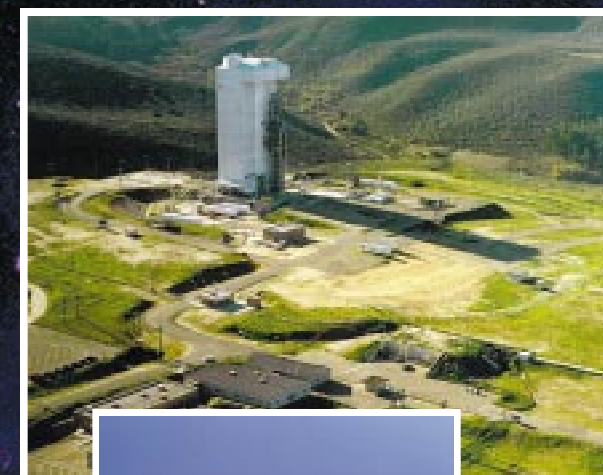
ATLAS

Atlas Launch System Mission Planner's Guide



ILS

INTERNATIONAL LAUNCH SERVICES

THE INDUSTRY SUPERPOWER

LOCKHEED MARTIN

Revision 7

December 1998

ATLAS LAUNCH SYSTEM MISSION PLANNER'S GUIDE

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REVISION NOTICE

This document supersedes the Mission Planner's Guide for the Atlas Launch Vehicle Family—Revision 6 February 1997

DISCLOSURE OF DATA LEGEND

These technical data are exempt from the licensing requirements of International Traffic in Arms Regulation (ITAR) under ITAR Part 125.4(b)(13) and are cleared for public release.
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FOREWORD

This *Atlas Launch System Mission Planner's Guide* presents information on the vehicle capabilities of the Atlas launch system. A range of vehicle configurations and performance levels is offered to allow an optimum match to customer requirements at low cost. The performance data are presented in sufficient detail for preliminary assessment of the Atlas vehicle family for your missions.

The guide includes essential technical and programmatic data for preliminary mission planning and preliminary spacecraft design. Interfaces are in sufficient detail to assess a first-order compatibility. A brief description of the Atlas vehicles and launch facilities is also given. See the companion *Atlas Launch Services Facility Guide* for spacecraft processing and launch services at Space Launch Complex 36. The *Atlas Launch Campaign Guide* defines the operations and hardware flow for the spacecraft and Atlas vehicle leading to encapsulation, spacecraft mate to the launch vehicle, and launch countdown procedures.

This guide is subject to changes and will be revised periodically. Revision 7 has extensive updates from Revision 6. All updates have been documented on the revisions page.

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ATLAS LAUNCH SYSTEM MISSION PLANNER'S GUIDE REVISIONS

Revision Date	Rev No.	Change Description	Approval
October 1998	7	• Section 1—Updates Include – Added Atlas IIIA/IIIB Information Where Appropriate – Updated Figure for Atlas Launch Vehicle Family – Atlas I and II Vehicle Program Now Complete – Updated Figure for IIA/IIAS Vehicle Systems Characteristics – Updated Figure for IIIA Vehicle Systems Characteristics – Added Figures for IIIB DEC & SEC Vehicle Systems Characteristics – Updated Vehicle and Ground System Interfaces – Updated Flight-Derived Guidance Accuracy Data – Updated for Current Launch Achievements – Revised List of Atlas Enhancements	
	7	• Section 2—Updates Include – Updated Number for Atlas Missions Flown – Combined Atlas IIA/IIAS and IIIA/IIIB Sequence of Table – Updated Atlas IIA/IIAS Performance Capabilities, Ascent Profile, and Sequence Data – Added Atlas IIIA/IIIB(DEC/SEC) Performance Capabilities, Ascent Profile, and Sequence Data – Updated Mission Accuracy for Current Flight History – Added Boost Phase Ascent Profile for Atlas IIIA/IIIB – Added RD-180 Throttle Profile for Atlas IIIA/IIIB – Added Centaur Phase Ascent Profile for Atlas IIIA/IIIB	
	7	• Section 3—Updates Include – Added Atlas IIIA/IIIB Information Where Appropriate – Updated Minimum Flow Rate Data – Updated Gas Impingement Velocity Data – Revised RF Environments at CCAS for Deleted RF Sources – Revised Discussion of Helium Environments – Added (FASSN) Discussion	
	7	• Section 4—Updates Include – Updated As Needed for Atlas IIIA/IIIB; Rewrote Entire Spacecraft to Launch Vehicle Interface Section – Minor Clarifications/Corrections to Interface Drawings – Expanded Static Payload Envelope – Increased Structural Capability for Improved Centaur Equipment Module – Updated for Current Range Safety Compliance Documentation	
	7	• Section 5—Updates Include – Updated As Needed for Atlas IIIA/IIIB – Deleted MWG Discussions – Revised Integration Schedule from an 18- to 12-month Process – Revised Coupled Loads Analysis Discussion – Revised RF Link Compatibility (Airborne) Discussions – Deleted FMEA Discussions – Updated Launch Scheduling Section To Include Contract to Launch Duration of 12 to 18 months – Updated Launch Scheduling Figure To Reflect 12 Launches Per Year from CCAS	
	7	• Section 6—Updates Include – Revisions for Current Launch Site Facilities Availability – Updates to Astrotech Facility Information for CCAS and VAFB – Updated CCAS MST Figure To Show Movable Platforms – Updates to SLC-3E Facility Information	

<table>
<tr><th>Revision Date</th><th>Rev No.</th><th>Change Description</th><th>Approval</th></tr>
<tr><td></td><td>7</td><td>• Section 7—Updates Include
– Updates To Avoidance of Lighting Requirements
– Updates To Adjust discussions</td><td></td></tr>
<tr><td></td><td>7</td><td>• Section 8—Updates Include
– Revised for Current Atlas Enhancement After Moving Centaur Extended Mission Kit and Short Burn Capability into Standard Capabilities
– Added Discussions for the Universal Spacecraft Separation Node (UASSN)
– Added to Discussions for the Fast Acting Shockless Separation Node (FASSN)
– Updates to Centaur Type F Adapter Discussions
– Updates to Strengthened Payload Adapter for Atlas IIIB Heavy Payload Capability
– Added Atlas V Discussions</td><td></td></tr>
<tr><td></td><td>7</td><td>• Appendixes—Updates Include
– Appendix A
• Updated As Needed for Atlas IIIA/IIIB
• Revised Centaur Development Discussions and Figures
• Revised History for Current Flight Accomplishments
• Updates to RD-180 Discussions
• Revised Centaur Propellant Utilization Discussion
• Updates to Vehicle Reliability Discussions
– Appendix B
• Updates to Mission Success® and Product Assurance Discussions
• Updates to Corrective and Preventive Action Process
– Appendix C
• Updates to CAD Data Transfer Process</td><td></td></tr>
</table>

TABLE OF CONTENTS

Page

Figure

Table

GLOSSARY

A

A	Ampere
ac	Alternating Current
A/C	Air Conditioning
ADDJUST	Automated Determination and Dissemination of Just Updated Steering Terms
ADJ	Attach, Disconnect, and Jettison
AF	Air Force
AFQA	Air Force Quality Assurance
AFR	Air Force Regulation
AFSCF	Air Force Satellite Control Facility
AFSCN	Air Force Space Control Network
AGE	Aerospace Ground Equipment
AMSC	American Mobile Satellite Corporation
ARCM	Atlas Roll Control Module
AREP	Atlas Reliability Enhancement Program
ARIA	Advanced Range Instrumentation Aircraft
ARSR	Air Route Surveillance Radar
AS	Atlas Station
ASCII	American Standard Code for Information Interchange
ATS	Advanced Technology Satellite
AWG	American Wire Gage

B

Batt	Battery
BECO	Booster Engine Cutoff
BOS	Booster on Stand
bpi	Bit(s) per Inch
BPJ	Booster Package Jettison
BPSK	Binary Phase Shift Key
Btu	British Thermal Unit

C

°C	Degree(s) Celsius
CAB	Customer Awareness Board
CAD	Computer-Aided Design
CCAM	Collision and Contamination Avoidance Maneuver
CCAPS	Computer-Controlled Atlas Pressurization System
CCAS	Cape Canaveral Air Station
CCLS	Computer-Controlled Launch Set
CCTV	Closed-Circuit Television
CCVAPS	Computer-Controlled Vent and Pressurization System
CDR	Critical Design Review
CERT	Composite Electrical Readiness Test
cg	Center of Gravity
CIB	Change Integration Board
CLA	Coupled Loads Analysis
CLS	Commercial Launch Services
CRES	Corrosion-Resistant Steel
CRRES	Combined Release and Radiation Effects Satellite
CS	Centaur Station
CSO	Complex Safety Officer
CT	Command Transmitter
CV	Computervision
CW	Continuous Wave
CWA	Controlled Work Area
CX	Complex

D

DAS	Data Acquisition System
DAT	Digital Audio Tape
dB	Decibel(s)
dBm	Decibel(s) Relative to 1 Milliwatt
dBW	Decibel(s) Relative to 1 Watt
dc	Direct Current
DEC	Digital Equipment Corporation
DEC	Dual-Engine Centaur
DLF	Design Load Factor
DMSP	Defense Meteorological Satellite Program
DOD	Department of Defense
DOF	Degree(s) of Freedom
DPF	Defense System Communications Satellite (DSCS) Processing Facility
DSCS	Defense System Communications Satellite
DSN	Deep Space Network
Dstr	Destructor
DUF	Dynamic Uncertainty Factor

E

EB	East Bay
ECA	Environmentally Controlled Area
ECS	Environmentally Controlled System
EED	Electro-Explosive Device
EHF	Extreme High Frequency
EIA	Electronics Industry Association
ELAN	Electronic Library for Analysis of Nonconformances
EM	Electromagnetic
EMARS	Electronic Martin Anomaly Reporting System
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EMK	Extended Mission Kit
EPA	Environmental Protection Agency
EPF	Extended Payload Fairing
ER	Eastern Range
ERB	Engineering Review Board
ERR	Eastern Range Regulation
ESABASE	Thermal Control Geometric Math Modeling Code
ESMCR	Eastern Space and Missile Center Regulation
EUTELSAT	European Telecommunications Satellite Organization
EVCF	Eastern Vehicle Checkout Facility
EWR	Eastern/Western Range Regulation
ε	Emissivity

F

°F	Degree(s) Fahrenheit
FAB	Final Assembly Building
FAST	Flight Analogy Software Test
FCDC	Flexible Confined Detonating Cord
FCS	Flight Control Subsystem
FDLC	Final Design Loads Cycle
FLSC	Flexible Linear-Shaped Charge
FLTSATCOM	Fleet Satellite Communications
FM	Flight Model
FM	Frequency Modulation
FMEA	Failure Modes and Effects Analysis
FOD	Foreign Object Damage
FOM	Figure of Merit
FOTS	Fiber-Optics Transmission System
FPA	Flight Plan Approval
FPR	Flight Performance Reserve
FSO	Flight Safety Officer
ft	Foot; Feet
FTP	File Transfer Protocol
FTS	Flight Termination System

G

GCS	Guidance Commanded Shutdown
GG	Gas Generator
GEO	Geosynchronous Orbit
GHe	Gaseous Helium
GIDEP	Government-Industry Data Exchange Program
GMM	Geometric Mathematical Model
G&N	Guidance and Navigation
GN&C	Guidance, Navigation, and Control
GN_2	Gaseous Nitrogen
GOES	Geostationary Operational Environmental Satellite
GOWG	Ground Operations Working Group
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
GSO	Geostationary Orbit
GSTDN	Ground Spaceflight Tracking and Data Network
GTO	Geosynchronous Transfer Orbit
GTR	Gantry Test Rack
GTS	Ground Telemetry Station
GTV	Ground Transport Vehicle

H

HAIR	High Accuracy Instrumentation Radar
HAWK	U.S. Government Radar System
HEAO	High-Energy Astronomy Observatory
HEO	High-Energy (High-Eccentricity) Orbit
HEPA	High-Efficiency Particulate Air
HP	Hewlett-Packard
HPF	Hazardous Processing Facility
hr	Hour(s)
Hz	Hertz

I

ICD	Interface Control Document
ICWG	Interface Control Working Group
I/F	Interface
IFR	Inflight Retargeting
IGES	Initial Graphics Exchange Specification
IIP	Instantaneous Impact Point
ILC	Initial Launch Capability
ILS	International Launch Services
IMS	Inertial Measurement Subsystem
in.	Inch(es)
INTELSAT	International Telecommunications Satellite Organization
INU	Inertial Navigation Unit
IPJ	Insulation Panel Jettison
IRD	Interface Requirements Document
ISA	Interstage Adapter
ISD	Interface Scheduling Document
ISDS	Inadvertent Separation Destruct System
I_{SP}	Specific Impulse
ITA	Integrated Thermal Analysis

J

JCSAT	Japan Communications Satellite
JOSLE	Jettison of Shroud with Latches and Elasticity
JPL	Jet Propulsion Laboratory

K

kg	Kilogram(s)
km	Kilometer(s)
kN	Kilonewton(s)
kPa	Kilopascal(s)
KSC	Kennedy Space Center
kV	Kilovolt(s)

L

LAE	Liquid Apogee Engine
LAN	Longitude of Ascending Node
lb	Pound(s)
lbf	Pound(s)-Force
LC	Launch Conductor
LCC	Launch Control Center
LEO	Low-Earth Orbit
LeRC	Lewis Research Center
LH_2	Liquid Hydrogen
LHe	Liquid Helium
LKEI	Lockheed Khrunichev Energia International
LM	Lockheed Martin
lm	Lumen(s)
LMCLS	Lockheed Martin Commercial Launch Services
LN_2	Liquid Nitrogen
LO_2	Liquid Oxygen
LOB	Launch Operations Building
LPF	Large Payload Fairing
LR	Load Ratio
LSB	Launch Services Building
LSC	Linear Shaped Charge
LV	Launch Vehicle
LVDC	Launch Vehicle Data Center
LVMP	Launch Vehicle Mission Peculiar

M

m	Meter(s)
MDRD	Module-Level Design Requirements Document
MDU	Master Data Unit
MECO	Main Engine Cutoff
MES	Main Engine Start
MICD	Mechanical Interface Control Drawing
MIL-STD	Military Standard
MILA	Merritt Island Launch Area Radar
min	Minute(s)
MLV	Medium Launch Vehicle
mm	Millimeter(s)
MOTR	Multiple Object Tracking Radar
MPDR	Mission-Peculiar Design Review
MPF	Medium Payload Fairing
MRB	Material Review Board
MRR	Mission Readiness Review
MRS	Minimum Residual Shutdown
MS&PA	Mission Success® and Product Assurance
MSPSP	Missile System Prelaunch Safety Package
MST	Mobile Service Tower
μV	Microvolt(s)
mV	Millivolt(s)
MWG	Management Working Group

N

N	Newton(s)
NASA	National Aeronautics and Space Administration
NCAR	National Center for Atmospheric Research
NEVADA	Thermal Control Geometric Math Modeling Code
NEXRAD	Next Generation Radar
N_2H_4	Hydrazine
NIOSH	National Institute of Occupational Safety and Health
NISE	Naval In-Service Engineering
NM	Not Measured
nmi	Nautical Mile(s)
NOAA	National Oceanic and Atmospheric Administration
NRZ-L	Non-Return-to-Zero Logic
ns	Nanosecond(s)
NVR	Nonvolatile Residue

O

OAO	Orbiting Astronomical Observatory
OASPL	Overall Sound Pressure Level
Oct	Octave(s)
OIS	Orbit Insertion Stage
ORD	Operational Requirements Document
OSS	Ocean Surveillance System
OTM	Output Transformation Matrix
Ω	Ohm(s)
ω_p	Argument of Perigee

P

Pa	Pascal
PA	Public Address
PCM	Pulse-Code Modulation
PCOS	Power Changeover Switch
PDLC	Preliminary Design Loads Cycle
PDR	Preliminary Design Review
PDRD	Program-Level Design Requirements Document
PFJ	Payload Fairing Jettison
PFM	Protoflight Model
PHA	Preliminary Hazard Analysis
PHSF	Payload Hazardous Servicing Facility
PLCP	Propellant Leak Contingency Plan
PLCU	Propellant Loading Control Unit
PLF	Payload Fairing
PLIS	Propellant Level Indicating System
PMPCB	Parts, Materials, and Processes Control Board
PMR	Preliminary Material Review
POD	Program Office Directive
PPF	Payload Processing Facility
psi	Pound(s) per Square Inch
psig	Pound(s) per Square Inch, Gage
PSS	Payload Separation System
PST	Product Support Team
PSW	Payload Systems Weight
PSWC	Payload Systems Weight Capability
PTC	Payload Test Conductor
PTC	Payload Transport Canister
PU	Propellant Utilization
PVA	Perigee Velocity Augmentation
PVC	Polyvinyl Chloride
P&W	Pratt & Whitney
Pwr	Power

Q

QA	Quality Assurance
QIC	Quarter-Inch Cartridge

R

RAAN	Right Ascension of Ascending Node
RCS	Reaction Control System
RCU	Remote Control Unit
R&D	Research and Development
RDU	Remote Data Unit
RDX	Research Department Explosive
RF	Radio Frequency
RGU	Rate Gyro Unit
RLCC	Remote Launch Control Center
Rm	Room
RNCO	Range Noncommissioned Officer
ROI	Return on Investment
RPO	Radiation Protection Officer
RSC	Range Safety Console
RTS	Remote Tracking Station

S

s	Second(s)
S/A	Safe and Arm
SAEF	Spacecraft Assembly and Encapsulation Facility

SARA	Superfund Amendments Reauthorization Act
SASU	Safe/Arm and Securing Unit
SC	Spacecraft
SCAPE	Self-Contained Atmospheric Protective Ensemble
SDP	Software Documentation Plan
SDRC	Structural Dynamics Research Corporation
SECO	Sustainer Engine Cutoff
SEC	Single-Engine Centaur
SEPP	Systems Effectiveness Program Plan
SFC	Spacecraft Facility Controller
SFTS	Secure Flight Termination System
SHA	System Hazard Analysis
SIL	Systems Integration Laboratory
SINDA	System-Improved Numerical Differencing Analyzer
SIU	Servo-Inverter Unit
SLC	Space Launch Complex
SLC	Spacecraft Launch Conductor
SOHO	Solar and Heliospheric Observatory
SPRB	Space Program Reliability Board
SQEP	Software Quality Evaluation Plan
SQP	Sequential Quadratic Programming
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SRR	System Requirements Review
Sta	Station
STC	Satellite Test Center
STDN	Spaceflight Tracking and Data Network
STM	Structural Test Model
STS	Space Transportation System
STV	Spacecraft Transport Vehicle
SW	Space Wing

T

tar	Tape Archive (File Format)
TBD	To Be Determined
TC	Telecommand
TC	Test Conductor
TCD	Terminal Countdown Demonstration
TCO	Thrust Cutoff
TDRSS	Tracking and Data Relay Satellite System
TEMP	Test and Evaluation Master Plan
TIM	Technical Interchange Meeting
Tlm	Telemetry
TMM	Thermal Mathematical Model
TMRSS	Triple Modular Redundant Securing System
TOPS	Transistorized Operational Phone System
TPQ	Transportable Radar Special
TRAJEX	Trajectory Executive
TRASYS	Thermal Radiation Analysis System
TRM	Tension Release Mechanism
TSB	Technical Support Building
TVC	Thrust Vector Control
TVCF	Transportable Vehicle Checkout Facility
TWG	Technical Working Group
3-D	Three-Dimensional

U

UHF	Ultra-High Frequency
UPS	Uninterruptible Power System
USAF	United States Air Force
USN	United States Navy
UT	Umbilical Tower

V

V	Volt(s)
Vac	Volt(s) Alternating Current
VAFB	Vandenberg Air Force Base
Vdc	Volt(s) Direct Current
VDD	Version Description Document
VHF	Very High Frequency
VSWR	Voltage Standing-Wave Ratio
VTF	Vertical Test Facility

W

W	Watt(s)
WB	West Bay
WDR	Wet Dress Rehearsal
WR	Western Range
WRR	Western Range Regulation
WSMCR	Western Space and Missile Center Regulation
WSR	Weather Service Radar

X

Xdcr	Transducer

1.0 INTRODUCTION

1.1 SUMMARY

The *Atlas Launch System Mission Planner's Guide* is designed to provide current and potential Atlas launch services customers with information about the Atlas launch vehicle family and related spacecraft services. The Atlas family (Fig. 1.1-1) includes the flight-proven Atlas IIA and IIAS versions and the IIIA and IIIB versions in development. A full range of technical planning data is included to allow the user to assess the compatibility of the user's payload with the various interfaces that comprise the Atlas system.

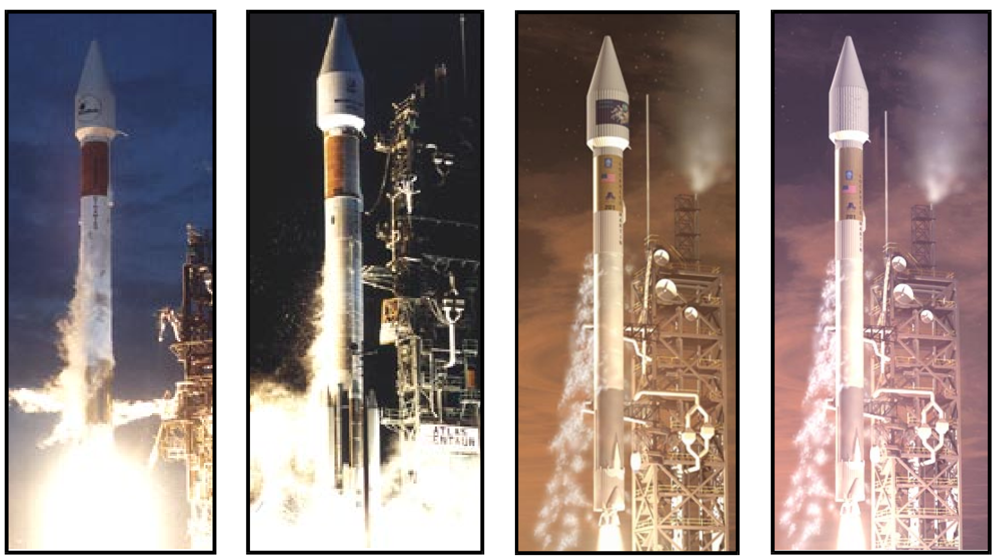
1.2 LAUNCH SERVICES

Atlas is offered to commercial and government launch services users through International Launch Services (ILS), a joint venture between the Lockheed Martin Corporation and the Lockheed-Khrunichev-Energia International Incorporated (LKEI) joint venture. ILS markets and manages Atlas and Proton launch services using a dedicated team of technical, management, and marketing specialists to readily define and refine the optimum and most cost-effective space transportation solutions for the launch services customer.

ILS operates as a strategic alliance between the Lockheed Martin Corporation, the manufacturer and operator of Atlas, and the Khrunichev State Research and Production Space Center and the Rocket Space Company Energia named for S. P. Korolev, the manufacturers and operators of the Russian Proton launch vehicle. Lockheed Martin Commercial Launch Services (CLS), a constituent company of ILS, operates as the legal contracting entity for all Atlas launch services. ILS's other constituent company, LKEI, operates as the legal contracting entity for Proton launch services.

The ILS organization is aligned to offer space launch services for Atlas and/or Proton, and, in addition, the unique services that are available with the alliance of the two mature, flight-proven launch systems using the shared resources of both constituent companies.

ATLAS



	ATLAS IIA	ATLAS IIAS	ATLAS IIIA	ATLAS IIIB
Initial Launch Capability	June 1992	December 1993	December 1998	September 2000
Performance to GTO	3,066 kg (6,760 lb)	3,719 kg (8,200 lb)	4,037 kg (8,900 lb)	4,500 kg (9,920 lb)

Figure 1.1-1 The Atlas Launch Vehicle Family

1.3 LAUNCH SERVICES ORGANIZATION AND FUNCTION

Through ILS, Lockheed Martin offers a full launch service, from spacecraft integration, processing and encapsulation, through launch operations and verification of orbit. The typical launch service includes:

1) Launch vehicle;
2) Launch operations services;
3) Mission-peculiar hardware and software design, test, and production;
4) Technical launch vehicle/spacecraft integration and interface design;
5) Mission management;
6) Launch facilities and support provisions;
7) Payload processing facilities;
8) Spacecraft support at the launch site;
9) Validation of spacecraft separation sequence and orbit;
10) Range safety interface.

With authority to proceed on a launch services contract, the Atlas program immediately assigns a single point of contact, the mission manager, to be responsible for program development and management. The manager acts as the customer's advocate to the various organizations within Lockheed Martin, its subcontractors, and suppliers. The manager's primary duty is to arrange the resources necessary for the successful completion of the launch services contract and to ensure complete customer satisfaction. In cases that involve the national security interests of the United States, separate integration management resources of the Lockheed Martin Astronautics organization are called on to handle the unique requirements typically present with integration and launch of these types of payloads.

As part of our Atlas launch services contract support, administrative guidance and assistance can be provided, when needed, in meeting government regulations, including import and export licenses, permits, and clearances from government and political entities

1.4 ADVANTAGES OF SELECTING ATLAS

All government and commercial agreements required to conduct Atlas launch services are maintained for our customer. Agreements are in place covering payload and Atlas launch vehicle processing facilities, services, and Range support at Cape Canaveral Air Station (CCAS) in Florida. Similar agreements are in work for comparable services at Vandenberg Air Force Base (VAFB) in California.

Our launch vehicles and services provide the following key advantages:

1) Flight-proven Atlas flight vehicle and ground system hardware and processes;
2) Moderate, fully validated, payload launch environments (e.g., shock, vibration, acoustic, thermal) that are generally lower than those of other launch vehicles;
3) Two launch pads at CCAS to ensure launch schedules and maintain commitments;
4) A West Coast launch facility for launch of high-inclination satellite missions;
5) Single-payload manifesting to ensure launch service dedication and responsiveness;
6) An experienced team that has launched more than 70 communications satellites;
7) Mission design flexibility demonstrated in a diverse array of mission types, including most U.S. planetary missions and numerous geostationary transfer orbit (GTO) missions;
8) A flexible mission design capability providing maximum spacecraft onorbit lifetime through optimized use of spacecraft and Centaur capabilities;
9) The combined resources and experience of the Atlas and Proton launch services team to meet the challenging commercial launch services needs of the future.

1.5 LAUNCH SYSTEM CAPABILITIES AND INTERFACES

From the user's perspective, the Atlas launch system is comprised of a number of hardware- and software-based subsystems and engineering, manufacturing, and operations processes designed to properly interface the spacecraft with our space transportation vehicle. The following paragraphs summarize the major interface and process components of the Atlas system. Each subject corresponds to an appropriate section of this document where much more detailed information can be found.

1.5.1 Atlas Launch System

The Atlas launch vehicle system consists of the Atlas booster, the Centaur upper stage, the payload fairing (PLF), and a payload interface (Figs. 1.5.1-1 and 1.5.1-2). The Atlas launch vehicle family includes the Atlas IIA, Atlas IIAS, Atlas IIIA, and Atlas IIIB launch vehicles.

With the evolution of commercial and government launch services requirements, Lockheed Martin is concentrating its Space Systems resources on the continued availability of the operational Atlas IIA and Atlas IIAS launch vehicles and development of the Atlas IIIA and Atlas IIIB launch vehicles for future missions. The Atlas I and Atlas II vehicle program is complete. Future Atlas I and Atlas II class payloads can use the Atlas IIA, IIIA, or IIIB single-engine Centaur (SEC) vehicle for space transportation requirements.

Atlas IIA and Atlas IIAS vehicles will continue to be offered well into the next decade. Figure 1.5.1-3 summarizes the characteristics of the IIA and IIAS Atlas vehicle systems.

The Atlas IIIA and Atlas IIIB vehicles are the next evolutionary versions and will be phased in beginning in late 1998. The upper-stage configuration for Atlas IIIB can be either the dual-engine Centaur (DEC) or SEC. Figure 1.5.1-4 summarizes the characteristics of the Atlas IIIA, Figure 1.5.1-5 summarizes the characteristics of Atlas IIIB (DEC), and Figure 1.5.1-6 summarizes the characteristics of the Atlas IIIB (SEC) vehicle systems.

1.5.2 Atlas Mission Design and Performance

On more than 100 past missions, Atlas has demonstrated its capability to deliver payloads of various volumes and masses to precisely targeted orbits. Because the Atlas vehicle is primarily designed for dedicated, single payload missions, a number of Atlas-unique, flight-proven mission trajectory and targeting enhancement options can be offered to maximize the benefits of the Atlas system to the spacecraft mission. As demonstrated by flight-derived guidance accuracy data in Table 1.5.2-1, Atlas provides the most accurate orbit placement capability in the expendable launch vehicle industry today. Section 2.0 discusses the launch vehicle mission and performance capabilities available for a full range of Atlas IIA, IIAS, IIIA, and IIIB launch missions.

1.5.3 Atlas Launch System Environments

The Atlas launch system provides the spacecraft preflight and flight environments that are typically more benign than those available with other launch systems. All environments specified for the Atlas launch system (e.g., shock, vibration, acoustic, thermal, electromagnetic) are based on engineering analyses and have been fully validated with test and flight telemetry data on all Atlas vehicle configurations. Verification that the launch service users' flight environments remained within specified levels can be obtained with use of additional instrumentation along with either a digital telepak or frequency modulation (FM)/FM telepak and instrumentation. This hardware enables telemetering of high-frequency measurements concerning the payload interface and environment. These supplemental measurements, when added to our standard low-frequency flight telemetry, can allow verification of the interface control document (ICD) flight environments for each Atlas mission (Mission Satisfaction Option). The environments to which spacecraft are exposed are fully discussed in Section 3.0.

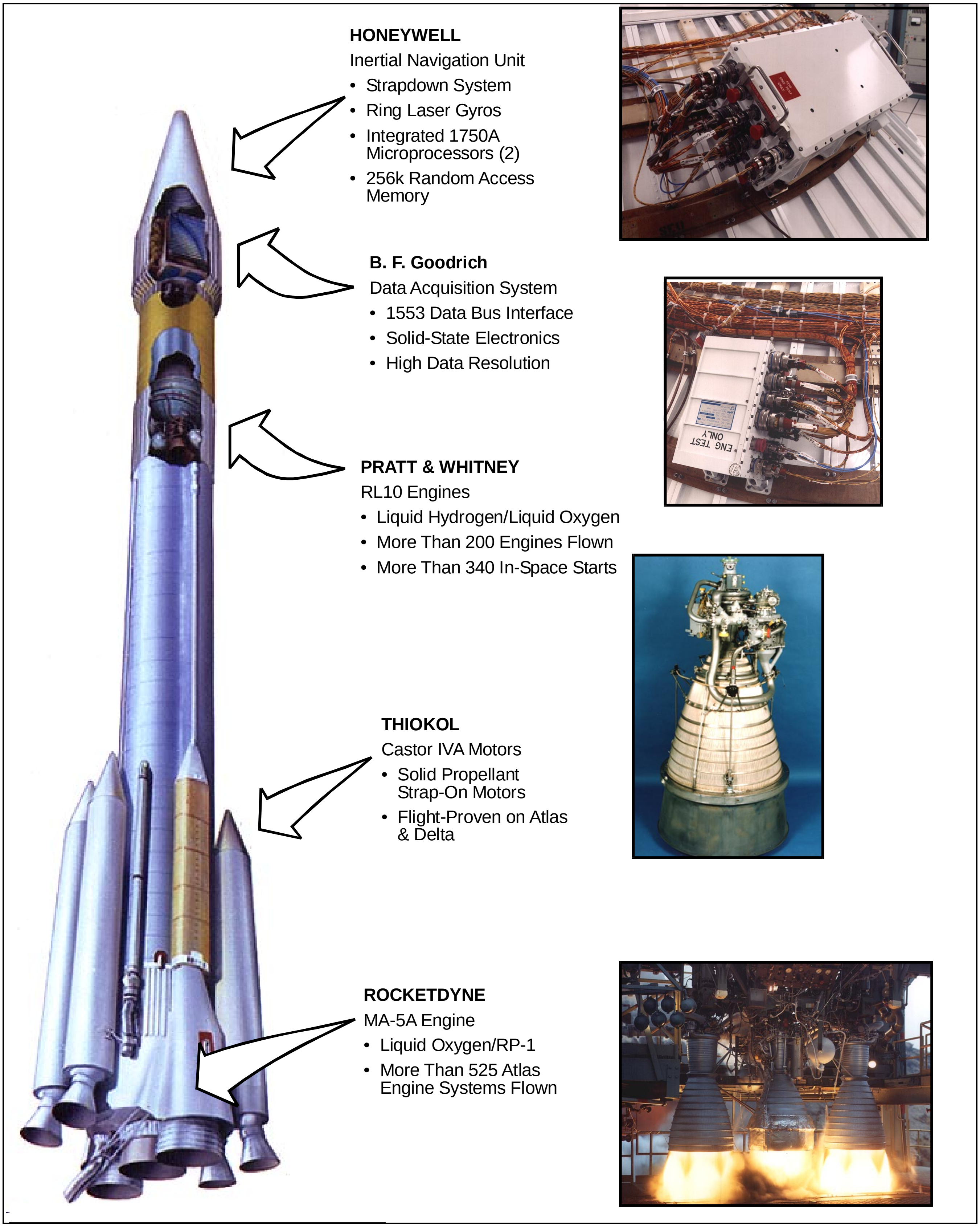


Figure 1.5.1-1 The Atlas IIA and IIAS launch vehicles offer flight-proven hardware.

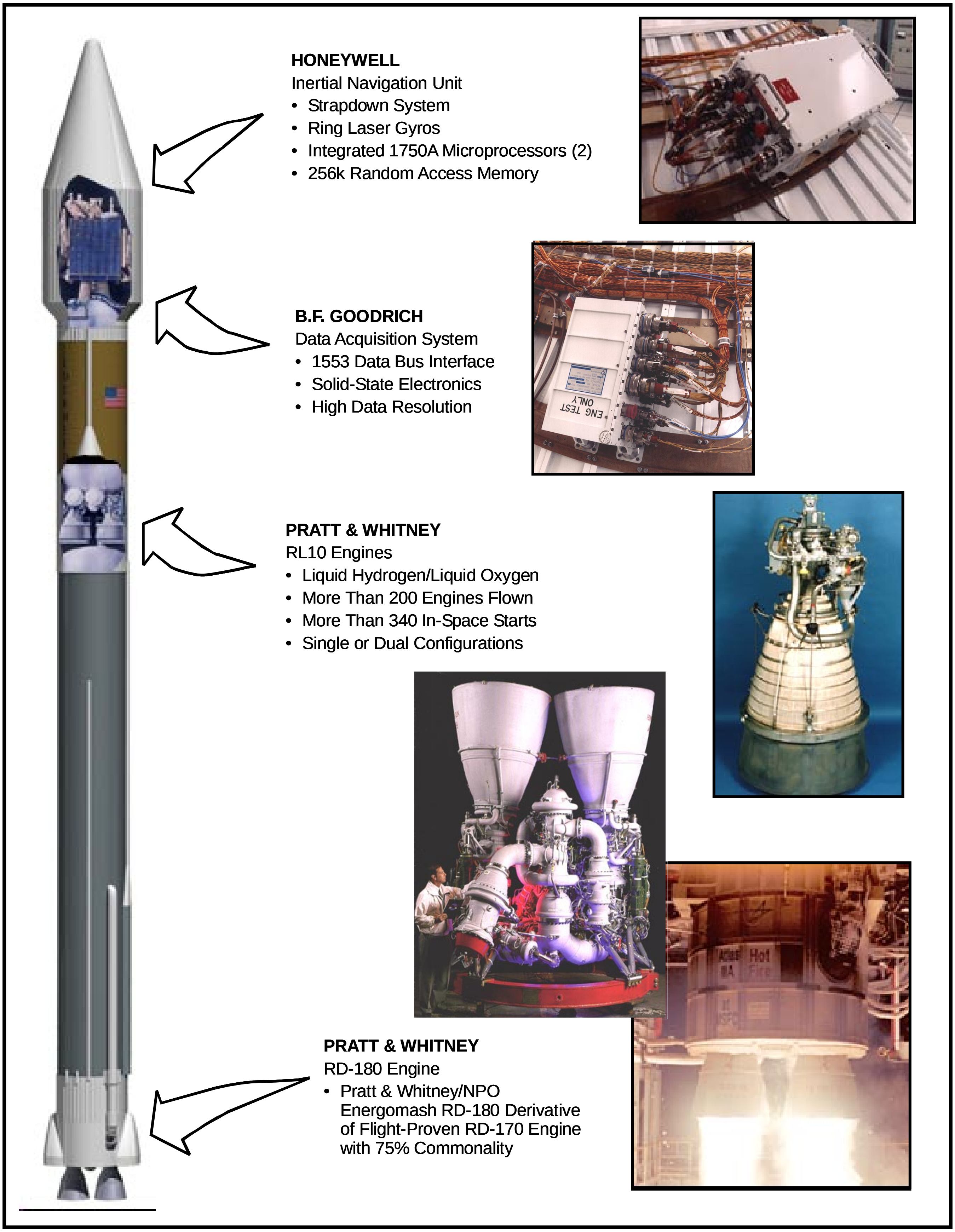
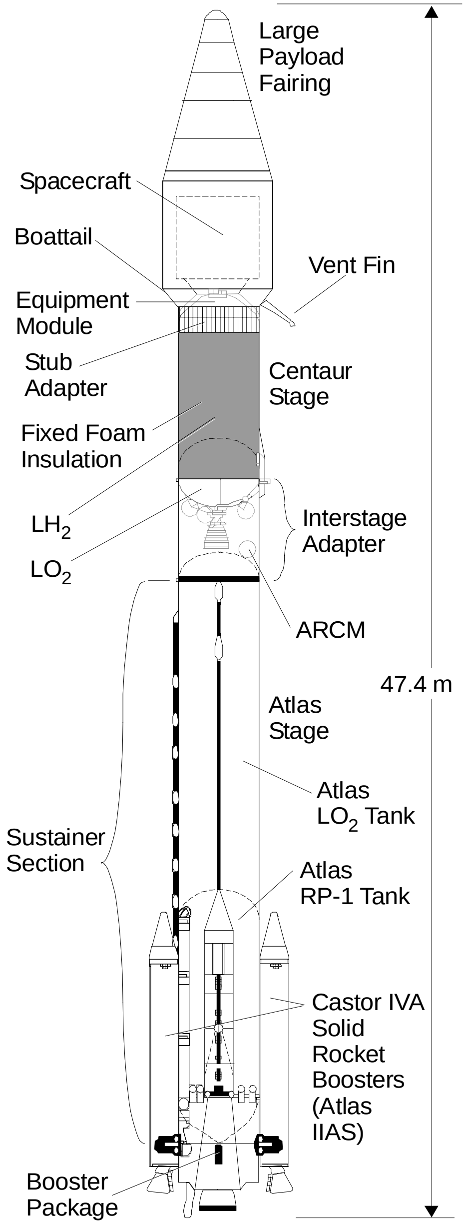


Figure 1.5.1-2 The Atlas IIIA and IIIB launch vehicles offer flight-proven hardware.



PAYLOAD FAIRING

Features:	Large (Shown)	Medium	Extended
Diameter:	4.2 m	3.3 m	4.2 m
Length:	12.2 m	10.4 m	13.1 m
Mass:	2,087 kg	1,409 kg	2,255 kg

Subsystems:	
Fairing:	Aluminum Skin Stringer & Frame Clamshell
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Clamshell
Separation:	Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE

Features:		
Size:	3.05-m Dia x 10.06-m Length	
Inert Mass:	1,840 kg	
Propellant:	16,930 kg LH_2 & LO_2	
Guidance:	Inertial	
Subsystems:		
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead	
Propulsion:	Two Pratt & Whitney Restartable Engines	
— Option:	RL10A-4	RL10A-4-1
— Thrust:	181.5 kN	198.4 kN
— I_{SP}:	449.0 s	450.5 s
	Two 51-cm Columbium Extendible Nozzle	
	(12) 27-N Hydrazine Thrusters	
Pneumatics:	Helium & Hydrogen Autogenous (Tank Pressurization)	
Hydraulics:	Provides Two Separate Systems for Gimbaling Main Engines	
Avionics:	Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, C-Band Tracking, Range Safety Command, Electrical Power	
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick),	

INTERSTAGE ADAPTER (ISA)

Features:	
Size:	3.05-m Dia x 3.99-m Length
Mass:	545 kg
Subsystems:	
Structure:	Aluminum Skin Stringer & Frame
Separation:	Flexible Linear-Shaped Charge
Propulsion:	Atlas Roll Control Module

ATLAS STAGE

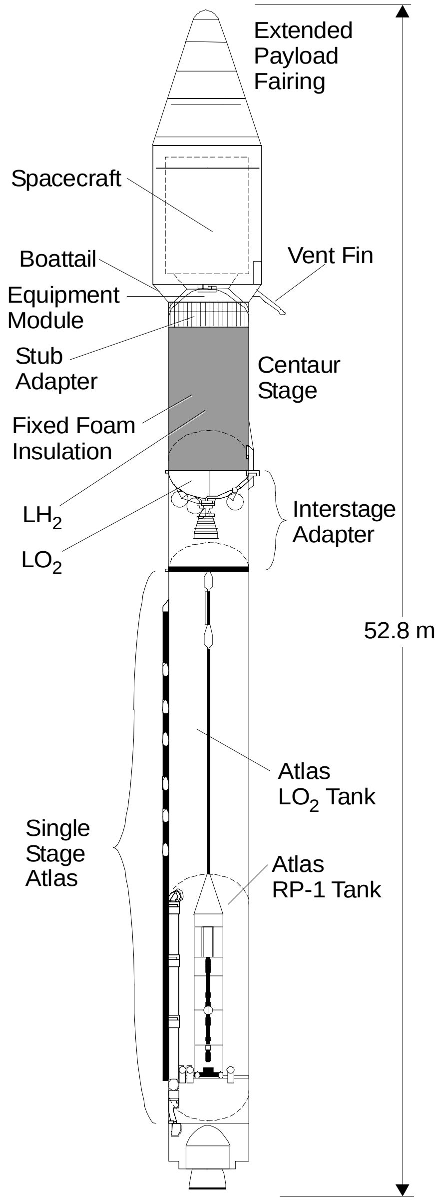
Features:	
Size:	3.05-m Dia x 24.90-m Length
Propellant:	156,400-kg LO_2 & RP-1
Guidance:	From Upper Stage
Subsystems:	
Structure:	Pressure-Stabilized Stainless Steel Tanks; Integrally Machined Thrust Structure
Separation:	Booster Package—10 Pneumatically Actuated Separation Latches Sustainer Section—8 Retro Rockets
Propulsion:	Rocketdyne MA-5A Booster Engine (2 Chambers) SL Thrust = 1,854 kN, I_{SP} = 262.1 s Vac Thrust = 2,065 kN, I_{SP} = 293.4 s Sustainer Engine SL Thrust = 266.0 kN, I_{SP} = 216.1 s Vac Thrust = 380.6 kN, I_{SP} = 311.0 s
Pneumatics:	Helium (Pressure Tanks), Hydraulics & Lubrication System
Hydraulics:	Provides Two Separate Systems for Main Engine Gimbal
Avionics:	Flight Control, Flight Termination, Telemetry, Computer-Controlled Atlas Pressurization System, Rate Gyro Unit

SOLID ROCKET BOOSTERS (SRB)

Four Thiokol Castor IVA	
Size:	102-cm Dia x 13.6-m Length
Mass:	11,567 kg (Each Fueled)
Thrust:	433.7 kN (Each)
I_{SP}:	237.8 s

Ground-Lit Motors 11° Cant Nozzles
Air-Lit Motors 7° Cant Nozzles
Inadvertent Separation Destruct System (ISDS)

Figure 1.5.1-3 Our modernized Atlas IIA and IIAS launch system is flight-proven and capable of meeting a wide variety of mission requirements.



PAYLOAD FAIRING

Features:	Large	Extended (Shown)
Diameter:	4.2 m	4.2 m
Length:	12.2 m	13.1 m
Mass:	2,087 kg	2,255 kg

Subsystems:

Fairing:	Aluminum Skin Stringer & Frame Clamshell
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Clamshell
Separation:	Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE

Features:

Size:	3.05-m Dia x 10.06-m Length
Inert Mass:	1,720 kg
Propellant:	16,930 kg LH_2 & LO_2
Guidance:	Inertial

Subsystems:

Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	One Pratt & Whitney Restartable Engine
— Type:	RL10A-4-1
— Thrust:	99.2 kN
— I_{SP}:	450.5 s
	One Electromechanically Actuated 51-cm Columbium Extendible Nozzle
	(12) 27-N Hydrazine Thrusters
Pneumatics:	Helium & Hydrogen Autogenous (Tank Pressurization)
Avionics:	Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, C-Band Tracking, Range Safety Command, Electrical Power, Redundant EMA Actuators for Gimbal Control
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

INTERSTAGE ADAPTER (ISA)

Features:

Size:	3.05-m Dia x 4.45-m Length
Mass:	465 kg

Subsystems:

Structure:	Aluminum-Lithium Skin Stringer & Frame
Separation:	Flexible Linear-Shaped Charge

ATLAS STAGE

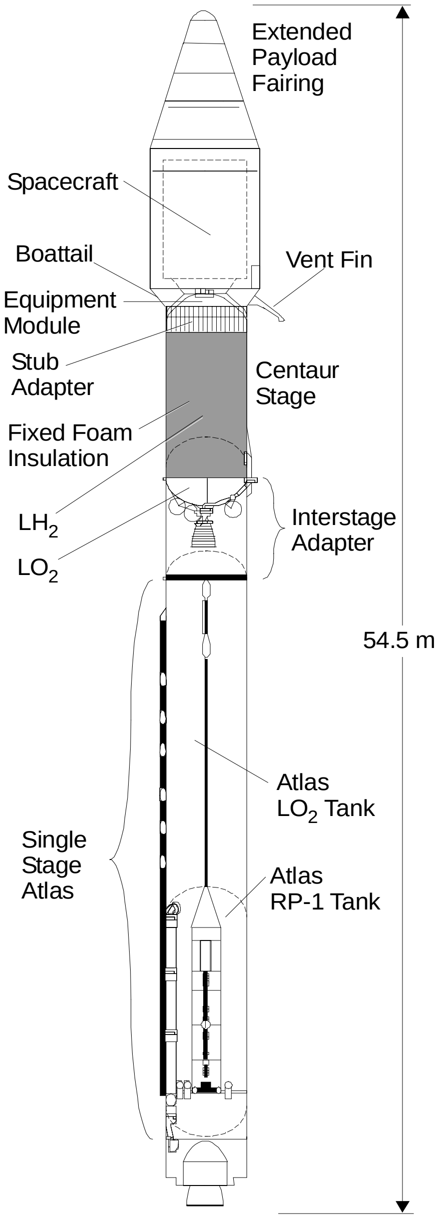
Features:

Size:	3.05-m Dia x 28.91-m Length
Propellant:	183,200-kg LO_2 & RP-1
Guidance:	From Upper Stage

Subsystems:

Structure:	Pressure-Stabilized Stainless Steel Tanks; Integrally Machined Thrust Structure
Separation:	Single Stage Atlas—12 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers) SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
Pneumatics:	Helium (Pressure Tanks)
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Computer-Controlled Atlas Pressurization System, Rate Gyro Unit, Remote Control Unit

Figure 1.5.1-4 The Atlas IIIA continues the proven Atlas IIA/AS design legacy with increased performance and reduced design complexity.



PAYLOAD FAIRING

Features:	Large	Extended (Shown)
Diameter:	4.2 m	4.2 m
Length:	12.2 m	13.1 m
Mass:	2,087 kg	2,255 kg

Subsystems:	
Fairing:	Aluminum Skin Stringer & Frame Clamshell
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Clamshell
Separation:	Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE

Features:	
Size:	3.05-m Dia x 11.74-m Length
Inert Mass:	2,110 kg
Propellant:	20,830 kg LH_2 & LO_2
Guidance:	Inertial

Subsystems:	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	Two Pratt & Whitney Restartable Engines
— Type:	RL10A-4-2
— Thrust:	198.4 kN
— I_{SP}:	450.5 s
	Two 51-cm Columbium Extendible Nozzle
	(12) 40-N Hydrazine Thrusters
Pneumatics:	Helium & Hydrogen Autogenous (Tank Pressurization)
Hydraulics:	Provides Two Separate Systems for Gimbaling Main Engines
Avionics:	Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, C-Band Tracking, Range Safety Command, Electrical Power
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

INTERSTAGE ADAPTER (ISA)

Features:	
Size:	3.05-m Dia x 4.45-m Length
Mass:	465 kg

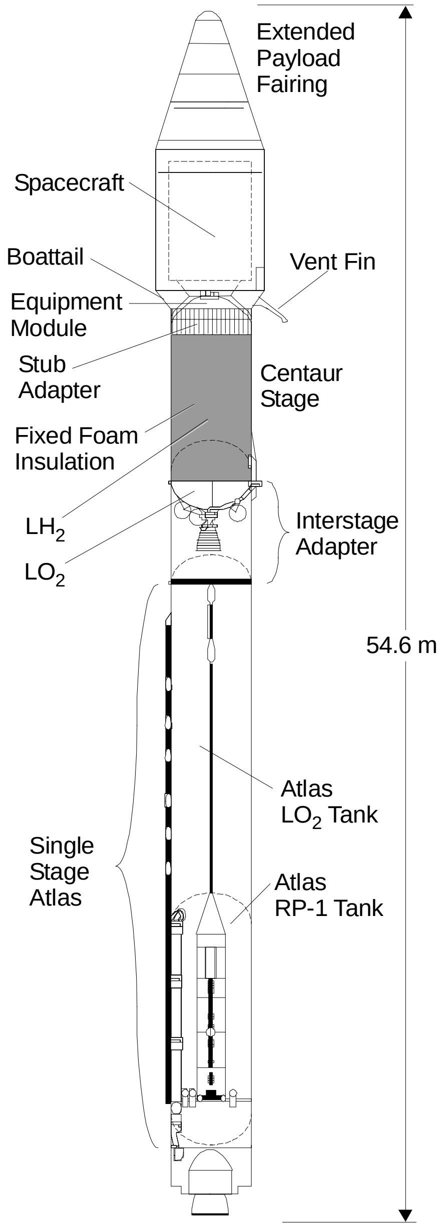
Subsystems:	
Structure:	Aluminum-Lithium Skin Stringer & Frame
Separation:	Flexible Linear-Shaped Charge

ATLAS STAGE

Features:	
Size:	3.05-m Dia x 28.91-m Length
Propellant:	183,200-kg LO_2 & RP-1
Guidance:	From Upper Stage

Subsystems:	
Structure:	Pressure-Stabilized Stainless Steel Tanks; Integrally Machined Thrust Structure
Separation:	Single Stage Atlas—12 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers)
	SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s
	Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
Pneumatics:	Helium (Pressure Tanks)
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Computer-Controlled Atlas Pressurization System, Rate Gyro Unit, Remote Control Unit

Figure 1.5.1-5 The Atlas IIIB (DEC) continues the proven Atlas IIA/AS design legacy with increased performance and reduced design complexity.



PAYLOAD FAIRING

Features:	Large	Large Extended (Shown)
Diameter:	4.2 m	4.2 m
Length:	12.2 m	13.1 m
Mass:	2,087 kg	2,255 kg

Subsystems:	
Fairing:	Aluminum Skin Stringer & Frame Clamshell
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Clamshell
Separation:	Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE

Features:	
Size:	3.05-m Dia x 11.89-m Length
Inert Mass:	1,930 kg
Propellant:	20,830 kg LH_2 & LO_2
Guidance:	Inertial
Subsystems:	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	One Pratt & Whitney Restartable Engine
— Type:	RL10A-4-2
— Thrust:	99.2 kN
— I_{SP}:	450.5 s
	One Electromechanically Actuated 51-cm Columbium Extendible Nozzle
	(12) 40-N Hydrazine Thrusters
Pneumatics:	Helium & Hydrogen Autogenous (Tank Pressurization)
Avionics:	Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, C-Band Tracking, Range Safety Command, Electrical Power, Redundant EMA Actuators for Gimbal Control
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

INTERSTAGE ADAPTER (ISA)

Features:	
Size:	3.05-m Dia x 4.45-m Length
Mass:	465 kg
Subsystems:	
Structure:	Aluminum-Lithium Skin Stringer & Frame
Separation:	Flexible Linear-Shaped Charge

ATLAS STAGE

Features:	
Size:	3.05-m Dia x 28.91-m Length
Propellant:	183,200-kg LO_2 & RP-1
Guidance:	From Upper Stage
Subsystems:	
Structure:	Pressure-Stabilized Stainless Steel Tanks; Integrally Machined Thrust Structure
Separation:	Single Stage Atlas—12 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers)
	SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s
	Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
Pneumatics:	Helium (Pressure Tanks)
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Computer-Controlled Atlas Pressurization System, Rate Gyro Unit, Remote Control Unit

Figure 1.5.1-6 The Atlas IIIB (SEC) continues the proven Atlas IIA/AS design legacy with increased performance and reduced design complexity.

Table 1.5.2-1 Flight-Derived Guidance Accuracy Data

Vehicle	Δ Apogee, km (nmi)	Δ Perigee, km (nmi)	Δ i, °	Δ ωp, °	Δ RAAN, °
AC-102	2.57 (1.39)	0.33 (0.18)	-0.002	0.011	–
AC-101	-5.82 (-3.14)	-0.48 (-0.26)	0.003	-0.026	–
AC-72	-10.17 (-5.49)	-0.28 (-0.15)	-0.005	0.008	–
AC-103	-20.50 (-11.07)	-0.72 (-0.39)	0.001	0.024	–
AC-74	-3.91 (-2.11)	-1.20 (-0.65)	0.001	0.000	–
AC-104	-11.45 (-6.18)	-0.39 (-0.21)	-0.002	0.008	–
AC-75	-7.59 (-4.10)	-0.35 (-0.19)	-0.009	0.009	–
AC-106	-6.50 (-3.51)	-0.46 (-0.25)	-0.001	0.015	–
AC-108	-1.61 (-0.87)	1.50 (0.81)	0.001	-0.026	–
AC-73	-7.80 (-4.21)	1.20 (0.65)	0.005	0.034	–
AC-76	-3.04 (-1.64)	-0.04 (-0.02)	-0.002	-0.002	–
AC-107	-10.76 (-5.81)	-0.06 (-0.03)	0.004	0.029	–
AC-111	-8.78 (-4.74)	0.57 (0.31)	-0.014	0.102	–
AC-110	-81.23 (-43.86)*	-0.65 (-0.35)	-0.020	-0.040	–
AC-113	-37.73 (-20.37)	-0.85 (-0.46)	-0.001	-0.005	–
AC-112	-12.15 (-6.56)	-0.46 (-0.25)	0.000	0.017	–
AC-115	-42.84 (-23.13)	N/A	-0.0003	0.030	–
AC-114	-17.69 (-9.55)	-0.17 (-0.09)	-0.053	-0.040	–
AC-77	-15.54 (-8.39)	-0.17 (-0.09)	-0.011	-0.002	–
AC-116	-13.70 (-7.40)	-0.39 (-0.21)	-0.0005	-0.000	–
AC-118	-2.6 (-1.4)	-0.26 (-0.14)	0.001	0.010	–
AC-117	-22.9 (-12.4)	-0.18 (-0.10)	0.034	0.100	–
AC-119	-14.4 (-7.8)	-0.52 (-0.28)	-0.003	0.000	–
AC-120	-1.8 (-1.0)	-0.87 (-0.47)	-0.102	-0.300	–
AC-126	-81.6 (-44.1)*	-0.46 (-0.25)	-0.001	-0.020	–
AC-122	-5.6 (-3.0)	-2.39 (-1.29)	0.006	0.000	–
AC-78	1.8 (1.0)	-2.96 (-1.60)	-0.001	N/A	–
AC-125	-8.5 (-4.6)	0.24 (0.13)	-0.007	0.020	–
AC-123	-32.2 (-17.4)	1.30 (0.70)	-0.003	0.020	–
AC-124	3.87 (2.09)	1.30 (0.70)	-0.003	0.020	–
AC-129	19.6 (10.6)	3.5 1.9)	-0.039	-0.150	–
AC-127	N/A	N/A	N/A	N/A	–
AC-128	-12.22 (-6.6)	1.33 (0.72)	0.004	0.10	–
AC-79	-10.37 (-5.6)	.17 (0.09)	-0.002	0.03	–
AC-133	4.26 (2.3)	-.61 (-0.33)	0.070	0.005	–
AC-146	-20.19 (-10.9)	-2.44 (-1.32)	0.033	0.093	–
AC-135	-12.41 (-6.7)	-1.67 (-0.90)	-0.005	0.192	–
AC-131	14.63 (7.9)	0.	0.002	0.019	–
AC-149	-14.07 (-7.6)	-1.94 (-1.05)	-0.032	0.150	–
AC-109	-20.372 (-11.0)	-1.11 (-0.60)	-0.007	0.014	–
AC-151	-5.0 (-2.7)	1.48 (0.80)	0.005	0.018	–
AC-132	-7.41 (-4.0)	-.24 (-0.13)	0.003	0.018	–
Note: *AC-110 & AC-126 Targeted to Greater Than 90,000-km (48,600-nmi) Apogee Altitude					

1.5.4 Vehicle and Ground System Interfaces

The Atlas launch system offers a range of interface options to meet spacecraft mission requirements. The primary interfaces between the launch vehicle and spacecraft consist of the payload adapter, which supports the spacecraft on top of the launch vehicle equipment module, and the payload fairing, which encloses and protects the spacecraft during ground operations and launch vehicle ascent. The Atlas program has eight standard payload adapters and three standard payload fairings available to the launch service customer (Fig. 1.5.4-1). These standard payload fairings and payload adapters are fully compatible with Atlas IIA, IIAS, IIIA, or IIIB configurations. Section 4.0 describes these components and specifies our flight vehicle and ground system interfaces with the spacecraft.

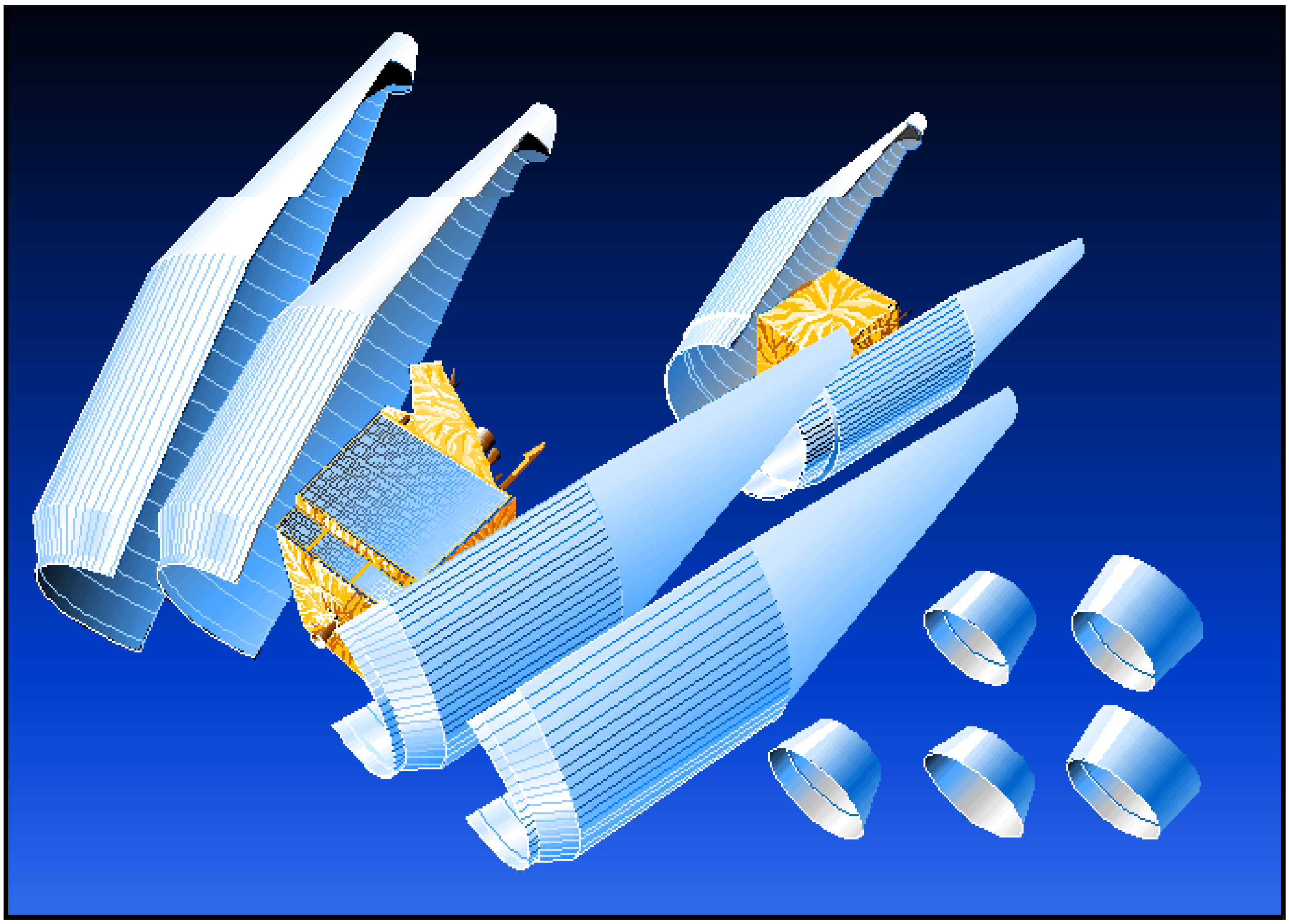


Figure 1.5.4-1 Atlas Fairings and Adapters

1.5.5 Atlas Mission Integration and Management

The Atlas mission integration and management process is designed to effectively use the engineering and production talents of Lockheed Martin Astronautics and spacecraft contractor organizations to integrate the customer's spacecraft onto the Atlas launch vehicle. Section 5.0 is an overview of the mission integration process and launch services management functions in place for commercial and government missions. For typical communications spacecraft missions, the 18-month integration schedule is discussed. Our management approach and a summary of integration analyses tasks are provided to enable the customer to fully understand the mission integration process.

1.5.6 Spacecraft and Launch Facilities

Upon arrival at the launch site, most spacecraft require the use of payload and/or hazardous processing facilities for final checkout of onboard systems before launch. Section 6.0 summarizes the payload processing facilities available for final spacecraft processing. In addition, the operational capabilities and interfaces of the Atlas launch complexes in operation at CCAS in Florida and the recently dedicated complex at VAFB in California are defined.

1.5.7 Atlas Launch Operations

Atlas launch operations processes require the involvement of the launch services customer and spacecraft contractor. Section 7.0 is a first-order overview of our operations processes, discussing issues that the potential launch services user may wish to consider early in the mission integration process.

1.5.8 Atlas Enhancements

Section 8.0 is designed to provide insight to the Atlas user community of Lockheed Martin's plans for enhancing the Atlas vehicle to meet the medium to intermediate launch services needs of the 21st Century. Some of the enhancements include:

1) Spacecraft dispensers and universal spacecraft separation node;
2) Centaur Type F adapter interface;
3) Payload adapter/payload separation system upgrades;
4) EELV family of vehicles development and performance data.

Contact information concerning requests for additional information can be found in the Foreword.

1.5.9 Supplemental Information

Three appendixes are provided in this document to address various issues in more detail:

1) Appendix A discusses the history of the Atlas and Atlas/Centaur launch vehicle. A more detailed description of the Atlas and Centaur stages and subsystems used for Atlas IIA, IIAS, IIIA, and IIIB is provided. The reliability growth methodology for evaluating mission and vehicle reliability also is summarized.
2) Appendix B details our Mission Success philosophy and quality assurance process firmly in place at our facilities and at our subcontractors and suppliers.
3) Appendix C defines the desired exchange of technical data requirements to support the initial mission development process. In addition, a discussion of the type and format of technical data we need is summarized to provide insight into the exchange of information between the spacecraft contractor, launch services customer, and ourselves that occurs during typical mission integration.

1.6 CONCLUSION

The members of the Atlas team are eager to assist in the definition and development of potential future Atlas missions. The Foreword of this document contains data to allow the potential launch services customer to contact the appropriate ILS/CLS representative to discuss mission needs.

2.0 ATLAS MISSION DESIGN AND PERFORMANCE

Over the past three decades, Atlas and Centaur stages have flown together as the Atlas/Centaur and with other stages (e.g., Atlas/Agena and Titan/Centaur) to deliver commercial, military, and scientific payloads to their target orbits. Based on our experiences with more than 550 Atlas launches, performance for each launch vehicle is determined by engineering analysis of developed and new hardware, emphasizing conservative performance prediction to ensure each vehicle meets design expectations. As all Atlas IIA and Atlas IIAS performance configurations are flight-proven, engineering estimates of Atlas family performance capabilities reflect flight-qualified hardware performance characteristics and our improved knowledge of developed hardware. Table 2-1 illustrates performance capabilities of the Atlas IIA/IIAS family; Table 2-2 illustrates performance capabilities of the Atlas IIIA/IIIB family.

As suggested by the table, Atlas is capable of being launched from Cape Canaveral Air Station (CCAS) in Florida and is planned to be launched from Vandenberg Air Force Base (VAFB) in California. This section further describes Atlas IIA/IIAS and IIIA/IIIB mission and performance options available with both East and West Coast launches.

Table 2-1 Atlas IIA/IIAS Performance Capabilities Summary

PSWC, kg (lb)		
Atlas IIA	**Atlas IIAS**	**Fairing**
GEOSYNCHRONOUS TRANSFER 167x35,788 km (90x19,324 nmi), i = 27.0°, ω_p = 180°, 99% GCS		
3,180 (7,010)	3,833 (8,450)	MPF
3,066 (6,760)	3,719 (8,200)	LPF
LOW-EARTH ORBIT FROM CCAS 185-km (100-nmi) Circular Orbit, i = 28.5°, 99.87% GCS		
7,316 (16,130)	8,618 (19,000)	LPF
LOW-EARTH ORBIT FROM VAFB 185-km (100-nmi) Circular Orbit, i = 90°, 99.87% GCS		
6,192 (13,650)	7,212 (15,900)	LPF
Note: PSWC: Payload Systems Weight Capability GCS: Guidance Commanded Shutdown MPF: Medium 3.3-m (11-ft) Payload Fairing LPF: Large 4.2-m (14-ft) Payload Fairing		

Table 2-2 Atlas IIIA/IIIB Performance Capabilities Summary

PSWC, kg (lb)		
Atlas IIIA	**Atlas IIIB**	**Fairing**
GEOSYNCHRONOUS TRANSFER 167 x 35,788 km (90 x 19,324 nmi), i = 27.0°, ω_p = 180°, 99% GCS		
4,037 (8,900)	4,477 (9,870)	EPF
4,060 (8,950)	4,500 (9,920)	LPF
—	*4,119 (9,080)	LPF
LOW-EARTH ORBIT FROM CCAS 185-km (100-nmi) Circular Orbit, i = 28.5°, 99.87% GCS		
8,641 (19,050)	10,718 (23,630)	EPF
8,686 (19,150)	10,759 (23,720)	LPF
LOW-EARTH ORBIT FROM VAFB 185-km (100-nmi) Circular Orbit, i = 90°, 99.87% GCS		
7,121 (15,700)	9,180 (20,240)	EPF
7,162 (15,790)	9,212 (20,310)	LPF
Note: PSWC: Payload Systems Weight Capability GCS: Guidance Commanded Shutdown EPF: Extended 0.91-m (36-in.) 4.2-m (14-ft) Payload Fairing LPF: Large 4.2-m (14-ft) Payload Fairing		
* Single Engine Centaur (IIIB LSCC)		

2.1 MISSION PERFORMANCE-LEVEL PHILOSOPHY

As Tables 2-1 and 2-2 illustrate, Lockheed Martin offers a range of performance levels for the Atlas family of launch vehicles. In addition, Lockheed Martin can meet performance requirements by customizing (or standardizing) mission and trajectory designs to meet specific mission desires. To meet evolving commercial satellite mission launch requirements, Lockheed Martin offers performance capability levels (as opposed to explicit hardware configurations) as part of its standard launch services package.

Besides offering the performance-level quotes that are associated with each vehicle, other performance requirements can be met in several ways. With spacecraft missions that may require less performance than a specific configuration may offer, additional mission constraints will be used that will use excess performance to benefit the launch services customer. Ascent trajectory designs can be shaped to include coverage of the Centaur second burn and spacecraft separation from the Ascension Island tracking station as opposed to prepositioning advanced range instrumentation aircraft (ARIA) or use of the Tracking and Data Relay Satellite System (TDRSS). The ascent profile can be standardized to reduce mission integration analyses and/or schedules. These options can allow a more cost-effective solution for cases where maximum vehicle performance is not required.

2.2 MISSION DESCRIPTIONS

Atlas is a reliable and versatile launch system, capable of delivering payloads to a wide range of low- and high-circular orbits, elliptical transfer orbits, and Earth-escape trajectories. Each Atlas launch vehicle, available with either a medium-, large-, or extended-length large payload fairing (MPF, LPF, or EPF), is dedicated to a single payload. The trajectory design for each mission can be specifically tailored to optimize the mission's critical performance parameter (e.g., maximum satellite orbit lifetime, maximum weight to transfer orbit) while satisfying satellite and launch vehicle constraints.

Atlas mission ascent profiles are developed using one or more Centaur upper-stage main engine burns. Each mission profile type is suited for a particular type of mission.

Direct Ascent Missions—With a one Centaur-burn mission design, the Centaur main engines are ignited just after Atlas/Centaur separation and the burn is continued until the Centaur and spacecraft are placed into the targeted orbit. Centaur/spacecraft separation occurs shortly after the burn is completed. Direct ascents are primarily used for low-Earth circular orbits and elliptic orbits with orbit geometries (i.e., arguments of perigee and inclinations) easily reached from the launch site. Orbits achievable with little or no launch vehicle yaw steering and those that can be optimally reached without coast phases between burns are prime candidates for the direct ascent mission design. Atlas/Centaur has flown 15 missions using the direct ascent design.

Parking Orbit Ascent Missions—The parking orbit ascent, used primarily for geosynchronous transfer missions, is the most widely used Atlas trajectory design. Performance capabilities are based on two Centaur burns injecting Centaur and the satellite into a transfer orbit selected to satisfy mission requirements. The first Centaur burn starts just after Atlas/Centaur separation and is used to inject the Centaur/spacecraft into a mission performance-optimal parking orbit. After a coast to the desired location for transfer orbit injection, the second Centaur main engine burn provides the impulse to place the satellite into the transfer and/or final orbit. If targeted to an elliptic transfer orbit, the satellite then uses its own propulsion system to achieve the final mission orbit. Missions requiring circular final orbits will use the second Centaur burn to circularize the satellite at the desired altitude and orbit inclination. More than 73 Atlas/Centaur missions have flown using the parking orbit ascent mission profile.

2.3 ATLAS ASCENT PROFILE

To familiarize users with Atlas and Centaur mission sequences, information is provided in the following paragraphs regarding direct and parking orbit ascent mission designs. Figures 2.3-1 and 2.3-2 show sequence-of-events data for a typical parking orbit ascent mission. Table 2.3-1 shows mission sequence data for each Atlas vehicle for a typical geosynchronous transfer mission. These data are representative; actual sequencing will vary to meet requirements of each mission. Atlas can be launched at any time of day to meet spacecraft mission requirements.

2.3.1 Booster Phase

2.3.1.1 Atlas IIA/IIAS—At liftoff, the booster ascent phase begins with ignition of the Rocketdyne MA-5A engine system, and for Atlas IIAS, the first pair of Thiokol Castor IVA solid rocket boosters (SRB).

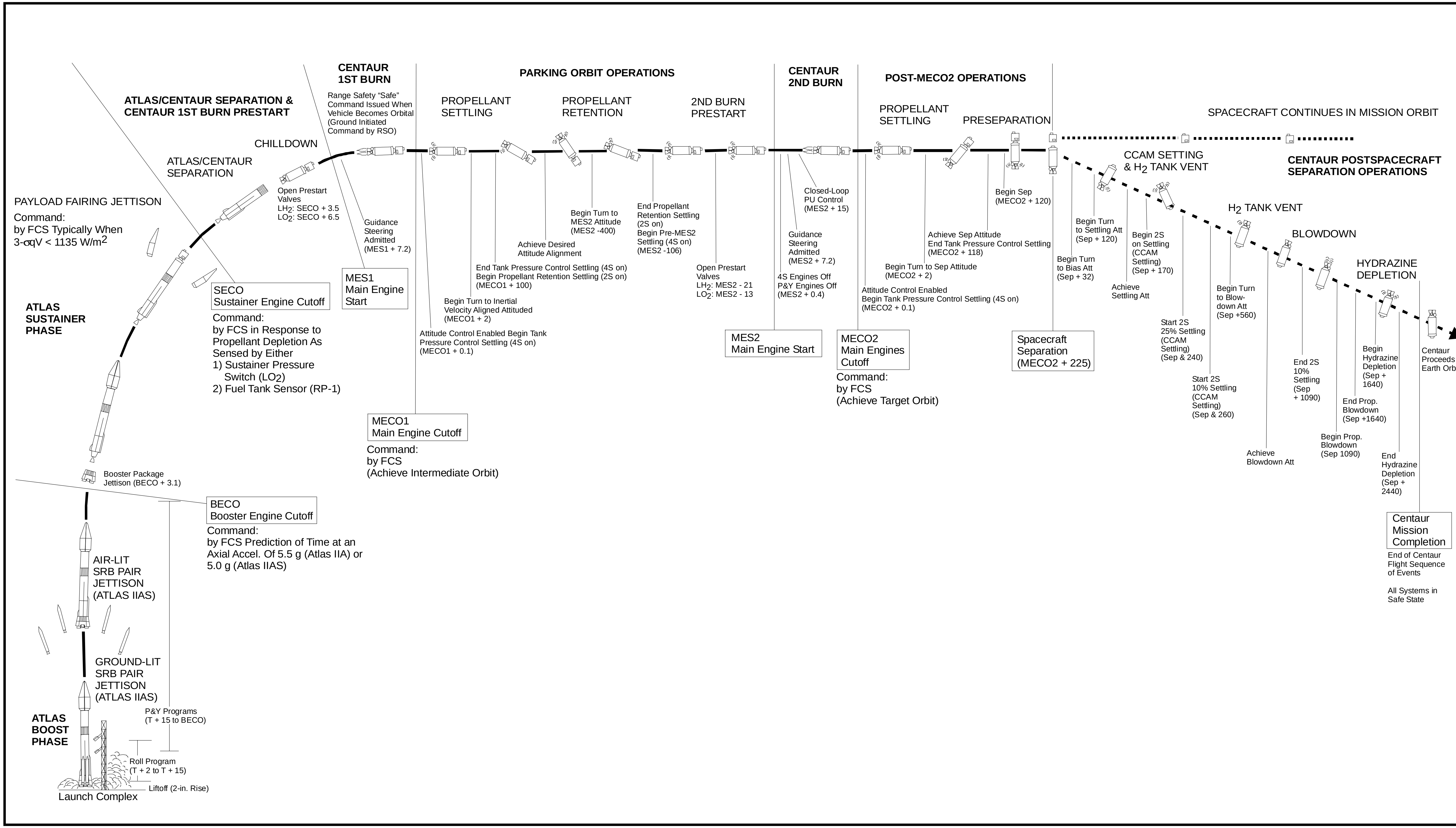


Figure 2.3-1 Typical Atlas IIA/IIAS Ascent Profile

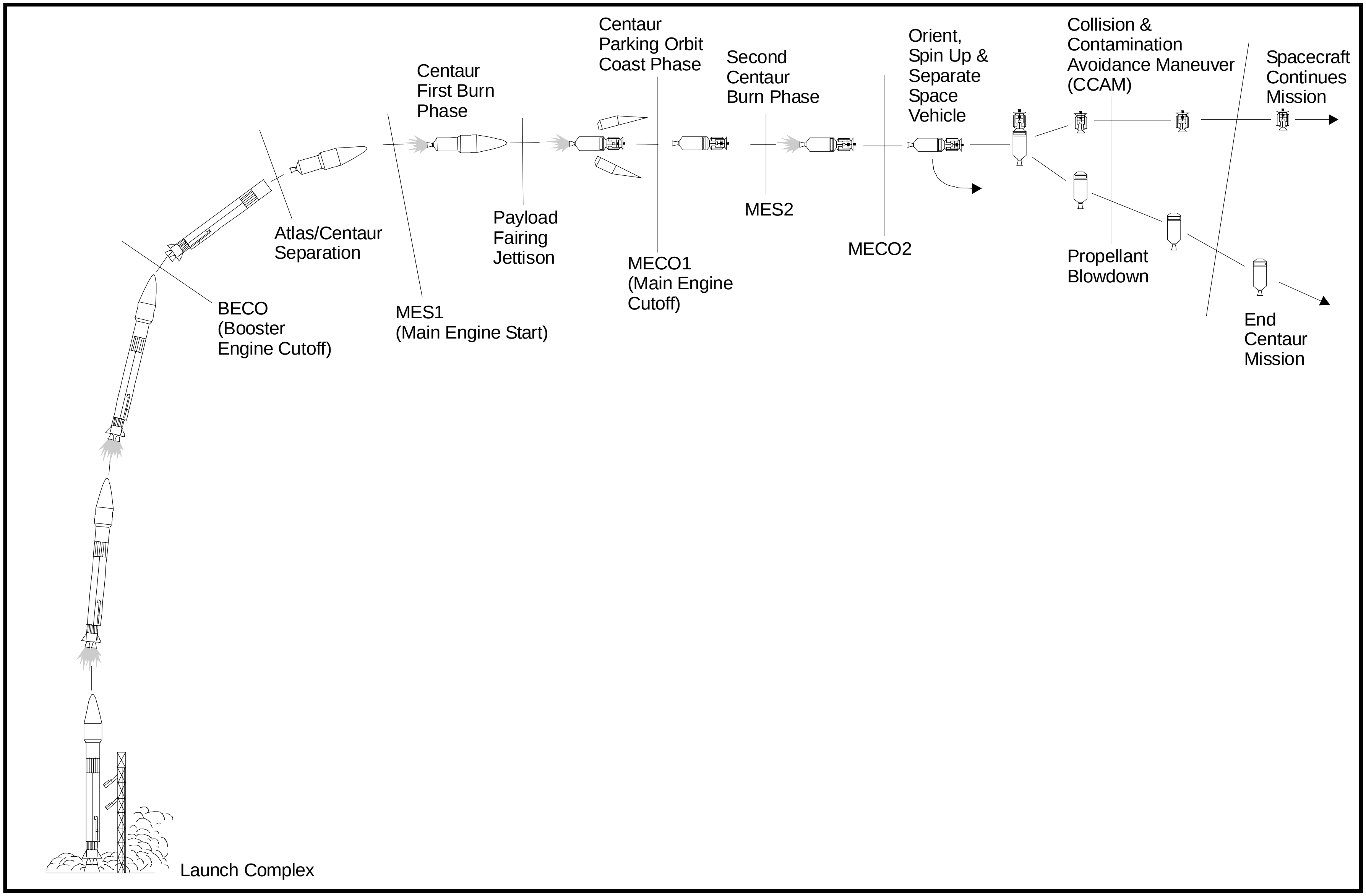


Figure 2.3-2 Typical Atlas IIIA/IIIB Summary Ascent Profile

Table 2.3-1 Typical Atlas IIA/IIAS GTO Mission Launch Vehicle Sequence Data

Event (Time in Seconds)	Atlas IIA	Atlas IIAS	Atlas IIIA	Atlas IIIB (DEC)	Atlas IIIB (SEC)
Guidance Go-Inertial	-11.0	-11.0	-11.0	-11.0	-11.0
Ground-Lit SRB Pair Ignition (Atlas IIAS)	—	-0.5	—	—	—
Liftoff	0.0	0.0	0.0	0.0	0.0
Ground-Lit Pair Burnout (IIAS)	—	54.7	—	—	—
Air-Lit SRB Pair Ignition (IIAS)	—	60.0	—	—	—
Ground-Lit SRB Pair Jettison (IIAS)	—	77.1	—	—	—
Air-Lit SRB Pair Burnout (IIAS)	—	115.3	—	—	—
Air-Lit SRB Pair Jettison (IIAS)	—	117.2	—	—	—
Atlas Booster Engine Cutoff (BECO)	165.1	163.3	184.0	182.0	181.9
Booster Package Jettison (BPJ)	168.2	166.4	—	—	—
Payload Fairing Jettison (PFJ) (IIA/IIAS)	227.8	214.5	—	—	—
Atlas Sustainer Engine Cutoff (SECO)	274.4	289.2	—	—	—
Atlas/Centaur Separation	278.4	293.3	195.0	193.0	192.9
Begin Extendible Nozzle Deployment (When Used)	279.9	294.8	196.5	194.5	194.4
End Extendible Nozzle Deployment	286.9	301.8	203.5	201.5	201.4
Centaur Main Engine Start (MES1)	294.9	309.8	212.9	202.9	202.8
Payload Fairing Jettison (PFJ) (IIIA, IIIB DEC/SEC)	—	—	220.9	225.5	210.8
Centaur Main Engine Cutoff (MECO1)	580.5	584.8	767.7	541.7	929.3
Start Turn to MES2 Attitude	1,196.3	1,180.8	1,101.2	982.0	1,085.1
Centaur Main Engine Start (MES2)	1,494.3	1,475.8	1,401.2	1,380.0	1,483.1
Centaur Main Engine Cutoff (MECO2)	1,581.0	1,571.9	1,585.3	1,494.8	1,663.9
Start Alignment to Separation Attitude	1,583.0	1,573.9	1,587.3	1,496.8	1,665.9
Begin Spinup	1,701.0	1,691.9	1,717.3	1,626.8	1,795.9
Separate Spacecraft (Sep)	1,808.0	1,798.9	1,810.3	1,719.8	1,888.9
Start Turn to CCAM Attitude	1,928.0	1,918.9	1,930.3	1,839.8	2,008.9
Centaur End of Mission	4,245.0	4,239.9	4,250.3	4,159.8	4,328.9

During the short vertical rise away from the pad, the vehicle rolls from the launch pad azimuth to the appropriate flight azimuth. At a vehicle-dependent altitude between 215 m (700 ft) and 305 m (1,000 ft), the vehicle begins pitching over into the prescribed ascent profile. At about 2,438 m (8,000 ft), the vehicle enters a nominal zero pitch and yaw angle of attack profile phase to minimize aerodynamic loads.

