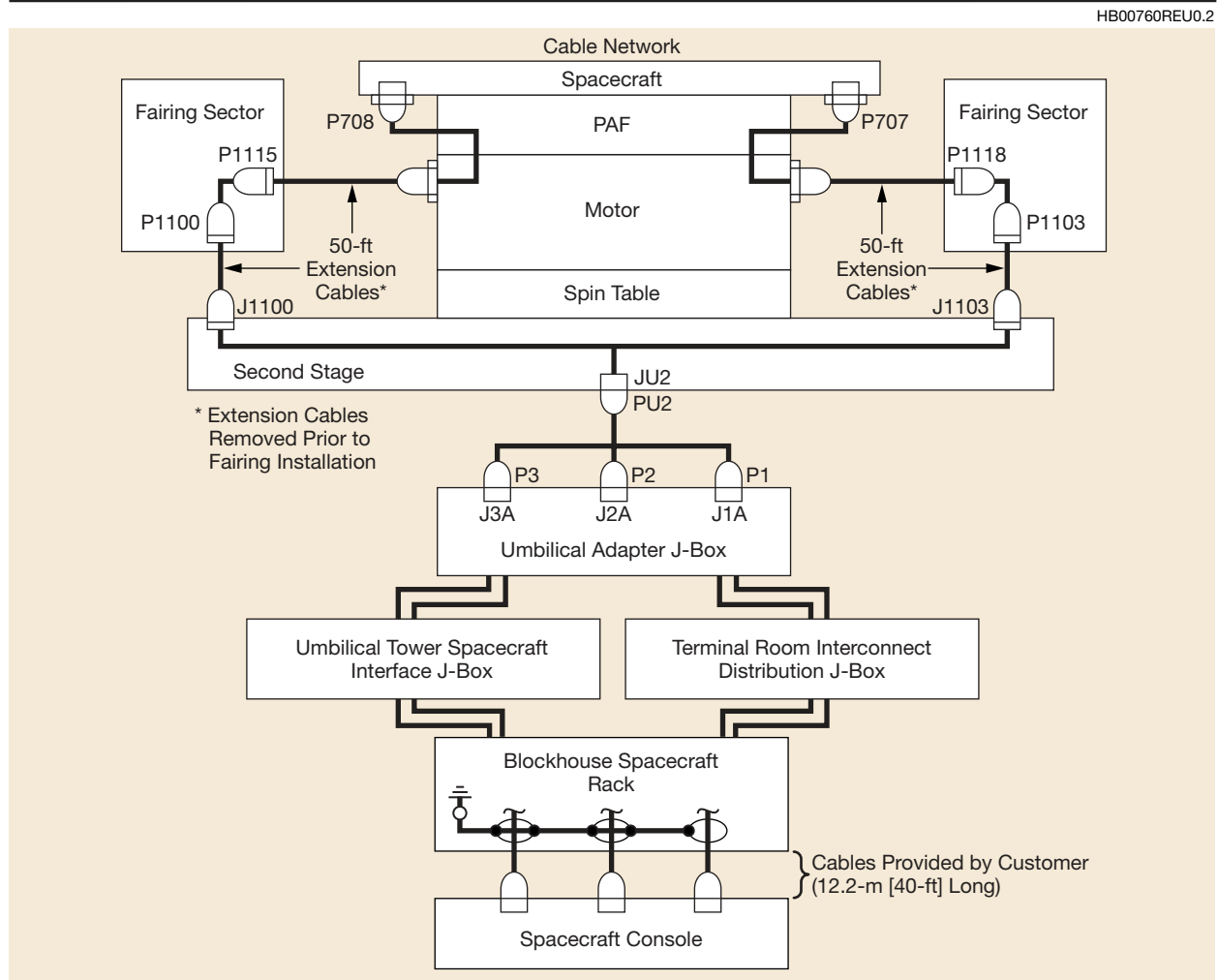


Figure 5-70. Typical Payload-to-Blockhouse Wiring Diagram for Three-Stage Missions at SLC-17

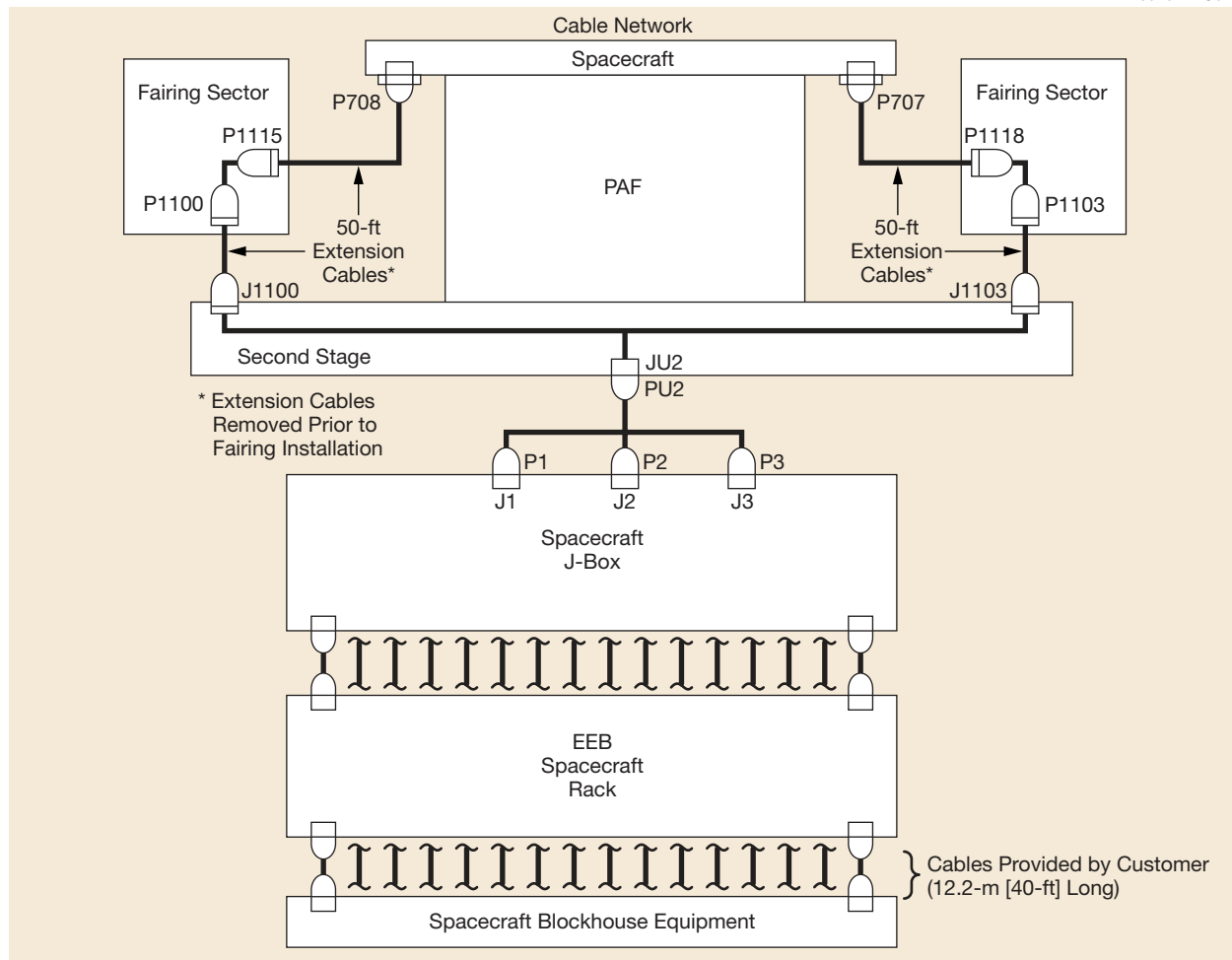


Figure 5-71. Typical Payload-to-EEB Wiring Diagram for Two-Stage Missions at SLC-2

Each wire in the baseline spacecraft-to-blockhouse wiring configuration has a current-carrying capacity of 6 A, wire-to-wire isolation of 50 megohms, and voltage rating of 600 VDC.

Typical one-way line resistance for any wire is shown in Table 5-5.

Table 5-5. Typical One-Way Line Resistance

Location	Function	No. of Wires	Fairing On*		Fairing Off**	
			Length (m/ft)	Resistance (ohm)	Length (m/ft)	Resistance (ohm)
CCAFS SLC-17A	Data/control	120	353/1157	3.4	384/1259	4.5
CCAFS SLC-17A	Data/control	24	365/1198	6.6	396/1300	7.8
CCAFS SLC-17A	Power	28	365/1198	1.2	396/1300	1.5
CCAFS SLC-17B	Data control	120	353/1157	4.5	384/1259	5.6
CCAFS SLC-17B	Data control	24	365/1198	6.6	396/1300	7.8
CCAFS SLC-17B	Data (TWINAX)	16	394/1293	13.1	425/1394	14.3
CCAFS SLC-17B	Power	28	365/1198	1.6	396/1300	1.9
VAFB SLC-2W EEB	Data/control	120	119/392	2.0	151/494	3.1
VAFB SLC-2W EEB	Data/(TWINAX)	16	119/392	4.3	151/494	5.5
VAFB SLC-2W EEB	Power	56	119/392	0.9	151/494	1.1

*Resistance values are for single wires between the fixed umbilical tower and the blockhouse

**Resistance values include fairing extension cable resistance

5.8.2 Spacecraft Umbilical Connectors

For spacecraft configurations in which the umbilical connectors interface directly with the payload attach fitting, the following connectors (conforming to MIL-C-81703) are recommended:

- MS3424E37-50S (flange-mount receptacle).
- MS3464E37-50S (jam nut-mount receptacle).

These connectors mate to a rack and panel mount interface connector (Deutsch part number D8179E37-0PN) on the payload attach fitting.

For spacecraft configurations in which the umbilical connectors interface directly with the fairing wire harnesses, the following connectors (conforming to MIL-C-81703) are recommended:

- MS3424E37-50S (flange-mount receptacle).
- MS3464E37-50S (jam nut-mount receptacle).

These connectors mate to a lanyard disconnect plug (Deutsch part number D8178E37-0PN) in the fairing.

The following alternative connectors, made by Deutsch and conforming to MIL-C-81703, may be used when spacecraft umbilical connectors interface with fairing-mounted wire harnesses or the payload attach fitting:

- D817*E61-OSN.
- D817*E37-OSN.

If “*” is 0, the receptacle is flange mounted; if 4, the receptacle is jam-nut mounted.

These connectors mate to a D8178E-series lanyard disconnect plug in the fairing or D8179E-series rack-and-panel plug on the PAF.

For spacecraft umbilical connectors that interface directly to the fairing wire harnesses, the spacecraft connector shall be installed so the polarizing key is in line with the longitudinal axis of the vehicle and facing forward (upward). The connector shall be within 5 deg of the fairing sector centerline. The face of the connector shall be within 2 deg of being perpendicular to the centerline. A typical spacecraft umbilical connector is shown in Figure 5-72. There should be no surrounding spacecraft intrusion within a 30-deg half-cone-angle separation clearance envelope at the mated fairing umbilical connector (Figure 5-73). Pull forces for the lanyard disconnect plugs are shown in Table 5-6. For spacecraft umbilical connectors interfacing with the PAF, the connector shall be installed so that the polarizing key is oriented radially outward. Spring compression and pin retention forces for the rack-and-panel connectors are shown in Table 5-7.

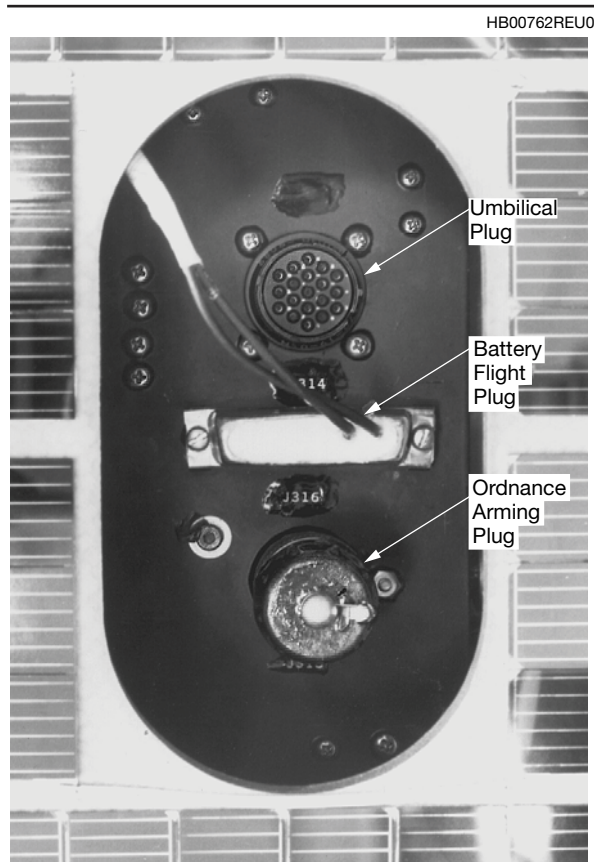


Figure 5-72. Typical Spacecraft Umbilical Connector

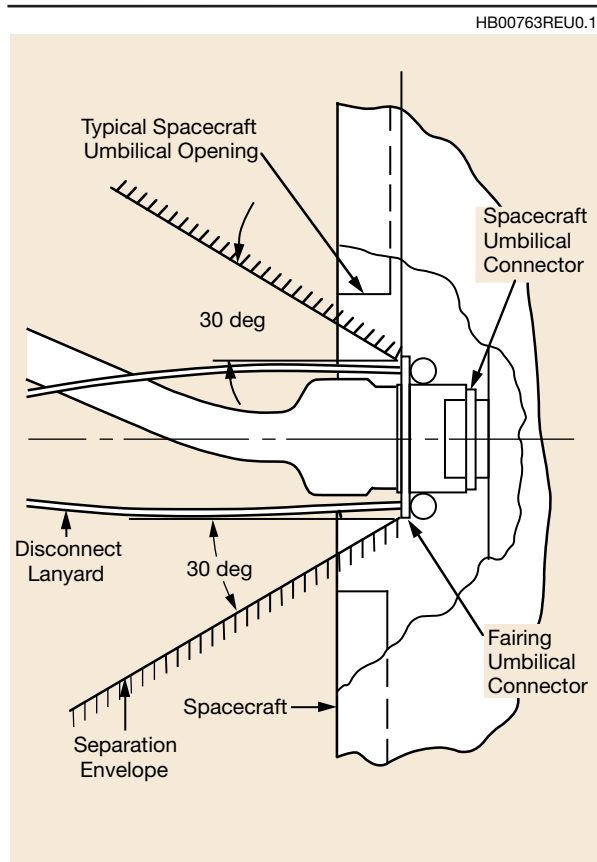


Figure 5-73. Spacecraft/Fairing Umbilical Clearance Envelope

Table 5-6. Disconnect Pull Forces (Lanyard Plugs)

Connector type	Shell size	Minimum force for disengagement		Maximum engagement and disengagement force	
		(lb)	(N)	(lb)	(N)
D817X	61	7.0	31.1	49.0	217.9
D817X	37	6.0	26.7	44.0	195.7

002253.2

Table 5-7. Disconnect Forces (Rack-and-Panel Connectors)

Connector type	Shell size	Maximum spring compression		Maximum pin retention	
		(lb)	(N)	(lb)	(N)
D817X	61	77	342.5	68	302.4
	37	48	213.5	50	222.4

002254.3

5.8.3 Spacecraft Separation Switch

To monitor vehicle/spacecraft separation, a separation switch can be installed on the spacecraft. The configuration must be coordinated with the Delta Program Office. This switch should be located to interface with the launch vehicle at the separation plane or within 25.4 mm (1 in.) below it. A special pad will be provided on the vehicle side of the interface. The design of the switch should provide for at least 6.4 mm (0.25 in.) over-travel in the mated condition. Typical spacecraft separation switch concepts are shown in Figure 5-74. The switch located over the separation spring is the preferred concept. An alternative for obtaining spacecraft separation indication is by the vehicle telemetry system.

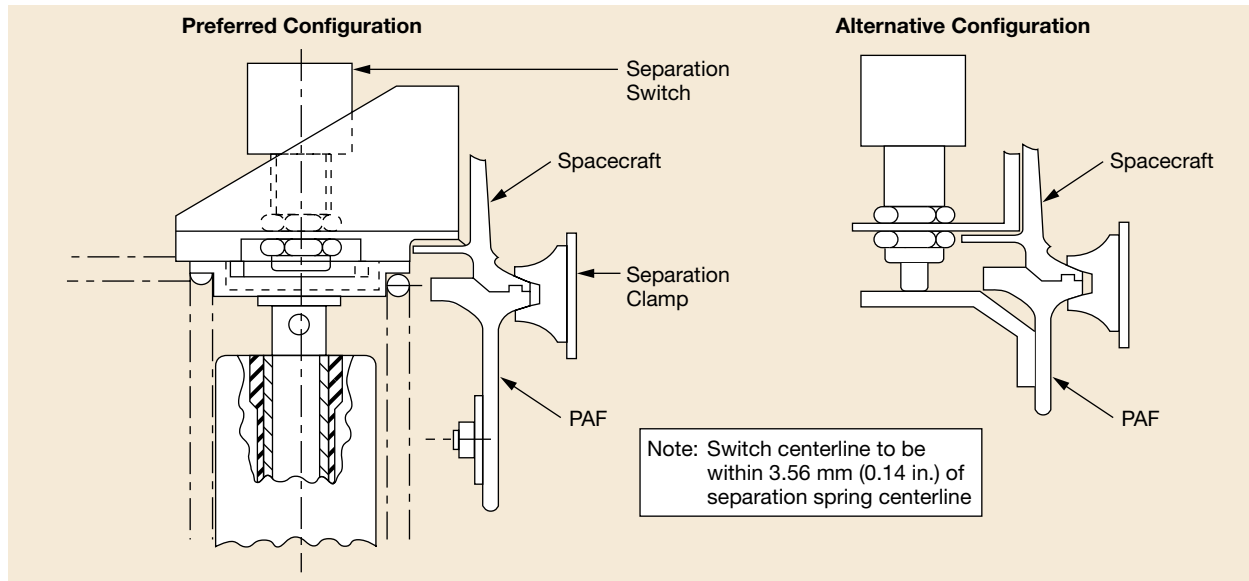


Figure 5-74. Typical Spacecraft Separation Switch and PAF Switch Pad

5.8.4 Spacecraft Safe and Arm Circuit

The spacecraft apogee motor S&A circuit (if applicable) must interconnect with the operations safety manager's console (OSMC) interface in the blockhouse or operations building. An interface diagram for the spacecraft console and the OSMC is given in Figure 5-75 for the existing blockhouse configuration and Figure 5-76 for the operations building configuration. Circuits for the S&A mechanism "arm permission" and the S&A talk-back lights are provided. This link is applicable at SLC-17 only and is not required at SLC-2.

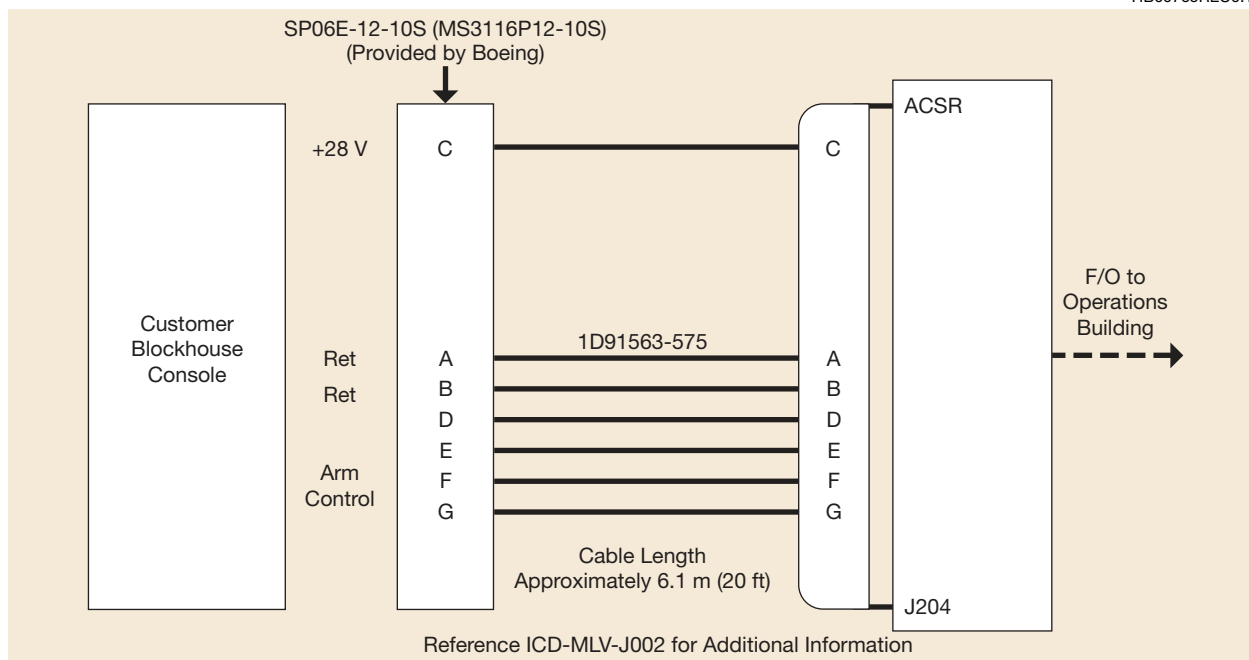
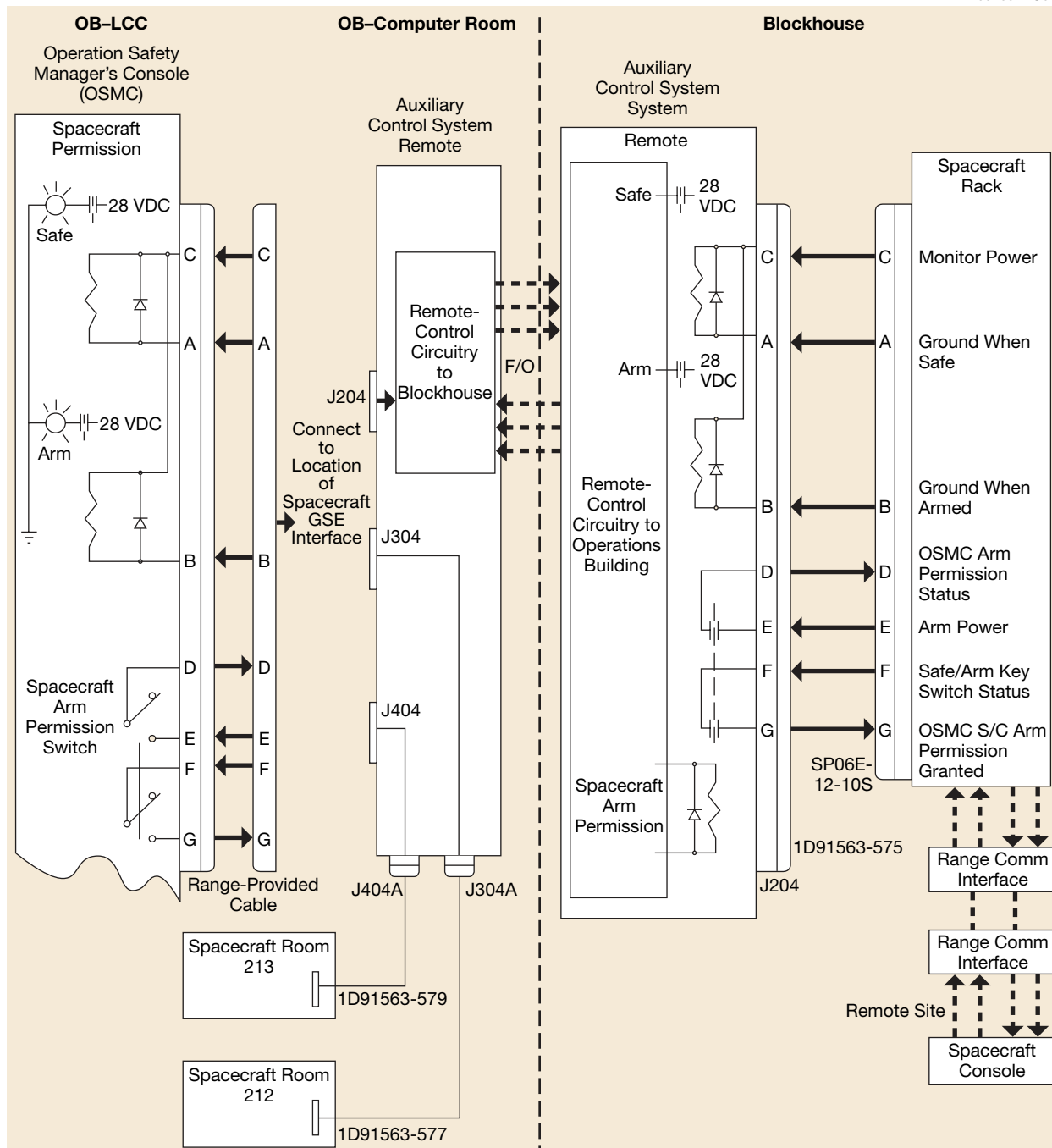


Figure 5-75. Blockhouse Spacecraft/Operation Safety Manager's Console Interface for SLC-17



- Pin A S&A Safe Position Status input to the OSMC – The presence of a Ground Indicates Safe position
- Pin B S&A Arm Position Status input to the OSMC – The presence of a Ground indicates Arm position
- Pin C Spacecraft manufacturer B/H Panel 28 VDC Monitor Power input to the OSMC
- Pin D Arm Permission Switch Position Status from OSMC – The presence of 28 VDC indicates Permission Granted
- Pin E Arming Power Switch input to the OSMC – The presence of 28 VDC indicates Spacecraft Blockhouse Console Arm Power Switch is On
- Pin F Safe/Arm Key Switch Position Status input to the OSMC – The presence of 28 VDC indicates Spacecraft Blockhouse Console Key Switch is in the Arm Position
- Pin G OSMC Arm Permission Command to Spacecraft – The presence of 28 VDC Arms the Spacecraft Blockhouse S&A

Figure 5-76. Spacecraft/Pad Safety Console Interface for SLC-17—Operations Building Configuration

Section 6
LAUNCH OPERATIONS AT EASTERN RANGE

This section presents a description of Delta launch vehicle operations associated with Space Launch Complex 17 (SLC-17) at the Cape Canaveral Air Force Station (CCAFS), Florida. Delta II prelaunch processing and spacecraft operations conducted prior to launch are presented.

6.1 ORGANIZATIONS

The Delta Program operates the Delta launch system and maintains a team that provides launch services to NASA, USAF, and commercial customers at CCAFS. The Delta Program provides the interface to the Department of Transportation (DOT) for the licensing and certification needed to launch commercial spacecraft using the Delta II. The Delta Program also has an established working relationship with Astrotech Space Operations that owns and operates a processing facility for commercial spacecraft in Titusville, Florida, in support of Delta missions. Utilization of these facilities and services is arranged by the Delta Program Office for the customer.

The Delta Program interfaces with NASA at Kennedy Space Center (KSC) through the Launch Services Program Office. NASA designates a launch site integration manager who arranges all of the support requested from NASA for a launch from CCAFS. The Delta Program Office has an established interface with the USAF Space and Missile Systems Center (USAF SMC) Delta II program office and the 45th Space Wing Directorate of Plans. The USAF designates a program support manager (PSM) to be a representative of the 45th Space Wing. The PSM serves as the official interface for all support and services requested. These services include range instrumentation and facilities/equipment operation and maintenance as well as safety, security, and logistics support. Requirements are described in documents prepared using the government's universal documentation system format. The Delta Program Office formally submits these documents to government agencies. The Delta Program Office and the customer generate the program requirements document (PRD).

The organizations that support a commercial launch are shown in Figure 6-1. A spacecraft coordinator from the Delta-CCAFS launch team is assigned early in the integration effort. The spacecraft coordinator will assist the spacecraft team during the launch campaign by helping to obtain safety approval of the spacecraft test procedures and operations, integrating the spacecraft operations into the launch vehicle activities, and serving as the interface between the spacecraft and test conductor in the launch control center during the countdown and launch.

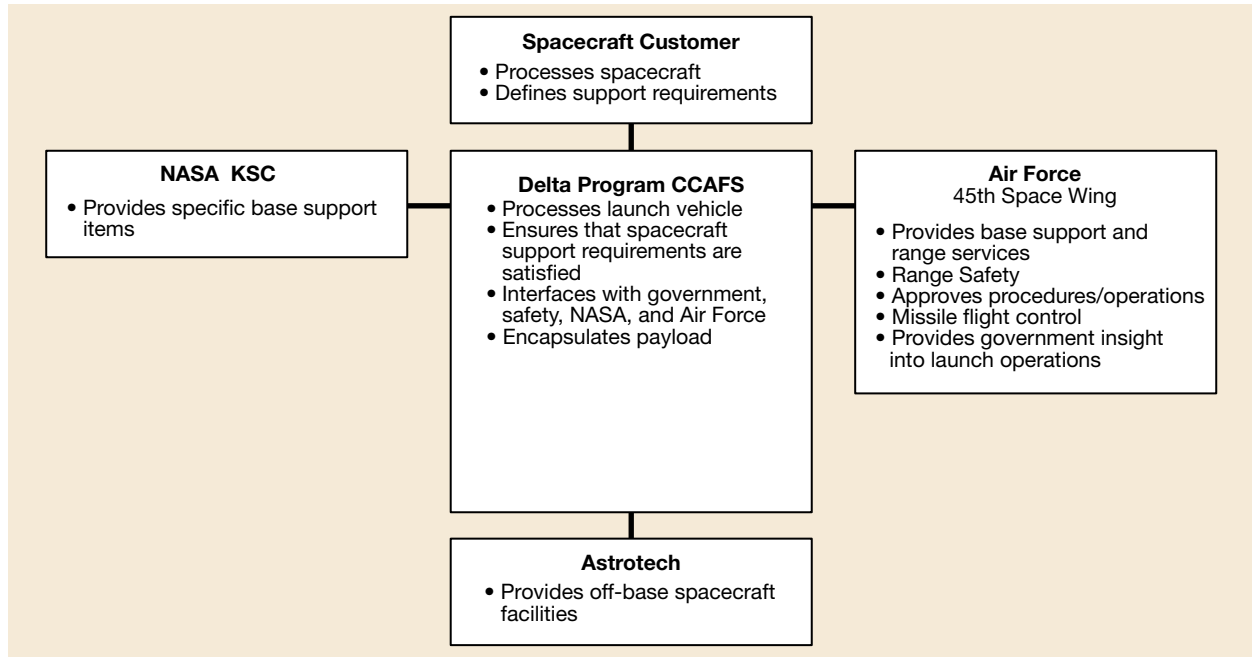


Figure 6-1. Organizational Interfaces for Commercial Users

6.2 FACILITIES

Commercial spacecraft will normally be processed through the Astrotech facilities. Other facilities on CCAFS, controlled by NASA and USAF, can be used for commercial spacecraft under special circumstances.

The spacecraft agency must provide its own test equipment for spacecraft preparations, including telemetry receivers and telemetry ground stations. Communications equipment, including some antennas, is available as base equipment for voice and data transmissions.

Transportation and handling of the spacecraft and associated equipment are provided by Boeing from the spacecraft processing facilities to the launch site. Equipment and personnel are also available for loading and unloading operations. Shipping containers and handling fixtures attached to the spacecraft are provided by the spacecraft agency.

Shipping and handling of hazardous materials, such as electro-explosive devices (EEDs) and radioactive sources, are the responsibility of the customer and must be in accordance with applicable regulations. It is the responsibility of the customer to identify these items and become familiar with such regulations; included are those imposed by NASA, USAF, and FAA (refer to Section 9).

6.2.1 Astrotech Space Operations Facilities

The Astrotech facility is located approximately 5.6 km (3 mi) west of the Gate 3 entrance to KSC near the intersection of state roads 405 and 407 in the Spaceport Industrial Park in Titusville, Florida (Figure 6-2). A complete description of the Astrotech facilities can be found on the Astrotech Web site: www.spacehab.com/aso/reference.htm.

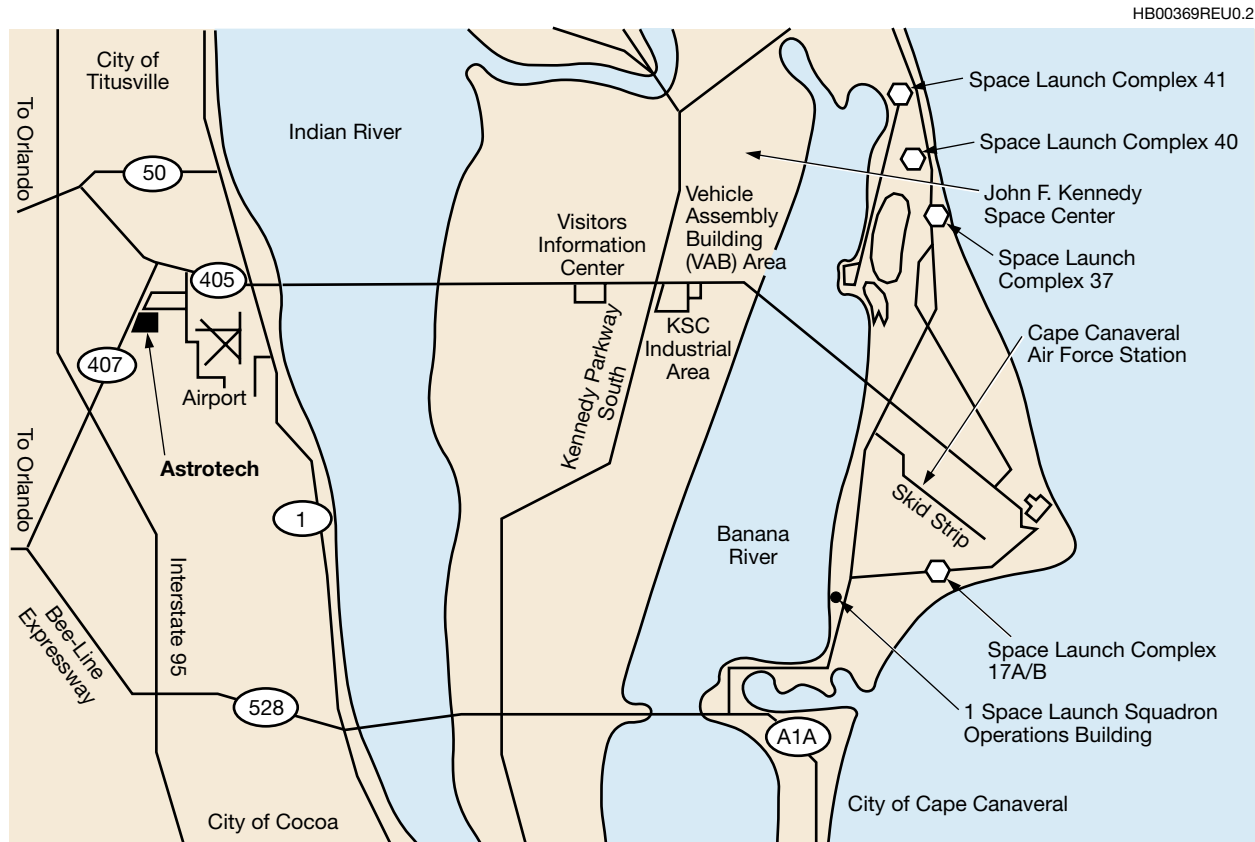


Figure 6-2. Astrotech Site Location

6.2.2 CCAFS Operations and Facilities

Commercial customers have the use of facilities and services on CCAFS based on “capacity available.” Typically, these facilities and services are arranged by Astrotech. Civil and military payloads arrange for the use of facilities and services through their sponsoring agencies. Typical areas used by commercial customers are described in the following paragraphs.

6.2.2.1 Mission Director Center (MDC). Launch operations and overall mission activities are monitored by the Mission Director (MD) and the supporting mission management team in the MDC (Figure 6-3) in building AE, where the team is informed of launch vehicle, spacecraft, and tracking network flight readiness. Appropriate real-time prelaunch and launch data are displayed to provide a presentation of vehicle launch and flight progress. During launch operations, the

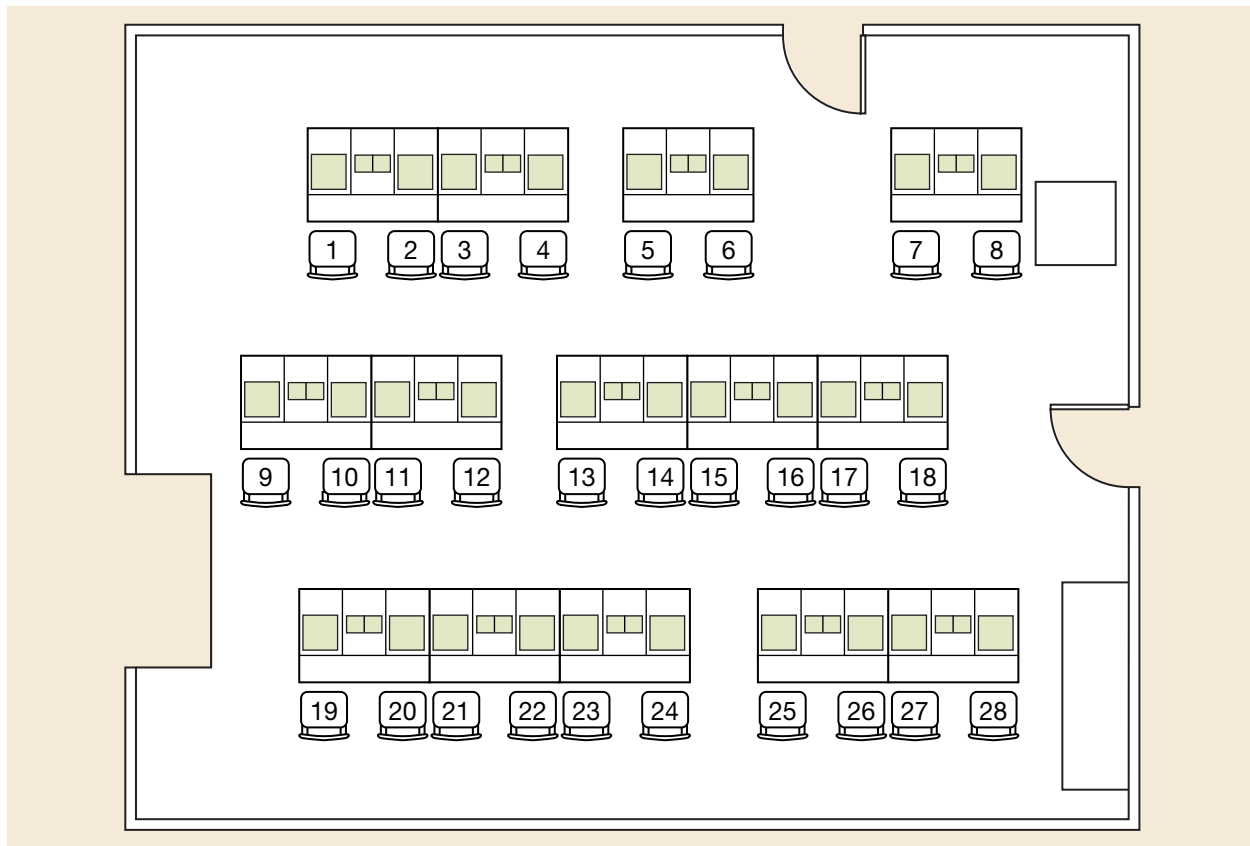


Figure 6-3. Building AE Mission Director Center

MDC also functions as an operational communications center from which all communication emanates to tracking and control stations.

At the front of the MDC are large illuminated displays that list the tracking stations and range stations in use and the sequence of events after liftoff. These displays are used to show present position and instantaneous impact point (IIP) plots. When compared to the theoretical plots, these displays give an overall representation of launch vehicle performance.

6.2.2.2 Solid-Propellant Storage Area. The facilities and support equipment in this area are maintained and operated by USAF range contractor personnel. Ordnance item transport is also provided by range contractor personnel. Preparation of ordnance items for flight (e.g., S&A device installation, thermal blanket installation) is performed by spacecraft contractor personnel according to range safety-approved procedures.

6.2.2.3 Storage Magazines. Storage magazines are concrete bunker-type structures located at the north end of the storage area. Only two of the magazines are used for spacecraft ordnance. One magazine, designated MAG H, is environmentally controlled to $23.9^{\circ} \pm 2.8^{\circ}\text{C}$ ($75^{\circ} \pm 5^{\circ}\text{F}$) with a maximum relative humidity of 65%. This magazine contains small ordnance items such as S&A devices, igniter assemblies, initiators, bolt cutters, and electrical squibs.

The second magazine, designated MAG I, is used for the storage of solid-propellant motors. It is environmentally controlled to $29.4^{\circ} \pm 2.8^{\circ}\text{C}$ ($85^{\circ} \pm 5^{\circ}\text{F}$) with a maximum relative humidity of 65%.

6.2.2.4 Electrical-Mechanical Testing Facility. The electrical-mechanical testing facility (EMT) (Figure 6-4), which is operated by range contractor personnel, is used for such functions as ordnance item bridgewire resistance checks and S&A device functional tests, as well as for test- firing small self-contained ordnance items.

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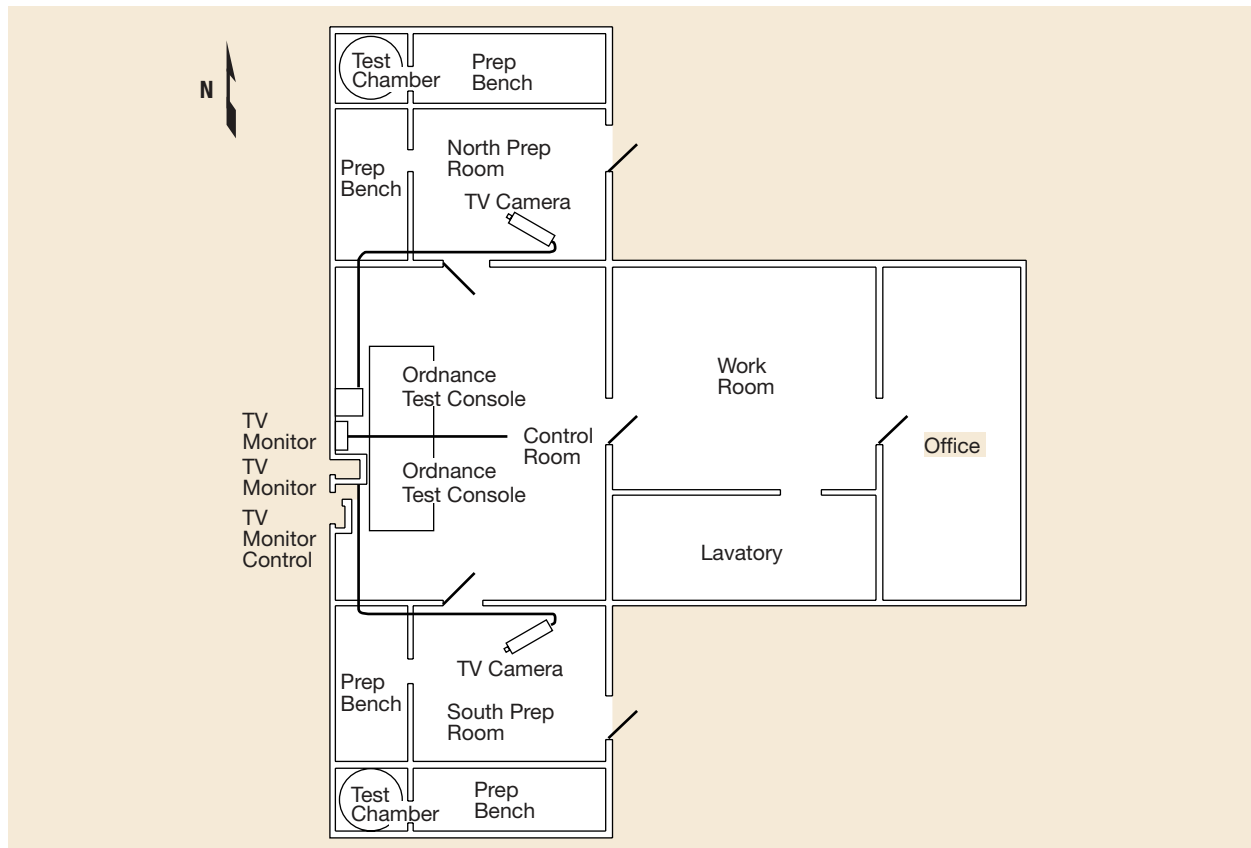


Figure 6-4. Electrical-Mechanical Testing Building Floor Plan

Electrical cables that provide the interface between the ordnance items and the test equipment already exist for most devices commonly used at CCAFS. These cables are tested before each use, and the test data are documented. If no cable or harness exists for a particular ordnance item, it is the responsibility of the spacecraft contractor to provide the proper mating connector for the ordnance item to be tested. A six-week lead time is required for cable fabrication.

The test consoles contain the items listed in Table 6-1. The tests are conducted according to spacecraft contractor procedures that have been approved by range safety personnel.

Table 6-1. Test Console Items

Resistance measurement controls	Alinco bridge and null meter
Digital current meter	Resistance test selector
Digital voltmeter	Digital ammeter
Auto-ranging digital voltmeter	Digital stop watch
Digital multimeter	Relay power supply
High-current test controls	Test power supply
Power supply (5 V)	Power control panel
High-current test power supply	Blower

002136

6.2.2.5 Liquid-Propellant Storage Area. Spacecraft contractor-provided liquid propellants can be stored in the liquid-propellant storage area on CCAFS. This climate-controlled area, operated by range contractor personnel, can store both fuel and oxidizer in Department of Transportation (DOT)-approved containers. Propellant servicing equipment can be cleaned/decontaminated in this area.

6.3 SPACECRAFT TRANSPORT TO LAUNCH SITE

After completion of spacecraft preparations and mating to the PAF in one of the payload processing facilities (PPFs) or hazardous processing facilities (HPFs), the flight-configured spacecraft is moved to SLC-17 to join with the Delta II launch vehicle. Boeing provides a mobile handling container to support spacecraft transfer to the launch pad.

The spacecraft handling container (Figure 6-5) is supported on a foam-filled, rubber-tired transporter and slowly towed to the pad with a Delta Program-provided tractor. The container (commonly called the handling can) can be configured for either two- or three-stage missions. The handling can height varies according to the number of cylindrical sections required for a safe envelope around the spacecraft. The spacecraft container is purged with GN₂ to reduce the relative humidity of the air inside the container and to maintain a slight positive pressure. When transporting the spacecraft, container temperature is not controlled directly but is maintained at acceptable levels by selecting the time of day when movement occurs. The transportation environment is monitored with recording instrumentation.

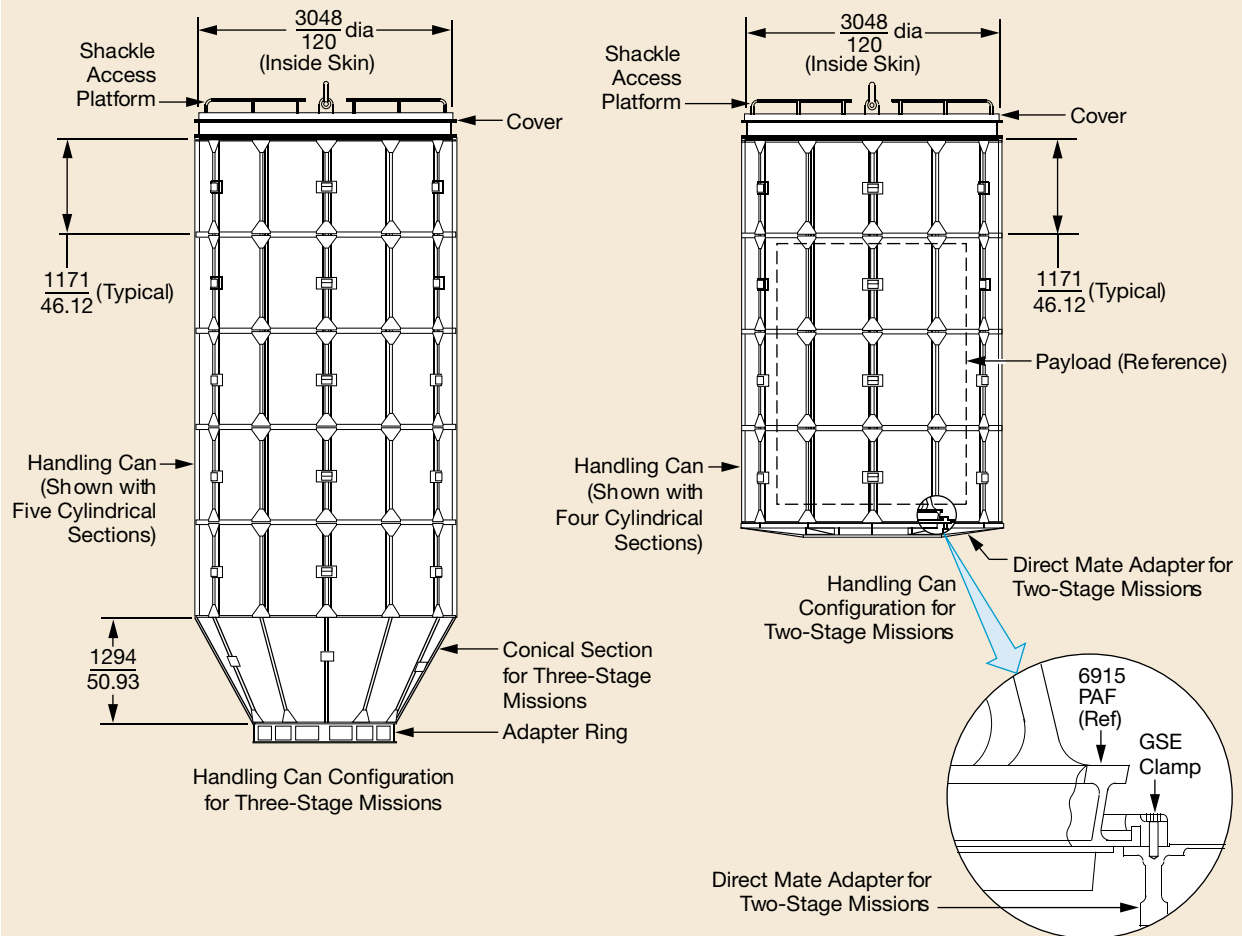
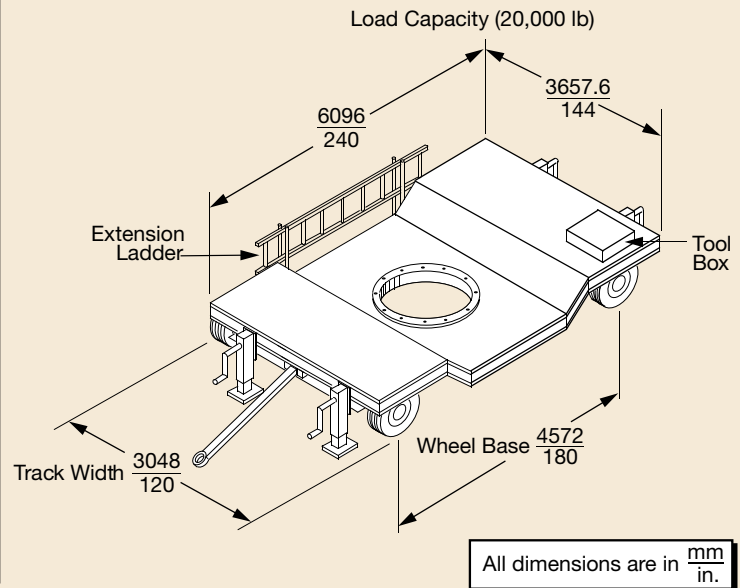
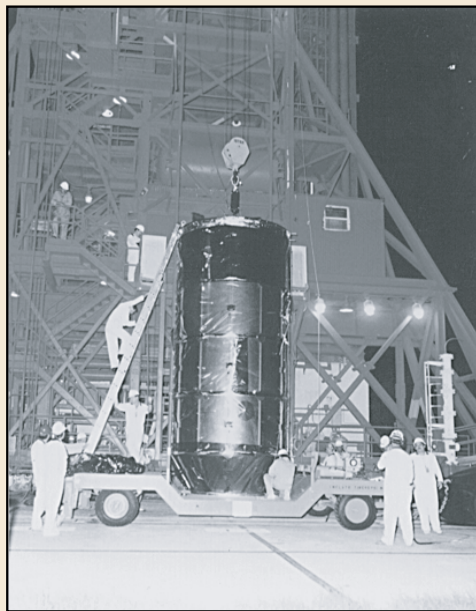


Figure 6-5. Delta II Upper-Stage Assembly Ground-Handling Can and Transporter

6.4 SLC-17, PADS A AND B (CCAFS)

SLC-17 is located in the southeastern section of CCAFS (Figure 6-6). It consists of two launch pads (17A and 17B), a blockhouse, ready room, shops, and other facilities needed to prepare, service, and launch the Delta II vehicle. The arrangement of SLC-17 is shown in Figure 6-7 and an aerial view in Figure 6-8.

Because all operations in the launch complex area involve or are conducted in the vicinity of liquid or solid propellants and explosive ordnance devices, the number of personnel permitted in the area, safety clothing to be worn, types of activities permitted, and equipment allowed are strictly regulated. Adherence to all safety regulations specified in Section 9 is required. Safety briefings on these subjects are given for those required to work in the launch complex area.

A clothing change room is provided on the mobile service tower (MST) level 9 in accordance with typical payload contamination guidelines.

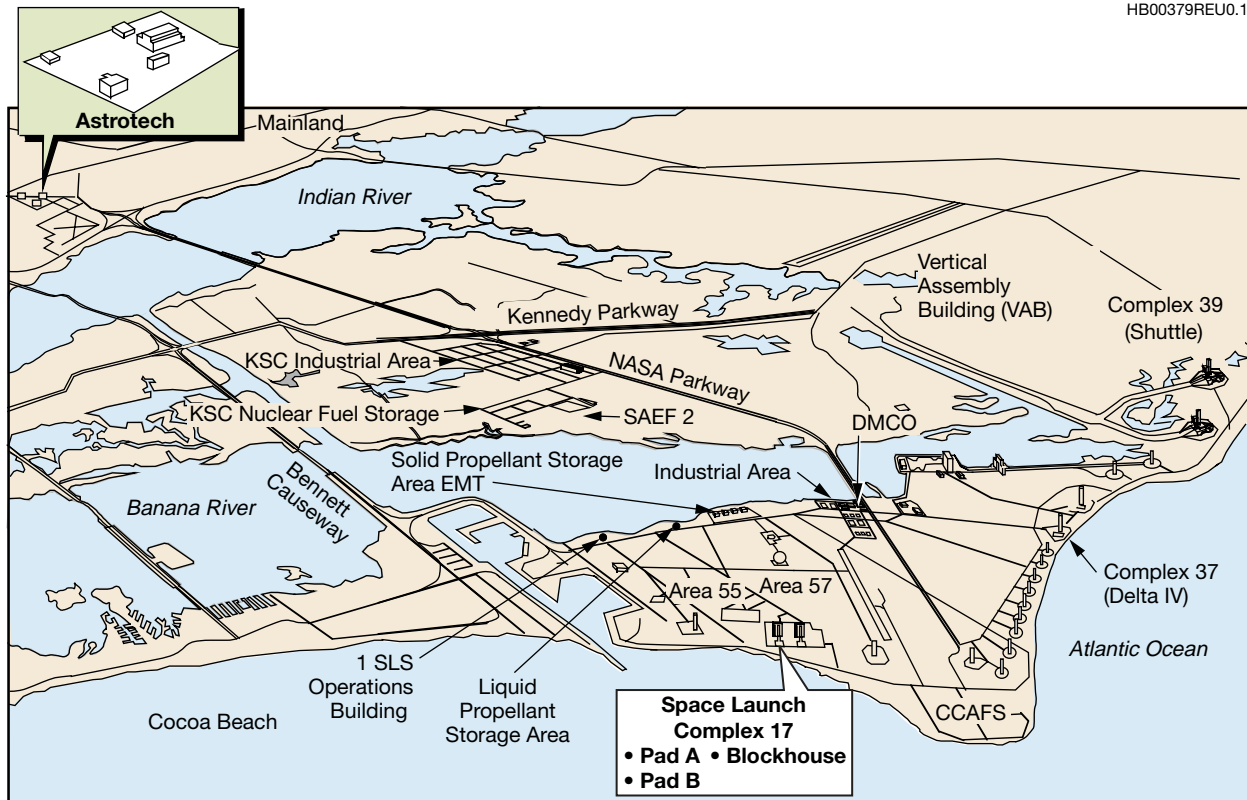


Figure 6-6. Delta Checkout Facilities

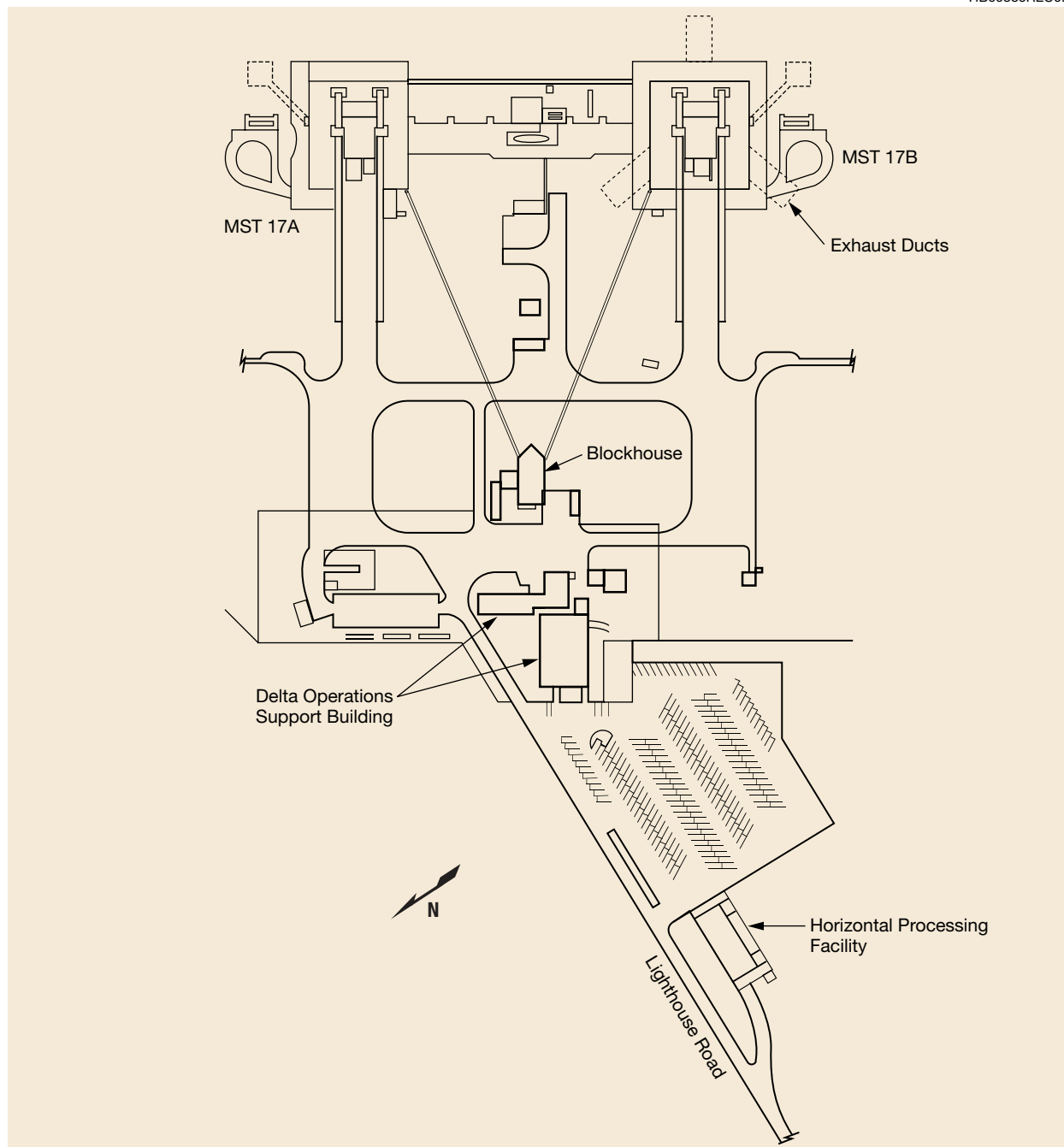


Figure 6-7. Space Launch Complex-17, Cape Canaveral Air Force Station



Figure 6-8. Space Launch Complex 17—Aerial View

6.4.1 MST Spacecraft Work Levels

The number of personnel admitted to the MST is governed by safety requirements and by the limited amount of work space on the spacecraft levels. The relationship of the vehicle to the MST is shown in Figure 6-9. Typical MST deck-level floor plans of pads 17A and 17B are shown in Figures 6-10A, 6-10B, 6-11A, 6-11B, 6-12A, and 6-12B.

6.4.2 Space Launch Complex 17 Blockhouse

Most hazardous operations, including launch, are no longer controlled from the SLC-17 blockhouse but are controlled from the 1st Space Launch Squadron Operations Building (1 SLS OB). The SLC-17 blockhouse remains and has floor space allocated for remotely controlled spacecraft consoles and battery-charging equipment. Terminal board connections in the spacecraft-to-block-house junction box (Figure 6-11) provide electrical connection to the spacecraft umbilical wires. If desired, the Delta Program will terminate the cables for the customer. Spacecraft umbilical wires should be tagged with the terminal board location identified, as indicated in the payload-to-blockhouse wiring diagram provided by the Delta Program in the interface control document.

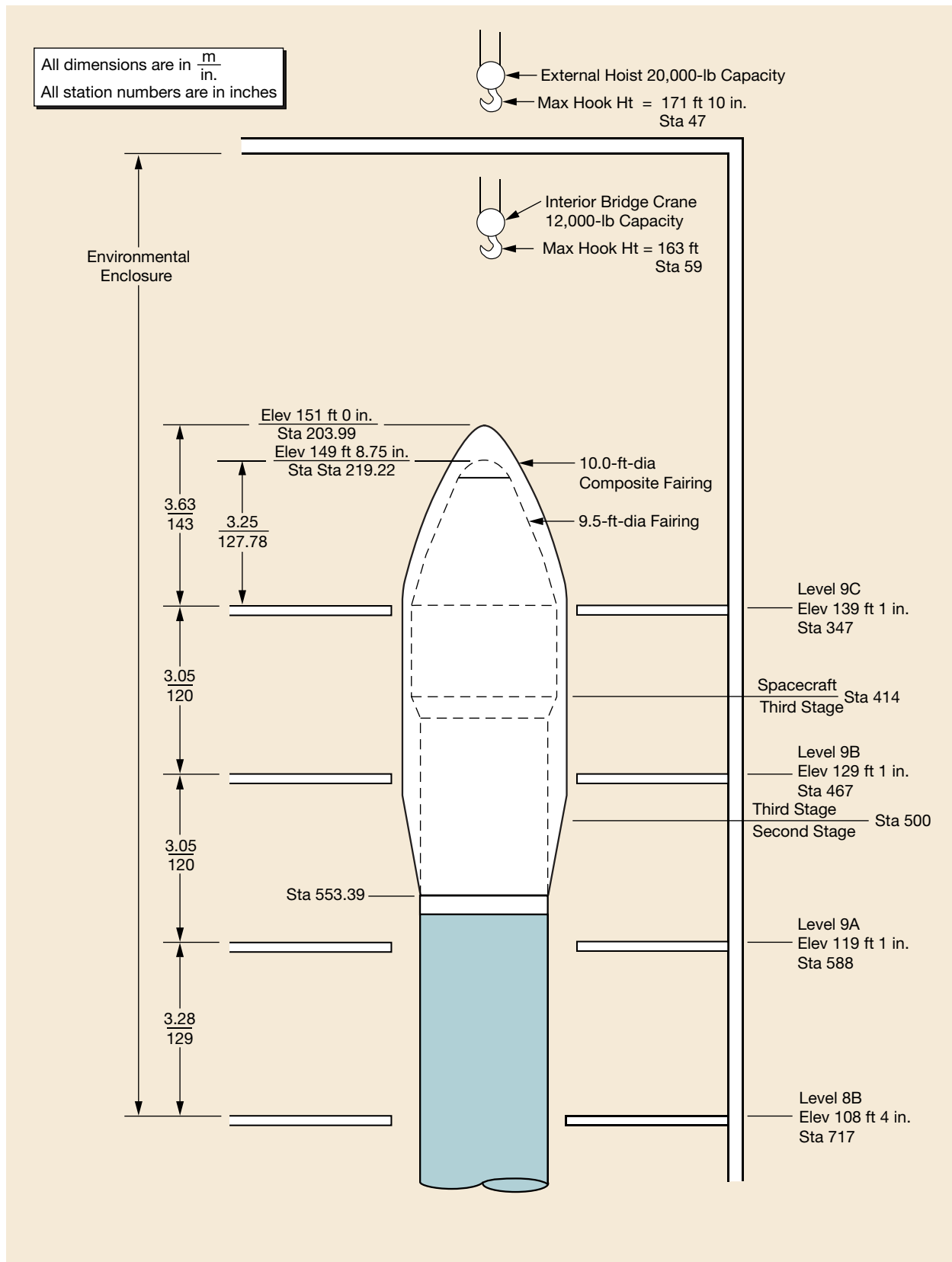


Figure 6-9. Environmental Enclosure Work Levels

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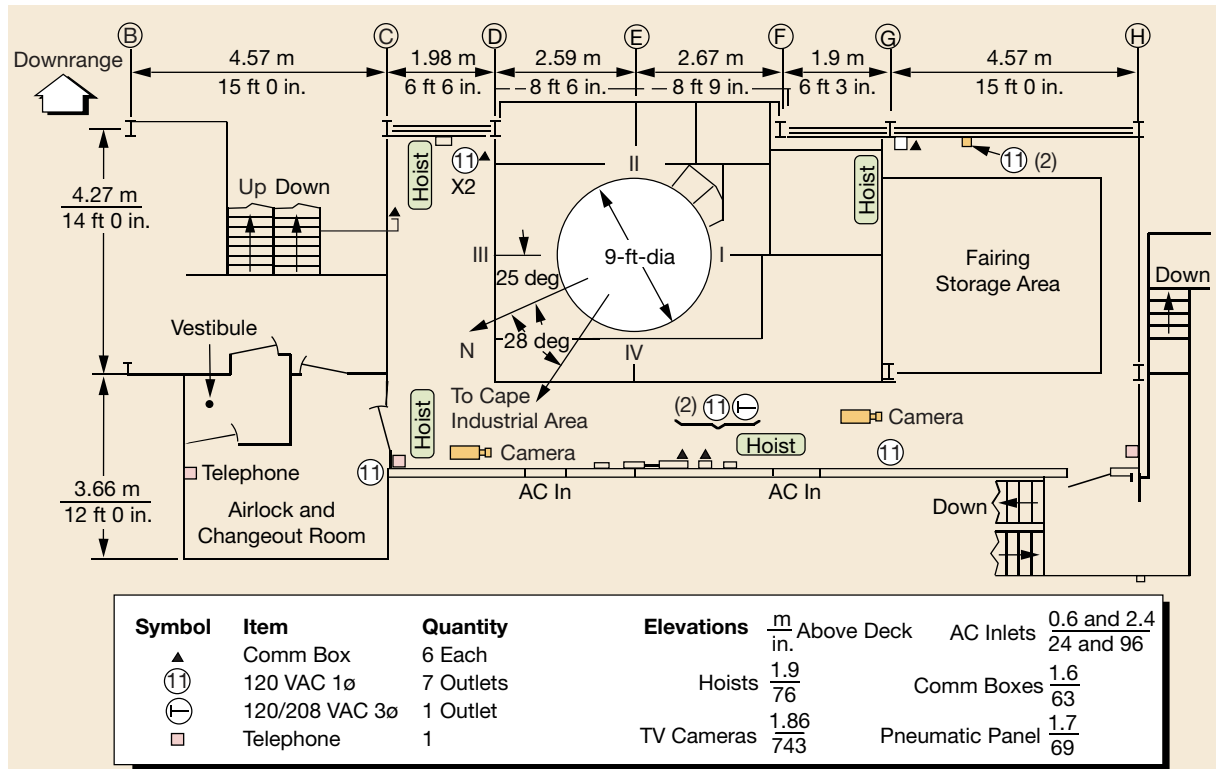


Figure 6-10A. Level 9A Floor Plan, Pad 17A

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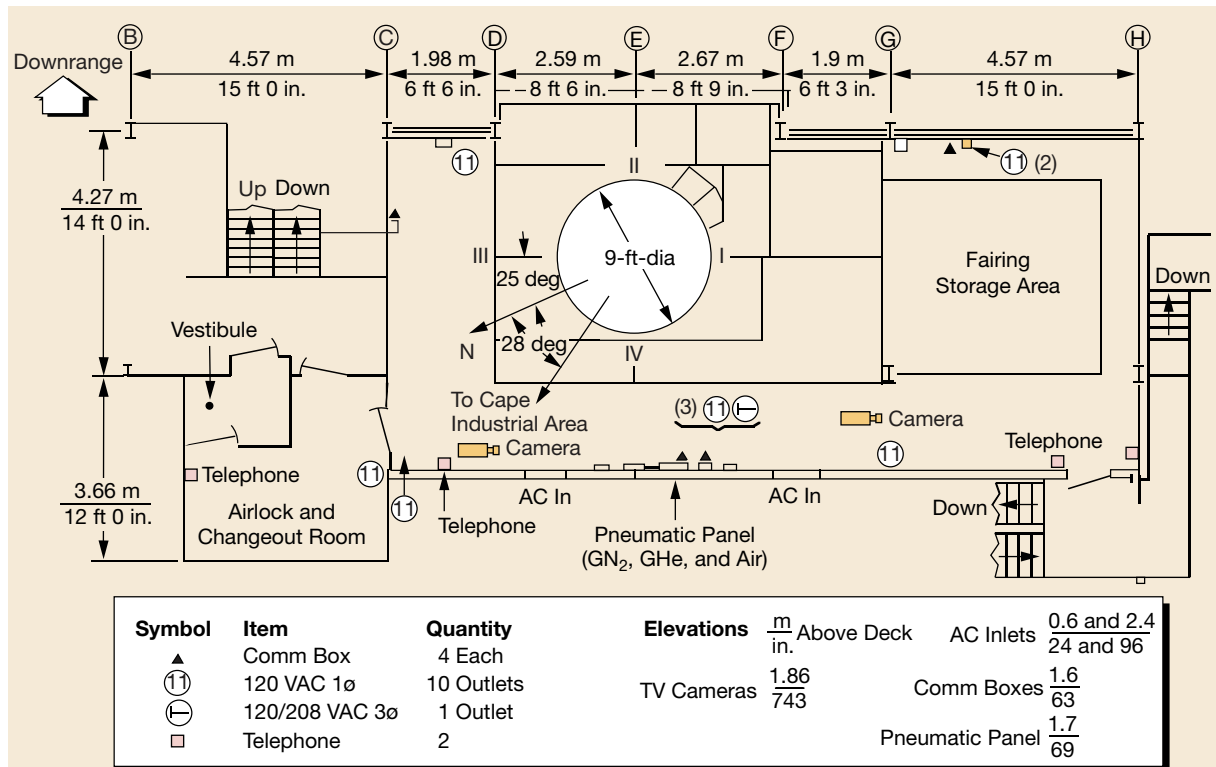


Figure 6-10B. Level 9A Floor Plan, Pad 17B

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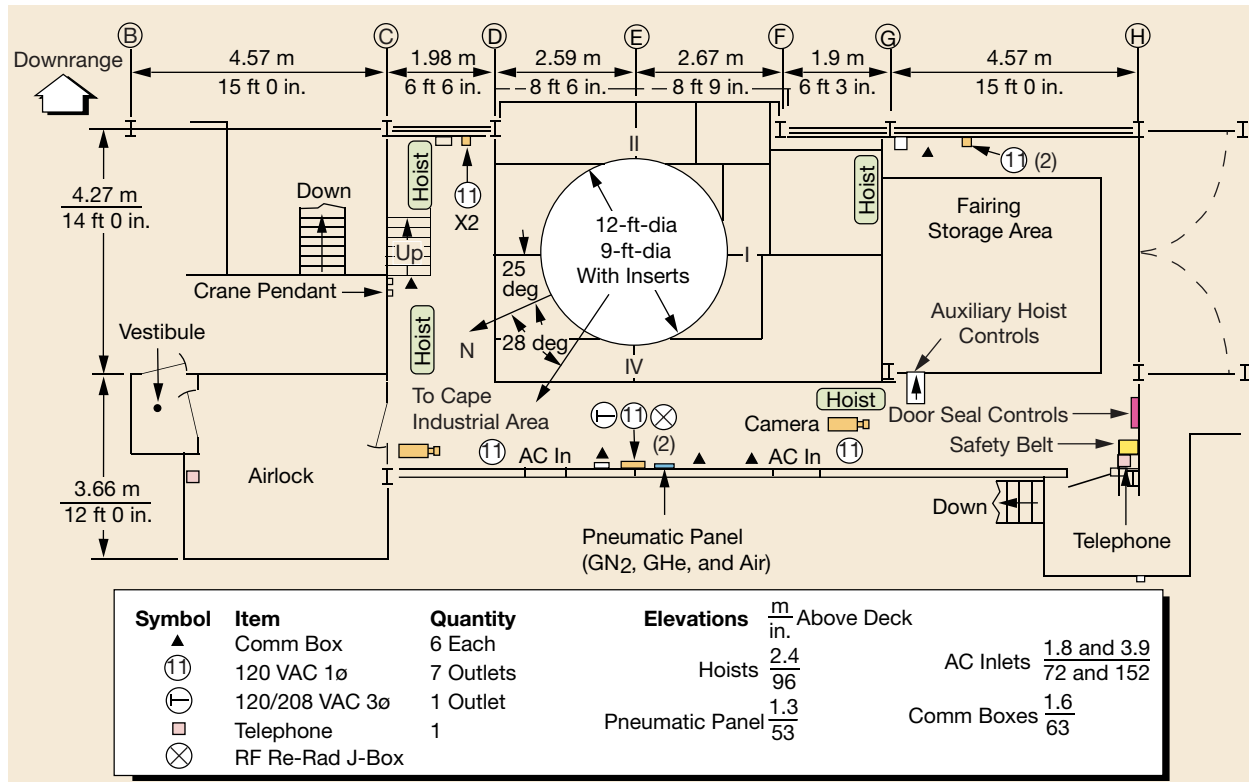


Figure 6-11A. Level 9B Floor Plan, Pad 17A

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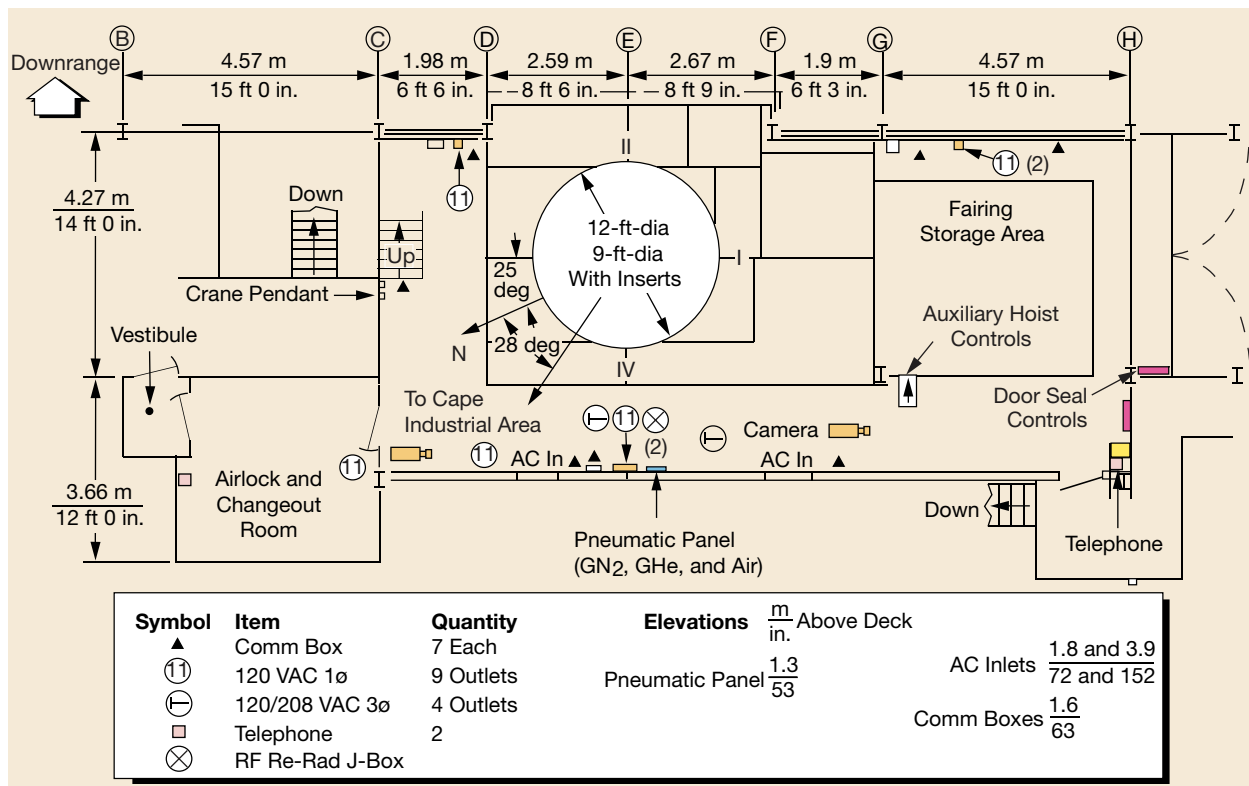


Figure 6-11B. Level 9B Floor Plan, Pad 17B

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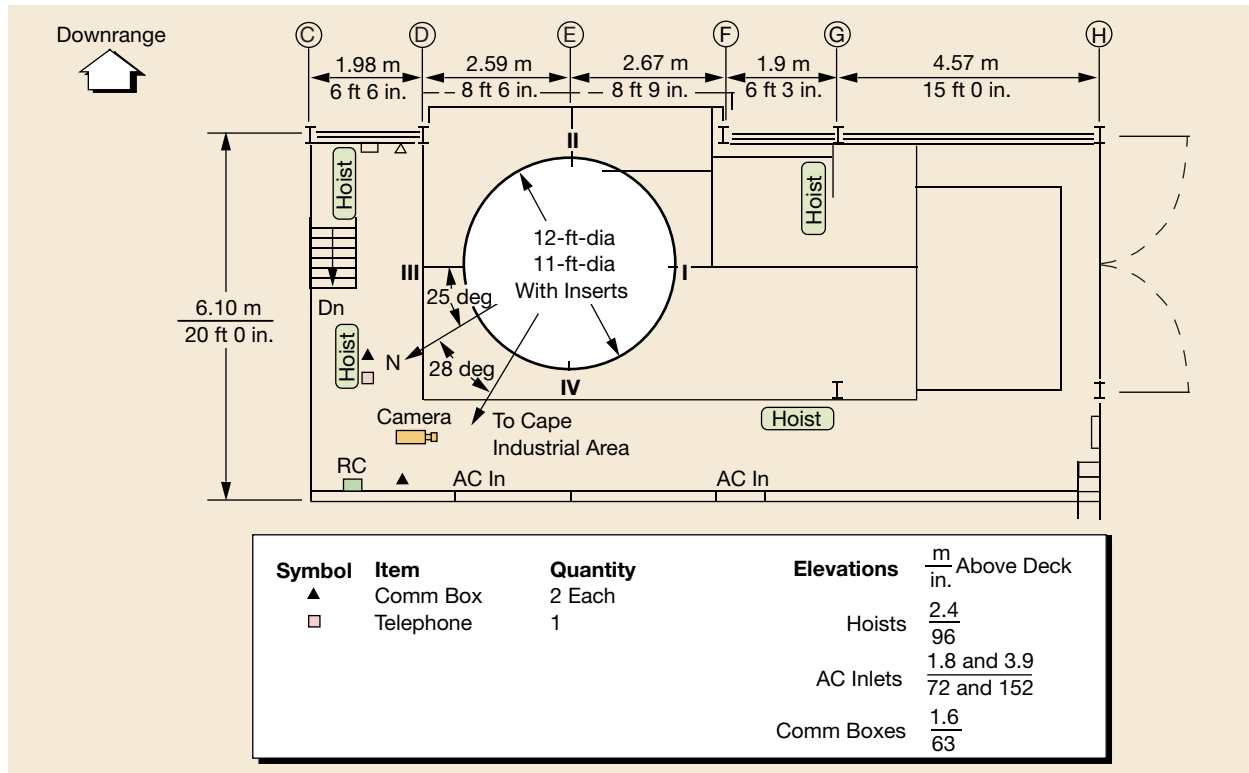


Figure 6-12A. Level 9C Floor Plan, Pad 17A

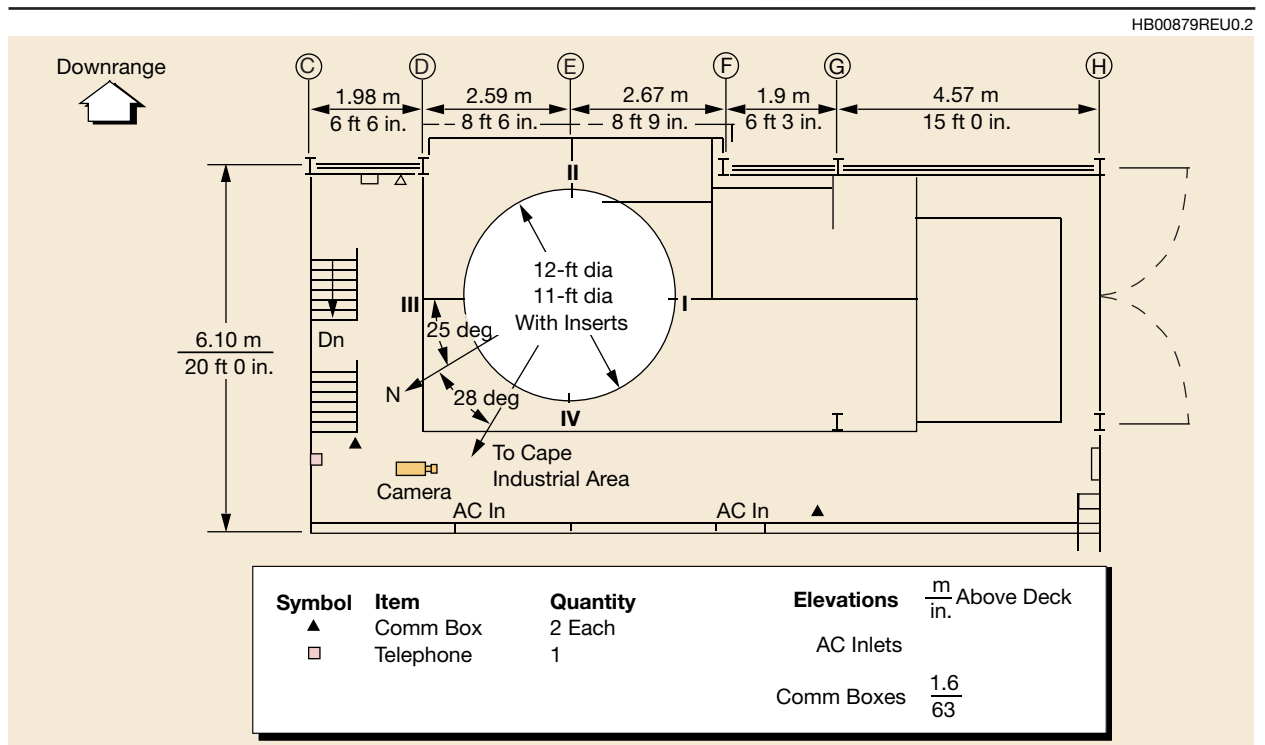


Figure 6-12B. Level 9C Floor Plan, Pad 17B

6.4.3 First Space Launch Squadron Operations Building (1 SLS OB)

All launch operations are controlled from the Launch Control Center (LCC) on the second floor of the 1 SLS OB. The launch vehicle and GSE are controlled and monitored from the OB via the advanced launch vehicle control system (ALCS). Also on the second floor, two spacecraft control rooms and office space adjacent to the LCC are available during processing and launch (Figure 6-13). Communication equipment, located in each control room, provides signal interface between the 1 SLS OB and the blockhouse (Figure 6-14). Standard bus interfaces (i.e., EIA-422, RS-485, EIA-232, and Ethernet) will be available for remote spacecraft equipment monitoring and control.

The remote spacecraft rack also provides limited discrete control/feedback and handles analog data from the blockhouse to the OB.

Provisions are made to interface the spacecraft safe and arm status and arm permission to the range operations safety manager's (OSM) console at the Auxiliary Control System Rack (ACSR) in the blockhouse and from OB spacecraft control rooms 1 and 2. The spacecraft interface with the OSM console is defined in Boeing ICD-MLV-J002.

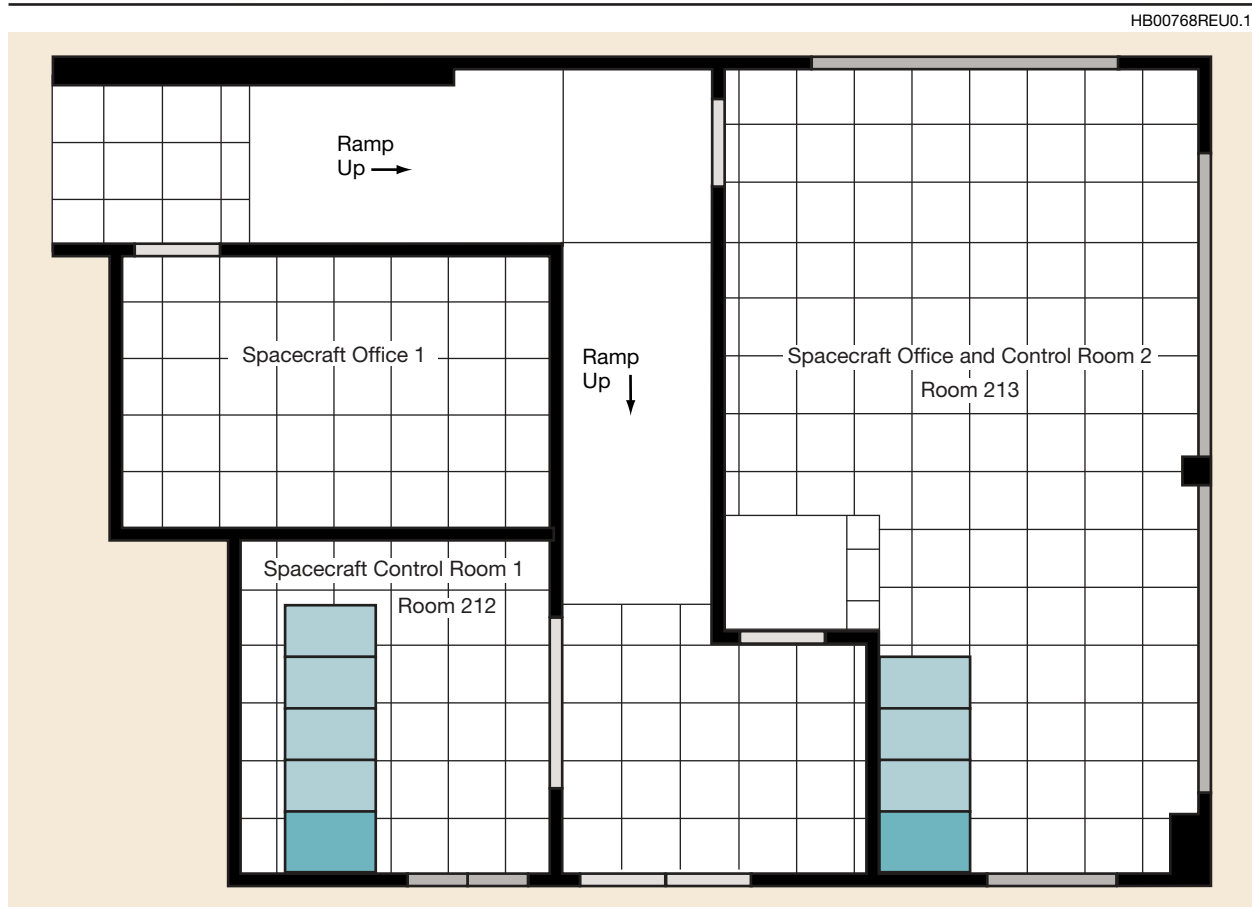


Figure 6-13. Spacecraft Customer Accommodations—Launch Control Center

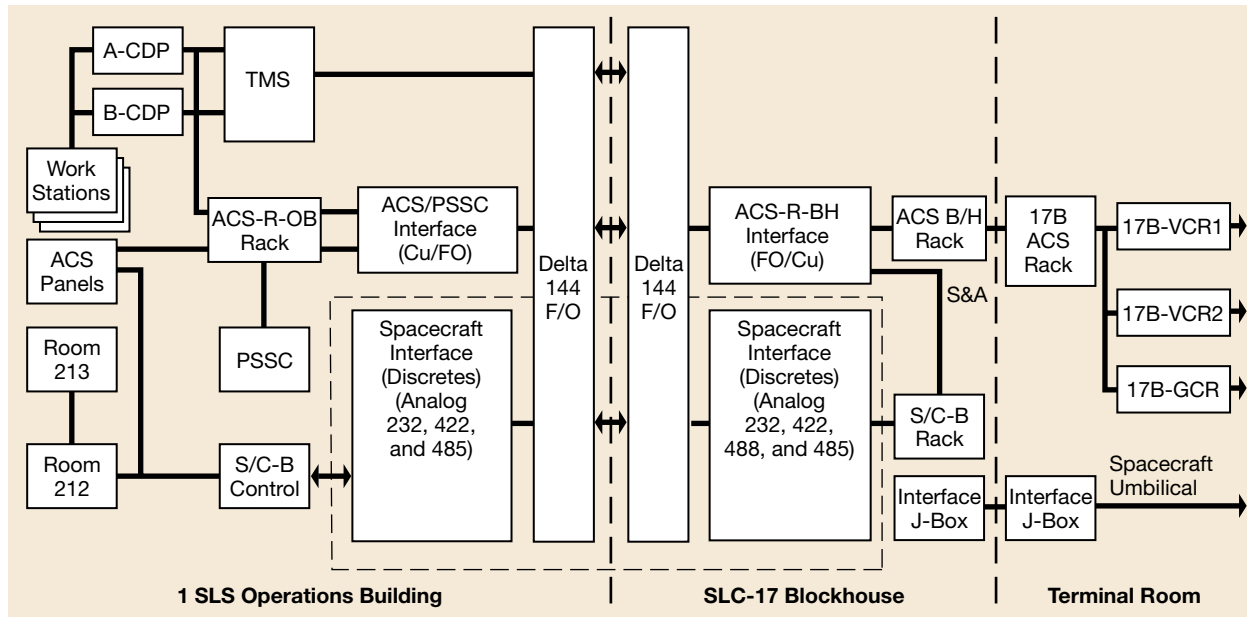


Figure 6-14. Interface Overview—Spacecraft Control Rack in 1 SLS Operations Building

6.5 SUPPORT SERVICES

6.5.1 Launch Support

For countdown operations, the Delta Program launch team is located in the 1 SLS OB engineering support area (ESA) and Hangar AE, with support from many other organizations. The following paragraphs describe the organizational interfaces and the launch decision process.

6.5.1.1 Mission Director Center (Hangar AE). The Mission Director Center provides the necessary seating, data display, and communications to control the launch process. Seating is provided for key personnel from the Delta Program Office, the Eastern Range, and the spacecraft control team. For NASA launches, key NASA personnel also occupy space in the Mission Director Center. Government launches incorporate additional reporting and decision responsibility.

6.5.1.2 Launch Decision Process. The launch decision process is conducted by the appropriate management personnel representing the spacecraft, the launch vehicle, and the range. Figure 6-15 shows the typical communication flow required to make the launch decision. For NASA missions, a Mission Director, launch management advisory team, engineering team, and quality assurance personnel will also participate in the launch decision process.

6.5.2 Weather Constraints

6.5.2.1 Ground-Wind Constraints. The Delta II vehicle is enclosed in the MST until approximately L-7 hours. The tower protects the vehicle from ground winds. The winds are measured using anemometers at the 9.1-m (30-ft) and 28.0-m (92-ft) levels of the tower.