For Atlas IIAS, the ground-lit pair of SRBs burns out at about 54 seconds into flight. Ignition of the air-lit pair is governed by structural loading parameters. Ground-lit pair jettison occurs when range safety parameters are met. The air-lit pair is jettisoned shortly after burnout.

The booster phase steering profile is implemented through our launch-day ADDJUST wind-steering programs, which enhance launch availability by controlling wind-induced flight loads. A zero total angle-of-attack flight profile is used from 2,438 m (8,000 ft) to minimize aerodynamic loads while in the low atmosphere. For Atlas IIA, a small alpha-bias angle-of-attack steering technique is used after reaching 24,380 m (80,000 ft) until booster engine cutoff (BECO) to reduce gravity and steering losses. For Atlas IIAS, alpha-bias steering starts at 36,580 m (120,000 ft).

Booster staging occurs when the desired axial acceleration is attained. For Atlas IIA geosynchronous transfer orbit (GTO) missions, BECO typically occurs at an axial acceleration of 5.5 g. For Atlas IIAS, BECO occurs at 5.0 g. Earlier staging for reduced maximum axial acceleration and/or optimum mission design is easily accomplished with minor associated changes in performance.

After jettison of the booster engine and associated thrust structure, flight continues in the sustainer phase. Sustainer engine cutoff (SECO) occurs when all available sustainer propellants are consumed.

Selected atmospheric ascent data from liftoff through first upper-stage burn accompany performance data for each launch vehicle (Ref Figs. 2.7-1 and 2.8-1).

For typical Atlas IIA and IIAS missions, the payload fairing is jettisoned before SECO, when 3-sigma free molecular heat flux falls below 1,135 W/m^2 (360 Btu/ft^2-hr). For sensitive spacecraft, payload fairing jettison can be delayed later into the flight with some performance loss.

2.3.1.2 Atlas IIIA/IIIB—The Atlas IIIA/IIIB consists of a single-booster stage, unlike the Atlas IIA and Atlas IIAS, which are one and one-half stage boosters. On the ground, the booster engine system (a single engine with two thrust chambers) is ignited to provide thrust for liftoff. The flight begins with a short vertical rise, during which the vehicle rolls to the flight azimuth. At an altitude of 152 m (500 ft) and time from liftoff greater than 9.0 seconds, the vehicle begins its initial pitch-over phase. At approximately 1,829 m (6,000 ft), the vehicle enters into a nominal zero-pitch and zero-yaw angle-of-attack phase to minimize aerodynamic loads. Zero-pitch and zero-yaw angle-of-attack and linear-tangent-steering phase is implemented through the ADDJUST-designed steering programs, which enhance launch availability by controlling wind-induced flight loads. Atlas flight continues in this guidance-steered phase until propellant depletion BECO. The boost phase of flight ends with 11 seconds after BECO. Atlas/Centaur separation occurs with firing of the separation charge attached to the forward ring of the ISA. Twelve retrorockets are then fired to push the spent Atlas booster stage away from the Centaur upper stage.

The RD-180 throttle profile design is based on launch vehicle constraints, the throttle profile is optimized on a mission specific basis to obtain the nominal prelaunch trajectory. Liftoff throttle setting is limited to reduce the liftoff, acoustics on the vehicle, and spacecraft. At 120 feet acoustics have decreased sufficiently to allow the engines to throttle up. The transonic throttle setting will be optimized and uplinked to mitigate the wind-induced variation in dynamic pressure. Based on the desire to maintain a robust design, inflight Mach is computed by the flight software and used as criteria for transonic throttle-down and throttle-up. This allows thrust dispersion effects to be sensed for real-time adjustment of these throttle events to control the variation in dynamic pressure. The RD-180 booster engine operates at almost full throttle until vehicle acceleration reaches a certain limit. The throttle setting is then ramped down at a predetermined rate. Guidance actively controls the throttle setting to maintain a constant acceleration until 10 seconds before booster engine cut-off.

2.3.2 Centaur Phase

2.3.2.1 Atlas IIA/IIAS—Centaur main engine start (MES or MES1) occurs 16.5 seconds after the Atlas stage is jettisoned. For direct ascent missions, the Centaur main burn injects the spacecraft into the targeted orbit and then performs a series of preseparation maneuvers. With parking orbit ascent missions, the Centaur first burn (typically the longer of the two) injects the spacecraft into an elliptic performance-optimal parking orbit. After first-burn main engine cutoff (MECO1), the Centaur and spacecraft enter a coast period. During the coast period (about 14 minutes for a typical geosynchronous transfer mission), the Centaur normally aligns its longitudinal axis along the velocity vector. Because typical parking orbit coasts are of short duration, most spacecraft do not require special pointing or roll maneuvers. Should a spacecraft require attitude maneuvers during coast phases, Centaur can accommodate all roll axis alignment requirements and provide roll rates up to 1.5 ±0.3°/s in either direction during nonthrusting periods. Greater roll rates can be evaluated on a mission-peculiar basis. Before the second Centaur burn main engine start (MES2), the vehicle is aligned to the ignition attitude and the engine start sequence is initiated.

At a guidance-calculated start time, Centaur main engines are reignited and the vehicle is guided to the desired orbit. After reaching the target, main engines are shut down (MECO2), and Centaur begins its alignment to the spacecraft separation attitude. Centaur can align to any attitude for separation. Pre-separation spinups to 5.0 ±0.5 rpm about the roll axis can be accommodated. In addition, a pitch/yaw plane transverse spin mode can also be used.

After Centaur/spacecraft separation, Centaur conducts its collision and contamination avoidance maneuver (CCAM) to prevent recontact and minimize contamination of the spacecraft.

2.3.2.2 Atlas IIIA/IIIB—The baseline Atlas IIIA Centaur uses a single-engine Centaur configuration. The Atlas IIIB Centaur can use either a dual-engine (DEC) or single-engine (SEC) configuration. After BECO and Atlas/Centaur separation, the Centaur stage ignites its main engines (MES1). The payload fairing (PLF) is then jettisoned while the 3-sigma free-molecular-heat flux is below 1,135 W/m^2 (360 Btu/ft^2-hr), but not before 8 seconds after MES1. The Centaur first burn continues, which is the longer of the two Centaur firings, and injects the vehicle into an elliptic, performance-optimal parking orbit. During the first Centaur burn, the instantaneous impact point (IIP) trace progresses across the Atlantic Ocean and is over the equatorial region of the African continent for less than 1 second.

After Centaur first-burn main-engine cutoff (MECO1), the Centaur and spacecraft enter a coast period. At a guidance-calculated start time near the Equator, the Centaur main engine is reignited (MES2). The vehicle is then steered by guidance into GTO. The Centaur mission on Atlas IIIA/IIIB from this point on is very similar to a Centaur mission on Atlas IIA/IIAS.

2.3.3 Injection Accuracy and Separation Control

Atlas' combination of precision guidance hardware with flexible guidance software provides accurate payload injection conditions for a wide variety of missions. In response to changing mission requirements, minimal data are required to specify targeted end conditions to provide for rapid preflight retargeting. These functional capabilities have been demonstrated on many low-Earth orbit (LEO), GTO, lunar, and interplanetary missions.

Injection accuracies for a variety of GTO and LEO missions are displayed in Table 2.3.3-1 and are typical of 3-sigma accuracies following final upper-stage burn. On all missions to GTO (82 to date), Atlas has met all mission injection accuracy requirements.

Past lunar and interplanetary mission accuracy requirements and achievements are shown in Table 2.3.3-2. With the exception of the SOHO mission, these accuracies resulted from use of the older inertial measurement group (IMG) flight computer and would be further improved with use of our current inertial navigation unit (INU) guidance system. On some planetary missions, the guidance requirement included orientation of a spacecraft-imbedded solid rocket kick stage to achieve proper

Table 2.3.3-1 Typical Injection Accuracies at Spacecraft Separation

Orbit at Centaur Spacecraft Separation				± 3-sigma Errors						
Mission	Apogee, km (nmi)	Perigee, km (nmi)	Inclination, °	Semimajor Axis, km (nmi)	Apogee, km (nmi)	Perigee, km (nmi)	Inclination, °	Eccentricity	Argument of Perigee, °	RAAN, °
GTO	35,941 (19,407)	167 (90)	27.0	N/A	117 (63.2)	2.4 (1.3)	0.02	N/A	0.23	0.26
GTO	35,949 (19,411)	167 (90)	22.1	N/A	109 (58.9)	2.2 (1.2)	0.02	N/A	0.19	0.21
Super-synchronous	123,500 (66,685)	167 (90)	27.5	625 (337)	1,250 (675)	2.8 (1.5)	0.01	N/A	0.17	0.26
Intermediate Circular Orbit	10,350 (5,589)	10,350 (5,589)	45.0	42.7 (23.0)	N/A	N/A	0.07	0.002	N/A	0.08
Elliptical Transfer	10,350 (5,589)	167 (90)	45.0	N/A	40.0 (21.6)	2.40 (1.3)	0.07	N/A	0.07	0.08
LEO (Circular)	1,111 (600)	1,111 (600)	63.4	19.4 (10.5)	N/A	N/A	0.15	0.004	N/A	0.11
Legend: N/A = Not Applicable or Available										

final planetary intercept conditions. The major error source on these missions was the uncertainty of solid rocket kick stage impulse.

2.3.3.1 Attitude Orientation and Stabilization—During coast phases, the guidance, navigation, and control (GN&C) system can orient the spacecraft to any desired attitude. The guidance system can reference an attitude vector to a fixed inertial frame or a rotating orthogonal frame defined by the instantaneous position and velocity vector. The reaction control system (RCS) autopilot incorporates three-axis-stabilized attitude control for attitude hold and maneuvering. In addition to a precision attitude control mode for spacecraft preseparation stabilization, Centaur can provide a stabilized spin rate to the spacecraft while maintaining roll axis orientation before separation. The Centaur system can accommodate spin rates up to 5.0 ±0.5 rpm, subject to some limitation due to space vehicle mass property misalignments. A detailed analysis for each Centaur/spacecraft combination will determine the maximum achievable spin rate.

Table 2.3.3-2 Lunar and Interplanetary Mission Accuracy

Mission	Figure of Merit (FOM), m/s: Mission Requirement	Figure of Merit (FOM), m/s: Guidance System 1σ
Surveyor	< 50	7.0
Mariner Mars	< 13.5	3.5
Mariner Venus Mercury	< 13.5	2.4
Pioneer 10	< 39	39*
Pioneer 11	< 36	36*
Viking 1	< 15	3.6
Viking 2	< 15	3.5
Voyager 1	< 21	16.2*
Voyager 2	< 21	17.6*
Pioneer Venus 1	< 7.5	2.3
Pioneer Venus 2	< 12.0	3.2
SOHO	< 10.3	6.9

Note: * Major Error Source-Solid Kickstage Motor

The extensive capabilities of the GN&C system allow the upper stage to satisfy a variety of spacecraft orbital requirements, including thermal control maneuvers, sun-angle pointing constraints, and telemetry transmission maneuvers.

2.3.3.2 Separation Pointing Accuracies—Pointing accuracy just before spacecraft separation is a function of guidance system hardware, guidance software, and autopilot attitude hold capabilities. In the nonspinning precision pointing mode, the system can maintain attitude errors less than 0.7°, and attitude rates less than 0.2, 0.2, and 0.5°/s about the pitch, yaw, and roll axes, respectively (before separation) (Table 2.3.3.2-1). Although the attitude and rates of a nonspinning spacecraft after separation (after loss of contact between the Centaur and the spacecraft) are highly dependent on mass properties of the spacecraft, attitude typically can be maintained within 0.7° per axis, body axis rates are typically less than 0.6°/s in the pitch or yaw axis and 0.5°/s in the roll axis. The angular momentum of the spacecraft after separation is often a concern. Total spacecraft angular momentum is typically less than 15 N-m-s. Separation conditions for a particular spacecraft are assessed during the mission-peculiar separation analysis.

Table 2.3.3.2-1 Summary of Guidance and Control Capabilities

Coast Phase Attitude Control		
• Roll Axis Pointing, °, Half Angle	≤ 1.6	
• Passive Thermal Control Rate, °/s (Clockwise or Counterclockwise)	1.5	±0.3
Centaur Separation Parameters at Separation Command (with No Spin Requirement)		
• Roll Axis Pointing, °, Half Angle	≤ 0.7	
• Body Axis Rates, °/s		
– Pitch	±0.2	
– Yaw	±0.2	
– Roll	±0.5	
Spacecraft Separation Parameters at Separation Command (with Transverse Spin Requirement)		
• Transverse Rotation Rate, °/s	≤ 7.0	
Spacecraft Separation Parameters Following Separation (with Nonspinning or Slowspinning Requirement)		
• Pitch, Yaw & Roll Axis Pointing (per Axis), °	≤ 0.7	
• Body Axis Rates, °/s		
– Pitch	±0.6	
– Yaw	±0.6	
– Roll	±0.5	
Spacecraft Separation Parameters Following Separation (with Longitudinal Spin Requirement)		
• Nutation, °, Half Angle	≤ 5.0	
• Momentum Pointing, °, Half Angle	≤ 3.0	
• Spin Rate, °/s	≤ 30.0 ±3.0	

Centaur can also use a transverse spin separation mode in which an end-over-end "rotation" is initiated before separating the payload. A rotation rate of up to 7°/s is possible about the pitch or yaw axis for typical spacecraft.

For a mission requiring preseparation spinup, conditions just before spacecraft separation combine with any tipoff effects induced by the separation system and any spacecraft principal axis misalignments to produce postseparation momentum pointing and nutation errors. Here, nutation is defined as the angle between the actual space vehicle geometric spin axis and the spacecraft momentum vector. Although dependent on actual spacecraft mass properties (including uncertainties) and the spin rate, momentum pointing and maximum nutation errors following separation are typically less than 3.0 and 5.0°, respectively.

2.3.3.3 Separation Velocity—The relative velocity between the spacecraft and the Centaur is a function of the mass properties of the separated spacecraft and the separation mechanism. Our separation systems provide a minimum relative velocity of 0.27 m/s (0.9 ft/s) for spacecraft weights up to 4,536 kg (10,000 lb), and are designed to preclude recontact between the spacecraft and Centaur.

2.4 PERFORMANCE GROUNDRULES

Atlas performance groundrules for various missions with launch from CCAS in Florida or VAFB in California are described in this section.

2.4.1 Payload Systems Weight Definition

Performance capabilities quoted throughout this document are presented in terms of payload systems weight (PSW). Payload systems weight is defined as the total mass delivered to the target orbit, including the separated spacecraft, the spacecraft-to-launch vehicle adapter, and all other hardware required on the launch vehicle to support the payload (e.g., payload flight termination system, harnessing). Table 2.4.1-1 provides masses for our standard payload adapters (Ref Sect. 4.1.2 for payload adapter details). Data are also provided estimating performance effects of various mission-peculiar hardware requirements. As a note, performance effects shown are approximate. The launch vehicle trajectory, spacecraft mass, and mission target orbit can affect the performance contributions of each mission-peculiar item.

2.4.2 Payload Fairings

2.4.2.1 Atlas IIA/IIAS—Most Atlas IIA and Atlas IIAS performance shown in this document is based on use of the 4.2-m (14-ft) diameter LPF unless noted otherwise. Lockheed Martin also offers a 3.3-m (11-ft) diameter MPF. Higher performance is available for those payloads that fit in the MPF. Performance gains are vehicle configuration and trajectory design-dependent, but for GTO missions the

Table 2.4.1-1 Performance Effects of Spacecraft-Required Hardware

Performance Effect of Payload Adapter Masses, kg (lb)			
Type A 44 (97)	Type B 49 (107)	Type C 24 (54)	Type D 54 (119)
Type A1 51 (113)	Type B1 53 (116)	Type C1 24 (54)	Type E 104 (230)
Performance Effect of Other Spacecraft-Required Hardware, kg (lb)			
Standard Package 8 (18)	PLF Acoustic Panels 11 (25) LPF 7 (16) MPF	PLF Thermal Shield 4 (9) LPF	Environment Verification Pkg (Telepak, Instruments) 9 (20)
Standard package on Centaur consists of flight termination system (FTS) and airborne harness; payload fairing standard package consists of two standard access doors, reradiating antenna, and customer logo.			
Other Hardware			
• Centaur Hardware Affects Performance at 1-kg (2.2-lb) Mass to 1-kg (2.2-lb) Performance Ratio • Payload Fairing Hardware Affects Performance at ~ 9-kg (19.8-lb) Mass to 1-kg (2.2-lb) Performance Ratio			

gain is about 113 kg (250 lb).

For spacecraft that require greater volume than the standard LPF, a 1-m (3-ft) stretch to our large fairing has been developed. Performance for a IIA or IIAS will degrade approximately 45 kg (100 lb) with its use. Additional fairing information is in Section 4.1.

2.4.2.2 Atlas IIIA/IIIB—Atlas IIIA performance is based on use of the 4.2-m (14-ft) diameter, 1-m (3-ft) stretch EPF. Atlas IIIB performance is based on use of the 4.2-m (14-ft) diameter LPF.

2.4.3 Launch Vehicle Performance Confidence Levels

Atlas missions are targeted to meet the requirements of each user. Historically, Atlas and most U.S.-launched missions have been designed with a 3-sigma performance confidence level (99.87%). With the flexibility of Atlas/Centaur hardware and flight software, performance confidence levels can be set based on each mission's requirements. The minimum residual shutdown (MRS) performance option, discussed later in this section, takes full advantage of this concept. All data in this document, except CCAS launch elliptical transfer orbit performance data, are based on the 3-sigma confidence level (i.e., performance shown will be attained or exceeded with a 99.87% probability).

For elliptical transfer orbit data, Lockheed Martin has baselined a 99% confidence-level performance reserve. Because many of today's communications satellites can benefit from reduced launch vehicle confidence levels (and associated nominal performance increases), MRS data are also discussed. Lockheed Martin will respond to any desired performance confidence-level requirement needed by the user.

2.4.4 Centaur Short-Burn Capability

For LEO mission applications, Lockheed Martin has evaluated launch vehicle requirements for short-duration Centaur second burns. With missions requiring short-duration second burns (10-30 seconds), propellant residuals will be biased to ensure proper engine propellant inlet conditions at MES. Centaur main engine burns as short as 10 seconds are possible. All performance data shown using short-duration burns include performance effects of propellant-level control at MES2.

2.4.5 Centaur Long-Coast Capability

A Centaur extended-mission kit has been developed to support long-duration Centaur parking orbit coasts. Coasts of up to 2-hours plus in duration are manageable, constrained by helium pressurant and hydrazine reaction control system (RCS) propellant capacities. The long-coast kit consists of a larger vehicle battery, shielding on the Centaur aft bulkhead, additional helium capacity, and an additional hydrazine bottle. Performance estimates using long parking orbit coasts include the effect of an extended-mission kit. See Section 8.1.1 for additional details.

2.4.6 Heavy-Payload Lift Capability

Centaur equipment module and payload adapters have been optimized for geosynchronous transfer missions. To manage the larger payload masses (typically greater than 4,080 kg [9,000 lb]), Atlas is capable of delivering to LEO two heavy-payload interfaces have been identified. Figure 2.4.6-1 illustrates the interfaces. The strengthened equipment module is in production. In both cases the user must account for the mass of a spacecraft-to-launch vehicle adapter. In addition, the stated performance penalty must be accounted for if the

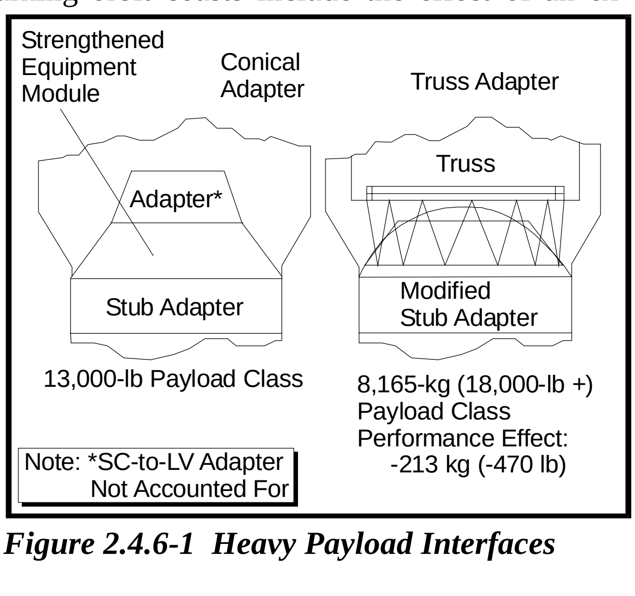


Figure 2.4.6-1 Heavy Payload Interfaces

truss adapter is exercised. See Section 8.2.1 for additional details.

2.5 GEOSYNCHRONOUS LAUNCH MISSION TRAJECTORY AND PERFORMANCE OPTIONS

Through Centaur's flexible flight software, a number of trajectory designs are possible. Depending on mission requirements, total satellite mass, dry mass-to-propellant mass ratio, and type of satellite propulsion (liquid or solid) system, one of the following trajectory design options will prove optimal:

1) Geosynchronous transfer (and reduced inclination transfers),
2) Supersynchronous transfer,
3) Subsynchronous transfer and perigee velocity augmentation.

2.5.1 Geosynchronous Transfer

The GTO mission is the standard mission design for communications satellite launches. Figure 2.5.1-1 illustrates the orbital mission profile involved. The transfer orbit inclination achieved depends on launch vehicle capability, satellite launch mass, and performance characteristics of both systems. Based on the performance of the Atlas family and enhanced capabilities of today's liquid apogee engine (LAE) subsystems, Lockheed Martin is finding that a 27° inclination is optimal for maximizing satellite beginning-of-life mass given an optimally sized satellite propulsion system. The 300-plus-second specific impulses of current LAEs have resulted in a shift in optimum inclination from 26.5 to 27°. With satellites weighing less than the GTO capability of the launch vehicle, excess performance can be used to further reduce inclination or raise perigee.

Although the GTO design is intuitively the standard launch option, single-payload manifesting allows the option of alternate designs that can extend geostationary satellite lifetimes. Supersynchronous transfers, subsynchronous transfers, and other mission enhancement options can enhance lifetime with satellites that use common sources of liquid propellant for orbit insertion and onorbit stationkeeping.

2.5.2 Supersynchronous Transfer

The supersynchronous trajectory design offers an increase in beginning-of-life propellants by minimizing the delta-velocity required of the satellite for orbit insertion. The Atlas injects the satellite into an intermediate transfer orbit with an apogee well above geosynchronous altitude. If the apogee al-

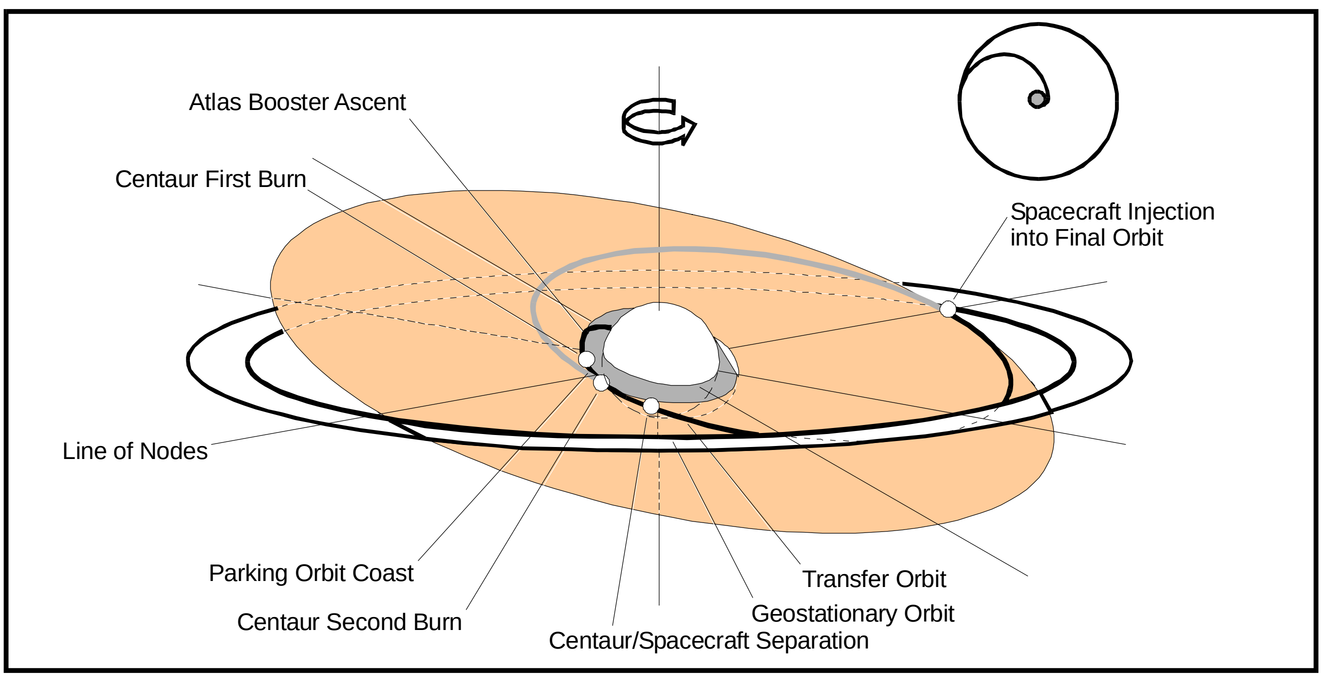


Figure 2.5.1-1 The Geosynchronous Transfer Orbit Mission Trajectory Profile

titude capability exceeds the satellite maximum allowable altitude, excess launch vehicle performance can be used to lower orbit inclination. At supersynchronous altitudes, the decreased inertial velocity allows the satellite to make orbit plane changes more efficiently. The satellite makes the plane change and raises perigee to geosynchronous altitude in one or more apogee burns. It then coasts to perigee and circularizes into final geostationary orbit (GSO). The total delta-velocity in this supersynchronous transfer design is less than would be required to inject from an equivalent performance reduced inclination geosynchronous transfer, resulting in more satellite propellants available for onorbit operations. Figure 2.5.2-1 illustrates the supersynchronous trajectory mission profile. Table 2.5.2-1 quantifies potential mission gains with the supersynchronous mission for a 1,850-kg (4,078-lb) satellite. Supersynchronous transfer trajectories have been flown on 18 missions starting with the Atlas II/EUTELSAT II (AC-102) mission launched in December 1991.

2.5.3 Subsynchronous Transfer and Perigee Velocity Augmentation

The perigee velocity augmentation (PVA) trajectory design, compared with the standard GTO design, can provide increased propellant mass at beginning-of-life on GSO. This is beneficial when propellant tank capacity is large with respect to the dry mass. Atlas delivers the satellite to a subsynchronous intermediate transfer orbit (apogee less than geosynchronous) with an inclination of approximately 27° because the satellite mass exceeds GTO launch capability. The separated satellite coasts to subsequent transfer orbit perigee(s), where the satellite supplies the required delta-velocity for insertion into geosynchronous transfer. At apogee, using one or more burns, the satellite lowers inclination and circularizes into GSO. As illustrated in Table 2.5.3-1, mass at beginning-of-life is enhanced. The orbit profile is shown in Figure 2.5.3-1. Several Atlas flights, including six UHF/EHF missions, have successfully used the subsynchronous transfer option.

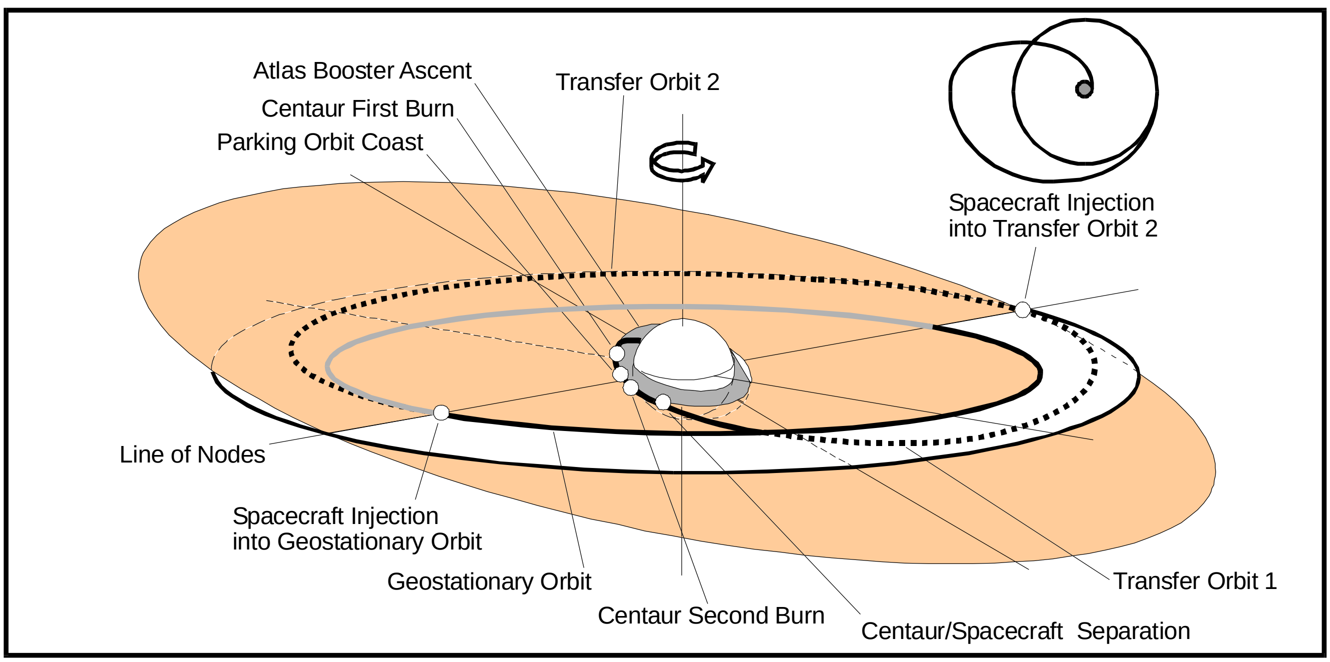


Figure 2.5.2-1 The Synchronous Transfer Orbit Mission Trajectory Profile

Table 2.5.2-1 Mission Benefits of Supersynchronous Transfer

Parameter	GTO	Super-synch
SC Mass, kg	1,850	1,850
Transfer Orbit Parameters		
• Perigee Altitude, km	391	183
• Apogee Altitude, km	35,786	50,000
• Orbit Inclination, °	19.6	21.2
• Argument of Perigee,°	180	180
Final Orbit	GSO	GSO
SC ΔV Required for GSO Insertion, m/s	1,643	1,607
Estimated Mission Lifetime, years	6.7	7.2
SC Gains Propellant for Additional 0.5 year of Lifetime		

Table 2.5.3-1 Mission Benefits of Subsynchronous Transfer

Parameter	GTO	Sub-synch
SC Mass, kg	3,654	3,654
SC Offload To Meet GTO Launch Capability, kg	-160	0
Transfer Orbit Parameters		
• Perigee Altitude, km	167	167
• Apogee Altitude, km	35,786	30,000
• Orbit Inclination, °	27.0	27.0
• Argument of Perigee, °	180	180
Final Orbit	GSO	GSO
SC ΔV Required for GSO Insertion, m/s	1,806	1,913
SC Mass at Beginning of Life, kg	1,916	1,933
Estimated Mission Lifetime, years	12.1	12.6
SC Gains Propellant for Additional 0.5 year of Lifetime		

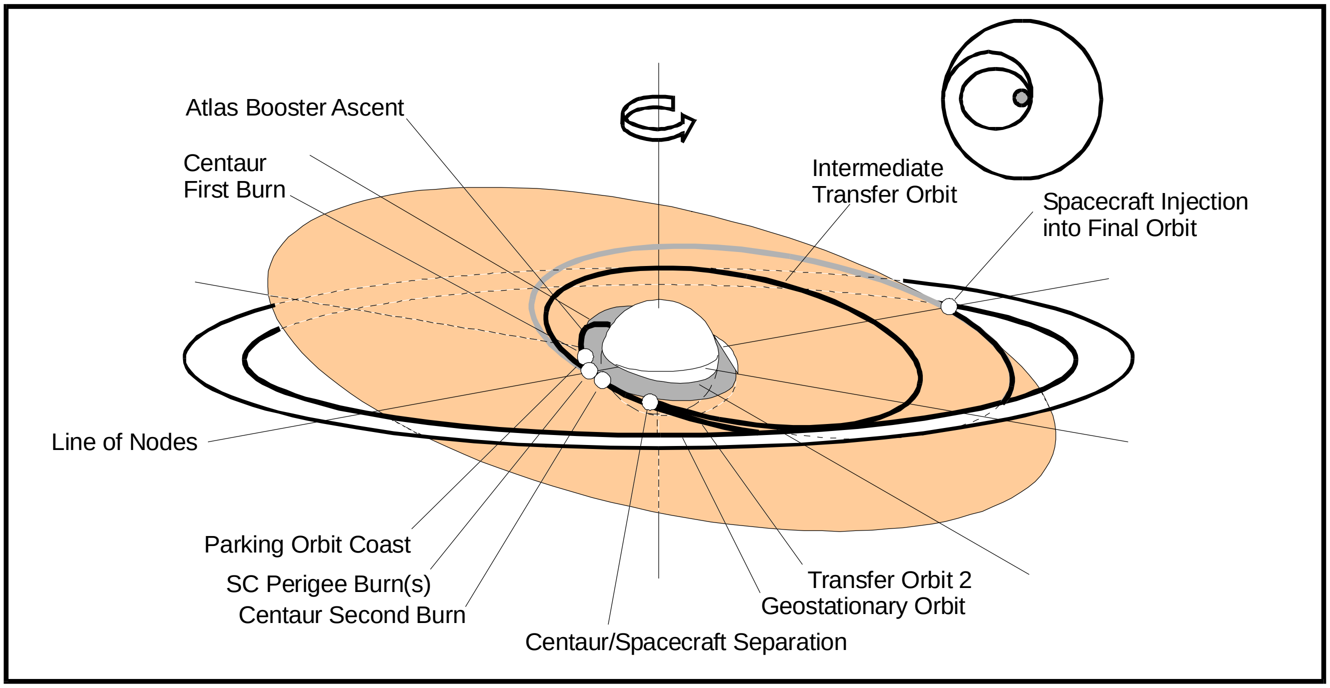


Figure 2.5.3-1 The Subsynchronous Transfer Orbit Mission Trajectory Profile

2.6 MISSION OPTIMIZATION AND ENHANCEMENT

Atlas trajectory designs are developed using an integrated trajectory simulation executive and a state-of-the-art optimization algorithm (sequential quadratic programming [SQP]). This optimization capability shapes the trajectory profile from liftoff through spacecraft injection into targeted orbit.

The SQP technique uses up to 50 independent design variables chosen to maximize a performance index and satisfy specified constraints. Typical control variables include boost-phase initial pitch and roll (launch azimuth) maneuvers, Atlas sustainer steering, and Centaur steering for all burns. In addition, spacecraft pitch and yaw attitudes and ignition times can be included as part of the total optimization process. The optimization program is formulated with up to 40 equality and inequality constraints on variables, such as dynamic pressure, tracker elevation angle, and range safety.

With Lockheed Martin's experience with launch of interplanetary and scientific spacecraft, an additional trajectory analysis tool is available to assist in the mission design/mission optimization process.

The N-BODY trajectory simulation program is used, with our optimization capability, in developing launch vehicle missions requiring precision inertial targeting in which perturbation effects of other celestial bodies are required to be considered.

These trajectory analysis tools, along with our extensive guidance and targeting capabilities, enable Atlas mission optimization to include spacecraft characteristics and programmatic goals. The most widely used mission enhancement options are:

1) Inflight retargeting (IFR),
2) Minimum residual shutdown (MRS),
3) Explicit right ascension of ascending node (RAAN) control.

2.6.1 Inflight Retargeting (IFR)

The software capability of the Centaur upper stage makes it possible to evaluate Atlas performance in flight and then target for an optimal injection condition that is a function of the actual performance of the booster stage. Centaur can be retargeted to a variable transfer orbit inclination, apogee, perigee, argument of perigee, or any combination of the above. IFR can provide a dual performance benefit. First, the nominal launch vehicle flight performance reserve (FPR) is reduced when the FPR contribution due to Atlas dispersions is eliminated. Second, whether Atlas performance is high, nominal, or low, the retargeting logic is calibrated to devote all remaining propellant margin to benefit the mission. With IFR, any desired level of confidence of a guidance shutdown can be implemented. IFR has been successfully flown on 10 missions starting with the Atlas II/EUTELSAT II mission.

2.6.2 Minimum Residual Shutdown

Centaur propellants may be burned to minimum residuals for a significant increase in nominal performance capability. When burning to minimum residuals, FPR propellants are eliminated to nominally gain additional delta-velocity from Centaur.

It is practical for Centaur to burn all its propellants when the satellite has a liquid propulsion system that is capable of correcting for variations in launch vehicle performance. This option is particularly attractive and appropriate when the trajectory design includes a supersynchronous or subsynchronous (PVA) transfer orbit. MRS is not an option for satellites using solid propellant (fixed impulse) OISs, because FPR propellants are required to ensure that the Centaur injection conditions will match the capability of the fixed impulse stage.

When Centaur burns all propellants to minimum residuals, the liquid propellant satellite corrects for the effects of launch vehicle dispersions. These dispersions primarily affect apogee altitude. Variations in other transfer orbit parameters are minor. The performance variation associated with MRS can also be quantified as an error in injection velocity that can be approximated as a dispersion in transfer orbit perigee velocity. Tables 2.6.2-1 and 2.6.2-2 document MRS perigee velocity injection variations for the Atlas IIA/IIAS and Atlas IIIA/IIIB, respectively. MRS has been successfully executed for 24 missions and has become the typical operations mode for GTO-type missions.

Table 2.6.2-1 Atlas IIA, IIAS Orbit Injection Performance Variations with MRS

Perigee Velocity Dispersions	2.33 sigma	3 sigma
Atlas IIA	68.9 m/s (226 ft/s)	88.4 m/s (290 ft/s)
Atlas IIAS	66.5 m/s (218 ft/s)	85.3 m/s (280 ft/s)

Table 2.6.2-2 Atlas IIIA and IIIB 3-sigma Performance Variations with MRS

Perigee Velocity Dispersions	2.33 sigma	3 sigma
Atlas IIIA	59.4 m/s (195 ft/s)	76.5 m/s (251 ft/s)
Atlas IIIB	60.4 m/s (198 ft/s)	77.7 m/s (255 ft/s)

2.6.3 Right Ascension of Ascending Node (RAAN) Control

Some satellite mission objectives may require launch-on-time placement into transfer and/or final orbit. For Earth orbital missions, this requirement typically manifests itself as a RAAN target value or range of values. Centaur's heritage of meeting the inertial orbit placement requirements associated with planetary missions makes it uniquely capable of targeting to an orbit RAAN (or range of RAANs dictated by actual launch time in a launch window) in addition to the typical or target parameters. With GTO missions, some satellite mission operational lifetimes can be enhanced by controlling RAAN of the targeted transfer orbit. A satellite intended to operate in a non-0° geosynchronous final orbit can benefit with proper RAAN placement. A drift toward a 0° inclination orbit can help reduce the typical north-south stationkeeping budget of the satellite, thereby increasing the amount of time the satellite can remain in an operational orbit.

Control of the node, and therefore RAAN, is obtained by varying the argument of perigee of the transfer orbit and the satellite burn location with respect to the Equator. The difference between the inclination of the transfer orbit and final orbits and the latitude of the satellite burn determines the amount of nodal shift between the transfer orbit and the mission orbit. Control of this shift is used to compensate for off-nominal launch times, keeping the inertial node of the final orbit fixed throughout a long-launch window. Centaur software can be programmed to control the argument of perigee (satellite burn location) as a function of time into the launch window to obtain the desired final orbit inertial node. Atlas IIA/INMARSAT-3 missions used this technique to gain an additional 3 years of mission lifetime.

2.7 MISSION PERFORMANCE DATA

For Atlas IIA, relevant atmospheric ascent parameters are shown in Figure 2.7-1. Detailed performance data are provided in Figures 2.7-2a through 2.7-11b. Data are shown for several types of launch missions and are based on our estimate of the vehicle's lift capability. Performance groundrules are shown on each curve and additional information is provided in the following paragraphs.

2.7.1 Elliptical Transfer Capability

The optimum trajectory profile for achieving elliptical transfer orbits is the parking orbit ascent. Atlas performance capability for 27° inclined orbits is shown in Figures 2.7-2 and 2.7-3. The 27°/s inclined orbit missions are launched at flight azimuths that have been approved and flown for many missions. The 27° inclination is near-optimal for geostationary transfer missions. The Centaur second burn is executed near the first descending node of the parking orbit (near the Equator). Performance is shown for both the 99% confidence-level GCS and MRS. Tabular performance data are provided in Tables 2.7.1-1 and 2.7.1-2.

For some missions, an ascending node injection into the transfer orbit may offer advantages. The performance degradation and mission constraints associated with this mission type can be analyzed on a mission-peculiar basis.

2.7.2 Reduced Inclination Elliptical Transfer Capability

The inclination effect on payload systems weight capability for a geosynchronous transfer orbit design for inclinations between 30° and 18° is shown in Figures 2.7-4 and 2.7-5. Performance degrades as inclination drops below 28.5° in part because the launch vehicle IIP trace is constrained by Range Safety requirements to remain a minimum range approved distance off the Ivory Coast of Africa. This requirement dictates that yaw steering be implemented in the ascent phase to meet Range Safety requirements while attempting to lower park orbit inclination toward the desired transfer orbit target value. The remainder of the inclination is completed with yaw steering in the Centaur second burn. Data are shown at the 99% confidence-level GCS and MRS for transfer orbit apogee altitudes of 35,788 km (19,324 nmi) and 100,000 km (53,996 nmi).

2.7.3 Earth-Escape Performance Capability

Earth-escape mission performance is shown in Figures 2.7-6. Centaur's heritage as a high-energy upper stage makes it ideal for launching spacecraft into Earth-escape trajectories. Performance data shown use the parking orbit ascent design and a near-planar ascent to an orbit that contains the outgoing asymptote of the escape hyperbola with a 700-s coast time between the upper-stage burns. The reference performance curves were developed using a 180° argument of perigee target and coast times reflect this constraint. The actual coast time necessary to achieve the desired departure asymptote will be determined by specific mission requirements. Our quoted performance assumes that 3-sigma (99.87% confidence level) flight performance reserves are held.

Additional performance data are shown for an optional vehicle configuration that uses the parking orbit ascent design with a customer-supplied third stage, a near-optimum size solid propellant orbit insertion stage (OIS) based on the Thiokol STAR 48B (TEM-711-18) motor. This vehicle configuration is advantageous for missions that require a very high-energy Earth departure, cases in which vehicle staging effects make it more efficient for a third stage to provide an additional energy increment. The reference performance mission was targeted similarly to the no-OIS case with the STAR 48B burn occurring just after Centaur/STAR 48B separation. This vehicle configuration is similar to the STAR 37-based Atlas configuration flown for the AC-27 and AC-30 Pioneer 10 (F) and Pioneer 11 (G) missions.

2.7.4 Low-Earth Orbit Capability

Atlas can launch payloads into a wide range of LEOs from Cape Canaveral using direct ascent or parking orbit ascent mission profiles. LEO capabilities typically require heavy-payload modifications.

Direct Ascent to Circular Orbit—Figure 2.7-7 shows circular orbit payload systems weight capability to LEO using the one Centaur burn mission profile. The maximum capability is available with planar ascent to a 28.5° inclination orbit. As shown, inclinations from 28.5 up to 55° are possible with the direct ascent. Given known Range Safety constraints, direct ascent performance to inclinations greater than 55° are not possible due to land overflight constraints up the Eastern seaboard of the United States and Canada. Direct ascent performance to reduced inclination orbits (down to ~22°) is also possible, but at the expense of substantial performance due to Range Safety overflight constraints over the Ivory Coast of Africa.

Direct Ascent to Elliptical Orbit—Figure 2.7-7 also shows elliptical orbit performance capability using the direct ascent with perigee altitude at 185 km (100 nmi). Similar Range Safety and orbital mechanics constraints limit inclinations available with a Florida launch.

Parking Orbit Ascent to Circular Orbit—Payload delivery to low-altitude circular orbit can be accomplished by two or more upper-stage burns. The first Centaur burn is used to inject the Centaur and payload into an elliptic parking orbit. A park orbit perigee altitude of 148 km (80 nmi) is assumed for our reference cases. Expected parking orbit coast durations will require use of the Centaur extended mission kit. The second Centaur burn will circularize the spacecraft into the desired orbit altitude.

Circular orbit performance capabilities for altitudes between 400 km (216 nmi) and 2,000 km (~1,000 nmi) are shown for each vehicle in Figure 2.7-7. Data are shown for 28.5, 45, 55, 60 and 63.4° inclinations. With high-inclination orbits (inclinations greater than 55°), Range Safety requirements require that Atlas meet instantaneous impact constraints up the Eastern seaboard of the United States and Canada. Additional inclination is added in the later stages of the Centaur first burn and with the second Centaur burn. As desired orbit inclination increases, performance degradations become more pronounced. High-inclination orbit performance capabilities are more optimally achieved with launch from VAFB (Sect. 2.7.6).

2.7.5 Intermediate Circular Orbits

Performance data are shown in Figure 2.7-8 for altitudes between 5,000 km (2,700 nmi) and 20,000 km (11,000 nmi). Similar groundrules apply to intermediate circular orbit data as to LEO circular orbit data except with respect to heavy payload requirements. The lower performance capabilities associated with the higher energy circular missions should allow use of standard payload interfaces.

2.7.6 VAFB Elliptical Orbit Transfer Capability

Atlas can launch payloads into high-inclination elliptical transfer from our West Coast launch site at VAFB. Transfer orbits at 63.4° are of interest because the rotation rate of the line of apsides is zero. These missions use a parking orbit ascent mission profile with the second burn executed near the first antinode (argument of perigee equaling 270°). Figure 2.7-9 shows performance capability for elliptical transfer orbits with apogee altitudes between 5,000 km (2,700 nmi) and 50,000 km (27,000 nmi). Performance is shown for both 99.87% confidence-level GCS and MRS.

2.7.7 VAFB LEO Performance

Figures 2.7-10 provides low-Earth orbit performance data for launches from our West Coast launch site. Orbits with inclinations ranging from 60° through Sun-synchronous are shown. Data are provided for mission types similar in scope to those discussed in Section 2.7.4.

2.7.8 VAFB High-Inclination, High-Eccentricity Orbit Capability

From VAFB, Atlas can insert payloads into orbits with 12-hour or 24-hour periods at an inclination of 63.4°. With the rotation rate of the line of apsides being zero, these orbits repeat their ground trace. These missions use a park orbit ascent mission similar to that described in Section 2.7.6. Figure 2.7-11 shows performance capability for these high-inclination, high-eccentricity orbits with perigee altitudes between 250 km (135 nmi) and 2,500 km (1,350 nmi). Performance capability is shown for 99.87% confidence-level GCS.

2.8 ATLAS IIAS PERFORMANCE DATA

Figures 2.8-1 through 2.8-11b and Tables 2.8-1 and 2.8-2 illustrate the performance of the Atlas IIAS vehicle to orbits described in Section 2.7.

2.9 ATLAS IIIA PERFORMANCE DATA

Figures 2.9-1 through 2.9-11b and Tables 2.9-1 and 2.9-2 illustrate the performance of the Atlas IIIA vehicle to orbits described in Section 2.7.

2.10 ATLAS IIIB (DEC) PERFORMANCE DATA

Figures 2.10-1 through 2.10-11b and Tables 2.10-1 and 2.10-2 illustrate the performance of the Atlas IIIB (DEC) vehicle to orbits described in Section 2.7.

2.11 ATLAS IIIB (SEC) PERFORMANCE DATA

Figures 2.11-1 through 2.11-3b and Tables 2.11-1 and 2.11-2 illustrate the performance of the Atlas IIIB (SEC) vehicle described in Sections 2.7.1 and 2.7.2 only.

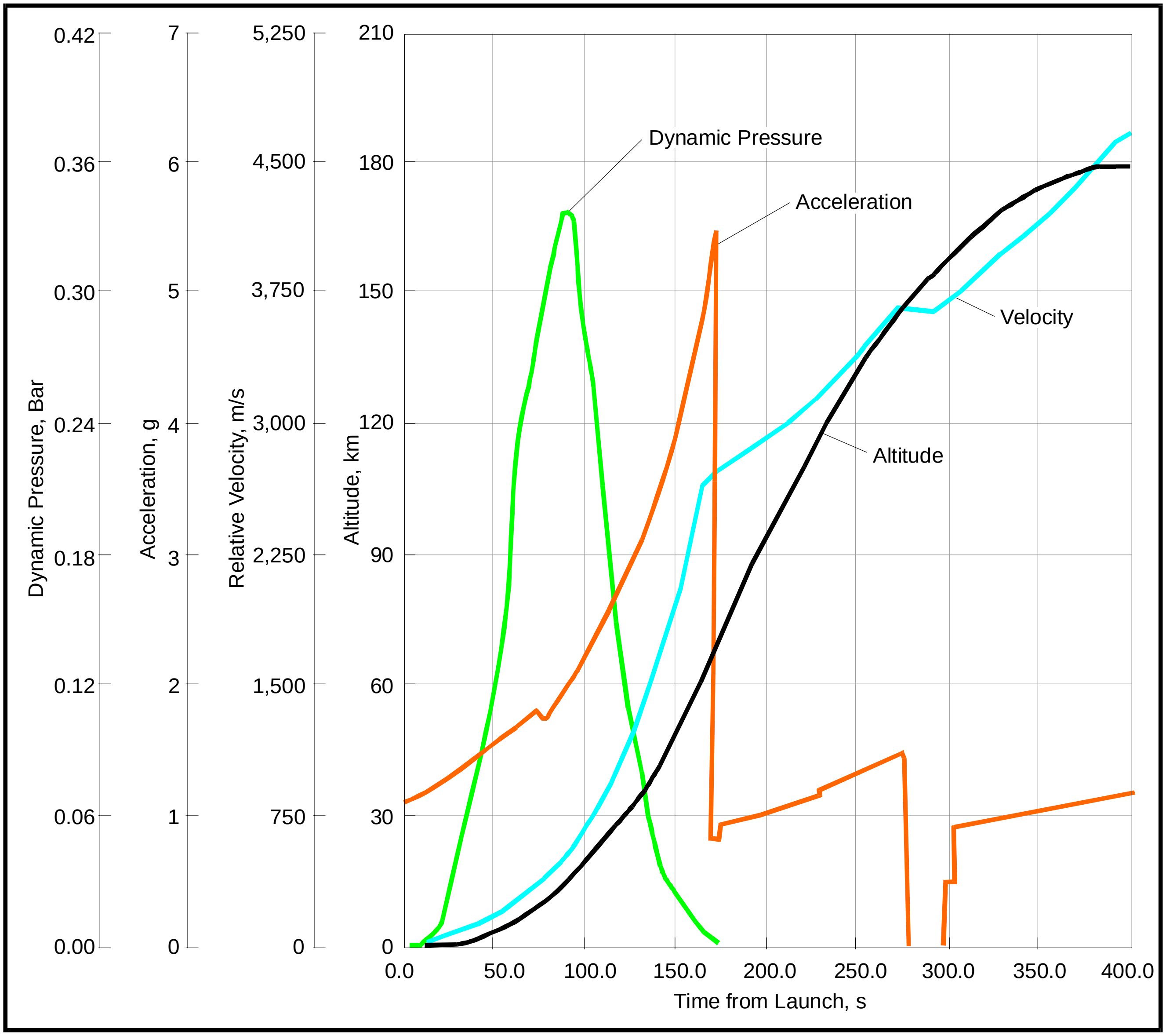


Figure 2.7-1 Atlas IIA Nominal Ascent Data

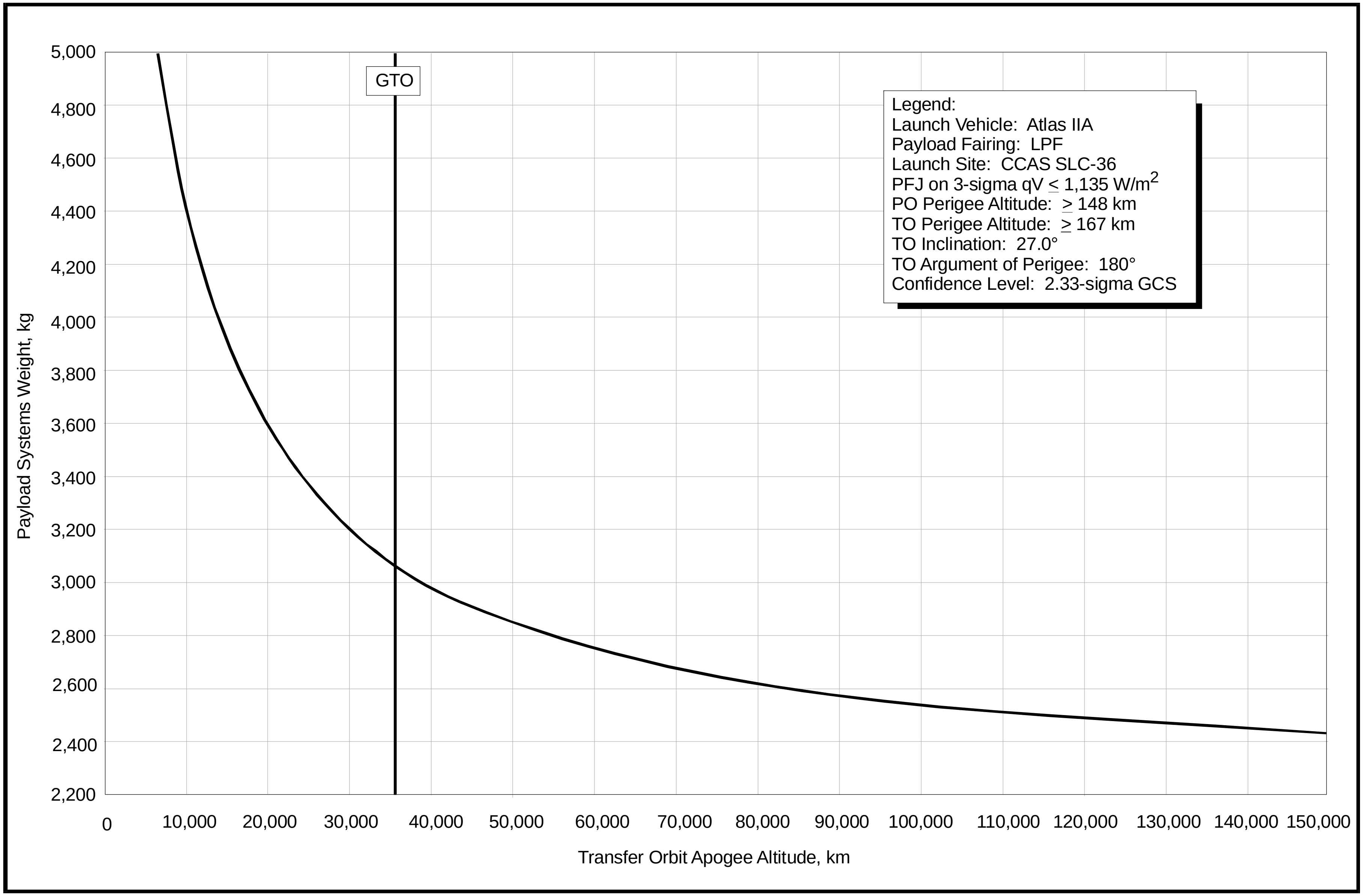


Figure 2.7-2a Atlas IIA CCAS Performance to Elliptical Transfer Orbit (GCS-Metric)

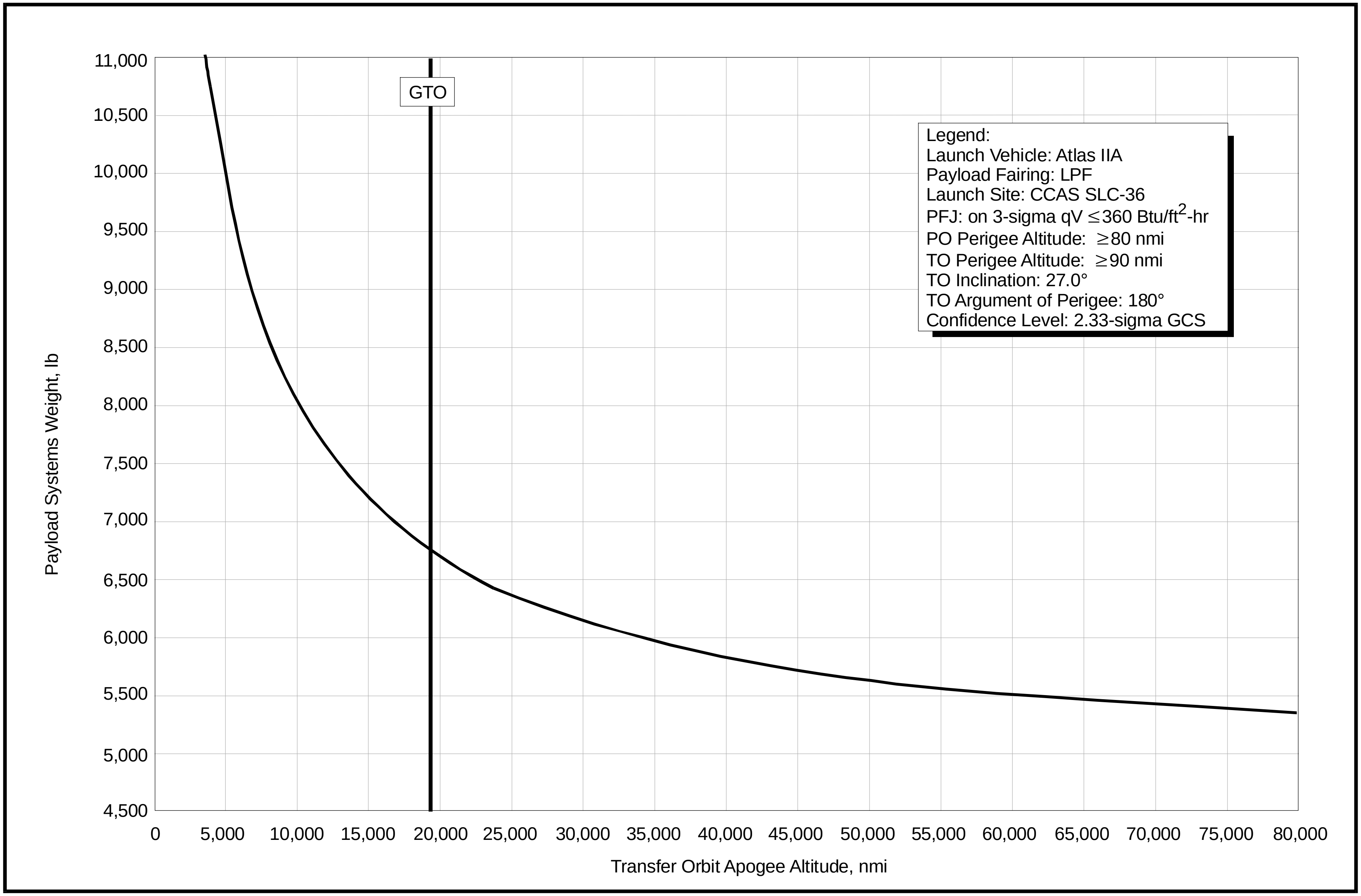


Figure 2.7-2b Atlas IIA CCAS Performance to Elliptical Transfer Orbit (GCS-English)

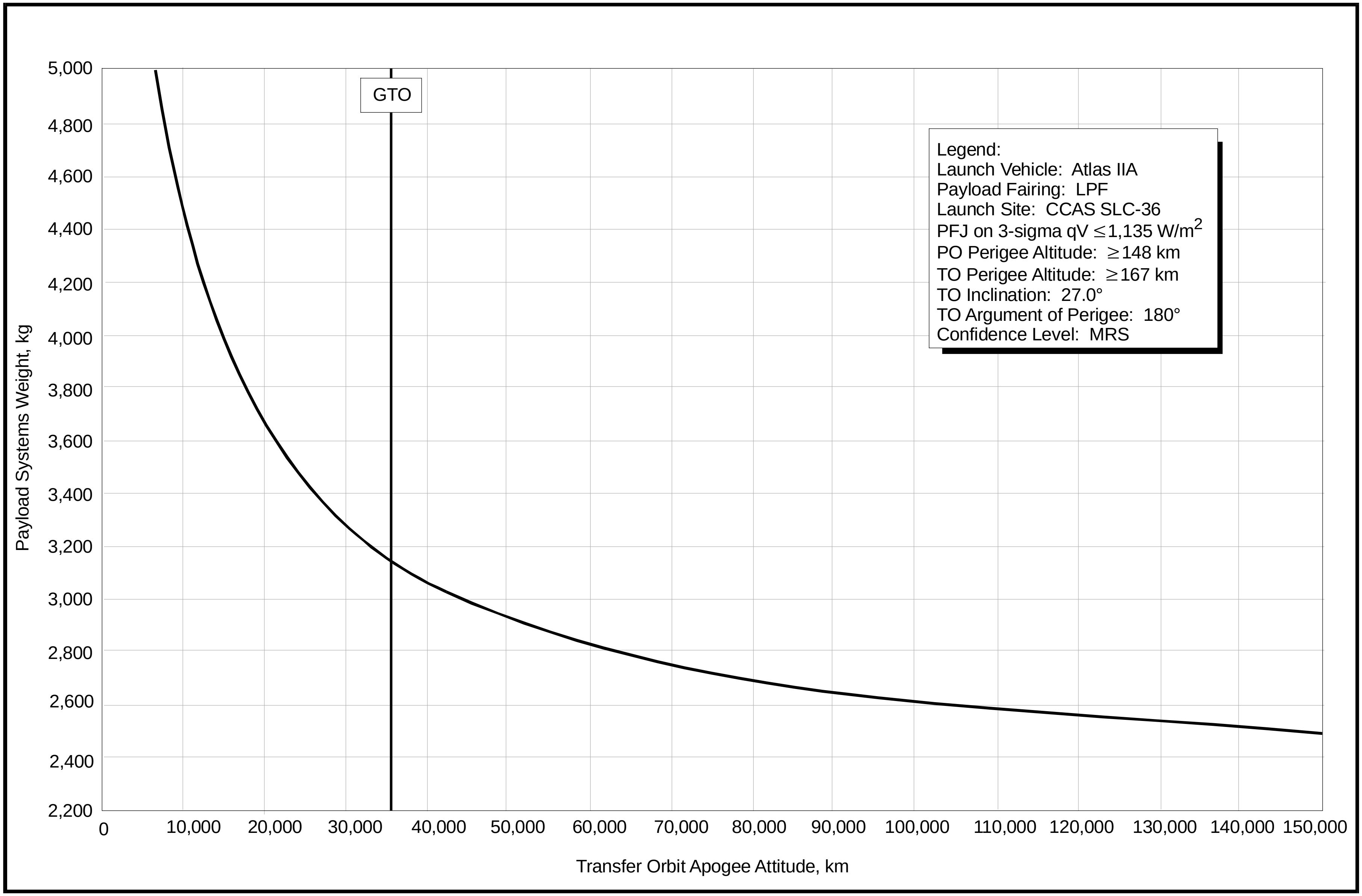


Figure 2.7-3a Atlas IIA CCAS Performance to Elliptical Transfer Orbit (MRS-Metric)

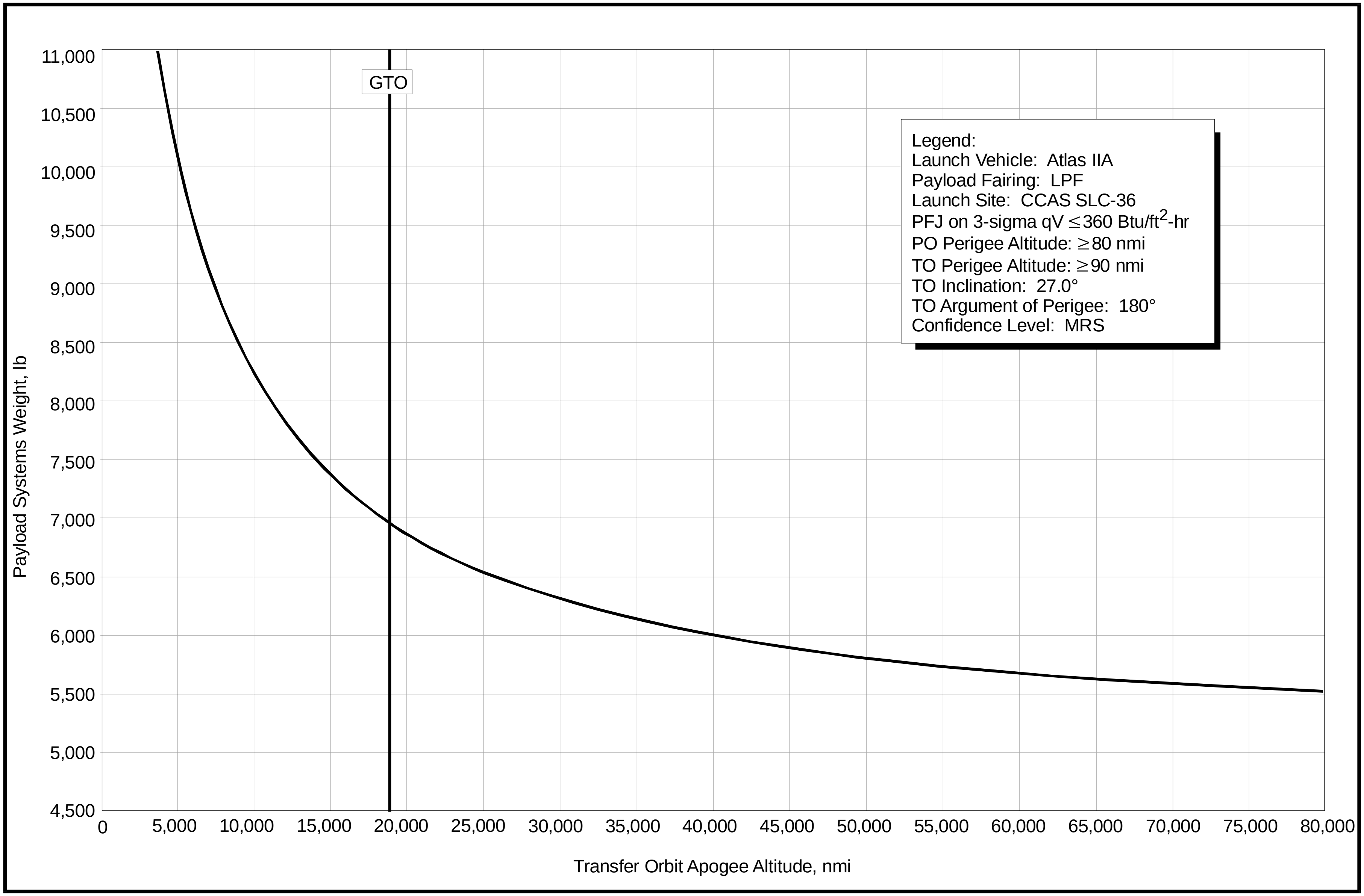


Figure 2.7-3b Atlas IIA CCAS Performance to Elliptical Transfer Orbit (MRS-English)

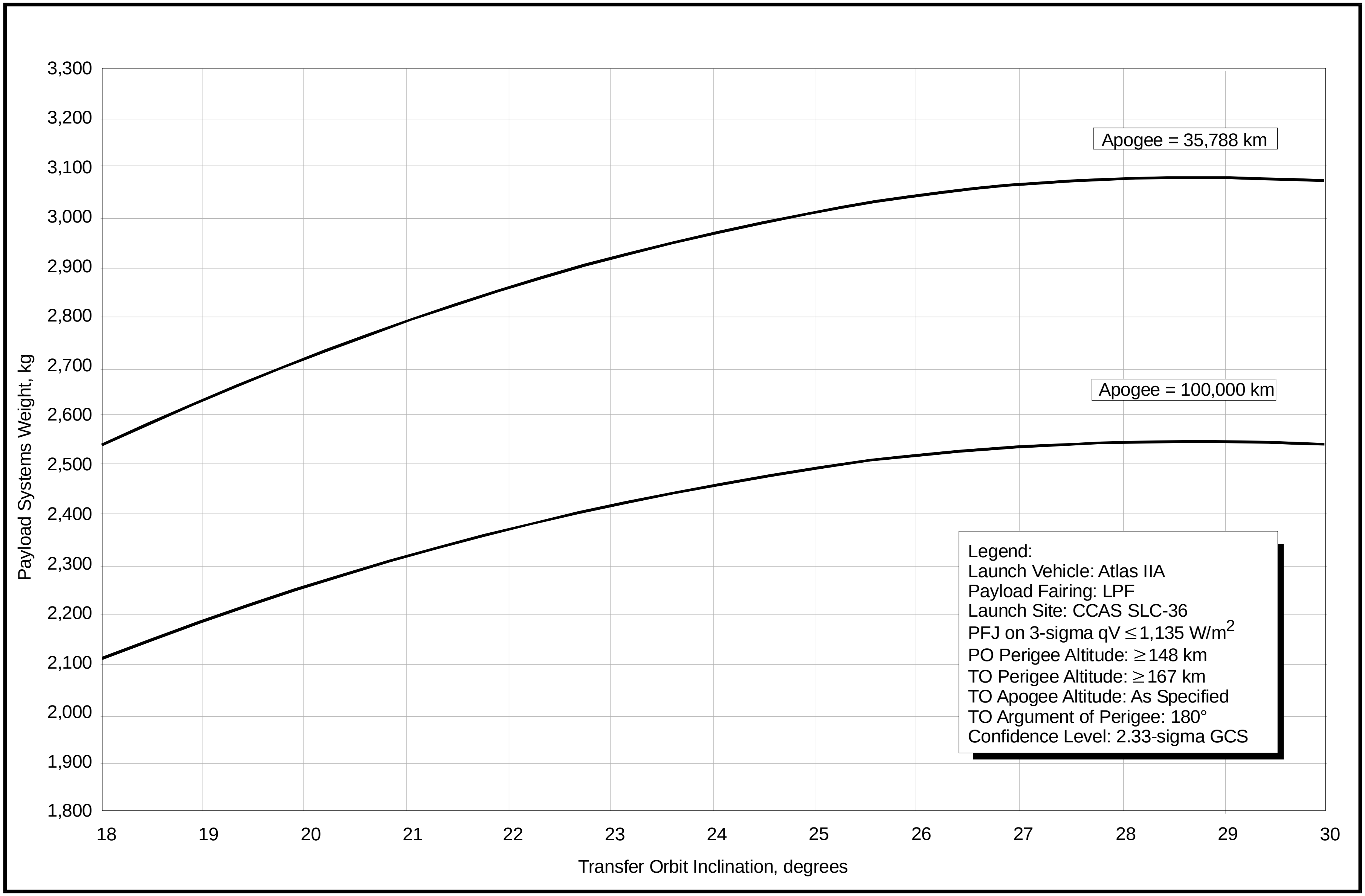


Figure 2.7-4a Atlas IIA CCAS Performance to Reduced Inclination Elliptical Transfer Orbits (GCS-Metric)

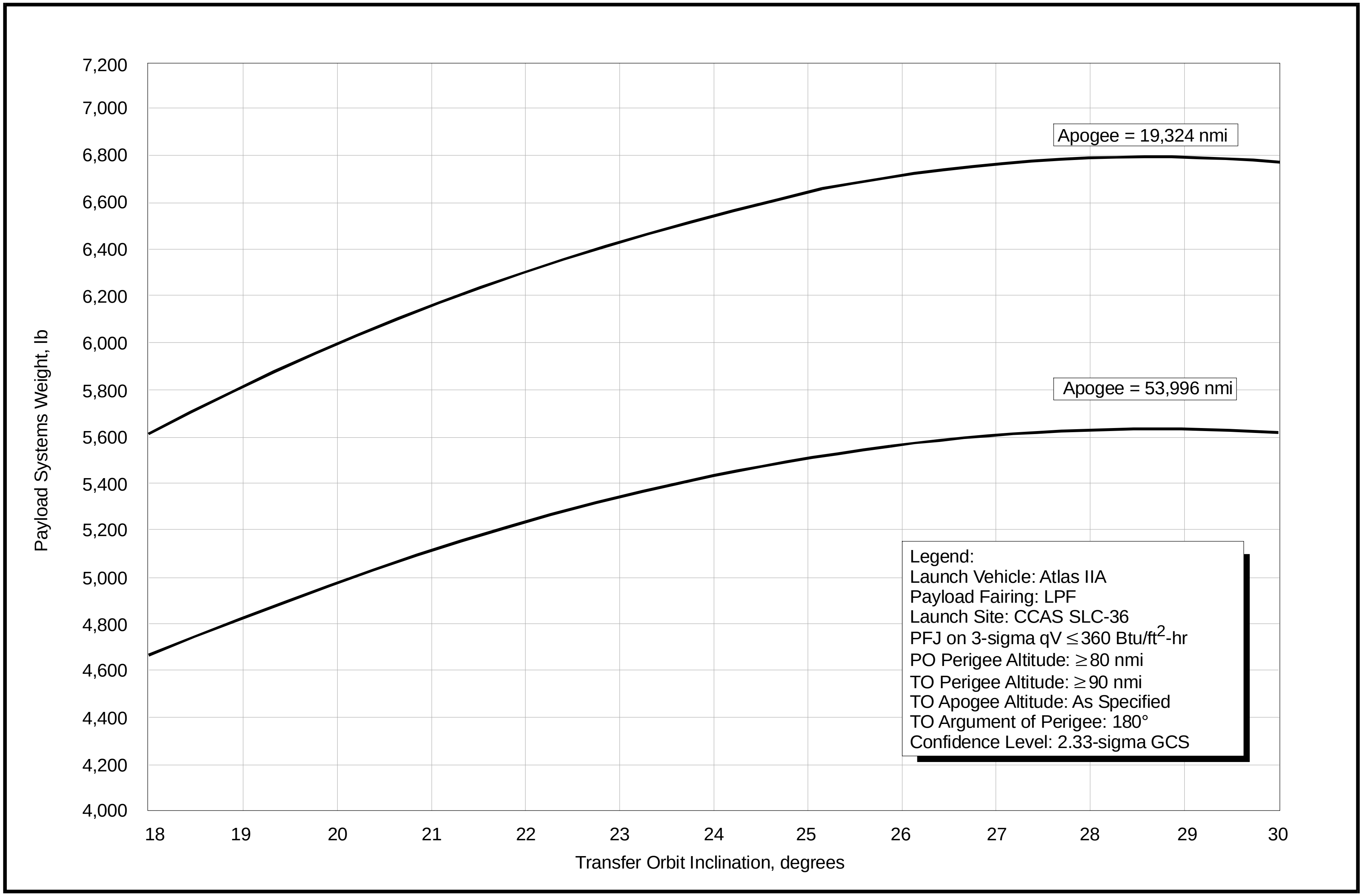


Figure 2.7-4b Atlas IIA CCAS Performance to Reduced Inclination Elliptical Transfer Orbits (GCS-English)

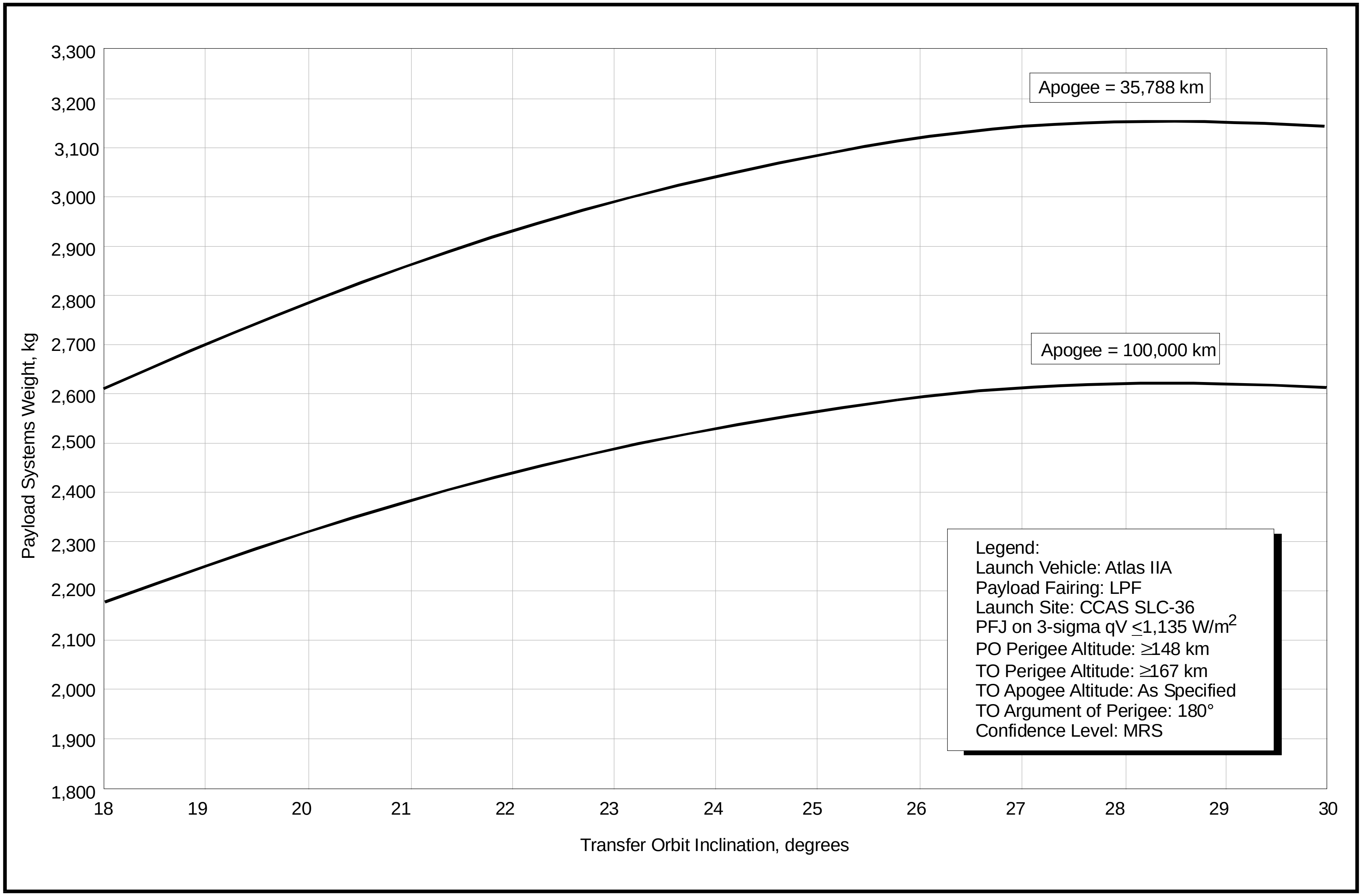


Figure 2.7-5a Atlas IIA CCAS Performance to Reduced Inclination Elliptical Transfer Orbits (MRS-Metric)