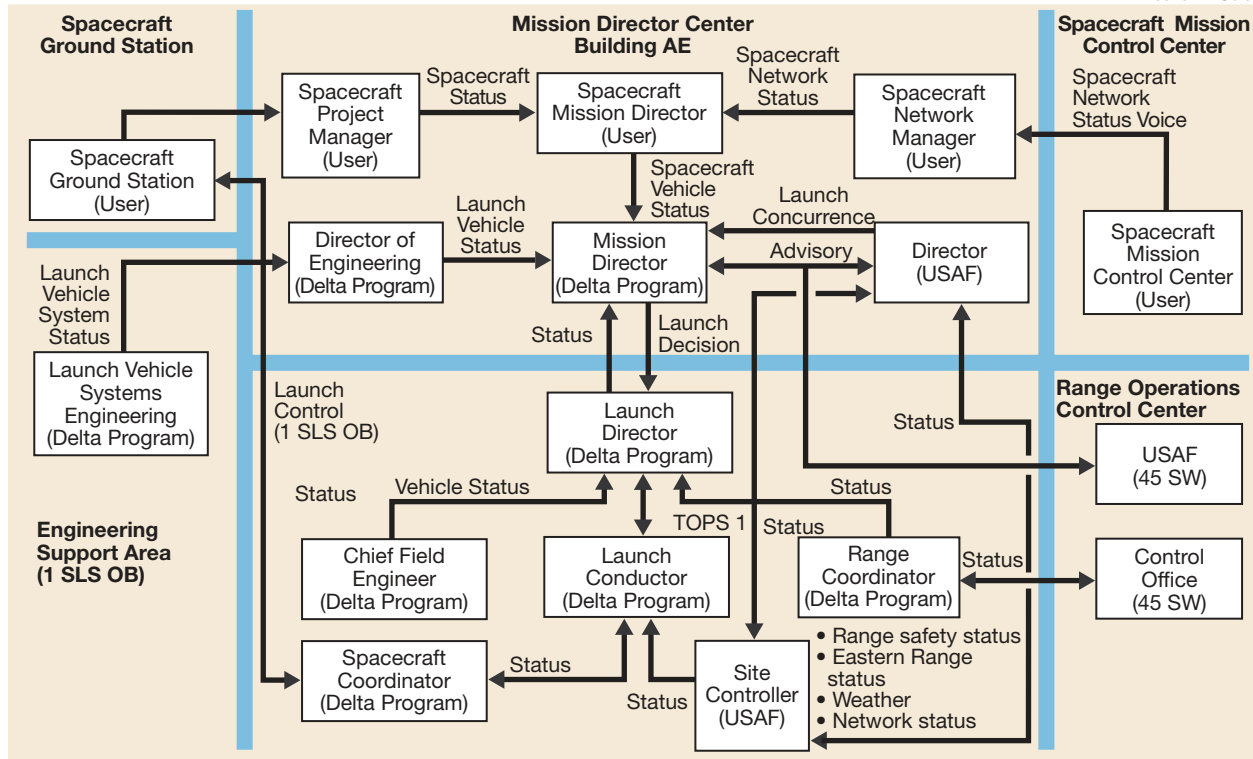


Figure 6-15. Launch Decision Flow for Commercial Missions—Eastern Range

The following limitations on ground winds (including gusts) apply:

- A. The MST shall not be moved from the Delta II if ground winds in any direction exceed 36 knots (41 mph) at the 9.1-m (30-ft) level.
- B. The maximum allowable ground winds at the 28.0-m (92-ft) level are shown on Figure 6-16 for 792X vehicles with lengthened nozzles on the air-ignited GEMs. As noted on the figure, the constraints are a function of the predicted liftoff solid-motor-propellant bulk temperature. This figure applies to both 9.5-ft and 10-ft-dia fairing configurations. The plot combines liftoff controls, liftoff loads, and on-stand structural ground wind restrictions.

6.5.2.2 Winds Aloft Constraints. Measurements of winds aloft are taken at the launch pad. The Delta II controls and loads constraints for winds aloft are evaluated on launch day by conducting a trajectory analysis using the measured wind. A curvefit to the wind data provides load relief in the trajectory analyses. The curvefit and other load-relief parameters are used to reset the mission constants just prior to launch.

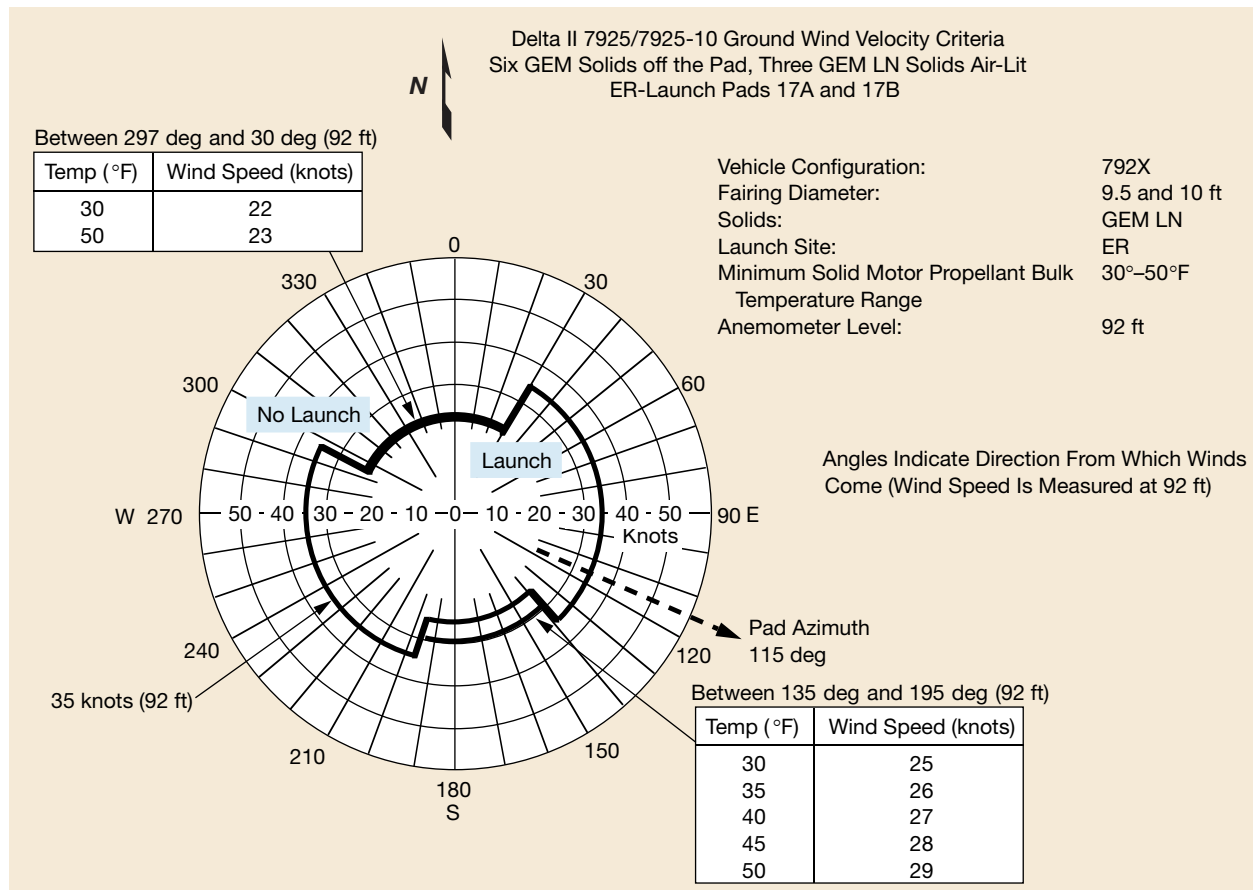


Figure 6-16. Delta II 792X Ground Wind Velocity Criteria, SLC-17

6.5.2.3 Lightning Activity. The following are Delta Program procedures for operating during lightning activity:

- A. Evacuation of the MST and fixed umbilical tower (FUT) is accomplished at the direction of the Boeing Test Conductor (Reference: Delta Launch Complex Safety Plan).
- B. First- and second-stage instrumentation may be operated during an electrical storm.
- C. If other vehicle electrical systems are powered when an electrical storm approaches, these systems may remain powered.
- D. If an electrical storm passes through after a simulated flight test, all electrical systems are turned on in a quiescent state, and all data sources are evaluated for evidence of damage. This turn-on is done remotely (pad clear) if any category A ordnance circuits are connected for flight. Ordnance circuits are disconnected and safed prior to turn-on with personnel exposed to the vehicle.

- E. If data from the quiescent turn-on reveal equipment discrepancies that can be attributed to the electrical storm, a flight program requalification test must be run subsequent to the storm and prior to a launch attempt.

Spacecraft personnel can follow the same procedures (which may be more restrictive).

6.5.3 Operational Safety

Safety requirements are covered in Section 9 of this document. In addition, it is the operating policy at both Boeing and Astrotech that all personnel will be given safety orientation briefings prior to entrance to hazardous areas. These briefings will be scheduled by the Delta Program Office spacecraft coordinator and presented by the appropriate safety personnel.

6.5.4 Security

6.5.4.1 Launch Complex Security. SLC-17 physical security is ensured by perimeter fencing, guards, and access badges. The MST white room is a Defense Investigative Service (DIS)-approved closed area with cypher locks on entry-controlled doors. Access can be controlled by a security guard on the MST eighth level.

6.5.4.2 CCAFS Security. For access to CCAFS, U.S. citizens must provide to the Delta Program security coordinator full name with middle initial if applicable, social security number, company name, and dates of arrival and expected departure. Delta Program security will arrange for entry authority for commercial missions or for individuals sponsored by the Delta Program. Access by NASA personnel or NASA-sponsored foreign nationals is coordinated at CCAFS by NASA KSC with the USAF. Access by other U.S. government-sponsored foreign nationals is coordinated by their sponsor directly with the USAF at CCAFS. For non-United States citizens, clearance information (name, nationality/citizenship, date and place of birth, passport number and date/place of issue, visa number and date of expiration, and title or job description) must be furnished to the Delta Program Office not later than 45 days prior to the CCAFS entry date. Failure to comply with the deadlines may result in access to CCAFS being denied by the Air Force. Government-sponsored individuals must follow NASA or US government guidelines as appropriate. The spacecraft coordinator will furnish visitor identification documentation to the appropriate agencies. After Delta Program security receives clearance approval, entry to CCAFS will be the same as for U.S. citizens.

6.5.5 Field-Related Services

Boeing employs certified propellant handlers, equipment drivers, welders, riggers, explosive ordnance handlers, and people experienced in most electrical and mechanical assembly skills such as torquing, soldering, crimping, precision cleaning, and contamination control. Boeing has under its control a machine shop, metrology laboratory, LO₂ cleaning facility, proof-load facility, and hydrostatic proof test equipment. Boeing operational team members are familiar with the payload processing facilities at the CCAFS, KSC, and Astrotech, and can offer all of these skills and services to the spacecraft project during the launch program.

6.6 DELTA II PLANS AND SCHEDULES

6.6.1 Integrated Schedules

The schedule of spacecraft activities varies from mission to mission. The extent of spacecraft field testing varies and is determined by the customer.

Spacecraft/launch vehicle schedules are similar from mission to mission, from the time of spacecraft weighing until launch.

Daily schedules are prepared on hourly timelines for these integrated activities. These schedules typically cover the integration effort in the HPF and launch pad activities after the spacecraft arrives. HPF tasks can include spacecraft weighing, spacecraft third-stage mate and interface verification, and transportation can assembly around the combined payload. The pad schedules provide a detailed, hour-by-hour breakdown of operations, illustrating the flow of activities from spacecraft erection through terminal countdown and reflecting inputs from the spacecraft project. These schedules comprise the integrating document to ensure timely launch pad operations.

Typical schedules of integrated activities from spacecraft weighing in the HPF until launch (Figures 6-17 through 6-29) are shown as launch minus (T-) workdays. Saturdays, Sundays, and holidays are typically not scheduled workdays and therefore are not T-days. The T-days, from spacecraft mate through launch, are coordinated with the customer to optimize on-pad testing. All operations are formally conducted and controlled using launch countdown documents. The schedules of spacecraft activities during that time, also called countdown bar charts, are controlled by the Boeing chief launch conductor. Tasks involving the spacecraft or tasks requiring that spacecraft personnel be present are shaded for easy identification.

Typical preparation tasks for a three-stage mission from CCAFS are as follows (stand-alone spacecraft and third-stage checkout are completed before T-11 day).

T-11 Tasks include equipment verification, precision weighing of the spacecraft by Boeing, and securing.

T-10 Spacecraft is lifted and mated to the third stage; clampband is installed, and initial clampband tension is established.

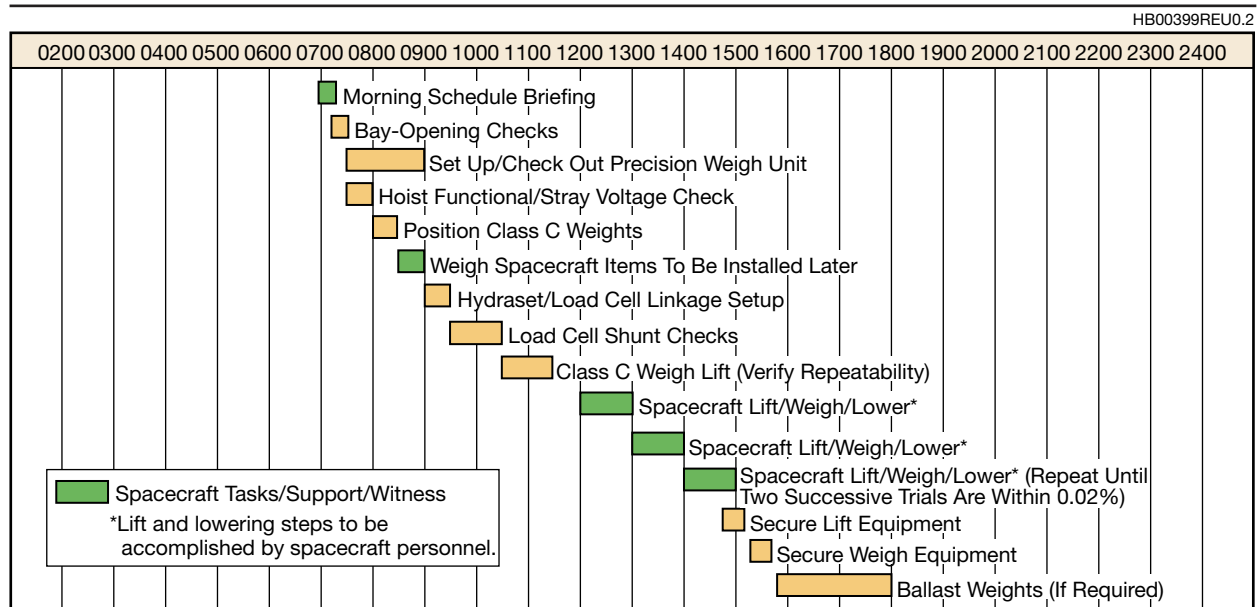


Figure 6-17. Typical Spacecraft Weighing (T-11 Day)

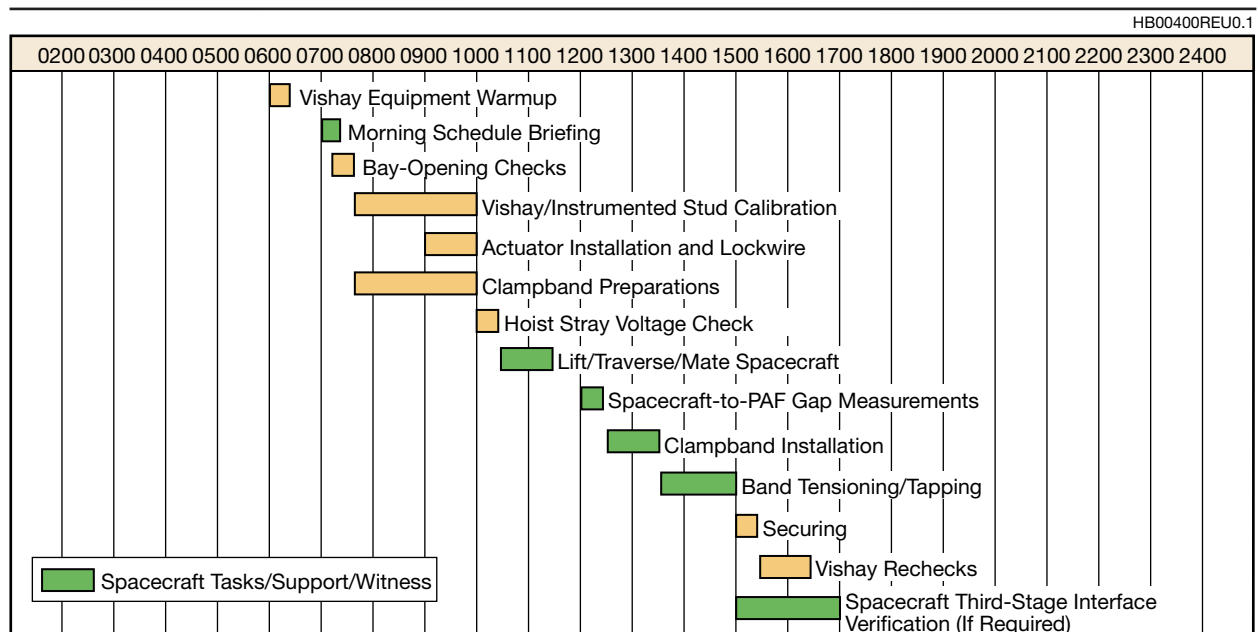


Figure 6-18. Typical Mating of Spacecraft and Third Stage (T-10 Day)

T-9 Final preparations are made prior to can-up for both spacecraft and third stage, and spacecraft/ third-stage interface is verified, if required.

T-8 The payload handling can is assembled around the spacecraft/third stage; handling can transportation covers are installed; the can is placed on its trailer; and the handling can purge is set up.

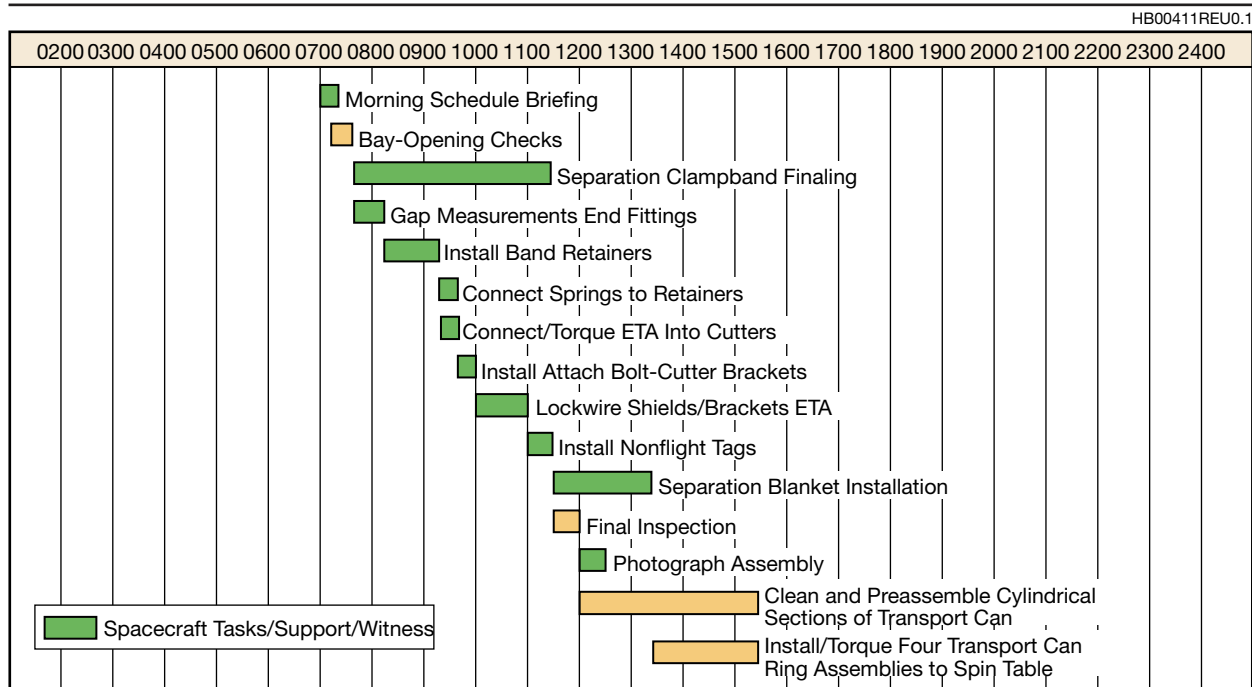


Figure 6-19. Typical Final Spacecraft Third-Stage Preparations (T-9 Day)

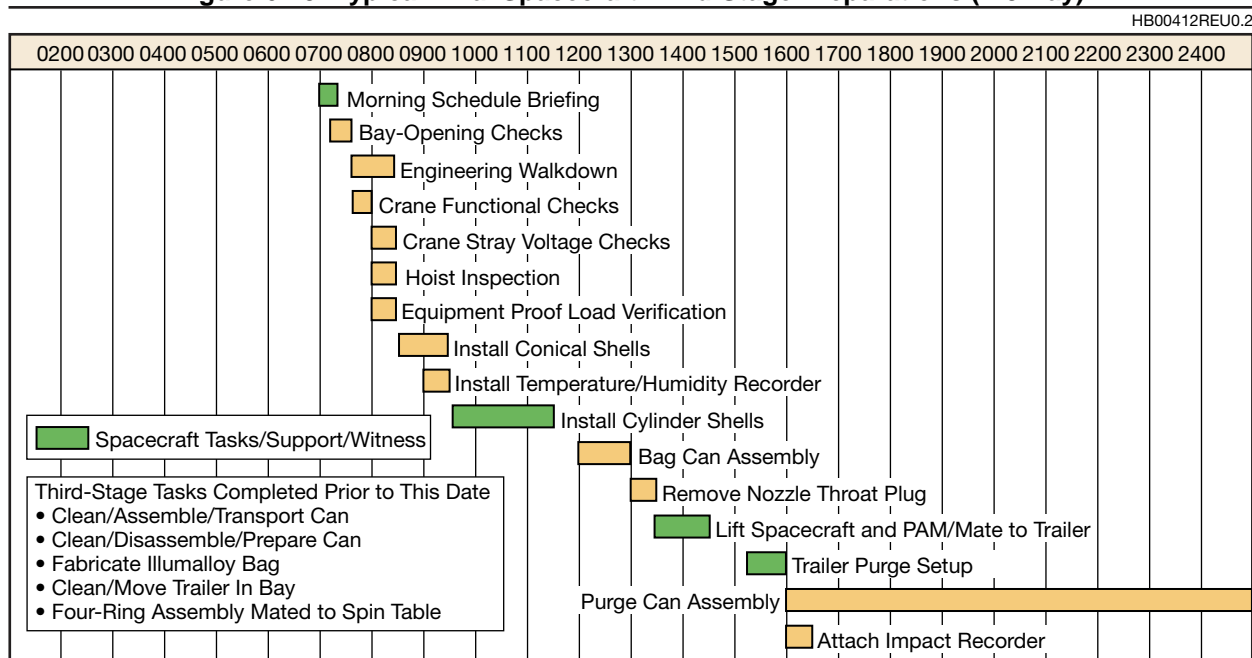


Figure 6-20. Typical Installation of Transportation Can (T-8 Day)

T-7 Tasks include transportation to the launch site, erection, and mating of the spacecraft/upper stage to the Delta II second stage in the MST cleanroom. Preparations are made for the launch vehicle flight program verification test.

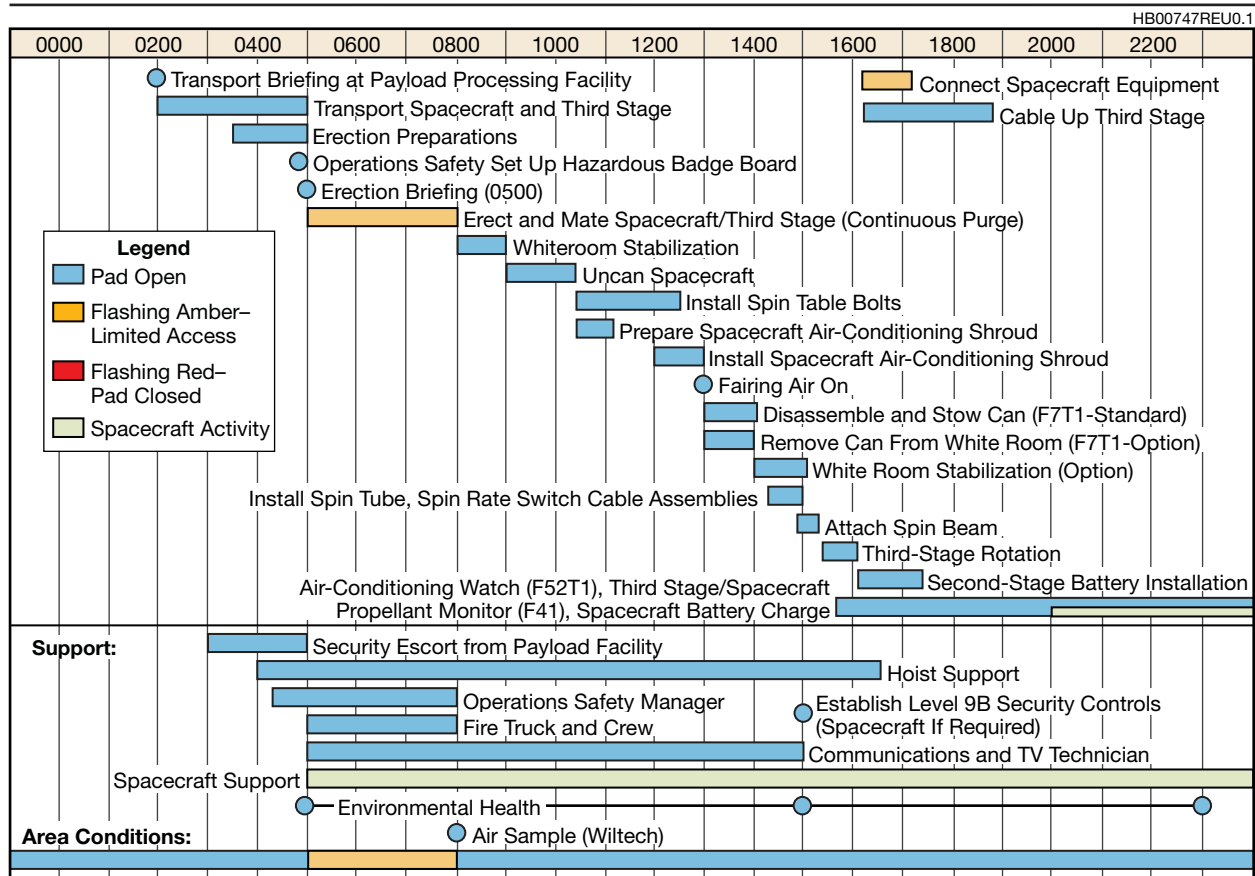


Figure 6-21. Typical Spacecraft Erection (T-7 Day)

T-6 The launch vehicle flight program verification test is performed, followed by the vehicle power-on stray-voltage test. Spacecraft systems powered at liftoff are turned on during the flight program verification test, and all data are monitored for electromagnetic interference (EMI) and radio frequency interference (RFI). Spacecraft systems to be turned on at any time between T-5 day and spacecraft separation are turned on in support of the vehicle power-on stray voltage test. Spacecraft support of these two vehicle system tests is critical to meeting the scheduled launch date.

T-5 The Delta II vehicle ordnance installation/connection, preparation for fairing installation, and spacecraft closeout operations are performed.

T-4,3 Spacecraft final preparations prior to fairing installation include Delta II upper-stage closeout, preparations for second-stage propellant servicing, and fairing installation.

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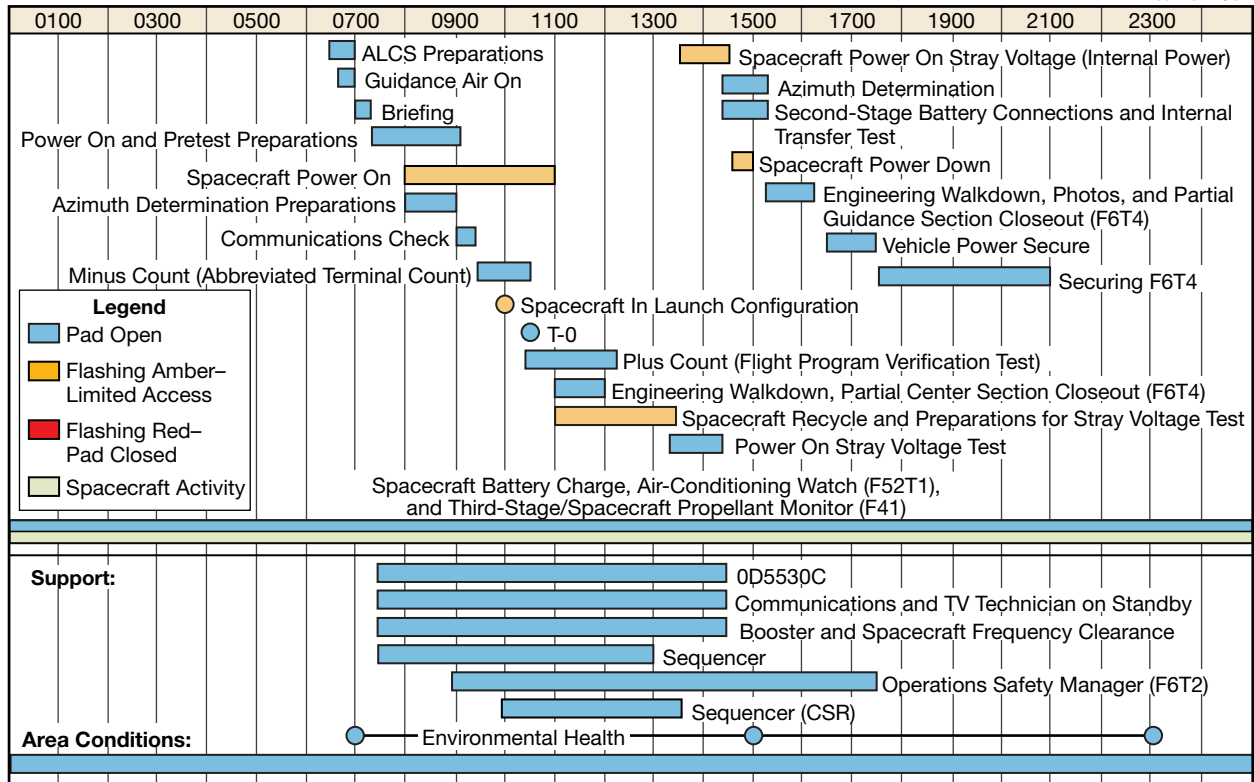


Figure 6-22. Typical Flight Program Verification and Stray-Voltage Checks (T-6 Day)

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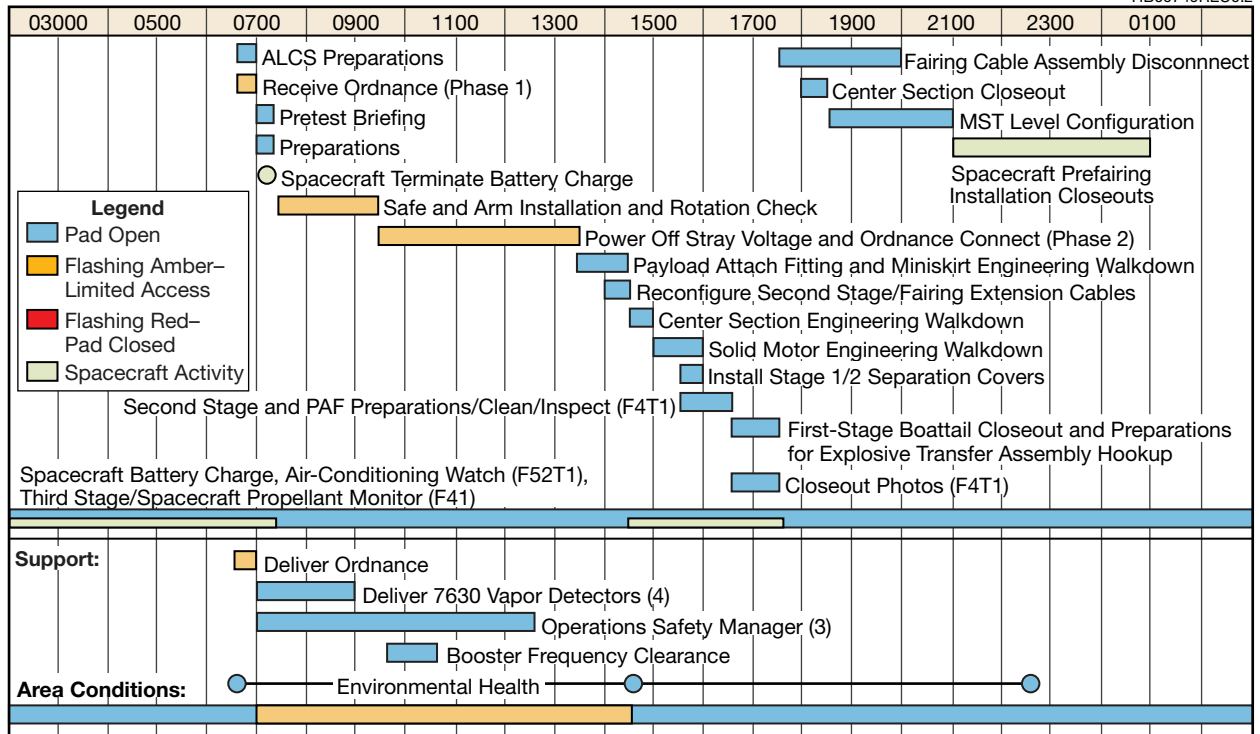


Figure 6-23. Typical Ordnance Installation and Hookup (T-5 Day)

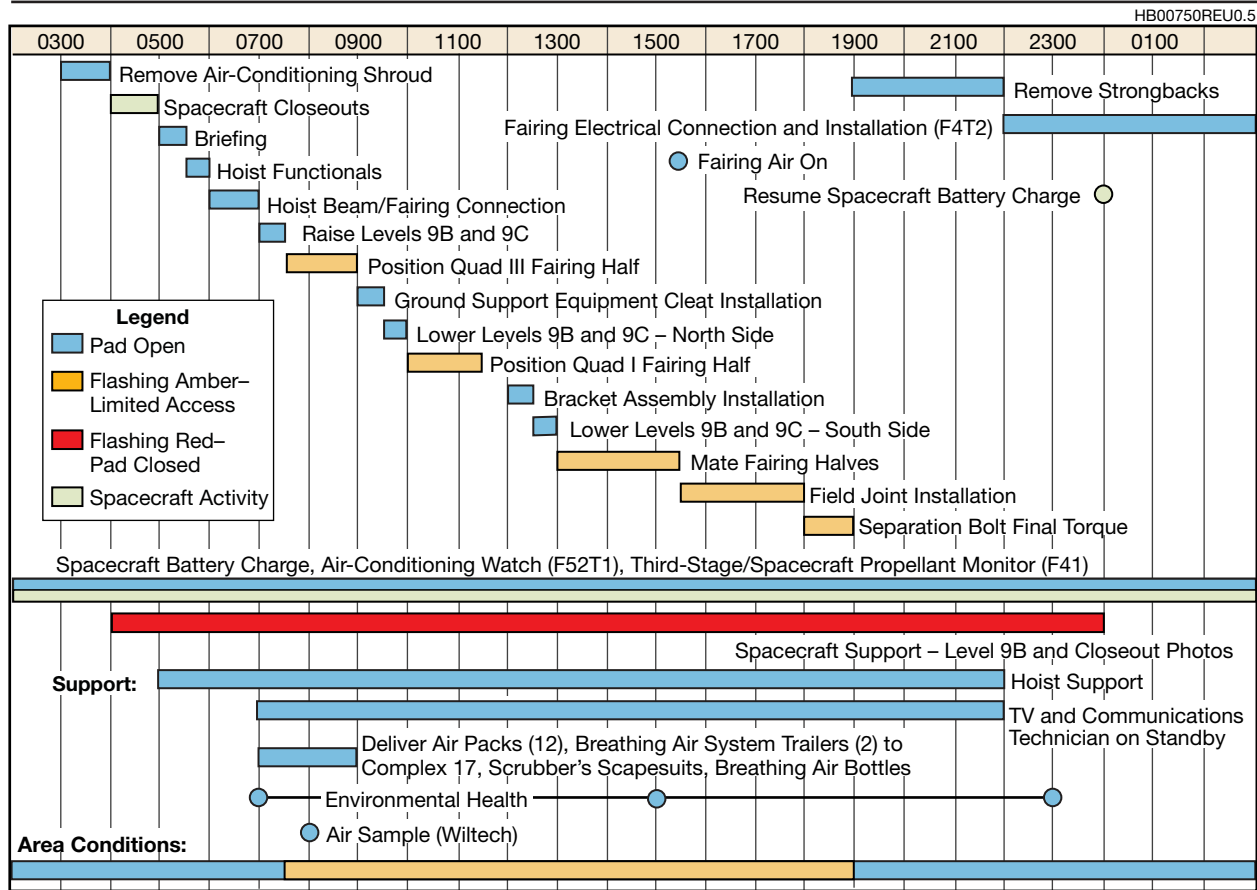


Figure 6-24. Typical Fairing Installation (T-4 Day)

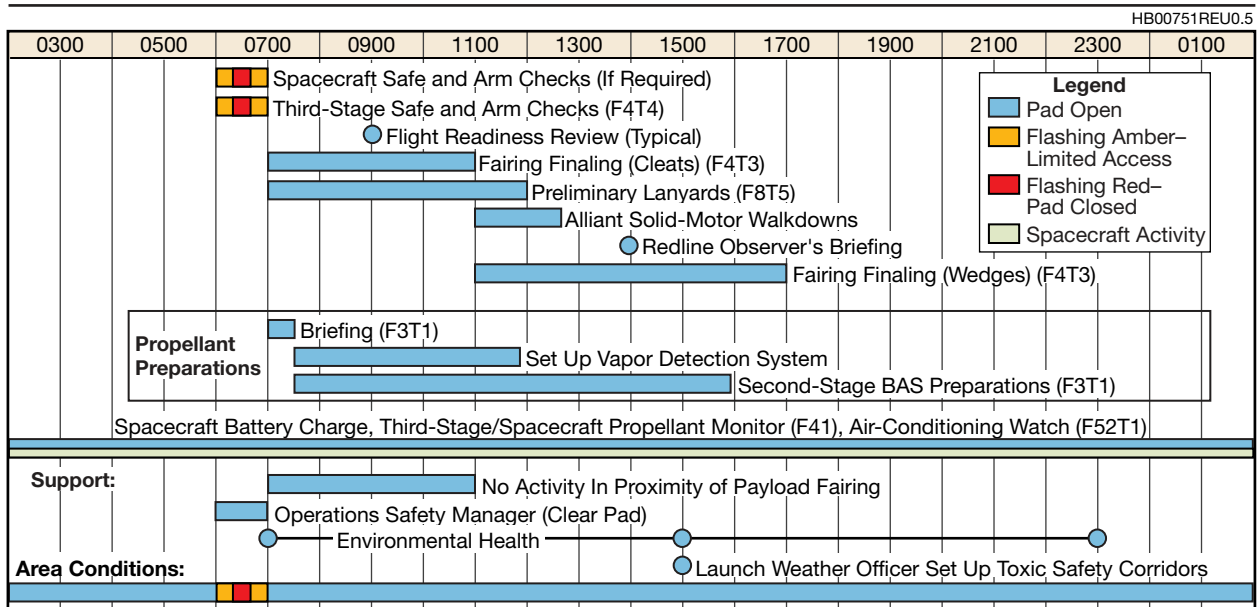


Figure 6-25. Typical Propellant Loading Preparations (T-3 Day)

T-2 Second-stage propellant is loaded.

T-1 Tasks include C-band beacon readout and azimuth verification, followed by the vehicle class A ordnance connection, spacecraft ordnance arming, final fairing preparations for MST removal, second-stage engine section closeout, and launch vehicle final preparations.

T-0 Launch day preparations include MST gantry removal, final arming, terminal sequences, and launch. Spacecraft should be in launch configuration immediately prior to T-4 minutes and standing by for liftoff. The nominal hold and recycle point is T-4 minutes.

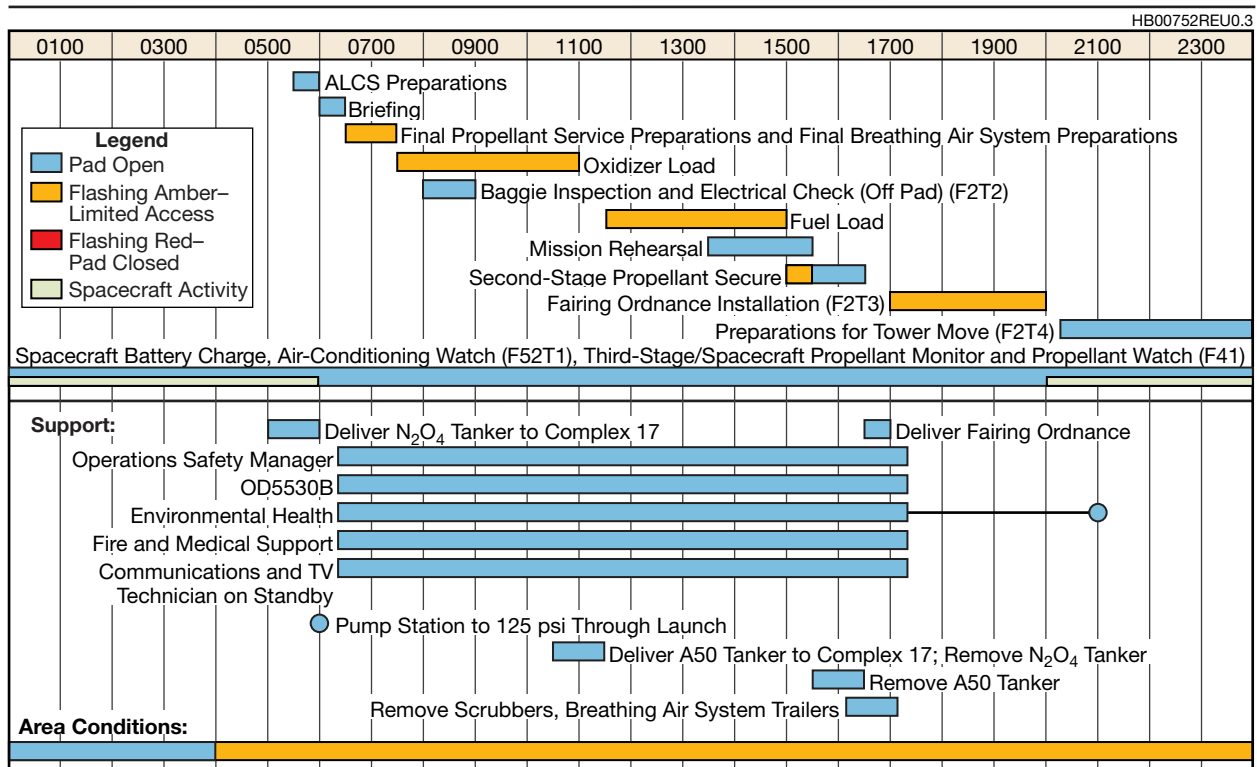


Figure 6-26. Typical Second-Stage Propellant Loading (T-2 Day)

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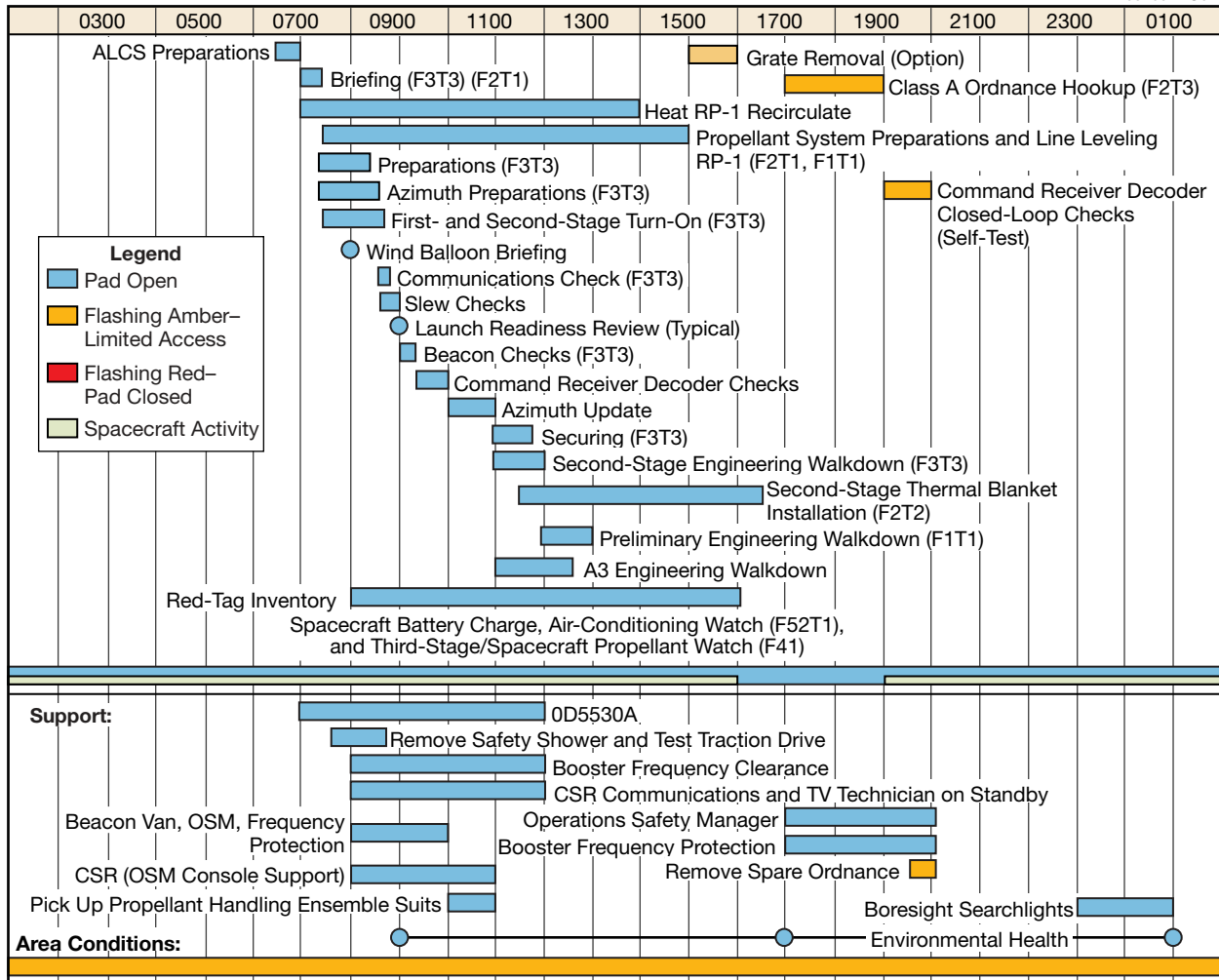


Figure 6-27. Typical Beacon, Range Safety, and Class A Ordnance (T-1 Day)

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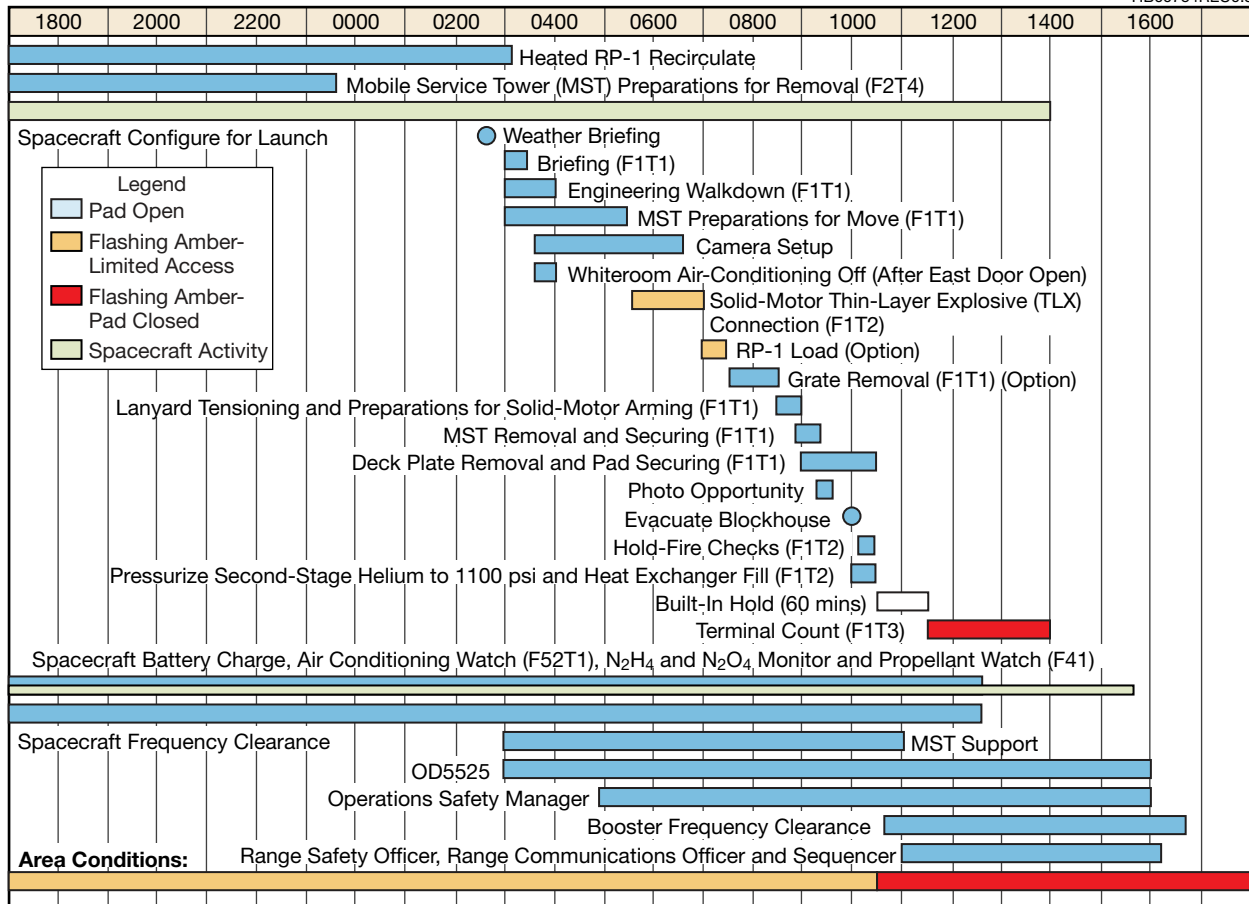


Figure 6-28. Typical Delta Countdown (T-0 Day)

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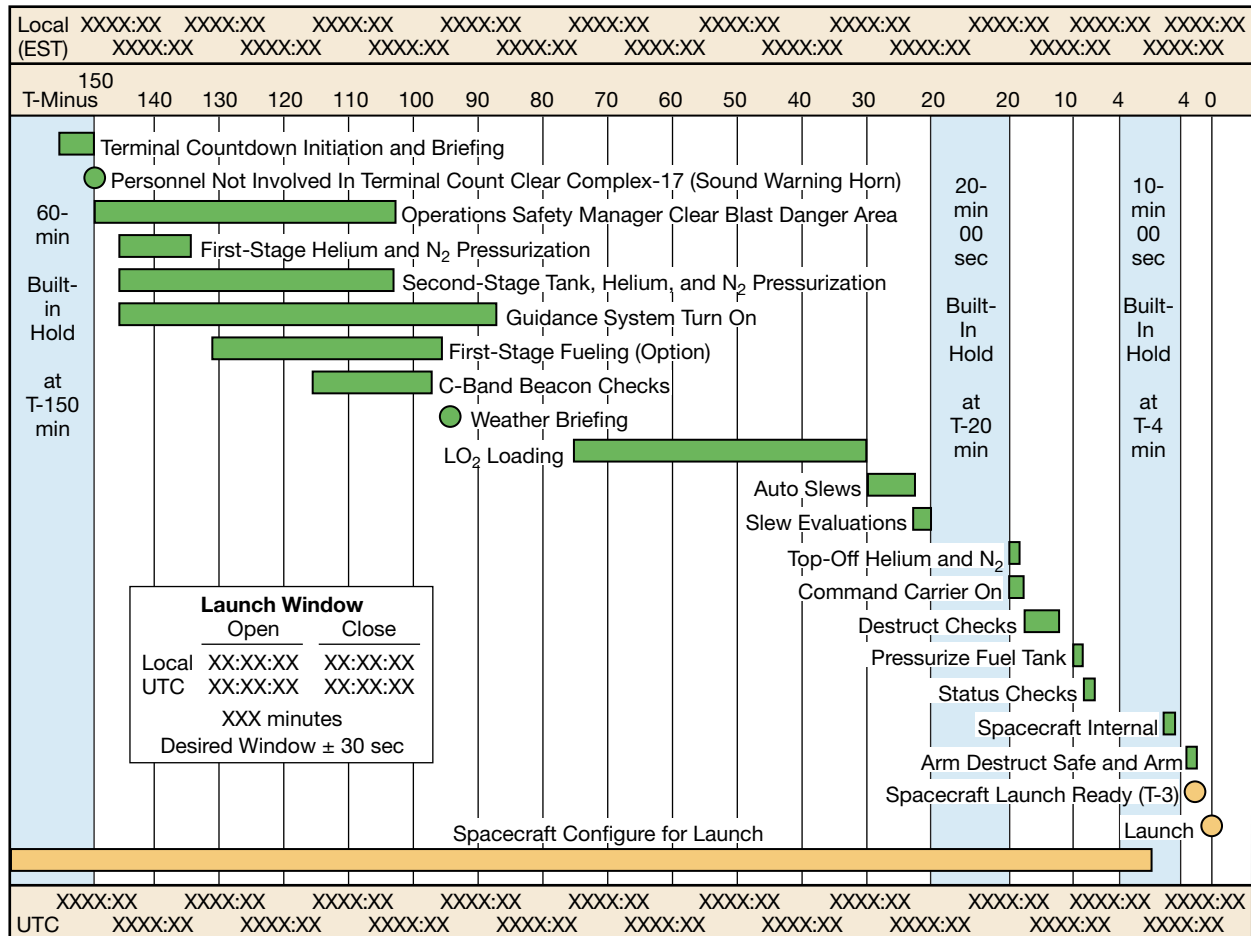


Figure 6-29. Typical Terminal Countdown (T-0 Day)

6.6.2 Launch Vehicle Schedules

One set of facility-oriented three-week schedules is developed, on a daily timeline, to show processing of multiple launch vehicles through each facility: i.e., for both launch pads, Delta Mission Checkout (DMCO), hangar M, solid-motor area, and each of the three PPFs as required. These schedules are revised daily and reviewed at the twice-weekly Delta status meetings. Another set of launch-vehicle-specific schedules are generated, on a daily timeline, covering a two- or three-month period to show the complete processing of each launch vehicle component. An individual schedule is made for each DMCO, third-stage HPF, and launch pad.

6.6.3 Spacecraft Schedules

The spacecraft project team will supply schedules to the Delta Program spacecraft coordinator who will arrange support as required.

6.7 DELTA II MEETINGS AND REVIEWS

During launch preparation, various meetings and reviews take place. Some of these will require spacecraft customer input while others allow the customer to monitor the progress of the overall mission. The Boeing spacecraft coordinator will ensure adequate spacecraft user participation.

6.7.1 Meetings

6.7.1.1 Delta Status Meetings. Status meetings, generally held twice a week at the OB, include a review of the activities that are scheduled or that have been accomplished since the last meeting; a discussion of problems and their solutions; and a general review of the mission schedule and specific mission schedules. SLC-17 activities are also reviewed. Spacecraft user representatives are encouraged to attend these meetings.

6.7.1.2 Daily Schedule Meetings. Daily schedule meetings are held in the OB and conference rooms by teleconference to provide the team members with their assignments and to summarize the previous or current day's accomplishments. These meetings are attended by the Test Conductor, Assistant Test Conductor, technicians, inspectors, engineers, supervisors, and the Spacecraft Coordinator. These meetings are held at the beginning of the first shift. Special circumstances may dictate that a meeting be held at the beginning of the second shift.

A daily meeting, usually at the end of the first shift, with the Delta Program launch conductor, spacecraft coordinator, and spacecraft representatives attending, is held starting approximately three days prior to the arrival of the spacecraft at the pad. Discussed are the status of the day's activities, the work remaining, problems, and the next day's schedule. This meeting may be conducted by telephone if required. The fully coordinated countdown bar charts are delivered to the payload customer at this meeting.

6.7.2 Reviews

Periodic reviews are held to ensure that the spacecraft and launch vehicle are ready for launch. The following paragraphs describe the Delta II readiness reviews.

6.7.2.1 Postproduction Review. This meeting, conducted at Decatur, Alabama, reviews the flight hardware at the end of production and prior to shipment to CCAFS.

6.7.2.2 Mission Analysis Review. This review is held at Huntington Beach, California, approximately 3 months prior to launch, to review mission-specific designs, studies, and analyses.

6.7.2.3 Pre-Vehicle-On-Stand (Pre-VOS) Review. This review is held at Huntington Beach subsequent to the completion of Delta mission checkout (DMCO) and prior to erection of the vehicle on the launch pad. It includes an update of the activities since the post-production review, the results of the DMCO processing, and any hardware history changes. Launch facility readiness is also reviewed.

6.7.2.4 Vehicle-On-Stand Readiness Review (VRR). This review is held at the launch site prior to first-stage erection. The status and processing history of the launch vehicle elements and ground support equipment are presented. The primary focus of this review is on the readiness of the first stage, solid motors, interstage, second stage, and fairing for erection and mate on the launch pad. Upon completion of this meeting and resolution of any concerns raised, authorization is given to proceed with erection activities.

6.7.2.5 Launch Site Readiness Review (LSRR). This review is held prior to erection and mate of the spacecraft. It includes an update of the activities since the pre-VOS review and verifies the readiness of the launch vehicle, launch facilities, and spacecraft for transfer of the spacecraft to the pad. Upon completion of this meeting and resolution of any concerns raised, authorization is given to proceed with spacecraft transfer to launch pad, immediately followed by erection and mate with the second stage.

6.7.2.6 Flight Readiness Review (FRR). This review, typically held on T-3 day, is an update of activities since the pre-VOS and is conducted to determine that checkout has shown that the launch vehicle and spacecraft are ready for countdown and launch. Upon completion of this meeting, authorization is given to proceed with the loading of second-stage propellants. This review also assesses the readiness of the range to support launch and provides a predicted weather status.

6.7.2.7 Launch Readiness Review (LRR). This review is normally held one day prior to launch and provides an update of activities since the FRR. All agencies and contractors are required to provide a ready-to-launch statement. Upon completion of this meeting and resolution of any concerns raised, an authorization to enter terminal countdown is given.

Section 7
LAUNCH OPERATIONS AT WESTERN RANGE

This section presents a description of Delta launch vehicle operations associated with Space Launch Complex 2 (SLC-2) at Vandenberg Air Force Base (VAFB), California. Prelaunch processing of the Delta II is presented, as is a discussion of spacecraft processing and operations that are conducted prior to launch day.

7.1 ORGANIZATIONS

The Delta Program operates the Delta launch system and maintains a team that provides launch services to NASA, USAF, and commercial customers at VAFB. The Delta Program provides the interface to the Department of Transportation (DOT) for the licensing and certification needed to launch commercial spacecraft using the Delta II.

NASA is responsible for the SLC-2 launch facilities at VAFB. For NASA and NASA-sponsored launches, NASA operates spacecraft processing facilities at VAFB that are used in support of Delta missions. The Delta Program interface with NASA is through the Kennedy Space Center (KSC) Launch Services Program Office. NASA maintains a resident office at VAFB, and NASA designates a launch site integration manager (LSIM) who arranges all the support (NASA launches only) required from NASA for a launch from VAFB. The Delta Program Office has established an interface with the 30th Space Wing Directorate of Plans. The Western Range has designated a range program support manager (PSM) to be a representative of the 30th Space Wing. The PSM serves as the official interface for all support and services requested. These services include range instrumentation, facilities/equipment operation and maintenance, safety, security, and logistics support. Requirements satisfied by NASA and/or USAF are described in the government's universal document system (UDS) format. The Delta Program Office and the spacecraft agency generate the program requirements document (PRD). Formal submittal of these documents to the government agencies is arranged by the Delta Program Office.

For commercial launches, the Delta Program Office makes all the arrangements for the payload processing facilities and services. The organizations that support a launch from VAFB are shown in Figure 7-1. A spacecraft coordinator from the Delta-VAFB launch team is assigned to each mission to assist the spacecraft team during the launch campaign. The coordinator shall arrange for support of the spacecraft, assist in obtaining safety approval of the spacecraft test procedures and operations, integrate the spacecraft operations into the launch vehicle operations, and, during the countdown and launch, serve as the interface between the spacecraft and test conductor in the remote launch control center (RLCC).

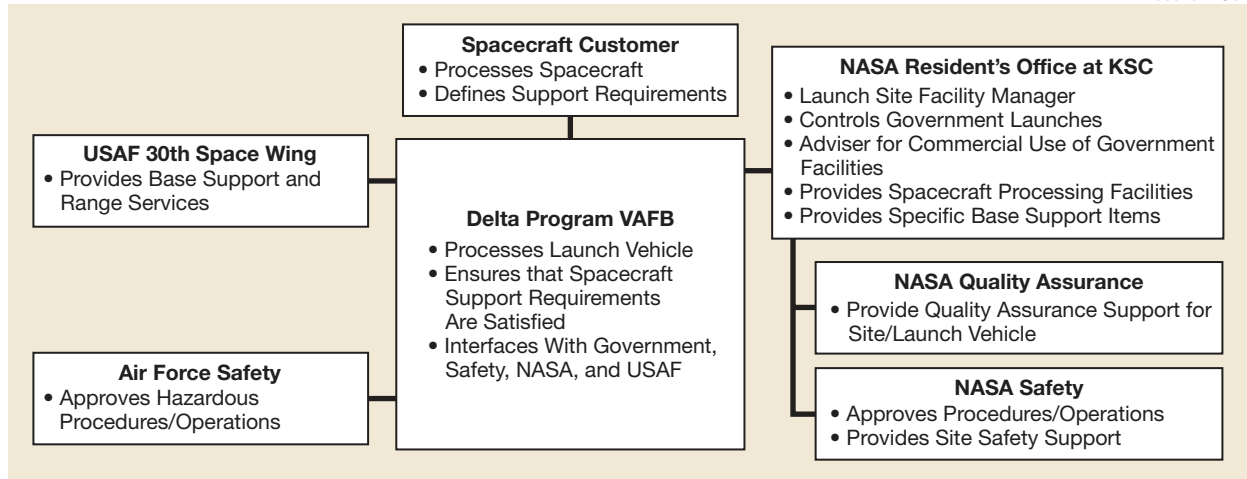


Figure 7-1. Launch Base Organization at VAFB

7.2 FACILITIES

In addition to the facilities required for Delta II launch vehicle processing, specialized facilities are provided for checkout and preparation of the spacecraft. Laboratories, cleanrooms, receiving and shipping areas, hazardous operations areas, and offices are provided for spacecraft project personnel. A map of VAFB is shown in Figure 7-2. The commonly used facilities at VAFB for NASA or commercial spacecraft are the following:

- A. Spacecraft payload processing facilities (PPF):
 1. NASA, building 836.
 2. Astrotech Space Operations, building 1032.
 3. Spaceport Systems International, building 375.
- B. Hazardous processing facilities (HPFs):
 1. NASA, building 1610.
 2. Astrotech Space Operations, building 1032.
 3. Spaceport Systems International, building 375.

While there are other spacecraft processing facilities located on VAFB that are under USAF control, commercial spacecraft will normally be processed through the commercial facilities of Astrotech Space Operations or Spaceport Systems International. Government facilities for spacecraft processing (USAF or NASA) can be used for commercial spacecraft only under special circumstances (use requires negotiations between the Delta Program Office, the spacecraft agency, and the USAF or NASA). The spacecraft agency must provide its own test equipment for spacecraft preparations including telemetry receivers and telemetry ground stations.

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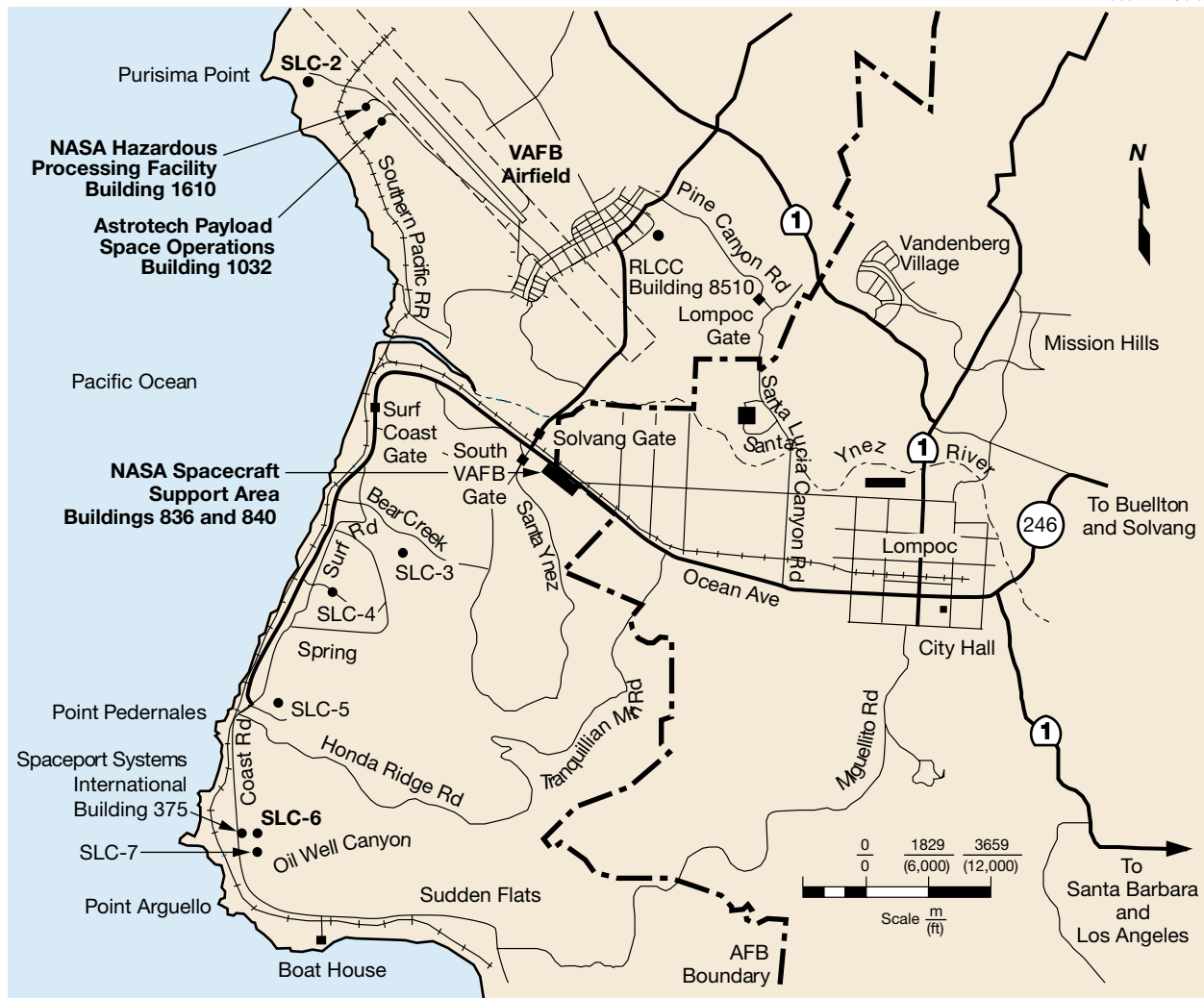


Figure 7-2. Vandenberg Air Force Base (VAFB) Facilities

After arrival of the spacecraft and its associated equipment at VAFB by road or by air (via the VAFB airfield), transportation to and from the payload processing facilities and to the launch site will be provided by the Delta Program or NASA, as appropriate. Equipment and personnel are also available for loading and unloading operations. It should be noted that the size of the shipping containers often dictates the type of aircraft used for transportation to the launch site. The air-freight carrier should be consulted for the type of freight unloading equipment that will be required at the western range. Shipping containers and handling fixtures attached to the spacecraft are provided by the spacecraft project.

Shipping and handling of hazardous materials such as electro-explosive devices, radioactive sources, etc., must be in accordance with applicable regulations. It is the responsibility of the spacecraft agency to identify these items and become familiar with such regulations. These regulations include those imposed by NASA, USAF, DOT, ATF, and FAA (refer to Section 9).

7.2.1 NASA Facilities on South VAFB

NASA spacecraft facilities are located in the NASA support area on South VAFB (SVAFB) (Figure 7-3). The spacecraft support area is adjacent to Ocean Avenue on Clark Street and is accessible through the SVAFB South Gate. The support area consists of the spacecraft laboratory (building 836), NASA technical shops, NASA supply, and NASA engineering and operations building (building 840).

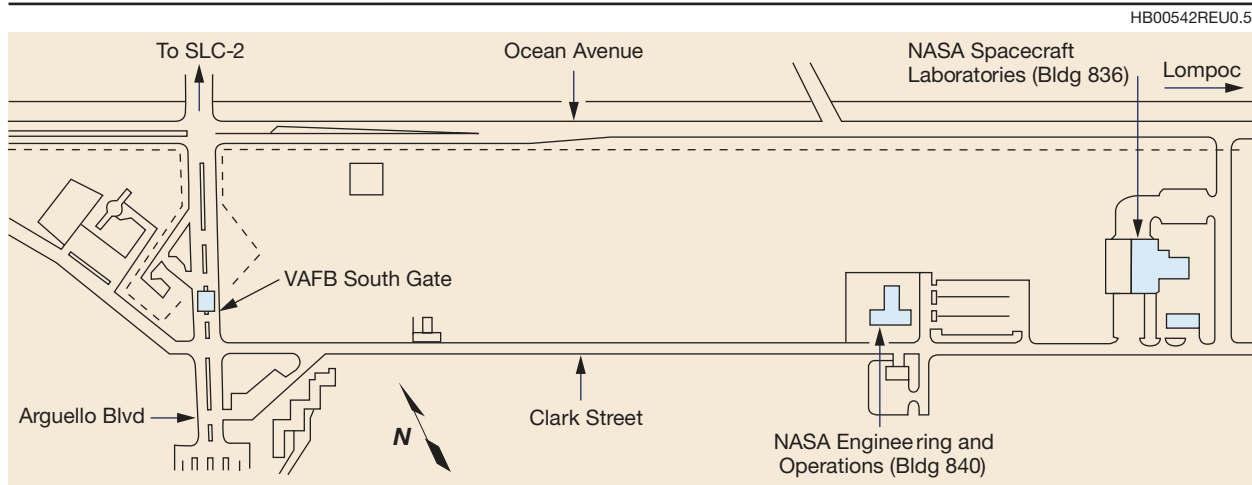


Figure 7-3. Spacecraft Support Area

7.2.1.1 NASA Telemetry Station and Spacecraft Laboratories. The NASA telemetry station and spacecraft laboratories, building 836 (Figure 7-4), are divided into work and laboratory areas and include the Mission Director Center (MDC), the Launch Vehicle Data Center (LVDC), spacecraft assembly areas, laboratory areas, cleanrooms, computer facility, office space, conference room, and the telemetry station.

Spacecraft laboratory 1 (Figure 7-5) consists of a high bay 20.4 m (67 ft) long by 9.8 m (32 ft) wide by 10.4 m (34 ft) high and an adjoining 167.2-m² (1800-ft²) support area. Personnel access doors and a sliding door 3.7 m (12 ft) by 3.7 m (12 ft) connect the two portions of this laboratory. The outside cargo entrance door to the spacecraft assembly room in laboratory 1 is 6.1 m (20 ft) wide by 7.7 m (25 ft 3 in.) high. A bridge crane, with an 8.8-m (29-ft) hook height and a 4545-kg (5-ton) capacity, is available for handling spacecraft and associated equipment. This assembly room contains a class 10,000 horizontal laminar flow cleanroom, 10.4 m (34 ft) long by 6.6 m (21.5 ft) wide by 7.6 m (25 ft) high. The front of the cleanroom opens for free entry of the spacecraft and handling equipment. The cleanroom has crane access in the front-to-rear direction only; however, the crane cannot operate over the entire length of the laboratory without disassembly because its path is obstructed by the horizontal beam that serves as the cleanroom divider. Spacecraft laboratory 1 will also support computer, telemetry, and checkout equipment in

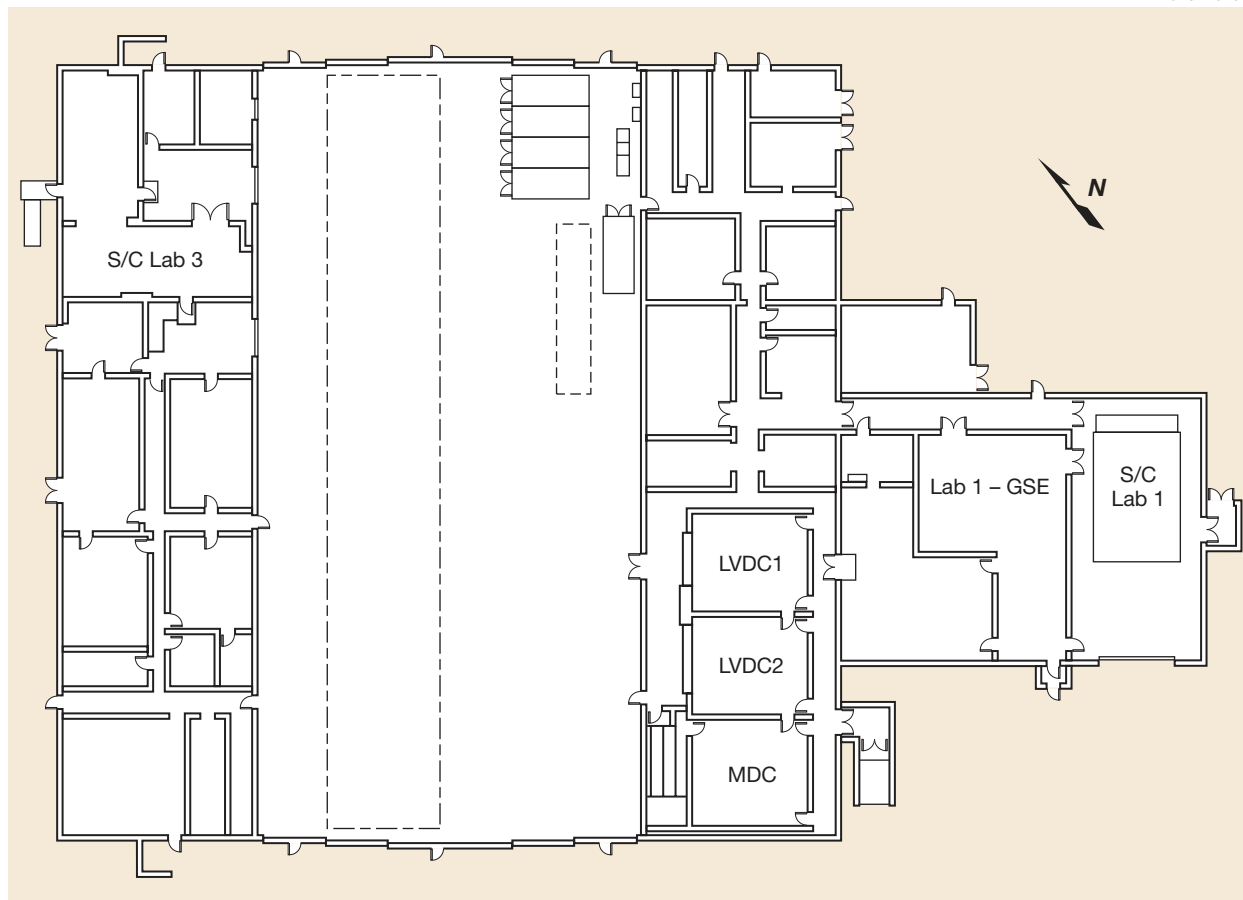


Figure 7-4. NASA Telemetry Station (Building 836)

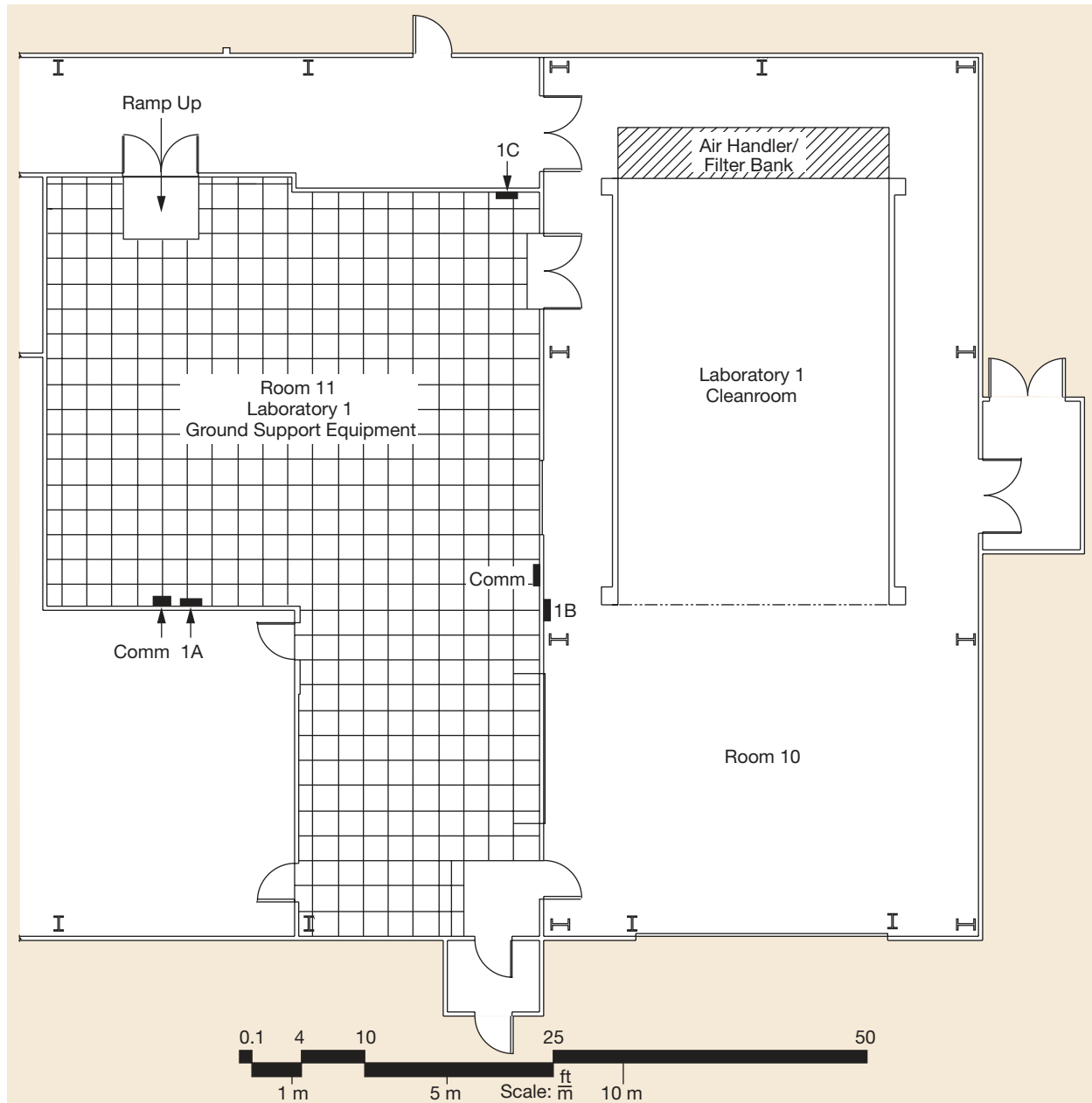


Figure 7-5. Spacecraft Laboratory 1 (Building 836)

a separate room containing raised floors and an under-floor power distribution system. This room has an area of approximately 167.2 m^2 (1800 ft^2).

Spacecraft laboratory 3 (Figure 7-6) has an area of 172.8 m^2 (1860 ft^2). This laboratory typically is assigned to the NOAA Environmental Monitoring Satellite Program, but could be used by other customers when not required by NOAA.

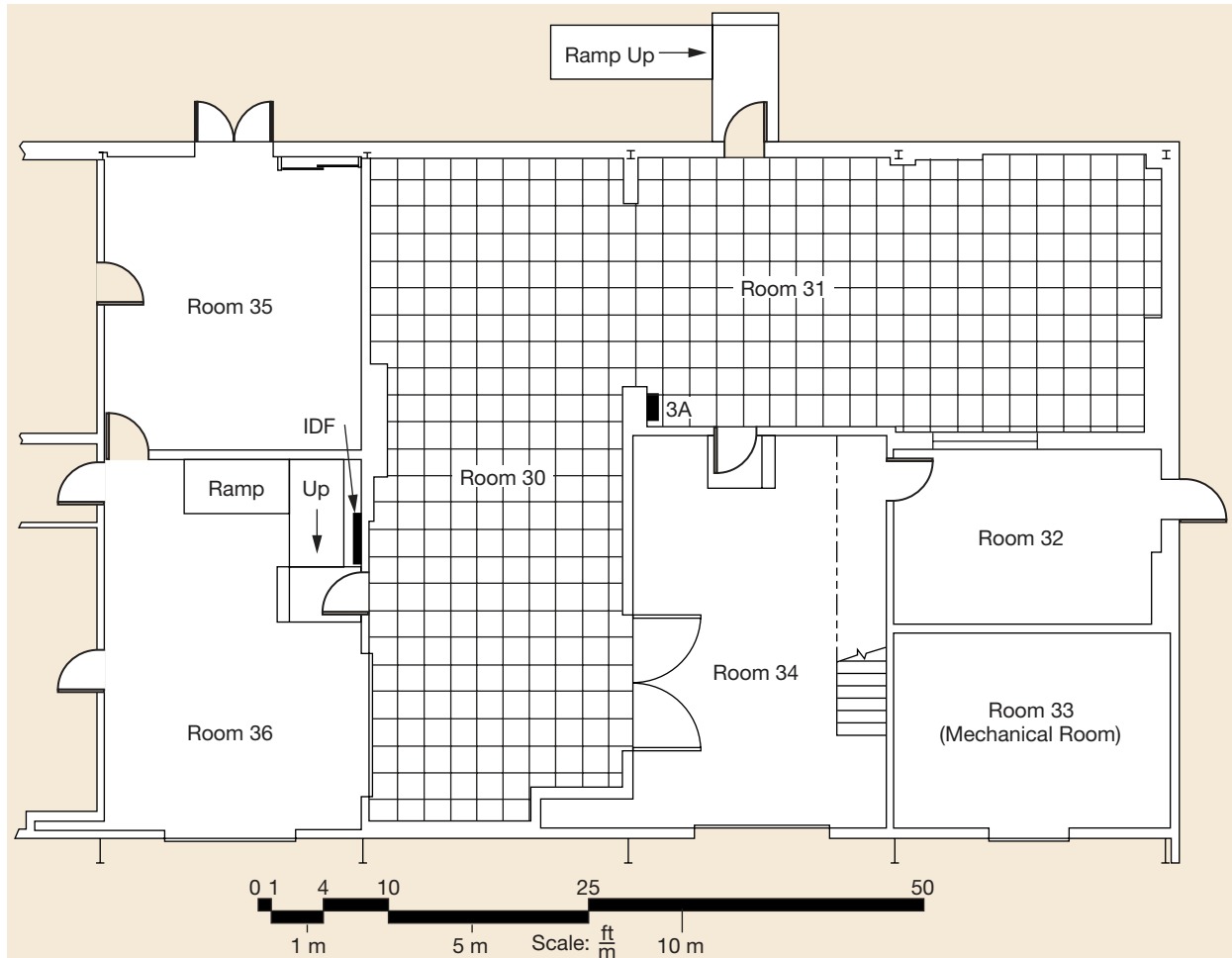


Figure 7-6. Spacecraft Laboratory 3 (Building 836)

Launch vehicle data center 1 (LVDC-1) (Figure 7-7) is an area containing 24 consoles for Delta Program Office management and technical support personnel. These positions are manned during countdown and launch to provide technical assistance to the launch team in the remote launch control center (RLCC) and to the Mission Director in the Mission Director Center (MDC). These consoles have individually programmed communications panels for specific mission requirements. This provides LVDC personnel with technical communications to monitor and coordinate both prelaunch and launch activities. Video data display terminals in the LVDC are provided for display of range and launch vehicle technical information.

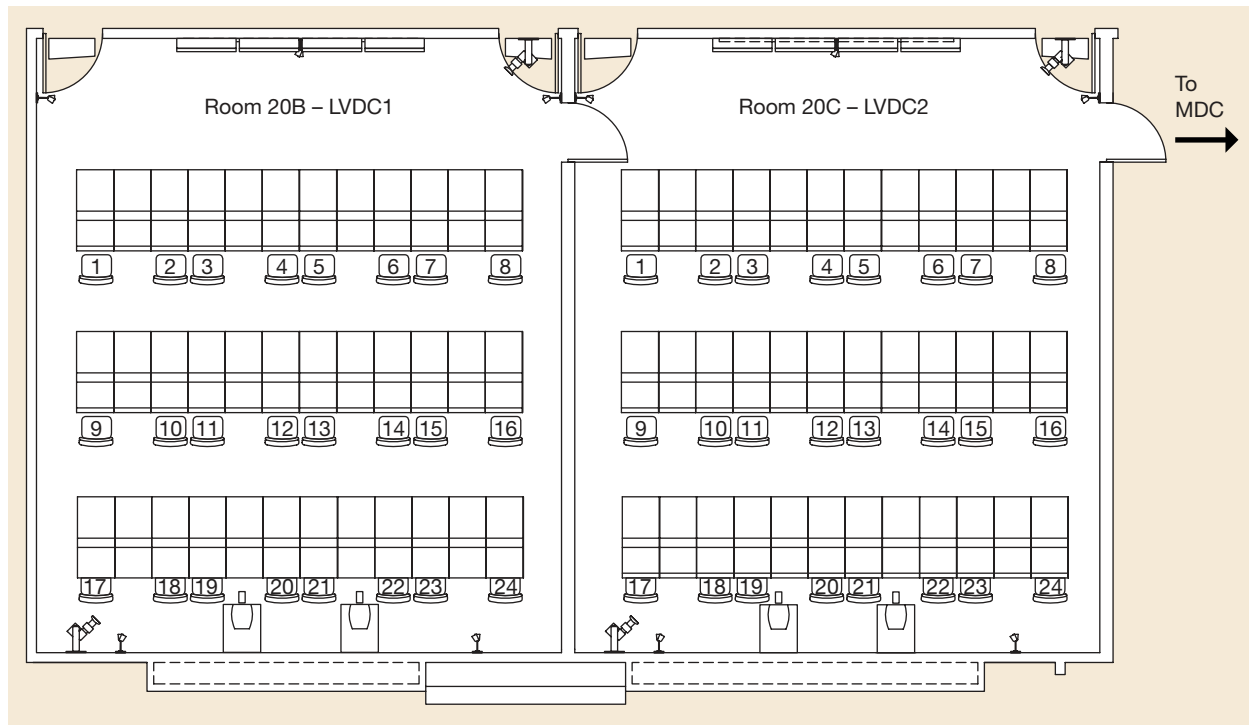


Figure 7-7. Launch Vehicle Data Center (Building 836)

Launch vehicle data center 2 (LVDC-2), a second data center, is provided with equipment similar to LVDC-1, and may also be used by spacecraft personnel.

The MDC (Figure 7-8) provides 32 communication consoles for use by the Mission Director, spacecraft and launch vehicle representatives, experimenters, display controller, and communications operators. These consoles have individually programmed communications for specific mission requirements. This provides Delta Program personnel with technical communications to monitor and coordinate both prelaunch and postlaunch activities.

Video data display terminals at the MDC are provided to display range and vehicle technical information. A readiness board and an events display board provide range and launch vehicle/spacecraft status during countdown and launch operations. Many TV display monitors (Figure 7-8) display preselected launch activities.

An observation room, separated from the MDC by a glass partition, is used for authorized visitors. Loudspeakers in the room monitor the communication channels used during the launch.

The high bay is a 30.5-m (100-ft) by 61-m (200-ft) (100-ft by 200-ft) area serviced by a 22,727- kg (25-ton) crane with a 7.6-m (25-ft) hook height. This area is ideal for handling heavy equipment and loading or unloading trucks. The high bay is heated and has 30.5-m (100-ft) wide by 9.1-m (30-ft) high sliding doors on both ends.

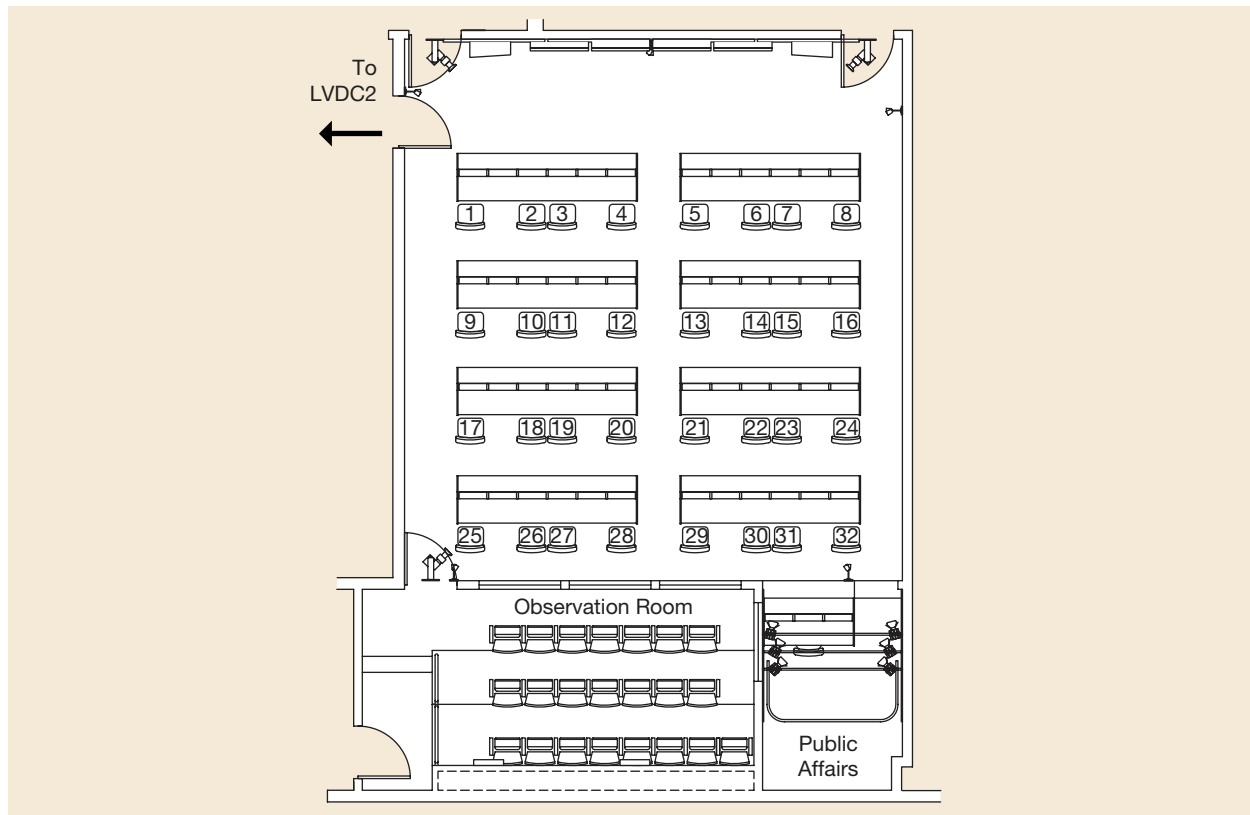


Figure 7-8. Mission Director Center (Building 836)

7.2.1.2 NASA Engineering and Operations Facility. The NASA engineering and operations facility in building 840 (Figure 7-9) is located on SVAFB at the corner of Clark and Scarpino Streets. It contains the NASA offices, NASA contractor offices, conference room, and other office space.