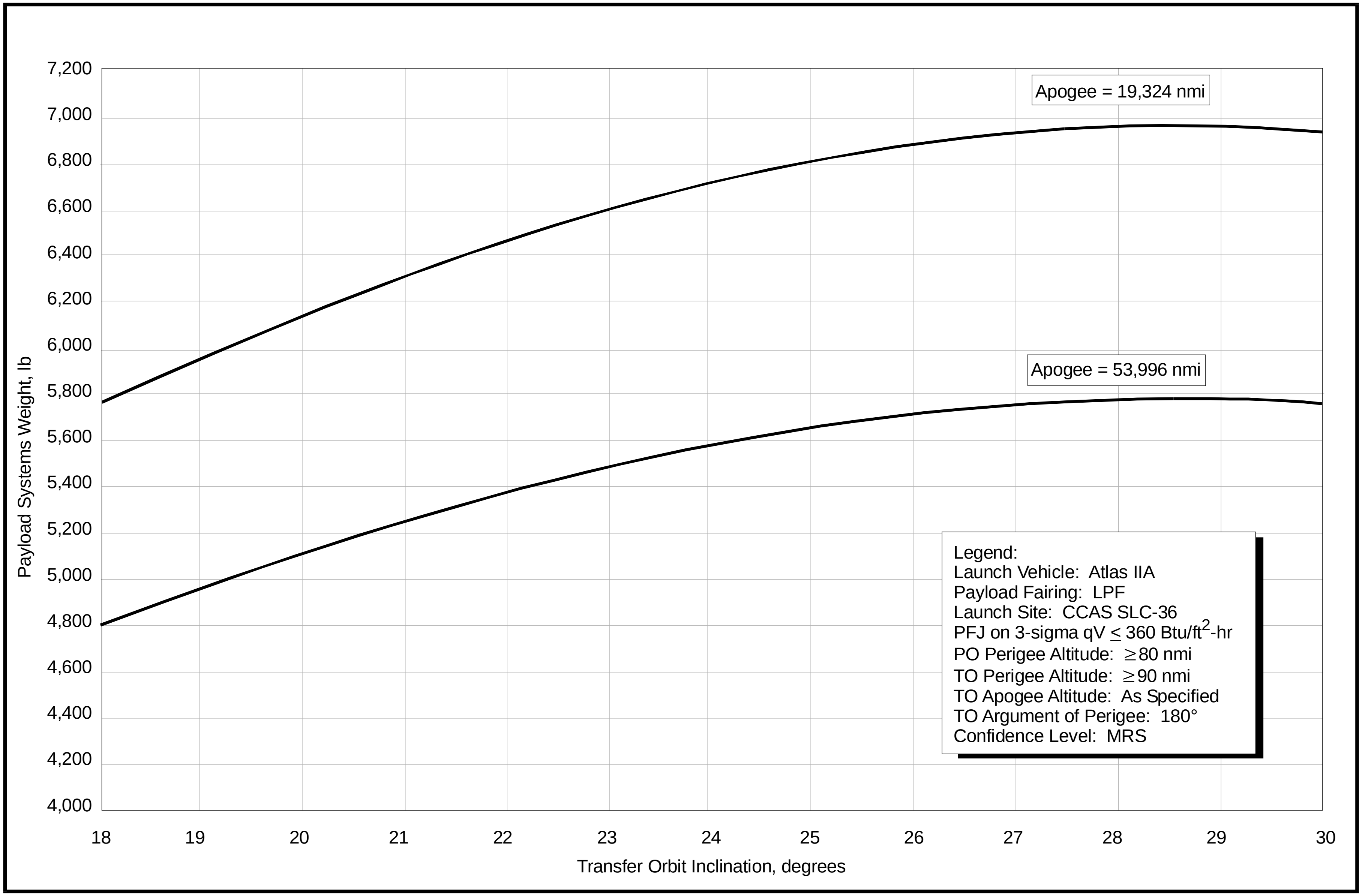


Figure 2.7-5b Atlas IIA CCAS Performance to Reduced Inclination Elliptical Transfer Orbits (MRS-English)

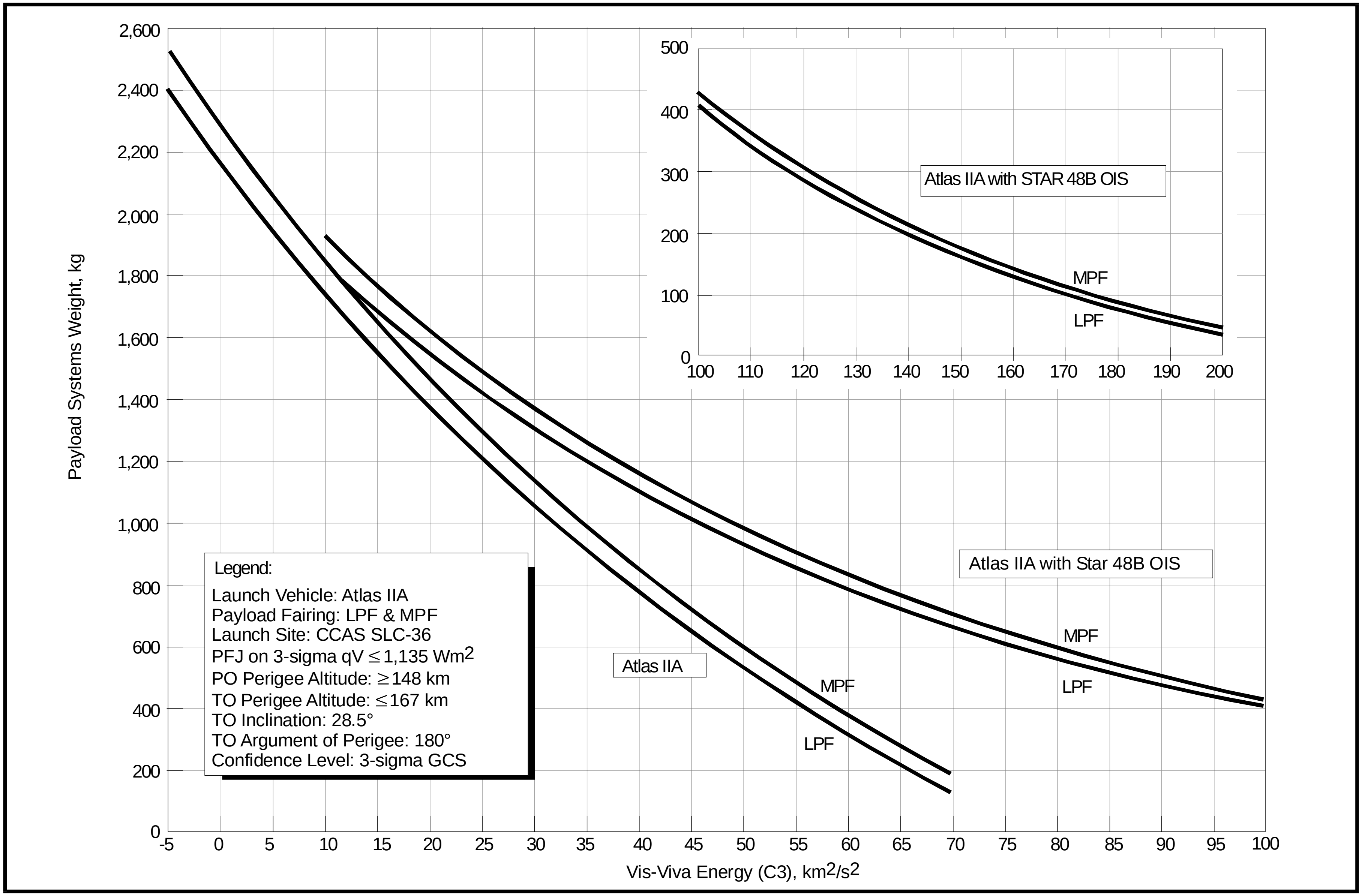


Figure 2.7-6a Atlas IIA CCAS Earth-Escape Performance (Metric)

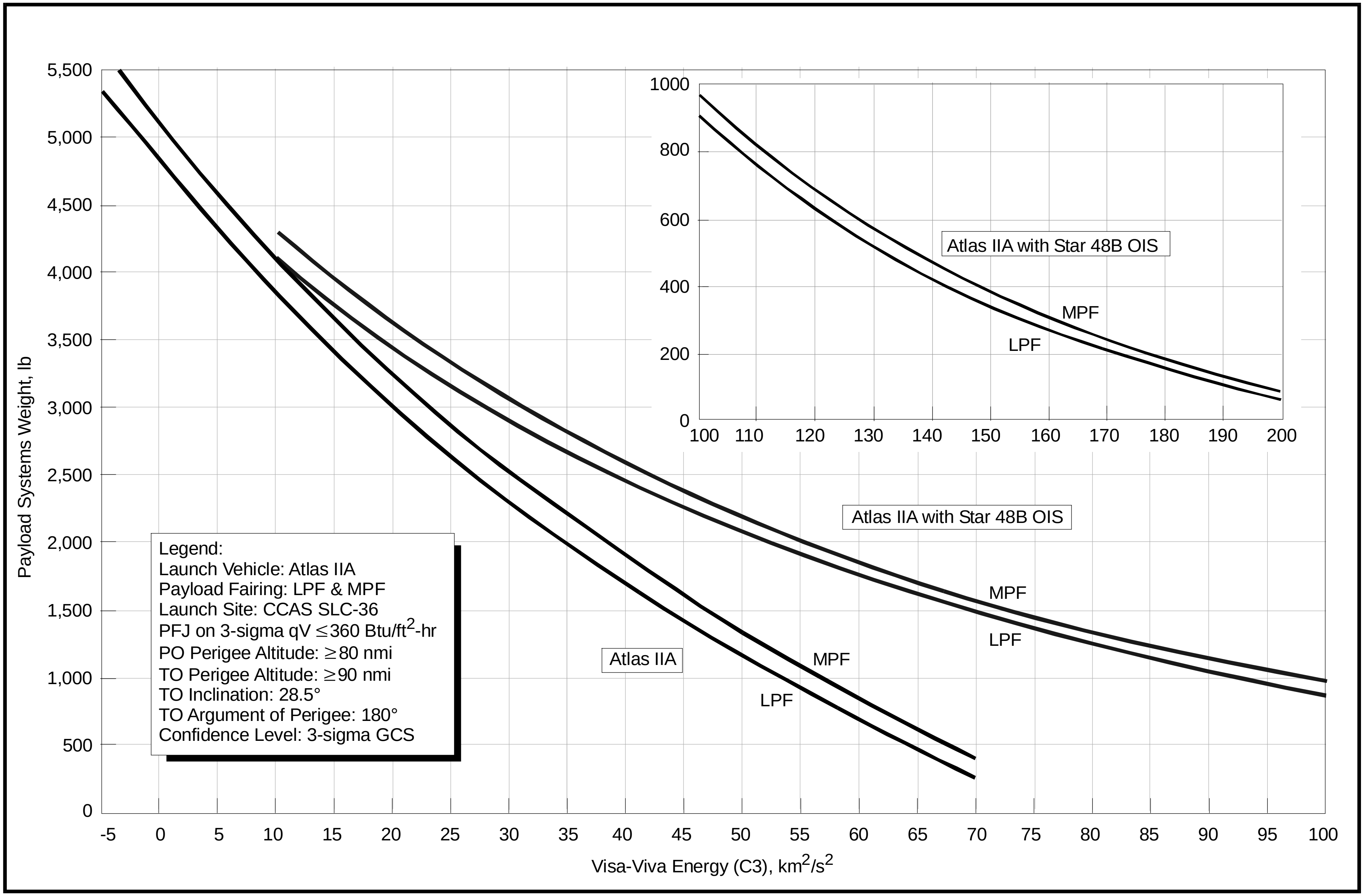


Figure 2.7-6b Atlas IIA CCAS Earth-Escape Performance (English)

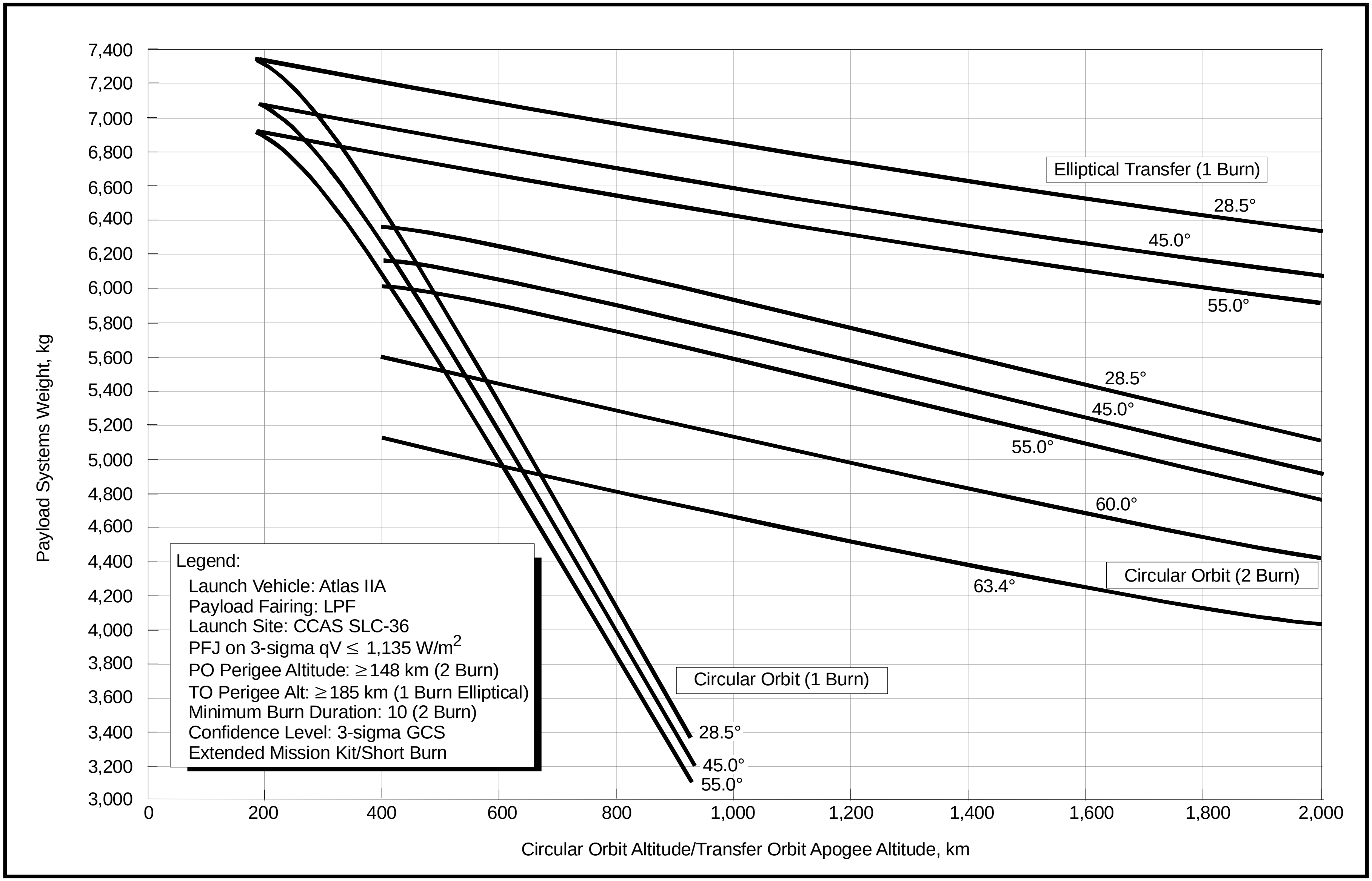


Figure 2.7-7a Atlas IIA CCAS Low-Earth Performance (Metric)

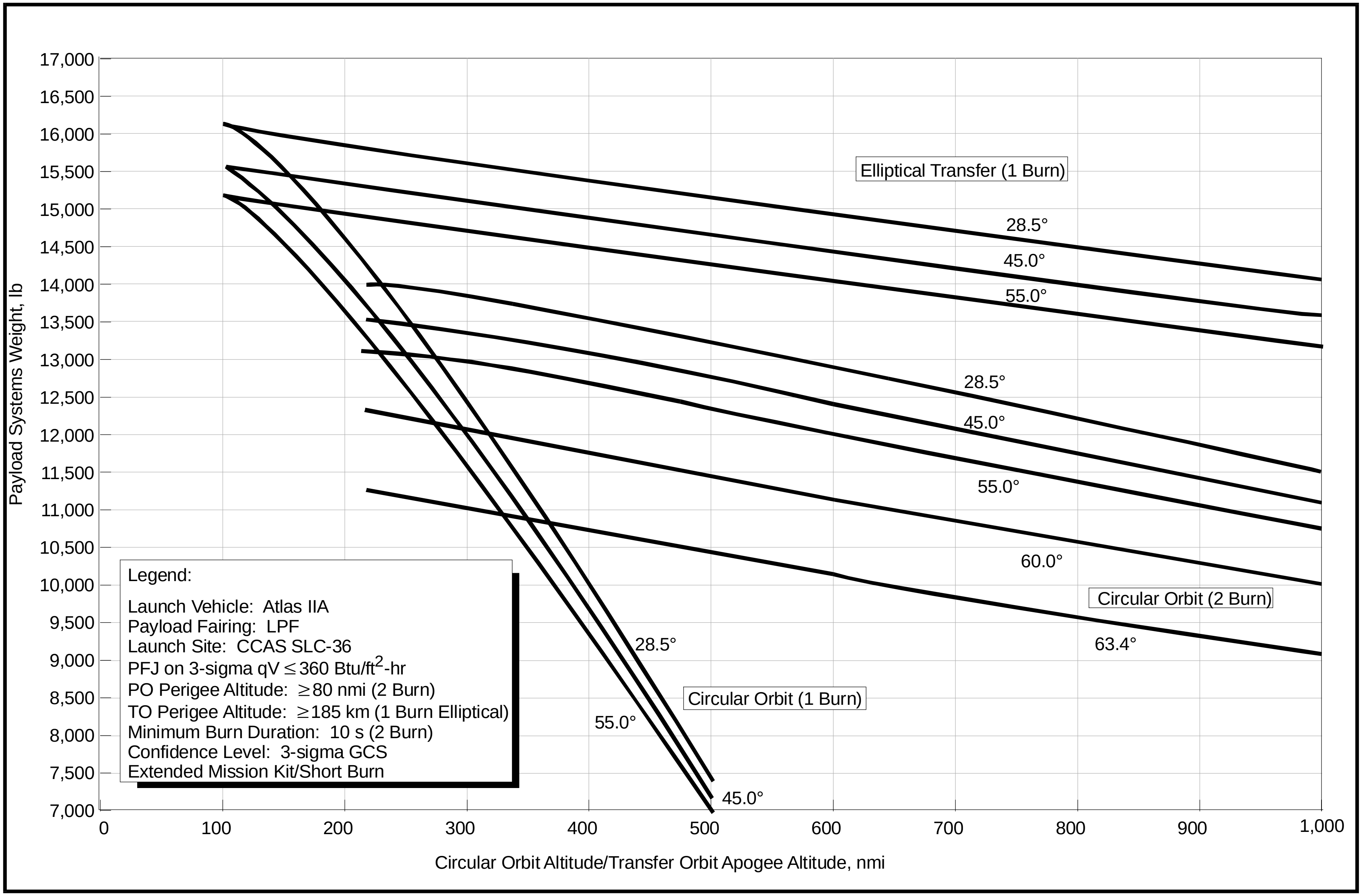


Figure 2.7-7b Atlas IIA CCAS Low-Earth Performance (English)

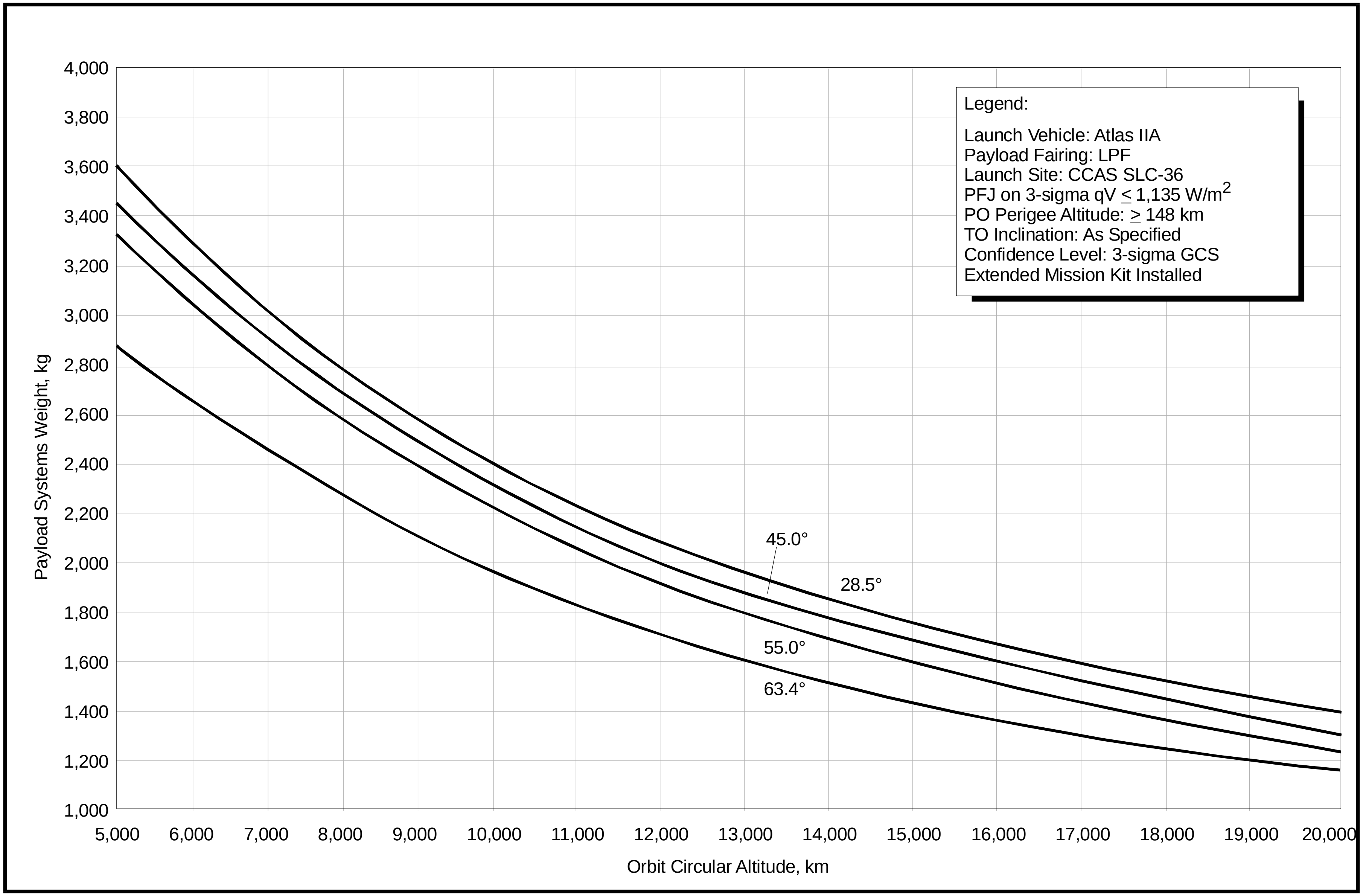


Figure 2.7-8a Atlas IIA CCAS Intermediate Circular Orbit Performance (Metric)

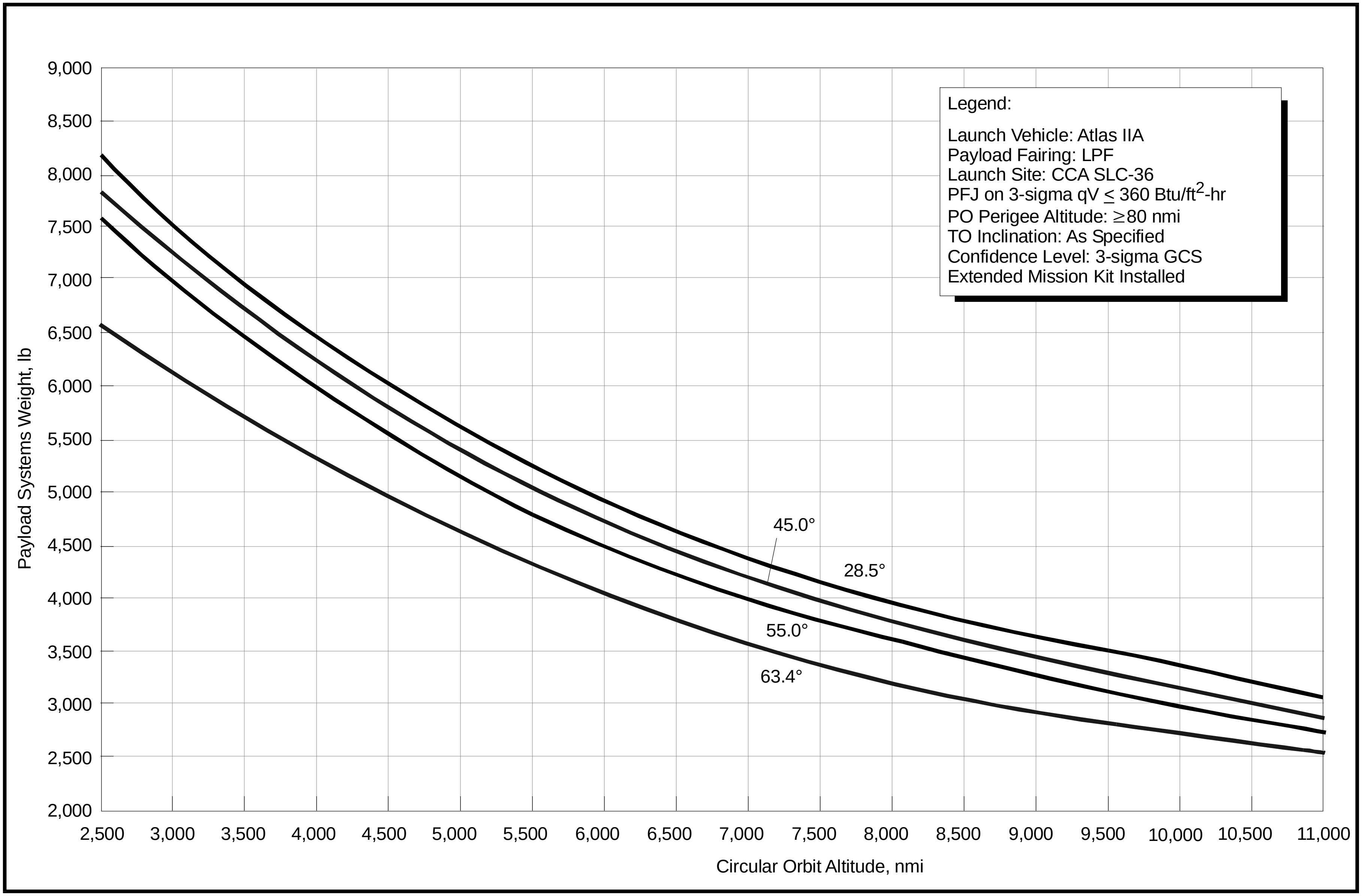


Figure 2.7-8b Atlas IIA CCAS Intermediate Circular Orbit Performance (English)

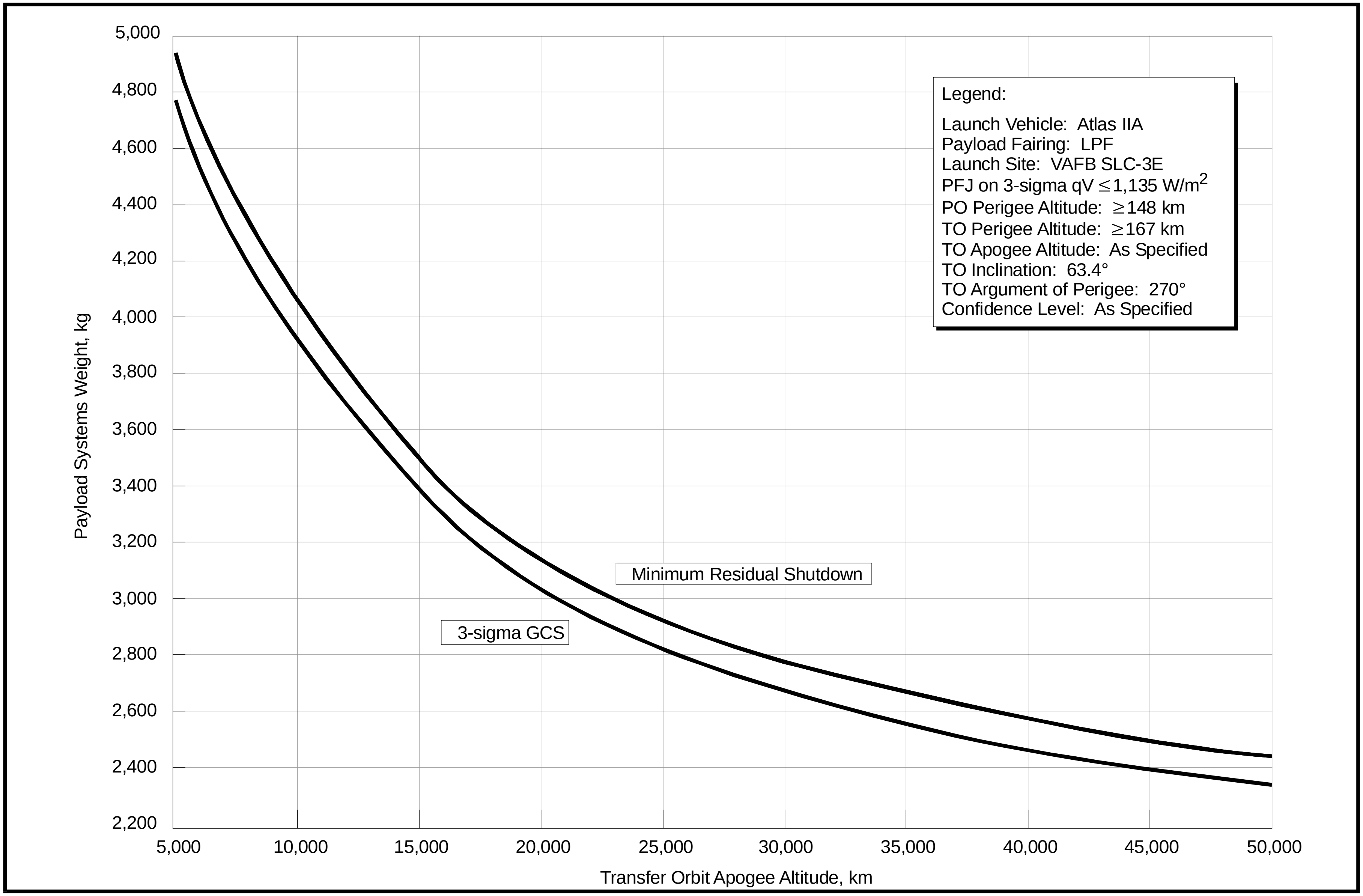


Figure 2.7-9a Atlas IIA VAFB Elliptical Orbit Performance (Metric)

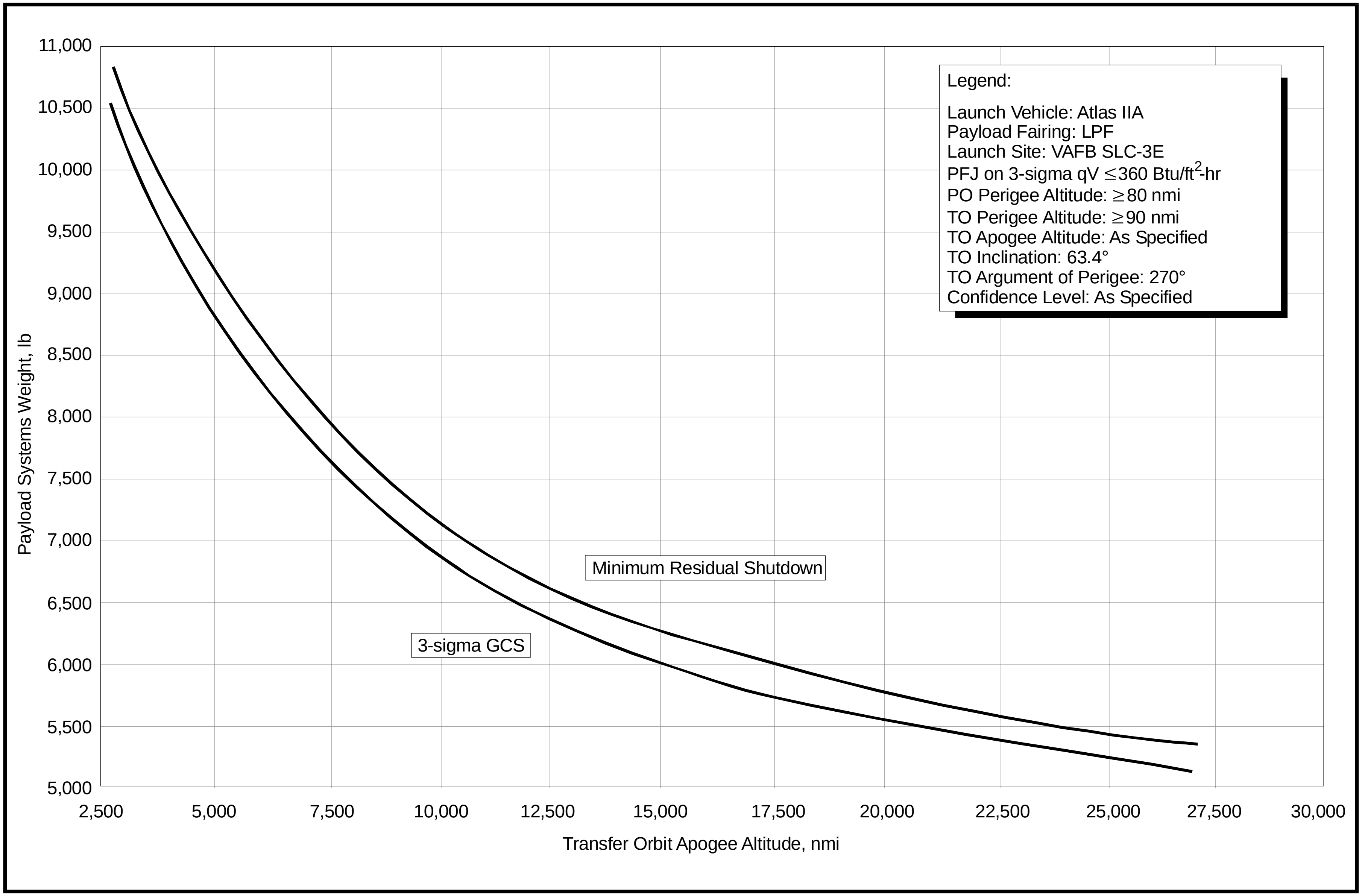


Figure 2.7-9b Atlas IIA VAFB Elliptical Orbit Performance (English)

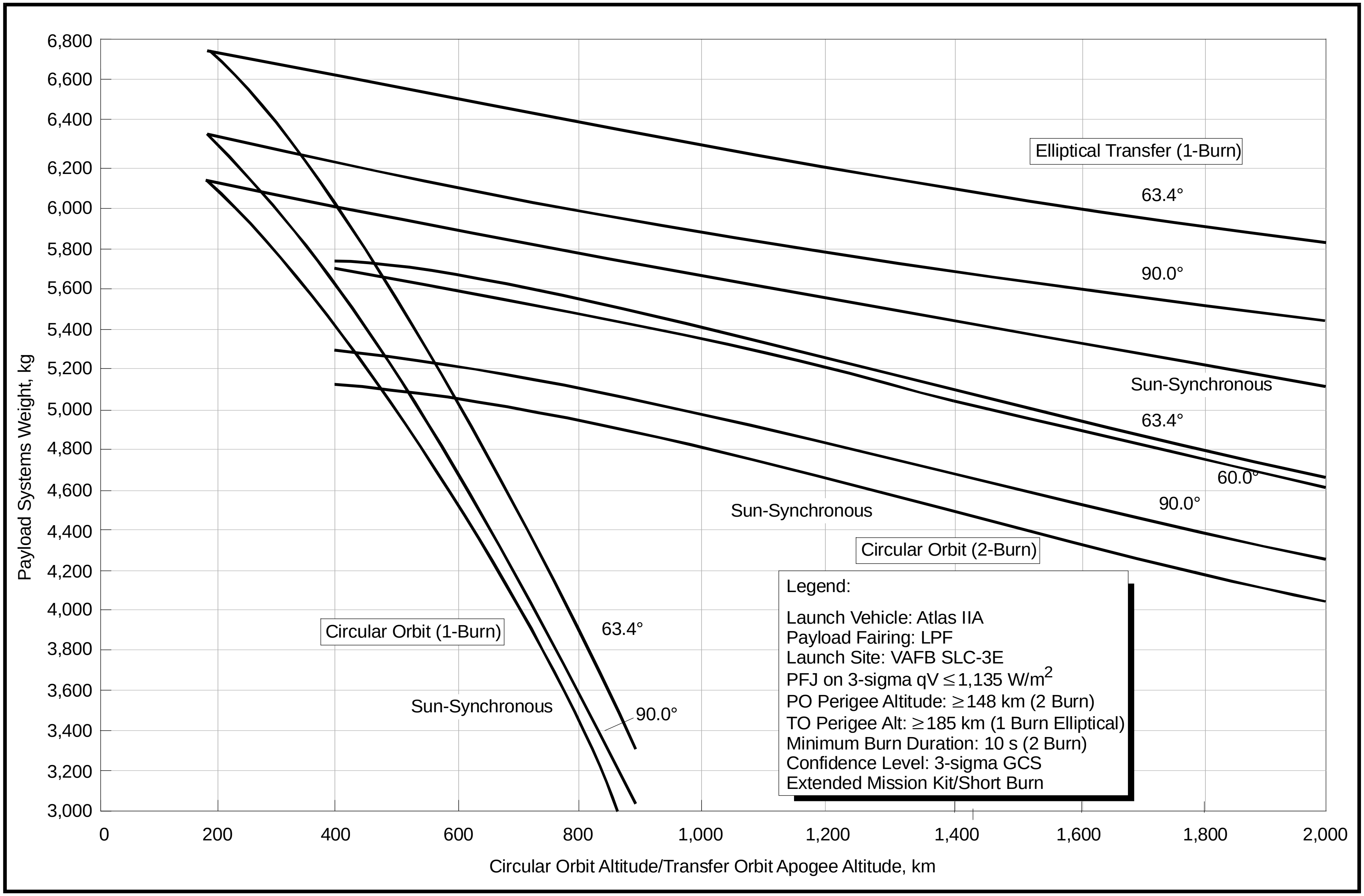


Figure 2.7-10a Atlas IIA VAFB Low-Earth Orbit Performance (Metric)

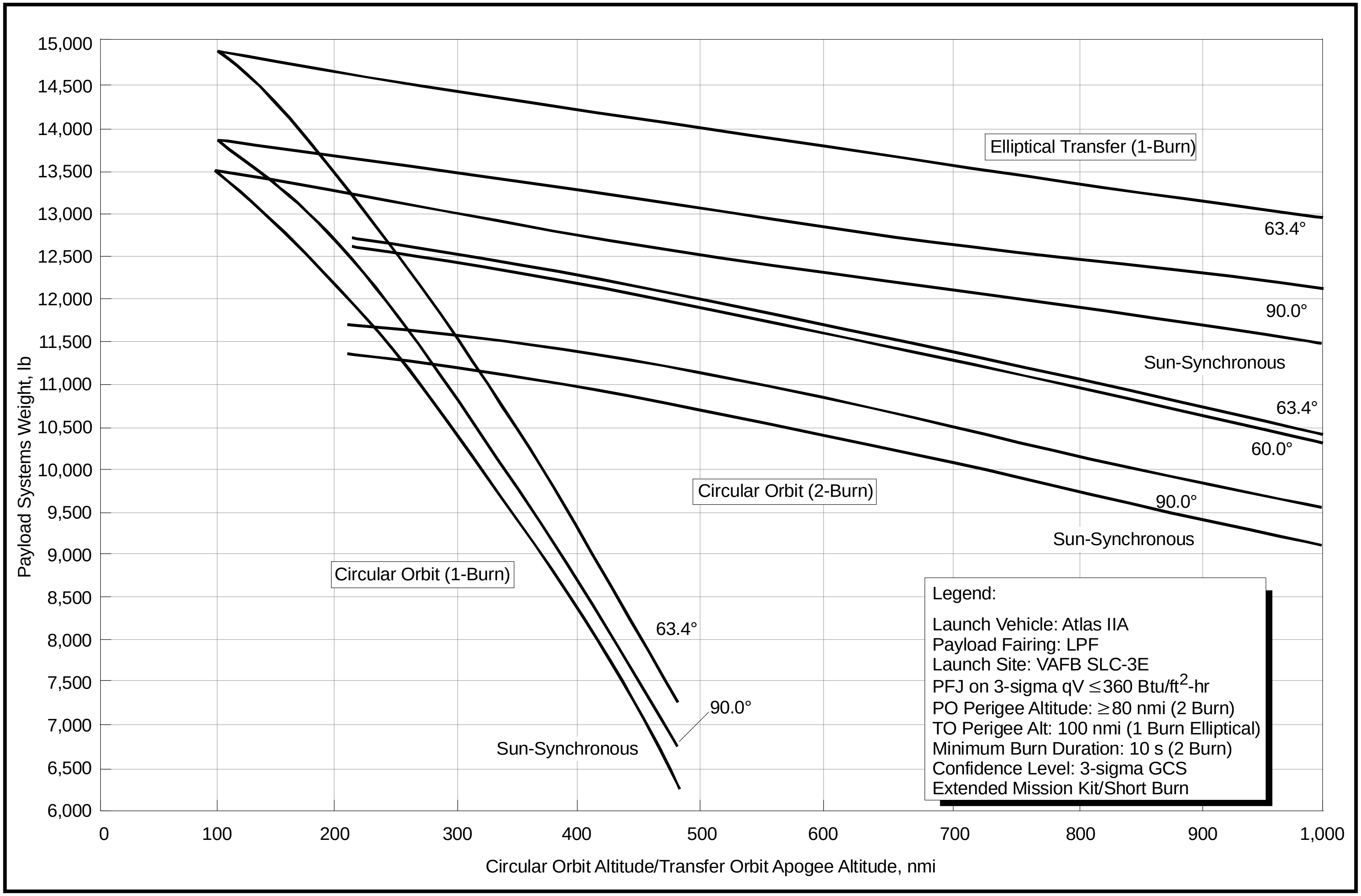


Figure 2.7-10b Atlas IIA VAFB Low-Earth Orbit Performance (English)

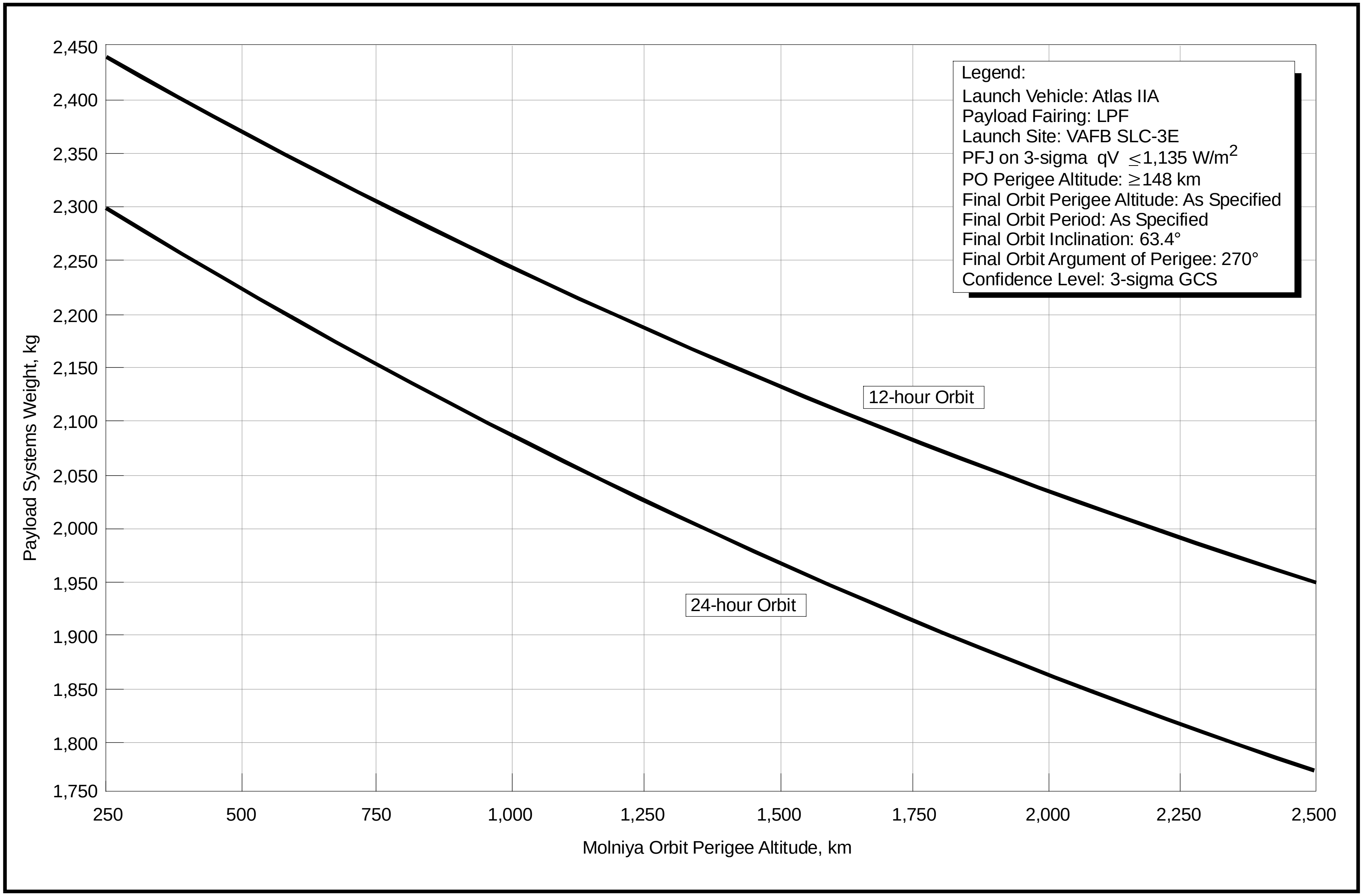


Figure 2.7-11a Atlas IIA VAFB High-Inclination, High-Eccentricity Orbit Performance (Metric)

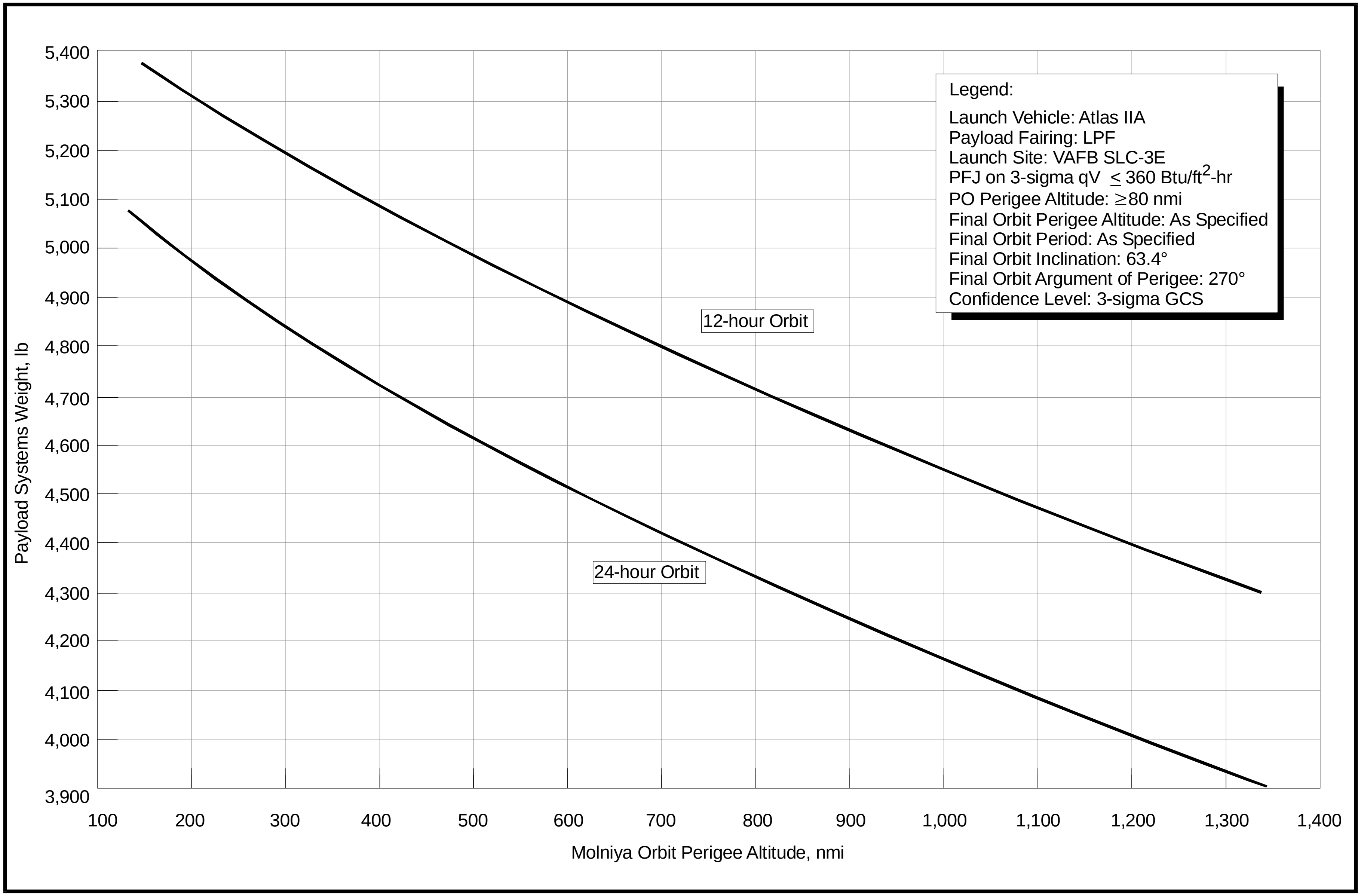


Figure 2.7-11b Atlas IIA VAFB High-Inclination, High-Eccentricity Orbit Performance (English)

Table 2.7.1-1 Atlas IIA Performance to Elliptical Transfer Orbit—PSW vs Apogee Altitude

		Payload Systems Weight, kg (lb)						Payload Systems Weight, kg (lb)			
Apogee Altitude		Atlas IIA				Apogee Altitude		Atlas IIA			
km	(nmi)	MRS		GCS		km	(nmi)	MRS		GCS	
150,000	(80,993.5)	2,497	(5,506)	2,431	(5,360)	40,000	(21,598.3)	3,064	(6,755)	2,987	(6,587)
140,000	(75,593.9)	2,514	(5,543)	2,447	(5,395)	37,500	(20,248.4)	3,109	(6,856)	3,032	(6,686)
130,000	(70,194.4)	2,533	(5,584)	2,466	(5,436)	35,788	(19,324.0)	3,144	(6,931)	3,066	(6,760)
120,000	(64,794.8)	2,554	(5,632)	2,487	(5,483)	35,000	(18,898.5)	3,160	(6,968)	3,083	(6,797)
110,000	(59,395.2)	2,580	(5,689)	2,512	(5,539)	32,500	(17,548.6)	3,218	(7,094)	3,139	(6,921)
100,000	(53,995.7)	2,610	(5,755)	2,542	(5,604)	30,000	(16,198.7)	3,283	(7,239)	3,203	(7,062)
95,000	(51,295.9)	2,627	(5,793)	2,559	(5,641)	27,500	(14,848.8)	3,358	(7,403)	3,277	(7,224)
90,000	(48,596.1)	2,647	(5,836)	2,578	(5,683)	25,000	(13,498.9)	3,455	(7,595)	3,362	(7,412)
85,000	(45,896.3)	2,668	(5,883)	2,599	(5,729)	22,500	(12,149.0)	3,546	(7,818)	3,461	(7,632)
80,000	(43,196.5)	2,692	(5,935)	2,622	(5,781)	20,000	(10,799.1)	3,666	(8,082)	3,579	(7,892)
75,000	(40,496.8)	2,718	(5,994)	2,648	(5,839)	17,500	(9,449.2)	3,810	(8,401)	3,721	(8,205)
70,000	(37,797.0)	2,749	(6,060)	2,678	(5,904)	15,000	(8,099.4)	3,988	(8,792)	3,896	(8,589)
65,000	(35,097.2)	2,783	(6,135)	2,711	(5,978)	12,500	(6,749.5)	4,210	(9,283)	4,115	(9,072)
60,000	(32,397.4)	2,822	(6,222)	2,750	(6,063)	11,000	(5,939.5)	4,373	(9,642)	4,275	(9,425)
55,000	(29,697.6)	2,868	(6,323)	2,795	(6,163)	10,000	(5,399.6)	4,498	(9,917)	4,398	(9,696)
52,500	(28,347.7)	2,894	(6,380)	2,820	(6,219)	9,000	(4,859.6)	4,638	(10,227)	4,536	(10,000)
50,000	(26,997.8)	2,922	(6,441)	2,848	(6,279)	8,000	(4,319.7)	4,797	(10,576)	4,691	(10,343)
47,500	(25,647.9)	2,952	(6,509)	2,878	(6,346)	7,000	(3,779.7)	4,978	(10,975)	4,869	(10,736)
45,000	(24,298.0)	2,986	(6,583)	2,911	(6,418)	6,000	(3,239.7)	5,186	(11,434)	5,074	(11,187)
42,500	(22,948.2)	3,023	(6,665)	2,948	(6,499)	5,000	(2,699.8)	5,428	(11,968)	5,312	(11,712)

Note: Large (4.2-m) Diameter Payload Fairing Jettison at 3-Sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude $\geq$ 148.2 km (80 nmi); Transfer Orbit Perigee Altitude $\geq$ 166.7 (90 nmi); Transfer Orbit Inclination = 27.0°; Argument of Perigee = 180°

Table 2.7.1-2 Atlas IIA Performance to Reduced Inclination Transfer Orbit—PSW vs Orbit Inclination

	Payload Systems Weight, kg (lb)			
	Atlas IIA			
Inclination, °	**MRS**		**GCS**	
18.00	2,613	(5,761)	2,544	(5,609)
18.50	2,658	(5,860)	2,588	(5,707)
19.00	2,701	(5,956)	2,631	(5,802)
19.50	2,744	(6,050)	2,673	(5,894)
20.00	2,785	(6,139)	2,713	(5,982)
20.50	2,824	(6,226)	2,752	(6,067)
21.00	2,862	(6,309)	2,789	(6,149)
21.50	2,897	(6,388)	2,824	(6,226)
22.00	2,931	(6,462)	2,857	(6,299)
22.50	2,963	(6,533)	2,889	(6,369)
23.00	2,993	(6,598)	2,918	(6,433)
23.50	3,020	(6,659)	2,945	(6,493)
24.00	3,045	(6,715)	2,970	(6,547)
24.50	3,068	(6,764)	2,992	(6,596)
25.00	3,088	(6,809)	3,012	(6,640)
25.50	3,106	(6,849)	3,029	(6,679)
26.00	3,121	(6,882)	3,044	(6,712)
26.50	3,134	(6,909)	3,056	(6,739)
27.00	**3,144**	**(6,931)**	**3,066**	**(6,760)**
27.50	3,151	(6,947)	3,073	(6,775)
28.00	3,155	(6,956)	3,077	(6,785)
28.50	3,156	(6,959)	3,079	(6,788)
29.00	3,156	(6,958)	3,078	(6,786)
29.50	3,154	(6,955)	3,077	(6,783)
30.00	3,153	(6,951)	3,075	(6,779)

Note: Large (4.2-m) Diameter Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude = 148.2 km (80 nmi); Transfer Orbit Perigee Altitude = 166.7 (90 nmi); Transfer Orbit Apogee Altitude = 35,788 km (19,324 nmi); Argument of Perigee = 180°

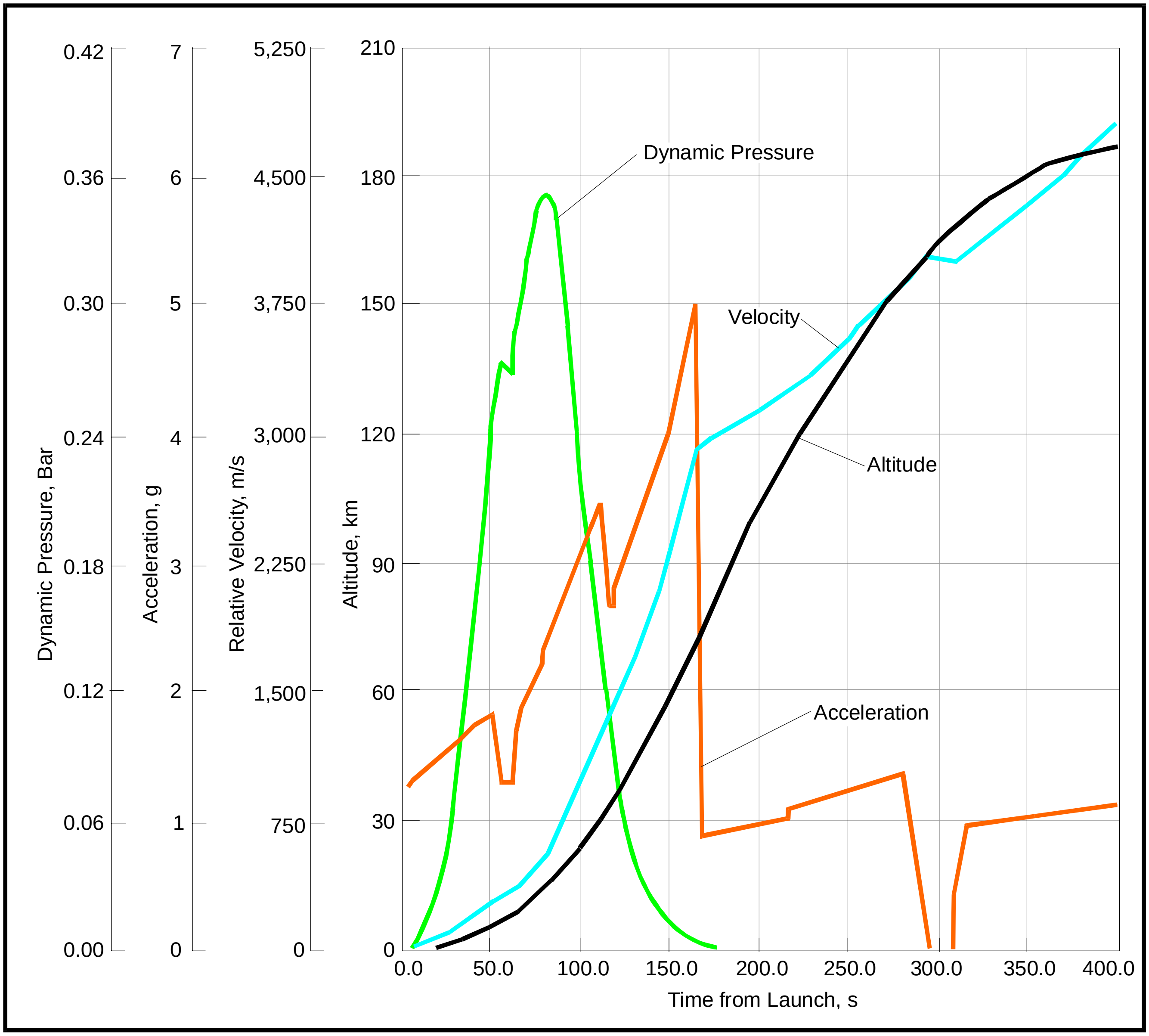


Figure 2.8-1 Atlas IIAS Nominal Ascent Data

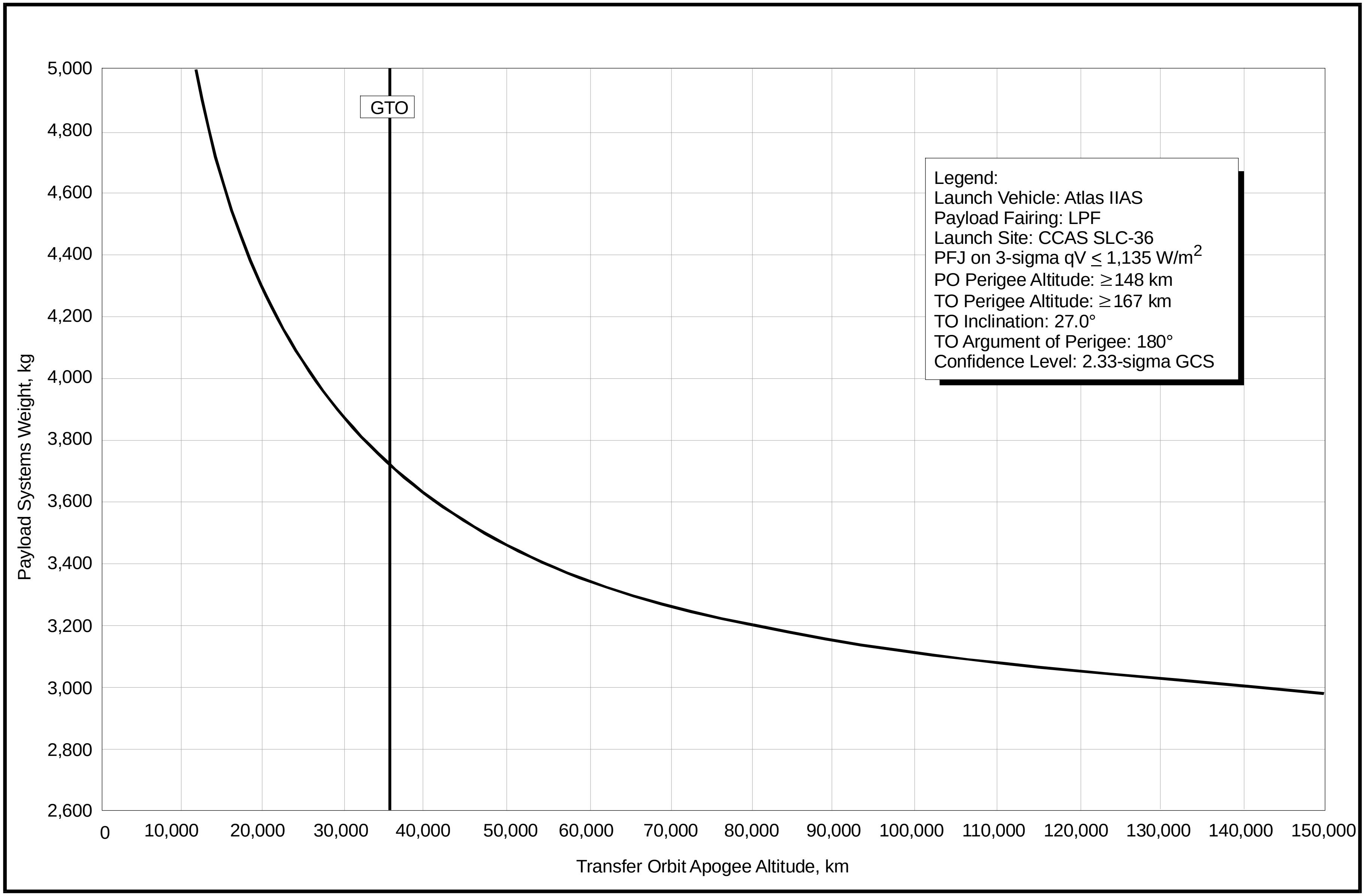


Figure 2.8-2a Atlas IIAS CCAS Performance to Elliptical Transfer Orbit (GCS-Metric)

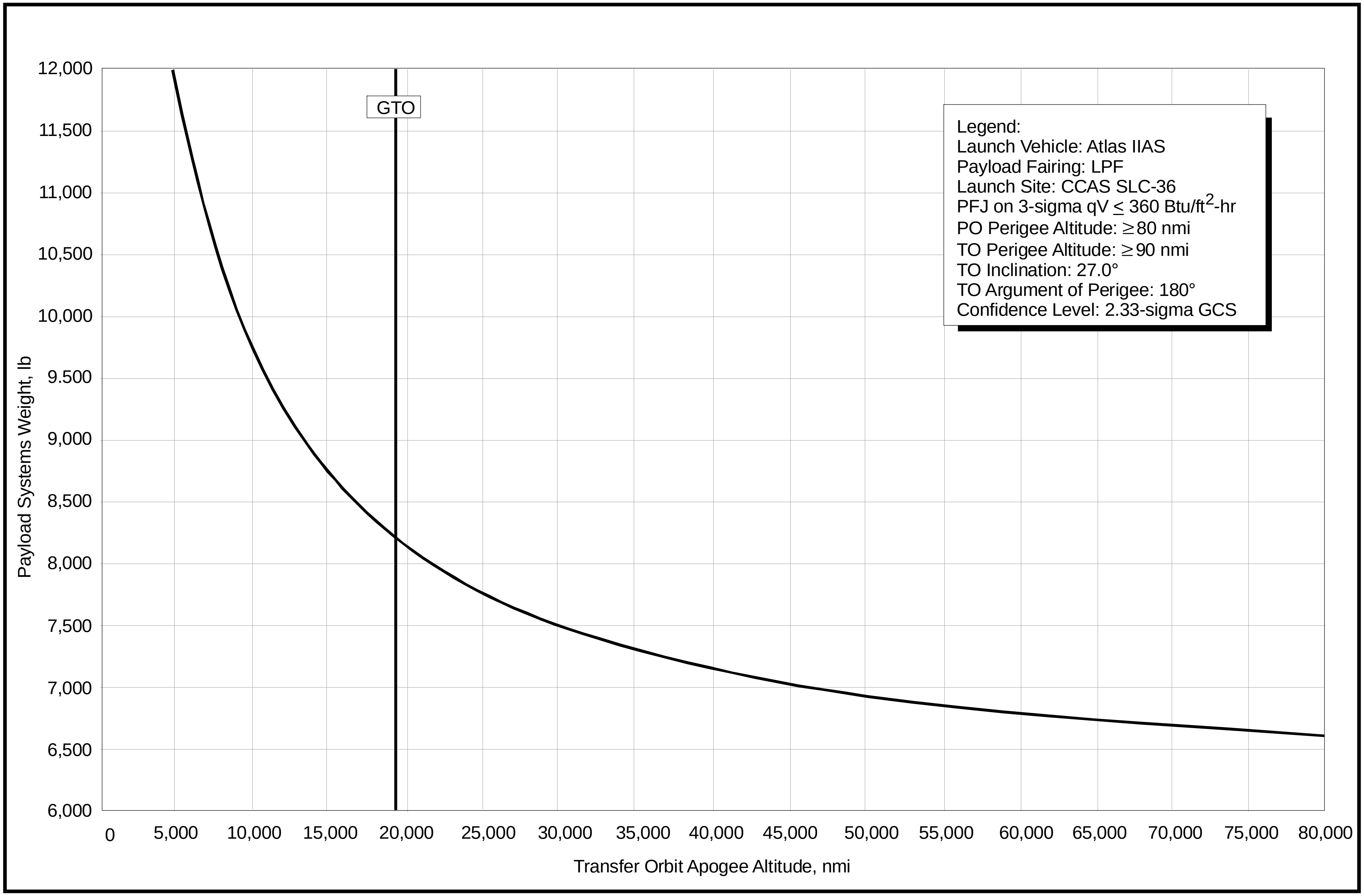


Figure 2.8-2b Atlas IIAS CCAS Performance to Elliptical Transfer Orbit (GCS-English)

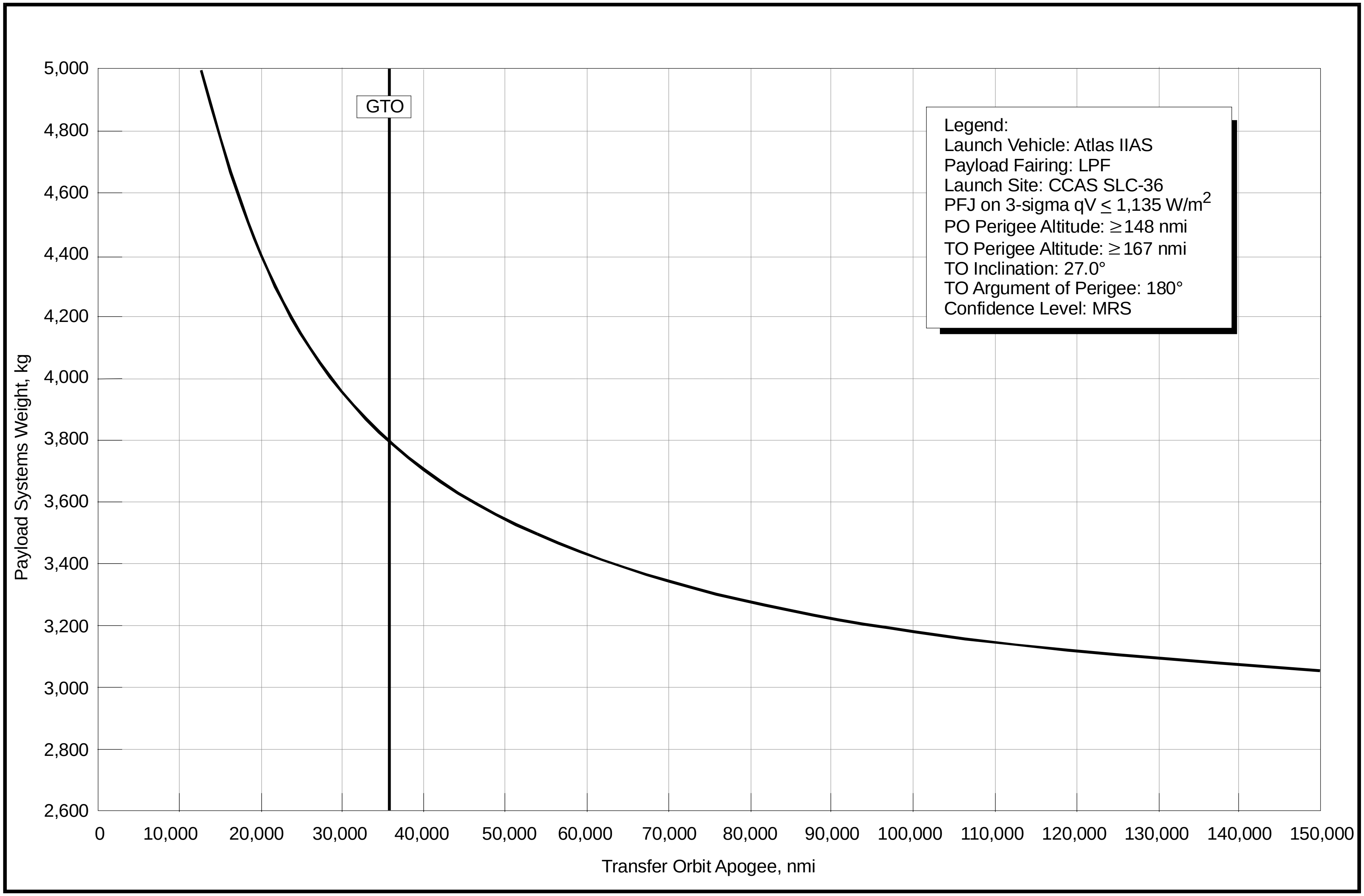


Figure 2.8-3a Atlas IIAS CCAS Performance to Elliptical Transfer Orbit (MRS-Metric)

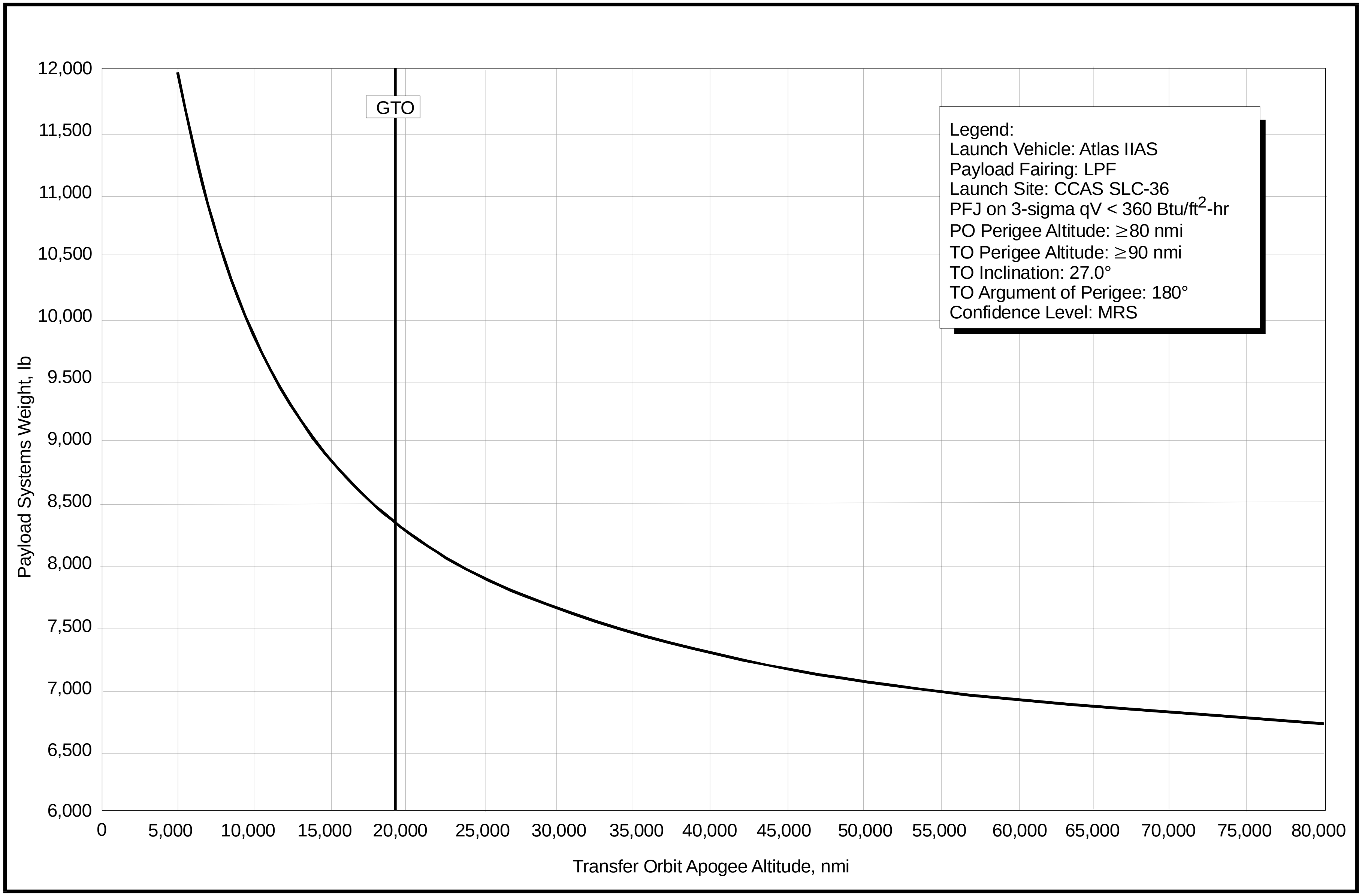


Figure 2.8-3b Atlas IIAS CCAS Performance to Elliptical Transfer Orbit (MRS-English)

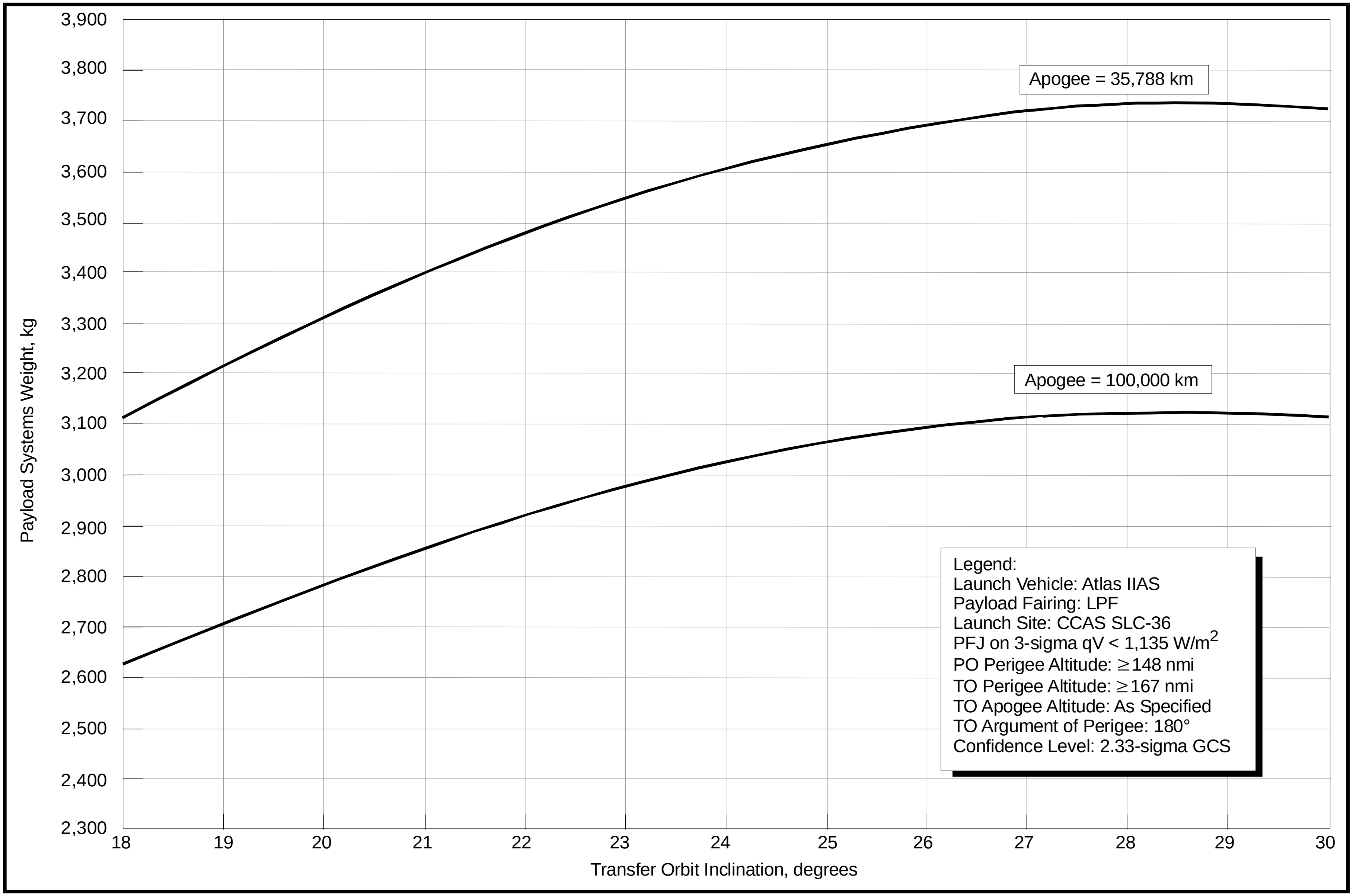


Figure 2.8-4a Atlas IIAS CCAS Reduced Inclination Elliptical Orbit Performance (GCS-Metric)