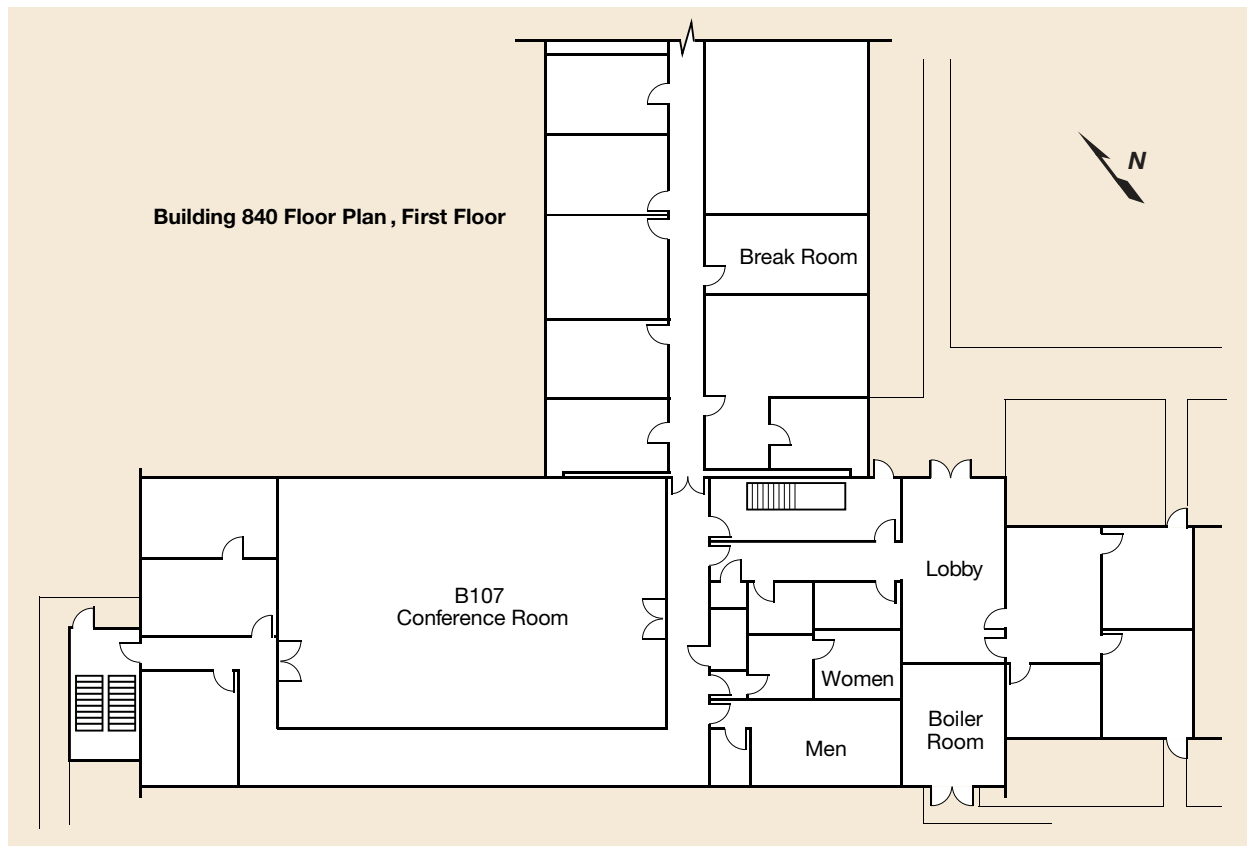


Figure 7-9. NASA Building 840

7.2.2 NASA Facilities on North Vandenberg

7.2.2.1 Hazardous Processing Facility (HPF). The NASA hazardous processing facility (building 1610) is located approximately 3.2 km (2 mi) east of SLC-2 and adjacent to Tangair Road (Figure 7-10). This facility (Figure 7-11) provides capabilities for the dynamic balancing of spacecraft, solid motors, and combinations thereof. It is also used for fairing processing, solid-motor buildup, spacecraft buildup, mating of spacecraft and solid motors, ordnance installation, and loading of hazardous propellants. It houses the Schenk treble dynamic balancing machine and equipment for buildup, alignment, and balancing of the third-stage solid-propellant motors and spacecraft. Composite spin balancing of the spacecraft/third-stage combination is not required. The spin-balancing machine is in a pit in the floor of building 1610. The machine interfaces with stages and/or spacecraft at floor level. Facilities consist of the hazardous processing facility (building 1610), control room (building 1605), UPS/generator building (building 1604), guard station, and fire pumping station. Hazardous operations are conducted in building 1610, which is separated from the control room by an earth revetment 4.6 m (15 ft) high. The two buildings are 47.2 m (155 ft) apart.

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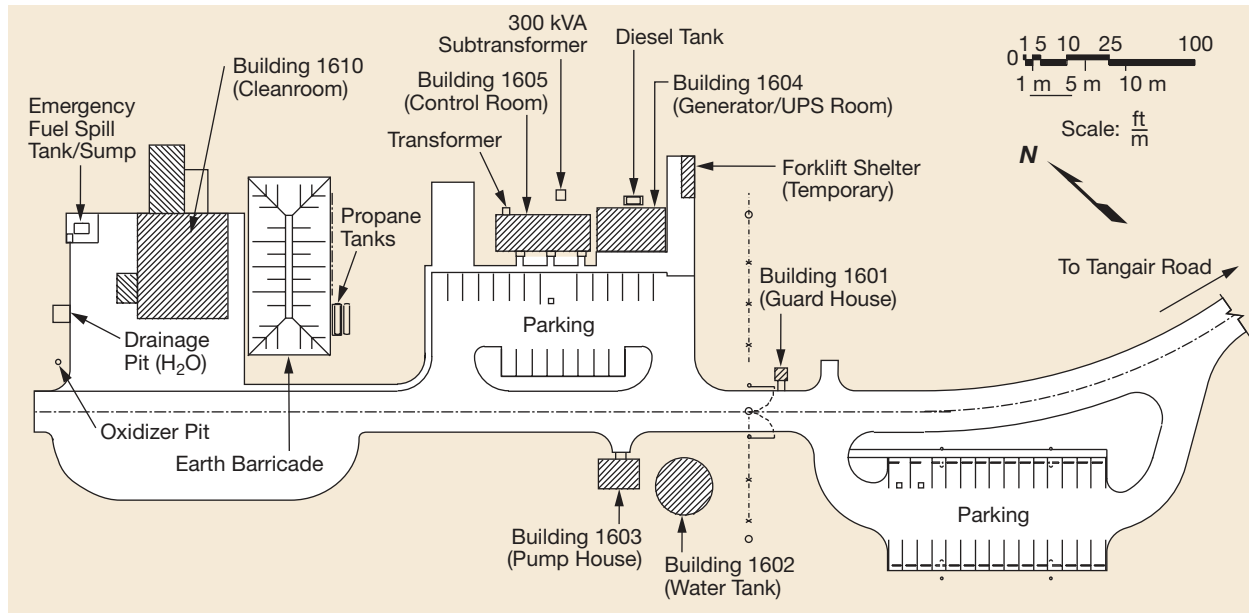


Figure 7-10. NASA Hazardous Processing Facility

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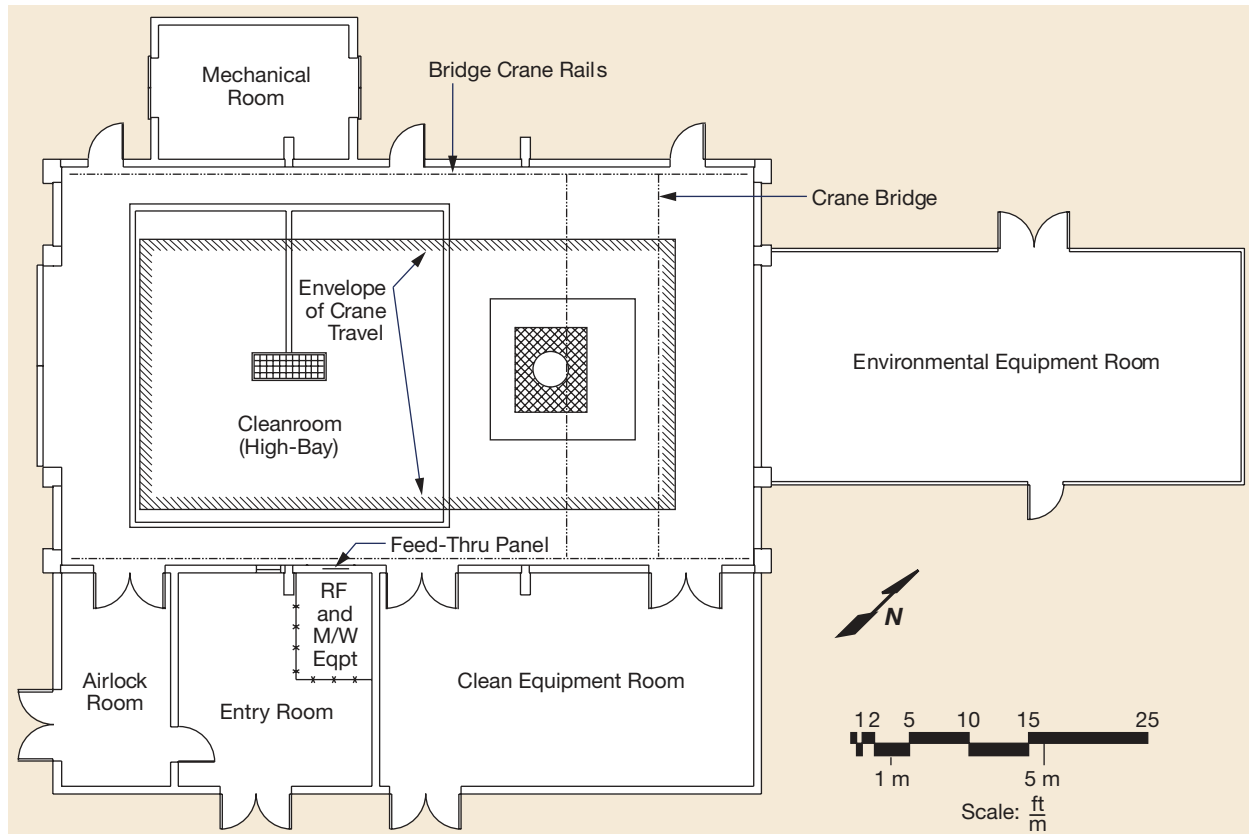


Figure 7-11. NASA Hazardous Processing Facility (Building 1610)

The HPF (Figure 7-11), is an approved ordnance-handling facility and was constructed for dynamic balancing of spacecraft and solid rocket motors. It is 17.7 m (58 ft) long by 10.4 m (34 ft) wide by 13.7 m (45 ft) high with personnel access doors and a flight equipment entrance door

opening that is 5.2 m (17 ft) wide and 9.1 m (29 ft 9 in.) high. The facility is equipped for safe handling of the hydrazine-type propellants used on many space vehicles for attitude control and supplemental propulsion. In the high bay, there is an overhead bridge crane with two 4545-kg (5-ton) capacity hoists. The working hook height is 10.4 m (34 ft). The spreader beam reduces the available hook height by 1 m (3 ft 2 in.) The HPF is a class 10,000 clean facility with positive pressure maintained in the room to minimize contamination from the exterior atmosphere. Positive-pressure clean air is provided by the air circulation and conditioning system located in a covered environmental equipment room at the rear of the building. Personnel gaining entry to the cleanroom from the entry room must wear appropriate apparel and must pass through an airlock. The airlock room has an access door to the exterior so that equipment can be moved into the cleanroom.

7.2.2.2 Control Room Building. The control room building (Figure 7-12) contains a control room, an operations ready room, a fabrication room, and a mechanical/electrical room. The control console for the dynamic balancing system is located within the control room. Television monitors and a two-way intercommunications system provide continuous audio and visual monitoring of operations in the spin test building.

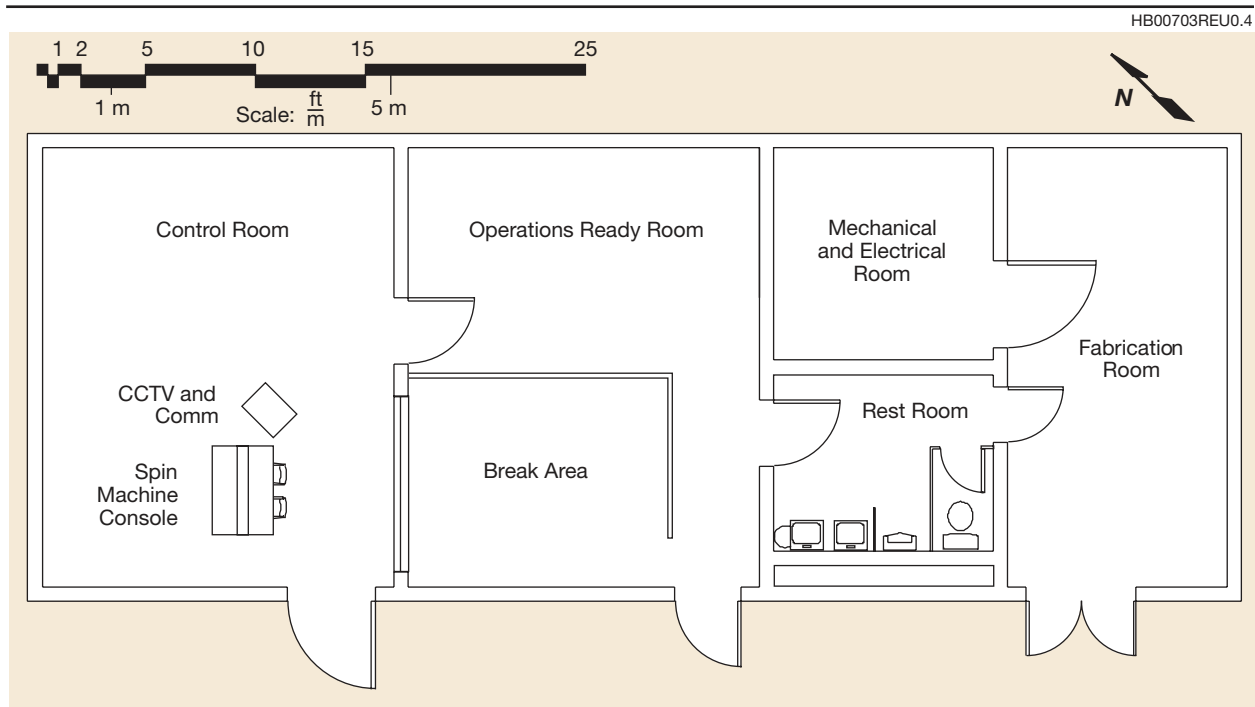


Figure 7-12. Control Rooms (Building 1605)

7.2.2.3 UPS/Generator Building. The UPS/generator building houses a 415-hp, autostart/autotransfer diesel generator. The generator produces 350 kVA, 240/208 VAC, 3-phase, 4-wire power. It is capable of carrying the entire facility power load approximately 8 hr after a loss of commercial power without a refueling operation. A 225 kVA uninterruptible power supply is also located in this building, which can carry all on-site power loads (except for HVAC) while the diesel is starting.

7.2.3 Astrotech Space Operations Facilities

The Astrotech facilities are located on 24.3 hectares (60 acres) of land at Vandenberg AFB approximately 3.7 km (2 mi) south of the Delta II launch complex (SLC-2) along Tangair Road (Figure 7-13). The complex is situated at the corner of Tangair Road and Red Road adjacent to the Vandenberg AFB runway. A complete description of the Astrotech facilities can be found on the Astrotech Web site: www.spacehab.com/aso/reference.htm.

7.2.4 Spaceport Systems International (SSI) Facilities

The SSI payload processing facility is located at SLC-6 on South Vandenberg adjacent to the SSI commercial spaceport. This processing facility is called the integrated processing facility (IPF) because both booster components and payloads (satellite vehicles) can be processed in the building at the same time. A complete description of the SSI facilities can be found on the Spaceport Systems International Web site: www.calospace.com.

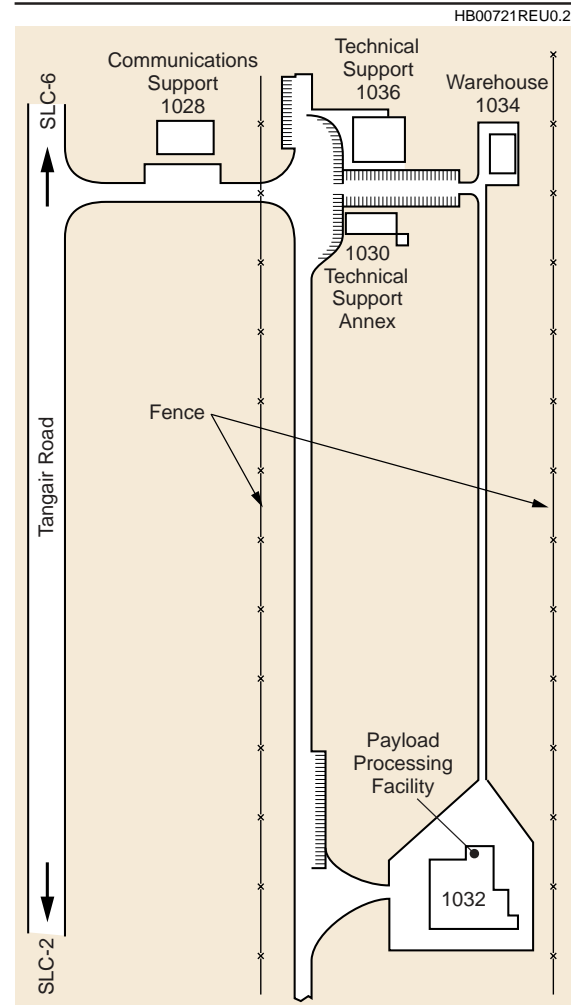


Figure 7-13. Astrotech Space Operations Facilities

7.3 SPACECRAFT TRANSPORT TO LAUNCH SITE

After completion of preparations in one of the spacecraft processing facilities, the flight-configured spacecraft is installed in a transportation handling can and moved to SLC-2 to be mated to the Delta II launch vehicle. Boeing provides the transportation container (Figure 7-14) to support transportation of the spacecraft to the launch pad. The container (ground handling can) can be configured for either three-stage or two-stage missions. The height of the handling can varies according to the number of cylindrical sections required for a safe envelope around the spacecraft.

The spacecraft, inside the handling can, is slowly transported to the launch pad on an air-ride trailer. The trailer travels in a convoy, with Delta Program-provided tractors and security personnel. The ground handling can is purged with GN₂ to reduce the relative humidity of the air inside the container and to maintain a slight positive pressure. Temperature is maintained at acceptable levels when transporting the spacecraft by selecting the time of day at which movement occurs and by adding protective covers. When required by mission specifications, the transportation environment is monitored with recording instrumentation. In addition, special handling can penetrations (feedthroughs, quick disconnects, etc.) may be provided, if required, to support customer-provided spacecraft support equipment (e.g., instrument purges, battery trickle charges).

7.4 SPACE LAUNCH COMPLEX 2

SLC-2 (Figure 7-15) consists of one launch pad (SLC-2W), a blockhouse, a Delta operations building, shops, a supply building, and other facilities necessary to prepare, service, and launch the Delta vehicle. An aerial view of SLC-2 is shown in Figure 7-15.

Because all operations in the launch complex involve or are conducted in the vicinity of liquid or solid propellants and/or explosive ordnance devices, the number of personnel permitted in the area, safety clothing to be worn, type of activity permitted, and equipment allowed are strictly regulated. Adherence to all safety regulations is required. Briefings on all these subjects are given to those required to work in the launch complex area.

The SLC-2 MST (Figure 7-16) is a 54.3-m (178-ft)-high structure with nine working levels designated as A, B, C, 1, 2, 3, 4, 5, and 6. An elevator gives access to eight of the levels, A through C and 1 through 5. The white room (spacecraft area) encloses Levels 4, 5, and 6 (Figures 7-17 and 7-18). However, Level 4 is not typically used for spacecraft work. Levels 4 and 5 are fixed platforms, and Level 6 is an adjustable platform with a range of 399 cm (157 in.) (Figure 7-19). The white room enclosure is constructed of RF-transparent panels. An internal bridge crane with a 4545-kg (5-ton) capacity is used for fairing and spacecraft equipment that must be moved within the MST. It has a maximum hook height of 9.83 m (32 ft 4 in.) above Level 5 (Figure 7-20). Space is available on Level 5 for spacecraft GSE. Placement of the GSE must be coordinated with the Delta Program Office and appropriate seismic restraints provided.

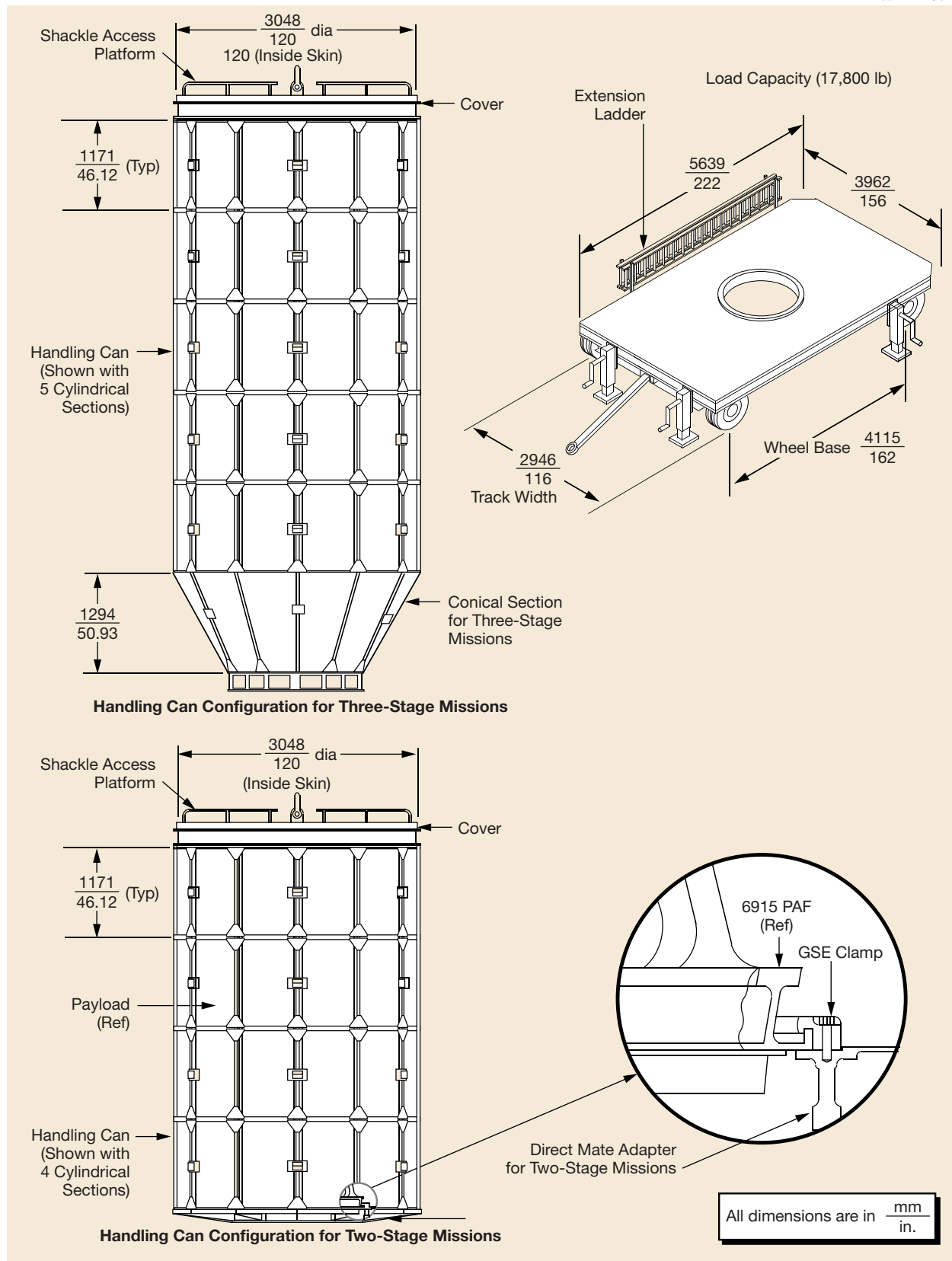


Figure 7-14. Second-Stage Assembly Ground Handling Can and Transporter



Figure 7-15. Space Launch Complex-2 at VAFB—Aerial View Looking West

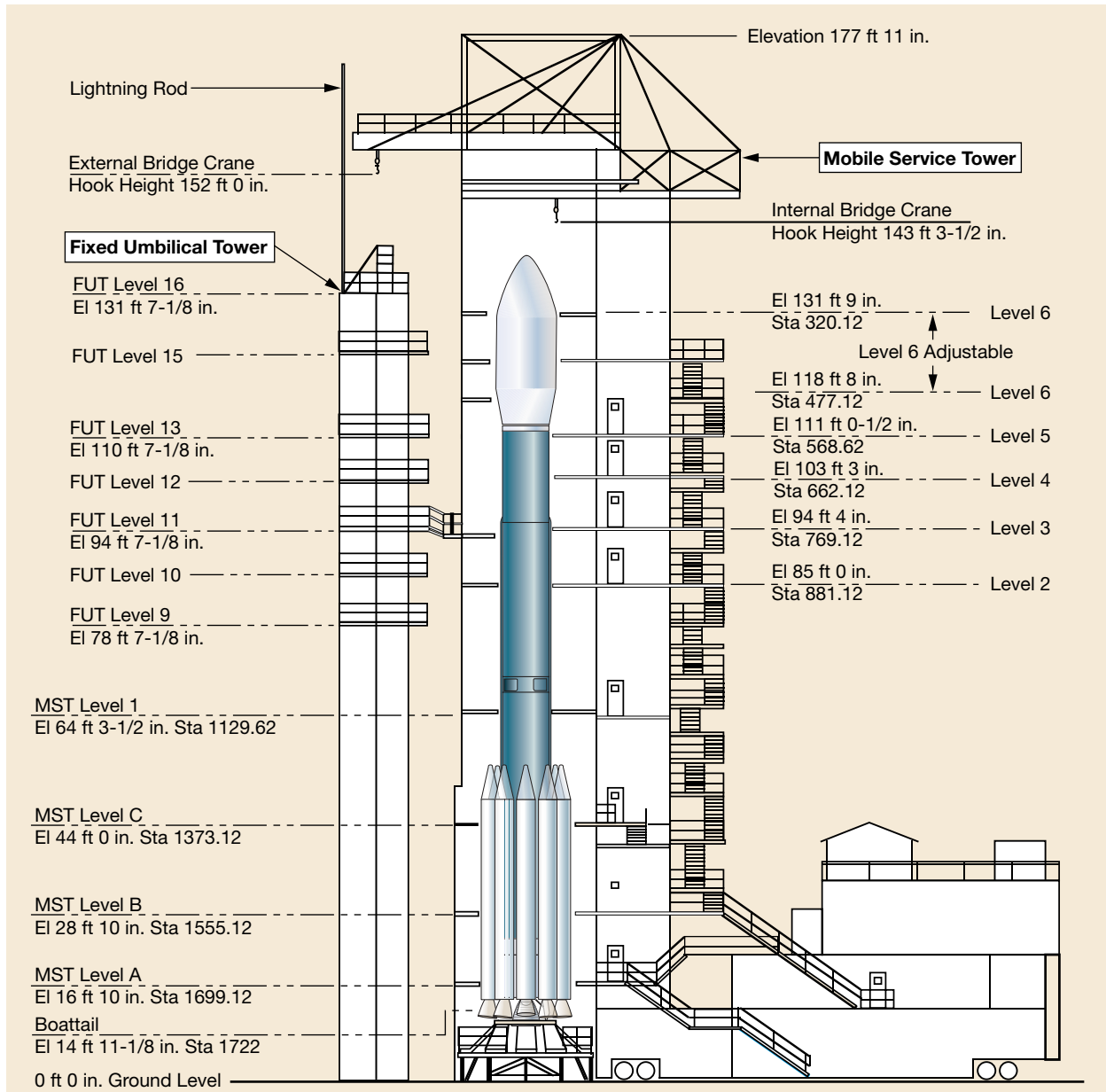


Figure 7-16. SLC-2 Mobile Service Tower/Fixed Umbilical Tower Elevations

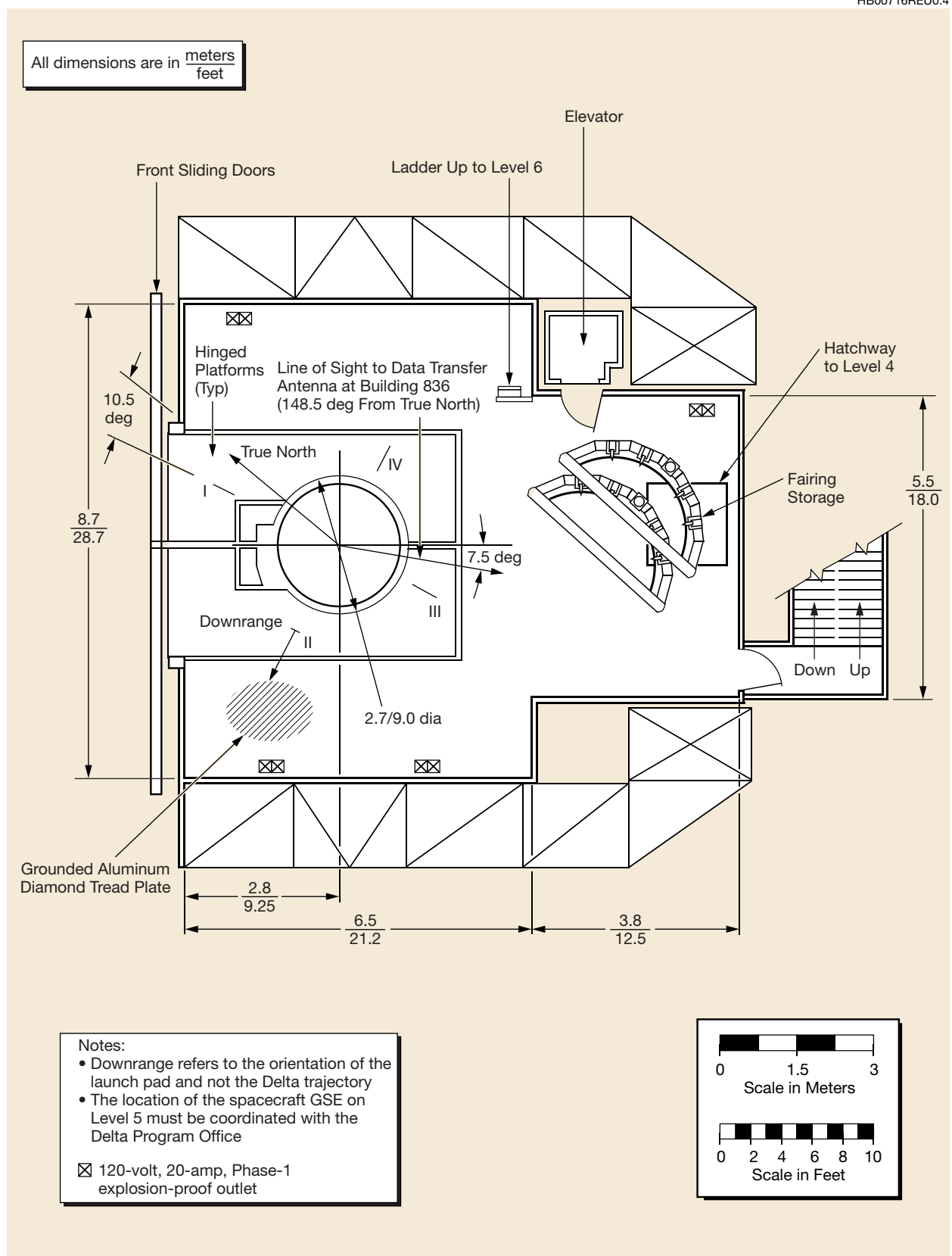


Figure 7-17. Level 5 of SLC-2 Mobile Service Tower—Plan View

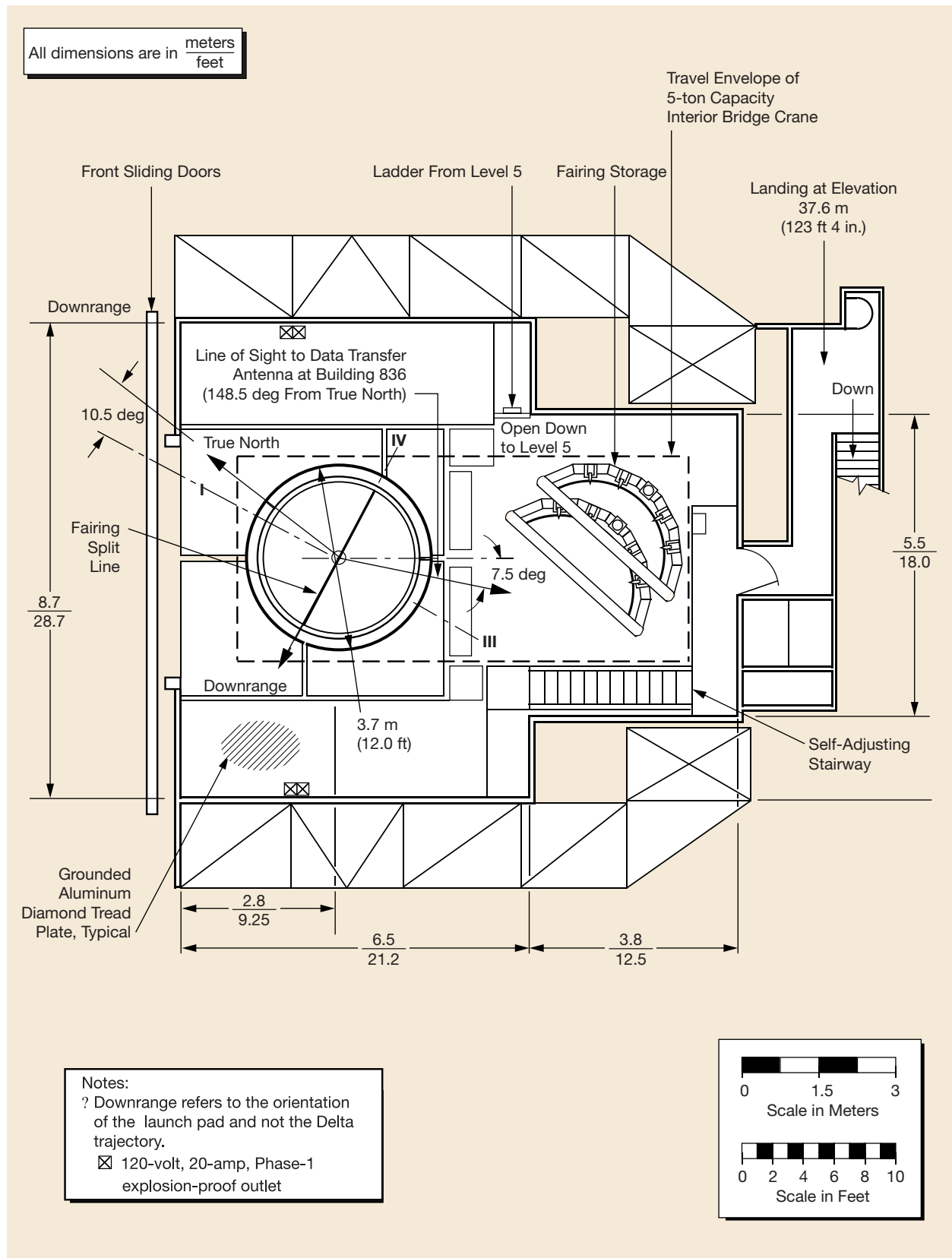


Figure 7-18. Level 6 of SLC-2 Mobile Service Tower—Plan View

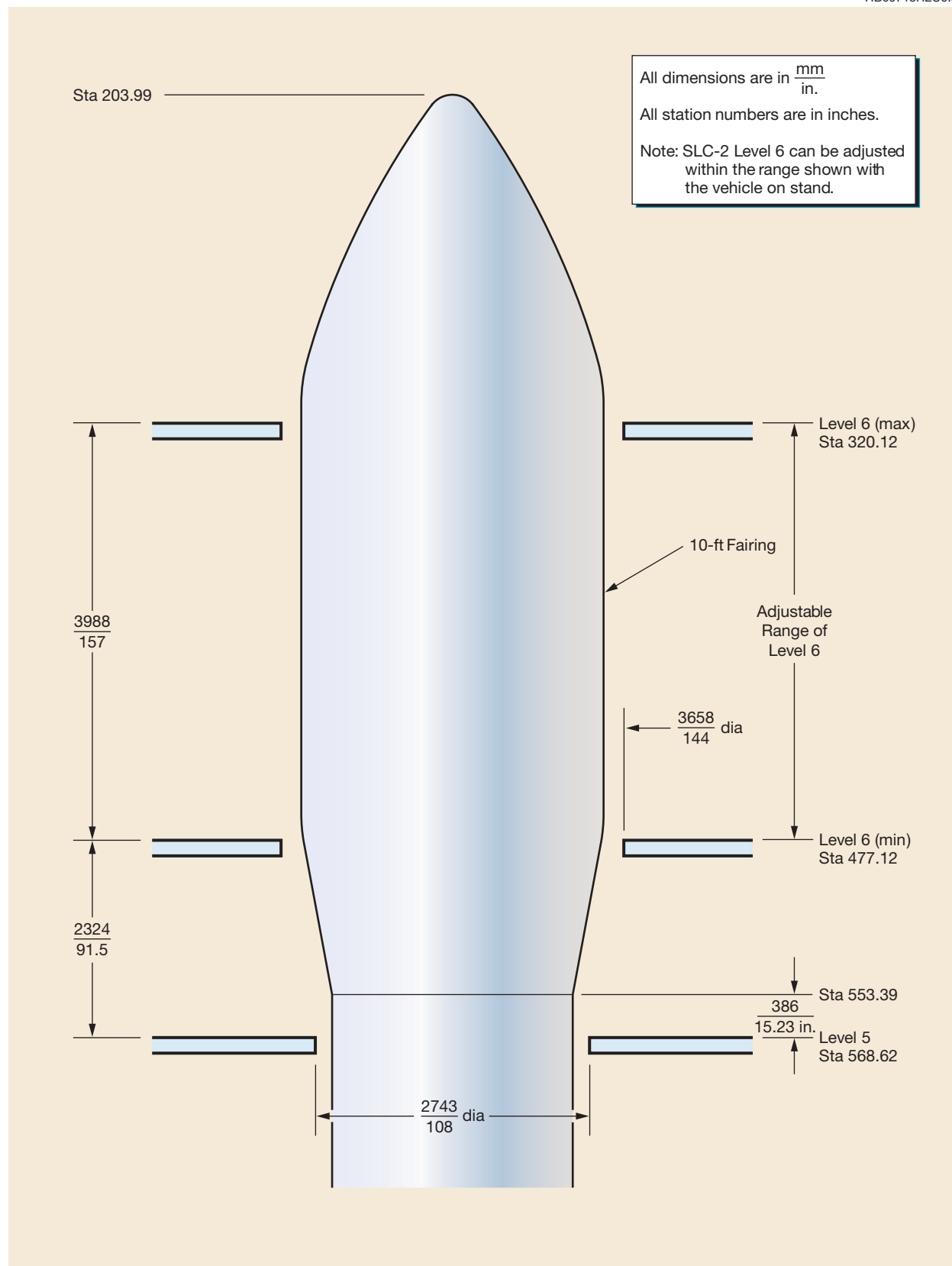


Figure 7-19. Spacecraft Work Levels in SLC-2 Mobile Service Tower—VAFB

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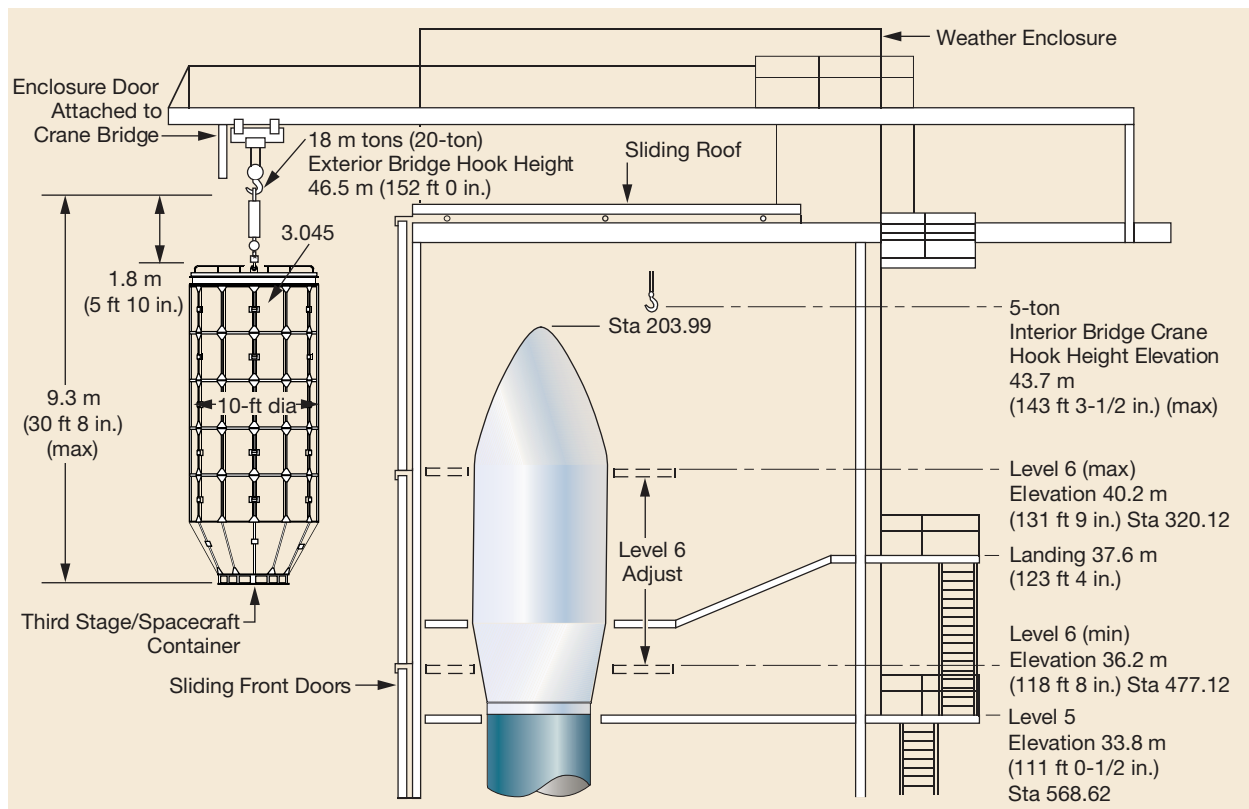


Figure 7-20. Whiteroom Elevations and Hook Heights—SLC-2 Mobile Service Tower

The entire MST is constructed to meet explosion-proof safety requirements. The restriction on the number of personnel admitted to the white room is governed by safety requirements as well as the limited amount of work space and the cleanliness level required on the spacecraft levels.

Launch operations are controlled from the blockhouse and the RLCC, which are equipped with vehicle monitoring and control equipment. Space is allocated for use by other equipment and spacecraft personnel in the RLCC, electrical equipment building (EEB) (Figure 7-21), and blockhouse. The EEB is located at the base of the FUT. In addition, a spacecraft console (Figure 7-22) is available that will accept a standard rack-mounted panel. Terminal board connections in the console provide electrical connections to the spacecraft umbilical wires. There are also a limited number of 28 VDC discrete commands circuits and discrete talkbacks circuits that provide the capability to remotely control and monitor spacecraft equipment in the EEB from the RLCC (Figure 7-23).