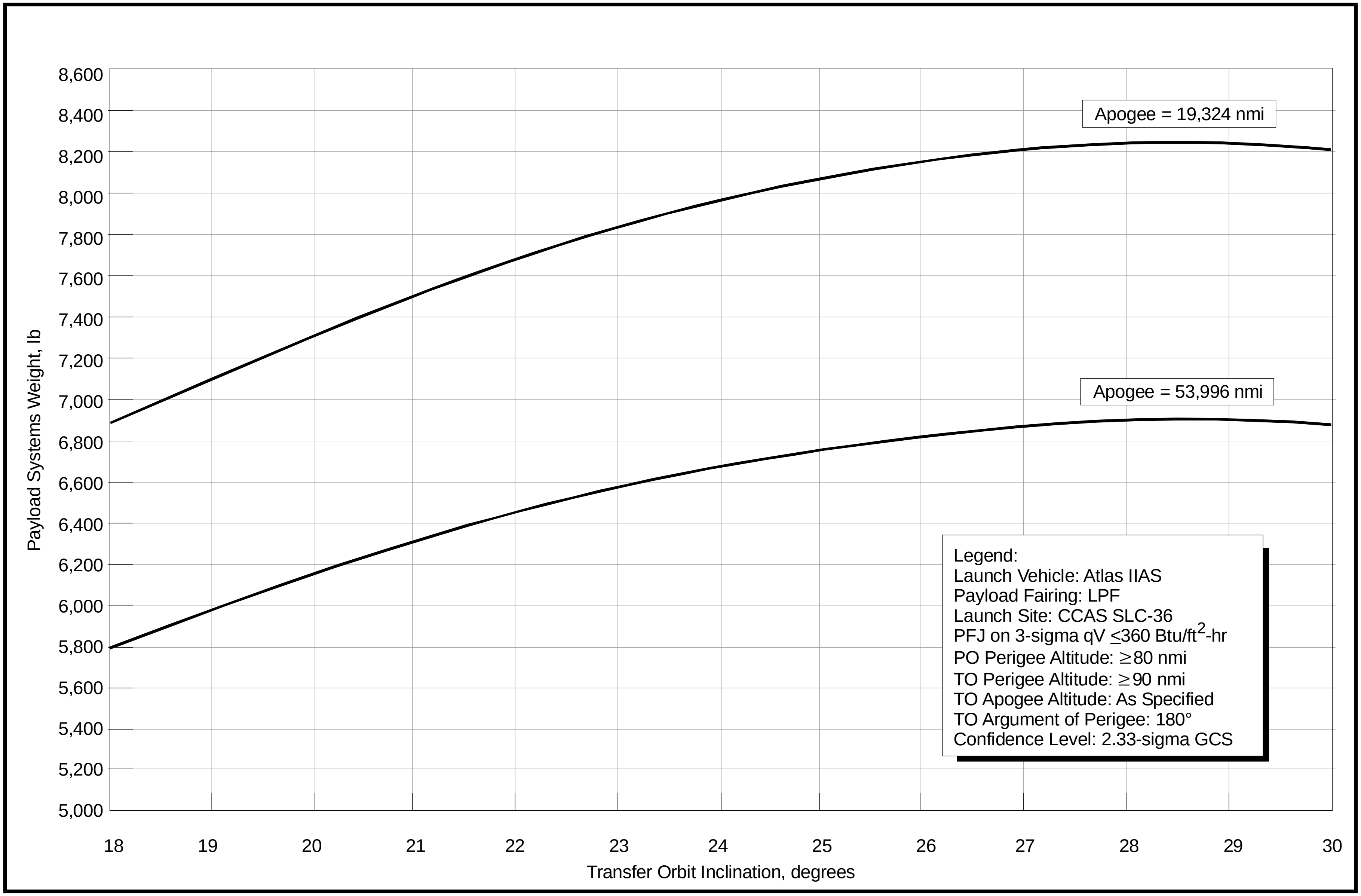


Figure 2.8-4b Atlas IIAS CCAS Reduced Inclination Elliptical Orbit Performance (GCS-English)

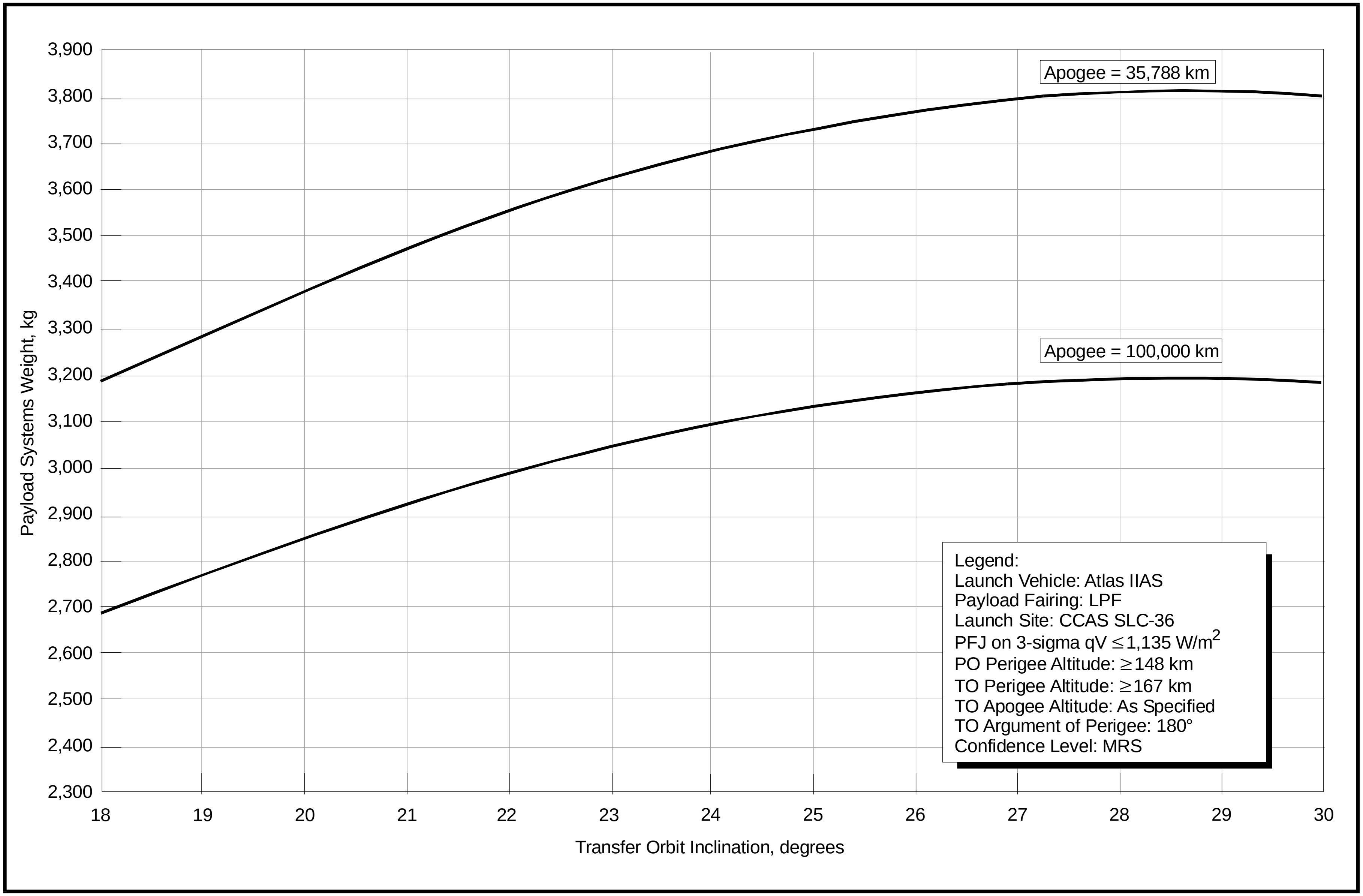


Figure 2.8-5a Atlas IIAS CCAS Reduced Inclination Elliptical Orbit Performance (MRS-Metric)

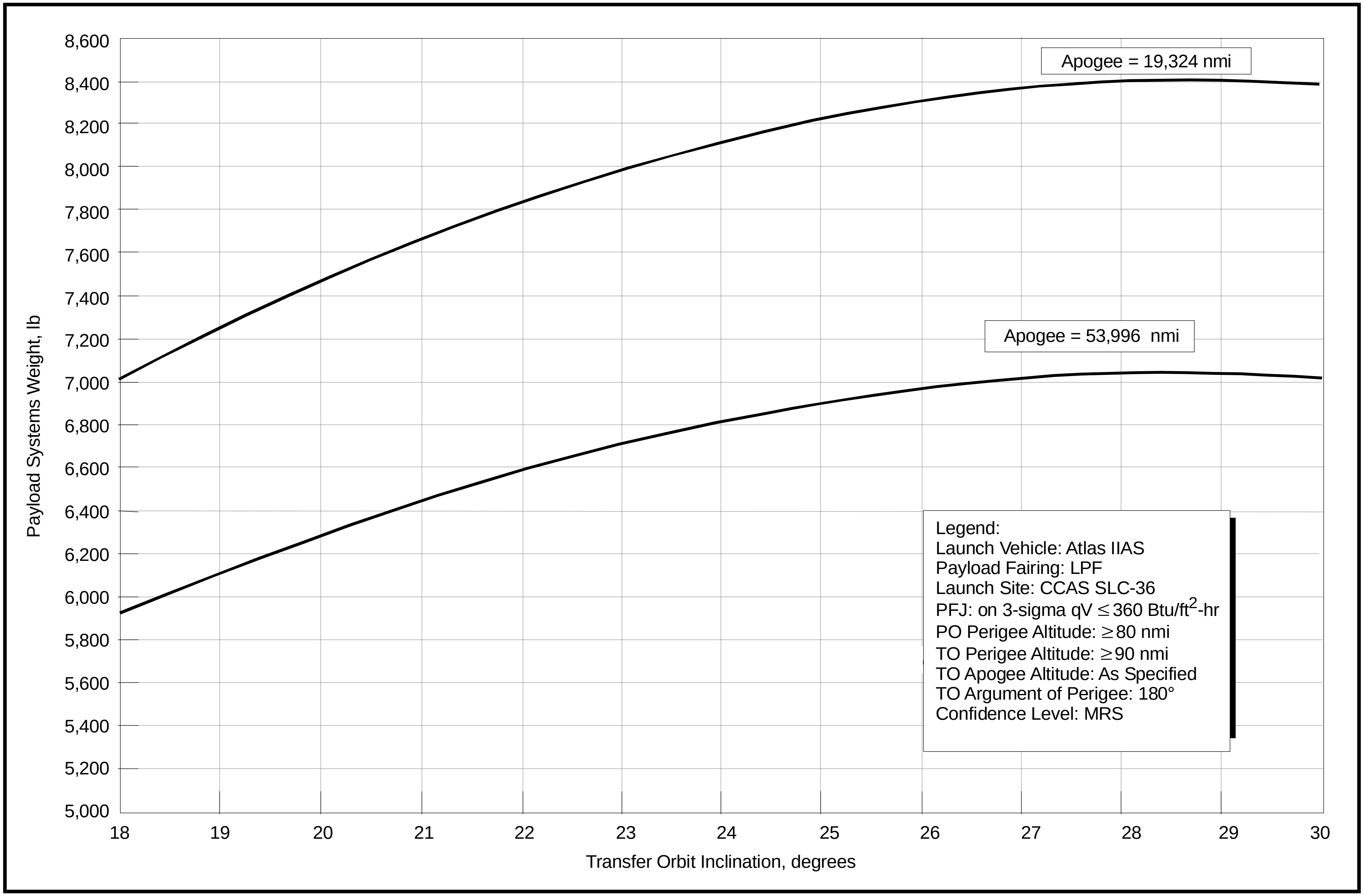


Figure 2.8-5b Atlas IIAS CCAS Reduced Inclination Elliptical Orbit Performance (MRS-English)

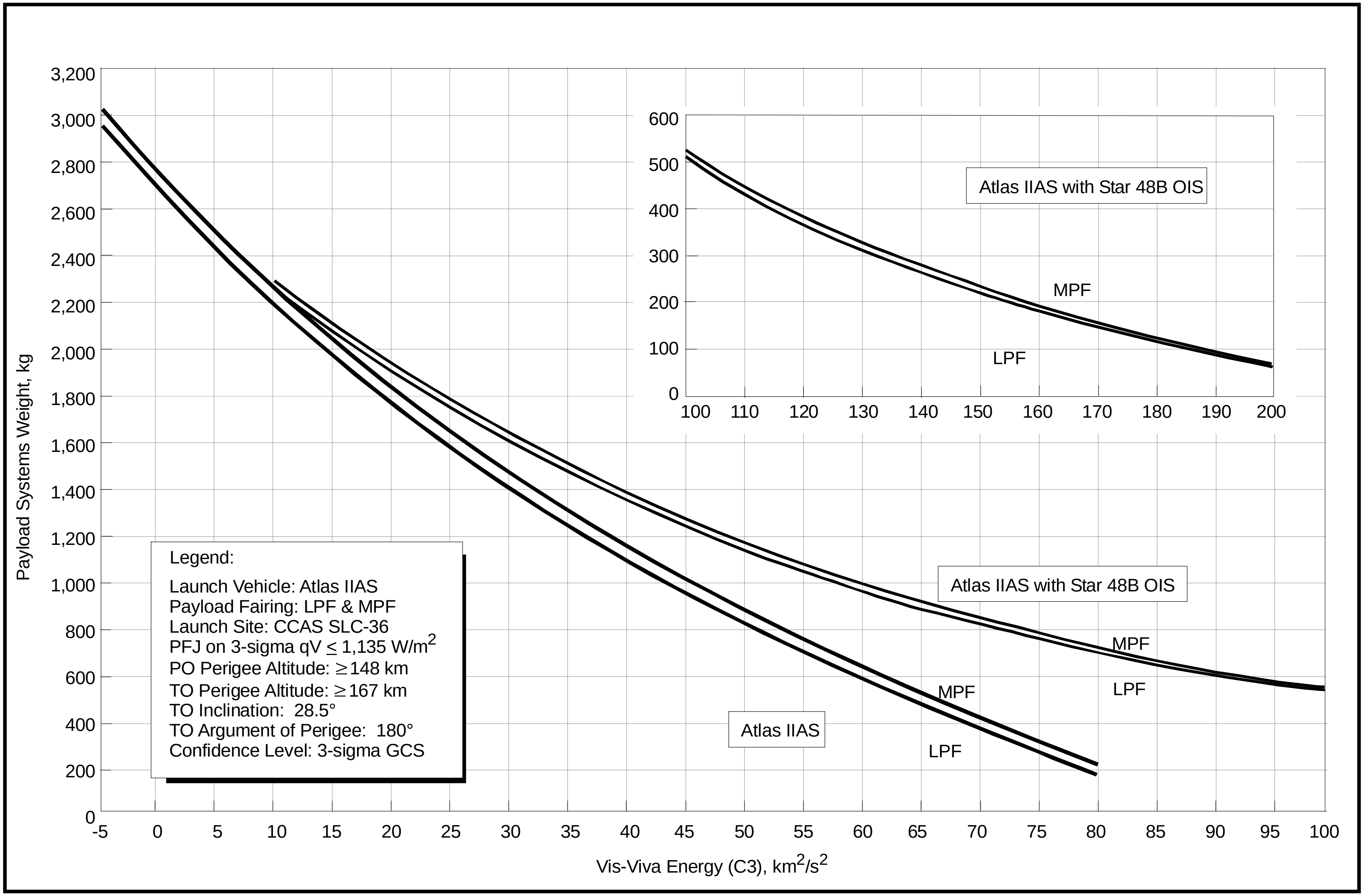


Figure 2.8-6a Atlas IIAS CCAS Earth-Escape Performance (Metric)

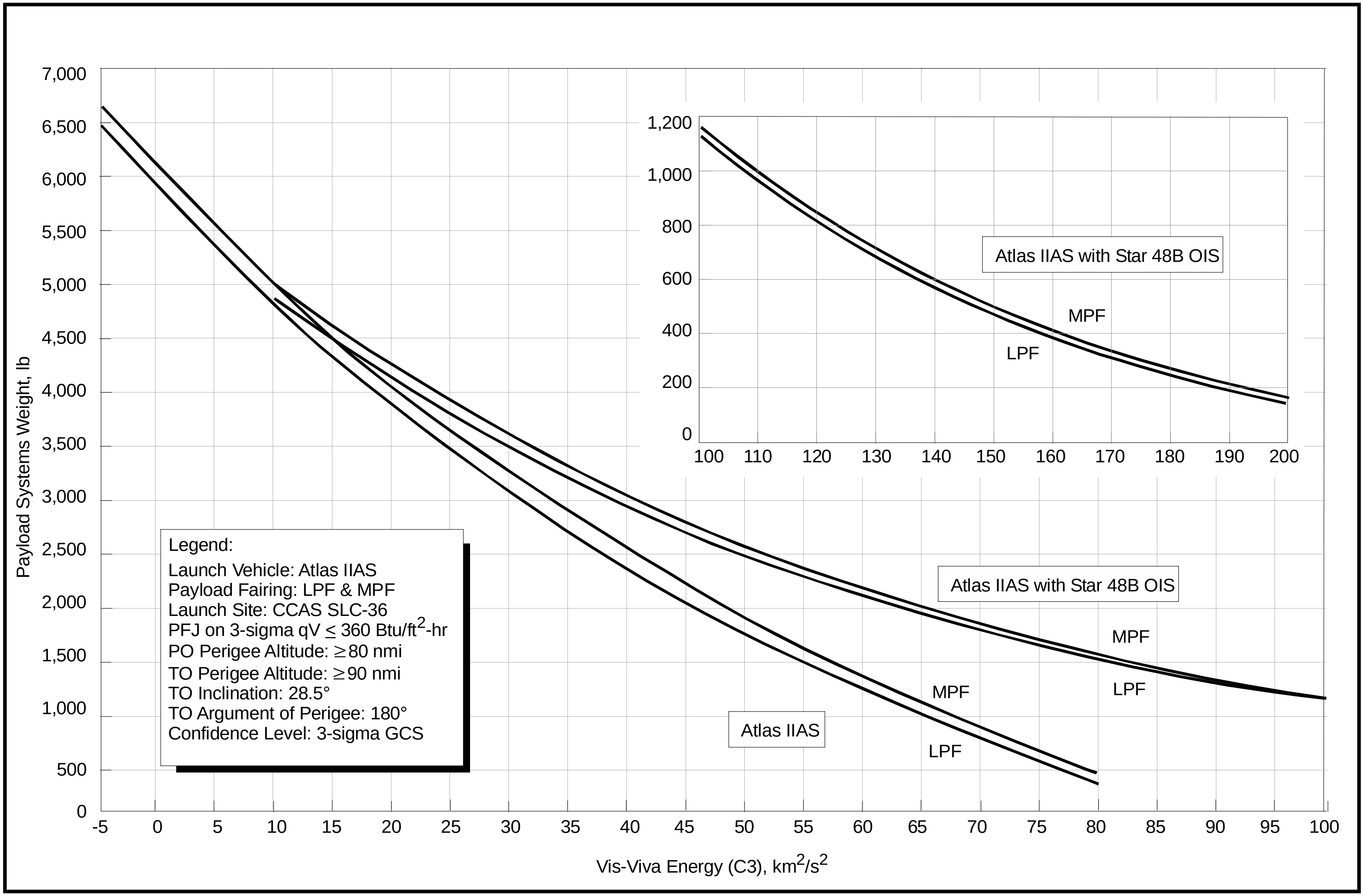


Figure 2.8-6b Atlas IIAS CCAS Earth-Escape Performance (English)

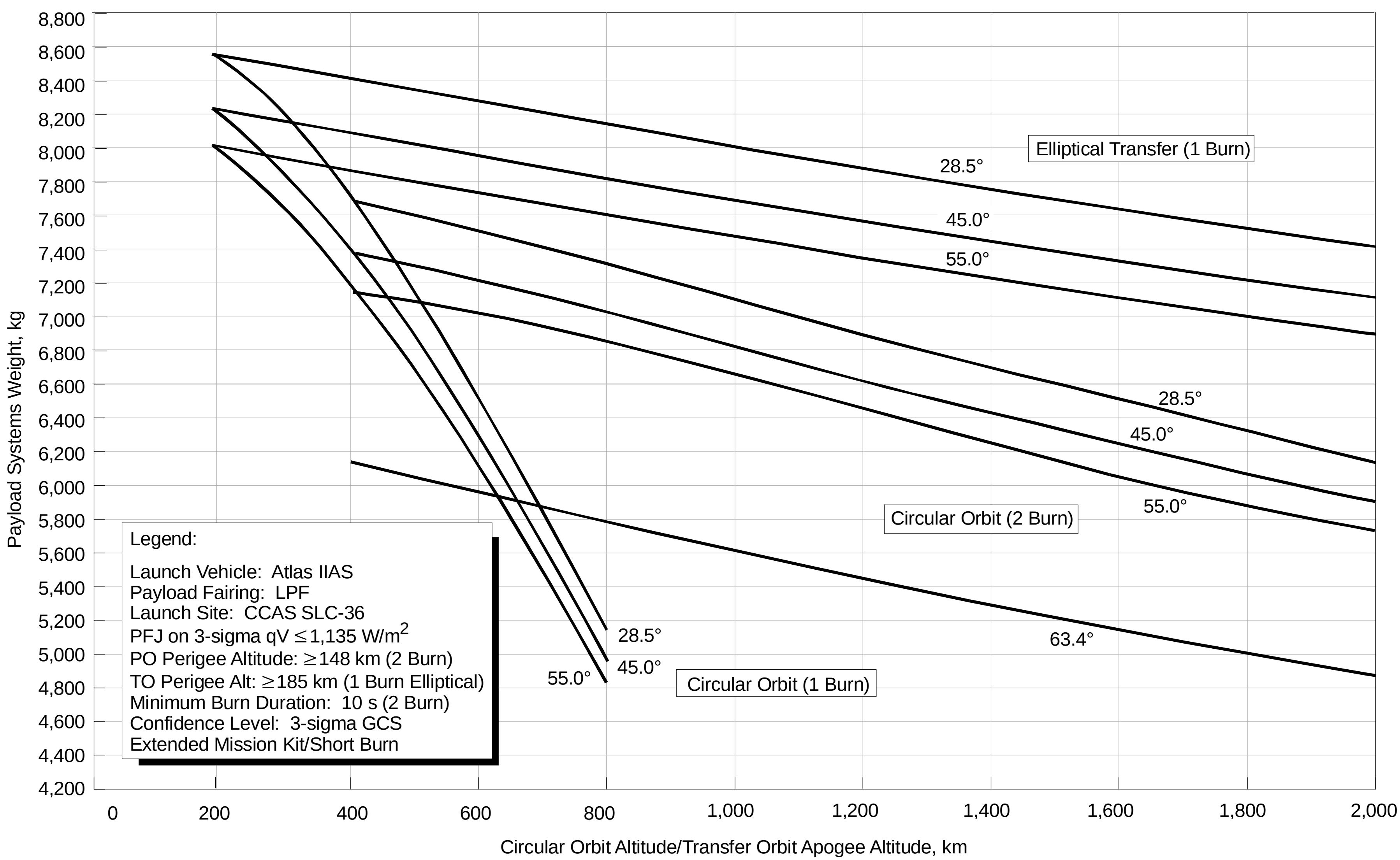


Figure 2.8-7a Atlas IIAS CCAS Low-Earth Orbit Performance (Metric)

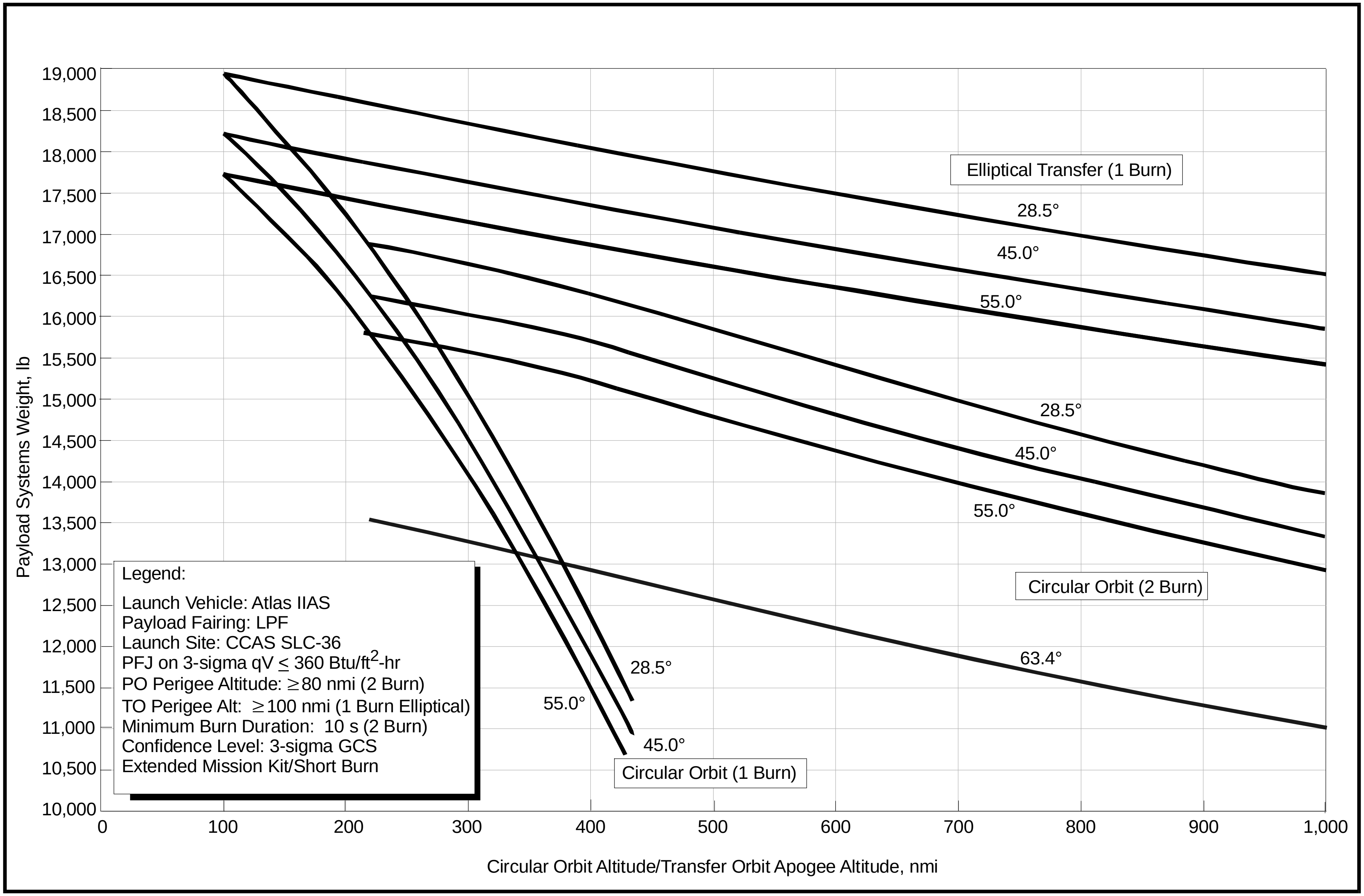


Figure 2.8-7b Atlas IIAS CCAS Low-Earth Orbit Performance (English)

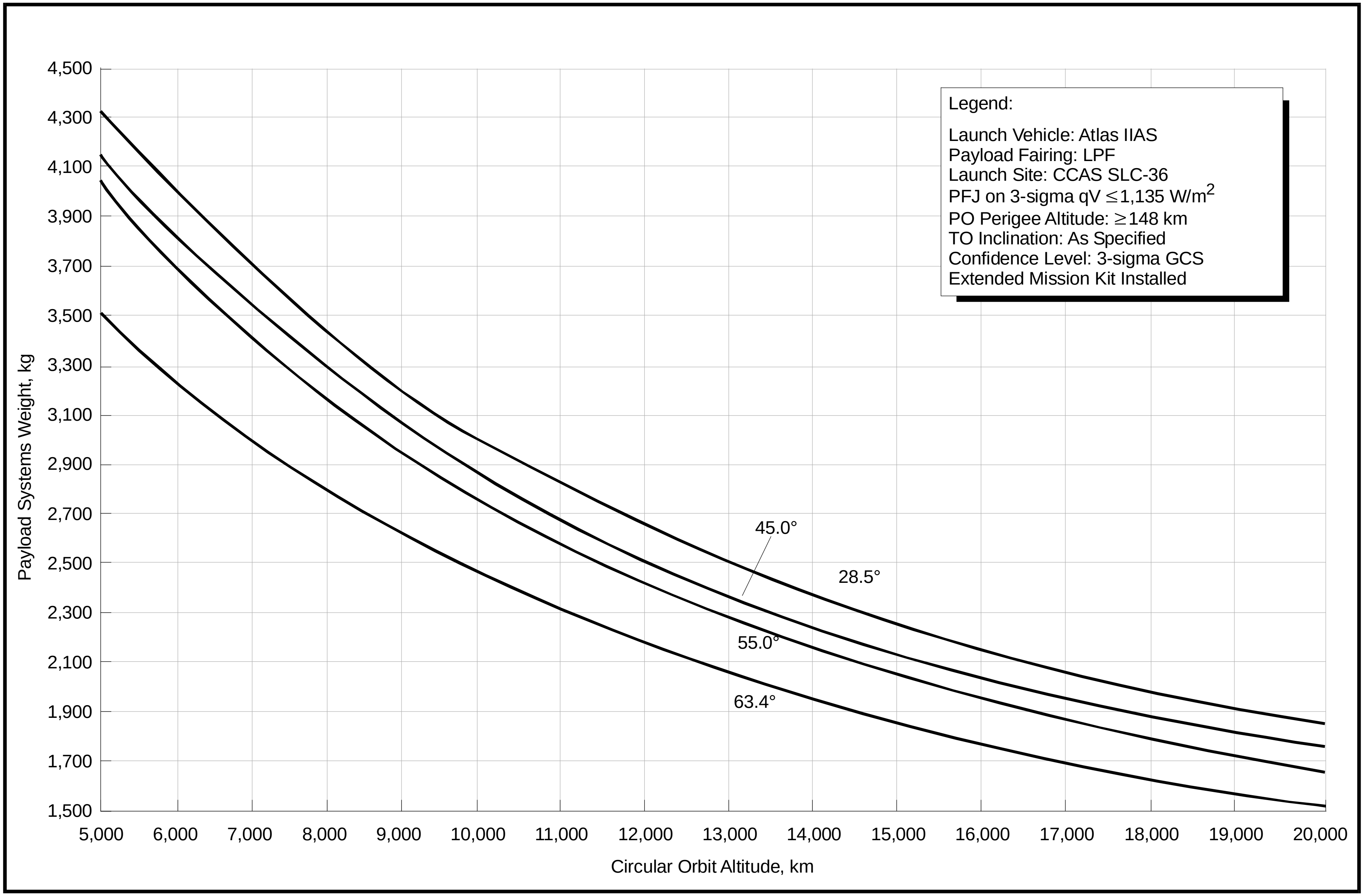


Figure 2.8-8a Atlas IIAS CCAS Intermediate Circular Orbit Performance (Metric)

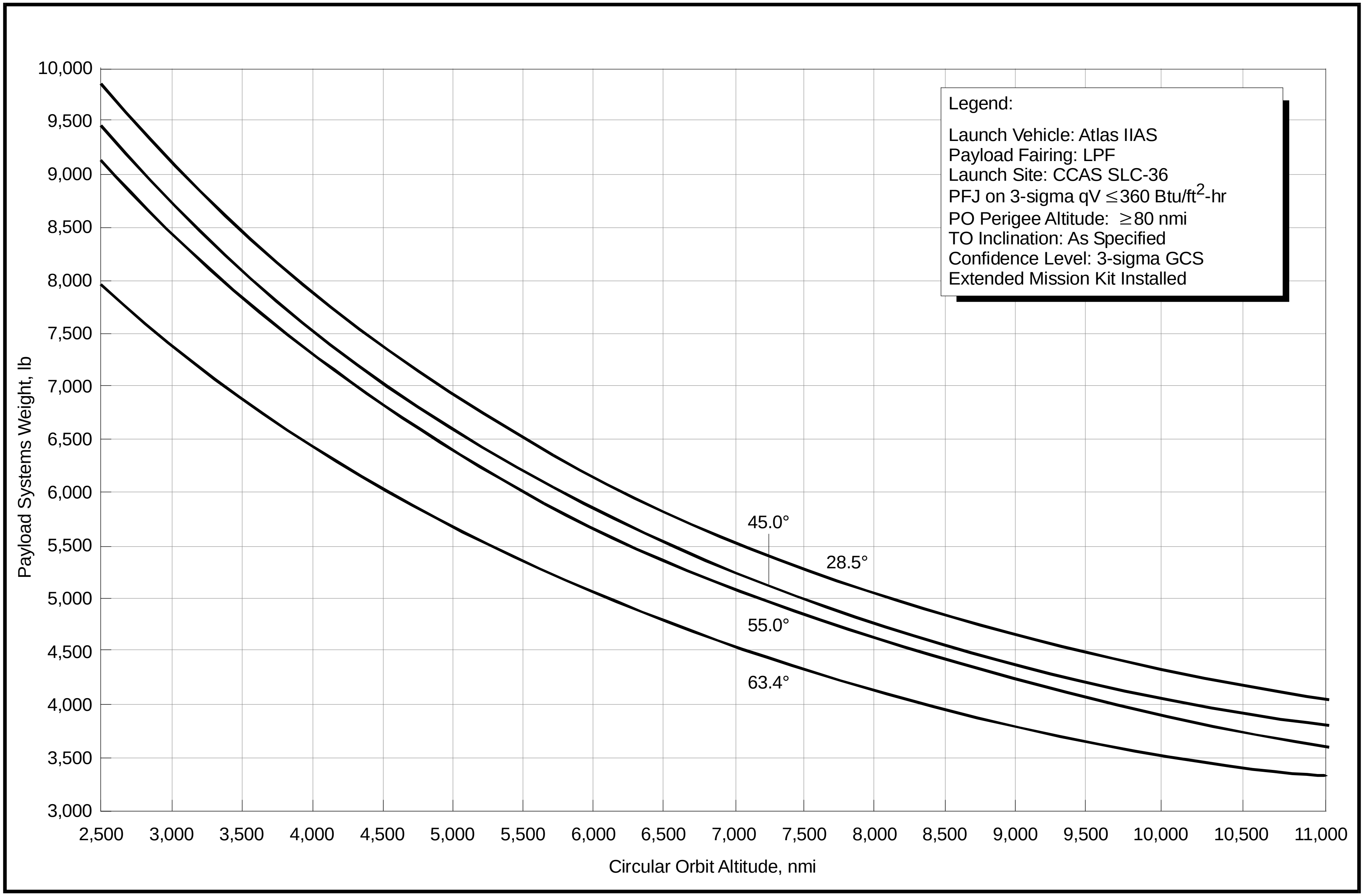


Figure 2.8-8b Atlas IIAS CCAS Intermediate Circular Orbit Performance (English)

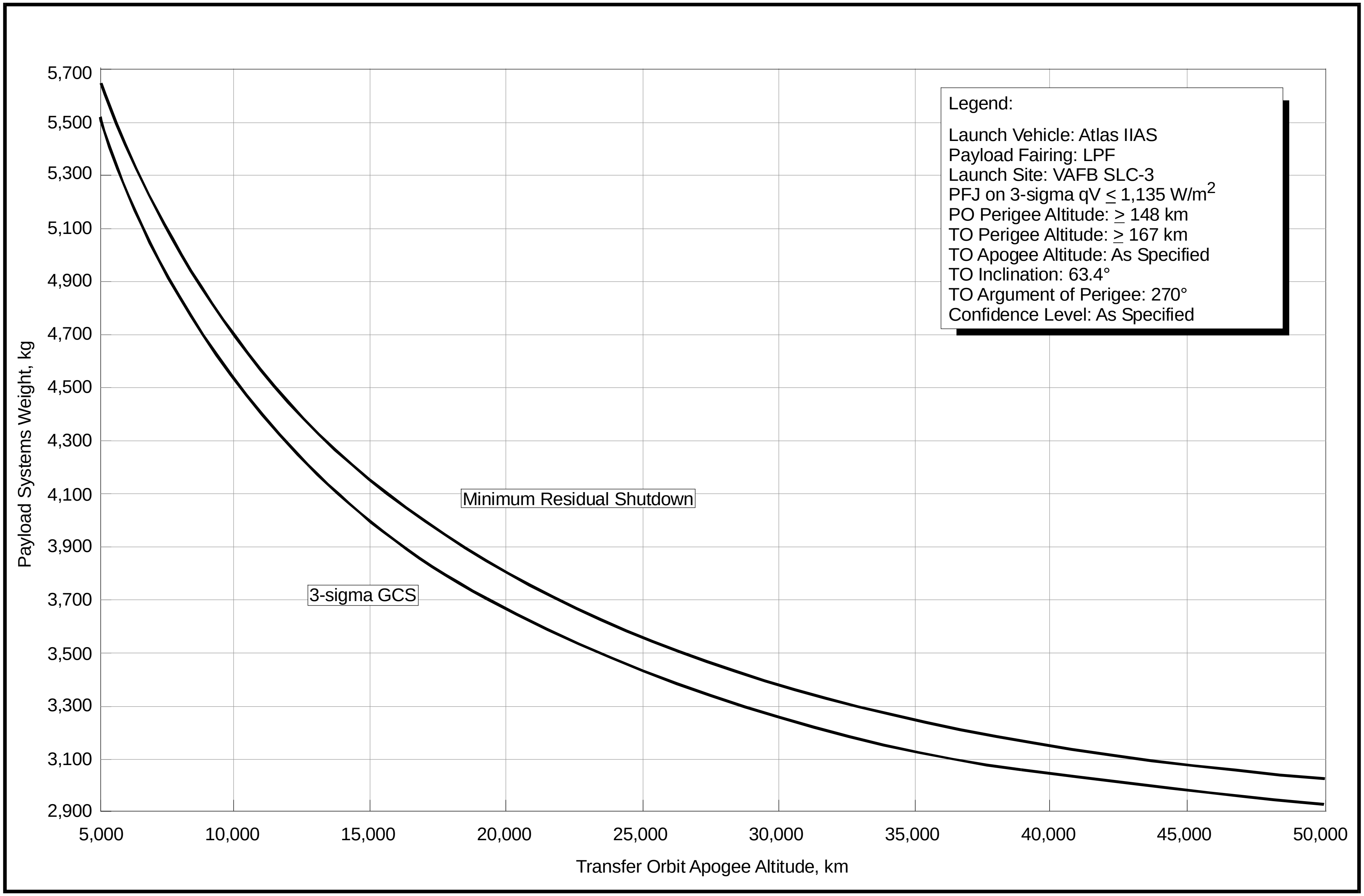


Figure 2.8-9a Atlas IIAS VAFB Elliptical Orbit Performance (Metric)

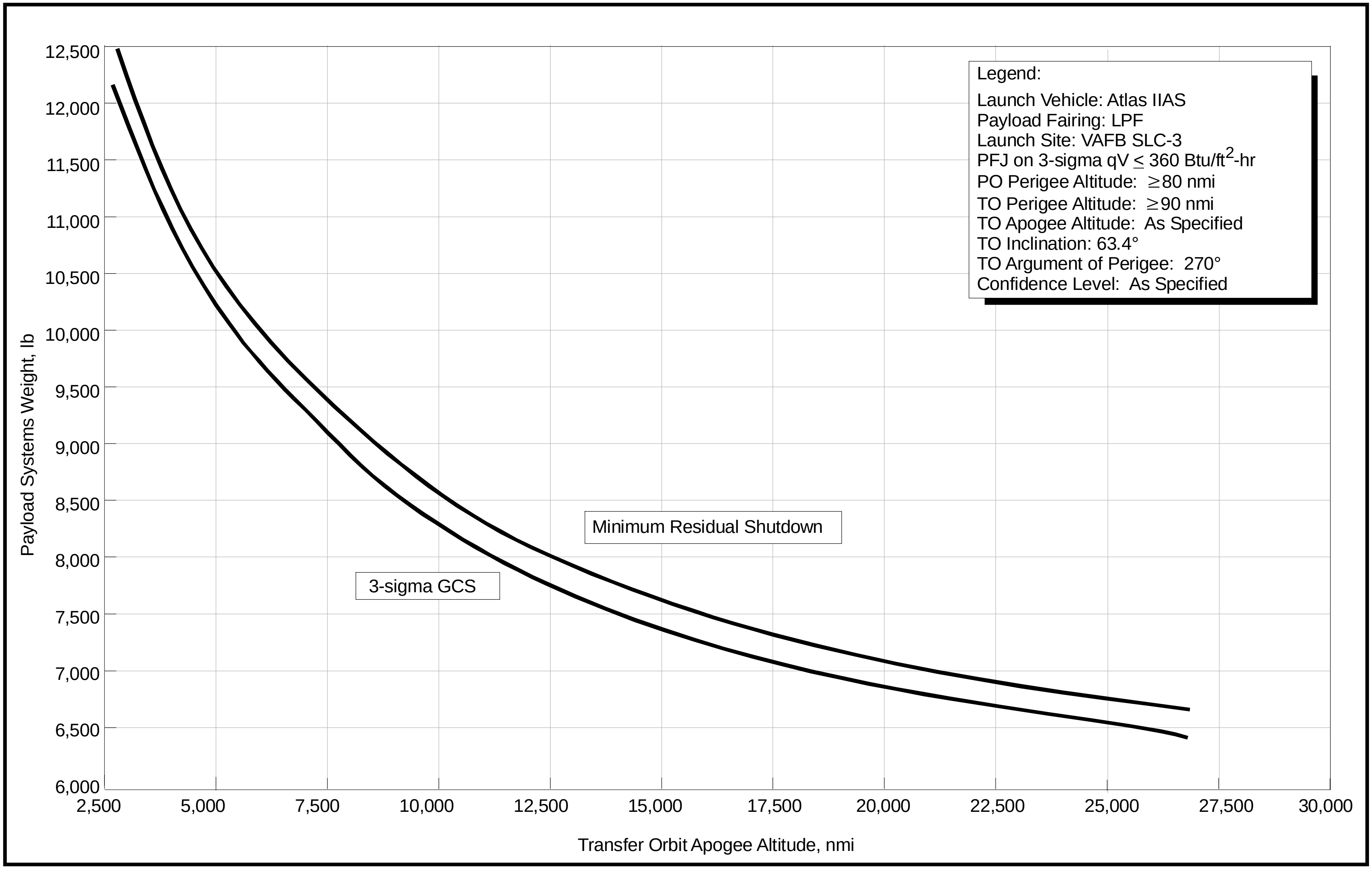


Figure 2.8-9b Atlas IIAS VAFB Elliptical Orbit Performance (English)

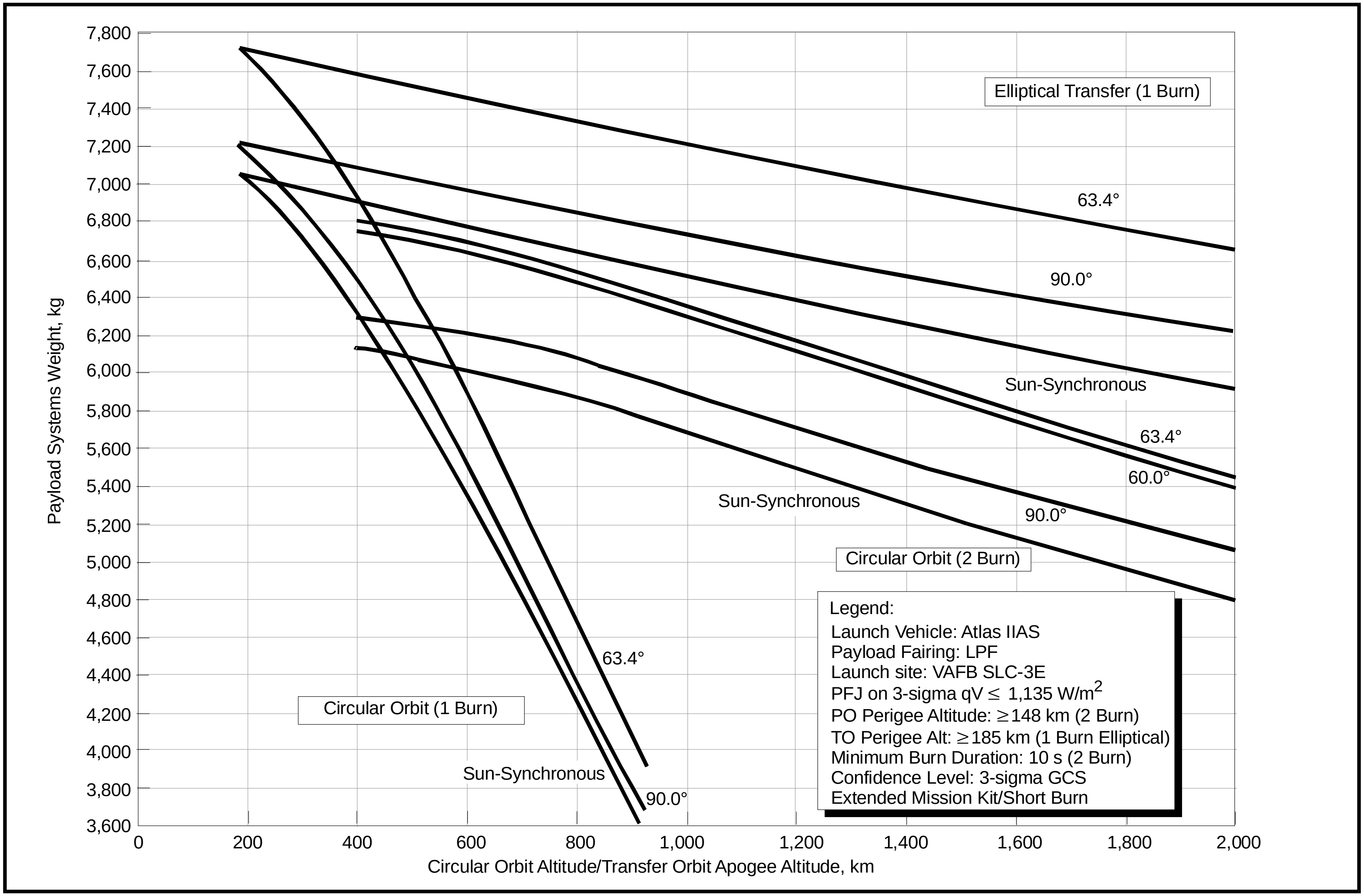


Figure 2.8-10a Atlas IIAS VAFB Performance to Low-Earth Orbit (Metric)

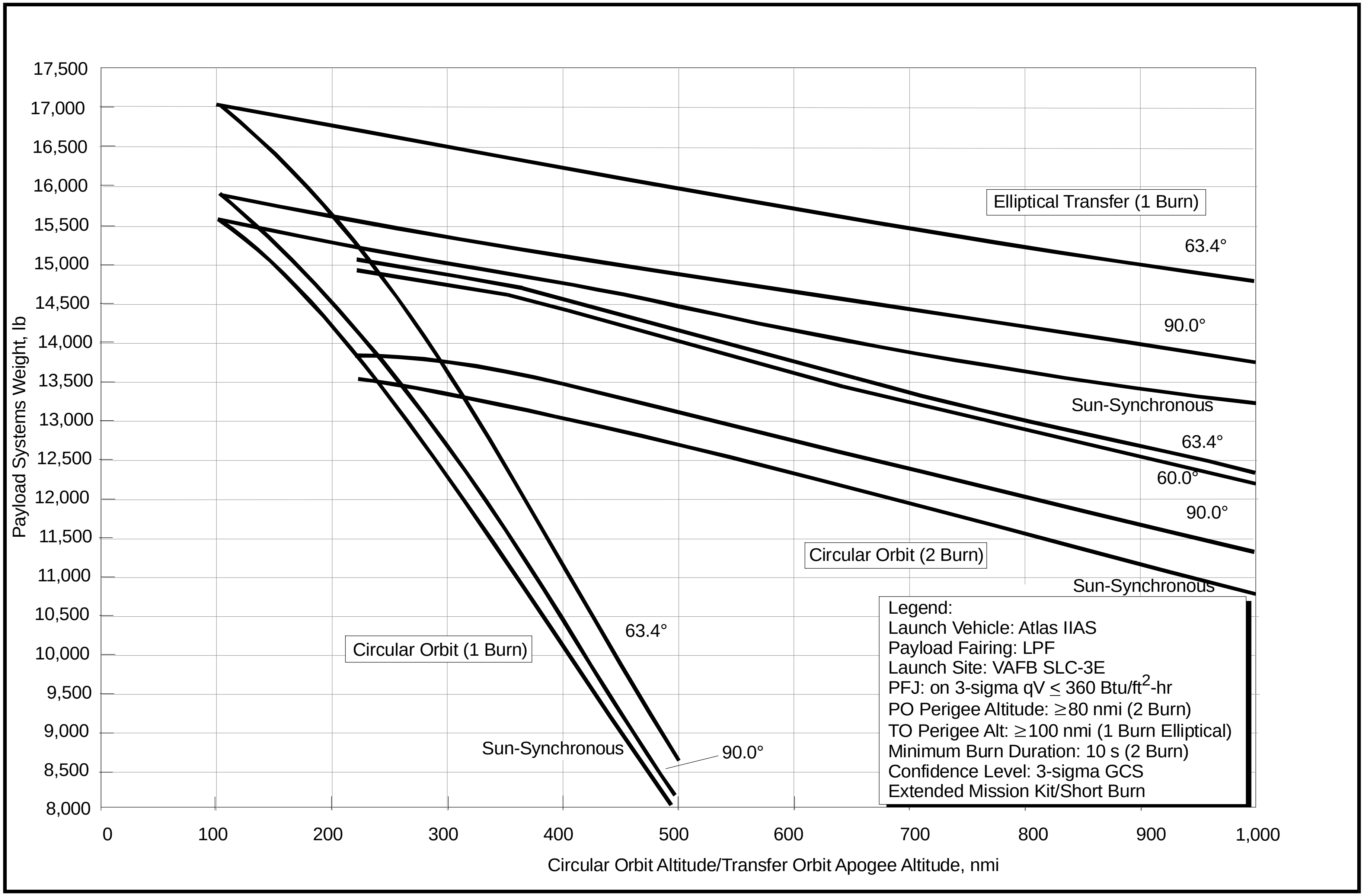


Figure 2.8-10b Atlas IIAS VAFB Performance to Low-Earth Orbit (English)

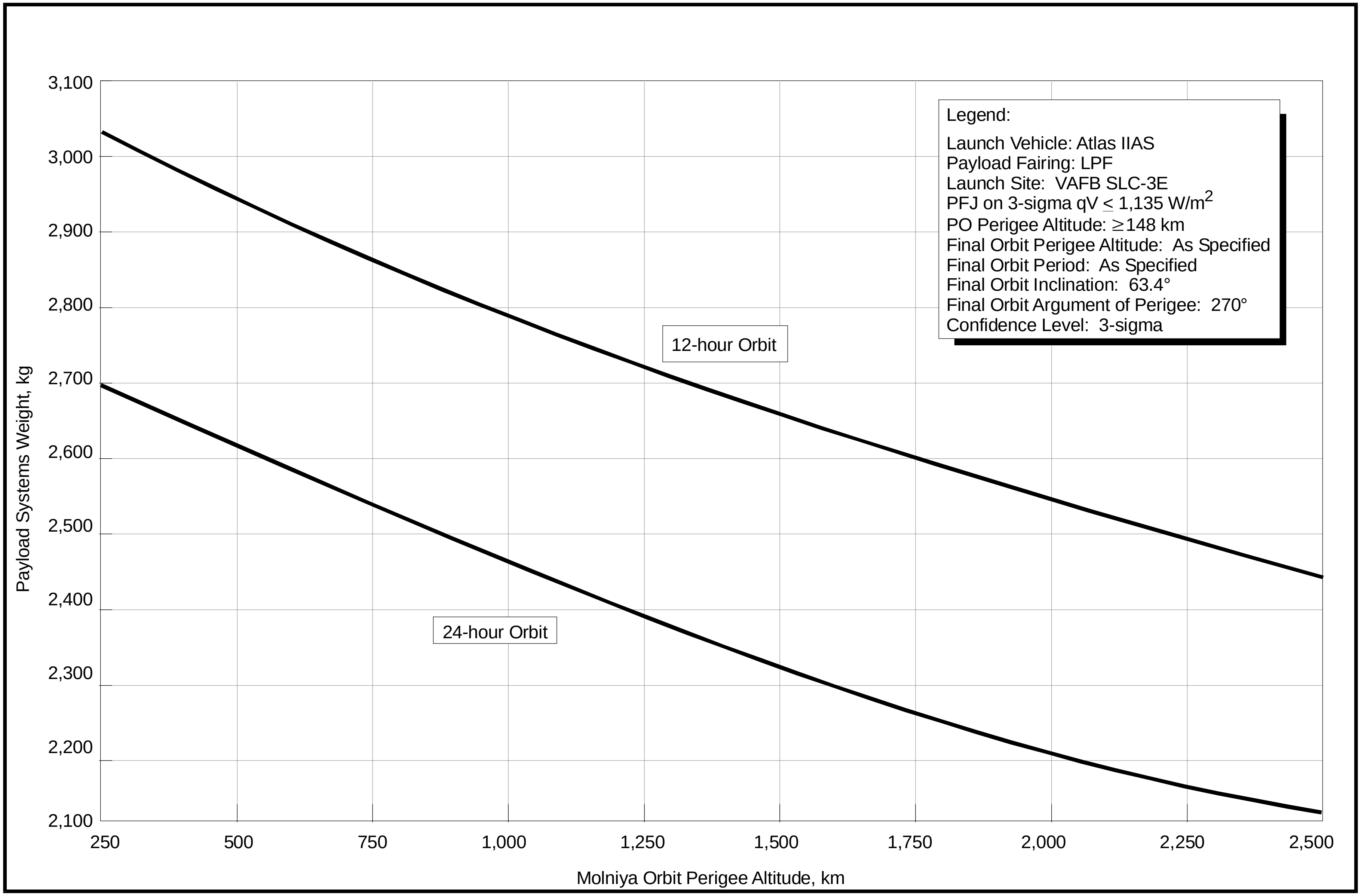


Figure 2.8-11a Atlas IIAS VAFB High-Inclination, High-Eccentricity Orbit Performance (Metric)

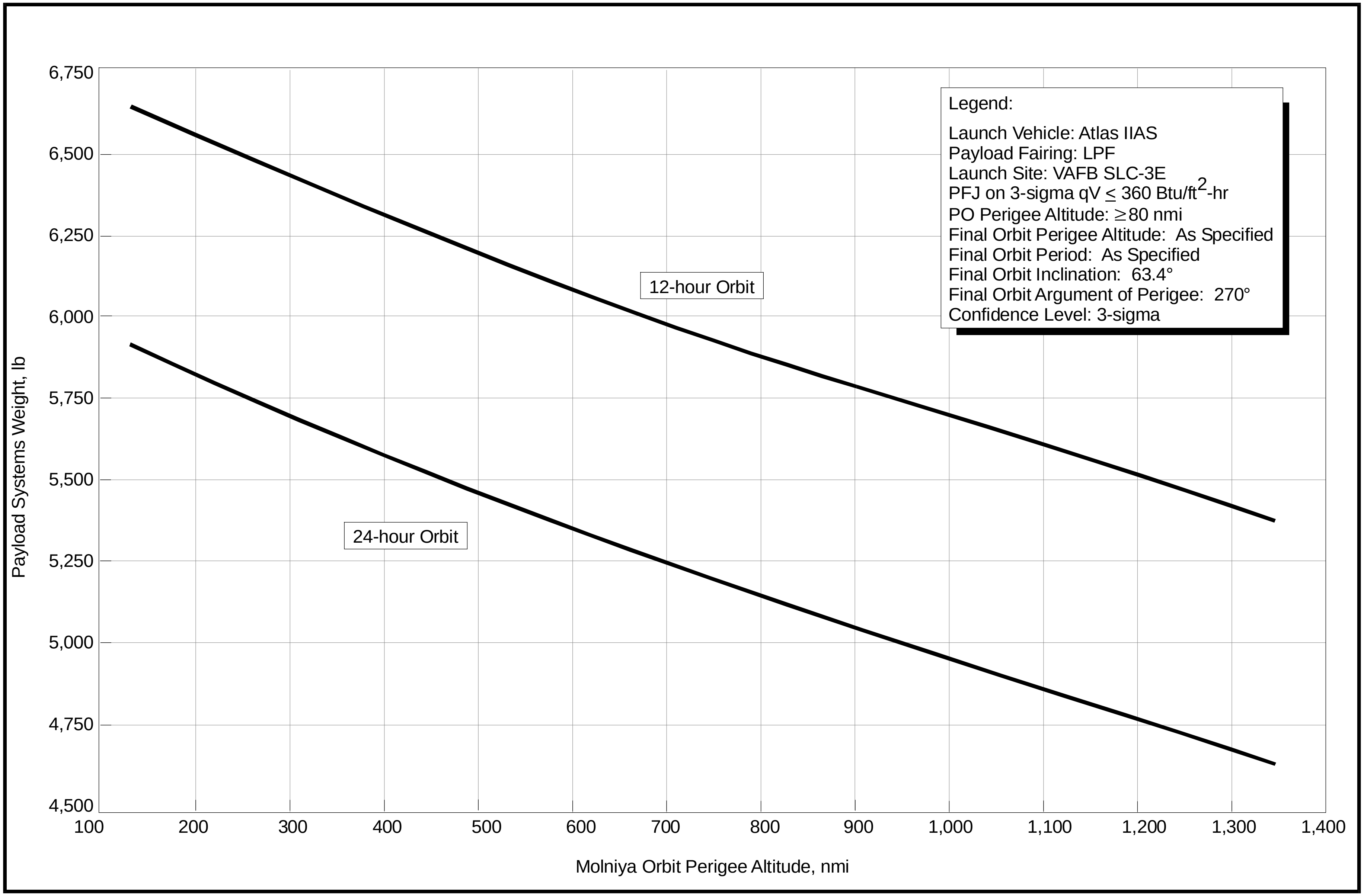


Figure 2.8-11b Atlas IIAS VAFB High-Inclination, High-Eccentricity Orbit Performance (English)

Table 2.8-1 Atlas IIAS Elliptical Transfer Orbit Performance—PSW vs Apogee Altitude

Payload Systems Weight, kg (lb)					
Apogee		Atlas IIAS			
km	(nmi)	MRS		GCS	
150,000	(80,993.5)	3,055	(6,735)	2,987	(6,586)
140,000	(75,593.9)	3,074	(6,777)	3,006	(6,627)
130,000	(70,194.4)	3,095	(6,824)	3,027	(6,673)
120,000	(64,794.8)	3,120	(6,879)	3,052	(6,728)
110,000	(59,395.2)	3,150	(6,944)	3,081	(6,792)
100,000	(53,995.7)	3,184	(7,020)	3,114	(6,867)
95,000	(51,295.9)	3,204	(7,065)	3,134	(6,910)
90,000	(48,596.1)	3,226	(7,113)	3,156	(6,958)
85,000	(45,896.3)	3,251	(7,167)	3,180	(7,011)
80,000	(43,196.5)	3,278	(7,228)	3,207	(7,071)
75,000	(40,496.8)	3,308	(7,294)	3,237	(7,136)
70,000	(37,797.0)	3,343	(7,371)	3,271	(7,212)
65,000	(35,097.2)	3,383	(7,458)	3,310	(7,297)
60,000	(32,397.4)	3,428	(7,559)	3,355	(7,396)
55,000	(29,697.6)	3,481	(7,674)	3,406	(7,510)
52,500	(28,347.7)	3,511	(7,740)	3,436	(7,575)
50,000	(26,997.8)	3,543	(7,811)	3,467	(7,644)
47,500	(25,647.9)	3,578	(7,889)	3,502	(7,721)
45,000	(24,298.0)	3,617	(7,974)	3,540	(7,805)
42,500	(22,948.2)	3,660	(8,069)	3,582	(7,898)
40,000	(21,598.3)	3,707	(8,173)	3,629	(7,000)
37,500	(20,248.4)	3,759	(8,288)	3,680	(8,114)
35,788	(19,324.0)	3,799	(8,376)	3,719	(8,200)
35,000	(18,898.5)	3,818	(8,418)	3,738	(8,241)
32,500	(17,548.6)	3,885	(8,565)	3,803	(8,386)
30,000	(16,198.7)	3,960	(8,731)	3,877	(8,549)
27,500	(14,848.8)	4,046	(8,921)	3,962	(8,736)
25,000	(13,498.9)	4,147	(9,142)	4,061	(8,954)
22,500	(12,149.0)	4,263	(9,400)	4,176	(9,207)
20,000	(10,799.1)	4,402	(9,706)	4,313	(9,508)
17,500	(9,449.2)	4,570	(10,075)	4,477	(9,872)
15,000	(8,099.4)	4,775	(10,528)	4,680	(10,317)
12,500	(6,749.5)	5,034	(11,100)	4,934	(10,879)
11,000	(5,939.5)	5,224	(11,517)	5,121	(11,290)
10,000	(5,399.6)	5,369	(11,837)	5,263	(11,605)
9,000	(4,859.6)	5,532	(12,197)	5,424	(11,959)
8,000	(4,319.7)	5,717	(12,605)	5,606	(12,360)
7,000	(3,779.7)	5,928	(13,070)	5,813	(12,817)
6,000	(3,239.7)	6,172	(13,607)	6,053	(13,346)
5,000	(2,699.8)	6,455	(14,232)	6,332	(13,961)

Note: Large (4.2-m) Diameter Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude = 148.2 km (80 nmi); Transfer Orbit Perigee Altitude = 166.7 (90 nmi); Transfer Orbit Inclination = 27.0°; Argument of Perigee = 180°

Table 2.8-2 Atlas IIAS Performance—PSW vs Transfer Orbit Inclination

Payload Systems Weight, kg (lb)				
Inclination, deg	Atlas IIAS			
	MRS		GCS	
18.00	3,188	(7,029)	3,118	(6,875)
18.50	3,240	(7,143)	3,169	(6,987)
19.00	3,290	(7,254)	3,219	(7,096)
19.50	3,339	(7,361)	3,266	(7,202)
20.00	3,386	(7,465)	3,313	(7,304)
20.50	3,431	(7,565)	3,357	(7,402)
21.00	3,474	(7,659)	3,399	(7,495)
21.50	3,515	(7,750)	3,440	(7,585)
22.00	3,554	(7,836)	3,479	(7,669)
22.50	3,591	(7,918)	3,515	(7,749)
23.00	3,625	(7,992)	3,548	(7,823)
23.50	3,657	(8,063)	3,579	(7,892)
24.00	3,686	(8,126)	3,608	(7,954)
24.50	3,712	(8,184)	3,634	(8,012)
25.00	3,735	(8,236)	3,657	(8,062)
25.50	3,756	(8,280)	3,676	(8,106)
26.00	3,773	(8,319)	3,694	(8,144)
26.50	3,788	(8,351)	3,708	(8,175)
27.00	3,799	(8,376)	3,719	(8,200)
27.50	3,807	(8,393)	3,727	(8,217)
28.00	3,812	(8,404)	3,732	(8,228)
28.50	3,813	(8,408)	3,733	(8,231)
29.00	3,813	(8,407)	3,733	(8,230)
29.50	3,811	(8,403)	3,731	(8,227)
30.00	3,809	(8,399)	3,729	(8,223)

Note: Large (4.2-m) Diameter Payload Fairing Jettison at 3-Sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude = 148.2 km (80 nmi); Transfer Orbit Perigee Altitude = 166.7 (90 nmi); Transfer Orbit Apogee Altitude = 35,788 km (19,324 nmi); Argument of Perigee = 180°

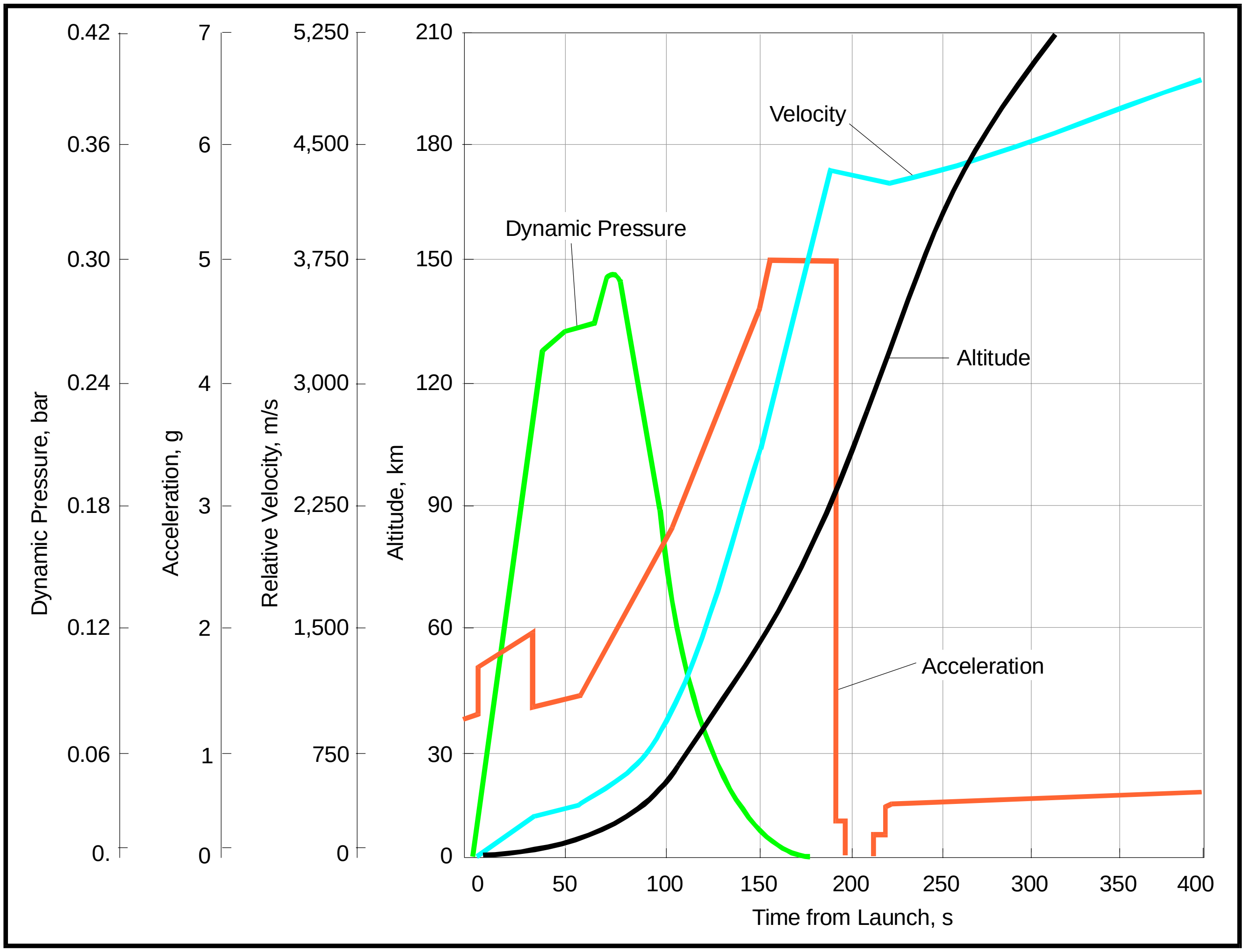


Figure 2.9-1 Atlas IIIA Nominal Ascent Data

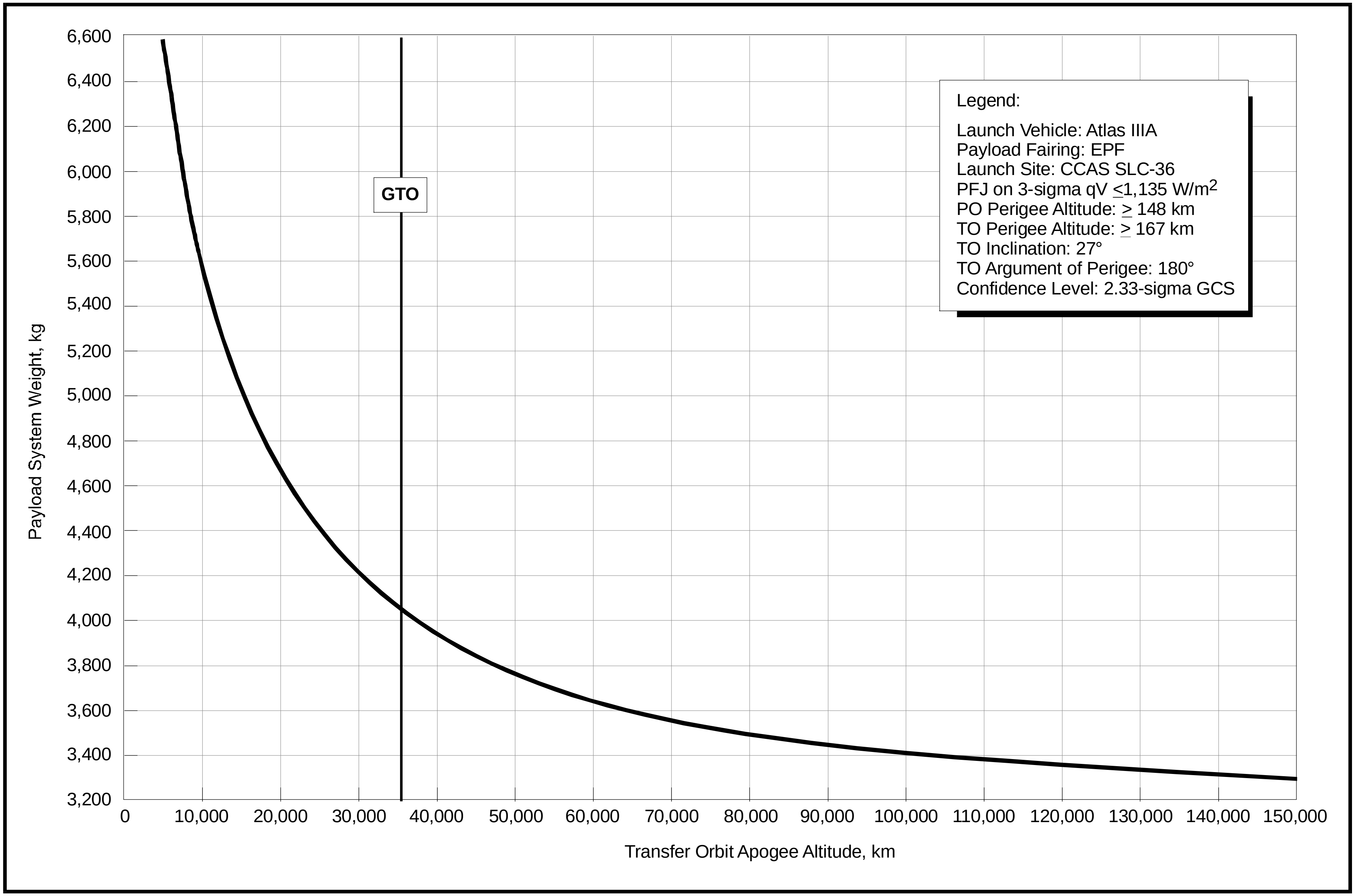


Figure 2.9-2a Atlas IIIA CCAS Performance to Elliptical Transfer Orbit (GCS-Metric)

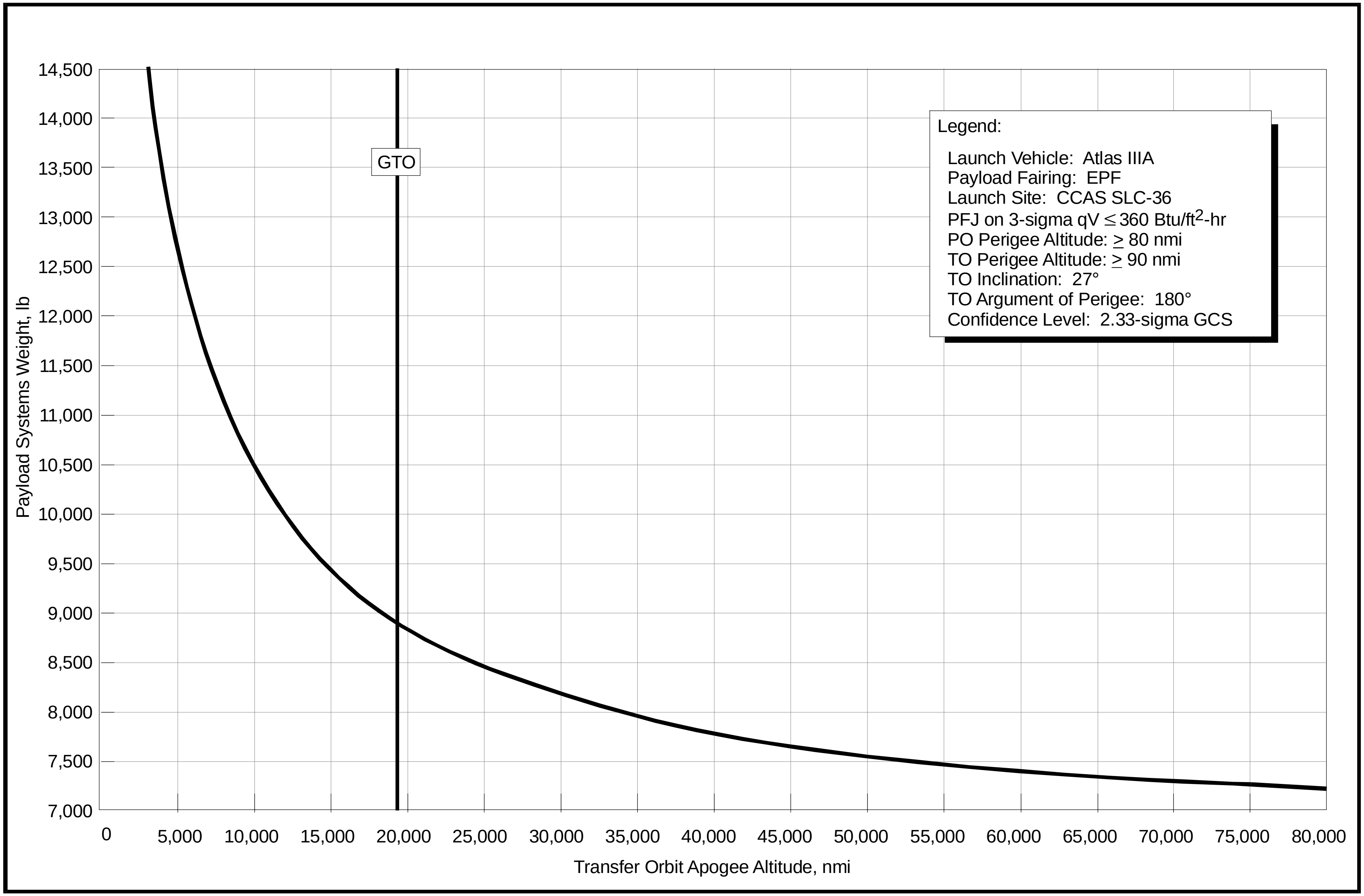


Figure 2.9-2b Atlas IIIA CCAS Performance to Elliptical Transfer Orbit (GCS-English)

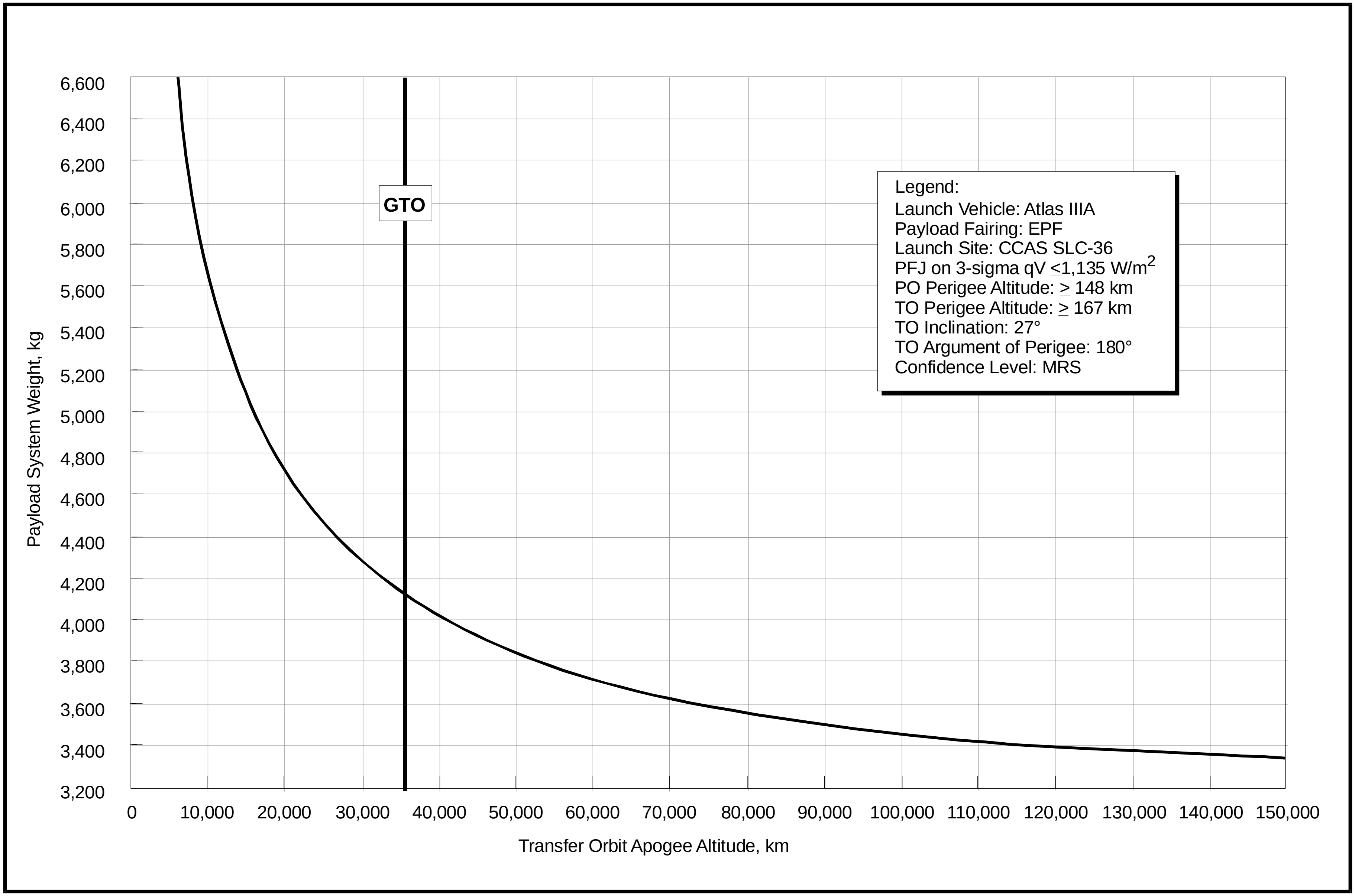


Figure 2.9-3a Atlas IIIA CCAS Performance to Elliptical Transfer Orbit (MRS-Metric)