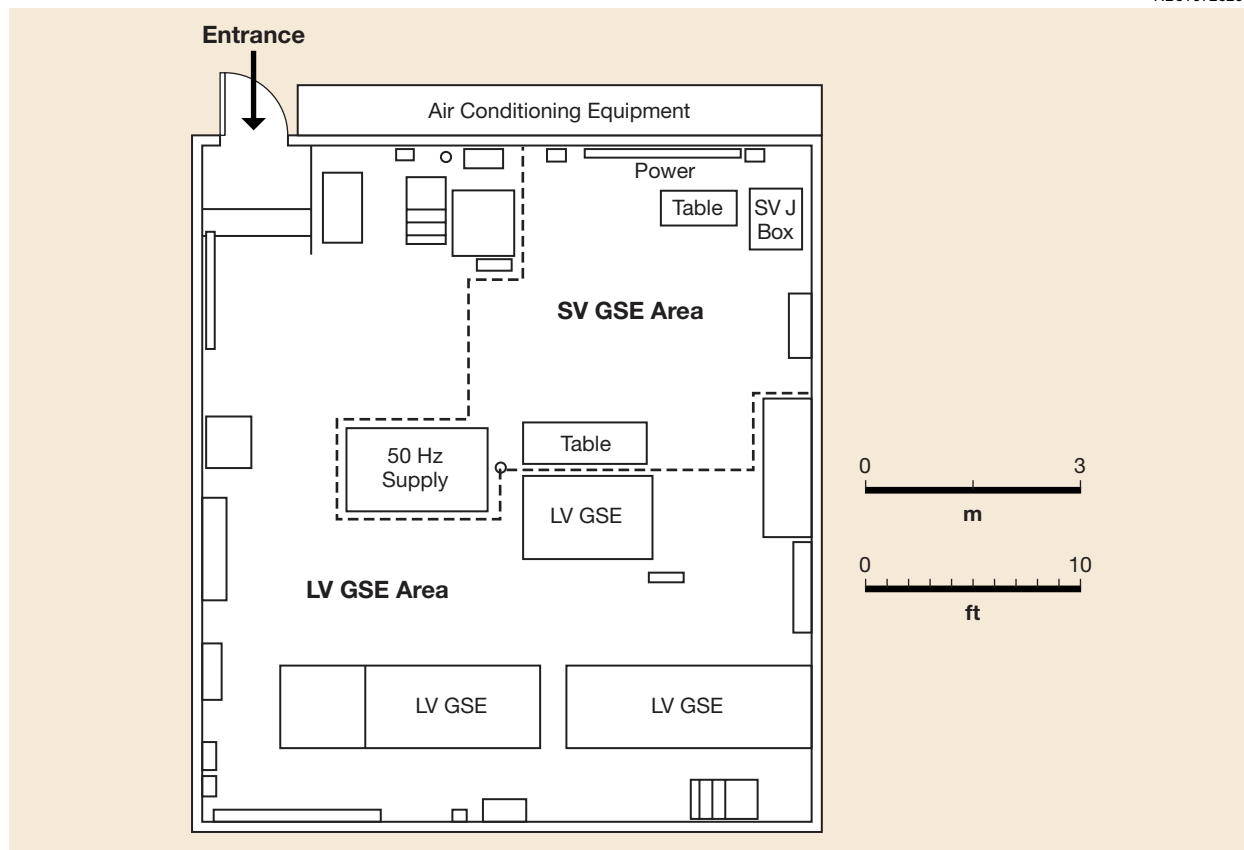


Figure 7-21. SLC-2 Electrical Equipment Building (EEB)

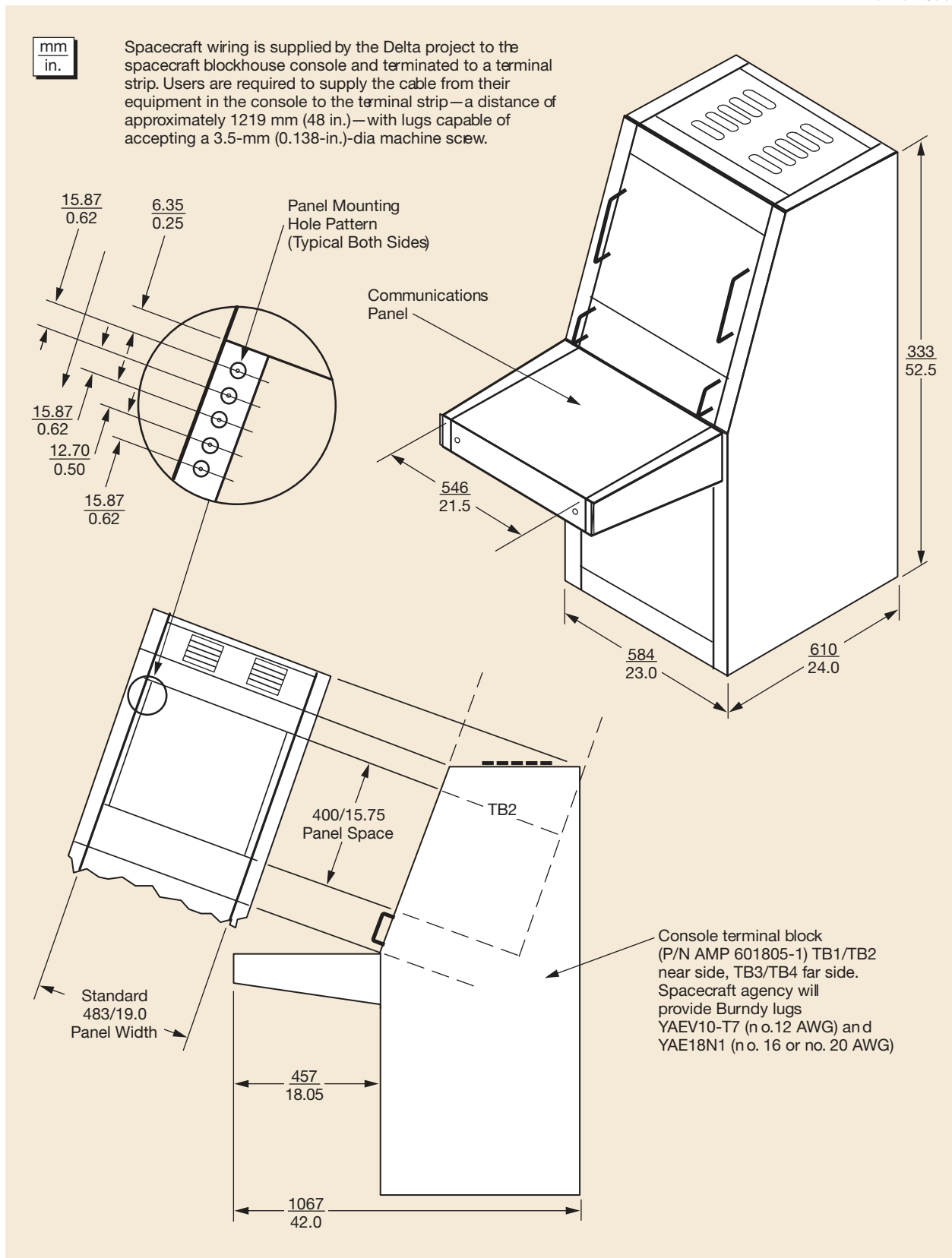


Figure 7-22. Spacecraft Blockhouse Console—Western Range

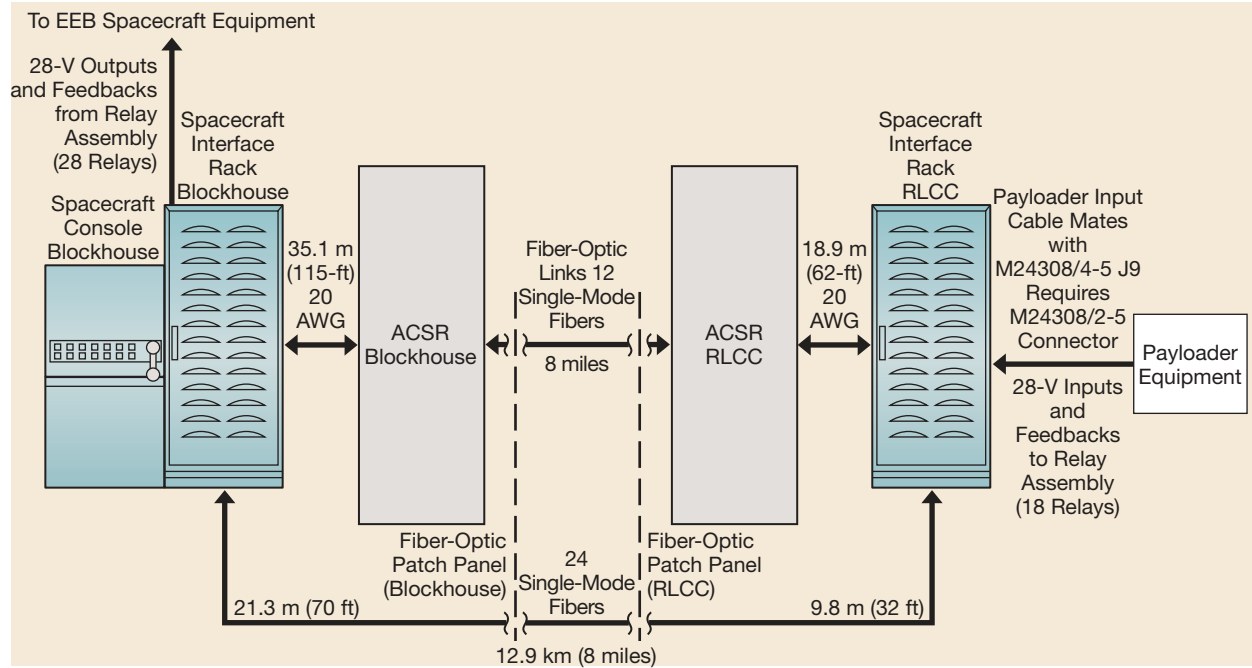


Figure 7-23. Auxiliary Control System Rack (ACSR) Blockhouse-to-RLCC Block Diagram

Located in the EEB and FUT are the spacecraft rack and the umbilical adapter junction box (J-box), respectively (Figure 7-24).

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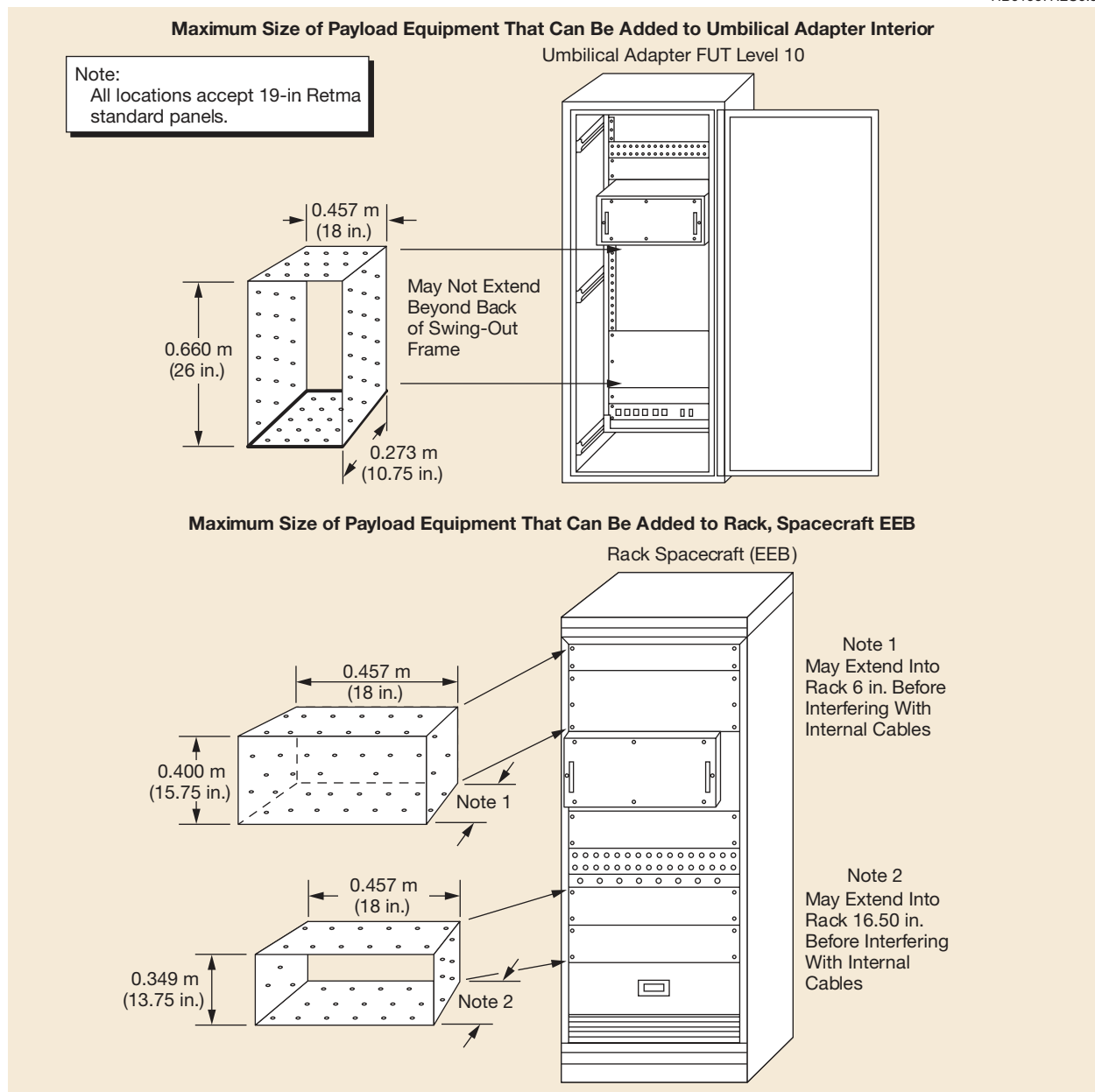


Figure 7-24. SLC-2 Spacecraft Rack and Umbilical Adapter J-Box

7.5 SUPPORT SERVICES

7.5.1 Launch Support

For countdown operations, the launch team is located in the remote launch control center in building 8510, and in the MDC and LVDC in building 836 and 840, with support from other base organizations.

7.5.1.1 Mission Director Center (Building 836). The Mission Director Center described in Section 7.2.1.1 and Figure 7-8, provides the necessary seating, data display, and communications to control the launch process. Seating is provided for key personnel from the Delta Program, the Western Range, and the spacecraft control team. For NASA launches, key NASA personnel will also occupy space in the mission director center.

7.5.1.2 Space Launch Complex 2 Blockhouse. Prelaunch operations are controlled from the blockhouse, which is equipped with vehicle monitoring and control equipment. Space is also allocated for the spacecraft blockhouse consoles and console operators. Terminal board connections in the spacecraft blockhouse junction box provide electrical connection to the spacecraft umbilical wires.

7.5.1.3 Remote Launch Control Center (RLCC) (Rooms 147 and 314 in Building 8510). Crew certification, second-stage propellant loading (approximately 3 days before launch), and all subsequent launch operations are controlled from the RLCC, which is equipped with a duplicate set of vehicle-monitoring-and-control equipment. Limited space is also allocated for spacecraft consoles and console operators in the RLCC.

7.5.1.4 Launch Decision Process. The launch decision process is made by the appropriate management personnel representing the spacecraft, launch vehicle, NASA, and range. Figure 7-25 shows the communications flow required to make the launch decision. For NASA missions, a mission director, launch management advisory team, engineering team, and quality assurance personnel will also participate in the launch decision process.

7.5.2 Operational Safety

Safety requirements are covered in Section 9 of this document. In addition, it is the operating policy at Boeing that all personnel will be given safety orientation briefings prior to entrance to hazardous areas such as SLC-2. These briefings will be scheduled by the Delta Program Office spacecraft coordinator and presented by the appropriate safety personnel.

7.5.3 Security

7.5.3.1 Astrotech Security. Physical security at the Astrotech facilities is provided by chain-link perimeter fencing, door locks, access badges, and guards. Spacecraft security requirements will be implemented through the Delta Program security coordinator.

7.5.3.2 SSI Security. Physical security at the SSI facilities is provided by chain-link perimeter fencing, a card-key entry system and cipher-locked doors, access badges, and guards. Each payload checkout cell security is independent of the other two cells and of the high bay. Spacecraft security requirements will be implemented through the Boeing security coordinator.

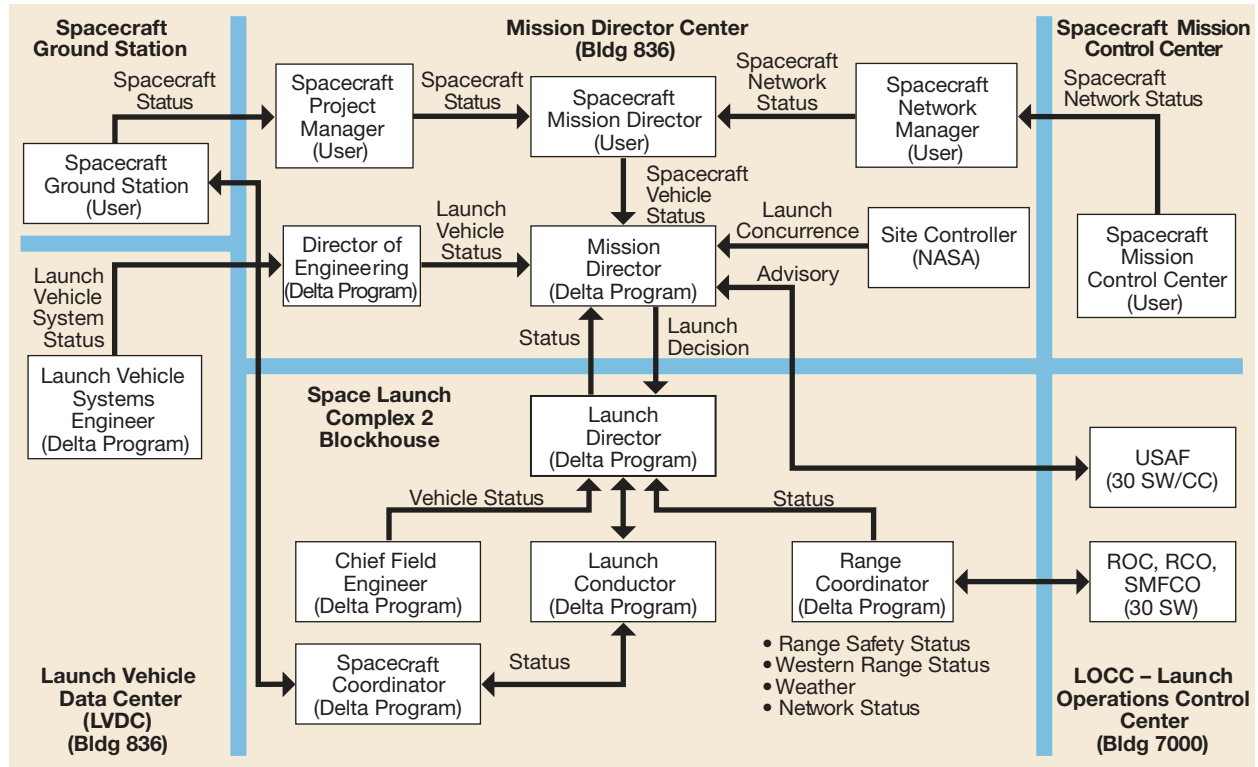


Figure 7-25. Launch Decision Flow for Commercial Missions—Western Range

7.5.3.3 Launch Complex Security. SLC-2 physical security is ensured by perimeter fencing, guards, access badges, and access lists. The MST white room is controlled with combination and key locks on entry-controlled doors. Access to spacecraft can be controlled by a security guard on the MST third level with badges and access lists.

7.5.3.4 VAFB Security. For access to VAFB, U.S. citizens must provide to the Delta Program security coordinator full name with middle initial if applicable, social security number, company name, and dates of expected arrival and departure. Delta Program security will arrange for entry authority for commercial missions or for individuals sponsored by the Delta Program. Access by NASA personnel or NASA-sponsored foreign nationals is coordinated by NASA KSC (at VAFB) with the USAF at VAFB. Access by other U.S. government-sponsored foreign nationals is coordinated by their sponsor directly with the USAF at VAFB. For non-United States citizens, clearance information (name, nationality/citizenship, date and place of birth, passport number and date/place of issue, visa number and date of expiration, and title or job description) must be furnished to the Delta Program Office not later than 2 weeks prior to the VAFB entry date. Government-sponsored individuals must follow NASA or U.S. government guidelines as appropriate. The spacecraft coordinator will furnish visitor identification documentation to the appropriate agencies. After Delta Program security gets clearance approval, entry to VAFB will be the same as for U.S. citizens.

7.5.4 Field-Related Services

Boeing employs certified equipment drivers, welders, riggers, and explosive ordnance handlers, in addition to personnel experienced in most electrical and mechanical assembly skills such as torquing, soldering, crimping, precision cleaning, and contamination control. Boeing has under its control a machine shop, metrology laboratory, precision cleaning facility, and proof-loading facility. Boeing operational team members are familiar with USAF and NASA payload processing facilities at VAFB and can offer all of these skills and services to the spacecraft project during the launch program.

7.6 DELTA II PLANS AND SCHEDULES

7.6.1 Mission Plan

A mission plan (Figure 7-26) is developed for each launch campaign showing major tasks on a weekly timeline format. The plan includes launch vehicle activities, prelaunch reviews, and spacecraft processing area occupancy times.

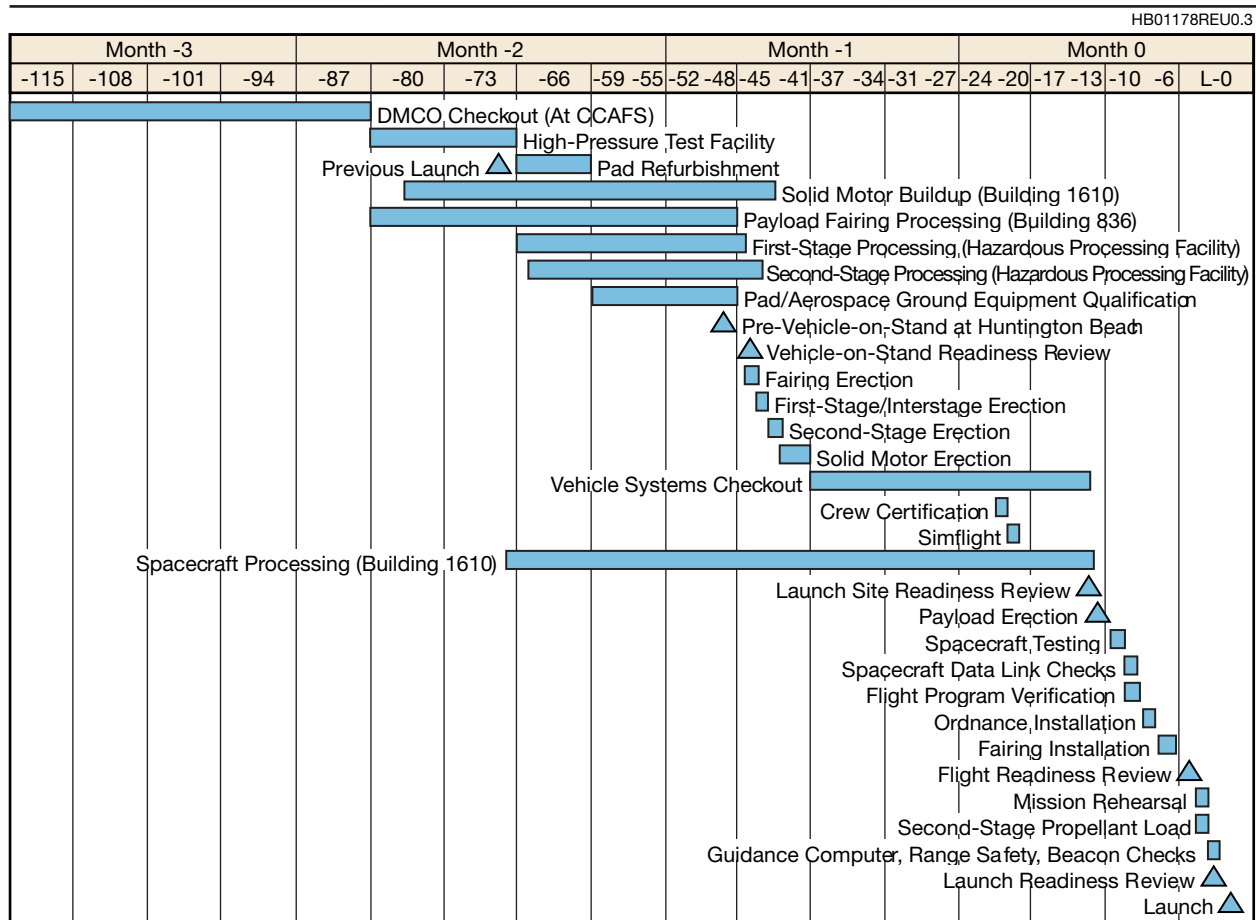


Figure 7-26. Typical Mission Plan

7.6.2 Integrated Schedules

The schedule of spacecraft activities before integrated activities in the payload processing facility varies from mission to mission. The extent of spacecraft field testing varies and is determined by the spacecraft agency. Spacecraft/launch vehicle schedules are similar from mission to mission from the time of spacecraft weighing until launch.

Daily schedules are prepared on hourly timelines for these integrated activities. These schedules cover 4 days of integrated effort in the payload processing facility and 8 days of launch countdown activities. Payload processing facility tasks include spacecraft weighing, spacecraft/third-stage mate and interface verification, and transportation can assembly around the combined payload. The countdown schedules provide a detailed hour-by-hour breakdown of launch pad operations, illustrating the flow of activities from spacecraft erection through terminal countdown, and reflecting inputs from the spacecraft project. These schedules comprise the integrating document to ensure timely launch pad operations. Typical schedules of integrated activities from spacecraft weighing in the payload processing facility until launch (Figures 7-27 through 7-39) are shown as launch minus (T-) workdays. Saturdays, Sundays, and holidays are not scheduled workdays and, therefore, are not T- days. The T- days, from spacecraft mate through launch, are coordinated with each spacecraft agency to optimize on-pad testing. All operations are formally conducted and controlled using launch processing documents. The schedule of spacecraft activities during that time is controlled by the Boeing launch operations manager. Tasks involving the spacecraft or tasks requiring that spacecraft personnel be present are shaded for easy identification. A typical mission from VAFB is as follows; spacecraft and third-stage (if applicable) checkout are completed before T-11 day.

T-7 Tasks include transportation to the launch site, erection and mating of the spacecraft/second stage to the Delta II vehicle in the MST whiteroom, whiteroom environment established, disassembly of the ground handling can, and removal of the can segments from the tower (Figure 7-31).

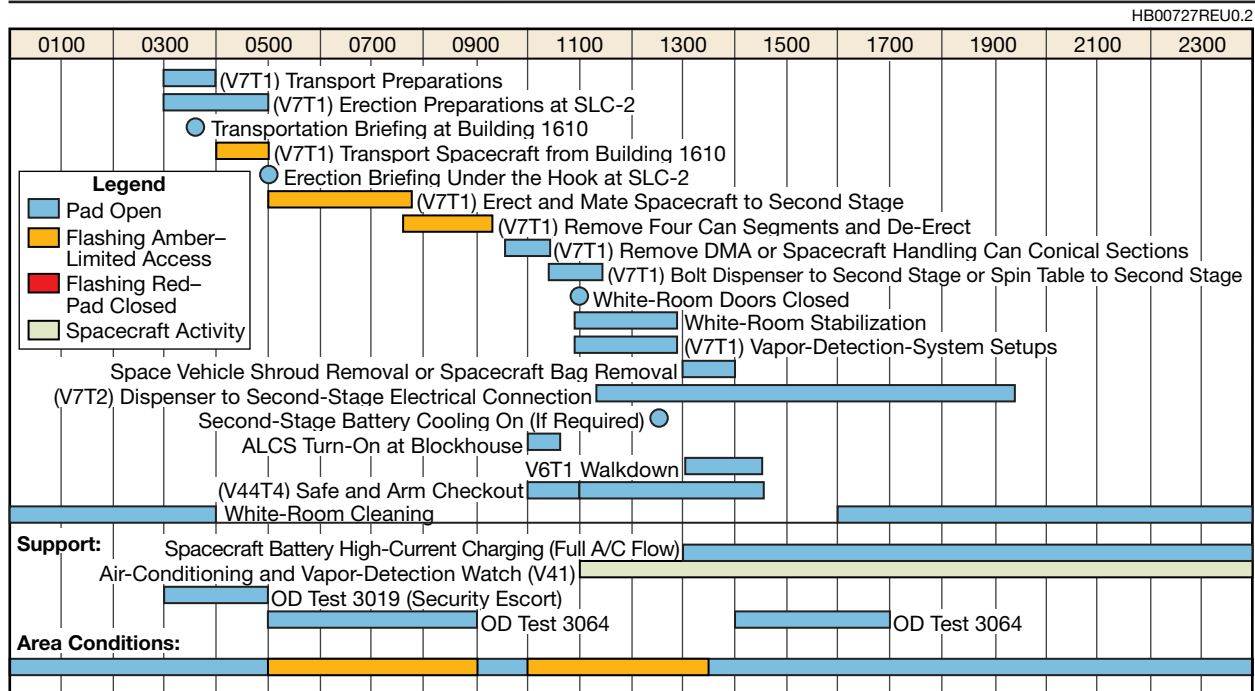


Figure 7-31. Typical Spacecraft Erection (T-7 Day)

T-6 The flight program verification test is performed followed by the vehicle power-on stray-voltage test. Spacecraft systems to be powered at liftoff are turned on during the flight program verification test, and all data are monitored for electro-magnetic interference (EMI) and radio frequency interference (RFI). All spacecraft systems that will be turned on at any time between T-6 day (stray-voltage checks) and T-0 day (spacecraft separation) will be turned on in support of the vehicle power-on stray-voltage test. Spacecraft support of these vehicle system tests is critical in meeting the scheduled launch date. They have priority over other spacecraft testing (Figure 7-32).

T-5 Tasks include Delta II vehicle ordnance installation/connection and preparation for fairing installation (Figure 7-33).

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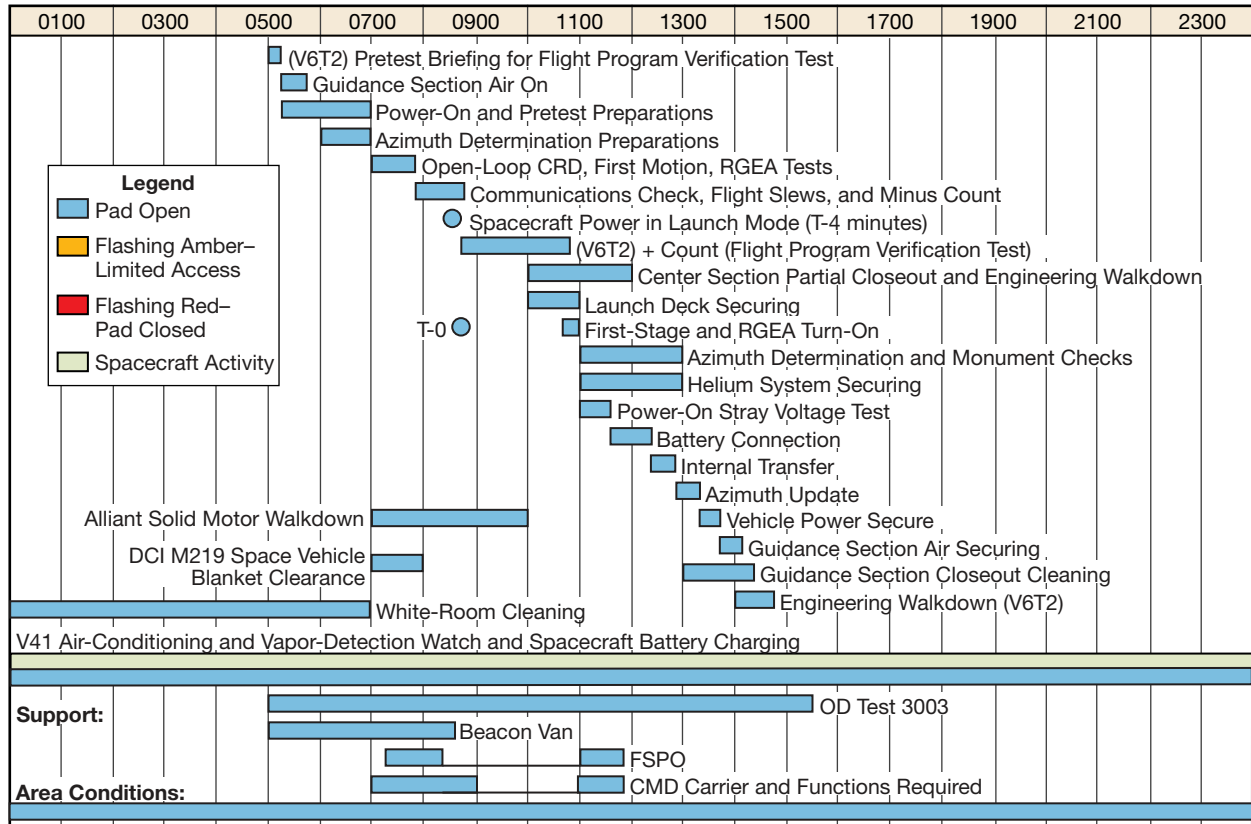


Figure 7-32. Typical Flight Program Verification and Stray Voltage Checks (T-6 Day)

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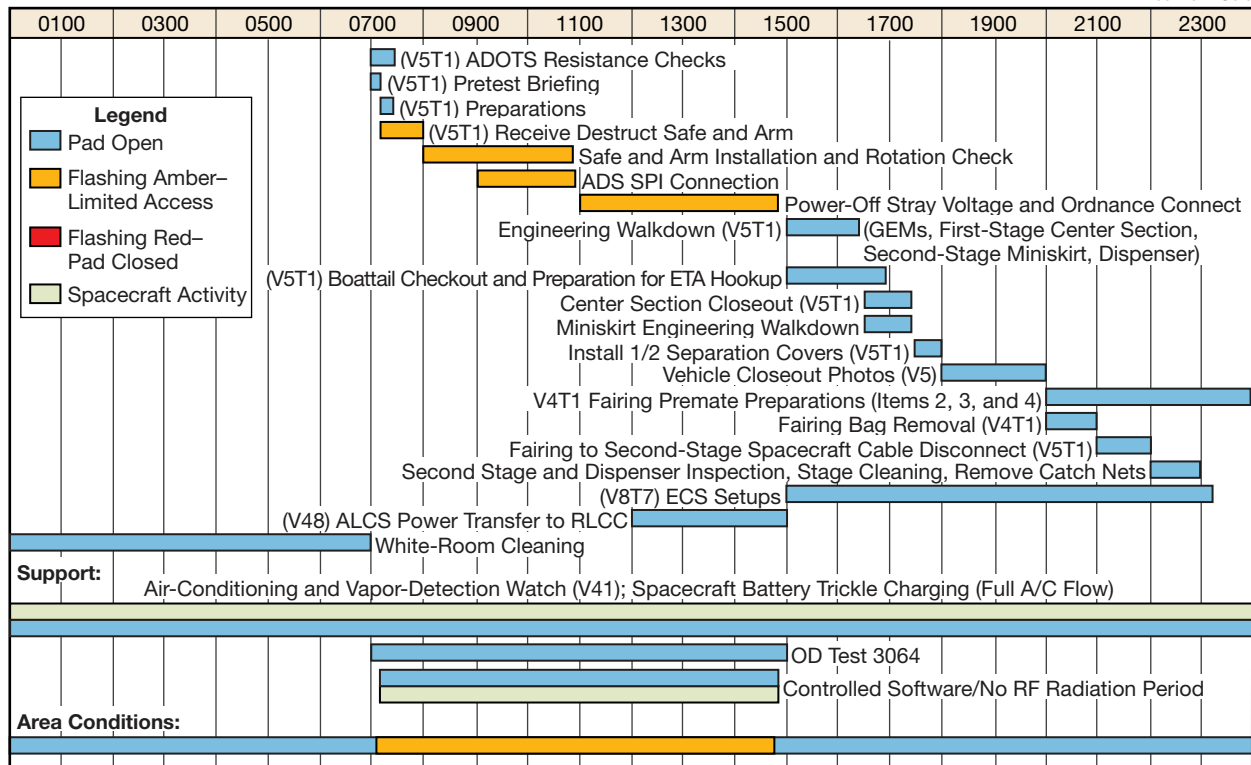


Figure 7-33. Typical Ordnance Installation (T-5 Day)

T-4 Spacecraft final preparations are made prior to fairing installation; included are Delta II second-stage closeout, second-stage propellant servicing preparations, and fairing installation (Figure 7-34).

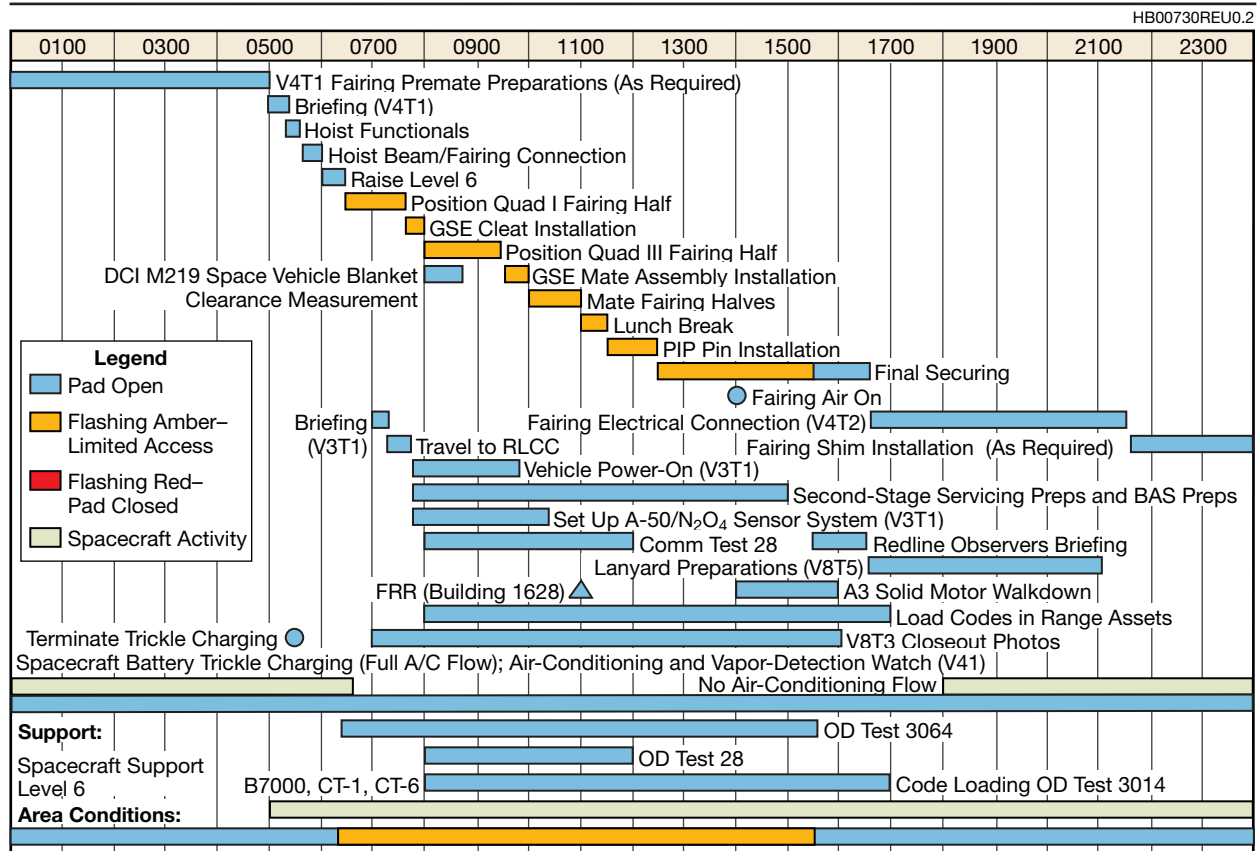


Figure 7-34. Typical Fairing Installation (T-4 Day)

T-3 Propellant is loaded into the second stage, and fairing ordnance is installed (Figure 7-35).

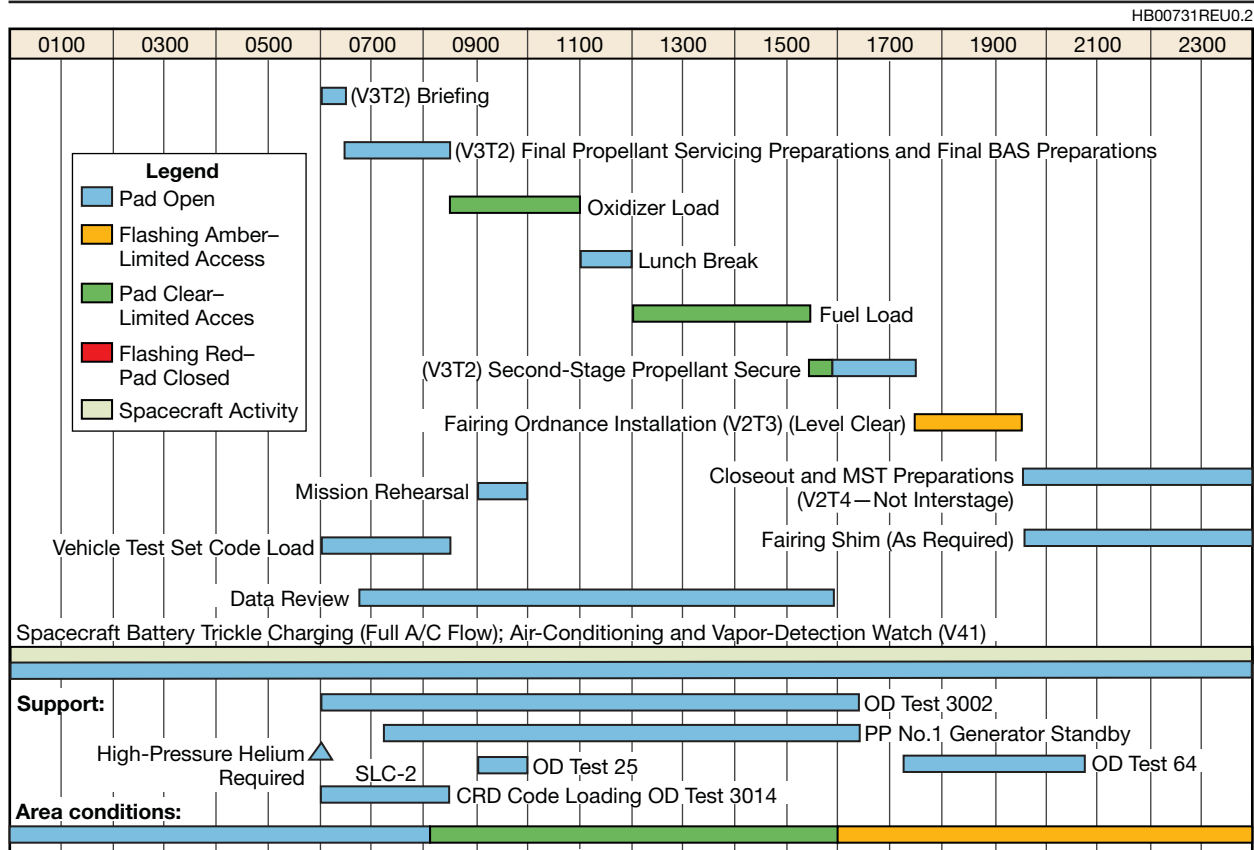


Figure 7-35. Typical Second-Stage Propellant Loading (T-3 Day)

T-2 Tasks include launch vehicle guidance turn-on, C-band beacon readout, guidance system azimuth update, range safety checks, and class A ordnance connection (Figure 7-36).

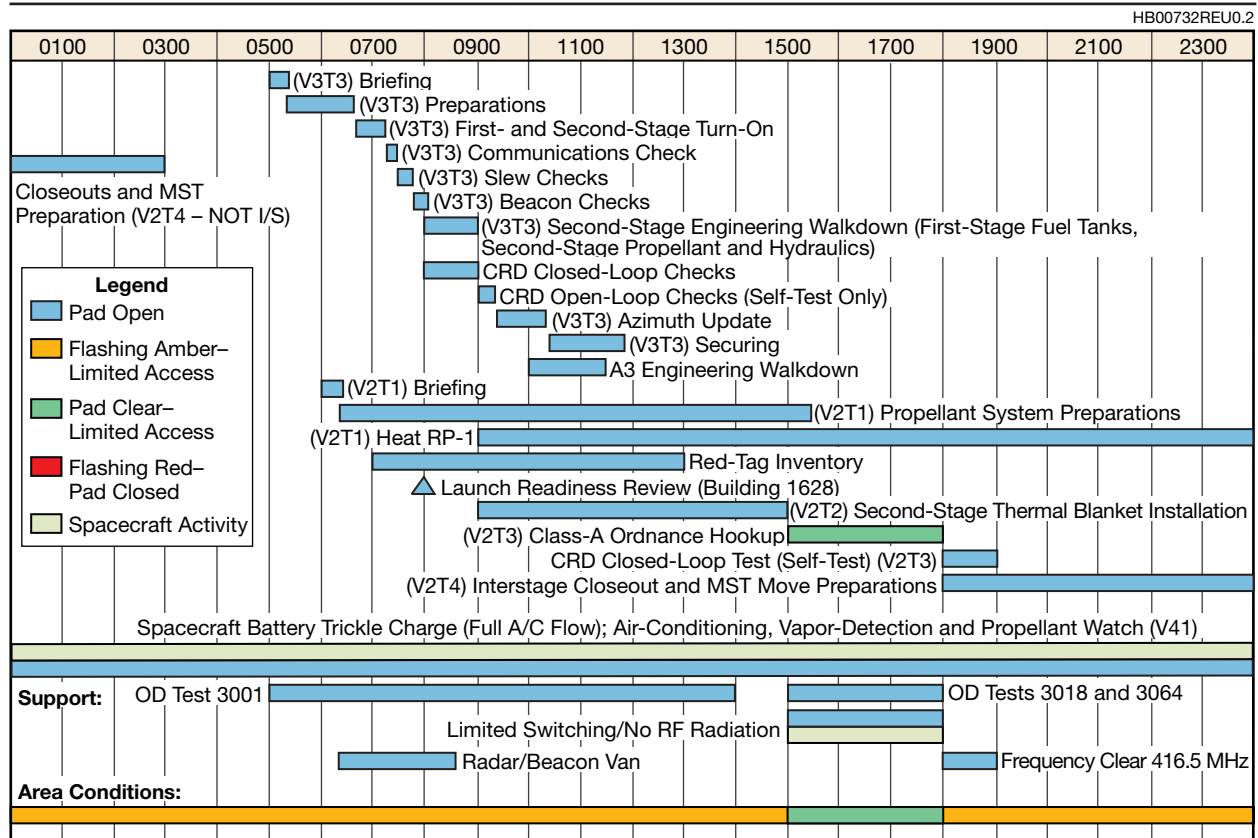


Figure 7-36. Typical Beacon and Range Safety Checks/Class-A Ordnance Connect (T-2 Day)

T-1 Final fairing and whiteroom preparations are made for MST removal, second-stage engine closeout, launch vehicle final preparations, and tower removal (Figures 7-37 and 7-38).

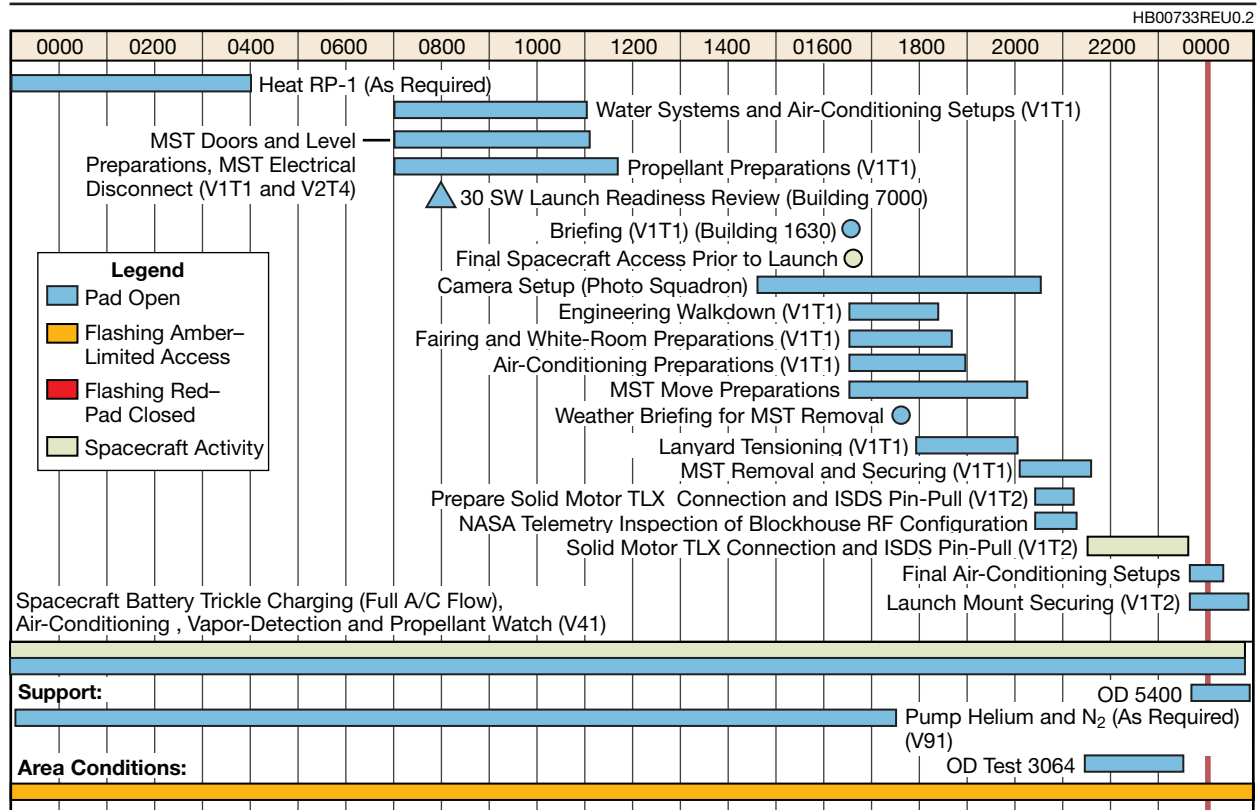


Figure 7-37. Typical Countdown Preparations (T-1 Day)

T-0 Launch day preparations include final spacecraft closeouts and fairing door installation, gantry removal, final arming, terminal sequences, and launch. Spacecraft should be in launch configuration immediately prior to T-4 min and standing by for liftoff. The nominal hold and recycle point is T-4 min. Launch is typically scheduled for a Thursday (Figures 7-38 and 7-39).

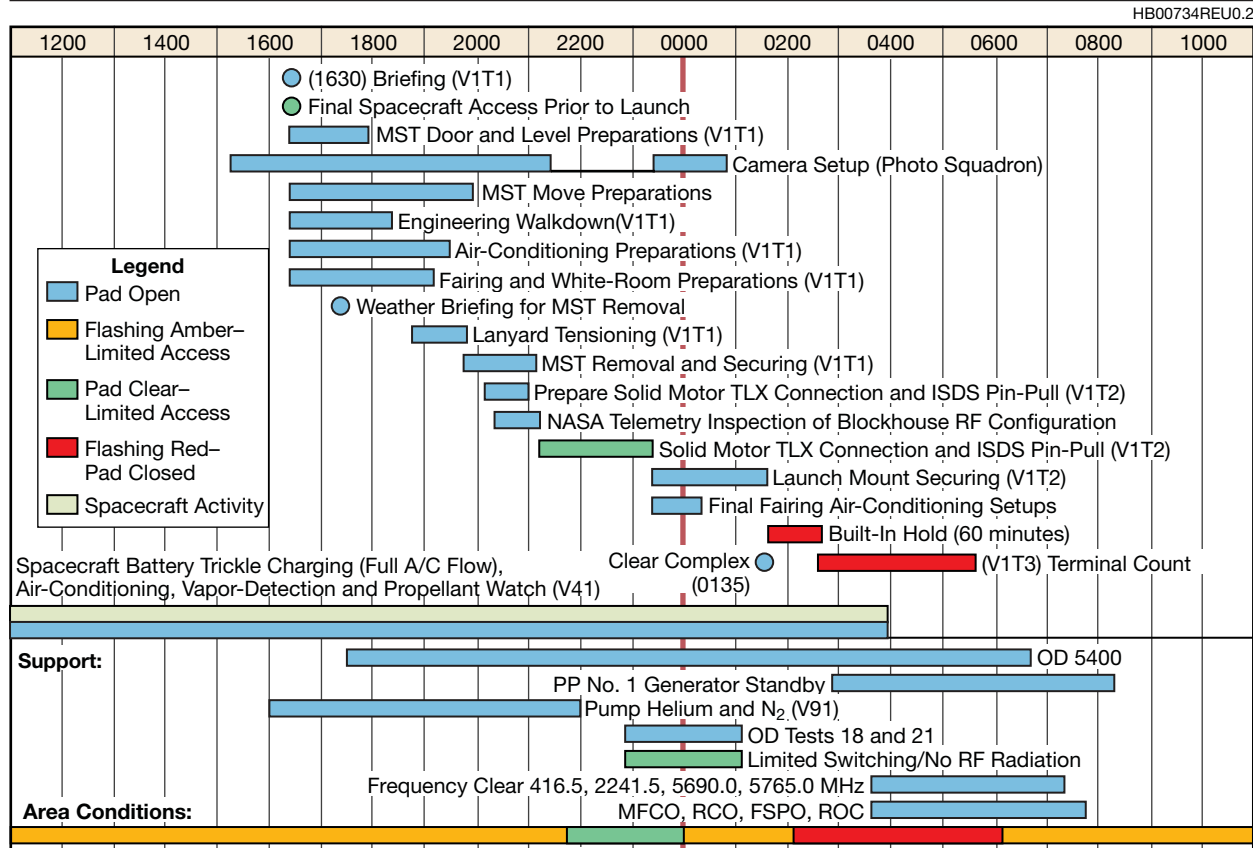
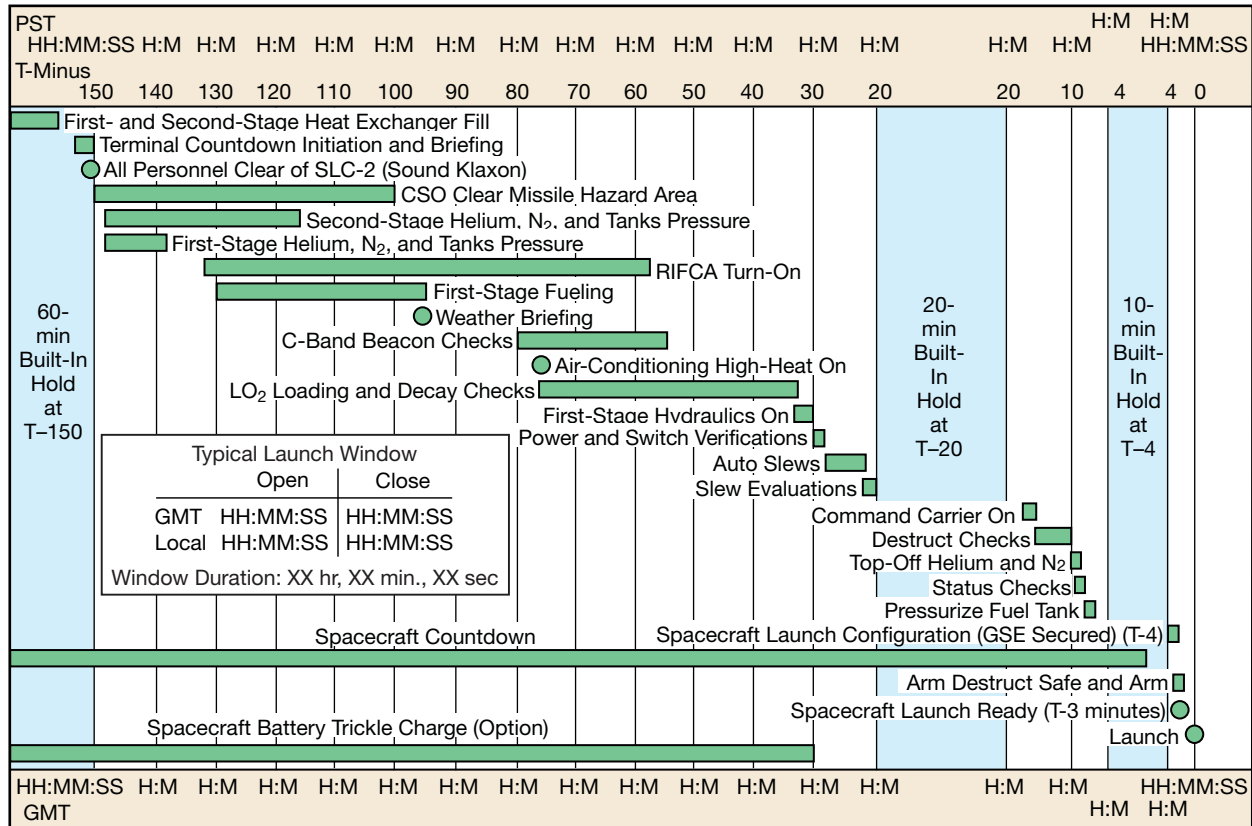


Figure 7-38. Typical Delta Countdown (T-1/T-0 Day)

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7.6.3 Spacecraft Schedules

The spacecraft project will supply schedules to the Boeing spacecraft coordinator, who will arrange support as required.

7.7 DELTA II MEETINGS AND REVIEWS

During the launch scheduling preparation, various meetings and reviews take place. Some of these will require user input while others allow the user to monitor the progress of the overall mission. The Boeing spacecraft coordinator will ensure adequate user participation.

7.7.1 Meetings

Delta Status Meetings. Status meetings are generally held twice a week. They include a review of the activities scheduled and accomplished since the last meeting, a discussion of problems and their solutions, and a review of the mission schedule. Spacecraft representatives are encouraged to attend these meetings.

Daily Schedule Meetings. Daily schedule meetings are held to provide the team members with their assignments and to summarize the previous or current day's accomplishments. These meetings are attended by the launch conductor, technicians, inspectors, engineers, supervisors,

and the spacecraft coordinator. Depending upon testing activities, these meetings are held at the beginning and the end of the first shift.

7.7.2 Prelaunch Review Process

Periodic reviews are held to ensure that the spacecraft and launch vehicle are ready for launch. The following paragraphs discuss the Delta II readiness reviews.

Postproduction Review. This meeting, conducted at Decatur, Alabama, reviews the flight hardware at the end of production and prior to shipment to DMCO at CCAFS, or to VAFB.

Mission Analysis Review. This review is held at Huntington Beach, California, approximately 3 months prior to launch, to review mission-specific designs, studies, and analyses.

Pre-Vehicle-On-Stand (VOS) Review. This review is held at Boeing-Huntington Beach subsequent to the completion of Delta mission checkout (DMCO) and prior to erection of the vehicle on the launch pad. It includes an update of the launch preparation activities since Decatur, the results of the DMCO processing, and any hardware history changes.

Vehicle-On-Stand Readiness Review (VRR). This review is held at the launch site prior to first-stage erection. The status and processing history of the launch vehicle elements and ground support equipment are presented. The primary focus of this review is on the readiness of the first stage, solid motors, interstage, second stage, and fairing for erection and mate on the launch pad. Upon completion of this meeting and resolution of any concerns raised, authorization is given to proceed with erection activities.

Launch Site Readiness Review (LSRR). This review is held at the launch site prior to erection and mate of the second stage and spacecraft to the launch vehicle. The status and entire launch site processing history of the launch vehicle elements and ground support equipment are reviewed. The primary focus of this review is on the readiness of the launch vehicle for erection and mate of the spacecraft to the second stage. Upon completion of this meeting and resolution of any concerns raised, authorization is given to proceed with spacecraft transfer to the launch pad, immediately followed by erection and mate with the second stage.

Flight Readiness Review (FRR). This review, typically held on T-3 day, is an update of activities since the pre-VOS and is conducted to determine that checkout has shown that the launch vehicle and spacecraft are ready for countdown and launch. Upon completion of this meeting, authorization is given to proceed with the loading of second-stage propellants. This review also assesses the readiness of the range to support launch and provides a predicted weather status.

Launch Readiness Review (LRR). This review is normally held one day prior to launch and provides an update of activities since the FRR. All agencies and contractors are required to provide a ready-to-launch statement. Upon completion of this meeting and resolution of any concerns raised, an authorization to enter terminal countdown is given.

Section 8 PAYLOAD INTEGRATION

This section describes the payload integration process, the supporting documentation required from the spacecraft customer, and the resulting analyses provided by the Delta Program Office.

8.1 INTEGRATION PROCESS

The integration process developed by the Delta Program is designed to support the requirements of both the launch vehicle and the payload. We work closely with our customers to tailor the integration activity to meet their individual program requirements. The typical integration process (Figure 8-1) encompasses the entire life of the launch vehicle/payload integration activities; L-date is defined as calendar day, including workdays and scheduled non-workdays such as holidays. At its core is a streamlined series of documents, reports, and meetings that are flexible and adaptable to the specific requirements of each program.

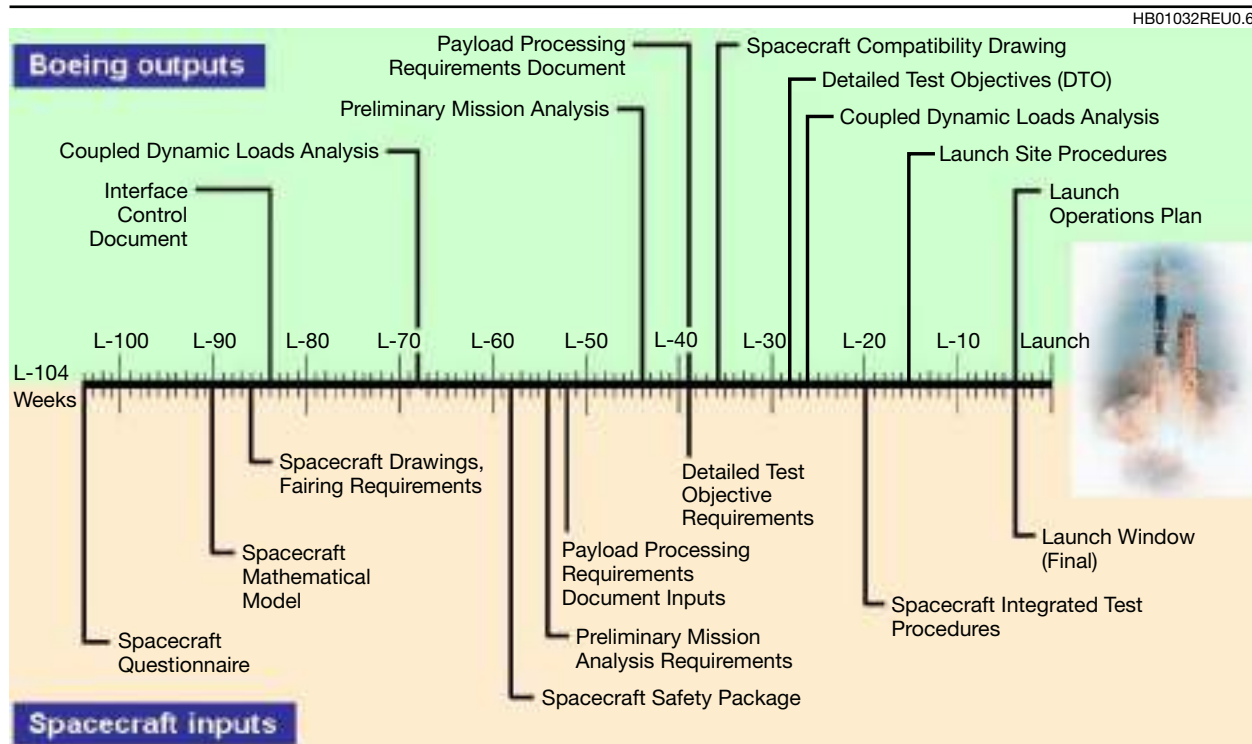


Figure 8-1. Typical Mission Integration Process

Mission integration for commercial missions is the responsibility of the Delta Program Office, which is located at the Boeing facility in Huntington Beach, California. The objective of mission integration is to coordinate all interface activities required to support the launch. This includes reaching a customer-Delta Program interface agreement and accomplishing: interface planning, requirements coordination, scheduling, and mission analyses.

The Delta Program Office assigns a mission integration manager to work with the customer and coordinate all mission-related interface activities. The mission integration manager develops

a tailored integration planning schedule for both the launch vehicle and the payload by defining the documentation and analyses required for the mission. The mission integration manager also synthesizes the payload requirements, engineering design, and launch environments into a controlled interface control document (ICD) that establishes and documents all agreed-to interface requirements.

The integration manager ensures that all lines of communication function effectively. To this end, all pertinent communications, including technical/administrative documentation, technical interchange meetings (TIM), and formal integration meetings, are coordinated through the mission integration manager and executed in a timely manner. These data exchange lines exist not only between the customer and the Delta Program, but also include all other agencies involved in the Delta II launch. Figure 8-2 illustrates the relationships among agencies involved in a typical Delta II mission.

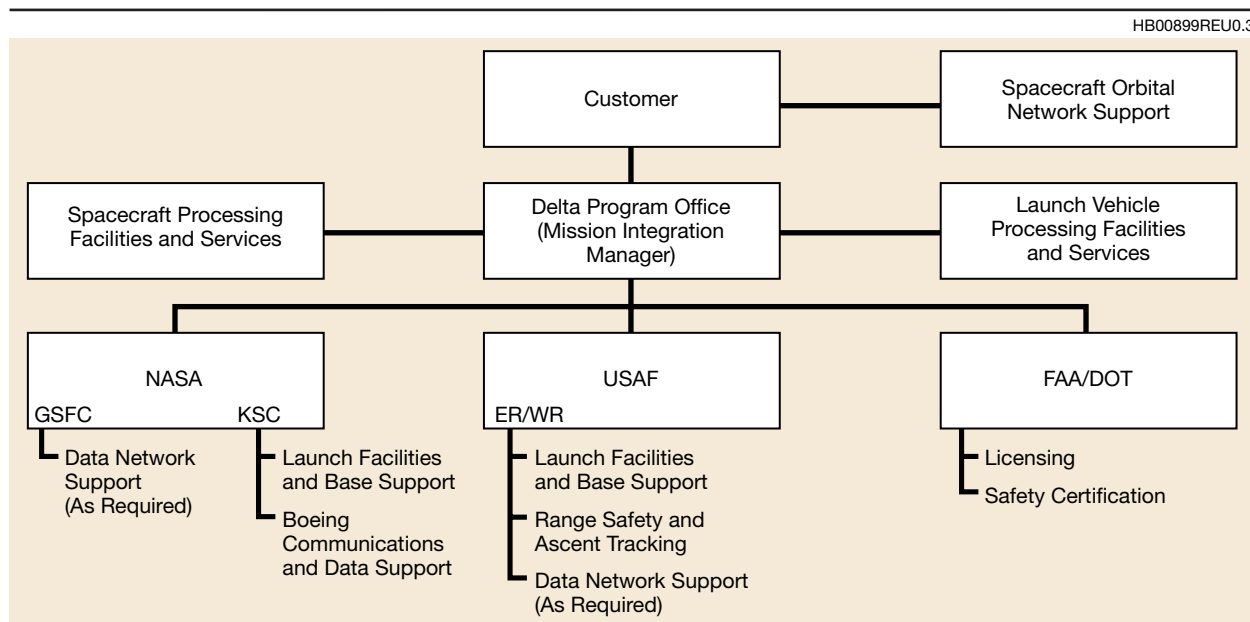


Figure 8-2. Typical Delta II Agency Interfaces

The mission integration process is identical for single, dual, and/or secondary payload missions. For a co-manifested mission using the dual payload attach fitting (DPAF), the Delta Program Office will assign a dedicated mission integration manager (MIM) to manage the integration effort associated with both payloads. This assures that the MIM maintains an integrated understanding of the overall mission objectives and requirements. Similarly, a MIM is assigned to manage all integration activities for missions flying both primary and secondary payloads.

8.2 DOCUMENTATION

Effective integration of the payload with the launch vehicle requires the diligent and timely preparation and submittal of required documentation. When submitted, these documents

represent the primary communication of requirements, safety data, system descriptions, etc., to each of the launch support agencies. The Delta Program Office acts as the administrative interface to assure proper documentation has been provided to the appropriate agencies. All data, formal and informal, are routed through the Delta Program Office. Relationships of the various categories of documentation are shown in Figure 8-3.

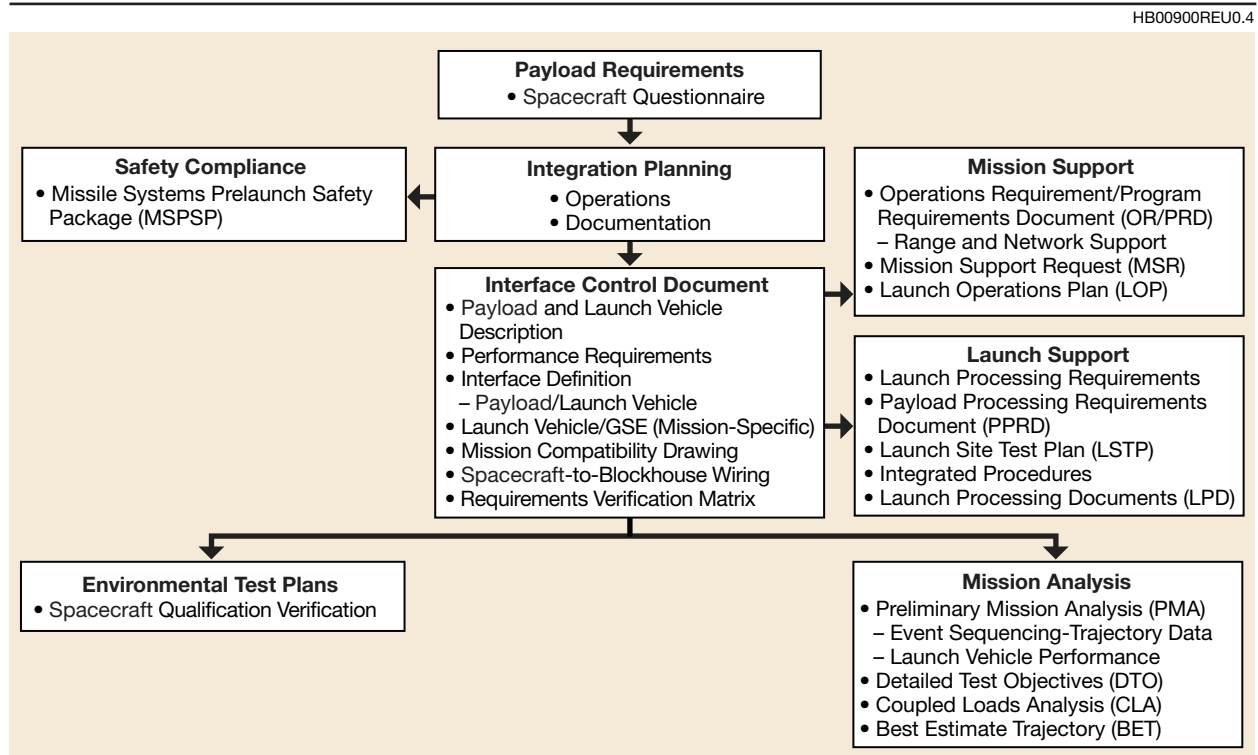


Figure 8-3. Typical Document Interfaces

The required documents for a typical mission are listed in Tables 8-1 and 8-2. Table 8-3 describes the contents of the program documents. Mission-specific schedules are established by agreement with each customer. The Spacecraft Questionnaire shown in Table 8-4 is normally completed by the customer 2 years prior to launch to provide an initial definition of payload characteristics and requirements. Table 8-5 is an outline of a typical payload launch-site test plan that describes the payload launch site activities and operations expected in support of the mission. Orbit data at burnout of the final stage are needed to reconstruct the performance of the launch vehicle following the mission. A complete set of orbital elements and associated estimates of 3-sigma (3σ) accuracy required to reconstruct this performance is presented in Table 8-6.

A typical integration planning schedule is shown in Figure 8-4. Each data item in Figure 8-4 has an associated L-date (weeks before launch). The responsible party for each data item is identified. Close coordination with the Delta mission integration manager is required to provide proper planning of the integration documentation.

Table 8-1. Customer Data Requirements

Description	Table 8-3 Reference	Nominal Due Weeks – or + Launch
Spacecraft Questionnaire	2	L104
Federal Aviation Administration (FAA) License Information	2	L104
Spacecraft Mathematical Model (Tested and verified)	3	L90/L-48
Spacecraft Drawings (Initial/Final)	18	L86/L44
Fairing Requirements	8	L86
Spacecraft Environmental Test Documents	5	L84
Interface Control Document Comments	4	30 days after receipt
Electrical Wiring Requirements	7	L80
Radiation Use Request/Authorization	10	L58
Combined Spacecraft/Third-Stage Nutation Time Constant and Mass Properties Statement (Initial/Final)for Three-Stage Missions	22	L54/L20
Spacecraft-Missile System Prelaunch Safety Package (MSPSP)	9	L58
Preliminary Mission Analysis Requirements (PMA)	11	L54/L39
Radio Frequency Application	30	L-52
Mission Operational and Support Requirements for Spacecraft	12, 13	L52
Payload Processing Requirements Document Inputs	14	L52
Spacecraft-to-Blockhouse Wiring Diagram Review	29	L40
Detailed Test Objectives (DTO) Requirements	17	L39
Launch Window (Initial/Final)	16	L39, L4
Vehicle Launch Insignia	15	L-39
Spacecraft Launch Site Test Plan	19	L34
Spacecraft Compatibility Drawing Comments	18	L29
Spacecraft Integrated Operations Inputs	21	L20
Spacecraft Launch Site Test Procedures	20	L18
Spacecraft Environments and Loads Test Report	5	L18
Best Estimate Trajectory (BET) Inputs	31	L-4
Postlaunch Orbit Confirmation Data	28	L+1 day

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Table 8-2. Delta Program Documents

Description	Table 8-3 Reference	Nominal Due Weeks – or + Launch
Interface Control Document (Initial)	4	L84
Coupled Dynamic Loads Analysis	6	L68, L-26
Spacecraft-to-Blockhouse Wiring Diagram (Preliminary/Final)	29	L50, L24
Preliminary Mission Analysis (PMA)	11	L44
Payload Processing Requirements Document	14	L39
Spacecraft Compatibility Drawing	18	L36, L17
Detailed Test Objectives (DTO)	17	L28
Spacecraft-Fairing Clearance Drawing	18	L27
Launch Site Procedures		As required*
Nutation Control System Analysis (if applicable)	23	L15
Spacecraft Separation Analysis	25	L12
Launch Operations Plan	26	L-12/L4
Integrated Countdown Schedule		L6
Vehicle Information Memorandum (VIM)	27	L3
Best Estimate Trajectory	31	L-1
*Approximately 2 weeks prior to use		

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Table 8-3. Required Documents

Item	Responsibility
<p>1. Feasibility Study (Optional) A feasibility study may be necessary to define the launch vehicle's capabilities for a specific mission or to establish the overall feasibility of using the vehicle for performing the required mission. Typical items that may necessitate a feasibility study are (1) a new flight plan with unusual launch azimuth or orbital requirements; (2) a precise accuracy requirement or a performance requirement greater than that available with the standard vehicle; and (3) spacecraft that impose uncertainties with regard to vehicle stability. Specific tasks, schedules, and responsibilities are defined before study initiation, and a final report is prepared at the conclusion of the study.</p>	Delta Program
<p>2. Spacecraft Questionnaire The Spacecraft Questionnaire (Table 8-4) is the first step in the process and is designed to provide the initial definition of spacecraft requirements, interface details, launch site facilities, and preliminary safety data to Delta's various agencies. It contains a set of questions whose answers define the requirements and interfaces as they are known at the time of preparation. The questionnaire is required not later than 2 years prior to launch. A definitive response to some questions may not be possible because many items are defined at a later date. Of particular interest are answers that specify requirements in conflict with constraints specified herein. Normally this document would not be kept current; it will be used to create the initial issue of the mission specification (Item 4) and in support of our Federal Aviation Administration (FAA)/Department of Transportation (DOT) launch permit. The specified items are typical of the data required for Delta II missions. The spacecraft customer is encouraged to include other pertinent information regarding mission requirements or constraints.</p>	Customer
<p>3 Spacecraft Mathematical Model for Dynamic Analysis A spacecraft mathematical model is required for use in a coupled loads analysis. Acceptable forms include (1) a discrete math model with associated mass and stiffness matrices or (2) a constrained normal mode model with modal mass and stiffness and the appropriate transformation matrices to recover internal responses. Required model information such as specific format, degree-of-freedom requirements, and other necessary information will be supplied. If the spacecraft has a propellant management device (e.g., diaphragm, bladder, baffles, etc.) the model should include spring-mass or pendulum parameters with maximum slosh mass, nominal slosh frequency and uncertainty range. If there are off-centerline propellant tanks, the model should include slosh modes and elastic modes relative to a fixed base at the spacecraft interface. The fundamental slosh modes should be in both directions for each off-centerline tank.</p>	Customer
<p>4. Interface Control Document (ICD) The Delta Program ICD functions as the Delta launch vehicle interface control document and describes all mission-specific requirements. It contains the spacecraft description, spacecraft-to-blockhouse wiring diagram interfaces, compatibility drawing, targeting criteria, special spacecraft requirements affecting the standard launch vehicle, description of mission-specific vehicle interfaces, a description of special aerospace ground equipment (AGE) and facilities the Delta Program Office is required to furnish, etc. The document is provided to spacecraft customers for review and concurrence and is revised as required. The initial issue is based on data provided in the spacecraft questionnaire and is provided approximately 84 weeks before launch. Subsequent issues are published as requirements and data become available. The mission-specific requirements documented in the ICD along with the standard interfaces presented in this manual define the spacecraft-to-launch vehicle interface.</p>	Delta Program (input required from customer)
<p>5. Spacecraft Environmental Test Documents The environmental test plan documents the spacecraft customer's approach for qualification and acceptance (pre flight screening) tests. It is intended to provide general test philosophy and an overview of the system-level environmental testing to be performed to demonstrate adequacy of the spacecraft for flight (e.g., static loads, vibration, acoustics, shock). The test plan should include test objectives, test specimen configuration, general test methods, and a schedule. It should not include detailed test procedures. Following the system-level structural loads and dynamic environment testing, test reports documenting the results shall be provided to the Delta Program Office. These reports should summarize the testing performed to verify the adequacy of spacecraft structure for the flight loads. For structural systems not verified by test, a structural loads analysis report documenting the analyses performed and resulting margins of safety should be provided to Boeing.</p>	Customer
<p>6. Coupled Dynamic Loads Analysis A coupled dynamic loads analysis is performed in order to define flight loads to major vehicle and spacecraft structure. The liftoff event, which generally causes the most severe lateral loads in the spacecraft, and the period of transonic flight and maximum dynamic pressure, causing the greatest relative deflections between spacecraft and fairing, are generally included in this analysis. Output for each flight event includes tables of maximum acceleration at selected nodes of the spacecraft model as well as a summary of maximum interface loads. Worst-case space craft-fairing dynamic relative deflections are included. Close coordination between the customer and the Delta Program Office is essential in order to decide on the output format and the actual work schedule for the analysis.</p>	Delta Program (input required from customer, Item 3)

Table 8-3. Required Documents (Continued)

Item	Responsibility
7. Electrical Wiring Requirements The wiring requirements for the spacecraft to the blockhouse and the payload processing facilities are needed as early as possible. Section 5 lists the Delta capabilities and outlines the necessary details to be supplied. The Delta Program Office will provide a spacecraft-to-blockhouse wiring diagram based on the spacecraft requirements. It will define the hard ware interface from the spacecraft to the blockhouse for control and monitoring of spacecraft functions after space craft installation in the launch vehicle. Close attention to the documentation schedule is required so that production checkout of the launch vehicle includes all of the mission-specific wiring. Any requirements for the payload processing facilities are to be furnished with the blockhouse information.	Customer
8. Fairing Requirements Early spacecraft fairing requirements should be addressed in the questionnaire and updated in the ICD. Final spacecraft requirements are needed to support the mission-specific fairing modifications during production. Any in-flight requirements, ground requirements, critical spacecraft surfaces, surface sensitivities, mechanical attachments, RF transparent windows, and internal temperatures on the ground and in flight must be provided.	Customer
9. Missile System Prelaunch Safety Package (MSPSP) (Refer to AFSPCMAN 91-710 for specific spacecraft safety requirements.) To obtain approval to use the launch site facilities and resources and for launch, a MSPSP must be prepared and submitted to the Delta Program Office. The MSPSP includes a description of each hazardous system (with drawings, schematics, and assembly and handling procedures, as well as any other information that will aid in appraising the respective systems) and evidence of compliance with the safety requirements of each hazardous system. The major categories of hazardous systems are ordnance devices, radioactive material, propellants, pressurized systems, toxic materials and cryogenics, and RF radiation. The specific data required and suggested formats are discussed in Section 3 of AFSPCMAN 91-710. Boeing will provide this information to the appropriate government safety offices for their approval.	Customer
10. Radiation Use Request/Authorization The spacecraft agency is required to specify the RF transmitted by the spacecraft during ground processing and launch intervals. A RF data sheet specifying individual frequencies will be provided. Names and qualifications are required covering spacecraft user personnel who will operate spacecraft RF systems. Transmission frequency bandwidths, frequencies, radiated durations, wattage, etc., will be provided. The Delta Program Office will provide these data to the appropriate range/government agencies for approval.	Customer
11. Preliminary Mission Analysis (PMA) This analysis is normally the first step in the mission-planning process. It uses the best available mission requirements (spacecraft weight, orbit requirements, tracking requirements, etc.) and is primarily intended to uncover and resolve any unusual problems inherent in accomplishing the mission objectives. Specifically, information pertaining to vehicle environment, performance capability, sequencing, and orbit dispersion is presented. Parametric performance and accuracy data are usually provided to assist the customer in selection of final mission-orbit requirements. The orbit dispersion data are presented in the form of variations of the critical orbit parameters as functions of probability level. A covariance matrix and a trajectory printout are also included. The mission requirements and parameter ranges of interest for parametric studies are due as early as possible but in no case later than 54 weeks before launch. Comments to the PMA are needed no later than launch minus 39 weeks for start of the detailed test objectives (DTO) (Item 17).	Delta Program (input required from customer)
12. Mission Operational and Support Requirements To obtain unique range and network support, the spacecraft customer must define any range or network requirements appropriate to its mission and then submit them to the Delta Program Office. Spacecraft customer operational configuration, communication, tracking, and data flow requirements are required to support document preparation and arrange required range support.	Customer
13. Program Requirements Document (PRD) To obtain range and network support, a spacecraft PRD must be prepared. This document consists of a set of pre-printed standard forms (with associated instructions) that must be completed. The spacecraft agency will complete all forms appropriate to its mission and then submit them to the Delta Program Office. The Delta Program Office will compile, review, provide comments, and, upon comment resolution, forward the spacecraft PRD to the appropriate support agency for formal acceptance.	Delta Program (input required from customer)
14. Payload Processing Requirements Document (PPRD) The PPRD is prepared if commercial facilities are to be used for spacecraft processing. The spacecraft customer is required to provide data on all spacecraft activities to be performed at the commercial facility. This includes detailed information of all facilities, services, and support requested by the Delta Program Office to be provided by the commercial facility. Spacecraft hazardous systems descriptions shall include drawings, schematics, summary test data, and any other available data that will aid in appraising the respective hazardous system. The commercial facility will accept spacecraft ground operations plans and/or MSPSP data as input to the PPRD.	Customer
15. Launch Vehicle Insignia The spacecraft customer is entitled to have a mission-specific insignia placed on the launch vehicle. The customer will submit the proposed design to the Delta Program Office not later than 9 months before launch for review and approval. Following approval, the Delta Program Office will have the flight insignia prepared and placed on the launch vehicle. The maximum size of the insignia is 2.4 m by 2.4 m (8 ft by 8 ft). The insignia is placed on the uprange side of the launch vehicle.	Customer

Table 8-3. Required Documents (Continued)

[illegible]

Table 8-3. Required Documents (Continued)

Item	Responsibility
24. RF Compatibility Analysis A radio frequency interference (RFI) analysis is performed to verify that spacecraft RF sources are compatible with the launch vehicle telemetry and tracking-beacon frequencies. Spacecraft frequencies defined in the mission specification are analyzed using a frequency-compatibility software program. The program provides a listing of all inter modulation products, which are then checked for image frequencies and inter-modulation product interference.	Delta Program
25. Spacecraft/Launch Vehicle Separation Memorandum An analysis is performed to verify that there is adequate clearance and separation distance between the spacecraft and expended payload attach fitting (PAF)/third stage. The principal parameters, including data from Item 22, that define the separation are the motor's residual thrust, half-cone angle, and spin rate. For two-stage missions this analysis verifies adequate clearance exists between the spacecraft and second stage during separation and second-stage post-separation maneuvers.	Delta Program (input required from customer)
26. Launch Operations Plan (LOP) This plan is developed to define top-level requirements that flow down into detailed range requirements. The plan contains the launch operations configuration, which identifies data and communication connectivity with all required support facilities. The plan also identifies organizational roles and responsibilities, the mission control team and its roles and responsibilities, mission rules supporting conduct of the launch operation, and go/no-go criteria.	Delta Program
27. Vehicle Information Memorandum (VIM) The Delta Program Office is required to provide a vehicle information memorandum to the U.S. Space Command 15 calendar days prior to launch. The spacecraft customer will provide to the Delta Program Office the appropriate spacecraft on-orbit data required for this VIM. Data required are spacecraft on-orbit descriptions, description of pieces and debris separated from the spacecraft, the orbital parameters for each piece of debris, spacecraft spin rates, and orbital parameter information for each different orbit through final orbit. The Delta Program Office will incorporate these data into the overall VIM and transmit to the appropriate U.S. government agency.	Delta Program
28. Postlaunch Orbit Confirmation Data To reconstruct Delta performance, orbit data at burnout (stage II or III) are required from the spacecraft customer. The spacecraft customer should provide orbit conditions at the burnout epoch based on spacecraft tracking data prior to any orbit-correction maneuvers. A complete set of orbital elements and associated estimates of 3-sigma accuracy are required (see Table 8-6).	Customer
29. Spacecraft-to-Blockhouse Wiring Diagram The Delta Program Office will provide, for inclusion in the mission specification, a spacecraft-to-blockhouse wiring diagram based on the spacecraft requirements. It will define the hardware interface from the spacecraft to the blockhouse for control and monitoring of spacecraft functions after spacecraft installation in the launch vehicle.	Delta Program
30. Radio Frequency Application If the customer plans, to radiate at the launch site, an FCC license should be obtained by the spacecraft customer. This will assure the customer that the spacecraft frequency will not be interfered with during use. The Delta Program office will assist the customer in this process.	Customer
31. Best Estimate Trajectory (BET) This analysis uses assigned stage one, two, and three (if present) propulsion predictions as well as actual launch vehicle and spacecraft weights in a guided simulation to provide a Best Estimate Trajectory for the mission. The guided simulation is based on targeting defined in the DTO trajectory (see Item 17 above), which can be adjusted slightly based on final customer inputs. The final spacecraft weight is also required as an input. The spacecraft is usually weighed by the Delta Program; however, if desired, a customer-furnished certified weight approved by the Delta Program Office may be submitted.	Delta Program (input required from customer)

002212.8

Table 8-4. Delta II Spacecraft Questionnaire

Note: When providing numerical parameters, please specify either English or Metric units.

1 Payload Characteristics

1.1 PAYLOAD DESCRIPTION

Provide a short description of the Payload, its major components, and how it is constructed.
Include two views of the payload (one in launch configuration and one on-orbit/in operation deployed configuration).

2 Applicable and Reference Document

2.1 APPLICABLE DOCUMENTS

These documents will form a part of the Interface Control Document (ICD) to the extent specified herein. In the event of conflicts the ICD will contain the superseding requirements.

2.2 REFERENCE DOCUMENTS

Document that contains additional information; e.g., Delta II Payload Planners Guide, document MDC 00H0016, dated October 2000.

3 Interface Requirements

3.1 MECHANICAL/STRUCTURAL INTERFACES

3.1.1 Coordinate Systems

3.1.1.1 Payload Coordinate System

Provide the spacecraft coordinate system for the moments and products of inertia and CG location.

3.1.2 Payload Fairing Interfaces

3.1.2.1 Payload Fairing Envelope (Refer to Chapter 3 of the Payload Planners Guide)

3.1.2.1.1 Payload Components Within 2.0 Inches of the Fairing Envelope

Table 3-1. Payload Components Within 2.0 Inches or Beyond the Fairing Envelope

Item	LV Vertical Station (unit)	Radial Distance from LV Centerline ¹	Payload Clocking (degree)	LV Clocking (degree) ²	Clearance from Stay-out Zone

Notes:

- Location of payload components should include maximum tolerances
- Clocking is measured from LV Quad IV (0/360°) toward LV Quad I (90°)

3.1.2.1.2 Payload Components Beyond the Separation Plane Envelope

Table 3-2. Payload Components beyond the Separation Plane Envelope

Item	LV Vertical Station (unit)	Radial Distance from LV Centerline ¹	Payload Clocking (degree)	LV Clocking (degree) ²	Clearance from Stay-out Zone

Notes:

- Location of payload components should include maximum tolerances
- Clocking is measured from LV Quad IV (0/360°) toward LV Quad I (90°)

3.1.2.2 Access Doors and RF Windows in Fairing

3.1.2.2.1 Access Doors

List known access door locations in Table 3-3.

3.1.2.2.2 RF Windows

List known RF window locations in Table 3-3.

Table 3-3. Access Doors and RF Windows

Size (in.)	LV Station (in.) ¹	LV Clocking (degree) ²	Purpose

Notes:

- Doors are centered at the locations specified
- Clocking is measured from LV Quad IV (0/360°) toward LV Quad I (90°)

- 3.1.3 Payload/Launch Vehicle Payload Attach Interface
- 3.1.3.1 Payload to Launch Vehicle Adapter Assembly
Provide payload adapter assembly interface design detail.
- 3.1.3.2 Electrical Bonding
Specify if the payload complies with the electrical bonding requirements of Chapter 4 of the Payload Planner's Guide.
- 3.1.3.3 Separation System
N/A if provided by Launch Vehicle.
- 3.1.4 Purge Interfaces
If a spacecraft purge system is required while on the pad and using the fairing – mounted purge tubing, fill in table below.

Table 3-4. Purge Interfaces

LV Station	Radius (inch)	Azimuth (degree)	Offset From Quad I Centerline

- 3.1.5 Special Vehicle Insignia
Include drawing of the payload insignia, if available.
- 3.1.6 Payload Mass Properties
- 3.1.6.1 Weight, Moments and Products of Inertia, and CG Location
List payload mass properties in Table 3-5.

Table 3-5. Payload Mass Properties

Description	Axis	Value	+/- 3 σ Uncertainty
Weight (lb.)/(kg)	N/A		
Center of Gravity (unit)	X		
	Y		
	Z		
Moments of Inertia (unit)	I _{xx}		
	I _{yy}		
	I _{zz}		
Products of (unit)	I _{xy}		
	I _{yz}		
	I _{xz}		

- 3.1.6.2 CG Offset
- 3.1.6.3 Principle Axis Misalignment
- 3.1.6.4 Nutation Time Constants (3 stage missions only)
- 3.2 ELECTRICAL INTERFACES
- 3.2.1 Payload/Payload Attach Fitting Electrical Connectors (if required)
- 3.2.1.1 Connector Types, Location, Orientation, and Part Number

Table 3-6. Interface Connectors (Spacecraft/Payload Attach Fitting)

Item	P1	P2
Vehicle connector		
Spacecraft mating connectors (J1 and J2)		
Distance forward of spacecraft mating plane		
Launch vehicle station		
Azimuth ¹		
Radial distance of connector centerline from vehicle centerline ¹		
Polarizing key location		
Maximum connector force (+compression, -tension)		

Note:

1. Positional tolerance defined in Chapter 5 of the Payload Planners Guide

- 3.2.1.2 Connector Pin Assignments

Table 3-7. Pin Assignments (Payload/Payload Attach Fitting)

Pin No.	Twisted and Shielded with	Function	Volt	Amp	Max resistance to EEB (ohm)	Polarity Requirement
1						
2						
3						
4						
5						

- 3.2.1.3 Payload Separation Indication
Provide any requirements related to spacecraft separation indication (breakwires).
- 3.2.1.4 Special Payload Functions
Provide appropriate subsections as needed for any requirements related to special spacecraft functions required of the launch vehicle (e.g., discrete commands to be provided by the launch vehicle to the spacecraft via SC/fairing connectors.)
- 3.2.1.5 Payload Data Requirements
Provide any requirements related to special spacecraft data to be transmitted by the launch vehicle SC/fairing connectors
- 3.2.2 Payload/Fairing Electrical Connectors (if required)
 - 3.2.2.1 Connector Types, Location, Orientation, and Part Number

Table 3-8. Interface Connectors (Payload/Fairing)

Item	P1	P2
Vehicle connector		
Spacecraft mating connectors (J1 and J2)		
Distance forward of spacecraft mating plane		
Launch vehicle station		
Azimuth ¹		
Radial distance of connector centerline from vehicle centerline ¹		
Polarizing key location		
Maximum connector force (+compression, -tension)		

Note:

- 1. Positional tolerance defined in Chapter 5 of the Payload Planners Guide

3.2.2.2 Connector Pin Assignments

Table 3-9. Pin Assignments (Payload/Fairing)

Pin No.	Twisted and Shielded with	Function	Volt	Amp	Max resistance to EEB (ohm)	Polarity Requirement
1						
2						
3						
4						
5						

- 3.2.2.3 Payload Separation Indication
- 3.2.2.4 Special Payload Functions
- 3.2.2.5 Payload Data Requirements
- 3.2.3 Separation Switches
 - 3.2.3.1 Separation Switches (Payload)
- 3.2.4 GSE Interfaces
 - 3.2.4.1 Payload GSE Electrical Interfaces
 - 3.2.4.2 Range Safety Console Interface
- 3.3 MISSION PARAMETERS
 - 3.3.1 Orbit Characteristics
State the type of orbit that is required. Complete Table 3-10 below.