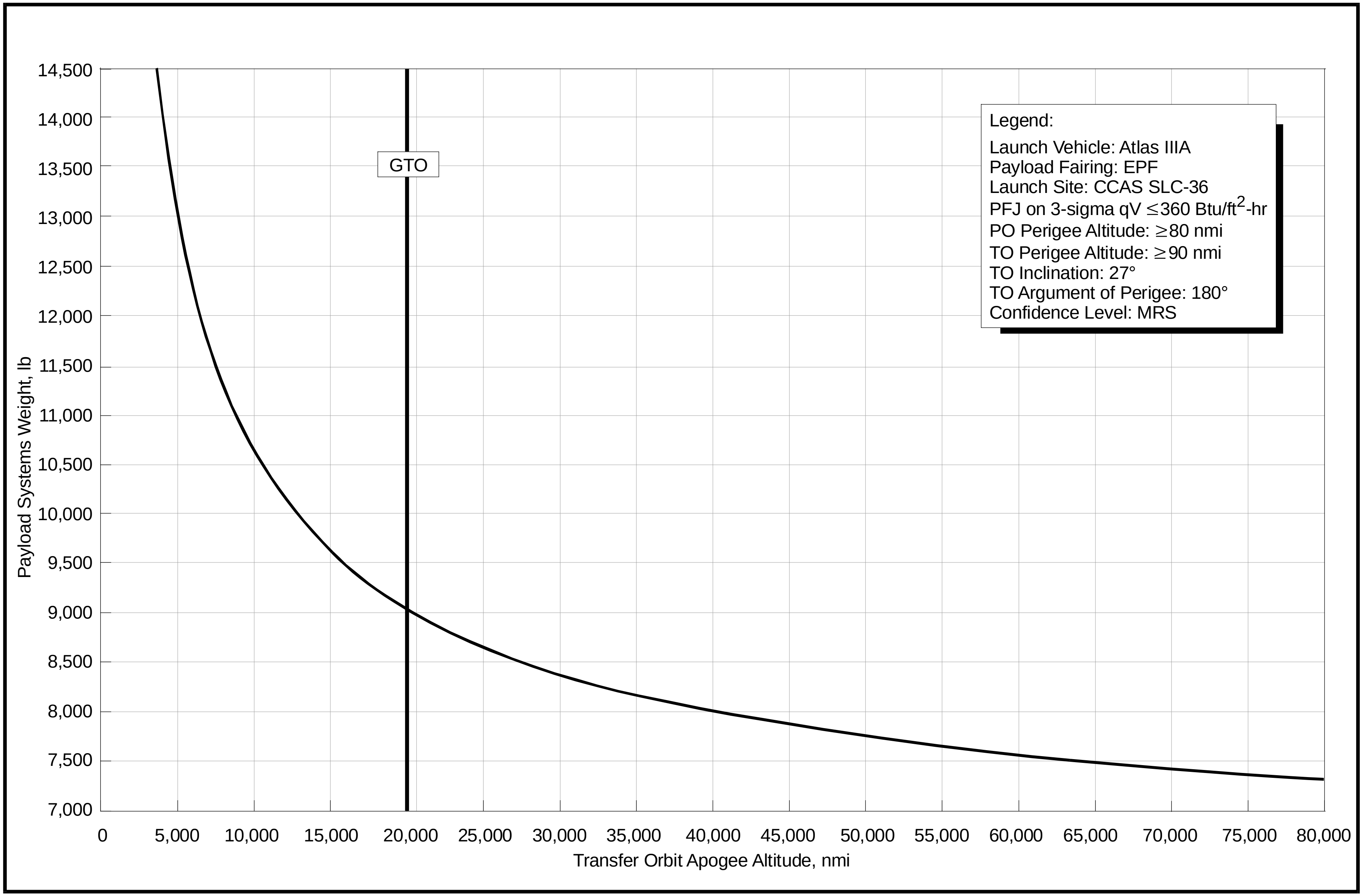


Figure 2.9-3b Atlas IIIA CCAS Performance to Elliptical Transfer Orbit (MRS-English)

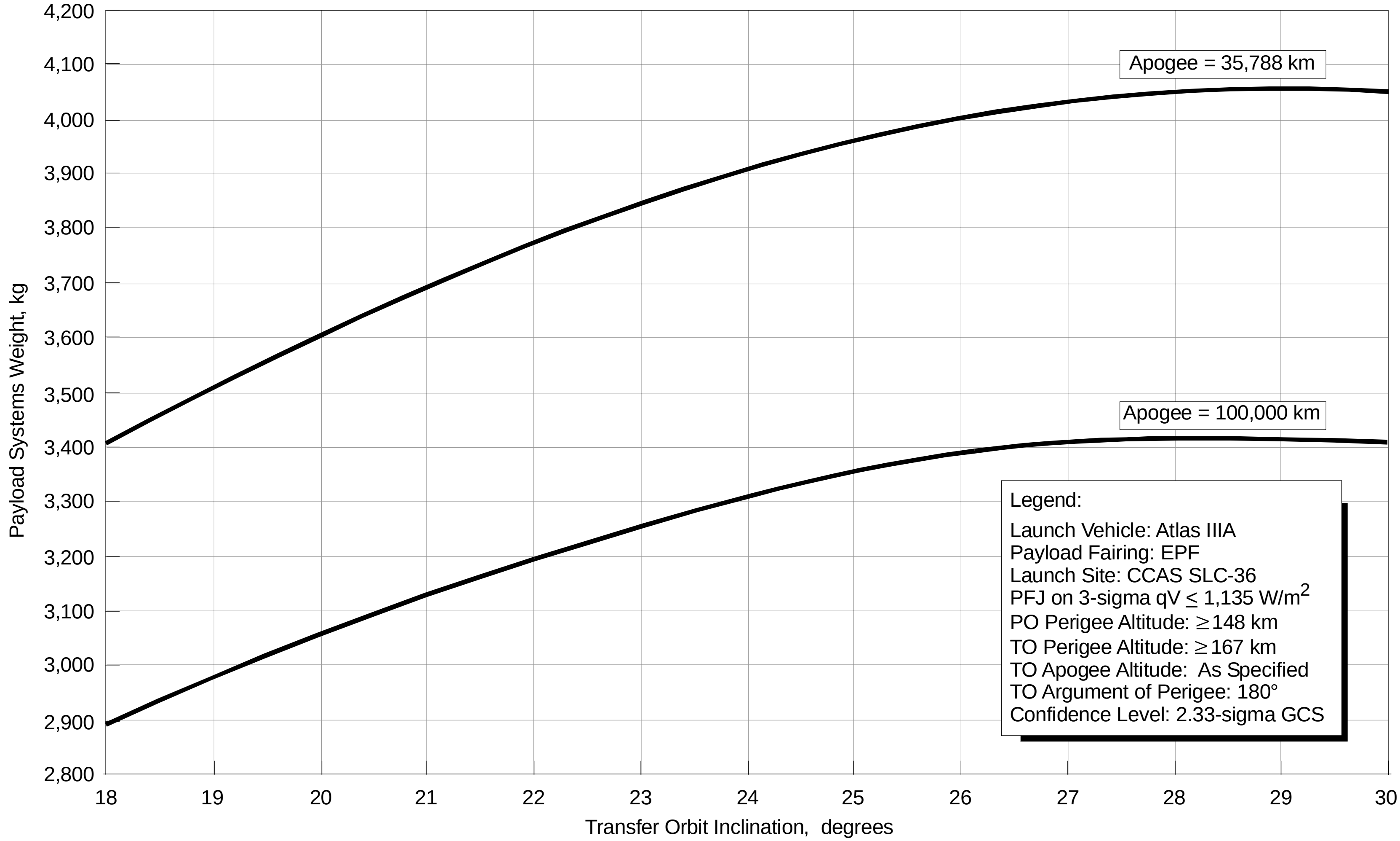


Figure 2.9-4a Atlas IIIA CCAS Reduced Inclination Elliptical Orbit Performance (GCS-Metric)

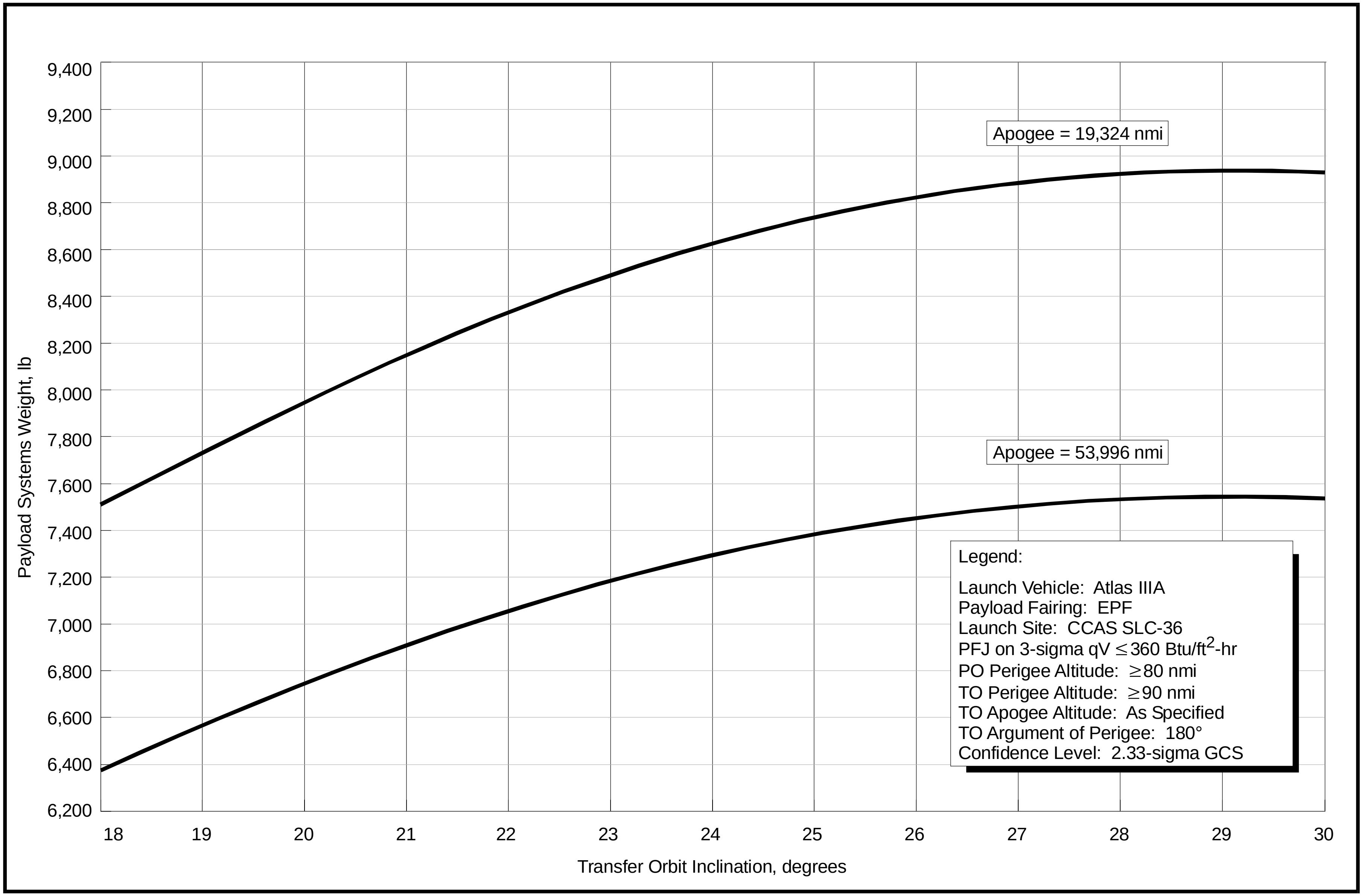


Figure 2.9-4b Atlas IIIA CCAS Reduced Inclination Elliptical Orbit Performance (GCS-English)

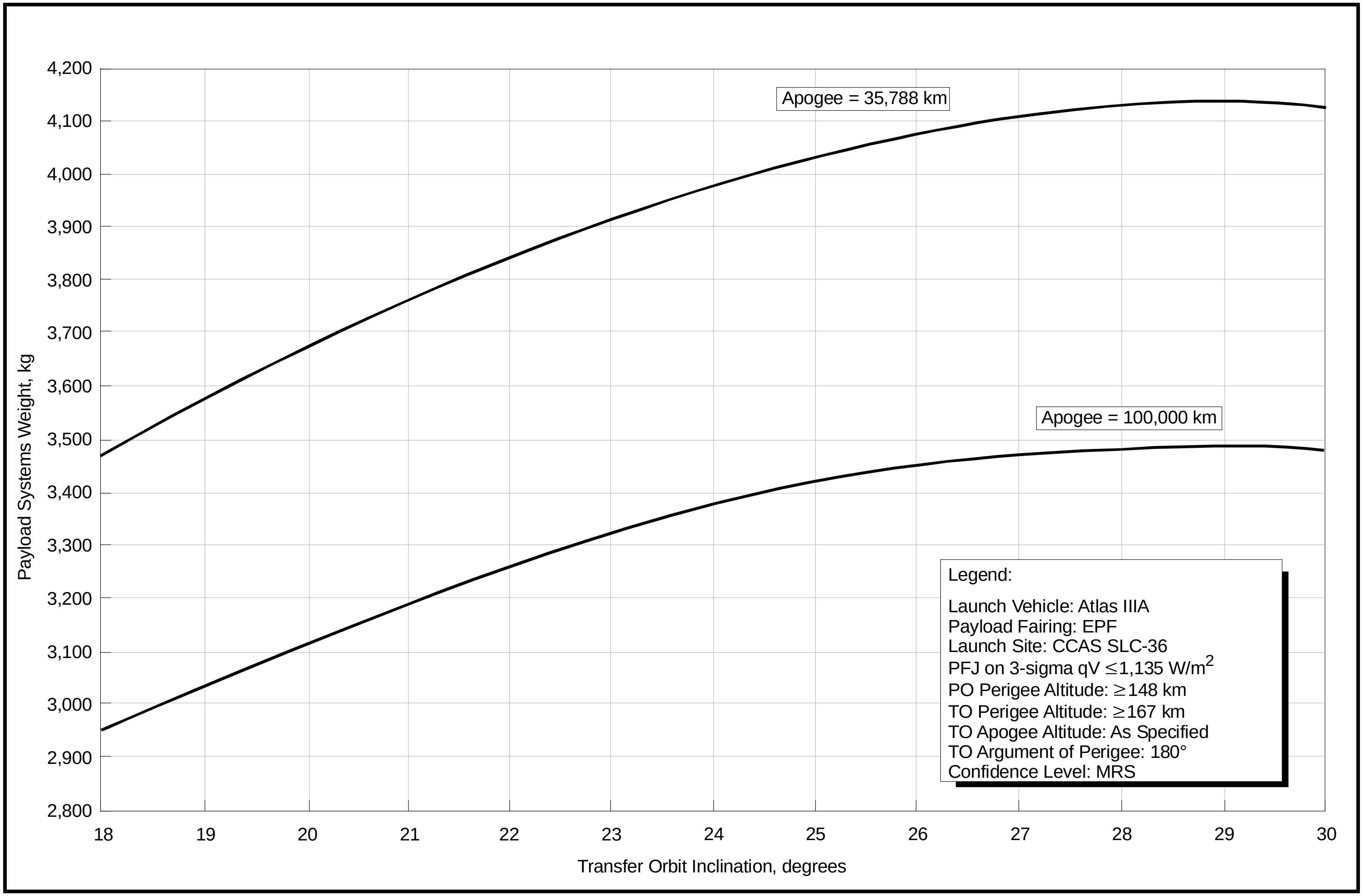


Figure 2.9-5a Atlas IIIA CCAS Reduced Inclination Elliptical Orbit Performance (MRS-Metric)

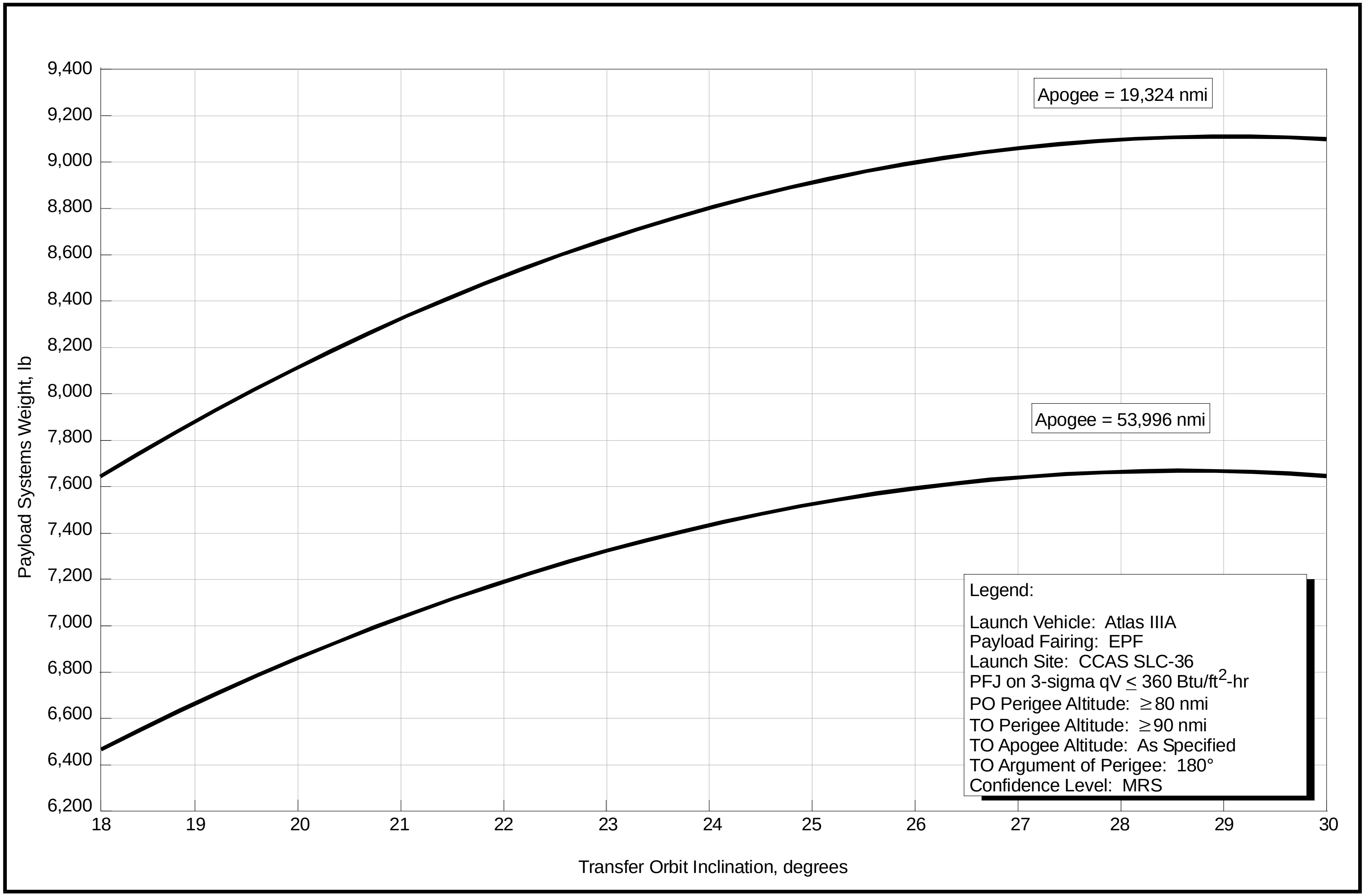


Figure 2.9-5b Atlas IIIA CCAS Reduced Inclination Elliptical Orbit Performance (MRS-English)

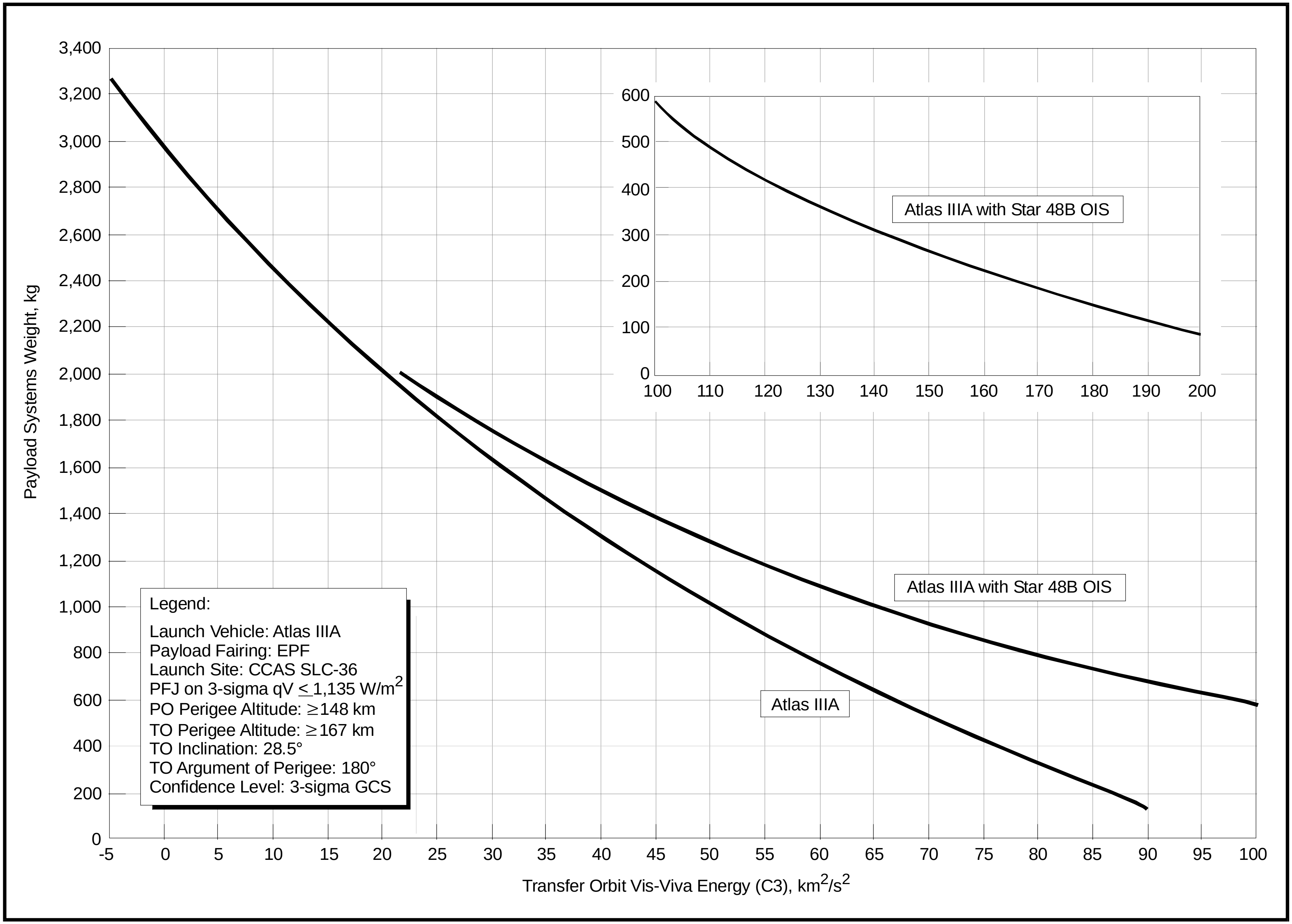


Figure 2.9-6a Atlas IIIA CCAS Earth-Escape Performance (Metric)

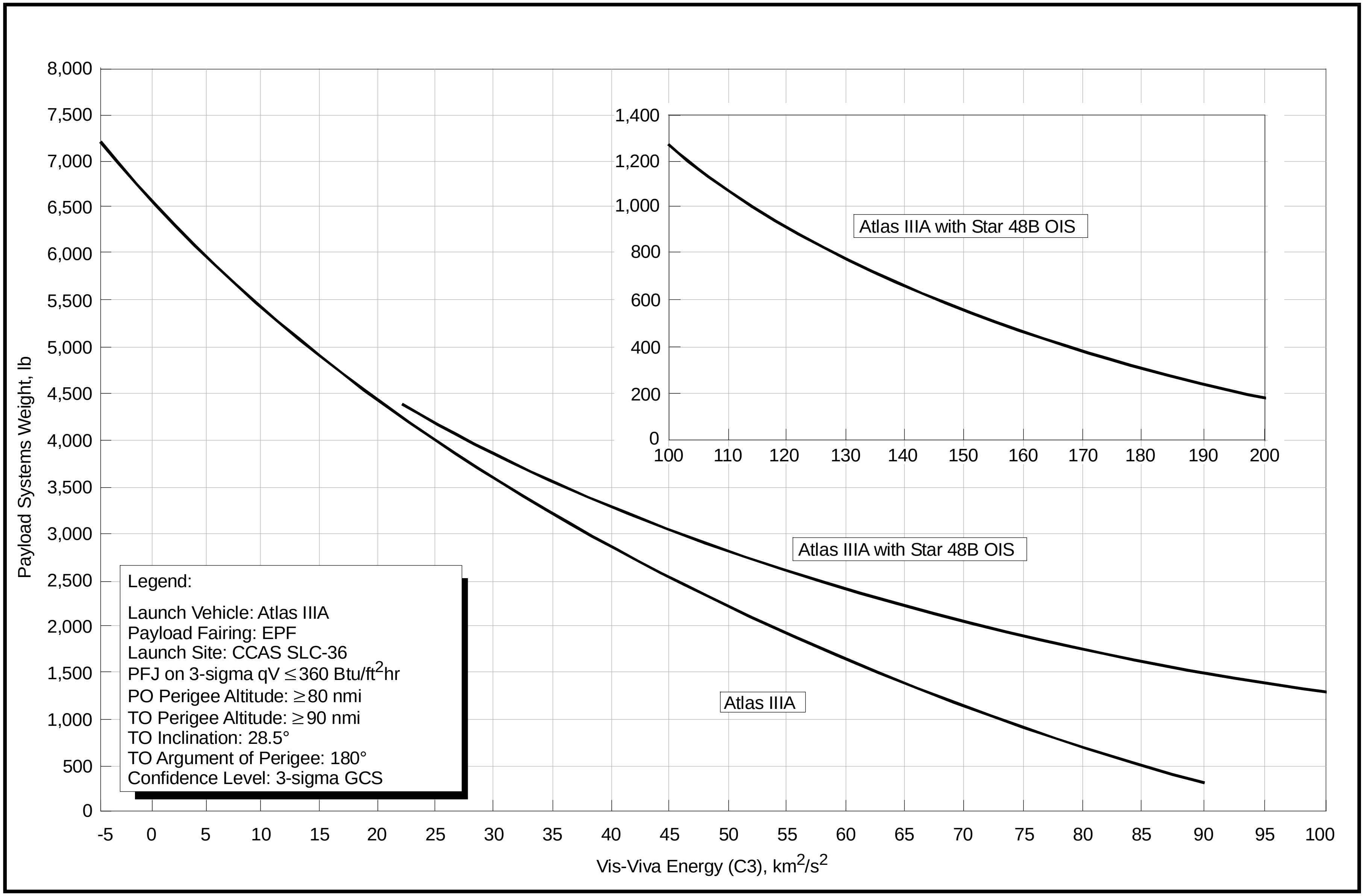


Figure 2.9-6b Atlas IIIA CCAS Earth-Escape Performance (English)

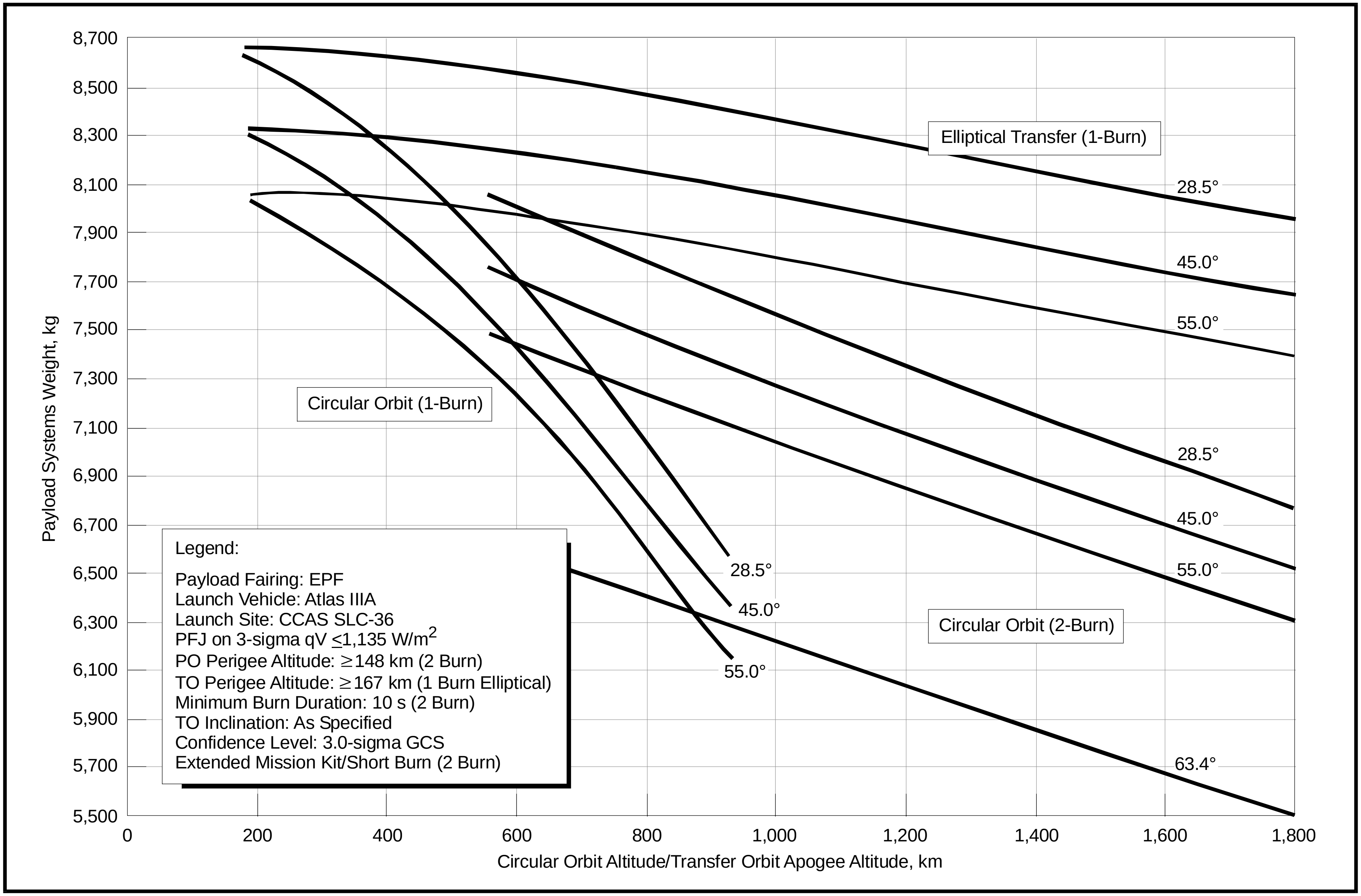


Figure 2.9-7a Atlas IIIA CCAS Low-Earth Performance (Metric)

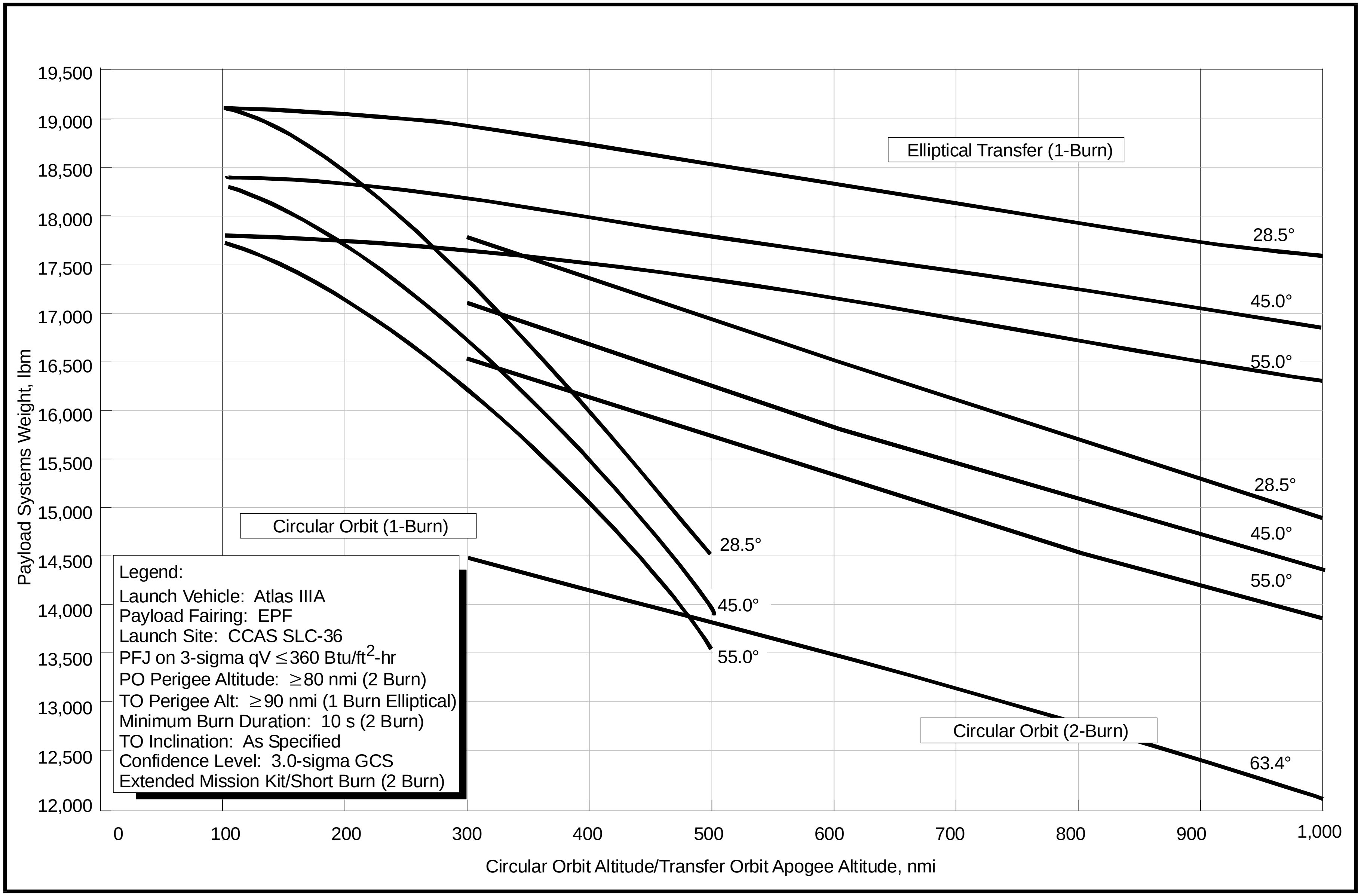


Figure 2.9-7b Atlas IIIA CCAS Low-Earth Performance (English)

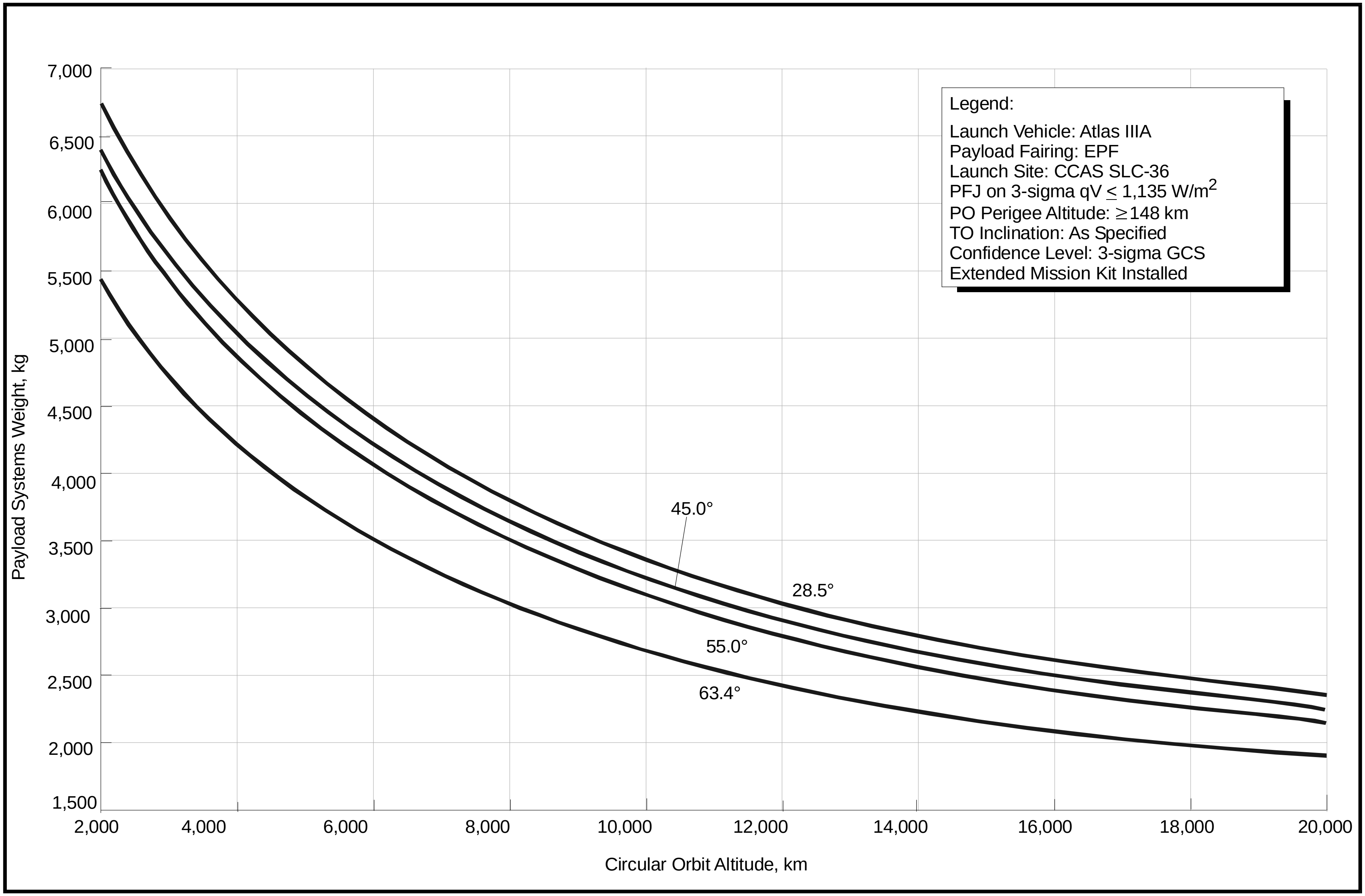


Figure 2.9-8a Atlas IIIA CCAS Intermediate Circular Orbit Performance (Metric)

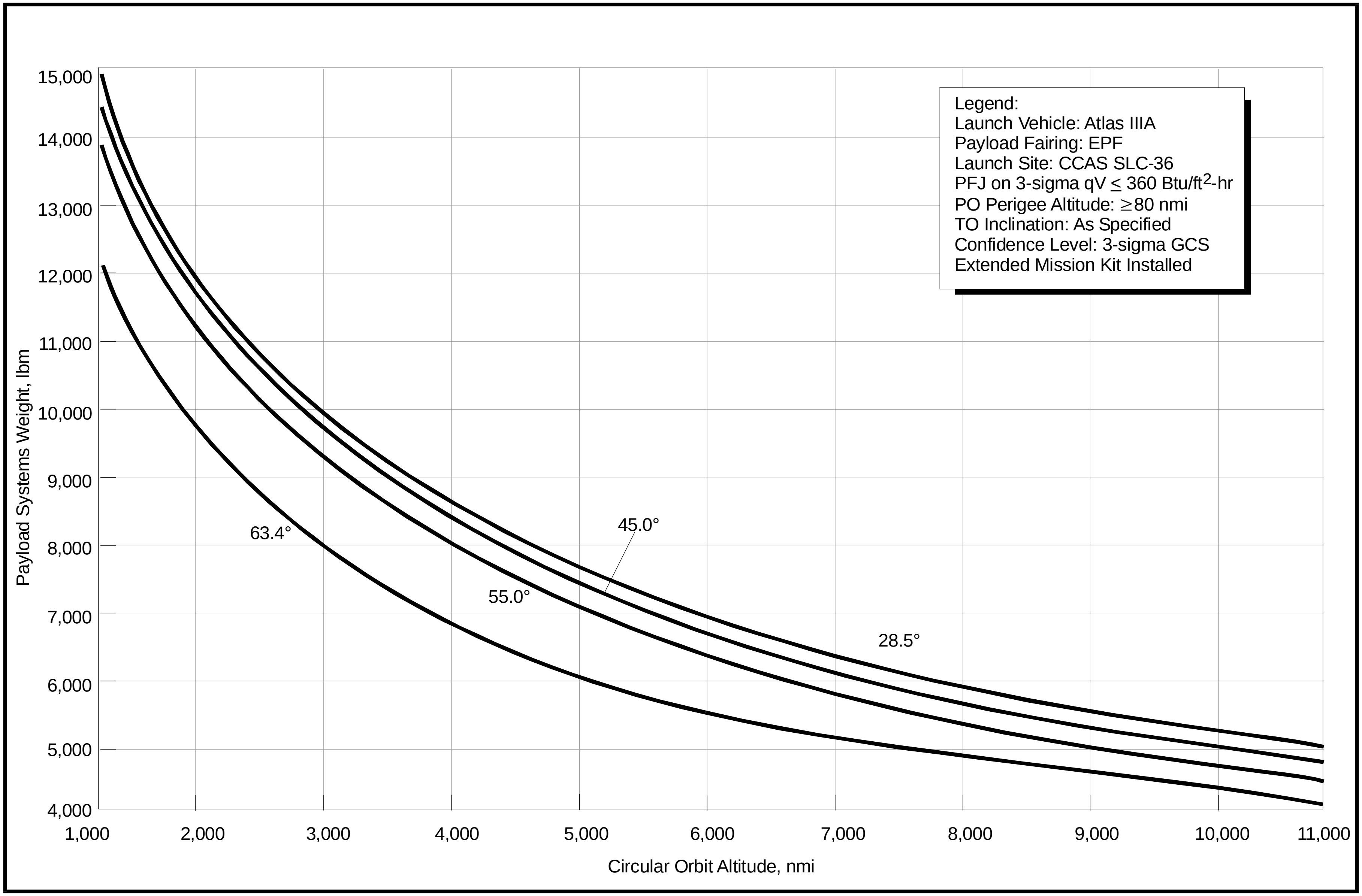


Figure 2.9-8b Atlas IIIA CCAS Intermediate Circular Orbit Performance (English)

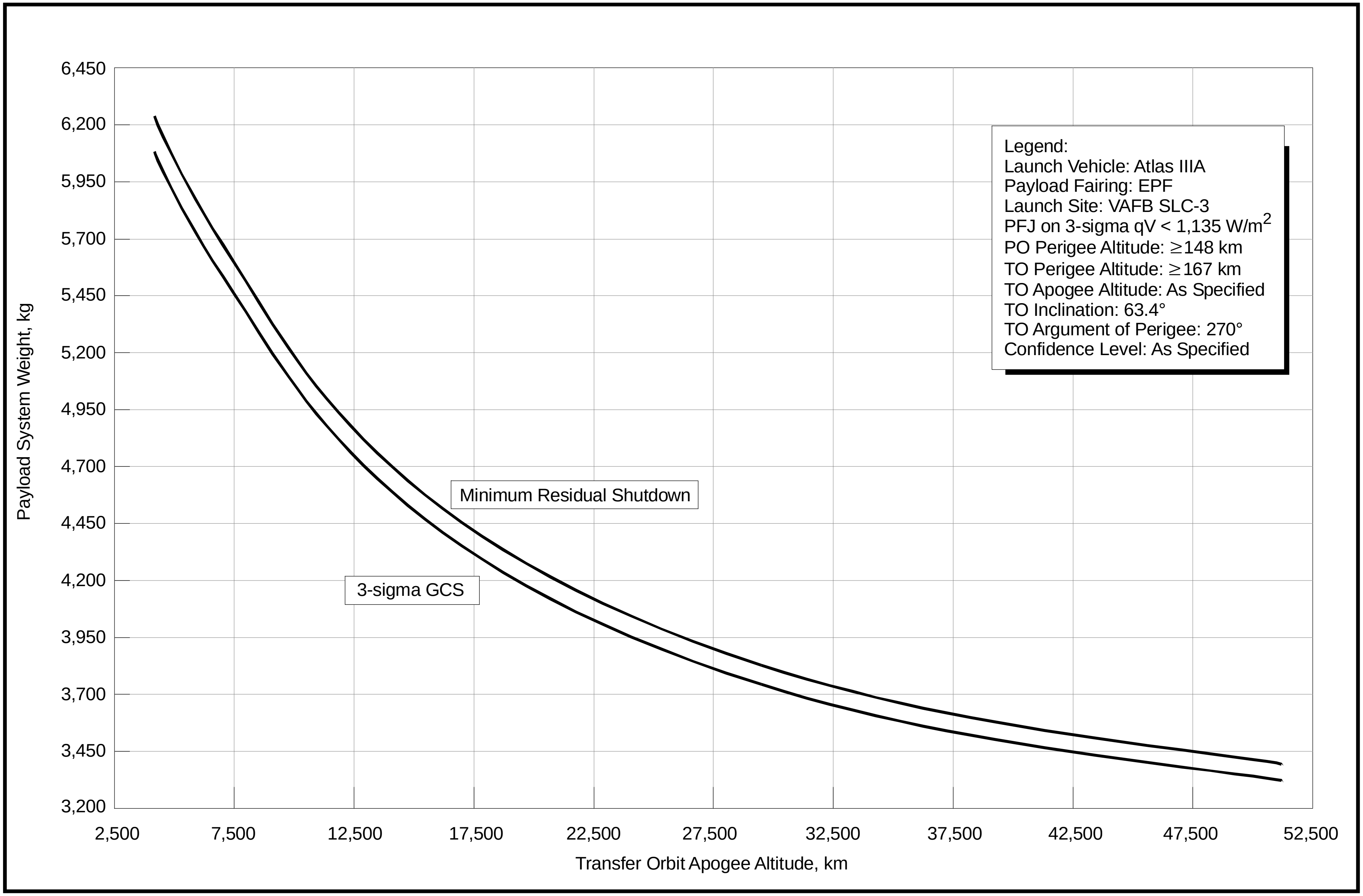


Figure 2.9-9a Atlas IIIA VAFB Elliptical Orbit Performance (Metric)

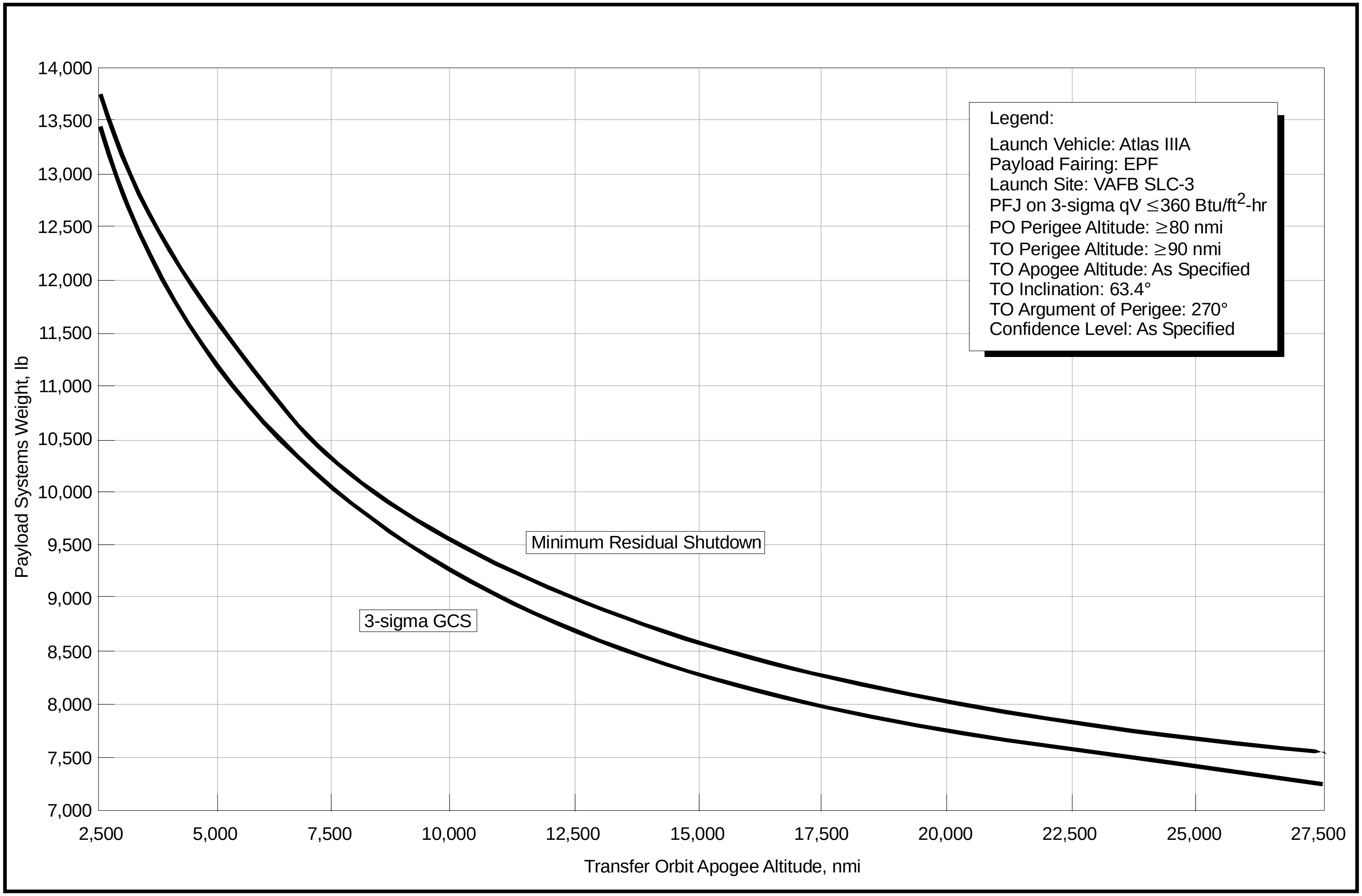


Figure 2.9-9b Atlas IIIA VAFB Elliptical Orbit Performance (English)

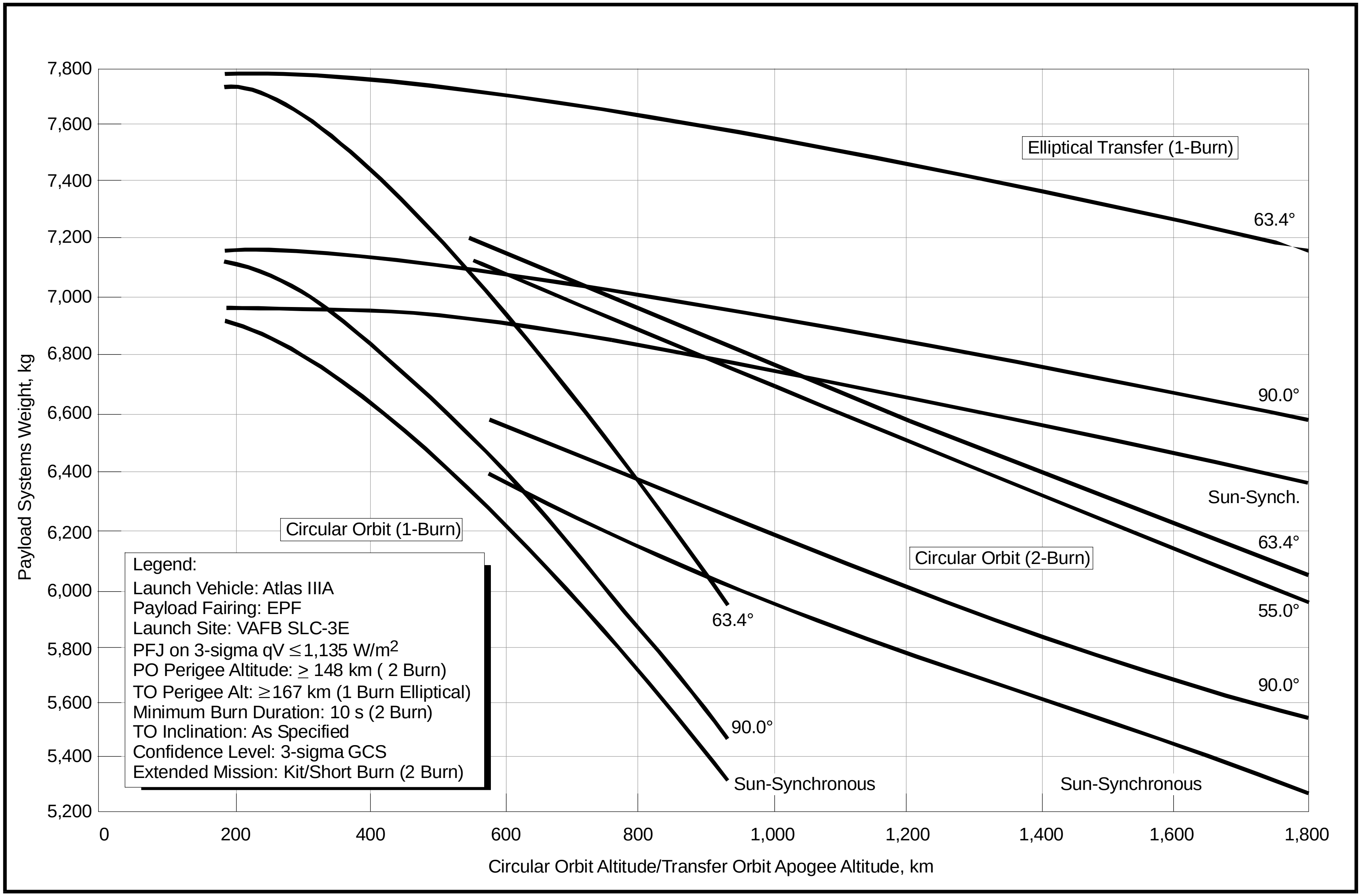


Figure 2.9-10a Atlas IIIA VAFB Performance to Low-Earth Orbit (Metric)

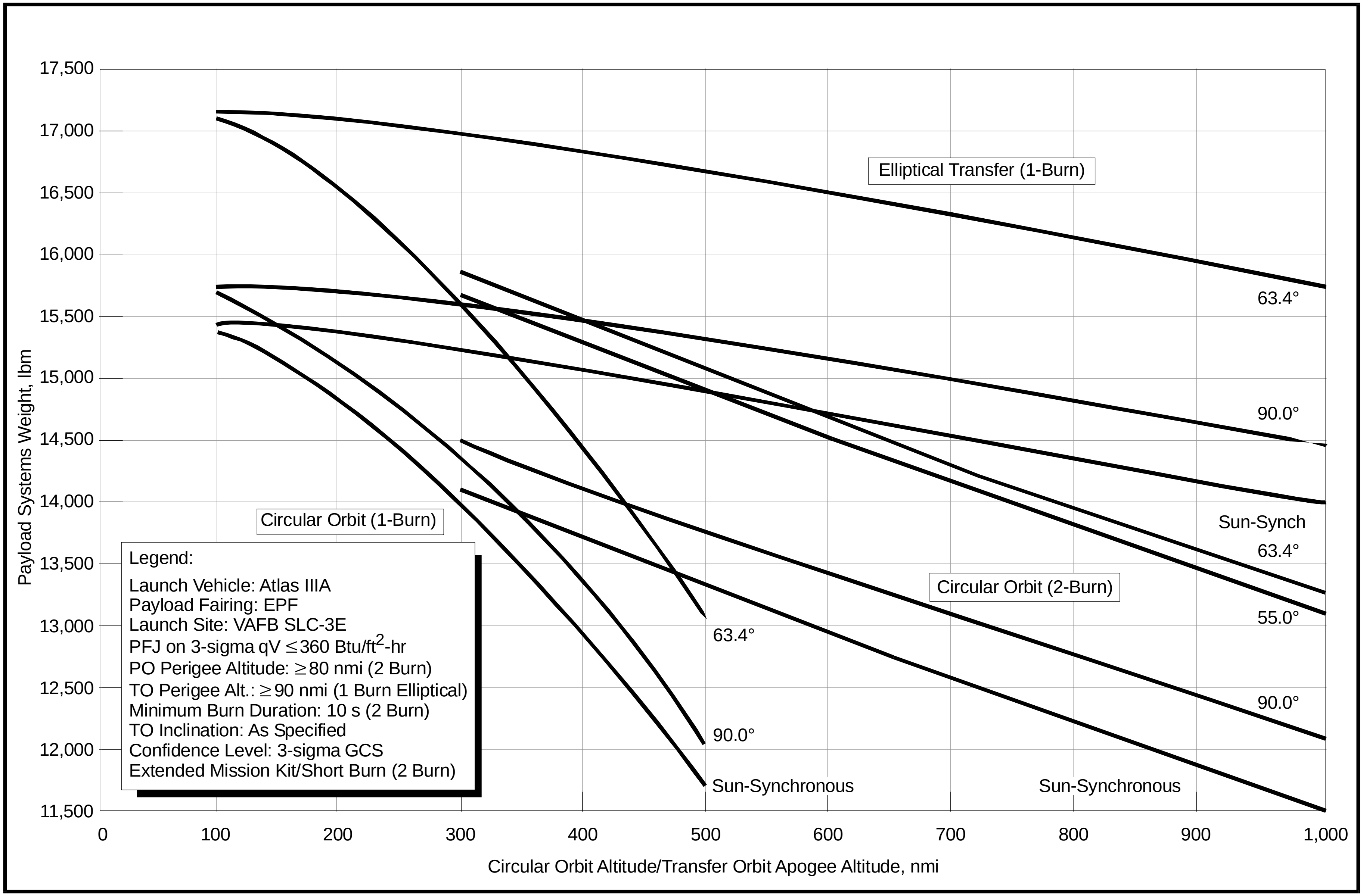


Figure 2.9-10b Atlas IIIA VAFB Performance to Low-Earth Orbit (English)

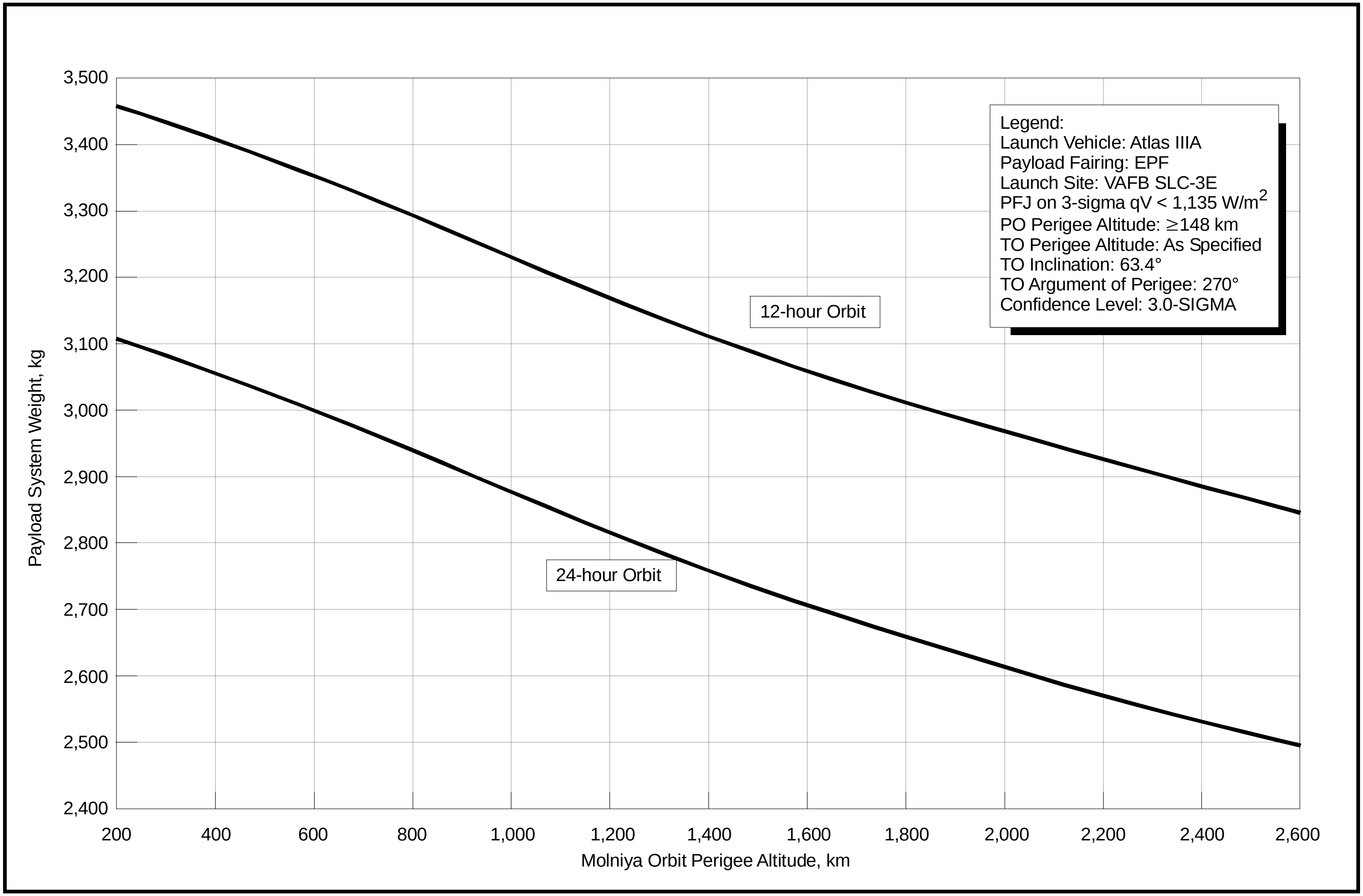


Figure 2.9-11a Atlas IIIA VAFB High-Inclination, High-Eccentricity Orbit Performance (Metric)

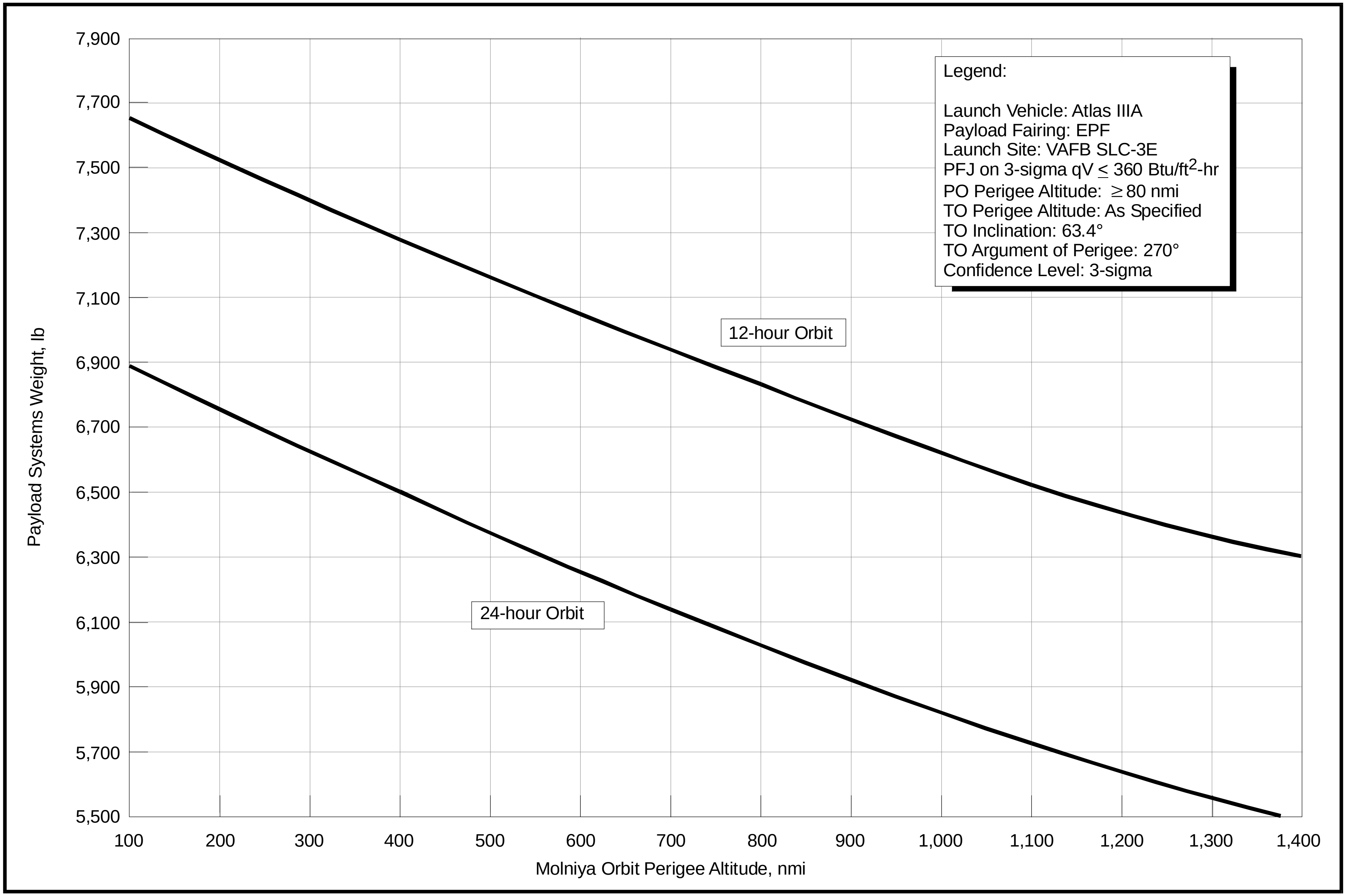


Figure 2.9-11b Atlas IIIA VAFB High-Inclination, High-Eccentricity Orbit Performance (English)

Table 2.9-1 Atlas IIIA Elliptical Transfer Orbit Performance—PSW vs Transfer Orbit Apogee Altitude

Payload Systems Weight, kg (lb)					
Apogee Altitude		Atlas IIIA			
km	(nmi)	MRS		GCS	
150,000	(80,993.5)	3,335	(7,352)	3,272	(7,214)
145,000	(78,293.7)	3,344	(7,373)	3,281	(7,234)
140,000	(75,594.0)	3,355	(7,396)	3,292	(7,257)
135,000	(72,894.2)	3,366	(7,420)	3,302	(7,280)
130,000	(70,194.4)	3,377	(7,446)	3,314	(7,306)
125,000	(67,494.6)	3,390	(7,474)	3,326	(7,333)
120,000	(64,794.8)	3,403	(7,503)	3,339	(7,362)
115,000	(62,095.0)	3,418	(7,536)	3,354	(7,394)
110,000	(59,395.2)	3,434	(7,571)	3,370	(7,429)
105,000	(56,695.5)	3,451	(7,609)	3,387	(7,467)
100,000	(53,995.7)	3,470	(7,651)	3,406	(7,508)
95,000	(51,295.9)	3,490	(7,694)	3,426	(7,553)
90,000	(48,596.1)	3,514	(7,748)	3,449	(7,603)
85,000	(45,896.3)	3,540	(7,805)	3,474	(7,659)
80,000	(43,196.5)	3,568	(7,867)	3,502	(7,721)
75,000	(40,496.8)	3,601	(7,938)	3,533	(7,790)
70,000	(37,797.0)	3,637	(8,018)	3,569	(7,869)
65,000	(35,097.2)	3,678	(8,109)	3,610	(7,958)
60,000	(32,397.4)	3,725	(8,213)	3,656	(8,061)
55,000	(29,697.6)	3,780	(8,334)	3,711	(8,181)
50,000	(26,997.8)	3,845	(8,477)	3,774	(8,321)
45,000	(24,298.1)	3,922	(8,647)	3,851	(8,489)
40,000	(21,598.3)	4,016	(8,854)	3,943	(8,692)
35,788	**(19,324.0)**	**4,112**	**(9,065)**	**4,037**	**(8,900)**
35,000	(18,898.5)	4,132	(9,109)	4,057	(8,944)
30,000	(16,198.7)	4,279	(9,434)	4,202	(9,264)
25,000	(13,498.9)	4,472	(9,860)	4,392	(9,683)
20,000	(10,799.1)	4,737	(10,444)	4,653	(10,258)
15,000	(8,099.4)	5,122	(11,291)	5,031	(11,092)
10,000	(5,399.6)	5,727	(12,626)	5,628	(12,407)
5,000	(2,699.8)	6,809	(15,011)	6,696	(14,762)

Note: Extended Payload Fairing Jettison at 3-sigma qv<1,135 W/m² (360 Btu/ft²-hr); Parking Orbit Perigee Altitude ≥ 148.2 km (80 nmi); Transfer Orbit Perigee Altitude ≥ 166.7 (90 nmi); Orbit Inclination = 27.0°; Argument of Perigee = 180°

Table 2.9-2 Atlas IIIA Performance to Reduced Inclination Transfer Orbit—PSW vs Orbit Inclination

Payload Systems Weight, kg (lb)				
	Atlas IIIA			
Inclination, °	MRS		GCS	
18.00	3,474	(7,659)	3,409	(7,515)
18.50	3,528	(7,778)	3,462	(7,631)
19.00	3,580	(7,892)	3,513	(7,745)
19.50	3,630	(8,004)	3,563	(7,854)
20.00	3,679	(8,111)	3,611	(7,960)
20.50	3,726	(8,215)	3,657	(8,062)
21.00	3,771	(8,313)	3,701	(8,159)
21.50	3,814	(8,408)	3,743	(8,252)
22.00	3,854	(8,497)	3,783	(8,340)
22.50	3,892	(8,580)	3,820	(8,423)
23.00	3,928	(8,659)	3,856	(8,500)
23.50	3,961	(8,732)	3,888	(8,572)
24.00	3,991	(8,799)	3,918	(8,638)
24.50	4,019	(8,860)	3,945	(8,697)
25.00	4,043	(8,914)	3,969	(8,751)
25.50	4,065	(8,962)	3,991	(8,798)
26.00	4,084	(9,003)	4,009	(8,839)
26.50	4,099	(9,037)	4,025	(8,873)
27.00	**4,112**	**(9,065)**	**4,037**	**(8,900)**
27.50	4,121	(9,086)	4,046	(8,920)
28.00	4,127	(9,099)	4,052	(8,933)
28.50	4,131	(9,106)	4,055	(8,940)
29.00	4,132	(9,109)	4,057	(8,943)
29.50	4,132	(9,109)	4,056	(8,943)
30.00	4,129	(9,102)	4,054	(8,937)

Note: Extended Payload Fairing Jettison at 3-sigma qv<1,135 W/m² (360 Btu/ft²-hr); Parking Orbit Perigee Altitude ≥ 148.2 km (80 nmi); Transfer Orbit Perigee Altitude ≥ 166.7 (90 nmi); Transfer Orbit Apogee Altitude = 35,788 km (19,324 nmi); Argument of Perigee = 180°

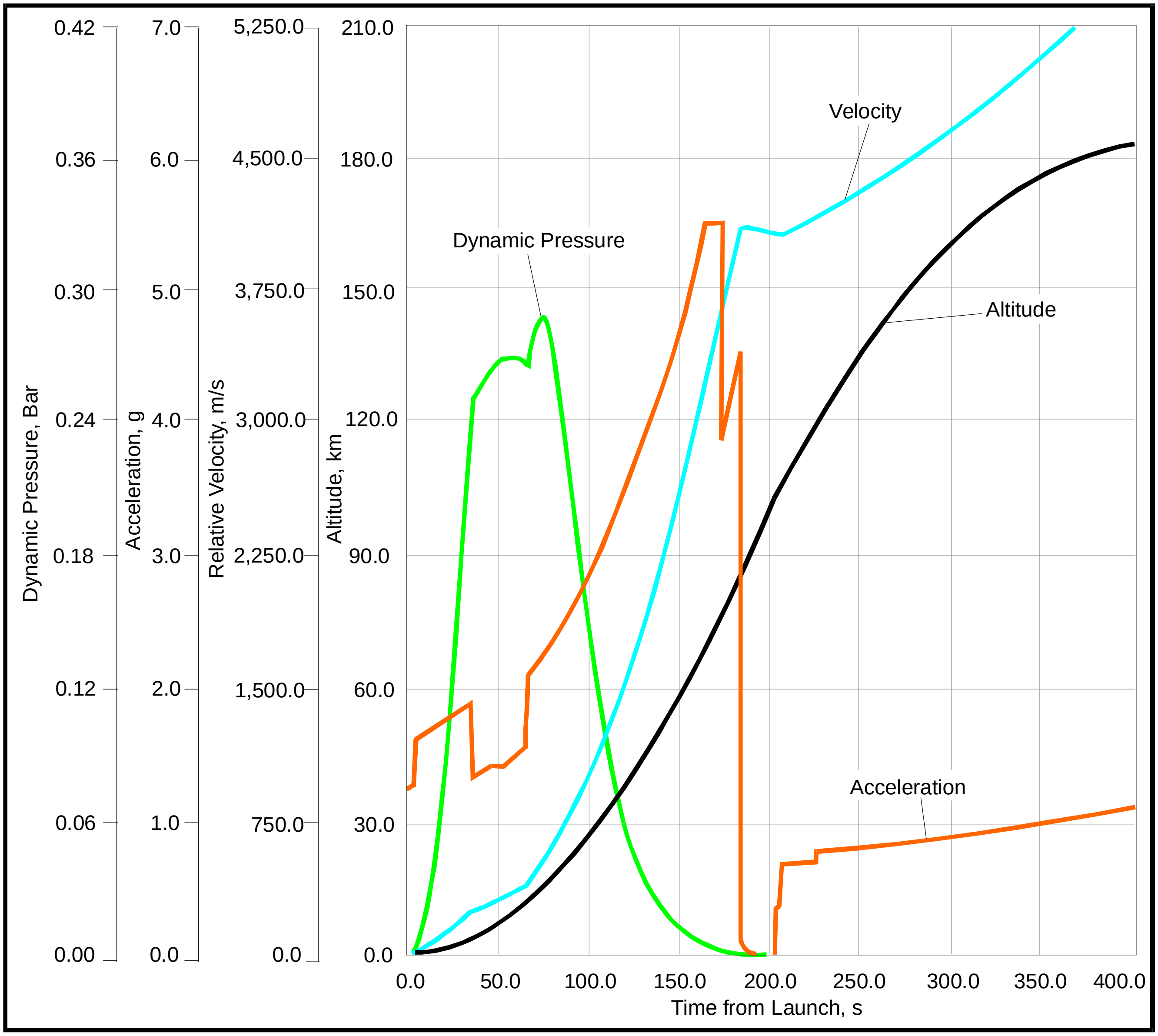


Figure 2.10-1 Atlas IIIB (DEC) Nominal Ascent Data

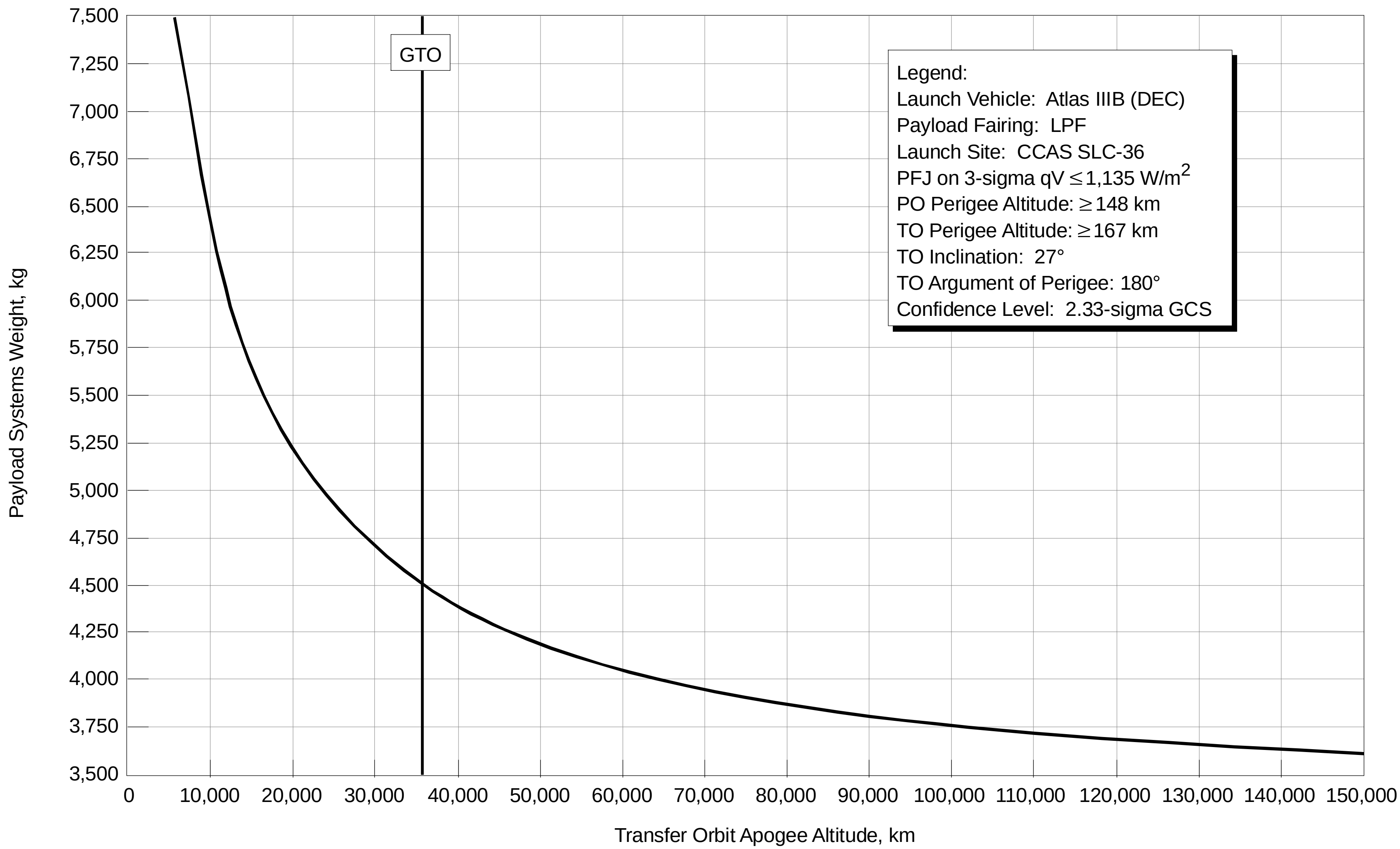


Figure 2.10-2a Atlas IIIB (DEC) CCAS Performance to Elliptical Transfer Orbit (GCS-Metric)

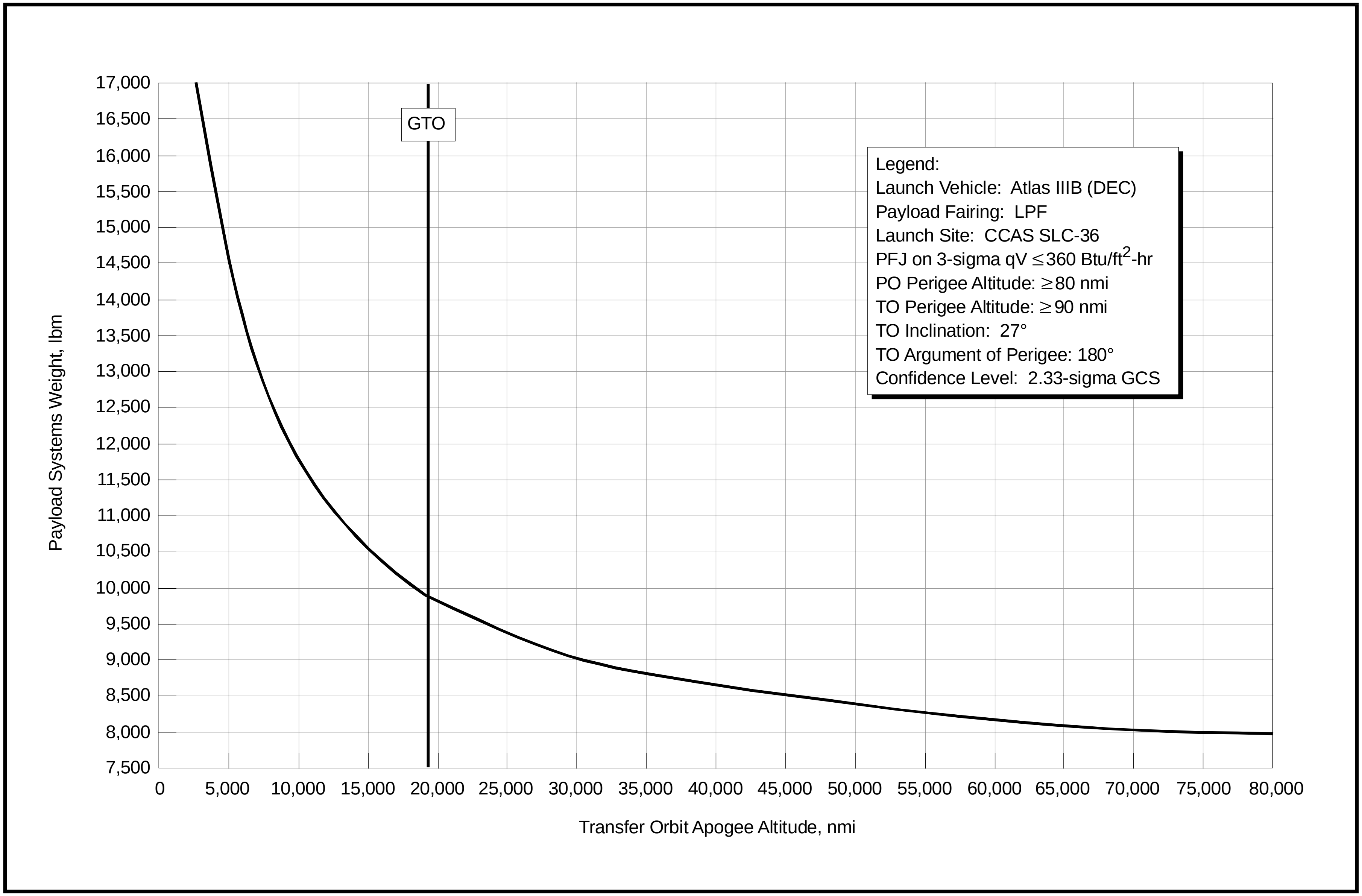


Figure 2.10-2b Atlas IIIB (DEC) CCAS Performance to Elliptical Transfer Orbit (GCS-English)