Table 3-10. Orbit Characteristics

Parameter	Value	Tolerance
Apogee		
Perigee		
Inclination		
Argument of perigee at insertion		
RAAN		
Probability of command shutdown		

3.3.2 Launch Windows

Table 3-11. Launch Windows

Local Time		GMT	
Window Open mm/dd/yy hh:mm:ss	Window Close mm/dd/yy hh:mm:ss	Window Open mm/dd/yy hh:mm:ss	Window Close mm/dd/yy hh:mm:ss

- 3.3.3 Payload Constraints on Mission Parameters
 - 3.3.3.1 Sun Angle Constraints
 - 3.3.3.2 Telemetry Constraints
 - 3.3.3.3 Thermal Attitude Constraints
 - 3.3.3.4 Contamination and Collision Avoidance Maneuver Constraints

3.3.4 Systems Activated Prior to Payload Separation

List all spacecraft events that will take place during the launch sequence, from liftoff to spacecraft separation, by completing the following chart:

Table 3-12. Events During Launch Phase

Event	Time from Liftoff	Constraint/Comment

3.3.5 Payload Separation Requirements

Table 3-13. Separation Requirements

Parameter	Value	Tolerance
Attitude pointing (two-stage mission)		
Tip - off angular rate (two-stage mission)		
Angular momentum vector pointing error (three-stage mission)		
Nutation cone angle (three-stage mission)		
Spin rate (three-stage mission)		

Note: The nutation coning angle is a half angle with respect to the angular momentum vector.

- 3.3.6 Flight Operations Requirements
 - 3.3.6.1 Payload Tracking Stations
 - 3.3.6.2 Payload Acquisition Assistance Requirements
 - 3.3.6.3 Special Payload Mission Operations Requirements
 - 3.3.6.4 Payload Uplink Requirements
 - 3.3.6.5 Payload Downlink Requirements

3.4 ENVIRONMENTAL REQUIREMENTS

3.4.1 Payload Stiffness (Frequency)

Note: To prevent dynamic coupling between the launch vehicle and the Payload in the low-frequency range for the three-stage Delta II 79XX and 79XXH configurations, the payload structural-stiffness should produce fundamental frequencies above 35 Hz in the thrust axis and 15 Hz in the lateral axes. For three-stage Delta II 73XX or 74XX configurations, the payload structure stiffness should produce fundamental frequencies above 35 Hz in the thrust axis and 20 Hz in the lateral axes of the payload. For all two-stage Delta II configurations, the payload structural stiffness should produce fundamental frequencies above 35 Hz in the thrust axis and 12 Hz in the lateral axes. The payload should meet these criteria, while hard-mounted at the payload separation plane (without compliance from the PAF and separation clampband). In addition, secondary structure mode frequencies should be above 35 Hz to prevent undesirable coupling with launch vehicle modes and/or large fairing-to-payload relative dynamic deflections.

3.4.2 Interface Loads

If payload construction does not provide uniform load distribution at the launch vehicle interface, provide details.

3.4.3 RF Environment

3.4.3.1 RF Inhibits

Note: To preclude RF fields that could be detrimental to launch vehicle ordnance or electronics, the Payload shall be designed with two independent inhibits to prohibit inadvertent RF transmissions.

3.4.3.2 RF Radiation levels (Personnel Safety)

Note: Distance at which RF radiation flux density equals 1m W/cm² for TBD antenna shall be TBD cm.

3.5 PAYLOAD HANDLING AND PROCESSING REQUIREMENTS

- 3.5.1 Facility and Ground Handling Environment (see Payload Planners Guide Chapter 4, Tables 4-1 and 4-2 for limitations)

Table 3-14. Launch Pad Environmental Requirements

Location	Temperature	Relative Humidity	Cleanliness
Mobile service tower	Clean room		
	Environmental shroud		
	Fairing		

3.5.2 Air Conditioning and Purges

3.5.2.1 Air Conditioning

State the required fairing airflow, maximum fairing airflow and any inspection or oversight requirements (i.e., AC air impingement restrictions or requirements).

3.5.2.2 GN² Purge

Provide requirements related to spacecraft GN² purges during processing and prior to launch. Typically this includes quality/cleanliness requirements, temperatures, and flow rate.

- 3.5.3 Contamination Control
 - 3.5.3.1 Payload Environmental Shroud
Specify any requirements for use of a spacecraft environmental shroud.
 - 3.5.3.2 Cleanliness Category
Specify the cleanliness class required during spacecraft assembly. Also, include the spacecraft and launch vehicle items that need to be cleaned prior to transport and/or prior to installation.
- 3.5.4 Payload Weighing and Balancing
 - 3.5.4.1 Payload Weighing
Note that Boeing must either witness or perform the weighing of the spacecraft. The weight must be within +/- 0.1% for two stage missions and +/-0.05% for three stage missions.
 - 3.5.4.2 Payload Balancing
Three stage missions only. State if the spacecraft manufacturer or Boeing is required to perform the spin balance test prior to integration with the Delta third stage.
- 3.5.5 Special Handling Requirements
State any additional Boeing handling requirements in the sections below. Specifically, include the size of any support equipments, their probable location, and any power requirements they may have.
 - 3.5.5.1 In Payload Processing Facility
 - 3.5.5.2 During Transport
 - 3.5.5.3 On Stand
 - 3.5.5.4 In Launch Support Building
- 3.5.6 Special Boeing – Supplied Equipment or Facilities
For example, stands and platforms for payload access.
- 3.5.7 Other Payload Ground Requirements
 - 3.5.7.1 Prior to Fairing Installation
List any payload ground access requirements that the payload will have prior to the fairing being installed in Table 3-15 below.

Table 3-15. Payload Access Requirements before Fairing Installation

Access item	Station Number	Angular Reference (degree in LV system)	Radial Distance from Centerline (inch)	Number of Personnel Required	Activity on Launch Minus Day

- 3.5.7.2 After Fairing Installation

Table 3-16. Payload Access Requirements after Fairing Installation

Access item	Station Number	Angular Reference (degree in LV system)	Radial Distance from Centerline (inch)	Number of Personnel Required	Activity on Launch Minus Day

4 Quality Assurance Provisions

4.1 VERIFICATION METHODS

The verification methods used in the verification matrix are defined in this section.

4.2 VERIFICATION MATRIX

A standard format verification matrix is maintained by Boeing and provided here. All interface requirements specified in the ICD are included in the verification matrix

5 Mission Information

This section contains information that may be included in the ICD but are not payload to launch vehicle interface requirements.

5.1 PAYLOAD CONFIGURATION

5.1.1 Payload Hazardous Systems

This section will be included in the Payload Safety Approval Package Inputs as required by the range. The information provided here is for reference only.

5.1.1.1 Propulsion System

Complete Table 5-1 below. If the spacecraft has more than one propulsion system, create an additional table for each system and attribute it to its spacecraft function.

Table 5-1. Propulsion System 1 Characteristics

Parameter	Value
Propellant type	
Propellant weight, nominal	
Propellant fill fraction	
Propellant density	
Propellant tank material	
Propellant tank location (SC coordinates)	
Station	
Azimuth	
Radius	
Diameter	
Shape	
Internal volume	
Capacity	
Internal description	
Operating pressure - flight	
Operating pressure - ground	
Design burst pressure - calculated	
Factor of safety (design burst/ground maximum expected operating pressure [MEOP])	
Proof pressure - test	
Actual burst pressure - test	
Pressure when Boeing personnel are exposed	
Number of vessels used	

5.1.1.2 Non-Propulsion Pressurized Systems

This section does not include batteries – refer to Section 5.1.1.3 below.

Complete Table 5-2 below. If the spacecraft has more than one non-propulsion pressurized system, create an additional table for each system and attribute it to its spacecraft function.

Table 5 2. Pressurized Tank 1

Parameter	Value
Purpose	
Vessel contents	
Tank material	
Capacity - launch	
Fill fraction	
Operating pressure - flight	
Operating pressure - ground	
Design burst pressure - calculated	
Factor of safety (design burst/ground MEOP)	
Proof pressure - test	
Actual burst pressure - test	
Pressure when Boeing personnel are exposed	

5.1.1.3 Payload Batteries

If the spacecraft has more than one battery, create an additional table for each one.

Table 5-3. Payload Battery 1

Parameter	Value
Battery type	
Battery capacity	
Electrolyte	
Cell pressure vessel material	
Number of cells	
Average voltage/cell	
Cell pressure (ground MEOP)	
Specification burst pressure	
Actual burst	
Proof tested	
Back pressure control (BPC) type	

5.1.1.3.1 Voltage of the Payload Battery and Polarity of the Ground

5.1.1.4 RF Systems

List the number of transmitters, receivers, and their components.

5.1.1.4.1 RF Characteristics

Fill out Table 5-4 below and add further charts as necessary.

5.1.1.4.2 RF Radiation Levels (Personnel Safety)

Describe the operations planned for the RF system while the payload is on the pad.

5.1.1.5 Deployable Systems

Identify and state the deployable systems on your payload (e.g., antennas, solar panels), their size, location and when they will be deployed.

5.1.1.6 Radioactive Devices

State and describe all the radioactive devices on the payload.

Table 5-4. Transmitters and Receivers

Parameter	Antennas			
	Rcvr 1	Xmttr 2	3	4
Nominal frequency (MHz)				
Transmitter tuned frequency (MHz)				
Receiver frequency (MHz)				
Data rates, downlink (kbps)				
Symbol rates, downlink (kbps)				
Type of transmitter				
Transmitter power, maximum (dBm)				
Losses, minimum (dB)				
Peak antenna gain (dB)				
Antenna gain 90 deg off boresight (dB)				
EIRP, maximum (dBm)				
Antenna location (base)				
Station (in.)				
Azimuth (deg)				
Radius (in.)				
1 mW/cm ² Distances (personnel safety)				
Planned operation:				
Prelaunch: In building _____				
Pre-launch: Pre-fairing installation				
Post-launch: Before payload separation				

5.1.1.7 Electroexplosive Devices (EEDs)

Table 5-5. Electro Explosive Devices

Qty	Type	Use	Firing Current (amps)		Bridgewire (ohm)	Where Installed	Where Connected	Where Armed
			No Fire	All Fire				

5.1.1.8 Non-EED Release Devices

Table 5-6. Non-electric Ordnance and Release Devices

Qty	Type	Use	Qty Explosives	Type	Explosives	Where Installed	Where Connected	Where Armed

5.1.1.9 Other Hazardous Systems

Note any hazardous materials or systems on the payload that have not already been identified.

5.1.2 Contamination – Sensitive Surfaces

Table 5-7. Contamination Sensitive Surfaces

Component	Sensitive To

- 5.1.3 Payload Venting
 - 5.1.3.1 Ventable Volume
Provide the total ventable volume of the spacecraft in cubic feet and cubic meters, stating the percentage accuracy of the data.
 - 5.1.3.2 Non-ventable Volume
Provide the total non-ventable volume of the spacecraft in cubic feet and cubic meters, stating the percentage accuracy of the data.
- 5.1.4 Payload Energy Dissipation Sources
List all components that will cause spacecraft energy dissipation (such as liquid propellants, passive nutation dampers, flexible antennas, heat pipes etc.) with a brief note so they can be identified on spacecraft drawings.
- 5.2 FLIGHT OPERATIONS REQUIREMENTS
 - 5.2.1 Location of Payload Operations Control Center

6 Additional Material

This section contains requests for information that will not be included in the ICD. Although later documents will collect much of this information and individual deadlines may vary, it is to the customer's advantage to respond in the Payload Questionnaire wherever this is possible.

6.1 MODELING

The Craig – Bampton format is the requested description of the Payload dynamic model. If possible, use the Nastran OP4 BCD format for the following items.

For SC with liquid tanks that are located off the centerline axis of the LV, the Payload dynamic model must include the slosh characteristics.

- 6.1.1 Mass Matrix
- 6.1.2 Stiffness Matrix
- 6.1.3 Response Recovery Matrix

6.2 TEST

6.2.1 Planning

- 6.2.1.1 Payload Development and Test Programs
- 6.2.1.2 Payload Development and Test Schedules
The most current information is required and will be updated in part from the *Payload Environmental Test* document.
- 6.2.1.3 Flow Chart and Test Schedule
The most current information is required and will be updated in part from the *Payload Environmental Test* document.
- 6.2.1.4 Test Schedule at Launch Site
The most current information is required and will be updated from the *Payload Launch Site Test Plan* and the *Payload Integrated Test Operations Procedure Inputs* documents.
- 6.2.1.5 Operations Flow Chart
The most current information is required and will be updated from the *Payload Launch Site Procedures* document.

6.2.2 Hardware/Personnel/Facilities Requirements

The most current information is required and will be updated in part from the *Payload Environmental Test* document.

- 6.2.2.1 Test PAF Requirements
Is a test PAF required? When?
- 6.2.2.2 Clamp Band Ordnance Requirements
Is clamp band ordnance required? When?
- 6.2.2.3 Special Test Requirements
- 6.2.2.4 Payload Spin Balancing

6.3 PROCESSING FACILITIES LOGISTICS REQUIREMENTS

- 6.3.1 Processing Facility Preference and Priority
- 6.3.2 Payload Processing Facility Dwell Time
- 6.3.3 Multishift Operation Plans
State whether a multishift operation is planned and give any available details.
- 6.3.4 Facility Crane Requirements
- 6.3.5 Facility Electrical Requirements
- 6.3.6 Hazardous Processing Facilities
- 6.3.7 NASA, USAF, or Commercially Provided Support Items
- 6.3.8 Facility Security

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Table 8-5. Typical Spacecraft Launch-Site Test Plan

1 General	
1.1	Plan Organization
1.2	Plan Scope
1.3	Applicable Documents
1.4	Spacecraft Hazardous Systems Summary
2 Prelaunch/Launch Test Operations Summary	
2.1	Schedule
2.2	Layout of Equipment (Each Facility) (Including Test Equipment)
2.3	Description of Event at Launch Site
2.3.1	Spacecraft Delivery Operations
2.3.1.1	Spacecraft Removal and Transport to Spacecraft Processing Facility
2.3.1.2	Handling and Transport of Miscellaneous Items (Ordnance, Motors, Batteries, Test Equipment, Handling and Transportation Equipment)
2.3.2	Payload Processing Facility Operations
2.3.2.1	Spacecraft Receiving Inspection
2.3.2.2	Battery Inspection
2.3.2.3	Reaction Control System (RCS) Leak Test
2.3.2.4	Battery Installation
2.3.2.5	Battery Charging
2.3.2.6	Spacecraft Validation
2.3.2.7	Solar Array Validation
2.3.2.8	Spacecraft/Data Network Compatibility Test Operations
2.3.2.9	Spacecraft Readiness Review
2.3.2.10	Preparation for Transport and Transport to Hazardous Processing Facility (HPF)
2.3.3	Solid Fuel Storage Area
2.3.3.1	Apogee Kick Motor (AKM) Receiving, Preparation, and X-Ray
2.3.3.2	Safe and Arm (S&A) Device Receiving, Inspection, and Electrical Test
2.3.3.3	Igniter Receiving and Test
2.3.3.4	AKM/S&A Assembly and Leak Test
2.3.4	HPF
2.3.4.1	Spacecraft Receiving Inspection
2.3.4.2	Preparation for AKM Installation
2.3.4.3	Mate AKM to Spacecraft
2.3.4.4	Spacecraft Weighing (Include Configuration Sketch and Approximate Weights of Handling Equipment)
2.3.4.5	Spacecraft/Third-Stage Mating
2.3.4.6	Preparation for Transport Installation Into Handling Can
2.3.4.7	Transport to Launch Complex
2.3.5	Launch Complex Operations
2.3.5.1	Spacecraft Hoisting and Removal of Handling Can
2.3.5.2	Spacecraft Mate to Launch Vehicle
2.3.5.3	Hydrazine Leak Test
2.3.5.4	Telemetry, Tracking, and Command (TT&C) Checkout
2.3.5.5	Preflight Preparations
2.3.5.6	Fairing Installation
2.3.5.7	Launch Countdown
2.4	Launch/Hold Criteria
2.5	Environmental Requirement for Facilities During Transport
3 Test Facility Activation	
3.1	Activation Schedule
3.2	Logistics Requirements
3.3	Equipment Handling
3.3.1	Receiving
3.3.2	Installation
3.3.3	Validation
3.3.4	Calibration
3.4	Maintenance
3.4.1	Spacecraft
3.4.2	Launch-Critical Mechanical Aerospace Ground Equipment (AGE) and Electrical AGE
4 Administration	
4.1	Test Operations—Organizational Relationships and Interfaces (Personnel Accommodations, Communications)
5 Security Provisions for Hardware	
6 Special Range-Support Requirements	
6.1	Real-Time Tracking Data Relay Requirements
6.2	Voice Communications
6.3	Mission Control Operations

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Table 8-6. Data Required for Orbit Parameter Statement

1.	Epoch: Stage burnout
2.	Position and velocity components (X, Y, Z, and \dot{X} , \dot{Y} , \dot{Z}) in equatorial inertial Cartesian coordinates.* Specify mean-of-date or true-of-date, etc.
3.	Keplerian elements* at the above epoch: Semimajor axis, a Eccentricity, e Inclination, i Argument of perigee, ω Mean anomaly, M Right ascension of ascending node, Ω
4.	Polar elements* at the above epoch: Inertial velocity, V Inertial flight path angle, γ_1 Inertial flight path angle, γ_2 Radius, R Geocentric latitude, ρ Longitude, μ
5.	Estimated accuracies of elements and a discussion of quality of tracking data and difficulties such as reorientation maneuvers within 6 hr of separation, etc.
6.	Constants used: Gravitational constant, μ Equatorial radius, R_E J_2 or Earth model assumed
7.	Estimate of spacecraft attitude and coning angle at separation (if available).
*Note: At least one set of orbit elements in Items 2, 3, or 4 is required	

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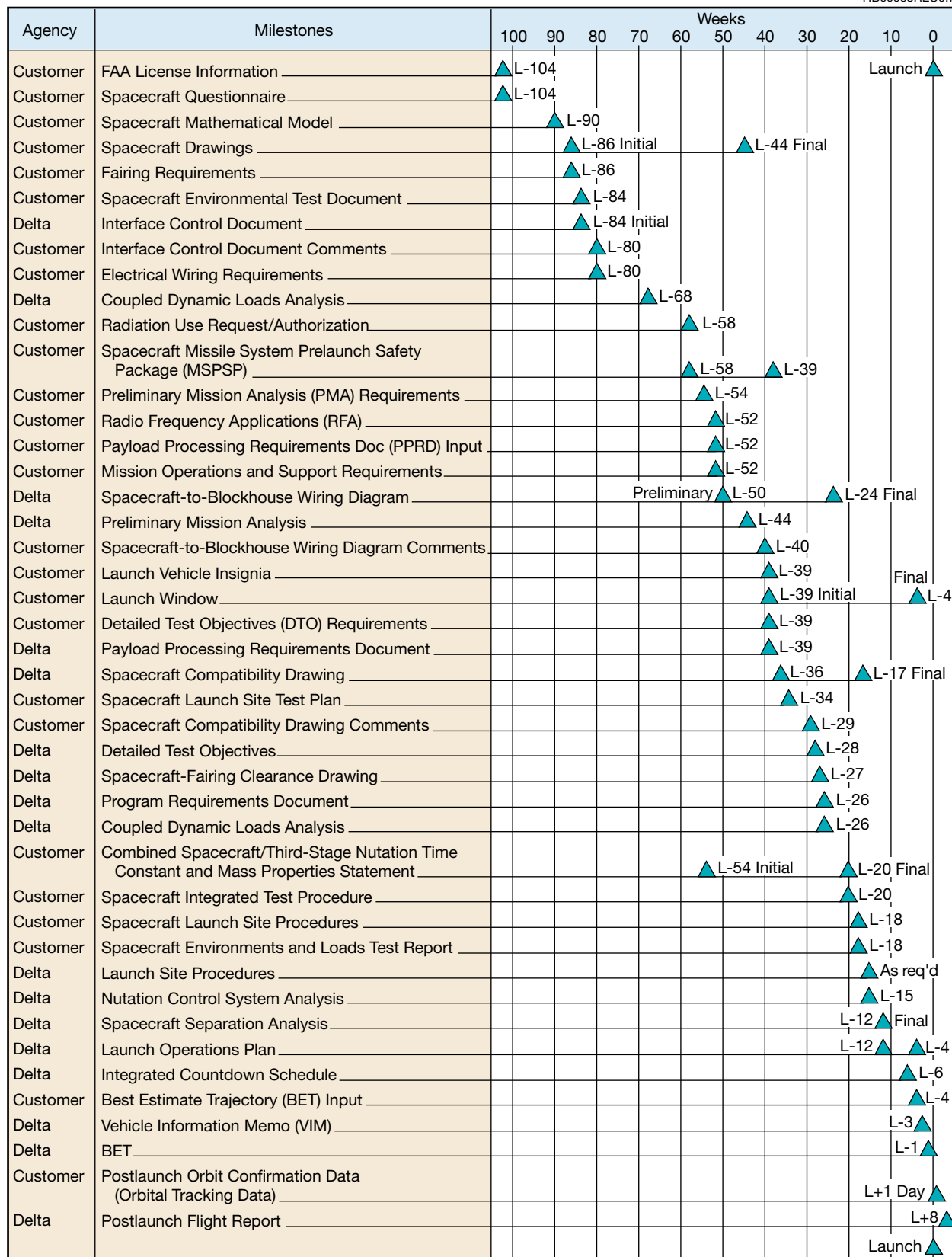


Figure 8-4. Typical Integration Planning Schedule

8.3 LAUNCH OPERATIONS PLANNING

The development of launch operations, range support, and other support requirements is an evolutionary process that requires timely inputs and continued support from the customer. The relationship and submittal schedules of key controlling documents are shown in Figure 8-5.

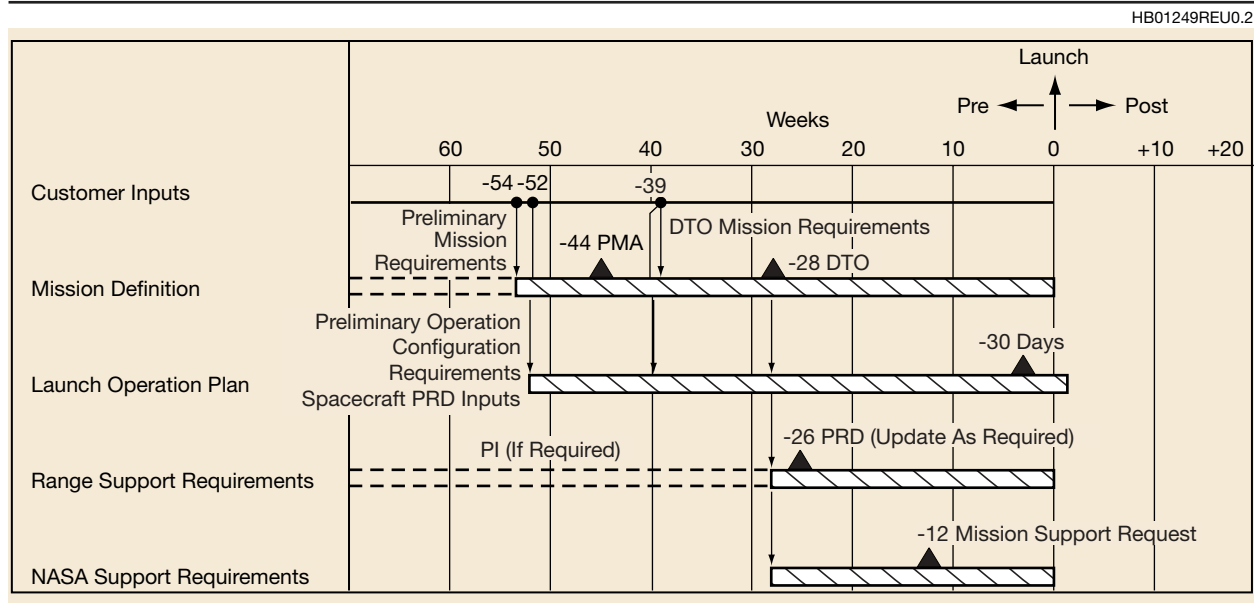


Figure 8-5. Launch Operational Configuration Development

8.4 SPACECRAFT PROCESSING REQUIREMENTS

The checklist shown in Table 8-7 is provided to assist the user in identifying the requirements at each processing facility. The requirements identified are submitted to the Delta Program Office for the program requirements document (PRD). Boeing coordinates with the appropriate launch site agency and implements the requirements through the program requirements document/payload processing requirements document (PRD/PPRD). The customer may add items to the list. Note that most requirements for assembly and checkout of commercial payloads will be met by the Astrotech or Spaceport Systems International (SSI) facility.

Table 8-7. Spacecraft Checklist

<p>1. General</p> <p>A. Transportation of spacecraft elements/ground support equipment (GSE) to processing facility (1) Mode of transportation _____ (2) Arriving at _____ (gate, skid strip) (date) _____</p> <p>B. Data-handling (1) Send data to (name and address) _____ (2) Time needed (real-time versus after-the-fact) _____</p> <p>C. Training and medical examinations for _____ crane operators</p> <p>D. Radiation data (1) Ionizing radiation materials _____ (2) Nonionizing radiation materials/systems _____</p> <p>2. Spacecraft Processing Facility (for nonhazardous work)</p> <p>A. Does payload require a cleanroom? (yes) _____ (no) _____ (1) Class of cleanroom required _____ (2) Special sampling techniques _____</p> <p>B. Area required (1) For spacecraft _____ (2) For ground station _____ (3) For office space _____ (4) For other GSE _____ (5) For storage _____</p> <p>C. Largest door size (1) For spacecraft/GSE _____ (high) _____ (wide) _____ (2) For ground station _____</p> <p>D. Material-handling equipment (1) Cranes a. Capacity _____ b. Minimum hook height _____ c. Travel _____ (2) Other _____</p> <p>E. Environmental controls for spacecraft/ground station (1) Temperature/humidity and tolerance limits _____ (2) Frequency of monitoring _____ (3) Downtime allowable in the event of a system failure _____ (4) Is a backup (portable) air-conditioning system required? (yes) _____ (no) _____ (5) Other _____</p> <p>F. Electrical power for payload and ground station (1) kVA required _____ (2) Any special requirements such as clean/quiet power, or special phasing? Explain _____ (3) Backup power (diesel generator) _____ a. Continuous _____ b. During Critical Tests _____</p> <p>G. Communications (list) (1) Administrative telephone _____ (2) Commercial telephone _____ (3) Commercial data phones _____ (4) Fax machines _____ (5) Operational intercom system _____ (6) Closed-circuit television _____ (7) Countdown clocks _____ (8) Timing _____</p>	<p>(9) Antennas _____ (10) Data lines (from/to where) _____ (11) Type (wideband/narrowband) _____</p> <p>H. Services general (1) Gases a. Specification _____ Procured by user? _____ KSC? _____ b. Quantity _____ c. Sampling (yes) _____ (no) _____ (2) Photographs/Video _____ (qty/B&W/color) _____ (3) Janitorial (yes) _____ (no) _____ (4) Reproduction services (yes) _____ (no) _____</p> <p>I. Security (yes) _____ (no) _____ (1) Safes _____ (number/type) _____</p> <p>J. Storage _____ (size area) _____ _____ (environment)</p> <p>K. Other _____</p> <p>L. Spacecraft payload processing facility (PPF) activities calendar (1) Assembly and testing _____ (2) Hazardous operations _____ a. Initial turn-on of a high-power RF system _____ b. Category B ordnance installation _____ c. Initial pressurization _____ d. Other _____</p> <p>M. Transportation of payloads/GSE from PPF to HPF (1) Will spacecraft agency supply transportation canister _____ If no, explain _____ (2) Equipment support, (e.g., mobile crane, flatbed) _____ (3) Weather forecast (yes) _____ (no) _____ (4) Security escort (yes) _____ (no) _____ (5) Other _____</p> <p>3. Hazardous Processing Facility</p> <p>A. Does spacecraft require a cleanroom? (yes) _____ (no) _____ (1) Class of cleanroom required _____ (2) Special sampling techniques (e.g., hydrocarbon monitoring) _____</p> <p>B. Area required (1) For spacecraft _____ (2) For GSE _____</p> <p>C. Largest door size (1) For payload _____ high _____ wide (2) For GSE _____ high _____ wide</p> <p>D. Material handling equipment (1) Cranes a. Capacity _____ b. Hook height _____ c. Travel _____ (2) Other _____</p> <p>E. Environmental controls spacecraft/GSE (1) Temperature/humidity and tolerance limits _____ (2) Frequency of monitoring _____ (3) Down-time allowable in the event of a system failure _____ (4) Is a backup (portable) system required? (yes) _____ (no) _____ (5) Other _____</p> <p>F. Power for spacecraft and GSE (1) kVA required _____</p>
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Note: Please specify units as applicable.

Table 8-7. Spacecraft Checklist (Continued)

<p>G. Communications (list)</p> <p>(1) Administrative telephone _____</p> <p>(2) Commercial telephone _____</p> <p>(3) Commercial data phones _____</p> <p>(4) Fax machines _____</p> <p>(5) Operational intercom system _____</p> <p>(6) Closed-circuit television _____</p> <p>(7) Countdown clocks _____</p> <p>(8) Timing _____</p> <p>(9) Antennas _____</p> <p>(10) Data lines (from/to where) _____</p> <p>H. Services general</p> <p>(1) Gases</p> <p>a. Specification _____</p> <p>Procured by user? _____ KSC? _____</p> <p>b. Quantity _____</p> <p>c. Sampling (yes) _____ (no) _____</p> <p>(2) Photographs/Video _____ (qty/B&W/color) _____</p> <p>(3) Janitorial (yes) _____ (no) _____</p> <p>(4) Reproduction services (yes) _____ (no) _____</p> <p>I. Security (yes) _____ (no) _____</p> <p>(1) Safes _____ (number/type) _____</p> <p>J. Storage _____ (size area) _____</p> <p>_____ (environment)</p> <p>K. Other _____</p> <p>L. Spacecraft HPF activities calendar _____</p> <p>(1) Assembly and testing _____</p> <p>(2) Hazardous operations _____</p> <p>a. Category A ordnance installation _____</p> <p>b. Fuel loading _____</p> <p>c. Mating operations (hoisting) _____</p> <p>M. Transportation of encapsulated payloads to launch pad</p> <p>(1) Equipment support, e.g., mobile crane, flatbed _____</p> <p>_____</p> <p>(2) Weather forecast (yes) _____ (no) _____</p> <p>(3) Security escort (yes) _____ (no) _____</p> <p>(4) Other _____</p> <p>4. Launch Complex White Room Mobile Service Tower (MST)</p> <p>A. Environmental controls payload/GSE</p> <p>(1) Temperature/humidity and tolerance limits _____</p> <p>(2) Any special requirements such as clean/quiet power? Please detail requirements _____</p>	<p>(3) Backup power (diesel generator)</p> <p>a. Continuous _____</p> <p>b. During critical tests _____</p> <p>(4) Hydrocarbon monitoring required _____</p> <p>(5) Frequency of monitoring _____</p> <p>(6) Down-time allowable in the event of a system failure _____</p> <p>_____</p> <p>(7) Other _____</p> <p>B. Power for payload and GSE</p> <p>(1) kVA required _____</p> <p>(2) Any special requirements such as clean/quiet power/phasing? Explain _____</p> <p>(3) Backup power (diesel generator) _____</p> <p>a. Continuous _____</p> <p>b. During critical tests _____</p> <p>C. Communications (list)</p> <p>(1) Operational intercom system _____</p> <p>(2) Closed-circuit television _____</p> <p>(3) Countdown clocks _____</p> <p>(4) Timing _____</p> <p>(5) Antennas _____</p> <p>(6) Data lines (from/to where) _____</p> <p>D. Services general</p> <p>(1) Gases</p> <p>a. Specification _____</p> <p>Procured by user? _____ KSC? _____</p> <p>b. Quantity _____</p> <p>c. Sampling (yes) _____ (no) _____</p> <p>(2) Photographs/Video _____ (qty/B&W/color) _____</p> <p>E. Security (yes) _____ (no) _____</p> <p>F. Other _____</p> <p>G. Stand-alone testing (does not include tests involving the launch vehicle)</p> <p>(1) Tests required _____</p> <p>(e.g., RF system checkout, encrypter checkout)</p> <p>(2) Communications required for _____</p> <p>(e.g., antennas, data lines)</p> <p>(3) Spacecraft servicing required _____</p> <p>(e.g., cryogenics refill)</p>
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Note: Specify units as applicable

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Section 9 *SAFETY*

This section presents an overview of safety process guidelines, rules, and regulations pertaining to the design, test, and prelaunch operations of payloads to be placed in orbit by a Delta II vehicle. These guidelines, rules, and regulations are applicable to missions from the Eastern Range (Cape Canaveral Air Force Station) or the Western Range (Vandenberg Air Force Base).

9.1 SAFETY REQUIREMENTS

Since all payloads eventually arrive on USAF property for processing, the governing safety document shall always be Air Force Space Command Manual (AFSPCMAN) 91-710, Range Safety User Requirements, 1 July 2004. Prelaunch processing facilities are described in Sections 6 and 7. Depending on the type of payload and which facility will be used for processing, the following safety documents are also applicable:

- Astrotech, Titusville Florida Astrotech Space Operations Safety Standard Operating Procedure (SOP), 1988
- Kennedy Space Center,
Florida Kennedy NASA Procedural Requirements (KNPR) 8715.3, KSC Safety Practices–Procedural Requirements, 18 November 2004
- Astrotech West, VAFB Astrotech Space Operations Safety Standard Operating Procedures at VAFB, Sept 1994
- NASA-KSC, VAFB KNPR 8715.3, KSC Safety Practices–Procedural Requirements, 18 November 2004
- California Spaceport, VAFB Spaceport Systems International (SSI) Integrated Processing Facility Site Safety Plan (SSI Doc. IPF-95-SA01), Rev 1, May 1995.

Before a payload moves onto USAF property, the customer must provide the appropriate Space Wing (SW) Safety Office with documentation verifying that the payload has been designed and tested in accordance with the requirements of AFSPCMAN 91-710, Range Safety User Requirements. The Space Wing Safety organizations encourage payload contractors to coordinate with them to generate a tailored version of these requirements that is specific to each program. This tailoring policy can work to the advantage of the payload contractor and greatly simplify the safety approval process. The Delta Program provides coordination and assistance to the payload contractor by facilitating the tailoring and approval process.

9.2 DOCUMENTATION REQUIREMENTS

Both USAF and NASA require formal submittal of safety documentation containing detailed information on all hazardous systems and associated operations. The 30th and 45th Space Wings

(30 SW and 45 SW) at the Western and Eastern Ranges require preparation and submittal of a Missile System Prelaunch Safety Package (MSPSP). Document content and format requirements are found in the AFSPCMAN 91-710, Range Safety User Requirements, and should shape the tailoring process. Data requirements for both ranges include design, test, and operational considerations. NASA requirements in almost every instance are covered by the USAF requirements; however, the spacecraft agency can refer to KNPR 8715.3 for details or additional requirements.

A Ground Operations Plan must be submitted describing hazardous and safety-critical operations for processing spacecraft systems and associated ground support equipment (GSE).

Test and Inspection Plans are required for the use of hoisting equipment and pressure vessels at the ranges. These plans describe testing methods, analyses, and maintenance procedures ensuring compliance with safety requirements.

The requirement for diligent and conscientious preparation of the required safety documentation cannot be overemphasized. Each of the USAF launch range support organizations retains final approval authority over all hazardous operations that take place within its jurisdiction. Therefore, the spacecraft agency should consider the safety requirements of Paragraph 9.1 from the outset of a program, follow them for design guidance, and submit the required data as early as possible.

The safety document is submitted to the appropriate government agency, or to Boeing for commercial missions, for review and further distribution. Sufficient copies of the original and all revisions must be submitted by the originator to enable a review by all concerned agencies. The review process usually requires several iterations until the system design and its intended use are considered to be final and in compliance with all safety requirements. The flow of spacecraft safety information is dependent on the range to be used, the customer, and contractual arrangements. Figure 9-1 illustrates the general documentation flow. Some differences exist depending on whether the payload is launching from the Eastern Range or the Western Range. Contact the Delta Program Office for specific details.

Each Air Force and NASA safety agency has a requirement for submittal of documentation for emitters of ionizing and nonionizing radiation. Required submittals depend on the location, use, and type of emitter and may consist of forms and/or analyses specified in the pertinent regulations and instructions.

An RF ordnance hazard analysis must be performed, documented, and submitted to confirm that the spacecraft systems and the local RF environment present no hazards to ordnance on the spacecraft or launch vehicle.

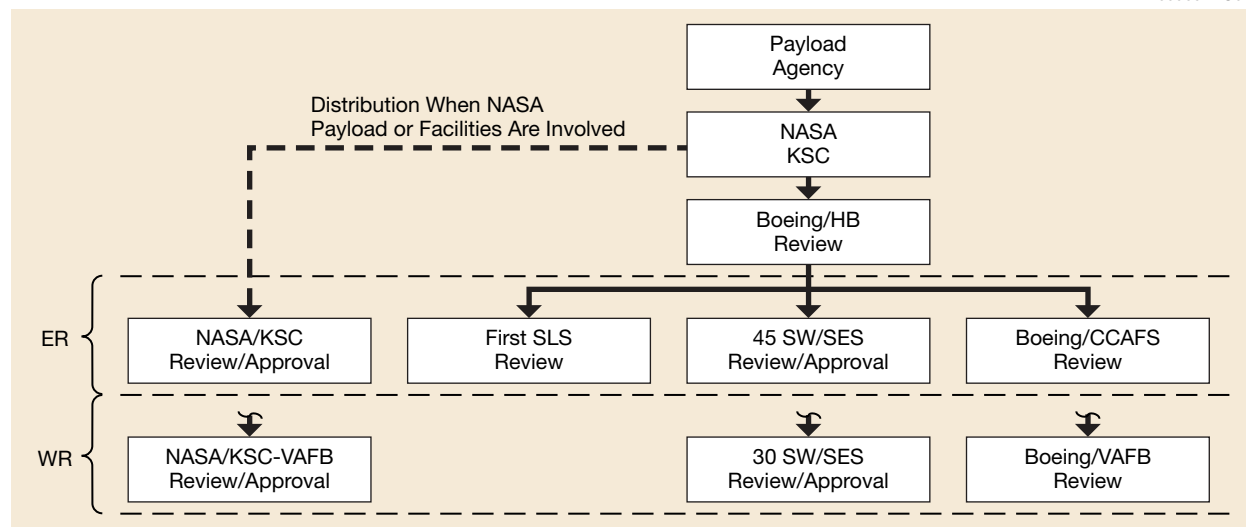


Figure 9-1. General Safety Documentation Flow

Each processing procedure that includes hazardous operations must have a written procedure approved by Space Wing Safety (and NASA Safety for NASA facilities). Those that involve Delta Program personnel or integrated operations with the launch vehicle must also be approved by Boeing Test and Operational Safety.

9.3 HAZARDOUS SYSTEMS AND OPERATIONS

The requirements cited in the Range Safety Regulations apply for hazardous systems and operations. However, Boeing safety requirements are, in some cases, more stringent than those of the launch range. The design and operations requirements governing activities involving Boeing participation are discussed in the following paragraphs.

9.3.1 Operations Involving Pressure Vessels (Tanks)

In order for Delta Program personnel to be safely exposed to pressurized vessels, the vessels must be designed, built, and tested to meet minimum factor-of-safety requirements (ratio between design burst pressure and operating pressure) in accordance with AFSPCMAN 91-710, Chapter 3. The Delta Program Office desires a minimum factor of safety of 2 to 1 for all pressure vessels that will be pressurized in the vicinity of Delta Program personnel. Analyses and test documentation verifying the pressure vessel safety factor must be included in the spacecraft safety documentation.

Any operation that requires pressurization at the launch site or after mating to Boeing equipment must be approved by the Delta Program Office and must be conducted remotely (no personnel exposure) after which a minimum 5-minute stabilization period must be observed prior to personnel exposure.

9.3.2 Nonionizing Radiation

The spacecraft nonionizing radiation systems are subject to the design criteria in the USAF and KSC manuals and the special Delta-imposed criteria as follows:

- Systems producing nonionizing radiation will be designed and operated so that the hazards to personnel are at the lowest practical level.
- Delta Program employees are not to be exposed to nonionizing radiation above 10 mW/cm^2 averaged over any 1-minute interval. Safety documentation shall include the calculated distances at which a level of 10 mW/cm^2 (194 V/m) occurs for each emitter of nonionizing radiation even if no operations are planned. This requirement is separate and distinct from the requirement to submit the radiation source documentation mentioned in Paragraph 9.2.
- Depending on power, frequency, and antenna locations, RF radiation (both planned and inadvertent) by the spacecraft can have a detrimental effect on launch vehicle electronics and ordnance. For this reason, all planned transmissions prior to spacecraft separation must be coordinated early to determine effects on the launch vehicle. Additionally, the Delta Program requires that two inhibits be incorporated into spacecraft designs to prevent unplanned RF emissions prior to separation. If this is not accomplished, actual designs must be reviewed for potential radiation and effects and approved by the Delta Program Office.

9.3.3 Liquid Propellant Offloading

Range Safety Regulations require that spacecraft be designed with the capability to offload liquid propellants from tanks during any stage of prelaunch processing. Any tank, piping, or other components containing propellants must be capable of being drained and then flushed and purged with inert fluids should a leak or other contingency necessitate propellant offloading to reach a safe state. Spacecraft designs should consider the number and placement of drain valves to maintain accessibility by technicians in Propellant Handler's Equipment (PHE) or a self-contained atmospheric protection ensemble (SCAPE) throughout processing. Coordinate with the Delta Program Office to ensure that access can be accomplished while the payload fairing is in place and that proper interfaces can be achieved with Delta equipment and facilities.

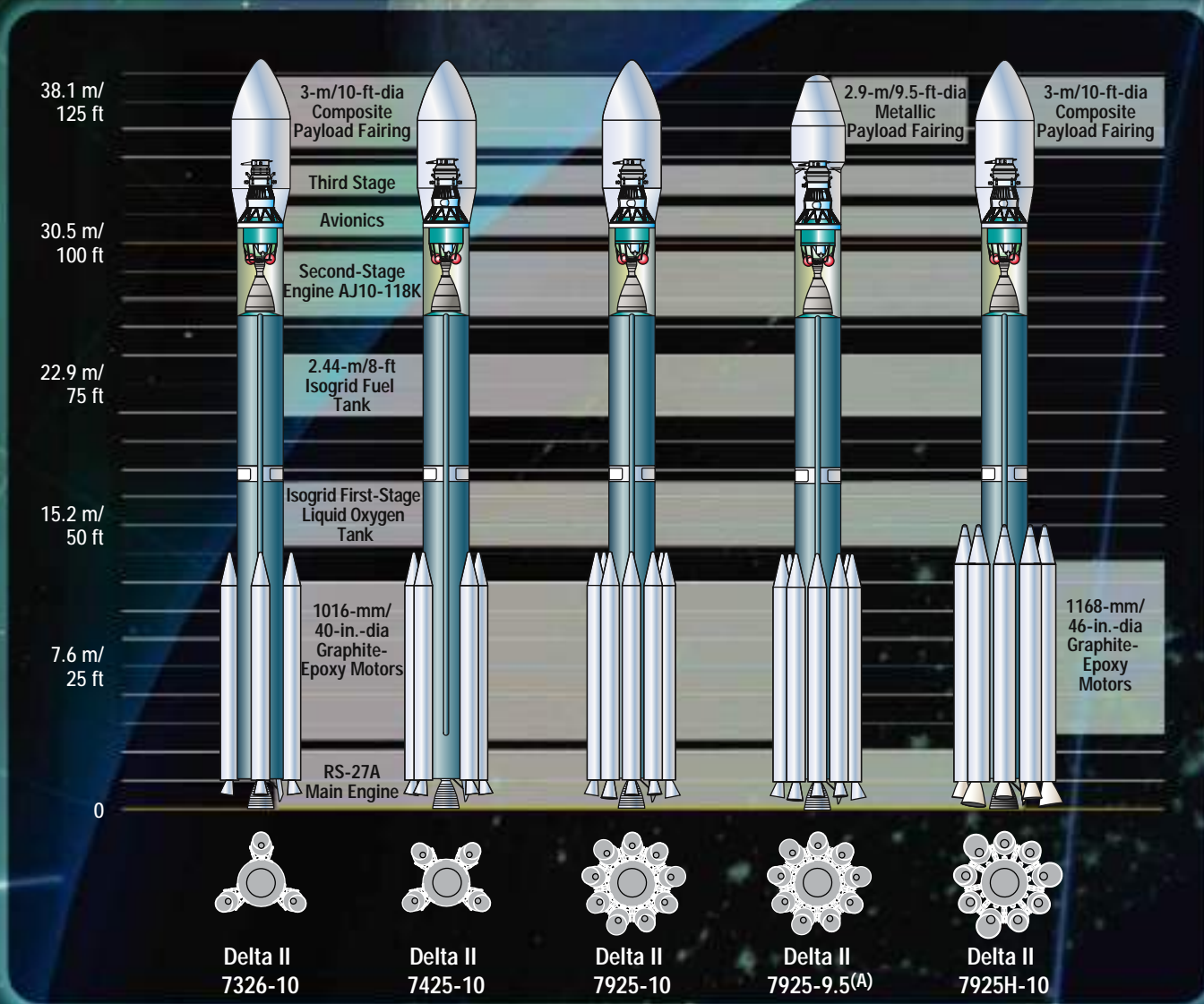
9.3.4 Safing of Ordnance

Manual ordnance safing devices (S&A or safing/arming plugs) for Range Category A ordnance are also required to be accessible with the payload fairing installed. Consideration should be given to placing such devices so that they can be reached through fairing openings and can be armed as late in the countdown as possible, and safed in the event of an aborted/scrubbed launch if required. Early coordination with the Delta Program Office is needed to ensure that the required fairing access door(s) can be provided.

9.4 WAIVERS

Space Wing Safety organizations discourage the use of waivers. They are normally granted only for spacecraft designs that have a history of proven safety. After a complete review of all safety requirements, the spacecraft agency should determine if waivers are necessary. A waiver or Meets Intent Certification (MIC) request is required for any safety-related requirement that cannot be met. If a noncompliant condition is suspected, coordinate with the appropriate Space Wing Safety organization to determine whether a Waiver or Meets Intent Certification will be required. Requests for waivers shall be submitted prior to implementation of the safety-related design or practice in question. Waiver or MIC requests must be accompanied by sufficient substantiating data to warrant consideration and approval. It should be noted that the USAF Space Wing Safety organizations determine when a waiver or MIC is required and have final approval of all requests. No guarantees can be made that approval will be granted.

DELTA II LAUNCH VEHICLE CONFIGURATIONS



HB6T017

Delta Launch Vehicle Programs