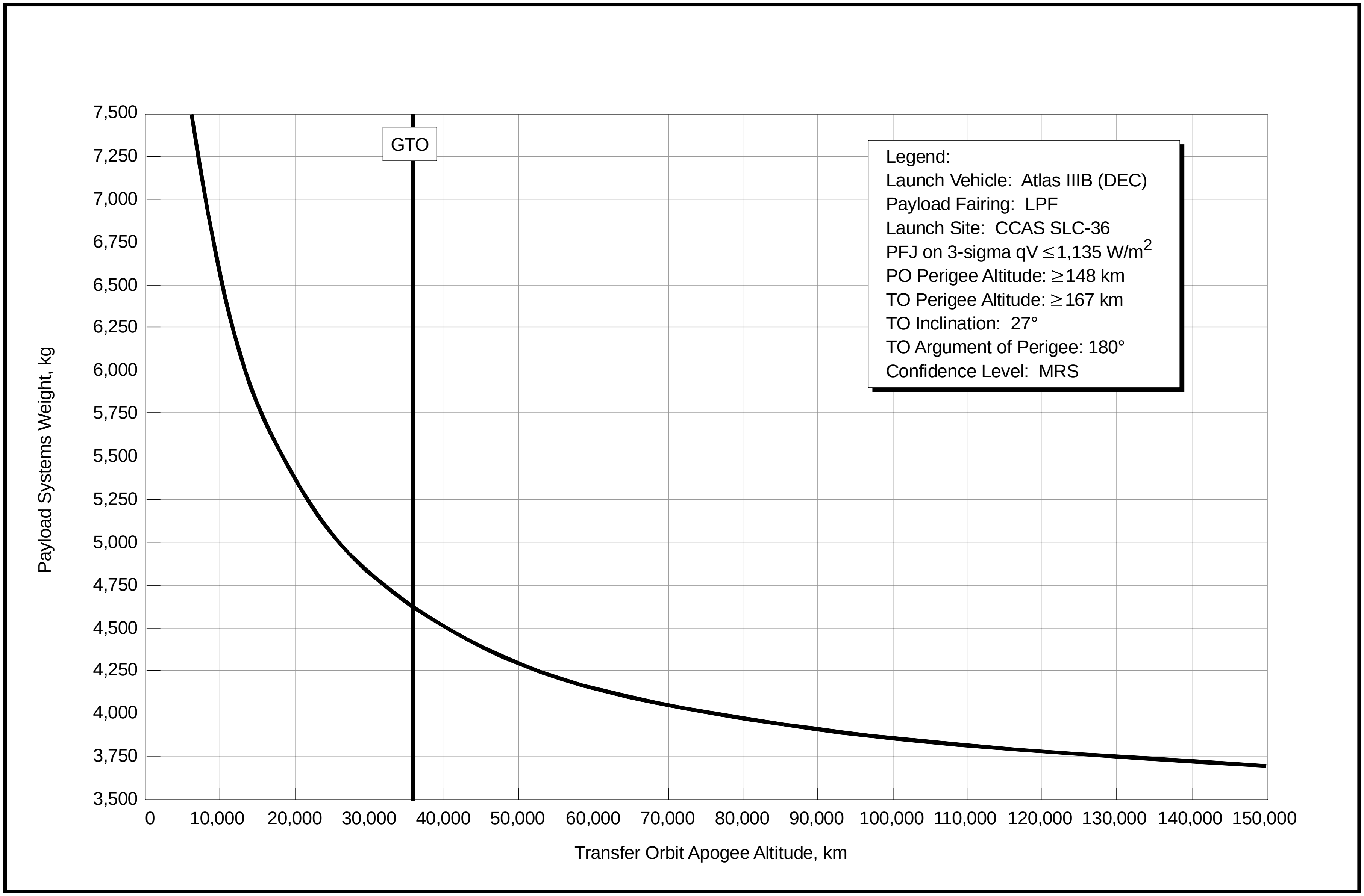


Figure 2.10-3a Atlas IIIB (DEC) CCAS Performance to Elliptical Transfer Orbit (MRS-Metric)

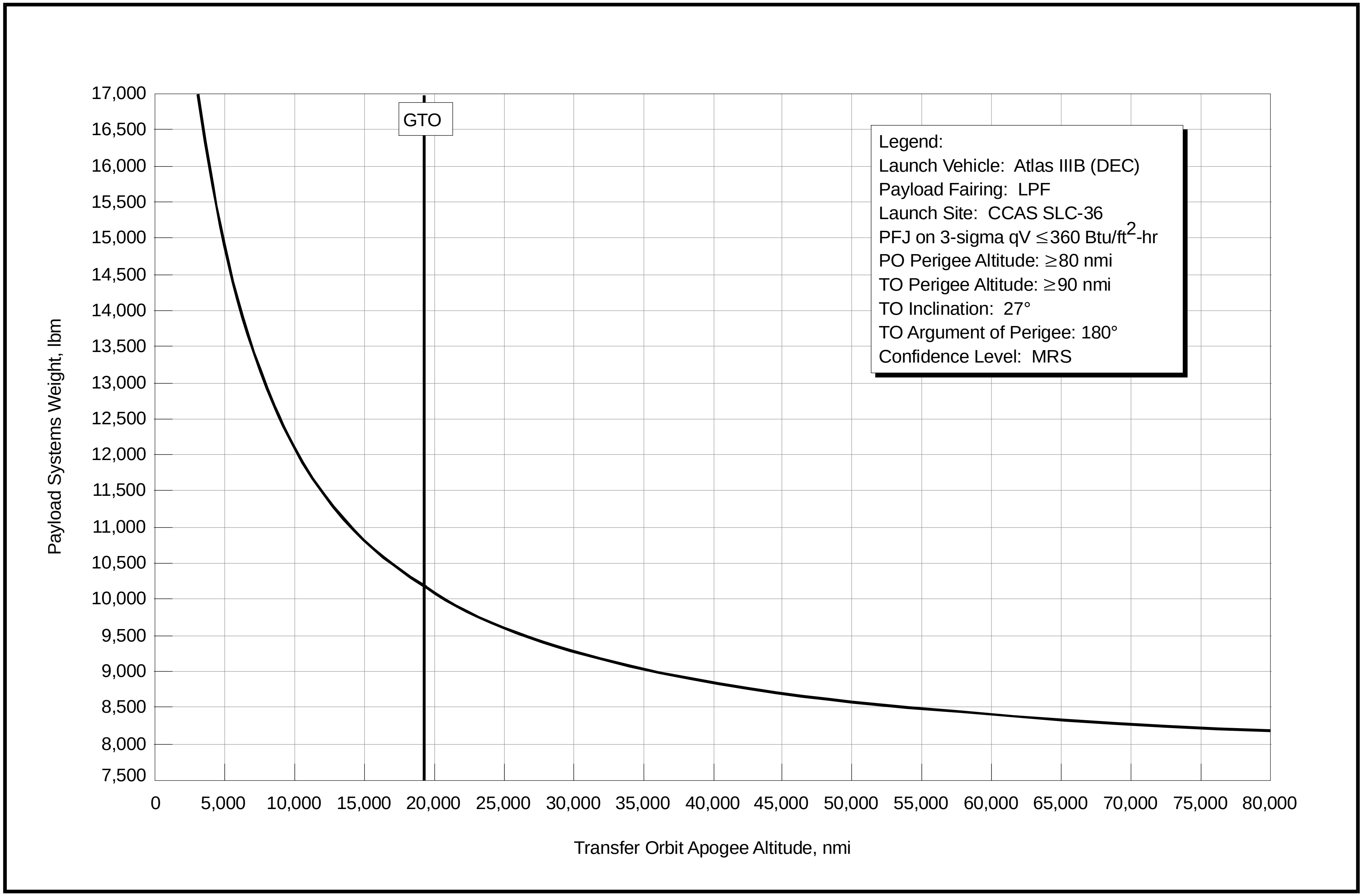


Figure 2.10-3b Atlas IIIB (DEC) CCAS Performance to Elliptical Transfer Orbit (MRS-English)

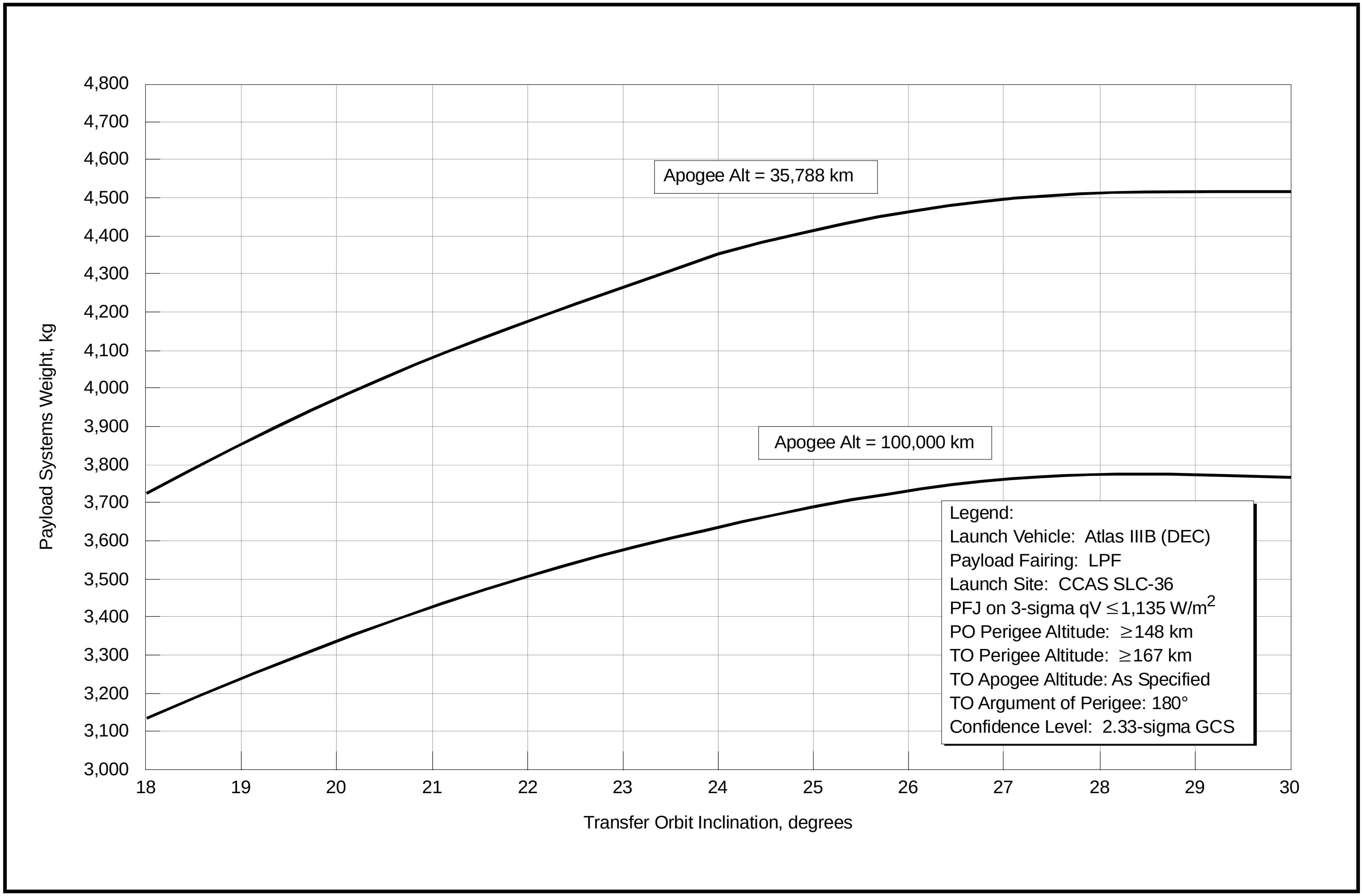


Figure 2.10-4a Atlas IIIB (DEC) CCAS Reduced Inclination Elliptical Orbit Performance (GCS-Metric)

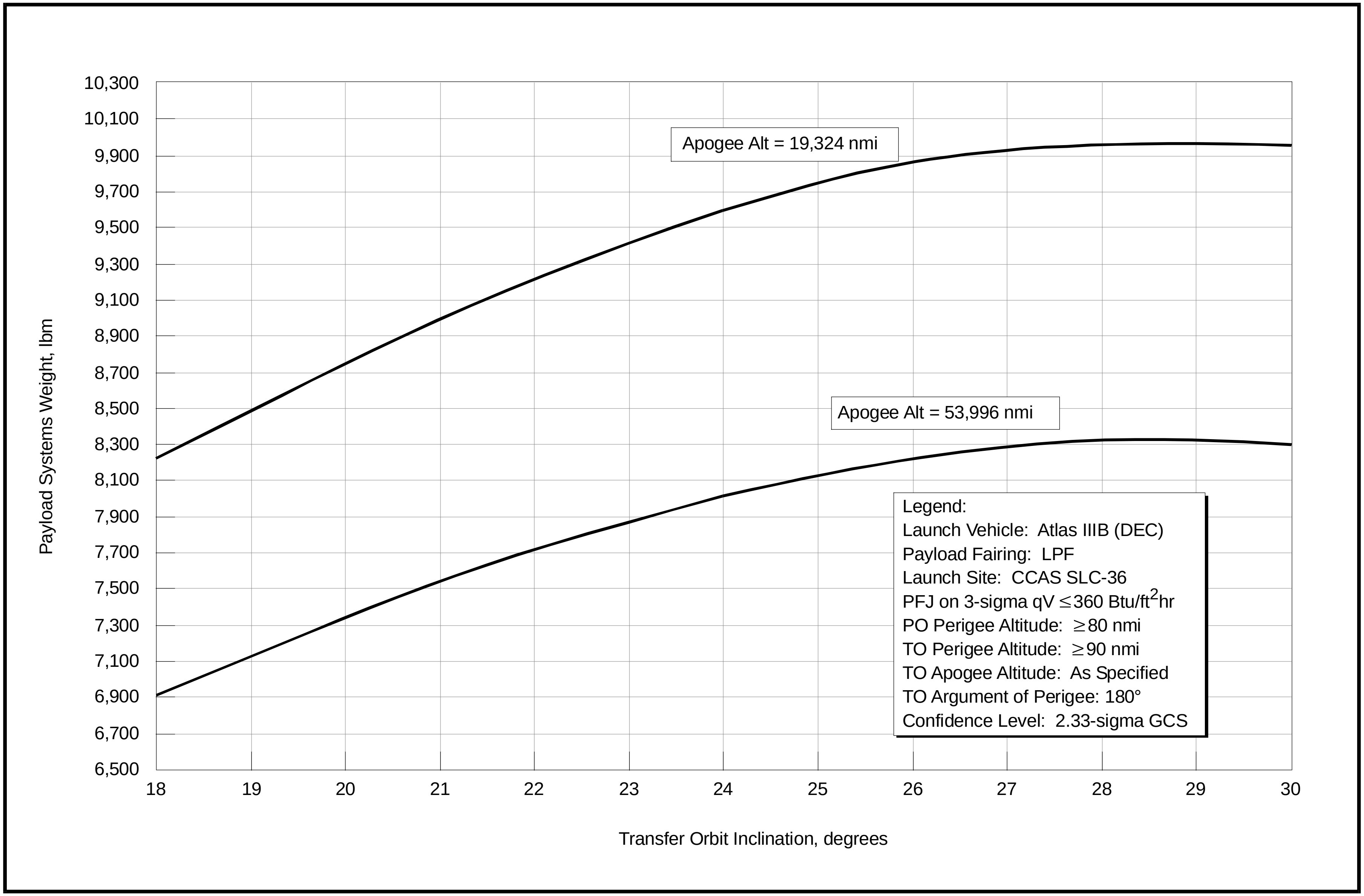


Figure 2.10-4b Atlas IIIB (DEC) CCAS Reduced Inclination Elliptical Orbit Performance (GCS-English)

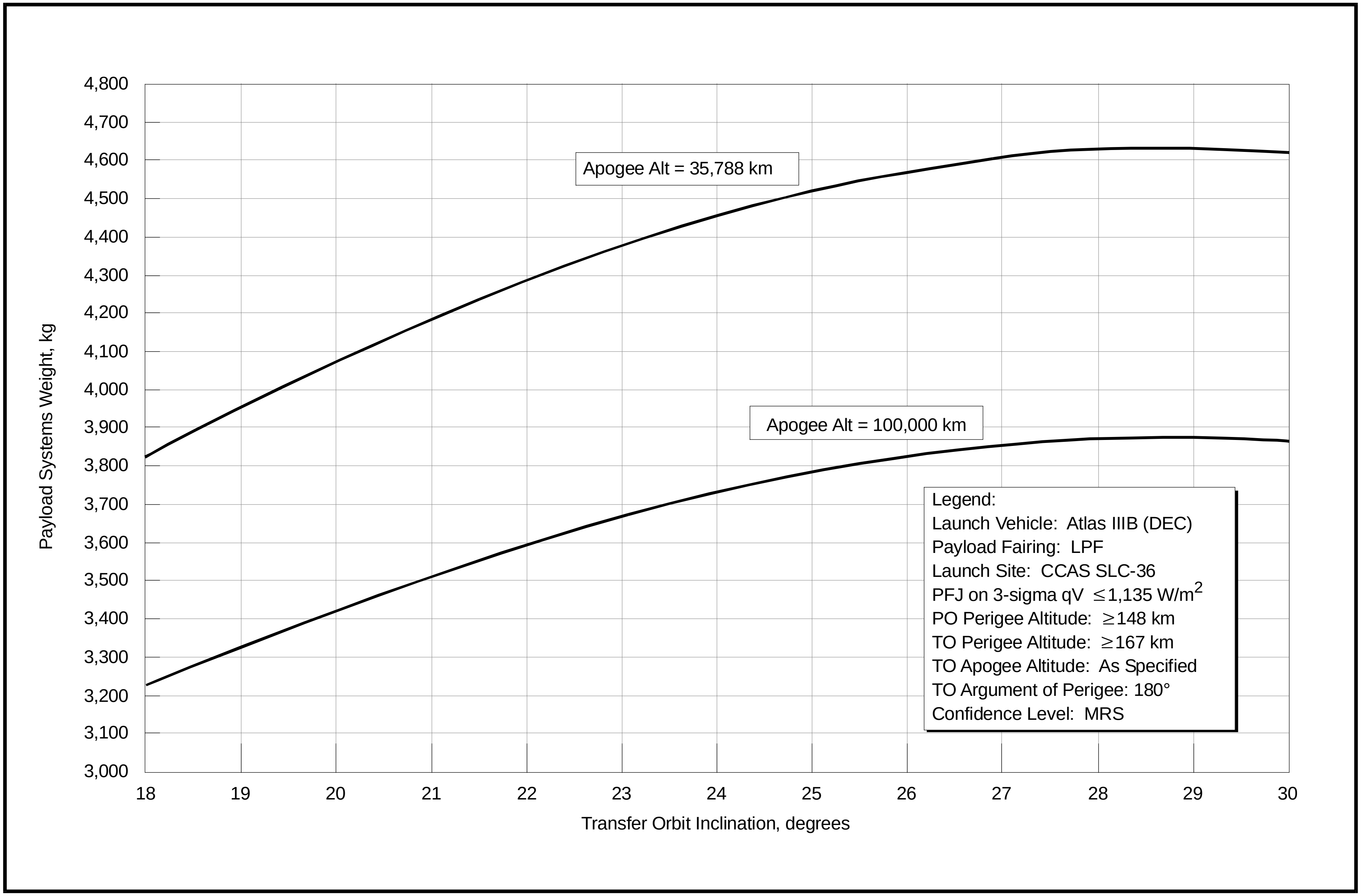


Figure 2.10-5a Atlas IIIB (DEC) CCAS Reduced Inclination Elliptical Orbit Performance (MRS-Metric)

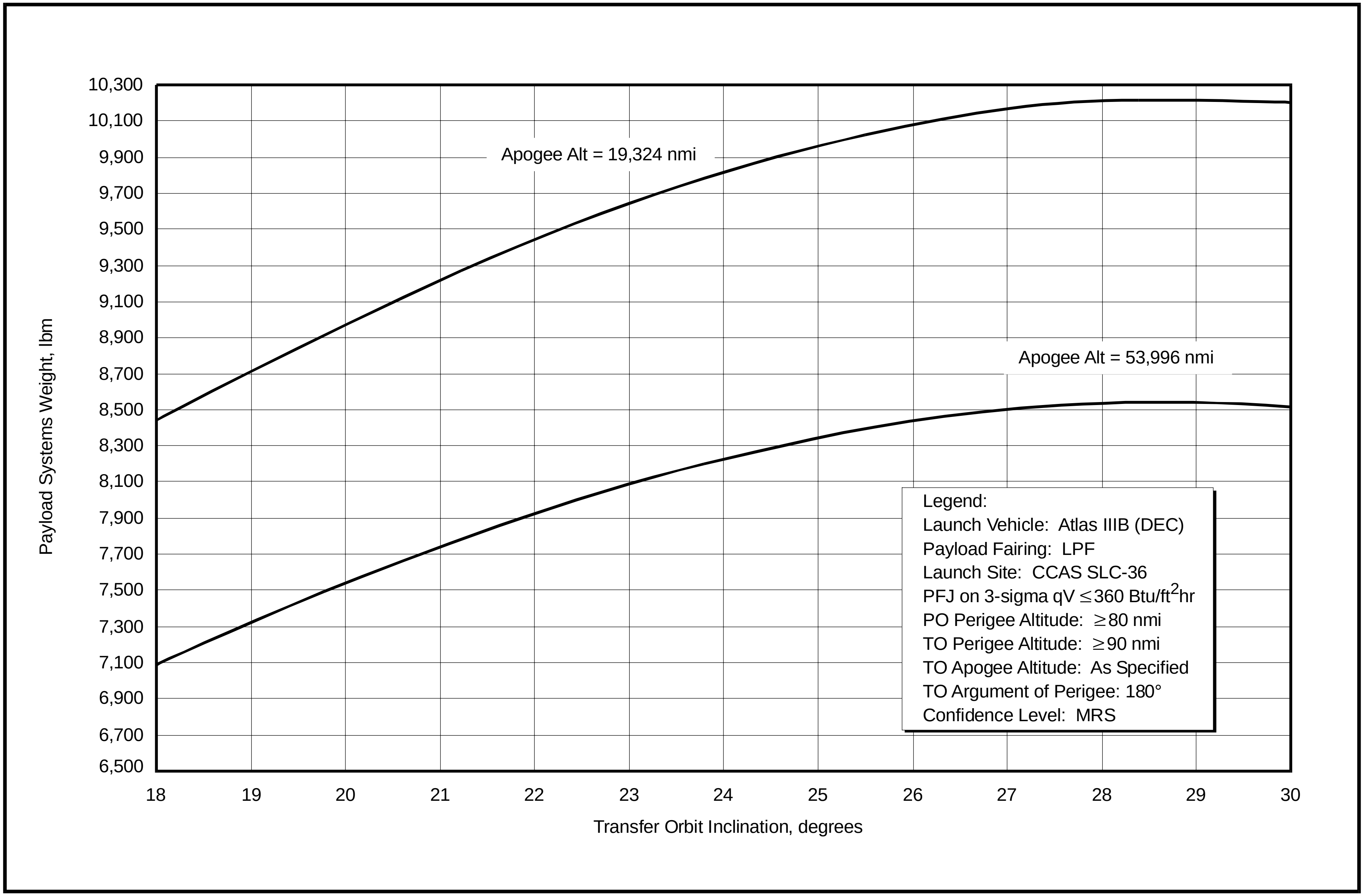


Figure 2.10-5b Atlas IIIB (DEC) CCAS Reduced Inclination Elliptical Orbit Performance (MRS-English)

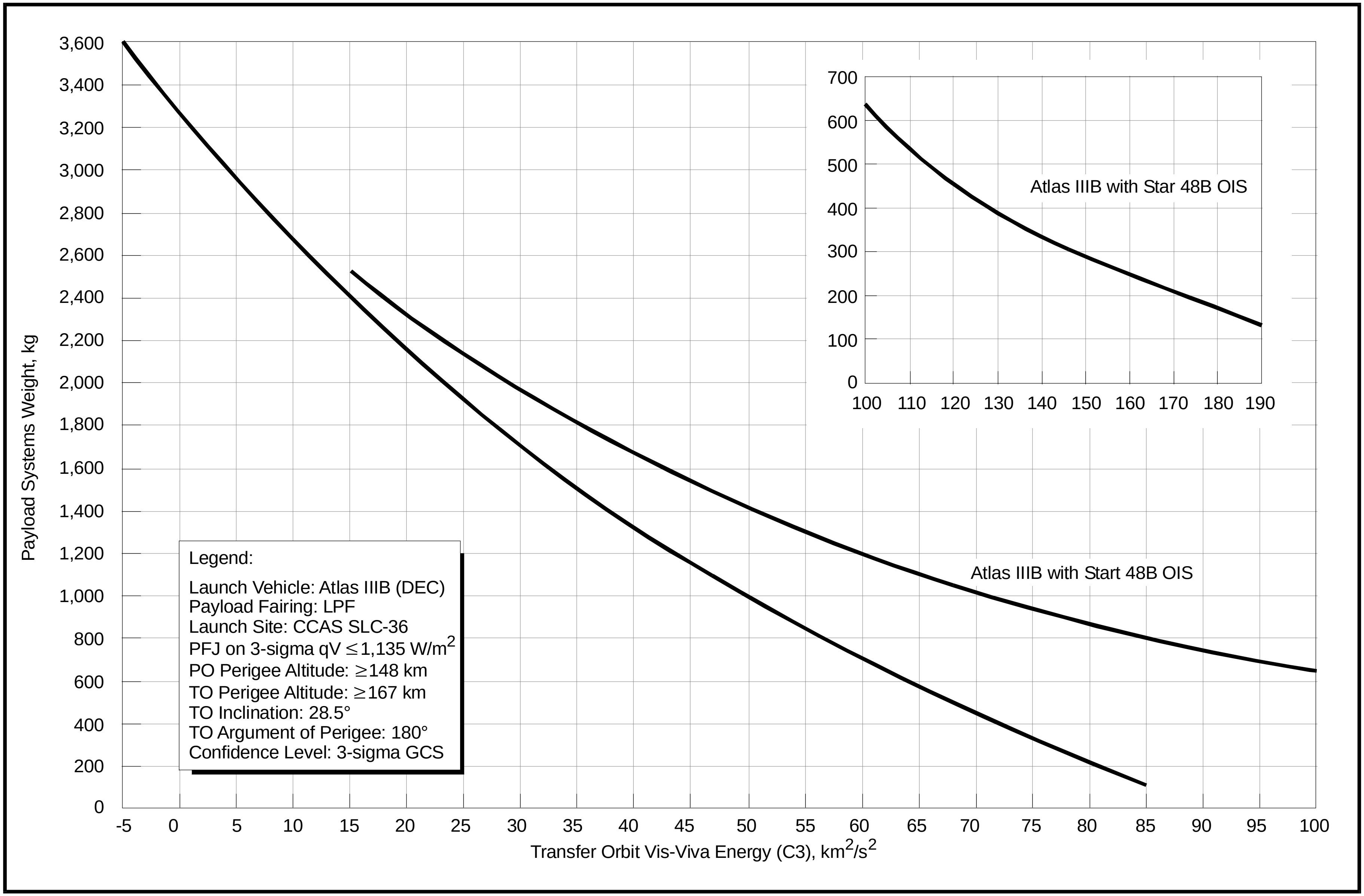


Figure 2.10-6a Atlas IIIB (DEC) CCAS Earth-Escape Performance (Metric)

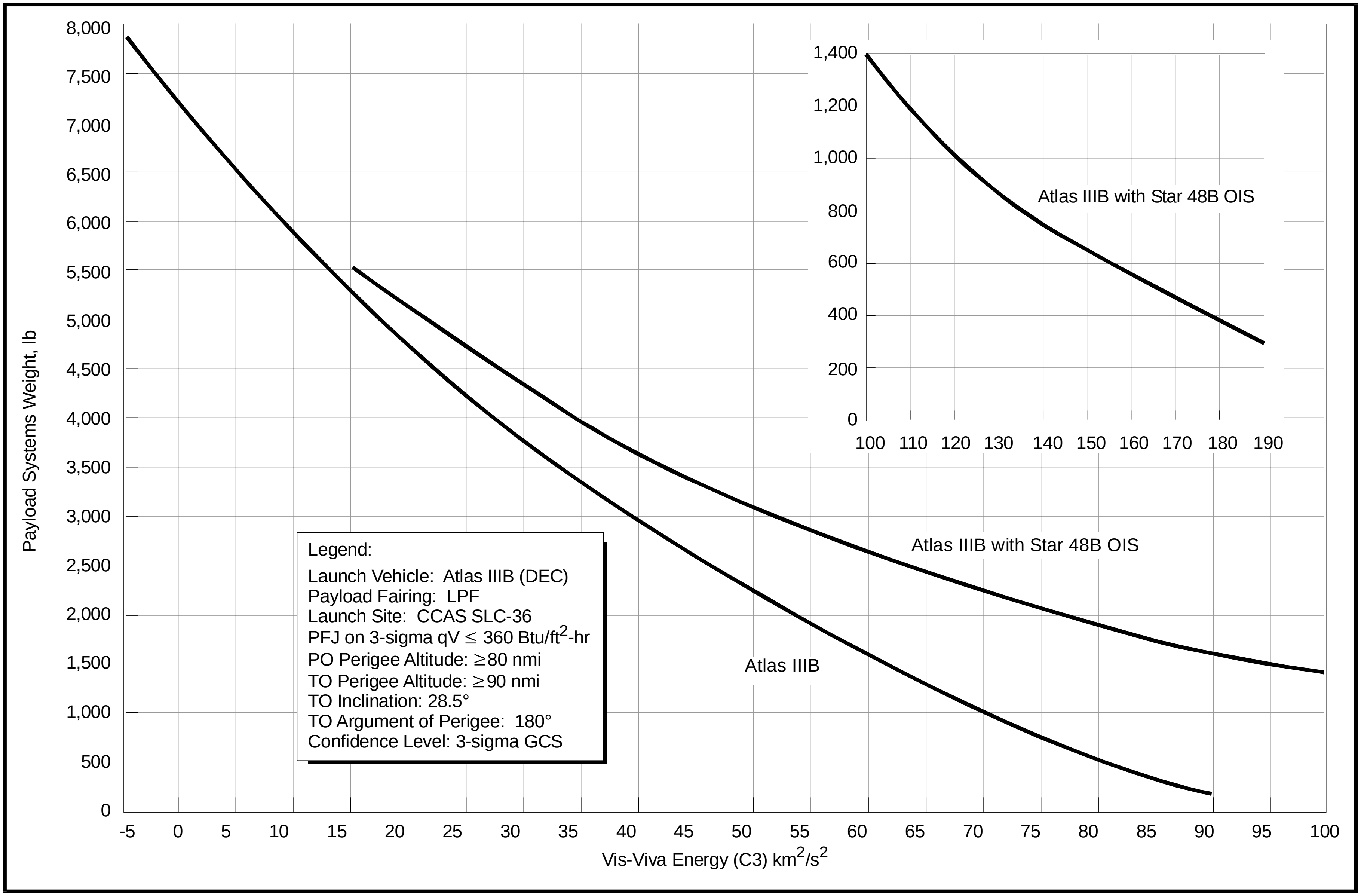


Figure 2.10-6b Atlas IIIB (DEC) CCAS Earth-Escape Performance (English)

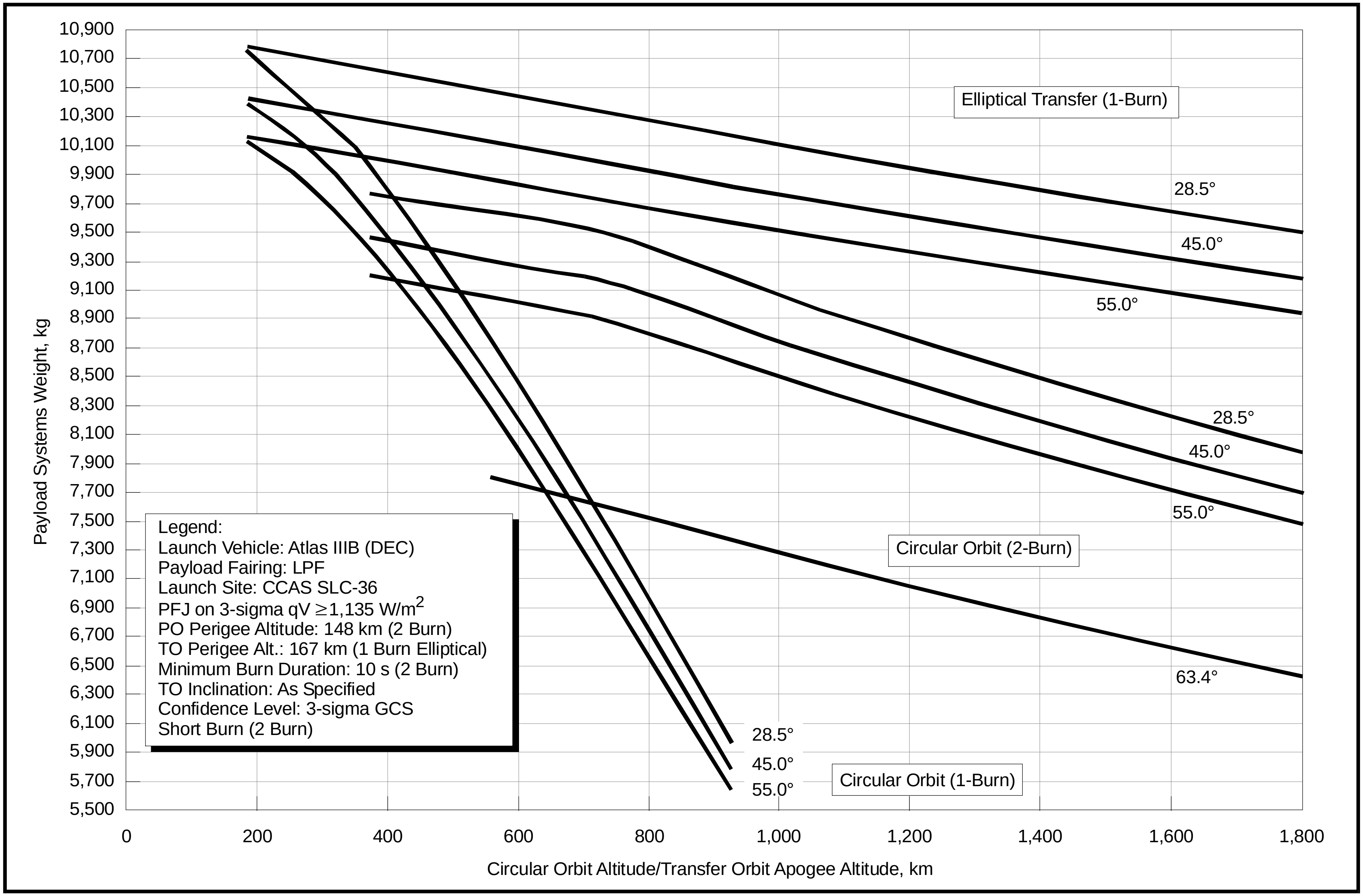


Figure 2.10-7a Atlas IIIB (DEC) CCAS Low-Earth Performance (Metric)

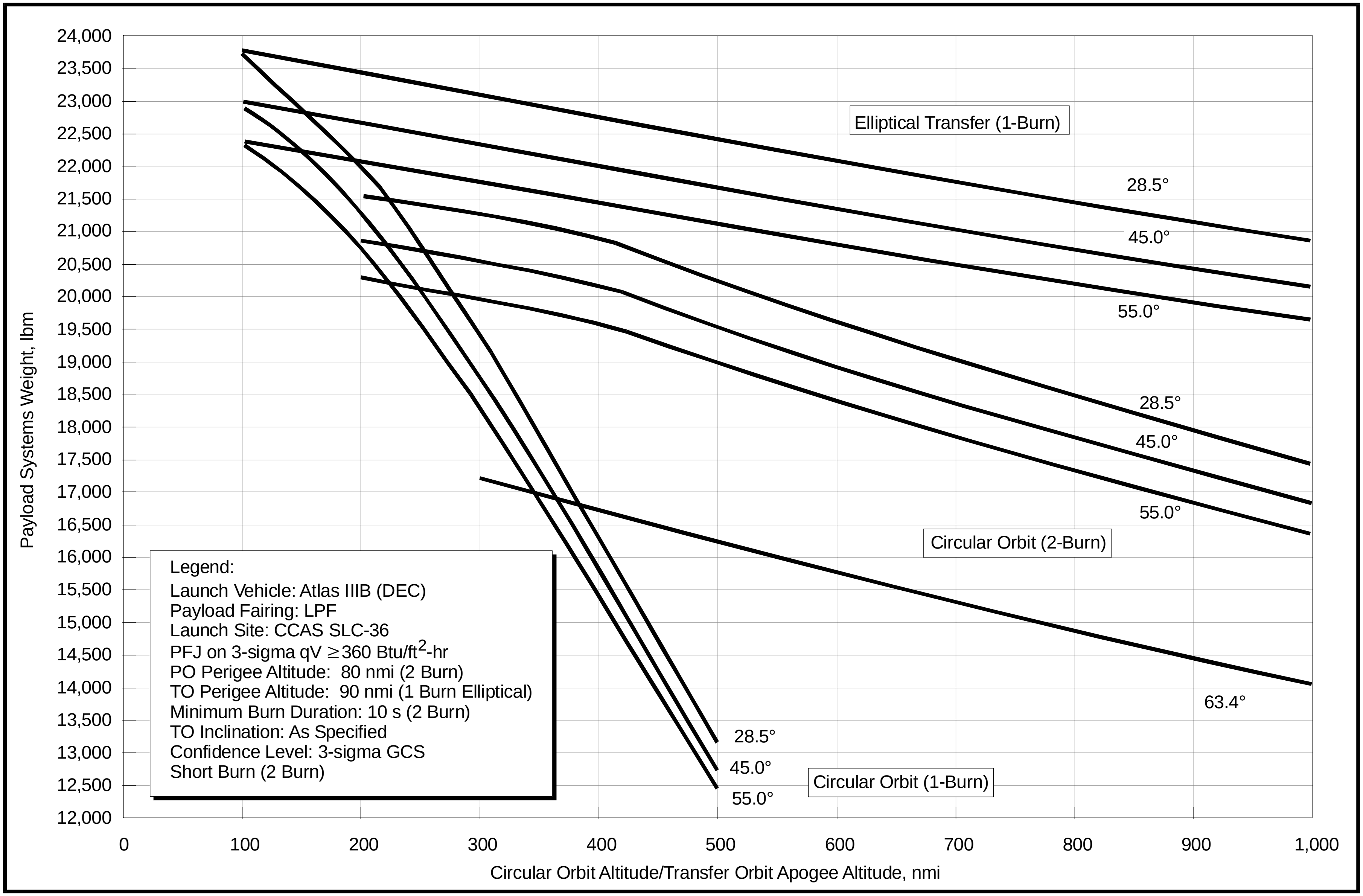


Figure 2.10-7b Atlas IIIB (DEC) CCAS Low-Earth Performance (English)

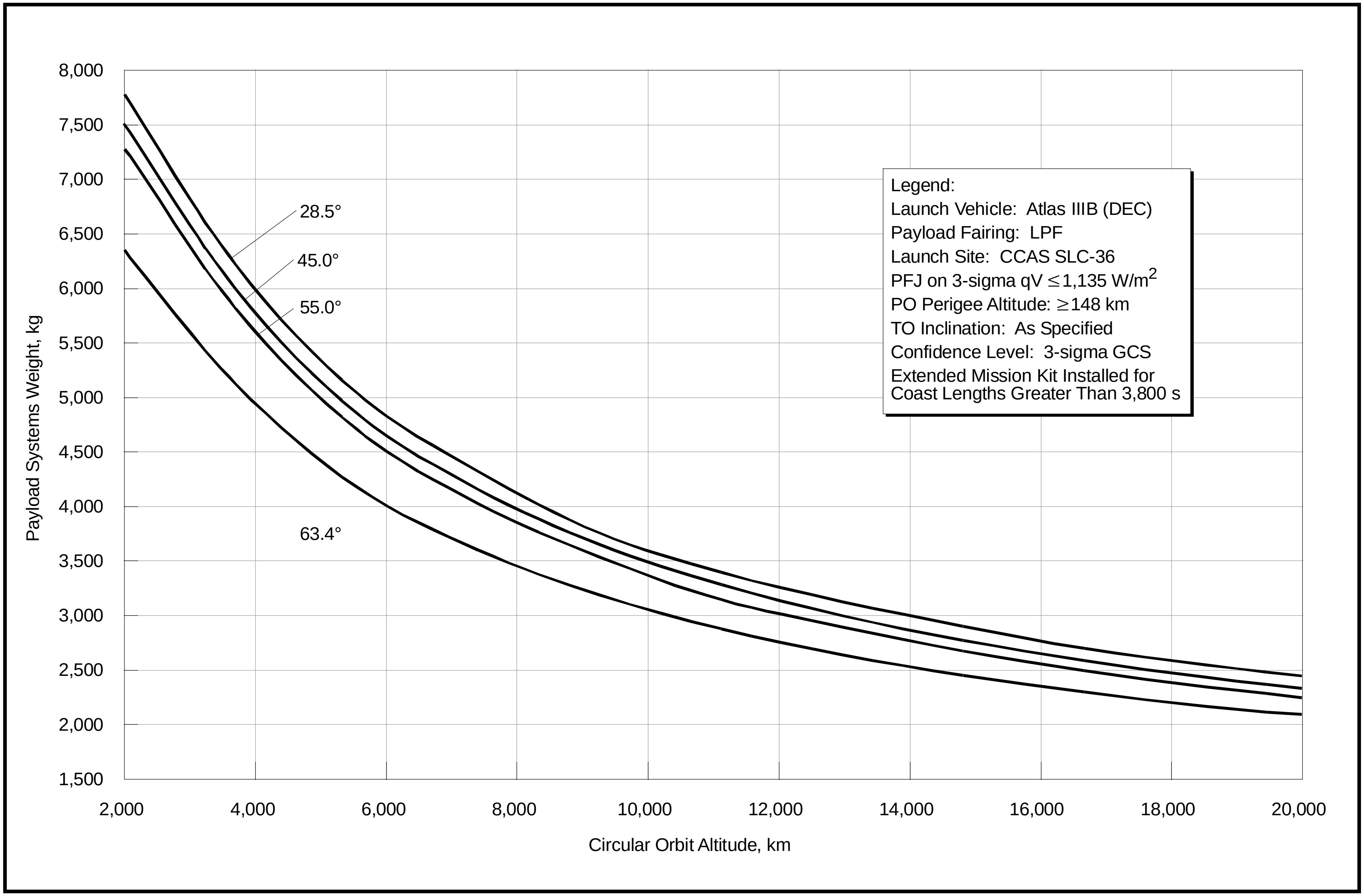


Figure 2.10-8a Atlas IIIB (DEC) CCAS Intermediate Circular Orbit Performance (Metric)

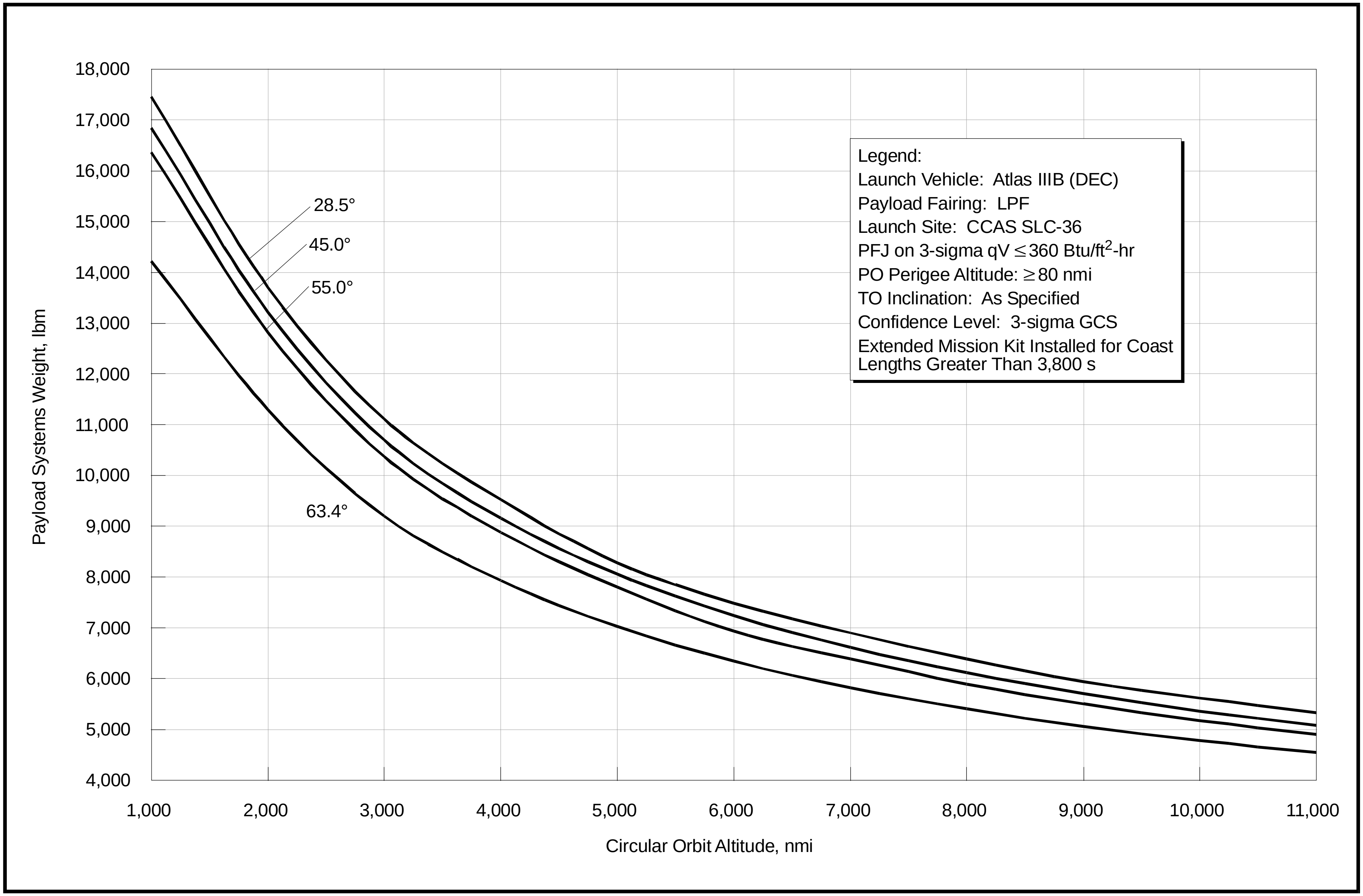


Figure 2.10-8b Atlas IIIB (DEC) CCAS Intermediate Circular Orbit Performance (English)

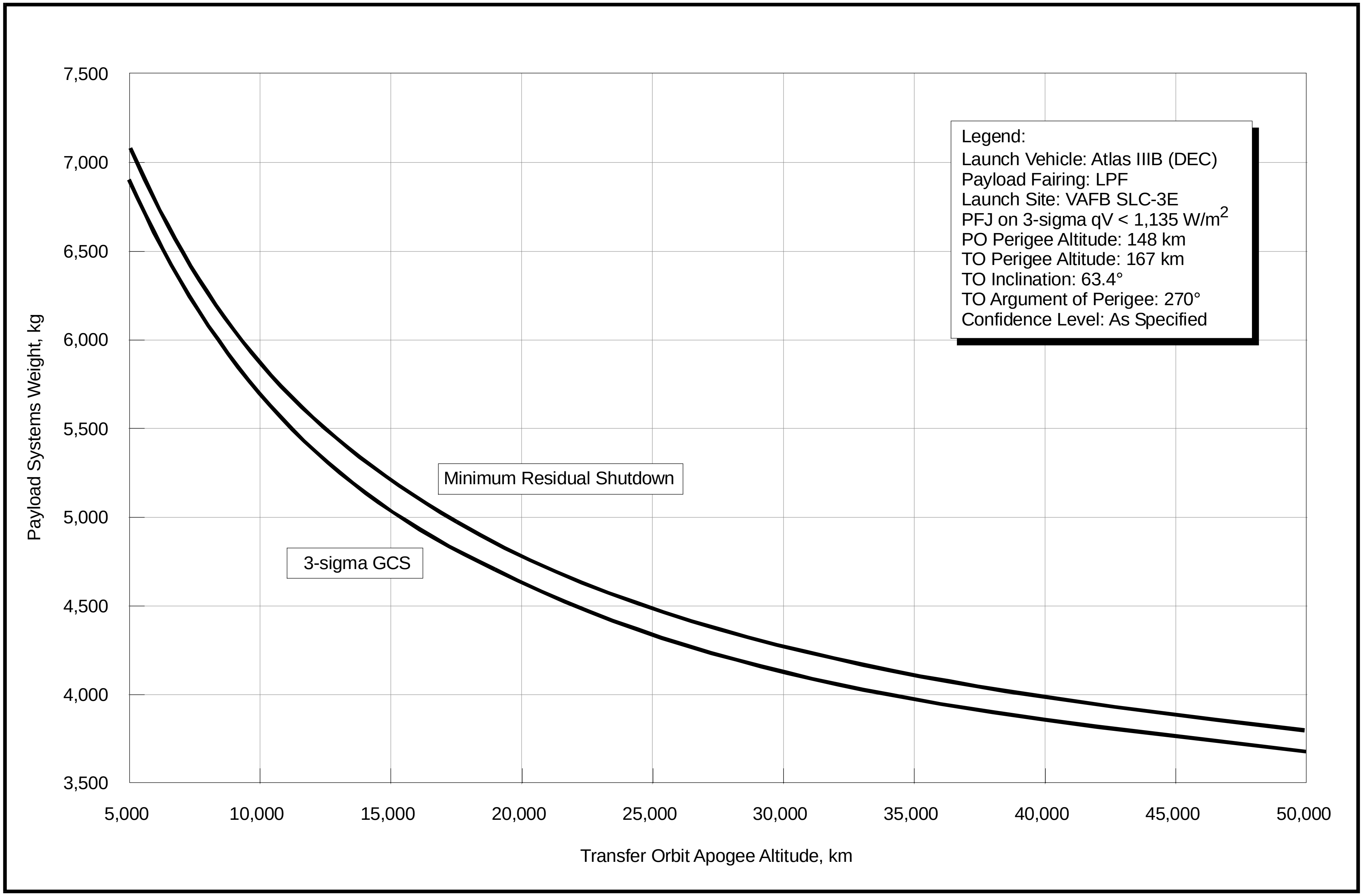


Figure 2.10-9a Atlas IIIB VAFB (DEC) Elliptical Orbit Performance (Metric)

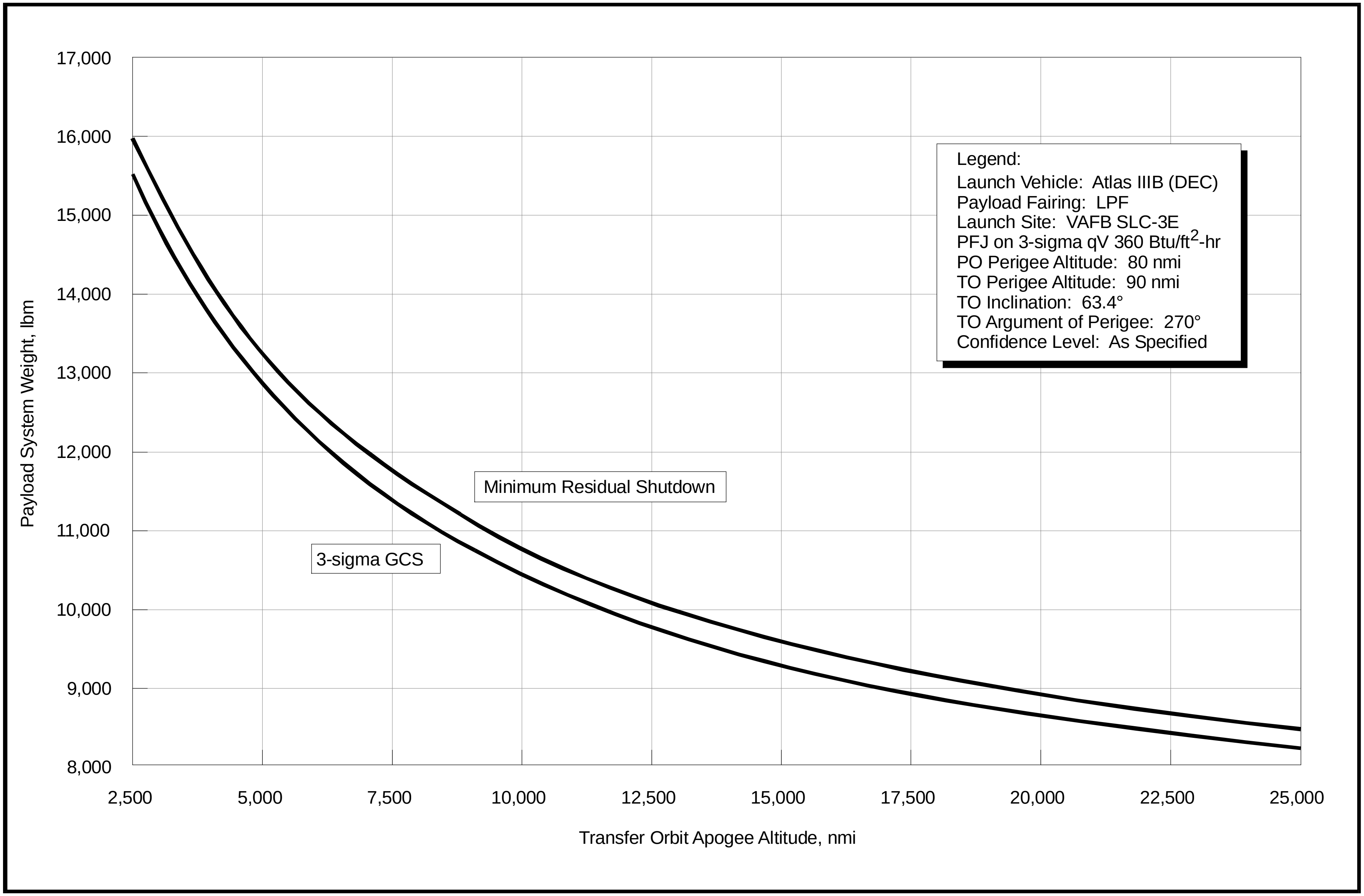


Figure 2.10-9b Atlas IIIB (DEC) VAFB Elliptical Orbit Performance (English)

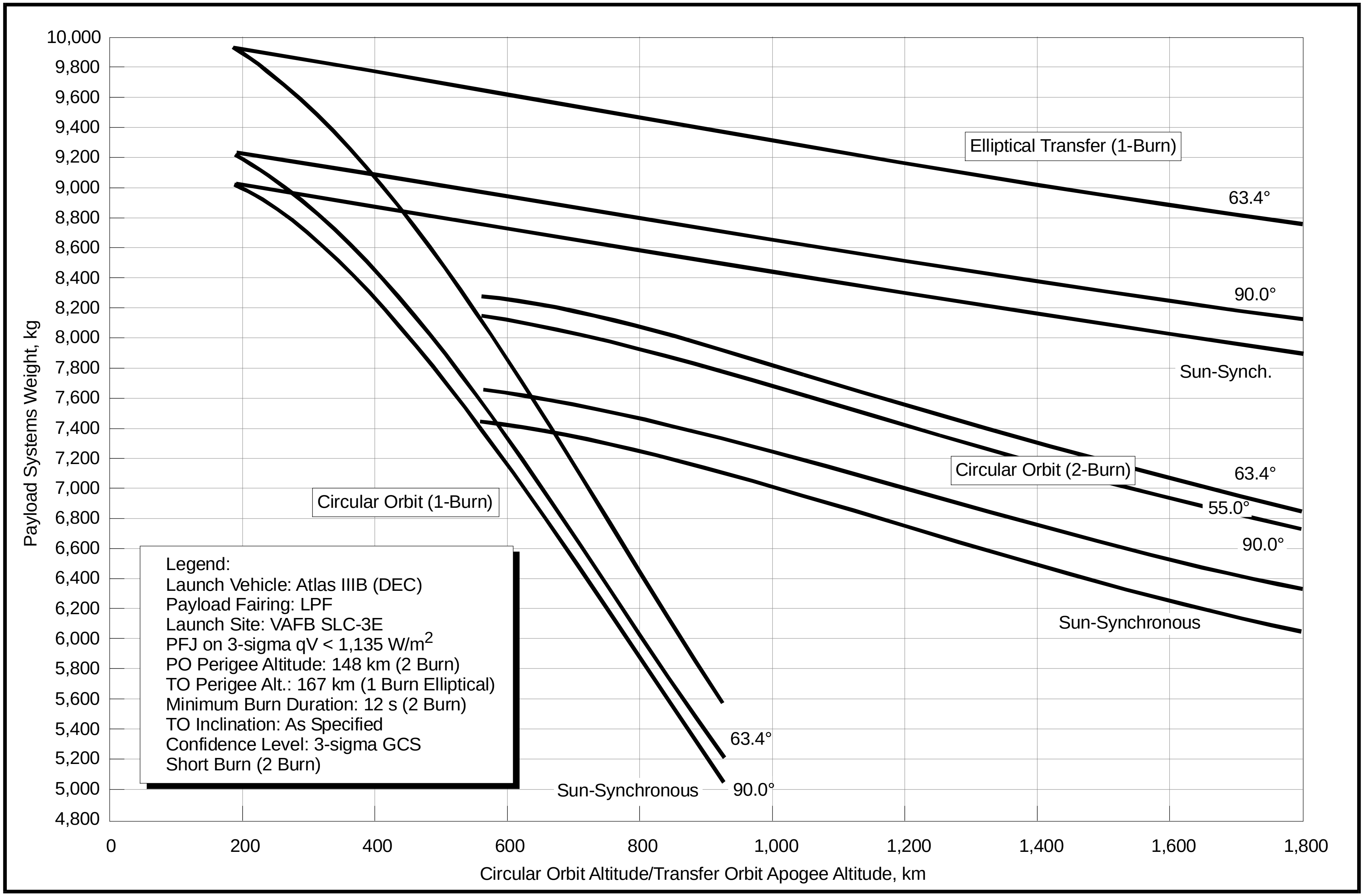


Figure 2.10-10a Atlas IIIB (DEC) VAFB Performance to Low-Earth Orbit (Metric)

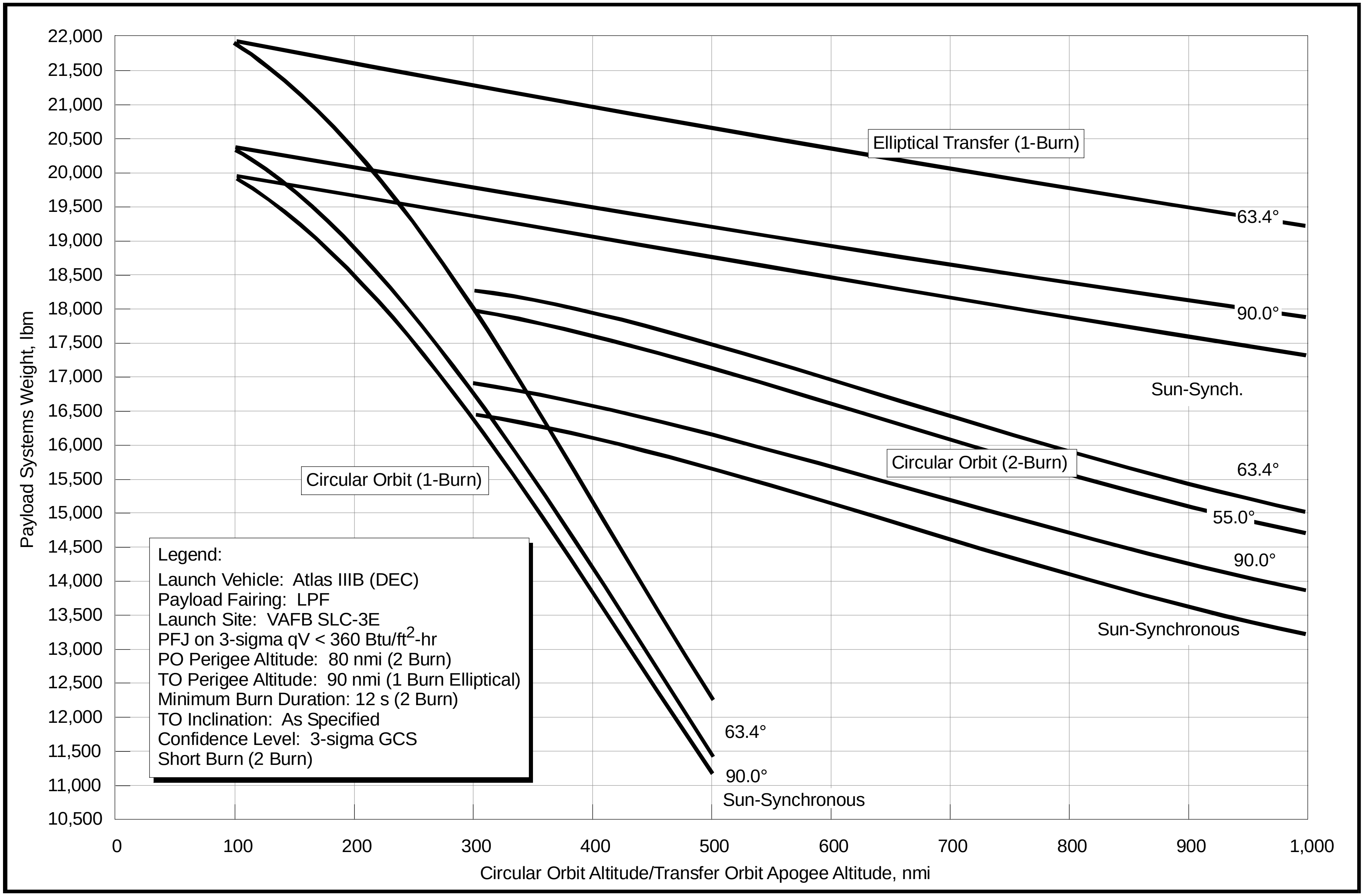


Figure 2.10-10b Atlas IIIB (DEC) VAFB Performance to Low-Earth Orbit (English)

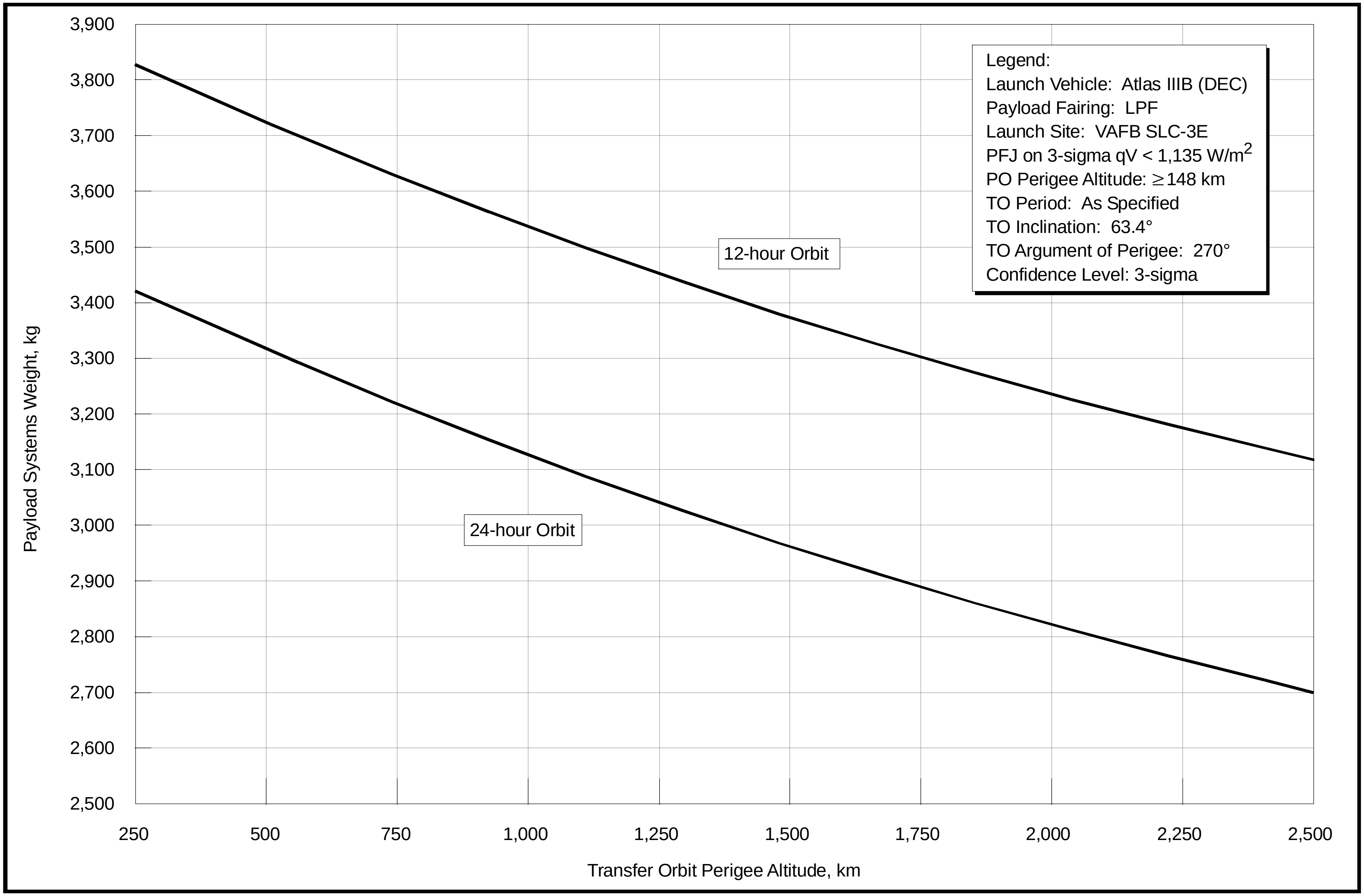


Figure 2.10-11a Atlas IIIB (DEC) VAFB High-Inclination, High-Eccentricity Orbit Performance (Metric

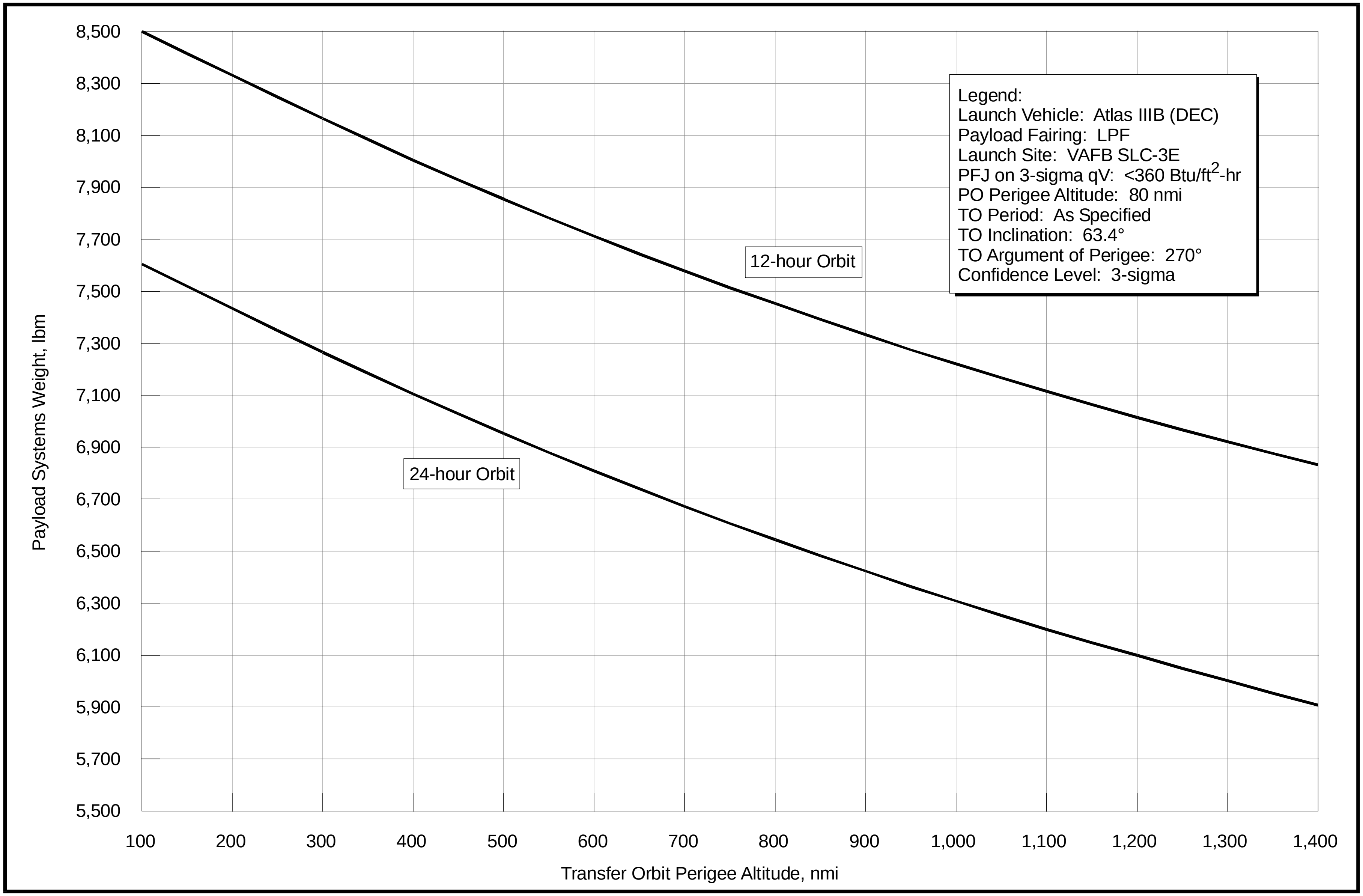


Figure 2.10-11b Atlas IIIB (DEC) VAFB High-Inclination, High-Eccentricity Orbit Performance (English)

Table 2.10-1 Atlas IIIB (DEC) Elliptical Transfer Orbit Performance—PSW vs Apogee Altitude

Payload Systems Weight, kg (lb)					
Apogee Altitude		Atlas IIIB			
km	(nmi)	MRS		GCS	
150,000	(80,993.5)	3,697	(8,150)	3,603	(7,944)
145,000	(78,293.7)	3,708	(8,174)	3,614	(7,968)
140,000	(75,594.0)	3,720	(8,201)	3,626	(7,994)
135,000	(72,894.2)	3,732	(8,229)	3,639	(8,022)
130,000	(70,194.4)	3,746	(8,259)	3,652	(8,051)
125,000	(67,494.6)	3,761	(8,291)	3,667	(8,083)
120,000	(64,794.8)	3,777	(8,326)	3,682	(8,118)
115,000	(62,095.0)	3,794	(8,364)	3,699	(8.155)
110,000	(59,395.2)	3,813	(8,405)	3,717	(8,195)
105,000	(56,695.5)	3,833	(8,450)	3,737	(8,239)
100,000	(53,995.7)	3,855	(8,499)	3,759	(8,288)
95,000	(51,295.9)	3,880	(8,553)	3,783	(8,340)
90,000	(48,596.1)	3,906	(8,612)	3,810	(8,399)
85,000	(45,896.3)	3,936	(8,678)	3,839	(8,464)
80,000	(43,196.5)	3,970	(8,752)	3,872	(8,536)
75,000	(40,496.8)	4,007	(8,834)	3,909	(8,617)
70,000	(37,797.0)	4,050	(8,928)	3,950	(8,709)
65,000	(35,097.2)	4,098	(9,034)	3,998	(8,814)
60,000	(32,397.4)	4,154	(9,157)	4,053	(8,934)
55,000	(29,697.6)	4,218	(9,299)	4,116	(9,074)
50,000	(26,997.8)	4,294	(9,467)	4,191	(9,239)
45,000	(24,298.1)	4,385	(9,668)	4,280	(9,436)
40,000	(21,598.3)	4,496	(9,911)	4,389	(9,675)
35,788	(19,324.0)	4,609	(10,161)	4,500	(9,920)
35,000	(18,898.5)	4,633	(10,213)	4,523	(9,972)
30,000	(16,198.7)	4,807	(10,597)	4,695	(10,350)
25,000	(13,498.9)	5,037	(11,104)	4,921	(10,848)
20,000	(10,799.1)	5,352	(11,800)	5,231	(11,533)
15,000	(8,099.4)	5,814	(12,817)	5,685	(12,532)
10,000	(5,399.6)	6,550	(14,440)	6,408	(14,127)
5,000	(2,699.8)	7,898	(17,413)	7,734	(17,051)

Note: Large Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude ≥ 148.2 km (80 nmi); Transfer Orbit Perigee Altitude ≥ 166.7 (90 nmi); Orbit Inclination = 27.0°; Argument of Perigee = 180°

Table 2.10-2 Atlas IIIB (DEC) Performance to Reduced Inclination Transfer Orbit—PSW vs Orbit Inclination

Payload Systems Weight, kg (lb)				
	Atlas IIIB			
Inclination, °	MRS		GCS	
18.00	3,826	(8,434)	3,730	(8,223)
18.50	3,890	(8,575)	3,793	(8,362)
19.00	3,952	(8,713)	3,854	(8,497)
19.50	4,012	(8,846)	3,914	(8,628)
20.00	4,071	(8,975)	3,971	(8,755)
20.50	4,127	(9,099)	4,027	(8,877)
21.00	4,182	(9,220)	4,080	(8,996)
21.50	4,235	(9,336)	4,132	(9,110)
22.00	4,285	(9,447)	4,182	(9,219)
22.50	4,332	(9,551)	4,228	(9,322)
23.00	4,377	(9,649)	4,272	(9,418)
23.50	4,418	(9,741)	4,313	(9,508)
24.00	4,456	(9,824)	4,350	(9,590)
24.50	4,491	(9,901)	4,384	(9,665)
25.00	4,522	(9,969)	4,415	(9,733)
25.50	4,549	(10,029)	4,442	(9,792)
26.00	4,573	(10,082)	4,465	(9,844)
26.50	4,593	(10,126)	4,484	(9,887)
27.00	4,609	(10,161)	4,500	(9,920)
27.50	4,621	(10,187)	4,512	(9,947)
28.00	4,628	(10,204)	4,519	(9,964)
28.50	4,633	(10,213)	4,523	(9,972)
29.00	4,632	(10,211)	4,522	(9,970)
29.50	4,628	(10,203)	4,519	(9,963)
30.00	4,624	(10,195)	4,515	(9,954)

Note: Large Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude ≥ 148.2 km (80 nmi); Transfer Orbit Perigee Altitude ≥ 166.7 (90 nmi); Transfer Orbit Apogee Altitude = 35,788 km (19,324 mn); Argument of Perigee = 180°

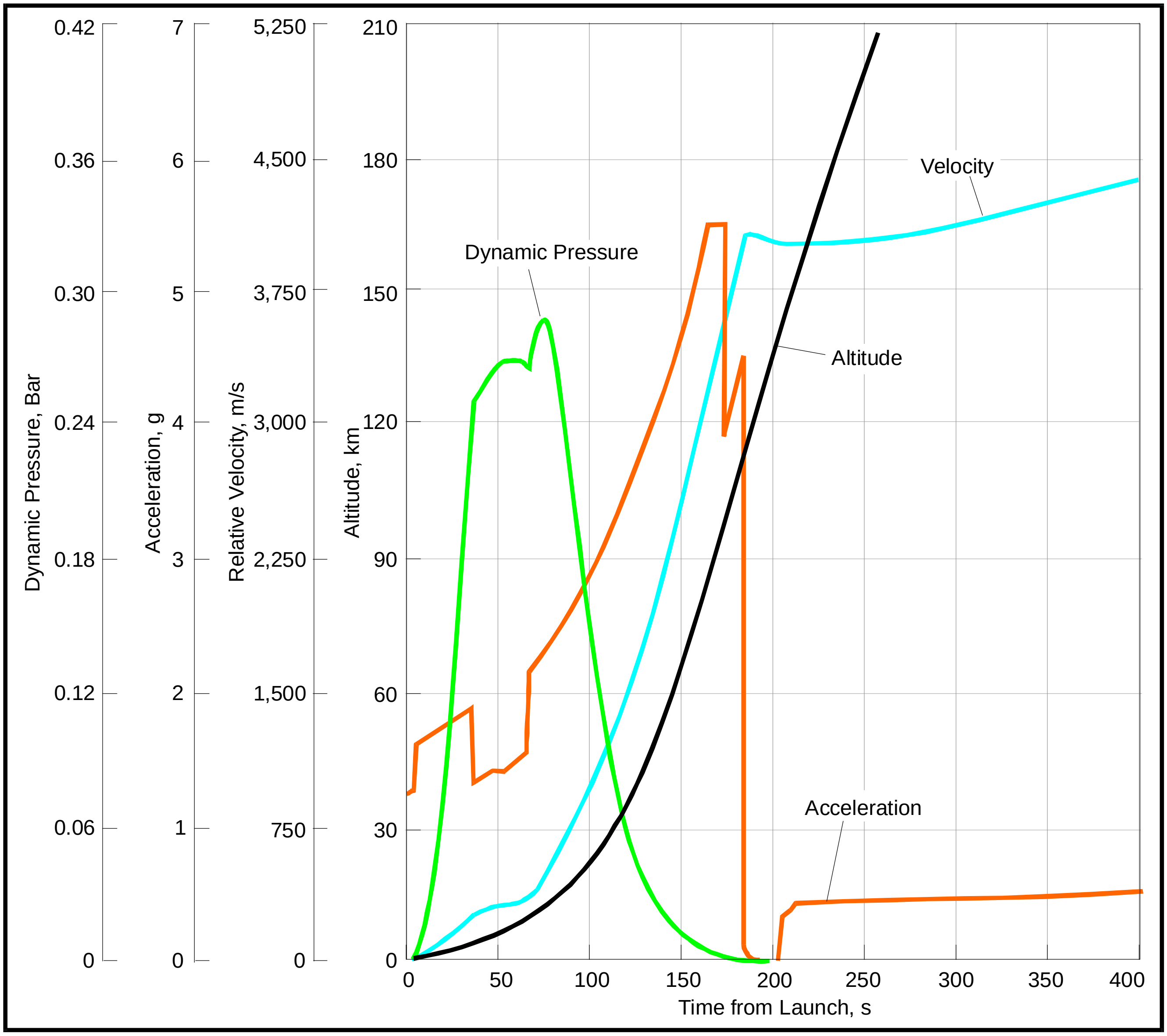


Figure 2.11-1 Atlas IIIB (SEC) Nominal Ascent Data

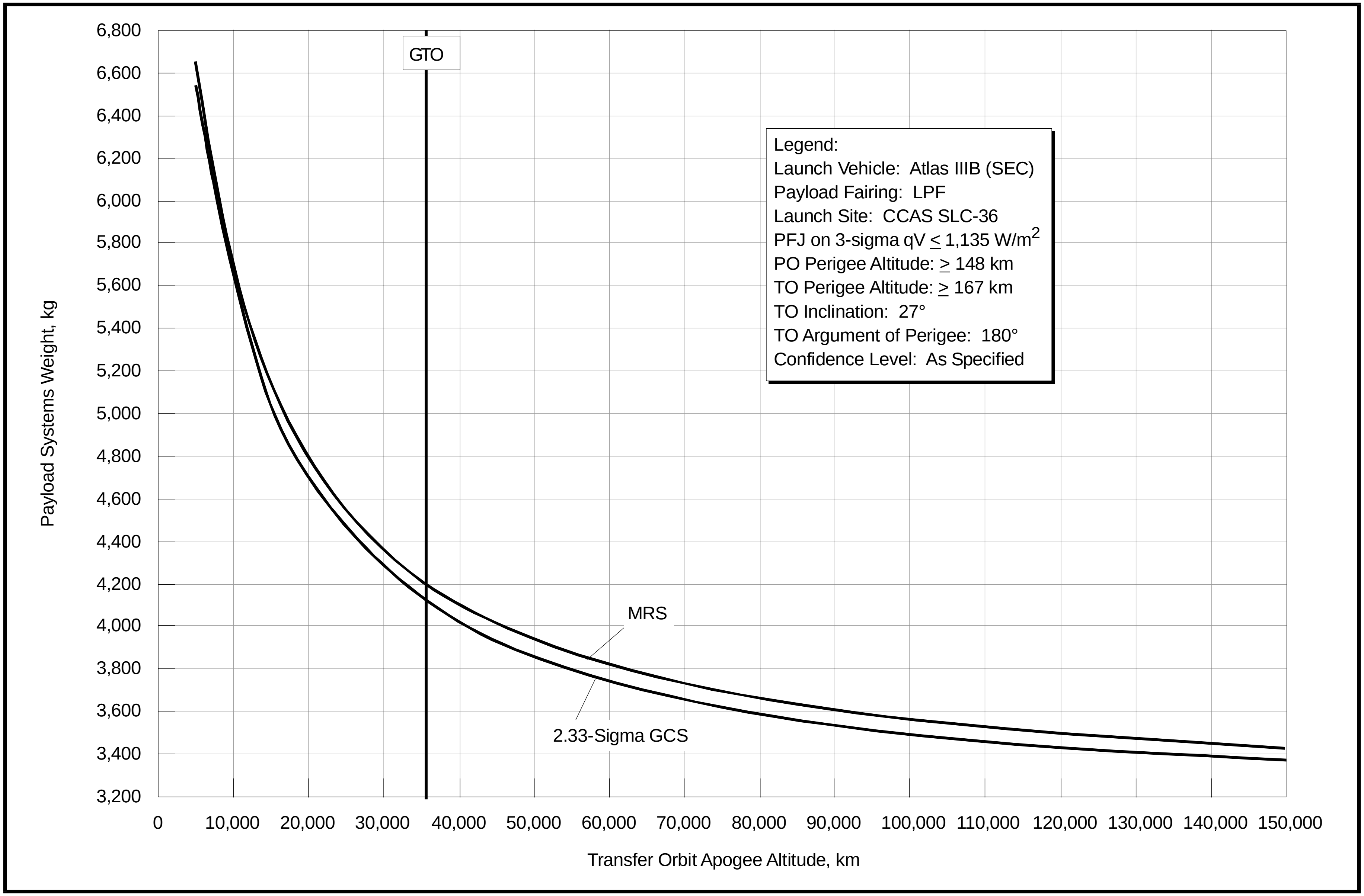


Figure 2.11-2a Atlas IIIB (SEC) CCAS Performance to Elliptical Transfer Orbit (Metric)

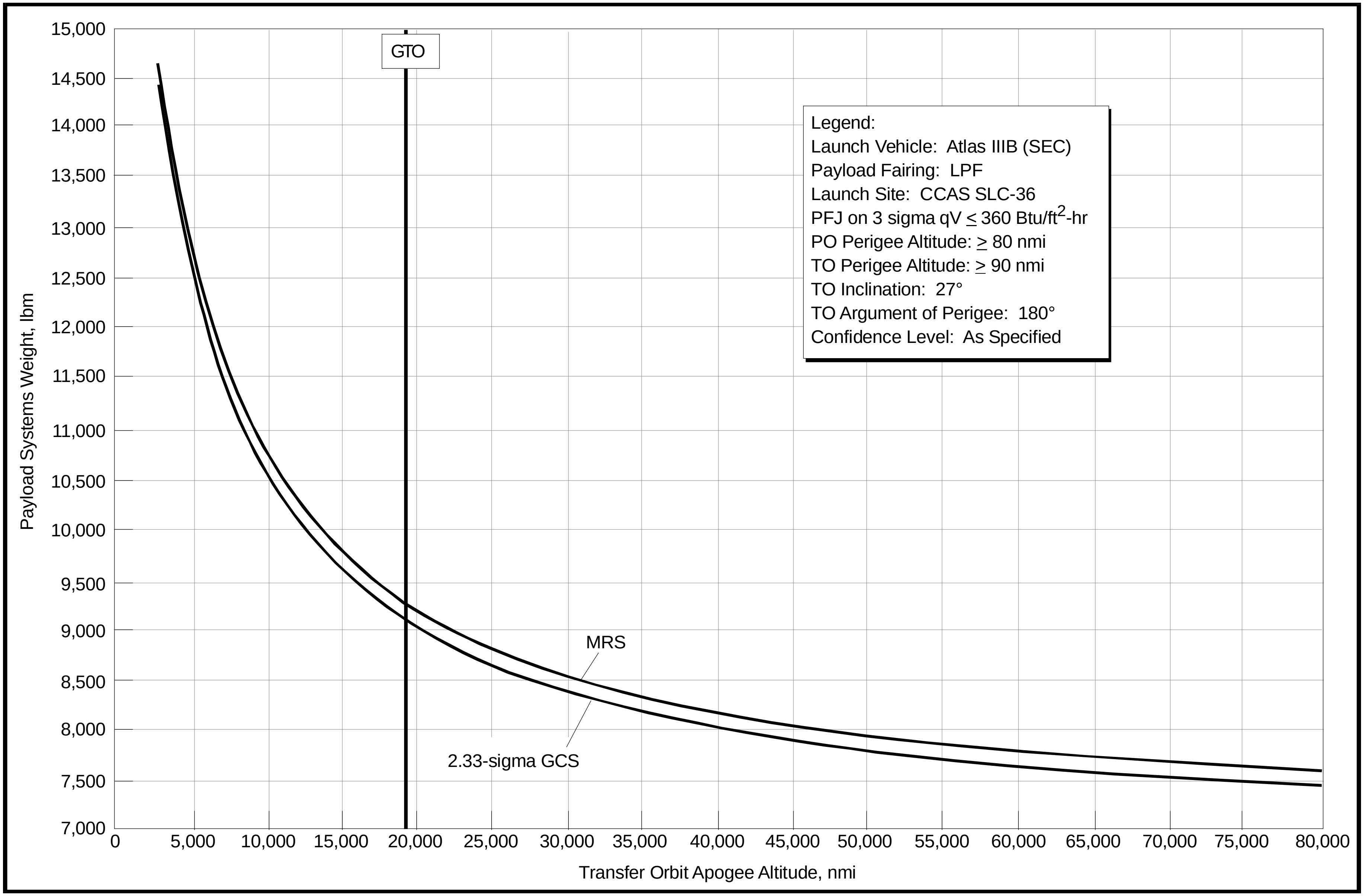


Figure 2.11-2b Atlas IIIB (SEC) CCAS Performance to Elliptical Transfer Orbit English)

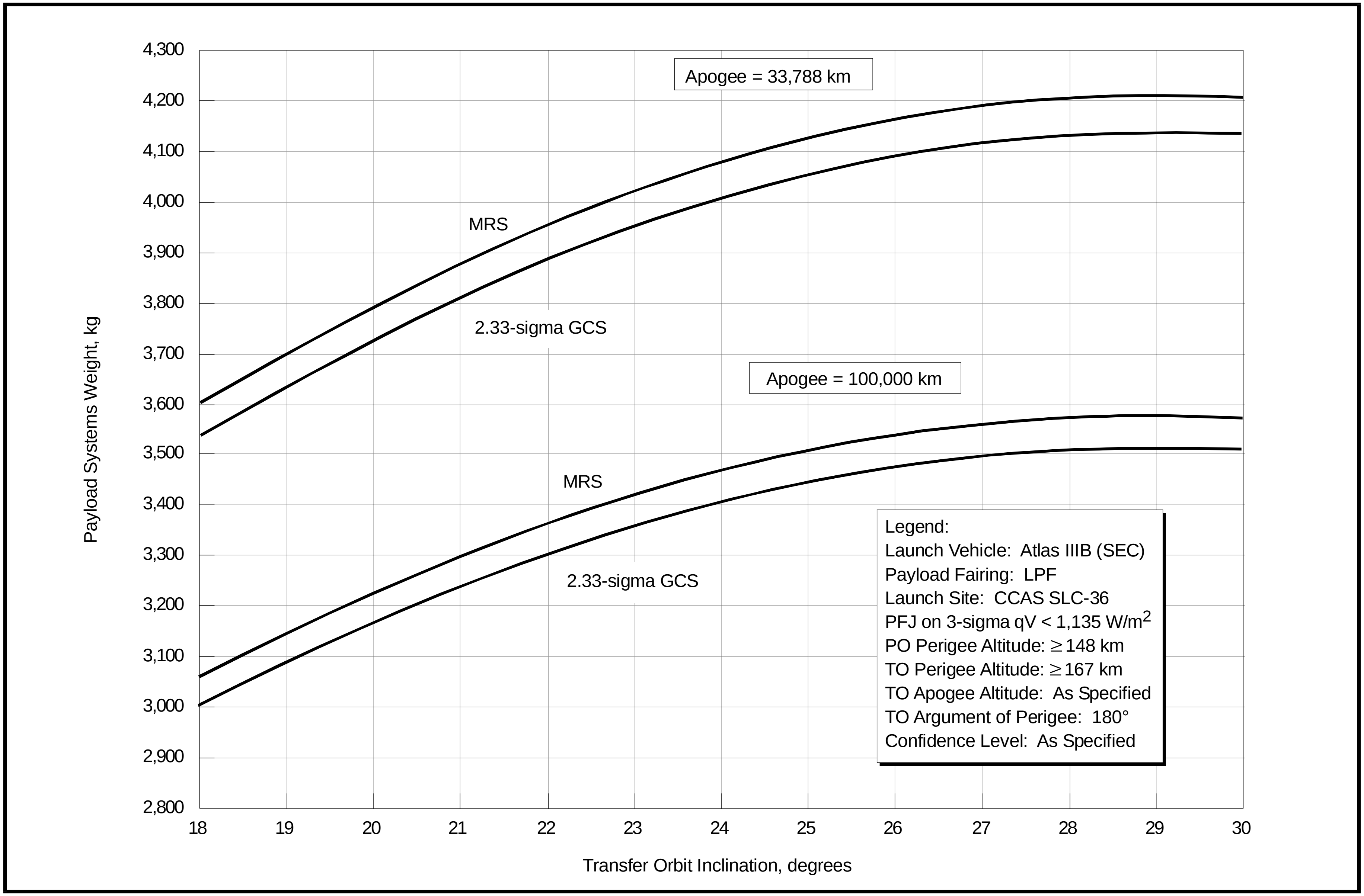


Figure 2.11-3a Atlas IIIB (SEC) CCAS Reduced Inclination Elliptical Transfer Orbit (Metric)

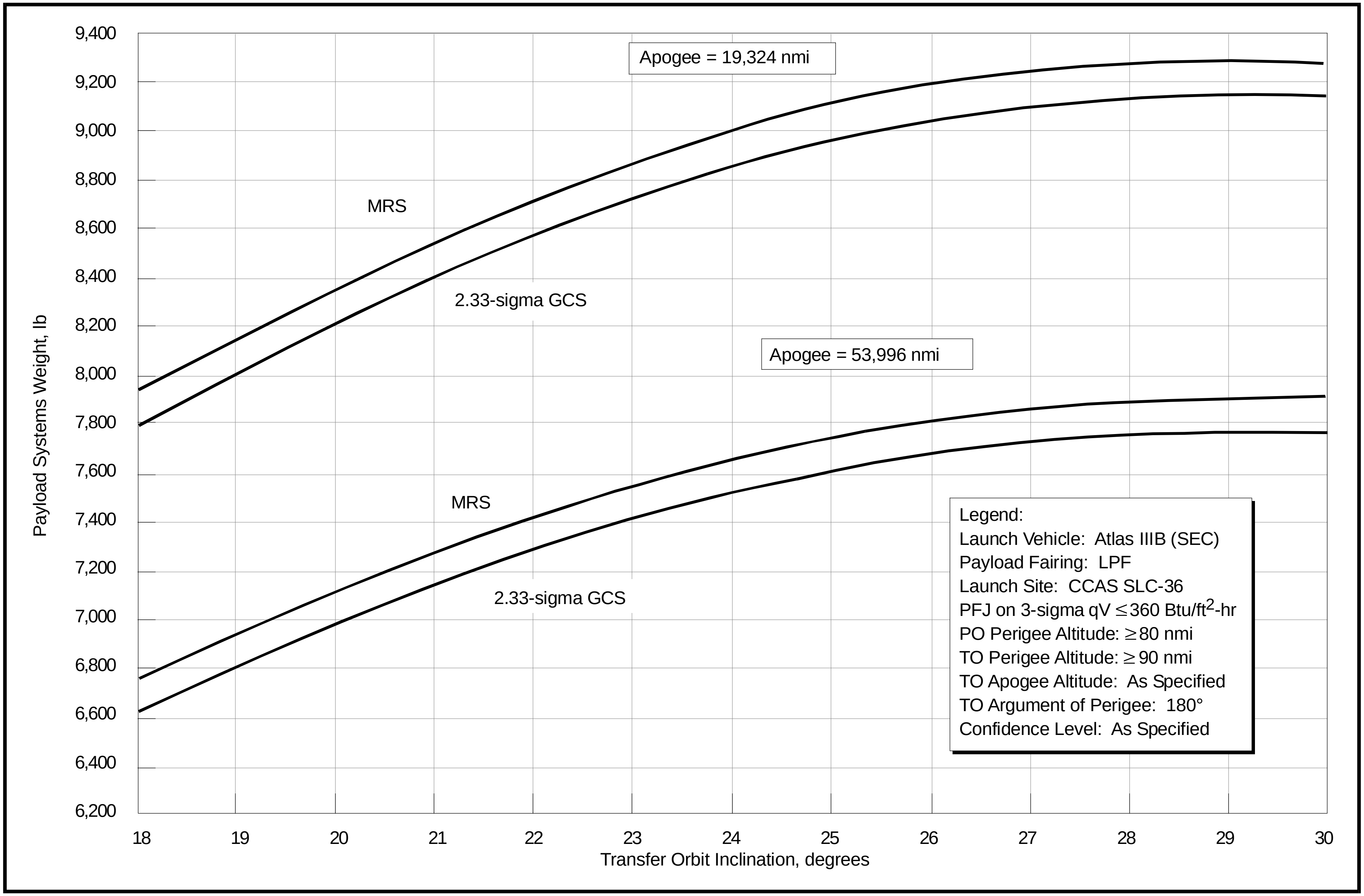


Figure 2.11-3b Atlas IIIB (SEC) CCAS Reduced Inclination Elliptical Transfer Orbit (English)

Table 2.11-1 Atlas IIIB (SEC) Performance to Reduced Inclination Transfer Orbit—PSW vs Orbit Inclination

Apogee Altitude		Payload System Weight, kg (lb) Atlas IIB-SEC			
km	(nmi)	MRS		GCS	
150,000	(80,993.5)	3,430	(7,561)	3,365	(7,419)
145,000	(78,293.7)	3,439	(7,582)	3,374	(7,439)
140,000	(75,594.0)	3,449	(7,604)	3,384	(7,461)
135,000	(72,894.2)	3,460	(7,628)	3,395	(7,485)
130,000	(70,194.4)	3,472	(7,654)	3,406	(7,510)
125,000	(67,494.6)	3,484	(7,681)	3,419	(7,538)
120,000	(64,794.8)	3,498	(7,711)	3,432	(7,567)
115,000	(62,095.0)	3,512	(7,743)	3,447	(7,599)
110,000	(59,395.2)	3,528	(7,778)	3,462	(7,633)
105,000	(56,695.5)	3,545	(7,816)	3,480	(7,671)
100,000	(53,995.7)	3,564	(7,858)	3,498	(7,712)
95,000	(51,295.9)	3,585	(7,904)	3,519	(7,757)
90,000	(48,596.1)	3,608	(7,954)	3,541	(7,807)
85,000	(45,896.3)	3,633	(8,010)	3,566	(7,861)
80,000	(43,196.5)	3,661	(8,072)	3,594	(7,923)
75,000	(40,496.8)	3,693	(8,142)	3,625	(7,992)
70,000	(37,797.0)	3,729	(8,220)	3,660	(8,069)
65,000	(35,097.2)	3,769	(8,310)	3,700	(8,158)
60,000	(32,397.4)	3,816	(8,413)	3,746	(8,259)
55,000	(29,697.6)	3,870	(8,532)	3,800	(8,377)
50,000	(26,997.8)	3,933	(8,671)	3,862	(8,514)
45,000	(24,298.1)	4,009	(8,838)	3,937	(8,679)
40,000	(21,598.3)	4,100	(9,039)	4,027	(8,877)
35,788	(19,324.0)	4,193	(9,244)	4,119	(9,080)
35,000	(18,898.5)	4,213	(9,287)	4,138	(9,122)
30,000	(16,198.7)	4,354	(9,600)	4,278	(9,432)
25,000	(13,498.9)	4,540	(10,009)	4,461	(9,835)
20,000	(10,799.1)	4,791	(10,562)	4,709	(10,382)
15,000	(8,099.4)	5,151	(11,355)	5,065	(11,166)
10,000	(5,399.6)	5,705	(12,578)	5,614	(12,376)
5,000	(2,699.8)	6,659	(14,681)	6,561	(14,465)

Note: Large Payload Fairing Jettison at 3-sigma qv<1,135 W/m2^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude = 148.2 km (80 nmi); Transfer Orbit Perigee Altitude = 166.7 km (90 nmi); Orbit Inclination = 27.0°; Argument of Perigee = 180°

Table 2.11-2 Atlas IIIB (SEC) Elliptical Transfer Orbit Performance—PSW vs Apogee Altitude

Inclination, °	Payload Systems Weight, kg (lb) Atlas IIIB-SEC			
	MRS		GCS	
18.00	3,602	(7,942)	3,534	(7,792)
18.50	3,653	(8,054)	3,585	(7,903)
19.00	3,702	(8,162)	3,633	(8,010)
19.50	3,750	(8,267)	3,680	(8,114)
20.00	3,795	(8,368)	3,725	(8,213)
20.50	3,839	(8,464)	3,769	(8,308)
21.00	3,881	(8,556)	3,810	(8,399)
21.50	3,920	(8,643)	3,849	(8,485)
22.00	3,958	(8,725)	3,886	(8,567)
22.50	3,993	(8,802)	3,920	(8,643)
23.00	4,025	(8,874)	3,953	(8,714)
23.50	4,056	(8,941)	3,983	(8,780)
24.00	4,083	(9,002)	4,010	(8,840)
24.50	4,108	(9,057)	4,035	(8,895)
25.00	4,131	(9,107)	4,057	(8,944)
25.50	4,151	(9,150)	4,077	(8,987)
26.00	4,167	(9,188)	4,093	(9,024)
26.50	4,182	(9,219)	4,107	(9,055)
27.00	4,193	(9,244)	4,119	(9,080)
27.50	4,201	(9,263)	4,127	(9,098)
28.00	4,208	(9,276)	4,133	(9,112)
28.50	4,212	(9,286)	4,137	(9,121)
29.00	4,214	(9,290)	4,139	(9,125)
29.50	4,213	(9,287)	4,138	(9,123)
30.00	4,209	(9,279)	4,134	(9,115)

Note: Large Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude = 148.2 km (80 nmi); Transfer Orbit Perigee Altitude = 166.7 (90 nmi); Transfer Orbit Apogee Altitude = 35,788 km (19,324 nmi); Argument of Perigee = 180°

3.0 ENVIRONMENTS

This section describes the environments to which the spacecraft is exposed. Detailed environmental data are provided for Cape Canaveral Air Station (CCAS) payload facilities, Space Launch Complex (SLC) 36, and our SLC-3 facility at Vandenberg Air Force Base (VAFB).

Prelaunch environments are described in Section 3.1; flight environments are described in Section 3.2; and spacecraft tests requirements are described in Section 3.3.

3.1 PRELAUNCH ENVIRONMENTS

3.1.1 Thermal

The spacecraft thermal environment is controlled during prelaunch activity, maintained during ground transport, and controlled after mate to the launch vehicle.

Environments in the spacecraft processing areas at Astrotech/CCAS are controlled at 21-27°C (70-80°F) and 50 ±5% relative humidity. Portable air conditioning units are available to further cool test equipment or spacecraft components as required.

During ground transport at CCAS, the temperature within the payload fairing (PLF) remains between 4-29°C (40-85°F), with positive pressure provided by a gaseous nitrogen (GN_2) purge. If required, an optional environmental control unit can maintain air temperatures between 10-25°C (50-77°F). In both cases, the relative humidity remains at or below 60%. At VAFB SLC-3, the transporter controls temperature between 10-27°C (50-80°F) with conditioned air and a GN_2 backup system.

During hoisting operations the encapsulated spacecraft is purged with dry GN_2 with relative humidity at or below 60%. If required, a high-flow rate conditioned air system can be used during hoist operations at CCAS.

After spacecraft mate to Centaur, gas conditioning is provided to the PLF at the required temperature, humidity, and flowrate. Air with a maximum dew point of 4°C (40°F) is used until approximately 2 hours before launch, after which GN_2 with a maximum dew point of -37°C (-35°F) is used. Table 3.1.1-1 summarizes prelaunch gas conditioning temperature capabilities for the nominal configuration and for a fairing with a mission-peculiar thermal shield. These environments envelope our current CCAS capabilities and those expected for VAFB. The optional thermal shield allows greater control over PLF internal temperatures during prelaunch gas conditioning. The shield consists of a noncontaminating membrane attached to the inboard surfaces of the PLF frames as shown in Figure 3.1.1-1. Mission-peculiar arrangements for dedicated purges of specific components can be provided.

The gas to the payload area is supplied through a ground/airborne disconnect on the PLF and is controlled by prime and backup environmental control units. These units provide air or GN_2 conditioned to the following parameters (Fig. 3.1.1-2):

Table 3.1.1-1 Gas Conditioning Capabilities

			Temperature Range Inside Payload Fairing**			
			MPF		LPF & EPF	
	Inlet Temp Capability*	**Flowrate Capability**	**Baseline**	**With Thermal Shield**	**Baseline**	**With Thermal Shield**
Inside Tower	10-29°C (50-85°F)	50-72.7 kg/min (110-160 lb/min)	12-24°C (54-75°F)	17-20°C (62-68°F)	LPF: 11-26°C (51-78°F) EPF: 10-26°C (50-79°F)	LPF: 16-21°C (61-69°F) EPF: 15-21°C (60-70°F)
After Tower Removal	10-29°C (50-85°F)	50-72.7 kg/min (110-160 lb/min)	8-28°C (47-83°F)	16-21°C (61-69°F)	LPF: 4-31°C (40-87°F) EPF: 3-32°C (38-89°F)	LPF: 16-21°C (60-70°F) EPF: 14-22°C (58-72°F)

Note: * Inlet temperature is adjustable (within system capability) according to spacecraft requirements.
** Temperature ranges are based on worst-case minimum & maximum external heating environments, Above ranges assume a 1.21-kg/s (160-lb/min) PLF ECS flow rate.

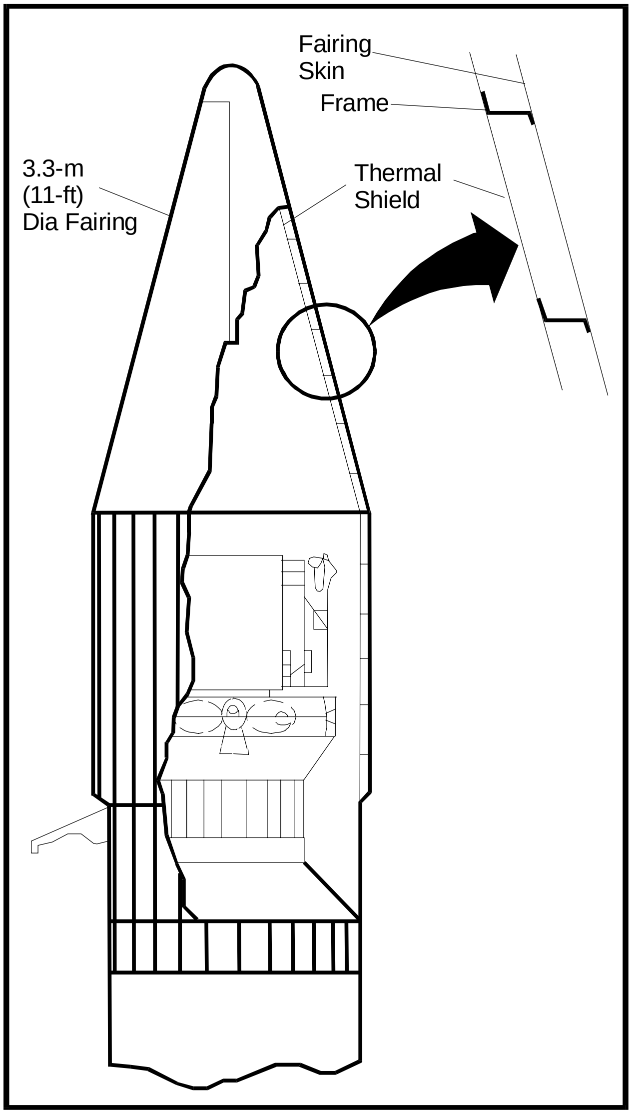


Figure 3.1.1-1 The Thermal Shield Option

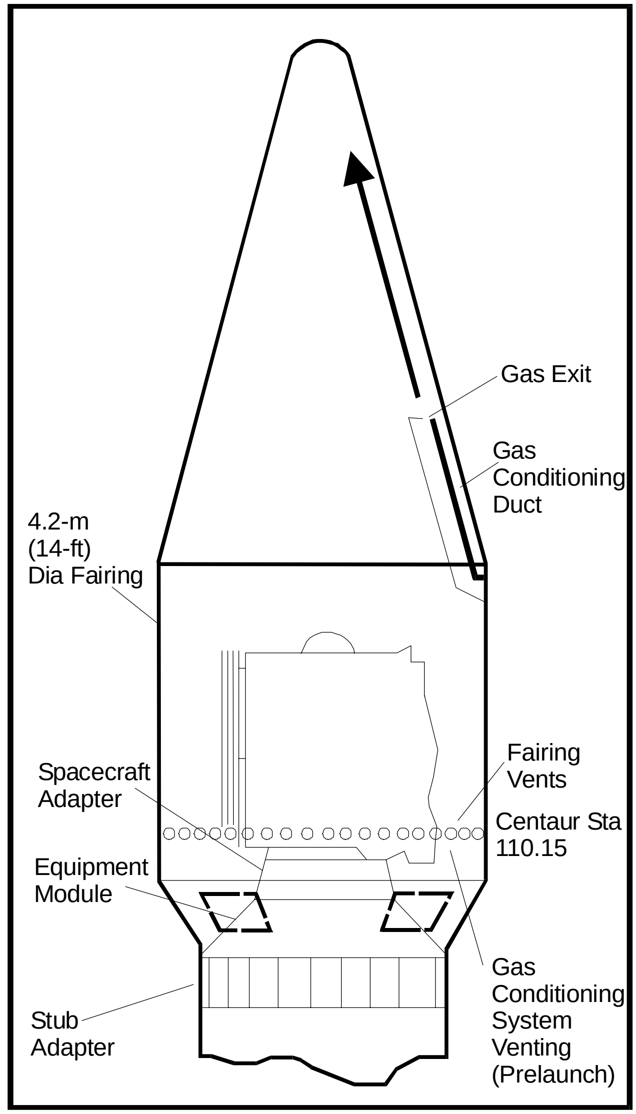


Figure 3.1.1-2 The PLF air conditioning system provides a controlled thermal environment during ground checkout and prelaunch.

1) Cleanliness: Class 10,000 per FED-STD-209B and Class 5,000 available on request at CCAS, Class 5,000 per FED-STD-209D at VAFB;
2) Inlet Temperature: Setpoint from 10-29°C (50-85°F);
3) Inlet Temperature Control: ±2°C (±3°F) at CCAS, ±1.1°C (±2°F) at VAFB;
4) Filtration: 0.3 microns at CCAS, 99.97% HEPA at VAFB;
5) Flow Rate: Adjustable to any setpoint between 50-72.7 kg/min (110-160 lb/min) at CCAS, between 22.7-72.7 kg/min (50-160 lb/min) at VAFB;
6) Dewpoint (Maximum): 4.4°C (40°F) air, -37.2°C (-35°F) GN_2.

Internal ducting directs the gas upward to prevent direct impingement on the spacecraft. The conditioning gas is vented to the atmosphere through one-way flapper doors in the aft end of the LPF and EPF, or in the split barrel when the MPF is used. This baseline design ensures that average gas velocities across spacecraft components are less than 4.9 m/s (16 ft/s) with medium fill-factor spacecraft that extend into the lower conical section of the payload fairing. For large fill-factor spacecraft, gas impingement velocities could conservatively approach 9 m/s (30 ft/s). If required, a computational fluid dynamics analysis can be performed to verify mission-unique gas impingement velocity limits.

3.1.2 Radiation and Electromagnetics

To ensure that electromagnetic compatibility (EMC) is achieved for each launch, the electromagnetic (EM) environment is thoroughly evaluated. The launch services customer will be required to provide all spacecraft data necessary to support EMC analyses (Appendix C tables) used for this purpose.

3.1.2.1 Launch Vehicle-Generated Radio Environment—Launch vehicle intentional transmissions are limited to the S-band telemetry transmitters at 10.8 dBW or 15.45 dBW (for operation with the Tracking and Data Relay Satellite System [TDRSS]) and the C-band beacon transponder at 28.5 dBW (peak) or 26 dBm (average).

Figure 3.1.2.1-1 shows the absolute worst-case antenna radiation environment generated by the launch vehicle. The curve is based on transmitter fundamental and "spurious output" requirements and assumes (1) maximum transmit output power, (2) maximum antenna gain and minimum passive line loss (i.e., measured values at transmit frequency applied across the entire frequency spectrum), and (3) straight-line direct radiation. Actual levels encountered by the satellite (influenced by many factors) can only be less than the levels depicted in the figure. Initial reductions are provided to the user on determination of which launch vehicle and payload adapter will be used.

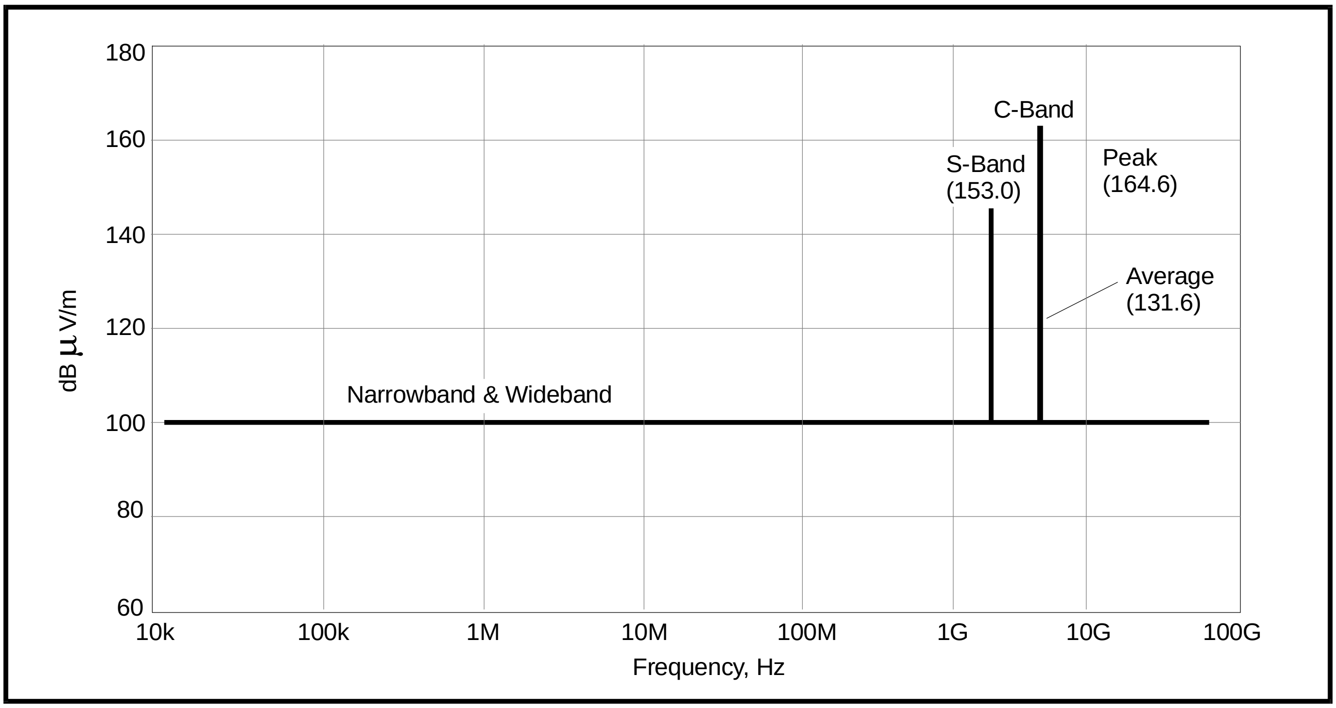


Figure 3.1.2.1-1 Launch Vehicle Field Radiation from Antennas

3.1.2.2 Launch Vehicle-Generated Electromagnetic Environment—The unintentional EM environments generated by the launch vehicle at the satellite location are depicted in Figures 3.1.2.2-1, 3.1.2.2-2, and 3.1.2.2-3. Actual levels to be encountered by the satellite will approach the typical levels depicted in the figures.

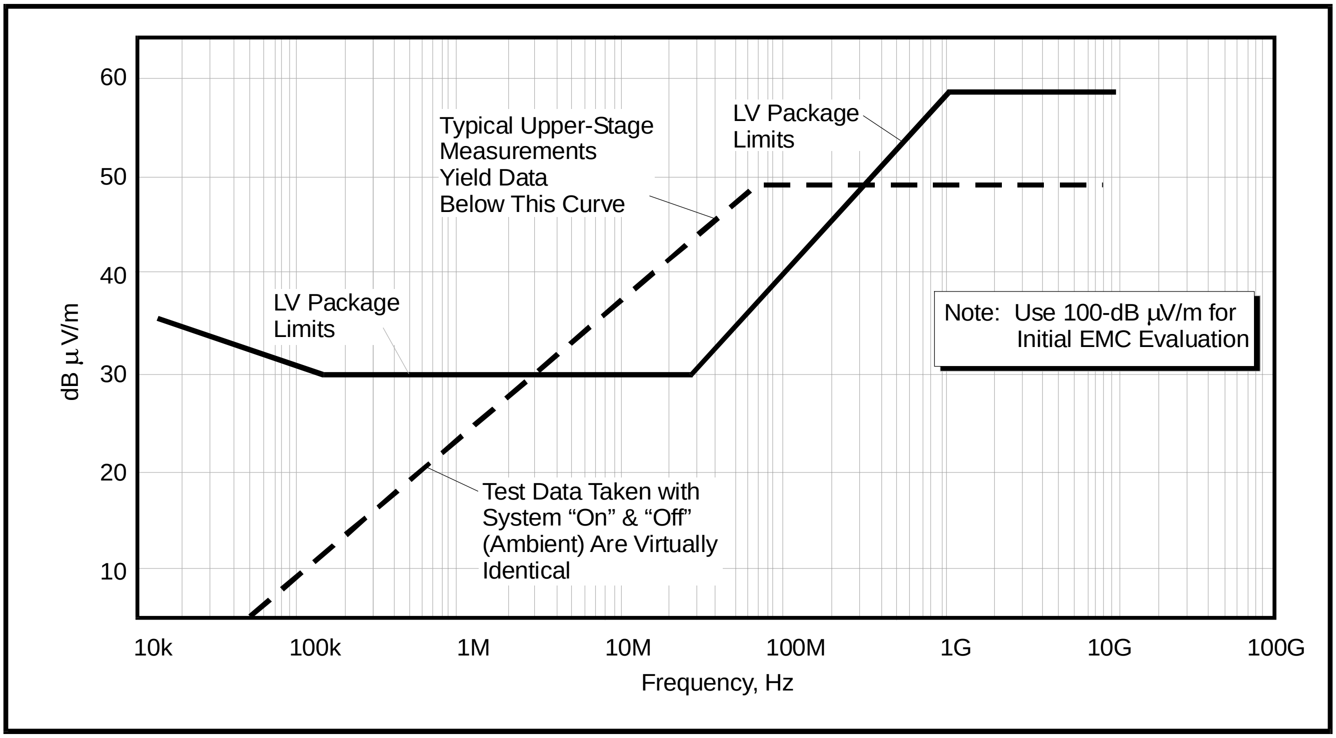


Figure 3.1.2.2-1 Launch Vehicle Electrical Field Radiation (Narrowband)

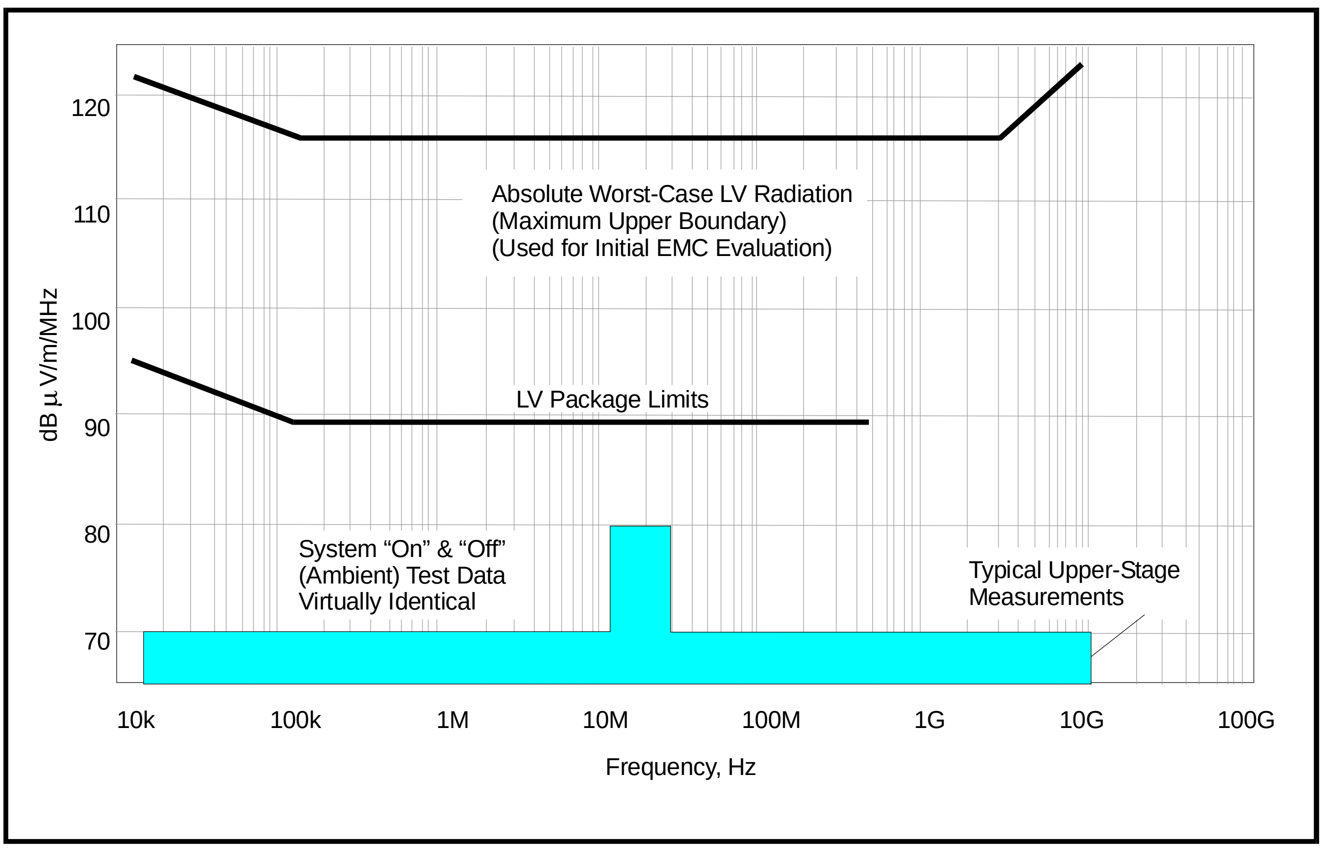


Figure 3.1.2.2-2 Launch Vehicle Electrical Field Radiation (Wideband)

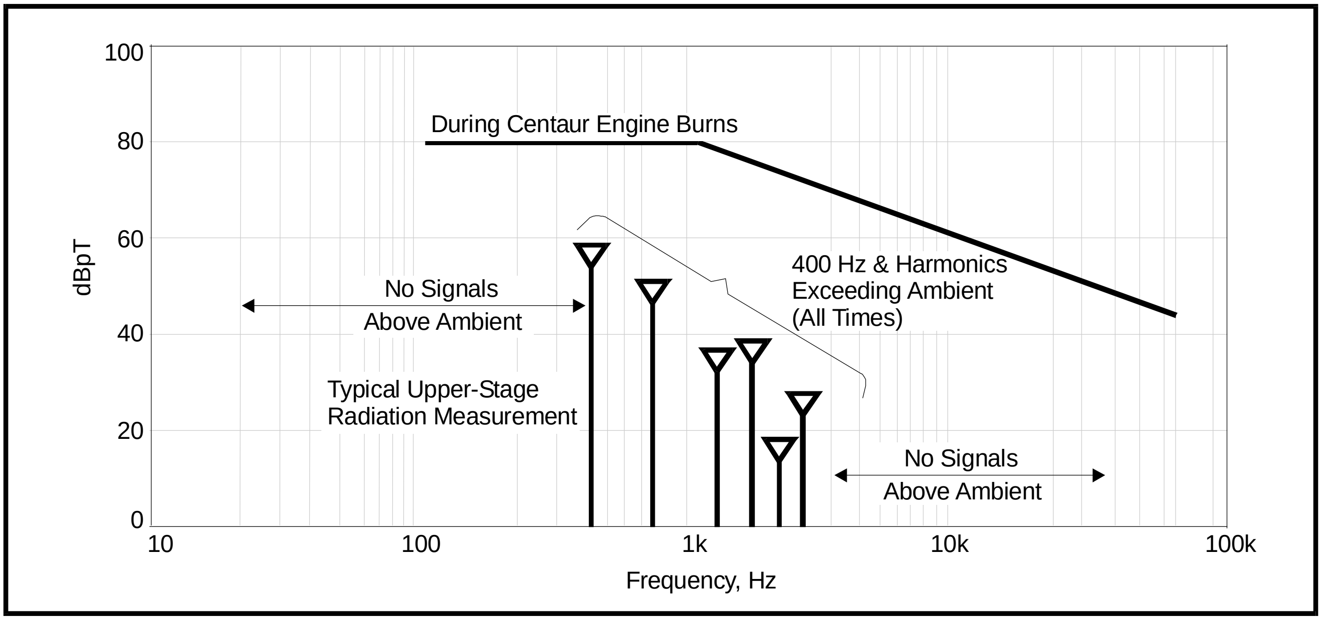


Figure 3.1.2.2-3 Launch Vehicle Magnetic Field Spurious Radiation (Narrowband)

3.1.2.3 Launch Range Electromagnetic Environment—The EM environment of the launch range is based on information contained in TOR-95 (5663)-1. An EMC analysis will be performed to ensure EMC of the spacecraft/launch vehicle with the range environment.

Table 3.1.2.3-1 documents EM emitters in the vicinity of CCAS. Field intensities at both Astrotech and Launch Complex 36 are also provided. Data can be provided for other locations on request.

Table 3.1.2.3-2 lists EM emitters in the vicinity of VAFB. These data are based on an EM site survey conducted at Atlas Launch Complex 3E in June 1997.

Table 3.1.2.3-1 Worst-Case RF Environment for CCAS

Combined SLC-36				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, v/m**	**Duty Cycle**	**Mitigation**
Radar 0.14	5,690	111.1	0.0016	Procedure Mask
Radar 1.16	5,690	169.5	0.00064	Procedure Mask
Radar 1.39	5,690 & 5,800	82.0	0.005	Procedure Mask
Radar 19.14	5,690	169.2	0.0016	Procedure Mask
Radar 19.17	5,690	28.7	0.0008	Procedure Mask
Radar 28.14	5,690	17.9	0.0016	Topography
Radar 1.8	9,410	4.7	0.0012	No (OD10040)
Radar ARSR-4	1,244.06 & 1,326.92	1.6	0.00006	None
Radar GPN-20	2,750 & 2,840	5.9	0.0008	None
WSR-74C	5,625	15.8	0.0064	None
WSR-88D	2,879	14.5	0.006	None
GPS Gnd Station	1,784	6.5	CW	Ops Min 3°
NASA STDN	2,025 & 2,120	5.8	CW	None
TVCF	1,761 & 1,842	5.4	CW	Ops Min 5°
Astrotech				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, v/m**	**Duty Cycle**	**Mitigation**
Radar 0.14	5,690	77.0	0.0016	Procedure Mask
Radar 1.16	5,690	26.3	0.00064	Procedure Mask[1]
Radar 1.39	5,690 & 5,800	18.5	0.005	Procedure Mask
Radar 19.14	5,690	116.7	0.0016	Procedure Mask[1]
Radar 19.17	5,690	36.6	0.0008	Procedure Mask[1]
Radar 28.14	5,690	16.5	0.0016	Topography

Table 3.1.2.3-1 (concl)

Astrotech				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, v/m**	**Duty Cycle**	**Mitigation**
Radar 1.8	9,410	0.8	0.0012	Procedure Mask
Radar ARSR-4	1,244.06 & 1,326.92	1.7	0.00006	None
Radar GPN-20	2,750 & 2,840	4.1	0.0008	None
WSR-74C	5,625	10.6	0.0064	None
WSR-88D	2,879	12.3	0.006	None
GPS Gnd Station	1,784	0.9	CW	Ops Min 3°
NASA STDN	2,025 & 2,120	7.1	CW	None
TVCF	1,761 & 1,842	0.9	CW	Ops Min 5°

Note: Sources Taken from Aerospace Report TOR-95(5663)-1, Rev 1, "Radio Frequency Environment, Eastern Range," 8/97
[1]See TOR-95(5663)-1, Sect. 2.2.1 Astrotech
Avg v/m = Pk, v/m*sqrt (Duty Cycle); CW = Continuous Wave
Shaded Blocks Indicate Emitters Without Specific Mechanical or Software Mitigation Measures
In-Flight Levels for Tracking Radars (0.14, 1.16, 1.39, 19.14, & 19.17) Are 20 v/m

Table 3.1.2.3-2 Worst-Case RF Environment for VAFB

SLC-3				
Emitter Name	**Frequency, MHz**	**Measured Intensity, v/m**	**Duty Cycle**	**Mitigation**
CT-2	416.5	2.54	CW	None
ARSR-4	1,319.15	1.4	0.0006	Procedure Mask
AN/GPN-12	2,800	8.07	0.000835	None
FPS-16-1	5,725	150.66	0.001	Procedure Mask
FPS-77	5,450 & 5,650	(Est)	0.0064	None
HAIR	5,400-5,900	279.25	0.002	Procedure Mask
MOTR	5,400-5,900	31.62	0.005	Procedure Mask
TPQ-18	5,840	874.0	0.0016	Procedure Mask
TPQ-39	9,180	50.47	0.00024	Procedure Mask
NEXRAD	2,890	(Est)	0.002	None
DRWP	404.37	NSO	0.067	Vertical Emitter
Astrotech				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, v/m**	**Duty Cycle**	**Mitigation**
CT-2	416.5	TBD	TBD	None
ARSR-1E	1,320	TBD	TBD	Procedure Mask
AN/GPN-12	2,800	TBD	TBD	None
FPS-16-1	5,725	TBD	TBD	Procedure Mask
FPS-77	5,450 & 5,650	TBD	TBD	None
HAIR	5,400-5,900	TBD	TBD	Procedure Mask
MOTR	5,400-5,900	TBD	TBD	Procedure Mask
TPQ-18	5,840	TBD	TBD	Procedure Mask
TPQ-39	9,180	TBD	TBD	Procedure Mask
NEXRAD	2,890	TBD	TBD	None
DRWP	404.37	TBD	TBD	None

Note: Theoretical intensity data taken from the "Field Strength Measurement Data Summary at the Western Range" March 7, 1994, prepared for the 30th Space Wing.
CT: Command Transmitter
Estimated (Est) values are theoretical and were not operating when measurement were taken.

3.1.2.4 Spacecraft-Generated Environment Limitation—During ground and launch operation timeframes through spacecraft separation, any spacecraft electromagnetic interference (EMI) radiated emissions (including antenna radiation) should not exceed values depicted in Figure 3.1.2.4-1. Launch vehicle/spacecraft external interfaces (EMI-conducted emissions) must be examined individually.

Each spacecraft will be treated on a mission-peculiar basis. Assurance of the launch vehicle/spacecraft EMC with respect to payload emissions will be a shared responsibility between Lockheed Martin and the individual spacecraft contractor.

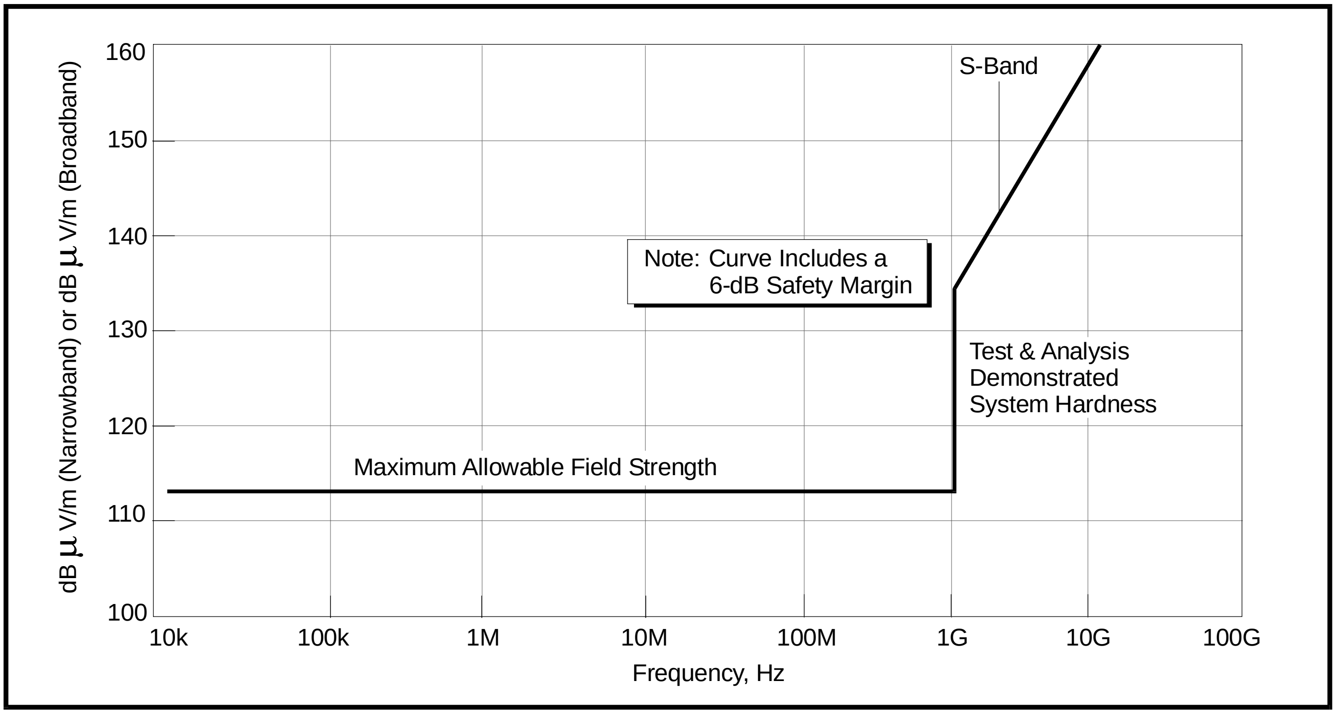


Figure 3.1.2.4-1 Spacecraft Electric Field Radiation Impingement on Launch Vehicle

3.1.3 Contamination and Cleanliness

Launch vehicle hardware that comes into contact with the payload environment has been designed and manufactured according to strict contamination control guidelines. This hardware is defined as contamination-critical and includes the Centaur forward equipment area, the payload adapter, and the interior surface of the PLF. In addition, ground operations at the launch site have been designed to ensure a clean environment for the spacecraft. A comprehensive Contamination Control Plan has been written to identify these requirements and procedures. Some guidelines and practices used in the plan follow.

1) Precautions are taken during manufacture, assembly, test, and shipment to prevent contamination accumulations on contamination-critical launch vehicle surfaces.
2) Launch vehicle contamination-critical surfaces are cleaned using approved material and procedures. A final visual inspection for cleanliness conformance occurs just before encapsulation.
3) The encapsulation process is performed in a facility that is environmentally controlled to Class 100,000 conditions per FED-STD-209B. All handling equipment is cleanroom-compatible and will be cleaned and inspected before entering the facility. These environmentally controlled conditions are available for all remote encapsulation facilities (i.e., Astrotech).
4) Personnel controls are used to limit access to the PLF to maintain spacecraft cleanliness. Contamination control training is provided to all launch vehicle personnel working in or around the

encapsulated PLF. Lockheed Martin provides similar training to spacecraft personnel working on the spacecraft while on the launch tower to ensure that they are familiar with the procedures.

3.1.3.1 Contamination Control Before Launch Site Delivery

Design and Assembly—Contamination control principles are used in design and manufacturing processes to limit the amount of contamination from launch vehicle components. Interior surfaces include maintainability features to facilitate the removal of manufacturing contaminants. The Centaur vehicle is assembled in a Class 100,000 facility to ensure that hardware surfaces, and in particular any entrapment areas, are maintained at an acceptable level of cleanliness before shipment to the launch site. Inspection points are provided to verify cleanliness throughout the assembly process.

Materials Selection—In general, materials are selected for contamination-critical hardware inside the PLF that will not become a source of contamination to the spacecraft. Metallic or nonmetallic materials that are known to chip, flake, or peel are prohibited from use. Materials that are cadmium-plated, zinc-plated, or made of unfused electro-deposited tin are restricted from use inside the PLF volume. Corrosion-resistant materials are selected wherever possible and dissimilar materials are avoided or protected according to MIL-STD-889B. Because most nonmetallic materials are known to exhibit some outgassing, these materials are evaluated against National Aeronautics and Space Administration (NASA) SP-R-0022 criteria before selection.

3.1.3.2 Contamination Control Before Spacecraft Encapsulation

Cleanliness Levels—Contamination-critical hardware surfaces are cleaned and inspected to specific criteria. These checks confirm the absence of all particulate and molecular contaminants visible to the unaided eye at a distance of 15.2-45.7 cm (6-18 in.) with a minimum illumination of 1,076 lumen/m^2 (100-ft candles). This criterion is Visibly Clean Level 2. Hardware that is cleaned to this criterion at the assembly plant is protected to maintain this level of cleanliness through shipping and encapsulation.

Contingency cleaning may also be required to reattain this level of cleanliness if the hardware becomes contaminated. Contingency cleaning procedures outside of the encapsulation facility before encapsulation are subject to Lockheed Martin engineering approval. Cleaning of the launch vehicle hardware that is required in the vicinity of the spacecraft must also be approved by the cognizant spacecraft engineer.

Certain payloads may require that contamination-critical hardware surfaces be cleaned to a level of cleanliness other than Visibly Clean Level 2. Because additional cleaning and verification may be necessary, these requirements are implemented on a mission-peculiar basis.

PLF Cleaning Techniques—Lockheed Martin recognizes that cleaning of large interior PLF surfaces depends on implementation of well-planned cleaning procedures. To achieve customer requirements, all cleaning procedures are verified by test and reviewed and approved by Material and Processes Engineering. Final PLF cleaning is performed in a Class 100,000 facility before encapsulation.

Cleanliness Verification—All contamination-critical hardware surfaces are visually inspected to verify Visibly Clean Level 2 criteria described above. The additional verification techniques shown below can be provided on a mission-unique basis:

1) Particulate Obscuration—Tape lift sampling,
2) Nonvolatile Residue (NVR)—Solvent wipe sampling,
3) Particulate and Molecular Fallout—Witness plates.

3.1.3.3 Contamination Control After Encapsulation

Contamination Diaphragm—After the two halves of the PLF are joined, encapsulation is completed by closing the aft opening with a ground support equipment (GSE) reinforced plastic film diaphragm. The doughnut-shaped diaphragm stretches from the payload adapter to the aft end of the PLF cylinder and creates a protected environment for the spacecraft through mating to the Atlas/Centaur. After the spacecraft is mated, the diaphragm remains in place until final PLF closeout when all flight doors are installed. This assists in protecting the spacecraft from possible contamination during Centaur operations performed at the launch tower after mating.

PLF Purge—After encapsulation, the PLF environment is continuously purged with filtered nitrogen or high-efficiency particulate air (HEPA)-filtered air to ensure the cleanliness of the environment.

Complex 36B—Access to the encapsulated spacecraft is performed from workstands situated on PLF Access Level 14. Work procedures and personnel control are established to maintain the spacecraft environment within the PLF to Class 100,000 standards. Garments are provided to personnel working inside the PLF to provide optimum cleanliness control as dictated by spacecraft requirements.

Complex 36A—Access to the encapsulated spacecraft is performed from workstands situated on PLF Access Level 14. Work procedures and personnel control are established to maintain the spacecraft environment within the PLF to Class 100,000 standards. Garments are provided to personnel working on this level to provide optimum cleanliness control as dictated by spacecraft requirements.

Complex 3E Controlled Work Area—An environmentally controlled area (ECA) is provided on Mobile Service Tower (MST) Levels 8 through 15. The conditioned air supply to this facility is HEPA-filtered and has the capability to control air temperature and humidity.

3.1.3.4 Payload Fairing Helium Environment in Prelaunch Operations—The volume between the LH_2 tank and the Centaur equipment module is purged with helium, while the vehicle is on the pad with the LH_2 tank loaded, to prevent condensation on the tank forward bulkhead. The Centaur equipment module is sealed; however, some helium leaks into the payload compartment. The helium mixes with the GN_2 being injected by the PLF and equipment module gas conditioning systems. At T-8 seconds, a pyroactivated helium vent door opens, venting the equipment module helium into the payload compartment. Gas conditioning and helium purge systems are terminated at T-0 in a normal launch. During ascent the payload compartment vents to negligible pressure approximately 3 minutes after launch. In case of an aborted launch attempt with the helium vent door open, the helium flow is shut off shortly after abort. The flow is allowed to continue during detanking if the vent door remains closed.

Measured helium concentrations above the Centaur equipment module are variable and no specific exposure level can be guaranteed until further testing is performed. Some solutions, such as GN_2 purge of the effected component, are available to helium sensitive spacecraft as a nonstandard service on a mission-unique basis. Long-term solutions are being investigated for future phase-in on a generic basis.

3.2 LAUNCH AND FLIGHT ENVIRONMENTS

This section describes general environmental conditions that may be encountered by a spacecraft during launch and flight with the Atlas launch vehicle. All flight environments defined in this section are maximum expected levels and do not include margins typically associated with qualification tests.

3.2.1 Spacecraft Design Loads

3.2.1.1 Design Load Factors—Design load factors (DLF) (Table 3.2.1.1-1) are used in preliminary design of primary structure and/or evaluation of the Atlas vehicle suitability for an existing spacecraft. Load factors are intended for application at the spacecraft's center of gravity (cg) to evaluate primary structure. The response of a spacecraft to launch vehicle transients will depend on its mass properties, stiffness, and amount of axial-to-lateral coupling. The load factors given are intended to provide a conservative design envelope for a typical spacecraft in the 1,814-kg (4,000-lb) to 4,500-kg (9,920-lb) weight class with first lateral modes above 10 Hz and first axial mode above 15 Hz. The load factors for preliminary design in Table 3.2.1.1-1 do not include a dynamic uncertainty factor (DUF). This factor is typically applied to preliminary coupled loads analysis responses to conservatively account for potential design changes or modal corrections before the final design load cycle. Although interface bending moments, shears, and axial forces calculated from preliminary design loads cycle (PDLC) analyses are generally enveloped by DLFs without applying a DUF, Lockheed Martin does not guarantee that this will be the case for all spacecraft and DUF combinations. Conservative design criteria treat DLF and PDLC responses with the same DUF. Consult Lockheed Martin if questions exist concerning this issue. Coupled loads analyses (CLA) during the integration activity will provide actual loads on the space vehicle for both primary and secondary structure. Load factors are separated into a quasi-steady-state and oscillatory dynamic. Total load factors in a direction are obtained by adding the steady-state and dynamic portion of the load factors.

3.2.1.2 Coupled Loads Analysis—Atlas/Centaur CLAs are performed to support the spacecraft contractor with accelerations, loads, and deflections for drawing release, test planning, and verification of minimum margins of safety.

The PDLC is typically performed with a spacecraft dynamic model representing preliminary sizing of hardware before drawing release or qualification testing. The determination of coupled responses early in the design and integration schedule allows optimization of spacecraft structure for Atlas environments and reduces the risk of costly redesign. Incorporation of a DUF to account for the preliminary nature of the dynamic characterization of the

Table 3.2.1.1-1 Spacecraft Limit Load Factors for Atlas IIA, IIAS, IIIA, and IIIB

Load Condition	Direction	Steady State, g	Dynamic, g
Launch	Axial	1.2	±1.1
	Lateral	—	±1.3 (IIAS, IIIA, IIIB)
		—	±1.0 (IIA)
Flight Winds	Axial	1.7—2.7	±0.8 (IIAS)
		1.3—2.7	±0.3 (IIA, IIIA, IIIB)
	Lateral	0.4	±1.6 (IIAS, IIIA, IIIB)
			±1.2 (IIA)
BECO/BPJ			
(Max Axial)	Axial	5.0	±0.5 (IIAS)
		5.5	±0.5 (IIA, IIIA, IIIB)
	Lateral	—	±0.5
(Max Lateral)	Axial	2.5-1.0	±1.0
	Lateral	—	±2.0 (IIA, IIAS) ± 1.5 (IIIA, IIIB)
SECO	Axial	2.0-0.0*	±0.4
	Lateral	—	±0.3
MECO			
(Max Axial)	Axial	4.8-0.0*	±0.5
	Lateral	—	±0.2
(Max Lateral)	Axial	0.0	±2.0
	Lateral	—	±0.6
Sign Convention			
Longitudinal Axis:	+ (Positive) = Compression		
	– (Negative) = Tension		
Pitch Axis:	± May Act in Either Direction		
Yaw Axis:	± May Act in Either Direction		
Lateral & Longitudinal Loading May Act Simultaneously During Any Flight Event			
Loading Is Induced Through the cg of the Spacecraft			
Note: * Decaying to Zero			

spacecraft is recommended. A DUF appropriate for preliminary models is typically in the range of 1.25 to 1.50, depending on company practice or policy.

The final design loads cycle (FDLC) requires a test-verified dynamic model of the spacecraft for determination of the maximum expected flight loads, as specified by the interface control document (ICD). These response data are provided for calculation of spacecraft minimum margins of safety.

Verification of minimum factors of safety is required by a combination of static and sinusoidal base acceleration testing of the spacecraft. Primary structural interfaces are most easily loaded during static testing, while the more complex dynamic responses are generated during the sinusoidal base acceleration test. Fidelity of CLA is specified in the 0- to 50-Hz range and notching of the base input in this frequency band is recommended. Notching philosophy is defined in early integration meetings.

3.2.2 Acoustics

The spacecraft is exposed to an acoustic environment throughout the boost phase of flight until the vehicle is out of the sensible atmosphere. Two portions of flight have significantly higher acoustic levels than the others. The highest acoustic level occurs for approximately 5 seconds during liftoff, when the acoustic energy of the engine exhaust is being reflected by the launch pad. The other significant level occurs for approximately 20 seconds during the transonic portion of flight and is due to transonic aerodynamic shock waves and a highly turbulent boundary layer. The acoustic level inside the PLF will vary slightly with different spacecraft. This is due to the acoustic absorption of each spacecraft depending on its size, shape, and surface material properties. Acoustic sound pressure levels for LPFs and EPFs are provided in Figures 3.2.2-1, 3.2.2-2, and 3.2.2-3. These figures represent the maximum expected environment based on a 95% probability and 50% confidence. The levels presented are for typical spacecraft of square cross-sectional area with 50-60% fill of the fairing by cross-sectional area. A mission-peculiar acoustic analysis is required for spacecraft with other fill factors to bound the acoustic environment. The spacecraft should be capable of functioning properly after 1-minute exposure to this level. For the LPF or EPF with acoustic panels, special consideration should be given to components located within 76 cm (30 in.) of PLF vents (MPF vents are fewer in number and located farther from the spacecraft envelope). Sound pressure levels for components near the vents are listed in Figure 3.2.2-4 for all vehicles.

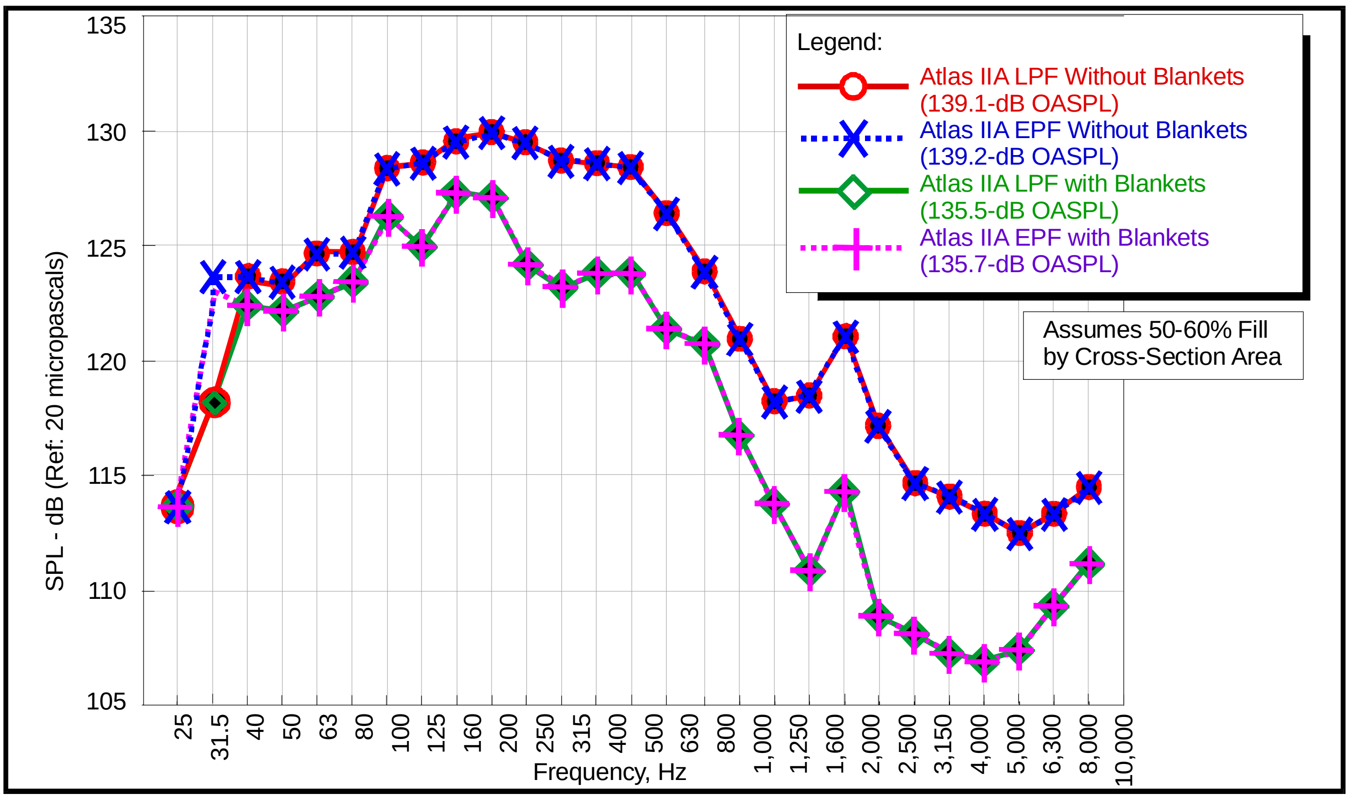


Figure 3.2.2-1 Acoustic Levels for Atlas IIA with Large or Extended-Length Large Fairings

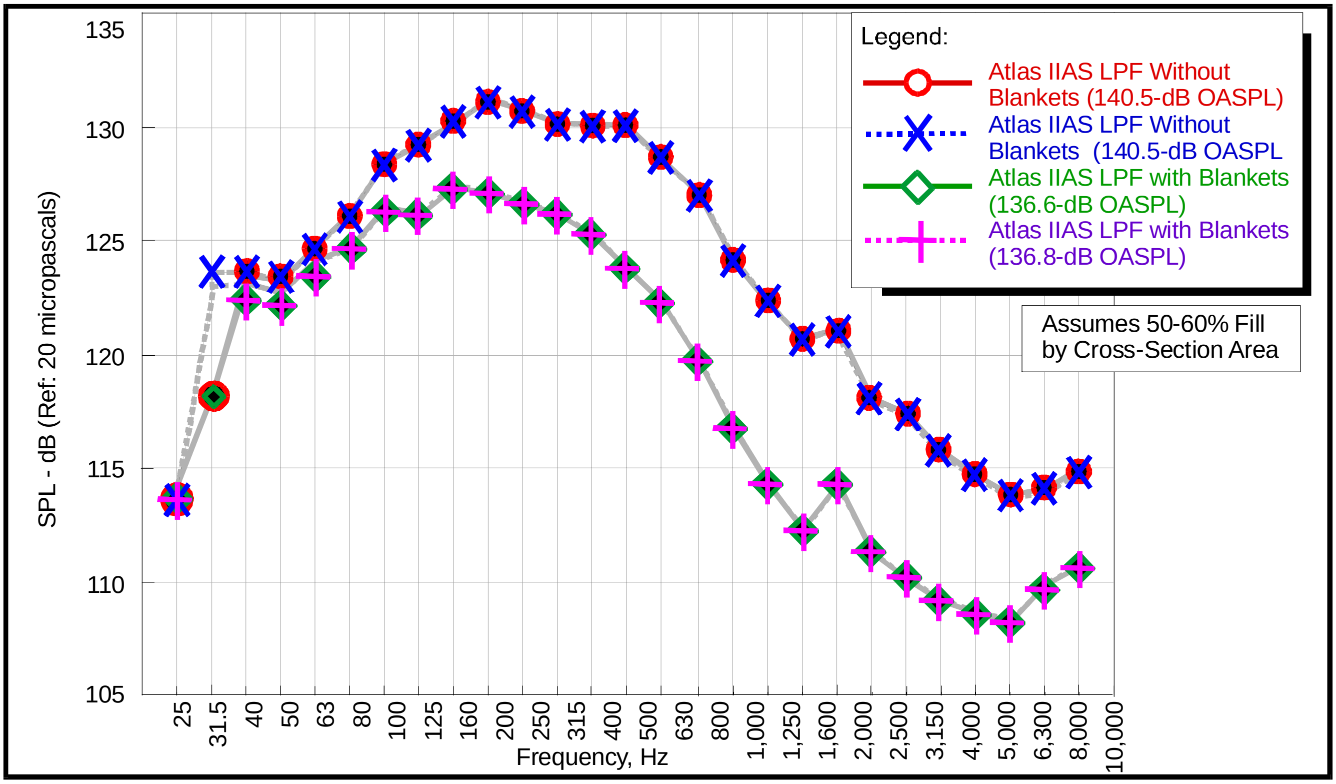


Figure 3.2.2-2 Acoustic Levels for Atlas IIAS with Large or Extended-Length Large Fairings

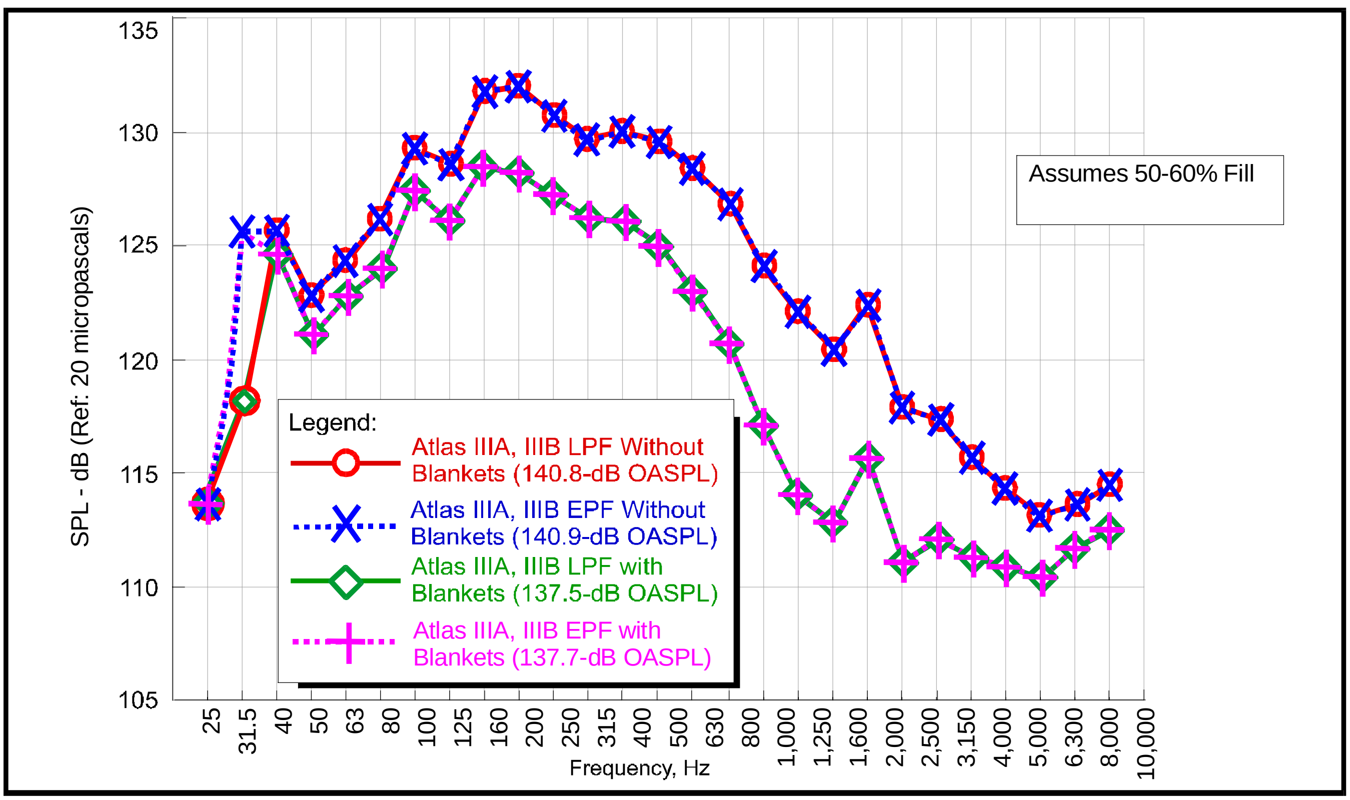


Figure 3.2.2-3 Acoustic Levels for Atlas IIIA and IIIB with Large or Extended-Length Large Fairings

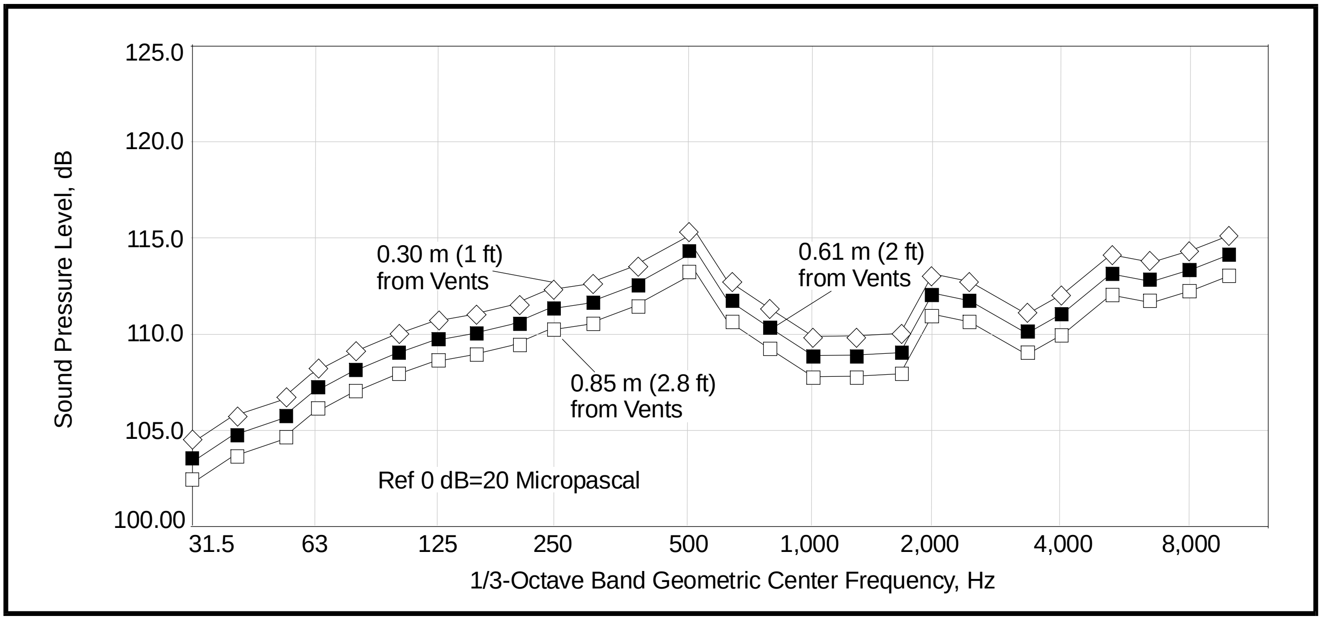


Figure 3.2.2-4 Acoustic Levels Near the Vents with the Large or Extended-Length Large Fairings

An optional mission-peculiar acoustic panel design is available (Fig 3.2.2-5) to reduce high-frequency acoustic energy within the payload envelope.

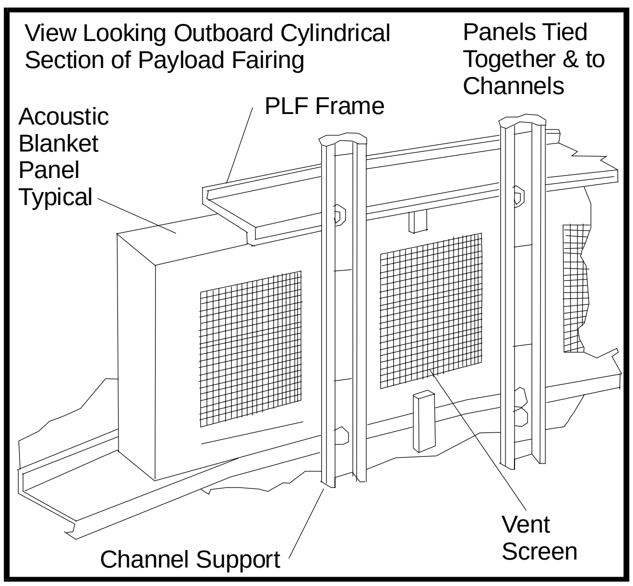


Figure 3.2.2-5 Acoustic Blanket Panels

3.2.3 Vibration

The spacecraft is exposed to a vibration environment that may be divided into two general frequency ranges: (1) low-frequency quasi-sinusoidal vibration, and (2) high-frequency broad-band random vibration.

The low-frequency vibration tends to be the design driver for spacecraft structure. An envelope of the Atlas/Centaur flight measured low-frequency vibration under 100 Hz near the spacecraft interface is shown in Figure 3.2.3-1. The peak responses occur for a few cycles during transient events, such as launch, gusts, booster engine cutoff (BECO), jettison events, and main engine cutoff (MECO).

The transient response environment during Atlas/Centaur flight is characterized by a combination of coupled dynamic response analyses (CLA) and equivalent sinusoidal vibration specified at the spacecraft interface.

Verification of minimum factors of safety is required by performance of a sine vibration test at the structural test model (STM) or protoflight model (PFM) stage and at the flight model (FM) stage of design as defined in Table 3.3-1. Design capability is the objective of STM and PFM testing, while verification of workmanship is accomplished by the acceptance-level FM test. Notching of the base input is recommended in the 0- to 50-Hz range to reduce component responses below design levels while demonstrating the minimum factor of safety over CLA values. Notching in the 50- to 100-Hz frequency range is not recommended without technical discussion concerning flight equivalent levels as related to spacecraft damping.

The high-frequency random vibration that the spacecraft experiences is primarily due to the acoustic noise field, with a very small portion being mechanically transmitted from the engines. The acoustically excited random vibration environment tends to be the design driver for components and small structure supports. The high-frequency vibration level will vary from one location to another depending on physical properties of each area of the spacecraft. Because the vibration level at the payload interface depends on the adjacent structure above and below the interface, the exact interface level depends on the

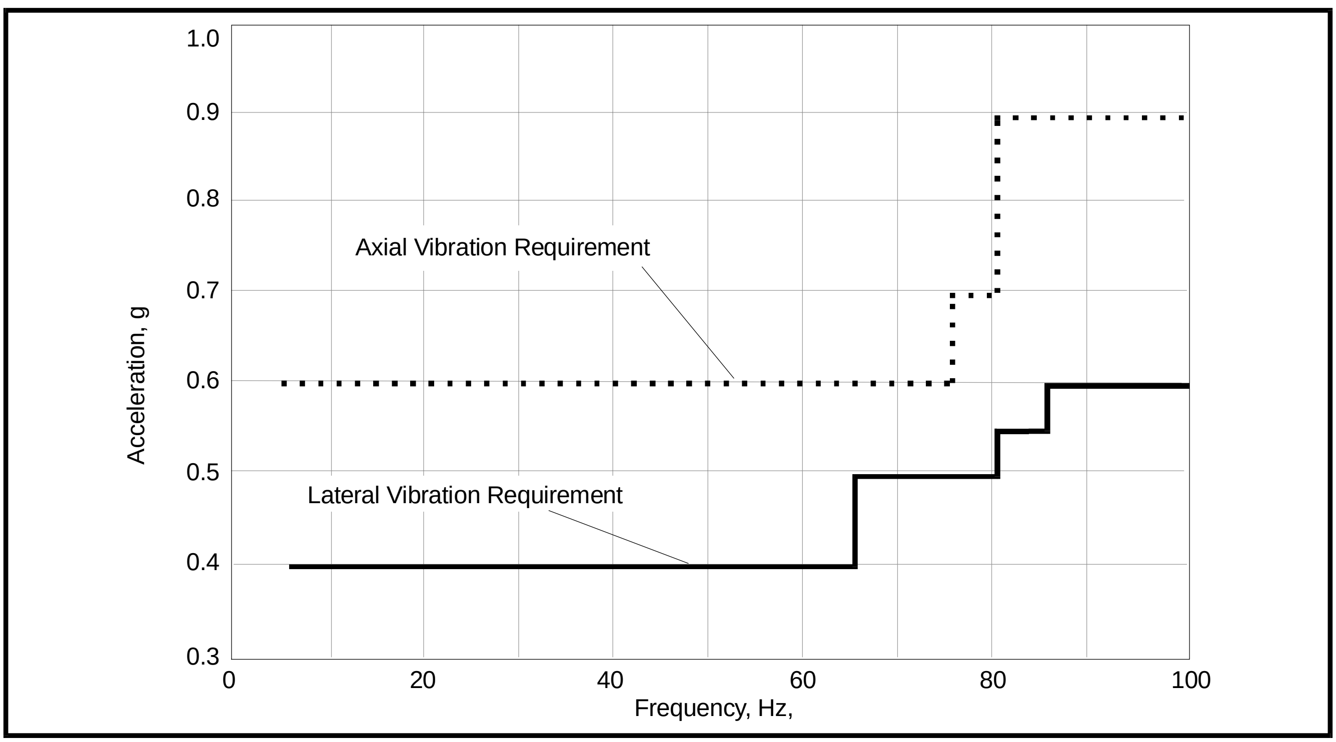


Figure 3.2.3-1 Quasi-Sinusoidal Vibration Requirement

structural characteristics of the lower portion of the spacecraft, the particular payload adapter, and how the acoustic field is influenced by the particular spacecraft.

An acoustic test of the spacecraft will more accurately simulate the high-frequency environment that it will undergo in flight than will a random vibration test. If the spacecraft is mounted to a test fixture that has structural characteristics similar to the PLA, then vibration levels at the interface will be similar to flight levels. It is not recommended to attach the spacecraft to a rigid fixture during the acoustic test because the interface vibration will be zero at the launch vehicle interface attach point, but at some distance away from the interface the vibration levels will be similar to flight levels.

3.2.4 Shock

Three pyrotechnic shock events occur during flight on the Atlas II family of vehicles: (1) payload fairing jettison (PFJ), (2) Centaur separation from the Atlas sustainer, and (3) spacecraft separation. Because the system for Centaur separation from Atlas is located far from the spacecraft, the shock is highly attenuated by the time it reaches the spacecraft and does not produce a significant shock at the spacecraft interface. Separation devices for PFJ are located closer to the spacecraft; thus, the shock at the spacecraft interface is noticeable. The spacecraft separation device is at the spacecraft/Centaur interface and produces the highest shock.

Figure 3.2.4-1 shows expected shock levels for spacecraft separation for a typical spacecraft at the spacecraft separation plane for Type A, A1, B, B1, and D adapters. Figure 3.2.4-2 shows the shock specification for the payload adapter side of a hard-point connection between the launch vehicle and spacecraft for the Type E and Type F payload adapters or similar nut-fired adapters. The Type F adapter may also be used with fast-acting shockless separation nuts (FASSN) with the shock levels as shown. Figures 3.2.4-1 through 3.2.4-3 represent the maximum expected environment based on a 95% probability and 50% confidence. Verification of spacecraft compatibility for clampband separation systems provided by Lockheed Martin is demonstrated by firing of a flight-configured system, typically following the FM matchmate. This test may be performed during STM or PFM testing to establish a mapping for component locations near the interface. Component unit qualification testing must envelop the mapped environment. For user-supplied adapters and separation systems, we recommend that the actual separation device be fired on a representative payload adapter and spacecraft to measure the actual level and/or qualify the spacecraft. Figure 3.2.4-3 shows the maximum acceptable shock level at the equipment module interface for a customer-provided separation system.

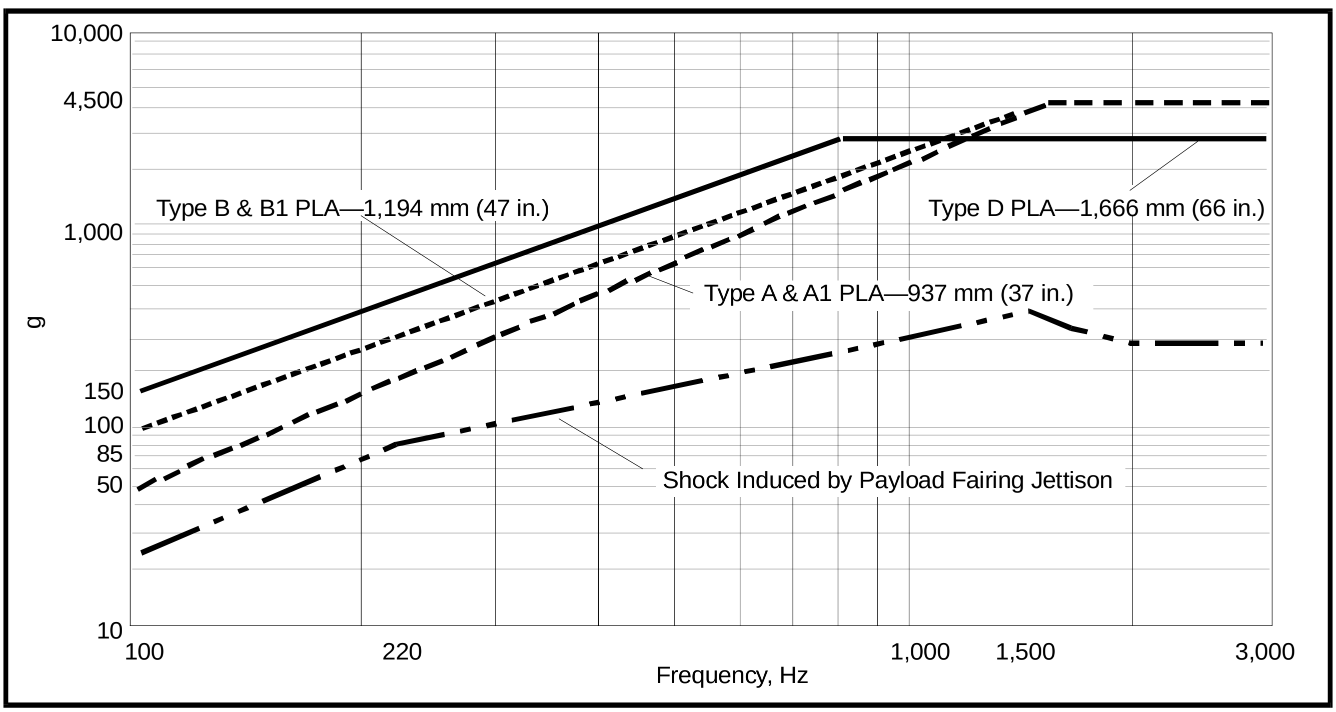


Figure 3.2.4-1 Typical Maximum Atlas Shock Levels—Type A, A1, B, B1, and D Adapters

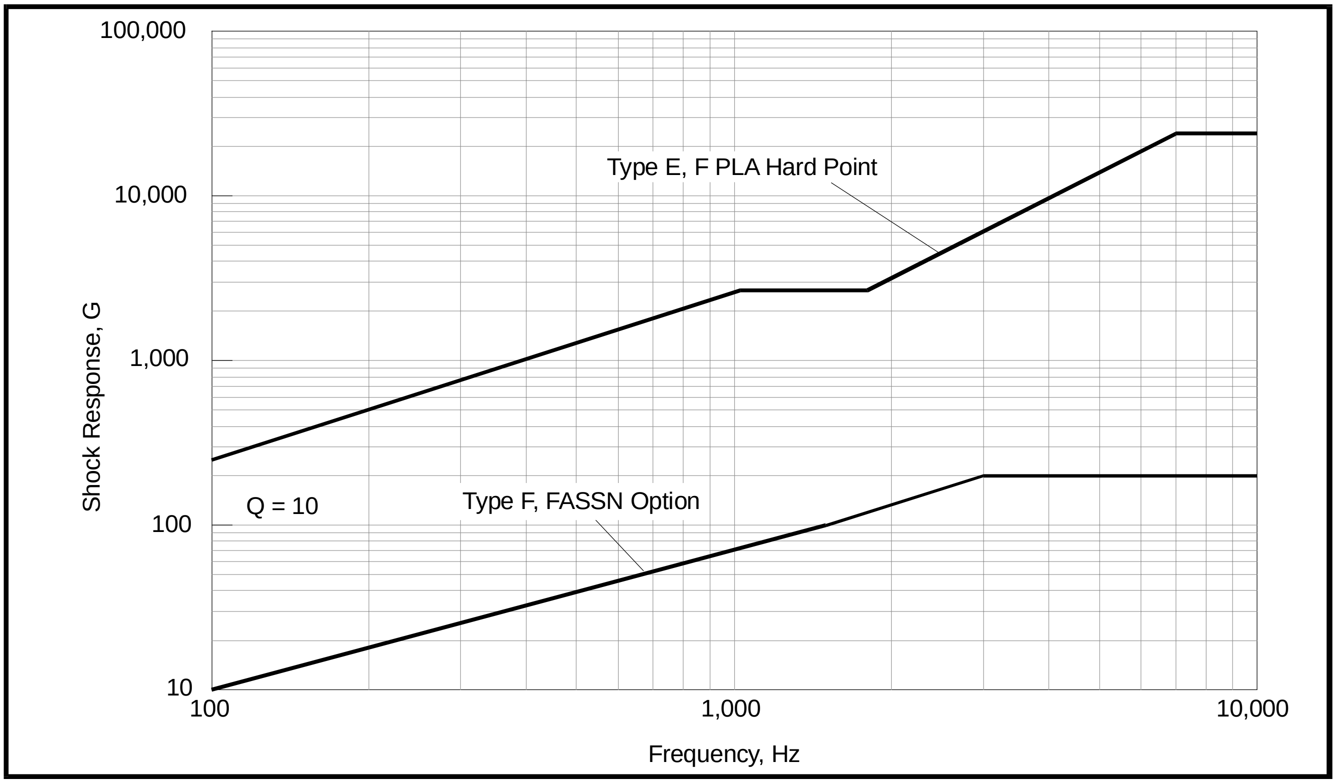
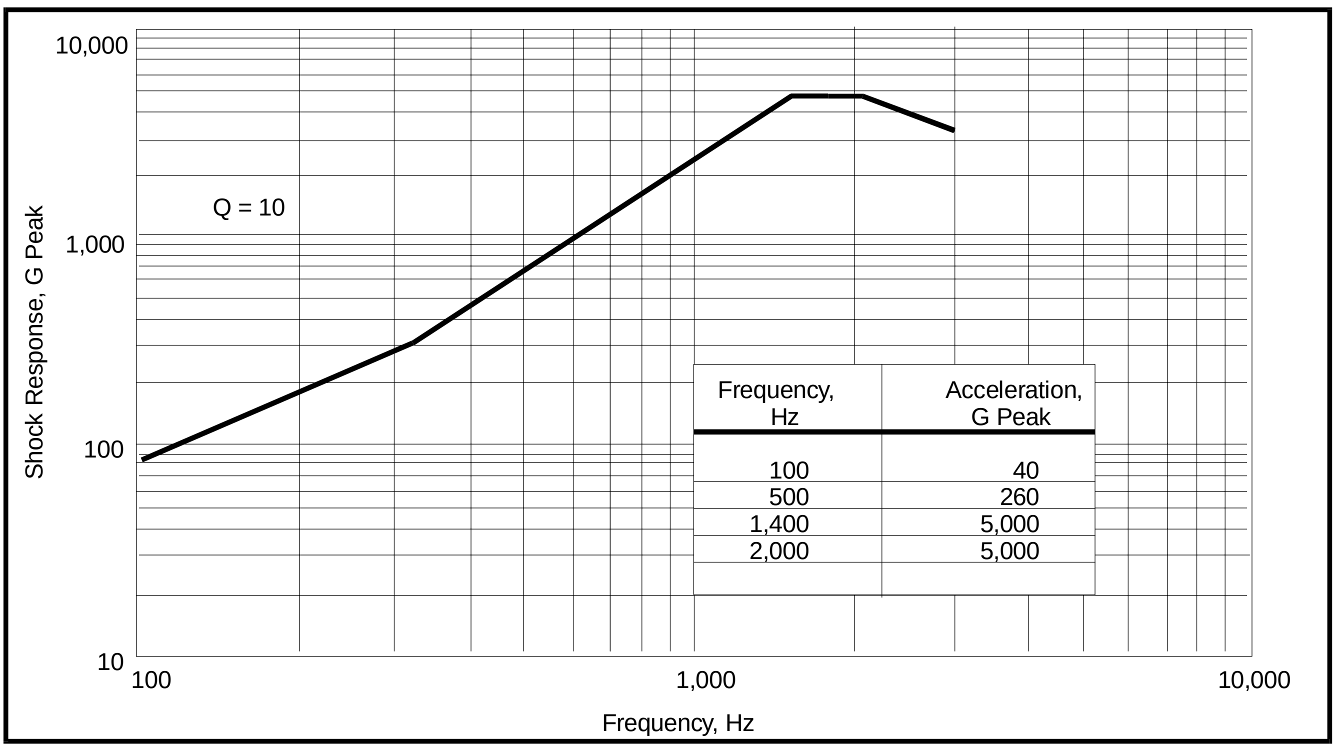


Figure 3.2.4-2 Maximum Atlas Shock Levels for the Type E, F, and FASSN Systems



Frequency, Hz	Acceleration, G Peak
100	40
500	260
1,400	5,000
2,000	5,000

Figure 3.2.4-3 Maximum Allowable Spacecraft-Produced Shock at Equipment Module Interface

3.2.5 Thermal

Within Fairing—The PLF protects the spacecraft during ascent to a nominal altitude of approximately 113,000 m (370,000 ft). Aerodynamic heating on the fairing results in a time-dependent radiant heating environment around the spacecraft before fairing jettison. The fairing uses cork on the external conical surface to minimize fairing skin temperatures. The inner surfaces of the cone and cylinder have a low-emittance finish ($\varepsilon < 0.1$) that minimizes heat transfer to the spacecraft. It should be noted that the PLF boattail and split barrel do not have a low-emittance coating ($\varepsilon \leq 0.9$). The peak heat flux radiated by the cone and cylinder surfaces is less than 400 W/m^2 (125 Btu/hr-ft^2) and peak temperatures remain below 212°C (414°F) at the warmest location.

After Fairing Jettison—Fairing jettison typically occurs when the 3-sigma maximum free molecular heat flux decreases to 1,135 W/m^2 (360 Btu/hr-ft^2). Jettison timing can be adjusted to meet specific mission requirements. Typical free molecular heating (FMH) profiles are shown in Figure 3.2.5-1. Because actual profiles are highly dependent on the trajectory flown, these data should not be used for design. As shown in Figure 3.2.5-1, peak FMH levels can be reduced by raising the parking orbit perigee altitude. However, raising perigee altitude will have a minor negative effect on delivered launch vehicle performance.

The spacecraft thermal environment following fairing jettison includes free molecular heating, solar heating, Earth albedo heating, Earth thermal heating, and radiation to the upper stage and to deep space.

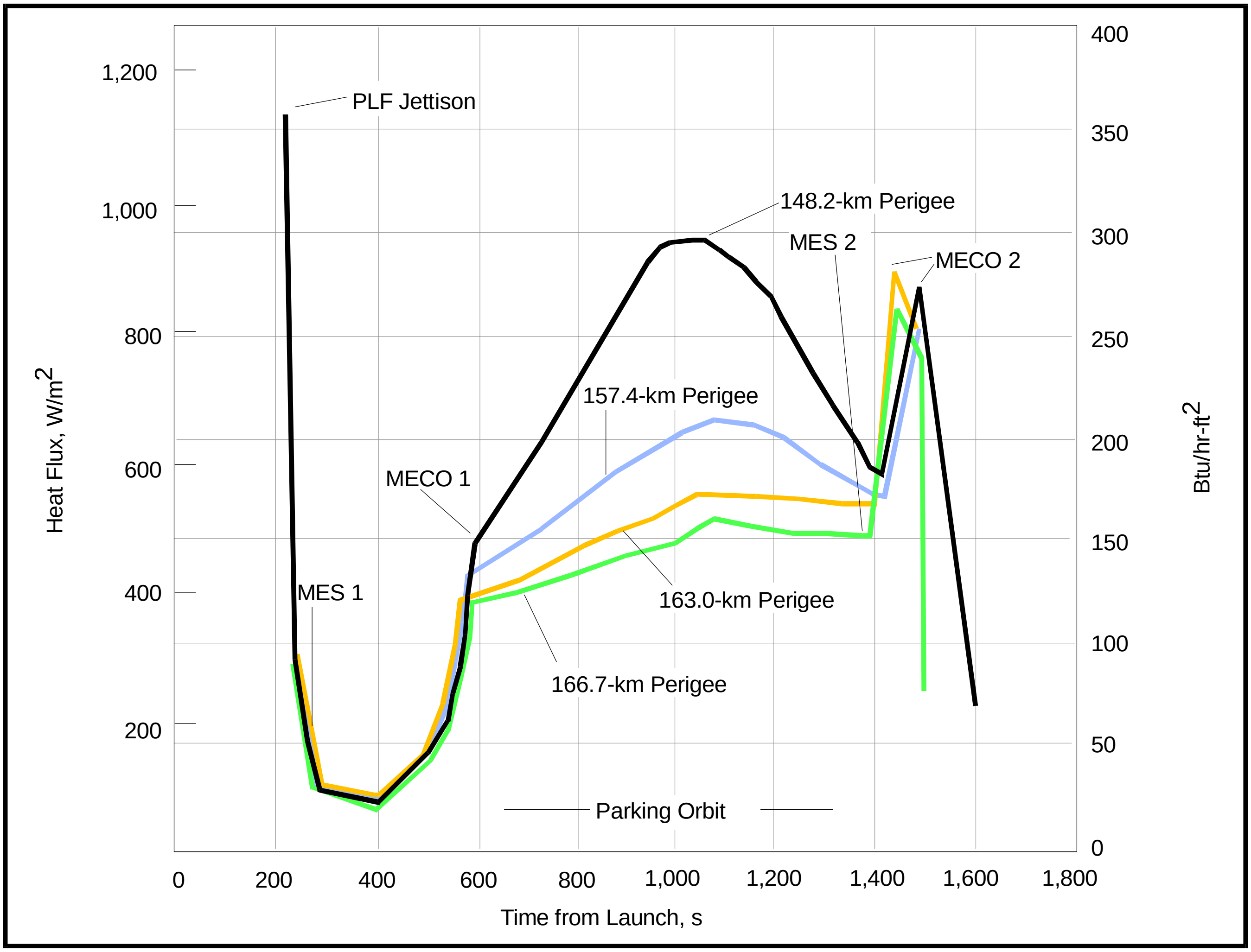


Figure 3.2.5-1 Typical FMH Flux Profiles

The spacecraft also is conductively coupled to the forward end of the Centaur upper stage through the spacecraft adapter. Solar, albedo, and Earth thermal heating can be controlled as required by the spacecraft by specification of launch times, vehicle orientation (including rolls), and proper mission design.

The Centaur nominally provides a benign thermal influence to the spacecraft, with radiation environments ranging from -45 to 52°C (-50 to 125°F) and interface temperatures ranging from 4 to 49°C (40 to 120°F) at the forward end of the spacecraft adapter. Neither upper-stage main engine plumes nor reaction control system (RCS) engine plumes provide any significant heating to the spacecraft. The main engine plumes are nonluminous due to the high purity of LH_2 and LO_2 reactants.

3.2.6 Static Pressure (PLF Venting)

The payload compartment is vented during the ascent phase through one-way vent doors. Payload compartment pressure and depressurization rates are a function of the fairing design and trajectory. The 4.26-m (14-ft) large payload fairing (LPF) was designed to have a depressurization rate of no more than 6.89 kPa (1.0 psi/s). The pressure decay rate will always be less than 2.07 kPa/s (0.3 psi/s), except for a short period around transonic flight when the decay rate shall not exceed 6.89 kPa/s (1.0 psi/s). Typical depressurization rates are less than these values. Figures 3.2.6-1 and 3.2.6-2 illustrate typical pressure profiles and depressurization rates for the LPF and the extended payload fairing (EPF).

The 3.35-m (11-ft) medium payload fairing (MPF) was designed for a depressurization rate of no more than 4.83 kPa/s (0.7 psi/s). The pressure decay rate will always be less than 2.1 kPa/s (0.3 psi/s), except for a short transonic period during which the decay rate shall not exceed 4.83 kPa/s (0.7 psi/s). Typical depressurization rates are less than these values. Figures 3.2.6-3 and 3.2.6-4 illustrate typical pressure profiles and depressurization rates for the MPF.

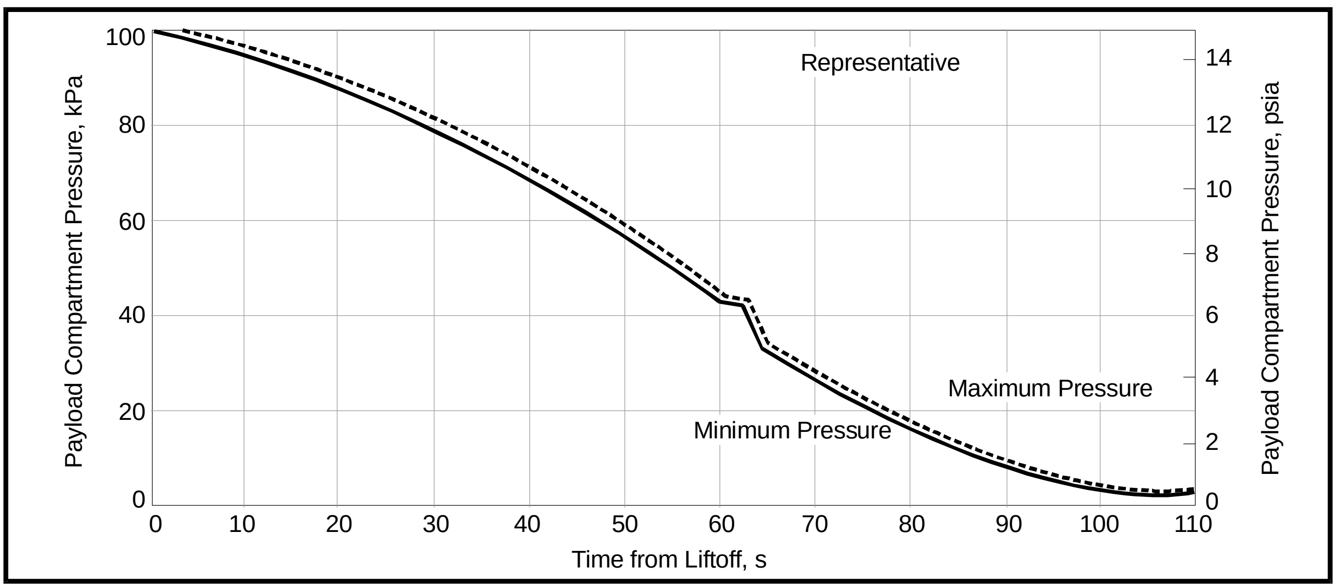


Figure 3.2.6-1 Typical Static Pressure Profiles Inside the Large and Extended-Length Large Fairings

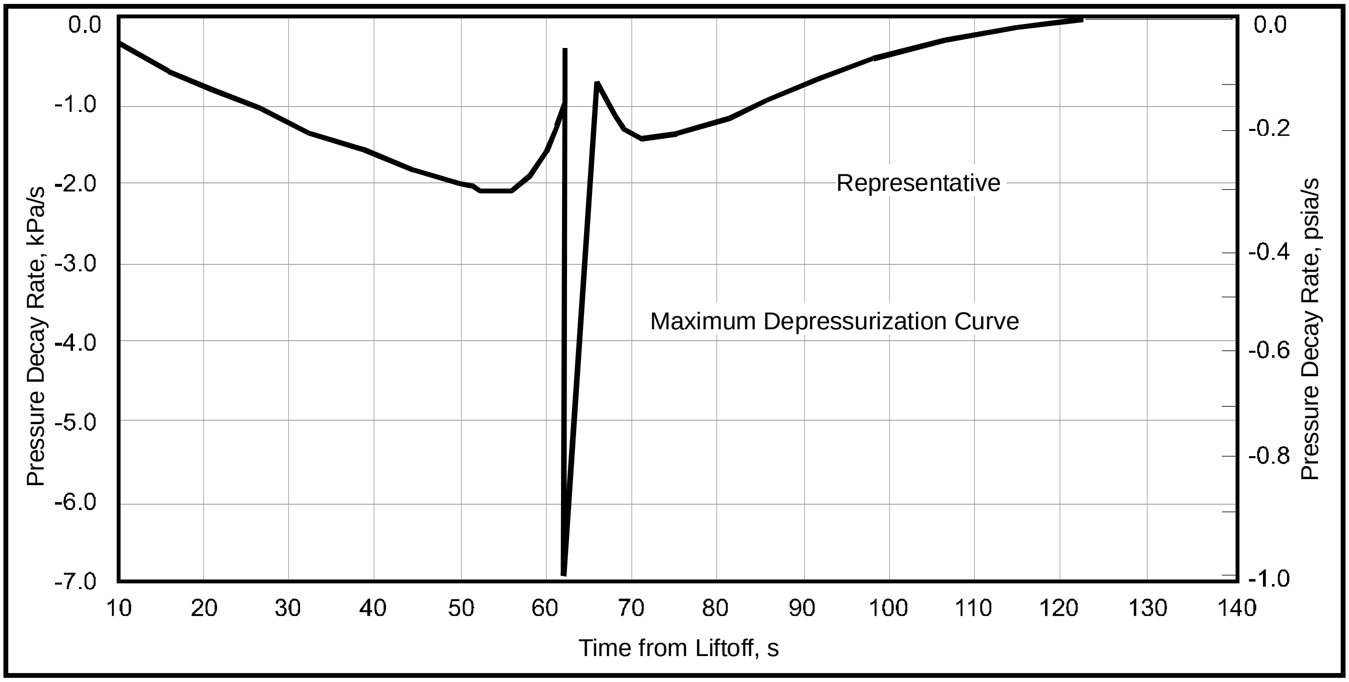


Figure 3.2.6-2 Typical Payload Compartment Pressure Decay Rate for the LPF and EPF

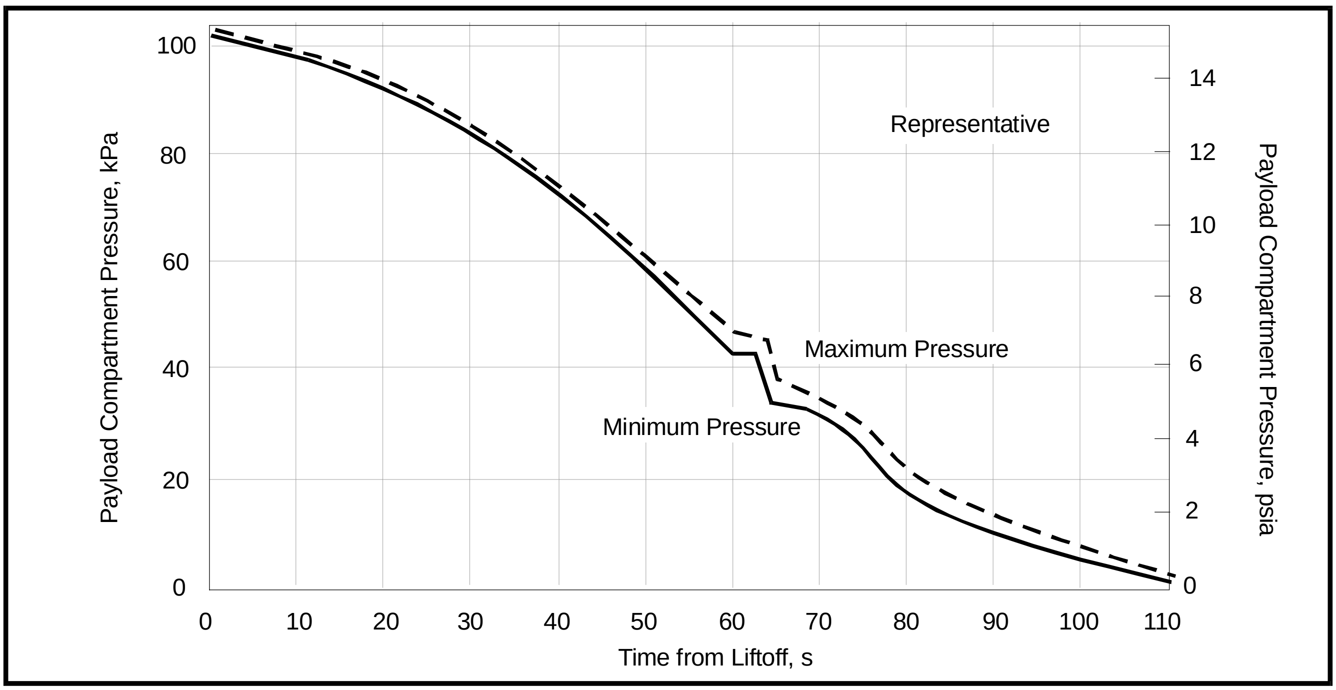
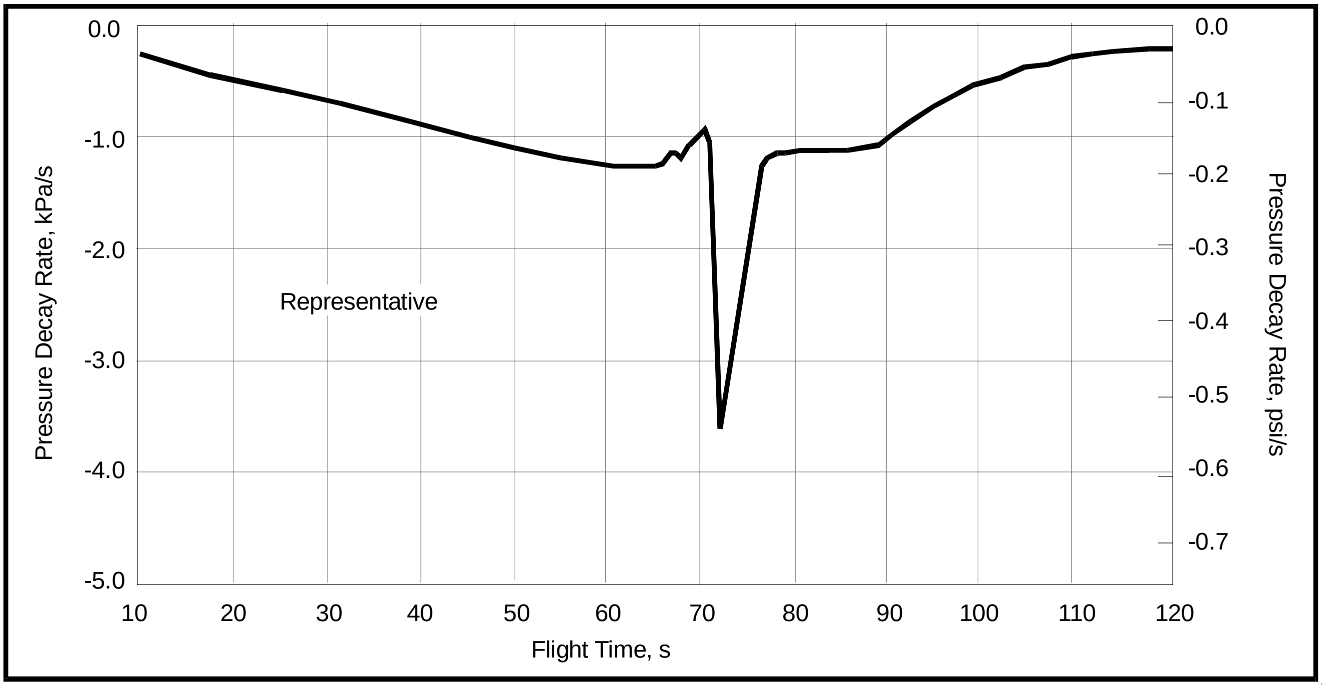


Figure 3.2.6-3 Typical Payload Compartment Pressure for the MPF



F igure 3.2.6-4 Typical Payload Compartment Pressure Decay Rate for the MPF

3.2.7 Contamination Control

3.2.7.1 Atlas Retrorockets—Approximately 270 seconds from launch, the Atlas sustainer is separated from Centaur. After separation, eight retrorockets near the aft end of the Atlas (Station 1133) are fired to ensure the expended Atlas stage moves away from Centaur. These eight retrorockets use solid propellants; exhaust products will consist of small solid particles and very low density gases.

Retrorocket nozzles are canted outboard 40° from the vehicle axis. This cant angle ensures that virtually no solid particles will impact the spacecraft. Exhaust gases that impinge on the spacecraft are rarefied and only a small fraction is condensable because spacecraft surfaces are still relatively warm from prelaunch payload compartment gas conditioning. Moderate molecular depositions can be expected on aft facing outlying surfaces (> 152-cm [60-in.] radius).

3.2.7.2 Upper-Stage Reaction Control System (RCS)—The upper-stage RCS consists of 12 27-N (6-lbf) hydrazine (N_2H_4) thrusters for settling, roll, and attitude control requirements. Four thrusters provide axial thrust and eight provide roll, pitch, and yaw control. Thrusters are located slightly inboard on the upper-stage aft bulkhead. Thrusters produce a plume that has an extremely low contaminant content.

Before upper-stage/spacecraft separation, the spacecraft will not be exposed to RCS exhaust plumes. The RCS thruster's inboard location on the aft bulkhead precludes direct line of access between the spacecraft and thrusters.

After separation, some minor spacecraft impingement from thruster exhaust plumes may occur during the collision/contamination avoidance maneuver (CCAM). CCAM is designed to prevent recontact of the Centaur with the spacecraft while minimizing contamination of the spacecraft.

An analysis of spacecraft contamination due to RCS motors predicts a maximum deposition of 6.03 x 10^{-9} g/cm^2 (0.6Å) on spacecraft surfaces.

A typical CCAM sequence is shown in Figure 3.2.7.2-1. This figure shows typical spacecraft motion after the separation event as longitudinal and lateral distance from the upper stage. Included are contour lines of constant flux density for the plumes of the aft-firing RCS settling motors during operation. The plumes indicate the relative rate of hydrazine exhaust product impingement on the spacecraft during the 2S-ON phase before blowdown and during hydrazine depletion. There is no impingement during the CCAM 4S-ON phase because the spacecraft is forward of the settling motors.

3.2.7.3 Upper-Stage Main Engine Blowdown—As part of the CCAM, hydrogen and oxygen are expelled through the engine system to safe the vehicle and to further increase upper-stage/spacecraft separation distance. Hydrogen is expelled out the engine cooldown ducts and oxygen is expelled out the main engine bells. The expelled products are hydrogen, oxygen, and trace amounts of helium, which are noncontaminating to the spacecraft. Figure 3.2.7.3-1 identifies typical main engine blowdown exhaust product impingement rates on the spacecraft.

3.2.7.4 Separation Event—Atlas uses a spacecraft separation system similar to those used throughout the launch vehicle industry. This system uses two pyrotechnically initiated bolt cutters to release two clamp halves that hold the spacecraft to the launch vehicle payload adapter. Upon actuation, small particles can be generated by this system. The kinetic energy of particles observed during separation system testing was approximately two orders of magnitude less than the micrometeroid design criteria specified in NASA SP 8013. The 99.9% largest expected particle has a mass of 0.008 grams and a longest dimension of 0.43 cm (0.17 in.).

3.2.8 Radiation and EMC

The description of environments in Section 3.1.2 encompasses worst-case flight environments.

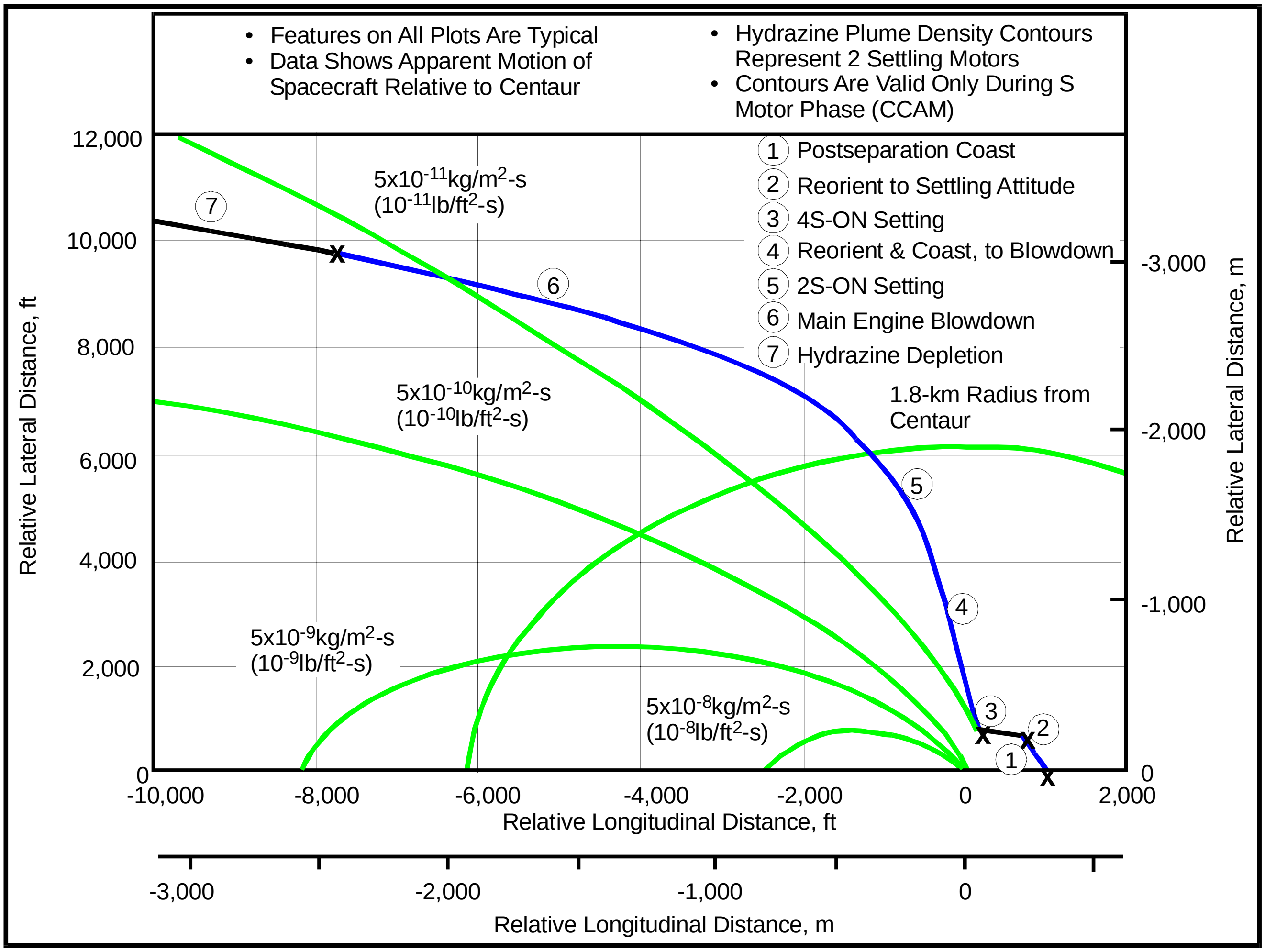


Figure 3.2.7.2-1 Typical Spacecraft Motion Relative to Centaur Upper Stage

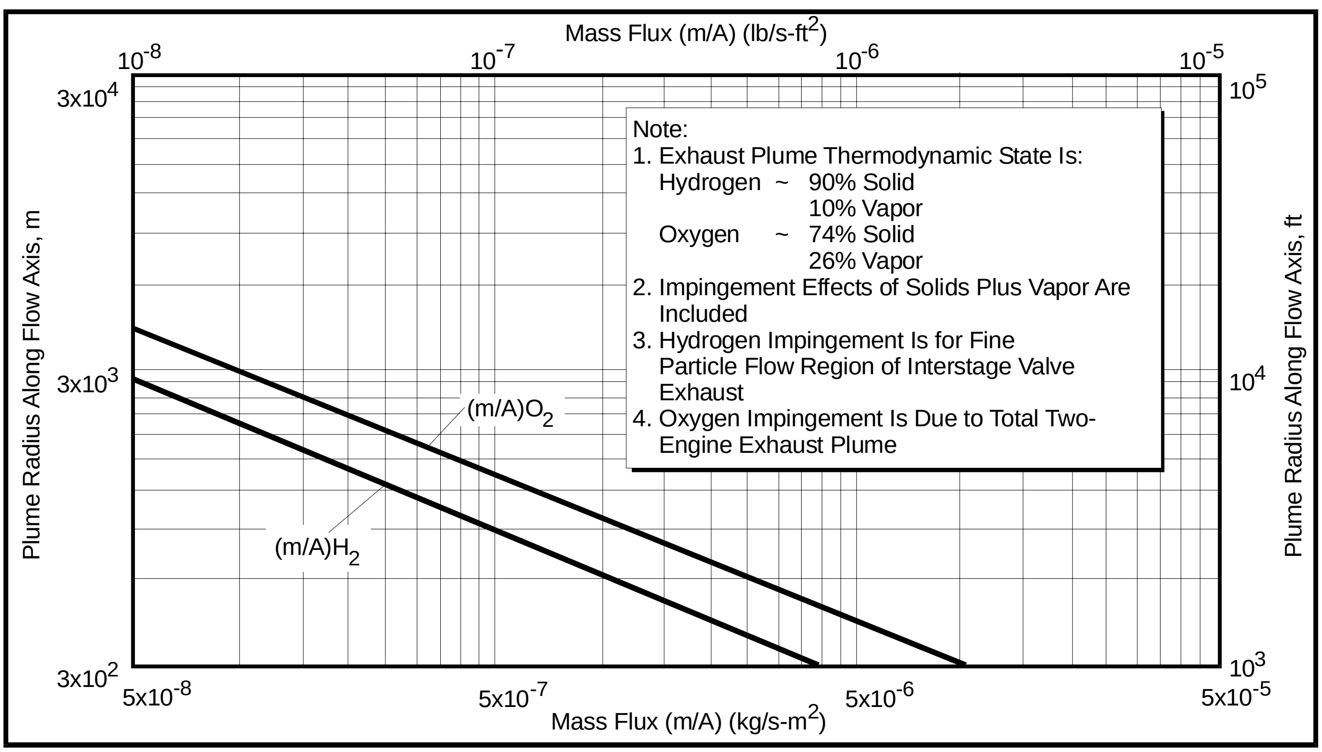


Figure 3.2.7.3-1 Typical Spacecraft Impingement Fluxes During Main Engine Blowdown

3.3 SPACECRAFT COMPATIBILITY TEST REQUIREMENTS

Lockheed Martin requires that the spacecraft be capable of experiencing maximum expected flight environments multiplied by minimum factors of safety to preclude loss of critical function. An environmental test report is required to summarize the testing performed and to document the adequacy of the spacecraft structure for flight loads.

The spacecraft testing required for demonstration of compatibility is listed in Table 3.3-1. This table describes tests, margins, and durations appropriate as a minimum for programs in three phases of development. The structural test model (STM) is considered a test-dedicated qualification unit with mass simulation of components to be tested in unit qualification programs. Data acquired during STM tests may be used to establish qualification levels for each component. The sine vibration test shall include wet propellant tanks on new designs if acceptance testing will not include this important effect.

Table 3.3-1 Spacecraft Structural Tests, Margins, and Durations

Test	STM (Qual)	PFM* (Protoflight)	FM (Flight)
Static • Level • Analyses	 1.25 x Limit (DLF or CLA)	 1.25 x Limit (CLA)	 1.1 x Limit (Proof Tests)
Acoustic • Level • Duration	 Limit + 3 dB 2 Min	 Limit + 3 dB 1 Min	 Limit Level 1 Min
Sine Vib • Level • Sweep Rate	 1.25 x Limit 2 Oct/Min	 1.25 x Limit 4 Oct/Min	 Limit Level 4 Oct/Min
Shock	1 Firing	1 Firing	1 Firing

*Note: The Protoflight test levels are also used for validation of ICD dynamic environments when supplemental FM measurements (Mission Satisfaction Option) are made for a specific mission.

The protoflight model (PFM) is the first flight article produced without benefit of a qualification or STM program. The flight-configured spacecraft is exposed to qualification levels for acceptance durations. The flight model (FM) is defined as each flight article produced after the qualification or protoflight article. Tests required for each FM are intended as proof-of-manufacturing only and are performed at maximum expected flight levels.

Lockheed Martin also suggests that the spacecraft contractor demonstrate the spacecraft capability to withstand thermal and EMI/EMC environments.

Flight hardware fit checks are performed to verify mating interfaces and envelopes. Table 3.3-2 identifies suggested spacecraft qualification and acceptance tests to ensure adequate compliance with Atlas environments. Specific test levels and margins, based on flight environments provided, are provided early in the integration process.

Table 3.3-2 Spacecraft Qualification and Acceptance Test Requirement

	Acoustic	Shock	Sine Vib	EMI/EMC	Modal Survey	Static Loads	Fit Check
Qual	X	X	X	X	X	X	
Accept	X		X				X

4.0 SPACECRAFT INTERFACES

4.1 SPACECRAFT-TO-LAUNCH VEHICLE INTERFACES

The primary interfaces between the launch vehicle and spacecraft consist of the payload adapter, which supports the spacecraft on top of the launch vehicle equipment module, and the payload fairing (PLF), which encloses and protects the spacecraft during ground operations and launch vehicle ascent. The Atlas program currently has eight standard payload adapters and three standard payload fairings available to the launch service customer. These standard components can be modified as necessary to provide mission-unique support services for the payload. Typical mission-unique modifications, shown in Figure 4.1-1, include access doors and thermal shields or acoustic blankets on the payload fairing and an extended mission kit which is used for missions requiring longer coast periods between Centaur main engine burns. All of these standard payload fairings, payload adapters and mission-unique options are fully compatible with either the Atlas IIA, IIAS, IIIA, or IIIB.

Sections 4.1.1 and 4.1.2 describe the payload fairing and payload adapter. The payload envelopes and vehicle interface information contained in these sections should be used only as a guideline. Modifications to these envelopes and adapters may be accommodated on a mission-peculiar basis. Ultimate control of interface information for a given mission is governed through the mechanical interface control drawing (MICD), developed and maintained during the mission integration process.

4.1.1 Mechanical Interface—Payload Fairings (PLF)

4.1.1.1 Payload Fairing Configuration—The Atlas PLF provides a protective (thermal, acoustic, electromagnetic, and environmental) enclosure for the payload and Centaur equipment module packages during prelaunch and ascent. The Atlas user has a choice between the standard length large payload fairing (LPF), the extended length large payload fairing (EPF), and medium payload fairing (MPF) configurations. These three fairings are depicted in Figure 4.1.1.1-1.

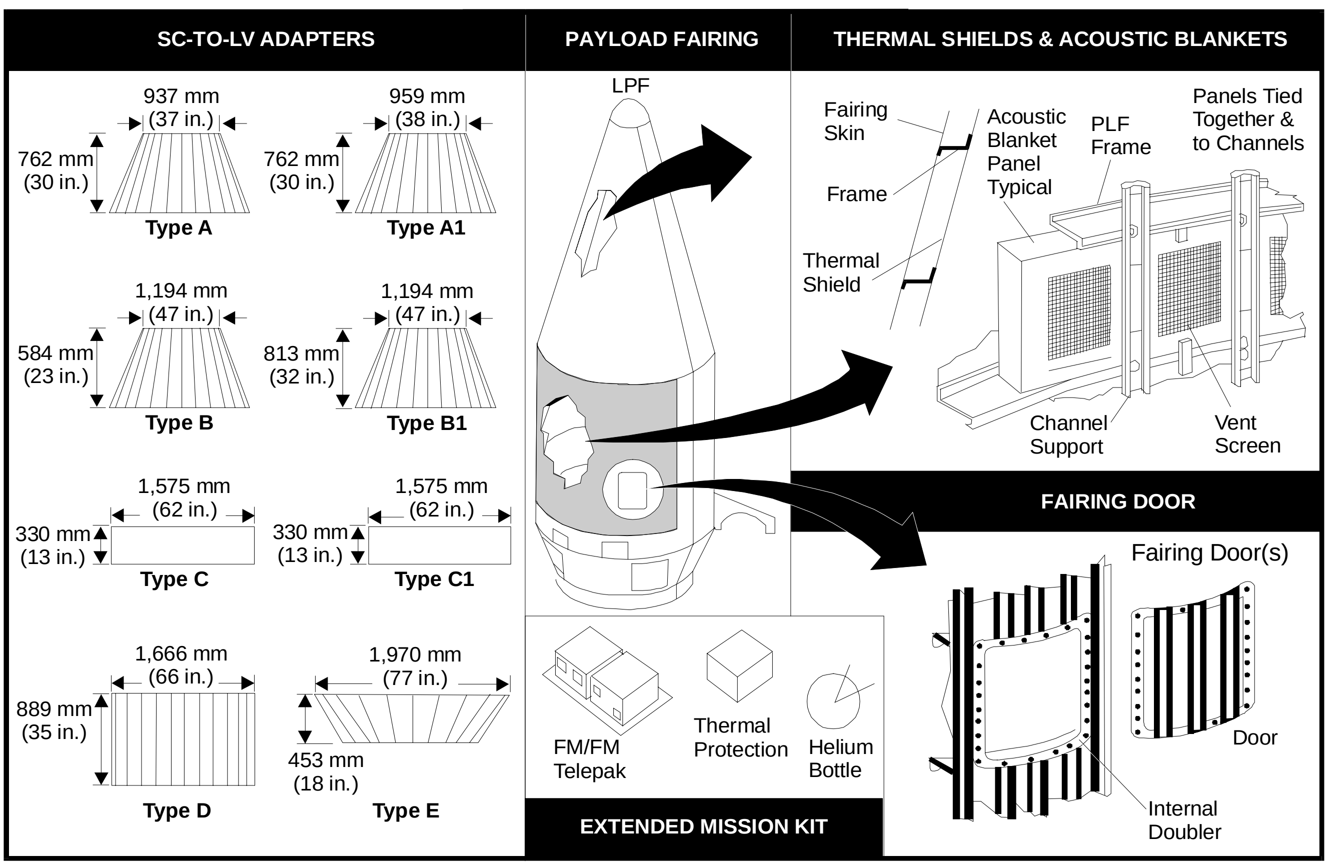


Figure 4.1-1 Existing Mission-Peculiar and mission-Unique hardware Available for Atlas Mission

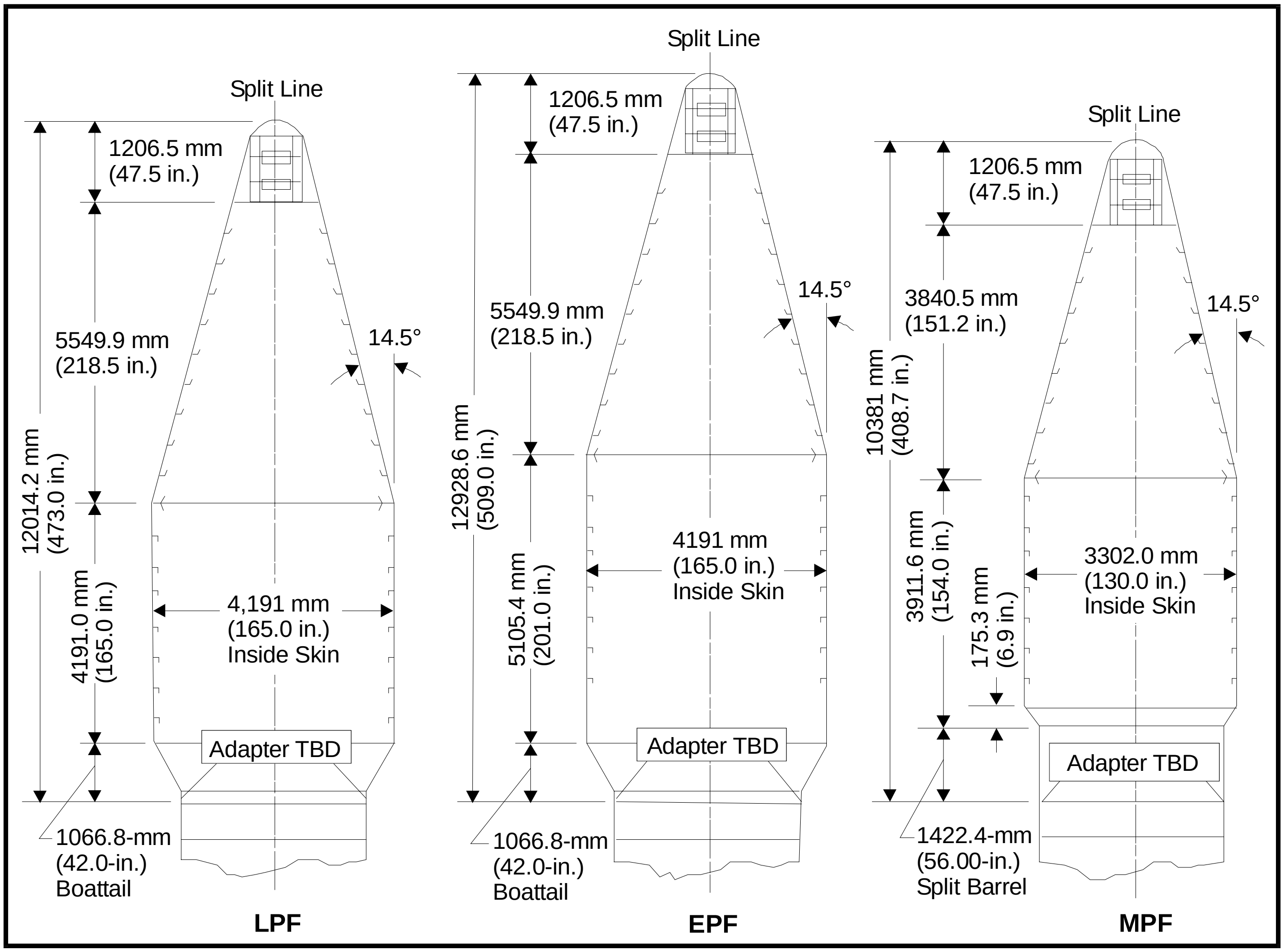


Figure 4.1.1.1-1 Payload Fairing Configurations

The Atlas LPF and EPF have a 4-m (13.2-ft) diameter cylindrical section which provides the major useable volume for the spacecraft. The LPF cylindrical section provides 4 m (13.2 ft) of length for the spacecraft. The EPF was developed to support launch of larger volume spacecraft by adding a 0.9-m (3-ft) cylindrical plug to the top of the cylindrical section of the standard length large fairing. The EPF cylindrical section provides 5.4 m (17.7 ft) of length for the spacecraft. For payloads that do not require the volume of the LPF, a 3.3-m (10.8-ft) diameter medium PLF is available. The MPF cylindrical section provides 3.9 m (12.8 ft) of length for the spacecraft.

The major sections of these fairings are the boattail (or split barrel for the MPF), the cylindrical section, and the nose cone which is topped by a spherical cap. All of these sections consists of an aluminum skin, stringer, and frame construction with vertical, split-line longerons that separate the fairing into bisectors for jettison. Portions of the external surfaces of the fairing are insulated with cork to limit temperatures to acceptable values. Noncontaminating thermal control coatings are used on internal surfaces of the fairing to reduce incident heat fluxes to the spacecraft.

The fairing also provides mounting provisions for various other systems. Payload compartment cooling system provisions are contained in the cylindrical and nose cone portion of the fairing. Vent holes and housings are mounted on the lower part of the cylindrical section for the LPF and EPF and on the split barrel for the MPF. Electrical packages required for the fairing separation system are mounted on the internal surface of the boattail or split barrel. Ducting for the upper-stage hydrogen tank venting system and cooling ducts for the equipment module packages are also attached to the boattail or split barrel. The four large doors in the aft (boattail or split barrel) portion of the PLFs provide primary access to Centaur equipment module packages and the encapsulated spacecraft. These doors (Fig. 4.1.1.1-2) provide an access opening approximately 762-mm (30-in.) wide by 660-mm (26-in.) tall (LPF and EPF boattail) and 1,016-mm (40-in.) wide by 914-mm (36-in.) tall (MPF split barrel) and are located one per fairing quadrant. Work platforms can be inserted through these doors into the payload compartment to allow access to spacecraft hardware near the aft end of the payload compartment.

If additional access to the spacecraft is required, doors can be provided on a mission-unique basis on the cylindrical section of each payload fairing. Doors can be located in most areas of the fairing cylindrical sections except near splitlines and interface planes. Typical access doors are shown in Figure 4.1.1.1.-2 for each fairing.

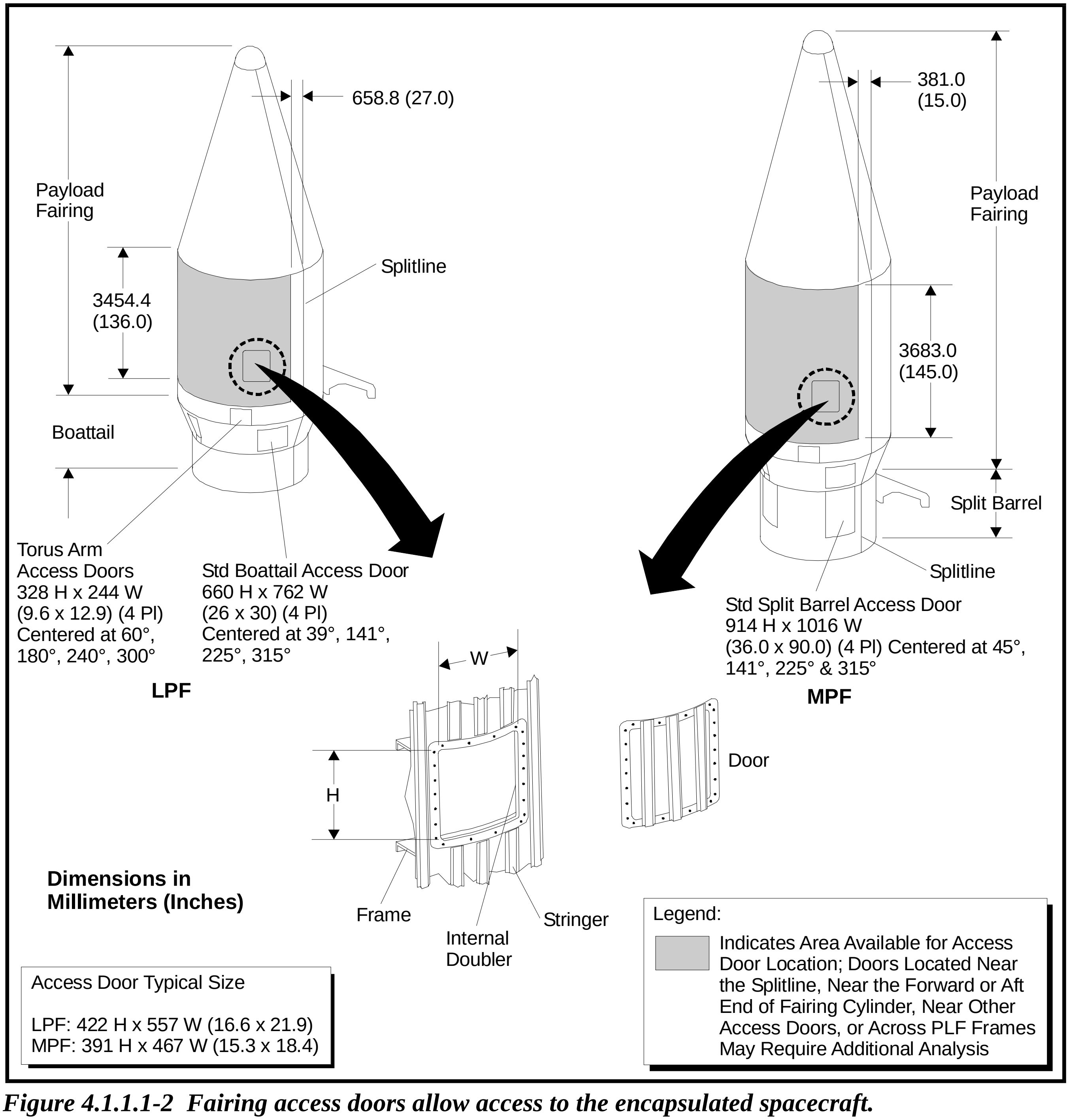


Figure 4.1.1.1-2 Fairing access doors allow access to the encapsulated spacecraft.

Additional mission-peculiar items that can be mounted on the payload fairing to provide support services for the payload include thermal shields, acoustic panels, and a radio frequency (RF) reradiating antenna. The thermal shields and acoustic blankets allow the spacecraft thermal and acoustic environments to be tailored for each mission. The reradiating antenna allows for spacecraft RF communications after the spacecraft is encapsulated inside of the payload fairing.

4.1.1.2 Static Payload Envelopes—The useable volume for a spacecraft inside of the payload fairing is defined by the static payload envelope. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. These envelopes include allowances for spacecraft and PLF static and dynamic deflections, manufacturing tolerances, out-of-round conditions, and misalignments and were established to insure that a minimum 1-in. clearance between the spacecraft and the payload fairing is maintained. Clearance layouts and analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured after the spacecraft is encapsulated inside of the fairing to ensure positive clearance during flight. To accomplish this, it is important for the spacecraft description to include an accurate physical location including the maximum manufacturing tolerances, of all points on the spacecraft that are within 50 mm (2 in.) of the allowable envelope.

The static payload envelope for the LPF is shown in Figure 4.1.1.2-1 The LPF provides a usable diameter of 3,750 mm (147.64 in.) in the cylindrical portion of the fairing. The aft portion of the envelope is reduced to allow for jettison clearance of the PLF hardware. The similar usable static envelope provided by the EPF is depicted in Figure 4.1.1.2-2.

The static payload envelope for the MPF envelope is shown in Figure 4.1.1.2-3. This provides a usable diameter of 2,921 mm (115 in.) in the cylindrical portion of the fairing. The aft portion of the envelope is also reduced to allow for jettison clearances of the fairing hardware.

The static payload envelope for the LPF and EPF can be increased to a 3,850-mm (151.57-in.) diameter in localized areas by incorporating modifications to the fairing. This is accomplished by notching the ring frames located on the inside of the payload fairing and adding ring splices and doublers to maintain fairing strength and stiffness. A typical static payload envelope incorporating these modifications is shown in Figure 4.1.1.2-4.

The envelope around the payload adapter is determined by the adapter configuration used. Envelopes surrounding Type A, Type A1, Type B, Type B1, Type C, Type C1, Type D, and Type E payload adapters and the equipment module are shown in Figures 4.1.1.2-5 through 4.1.1.2-12.

The spacecraft envelopes shown in Figures 4.1.1.2-1 to -12 were defined using a 3,720-kg (8,200-lb) spacecraft with the center-of-gravity (cg) 1,650 mm (65 in.) above the separation plane assuming that spacecraft primary structure first lateral modes are above 10 Hz and first axial modes are above 15 Hz. A reduced envelope may be required for spacecraft with characteristics other than these. No allowances for LPF and EPF thermal shields or acoustic blankets have been made. If either mission-peculiar option is selected, an envelope reduction would be required. A larger envelope may be permitted on a mission-peculiar basis for spacecraft secondary structure (e.g., antennas, thermal shields) that extends outside the envelopes shown. Coordination with the Atlas program is required to define appropriate envelopes for these situations.

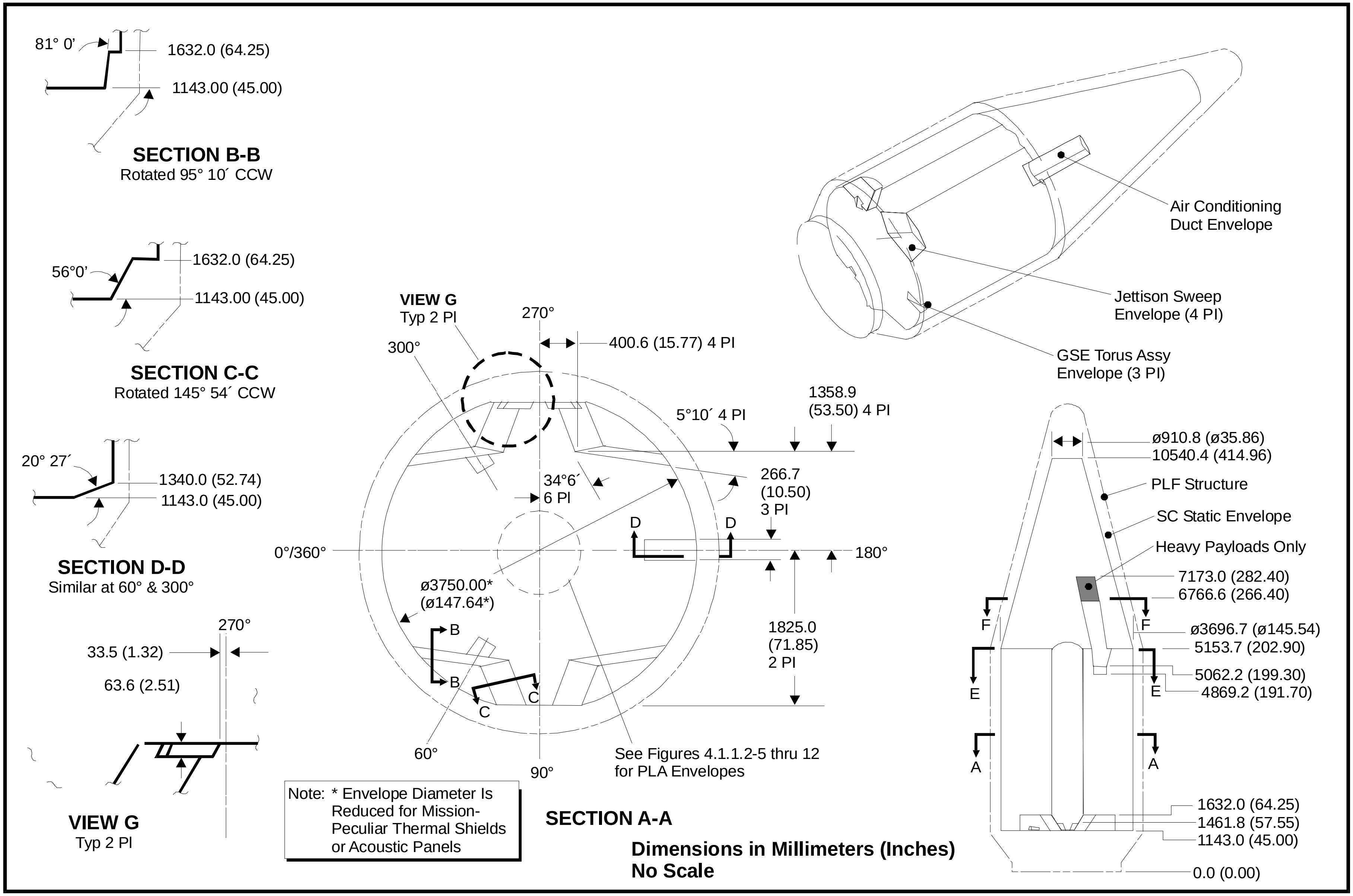


Figure 4.1.1.2-1 Large Payload Fairing (LPF) Envelope

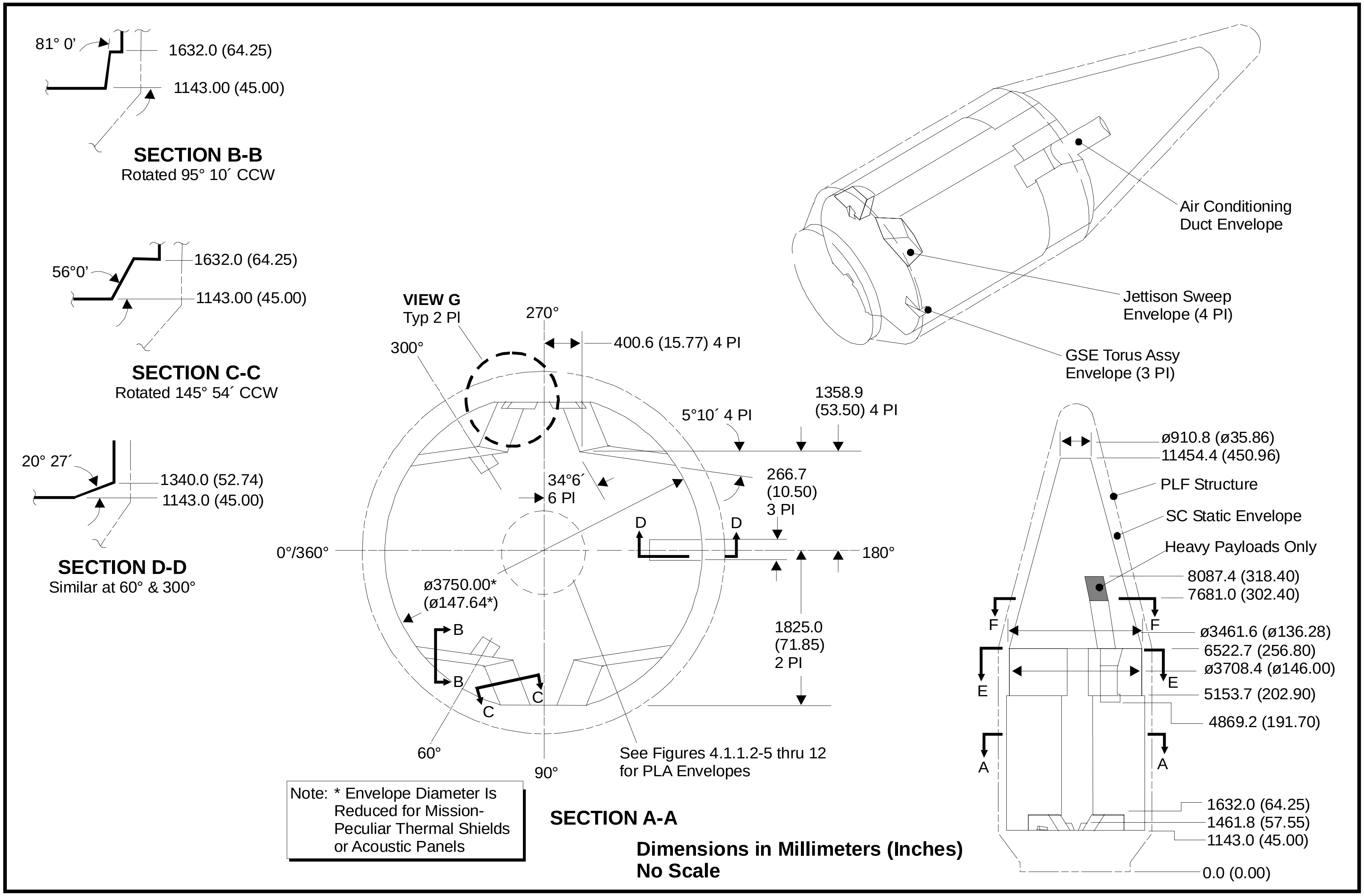


Figure 4.1.1.2-2 Extended Length Large Payload Fairing (EPF) Envelope

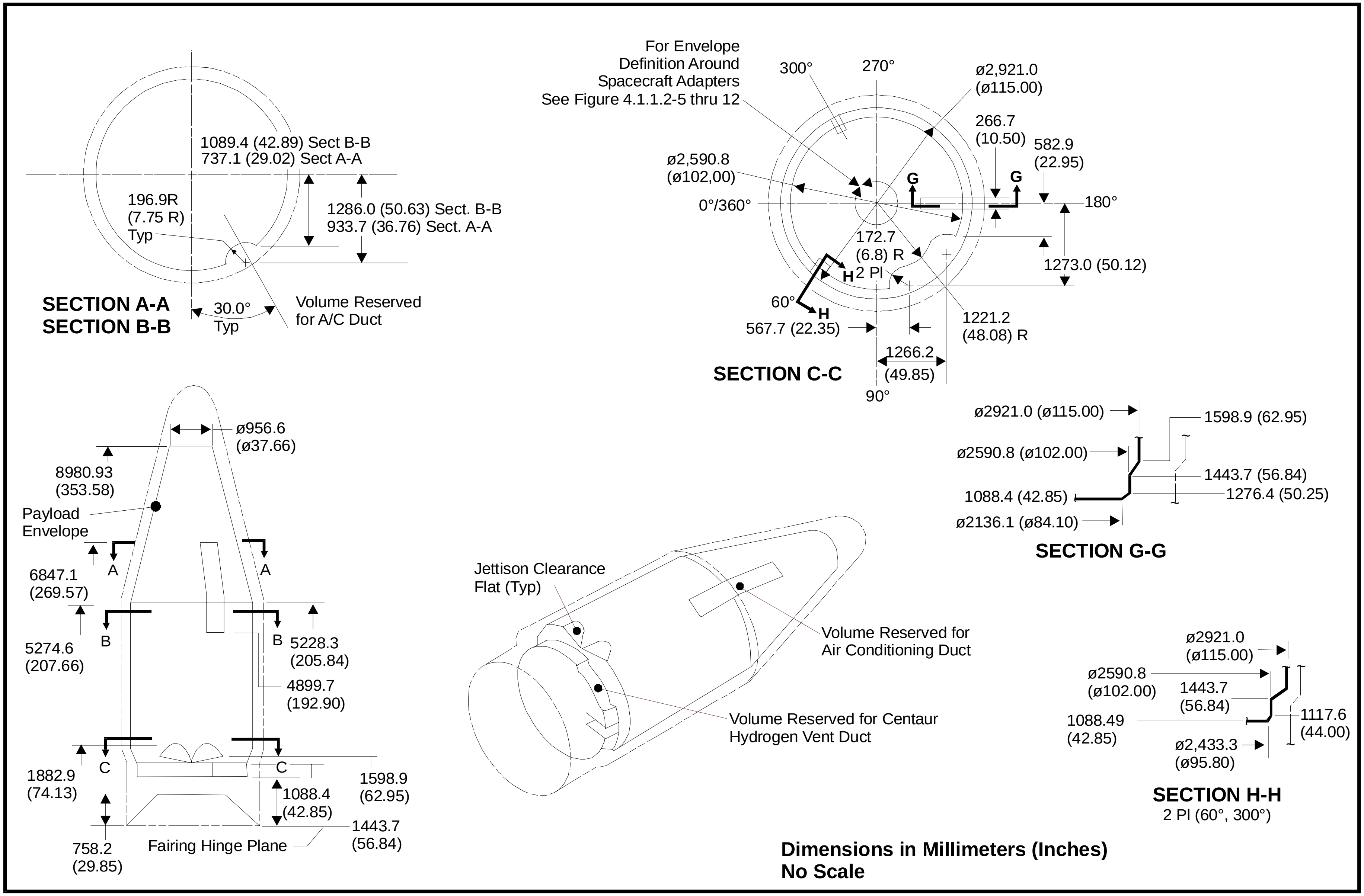


Figure 4.1.1.2-3 Medium Payload Fairing (MPF) Envelope

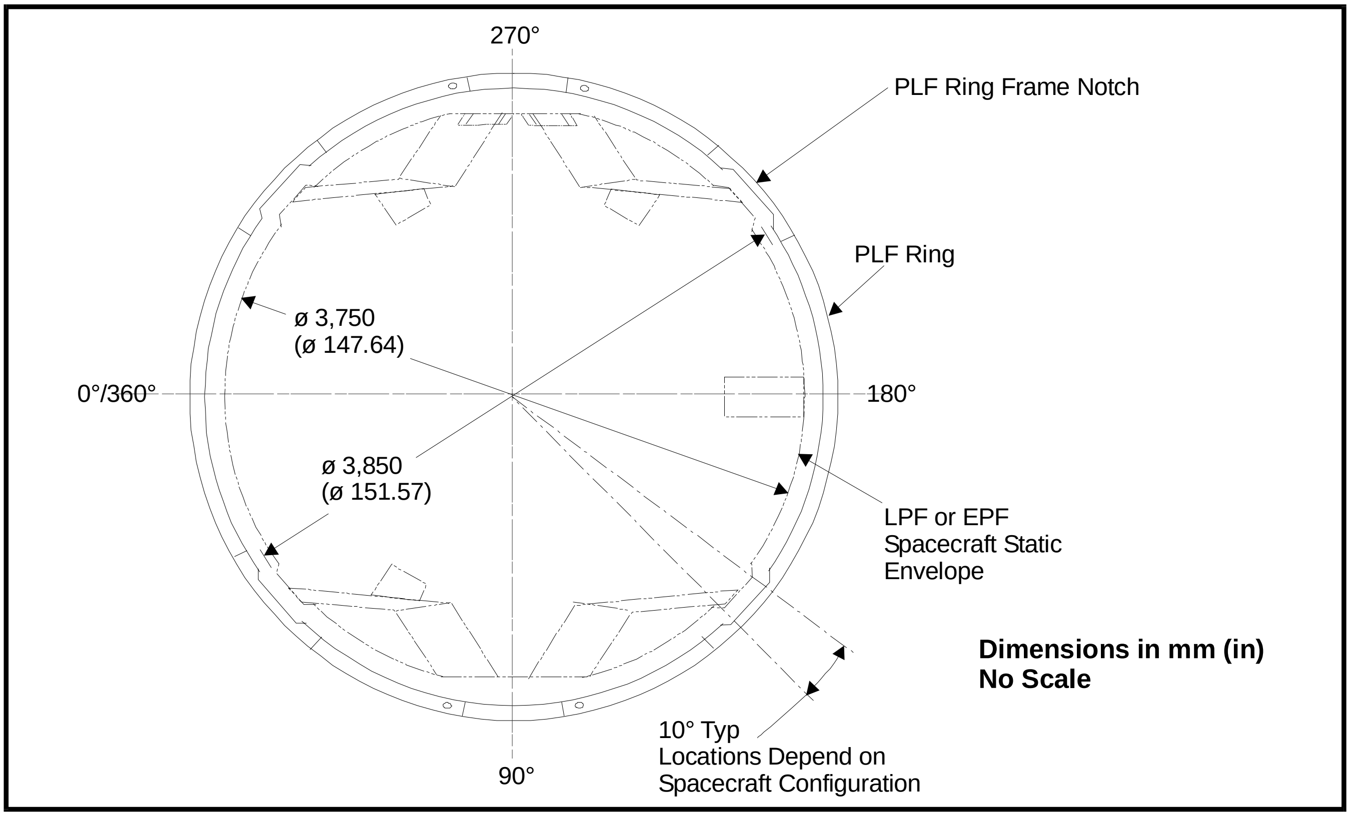


Figure 4.1.1.2-4 LPF or EPF Static Envelope Notch Option

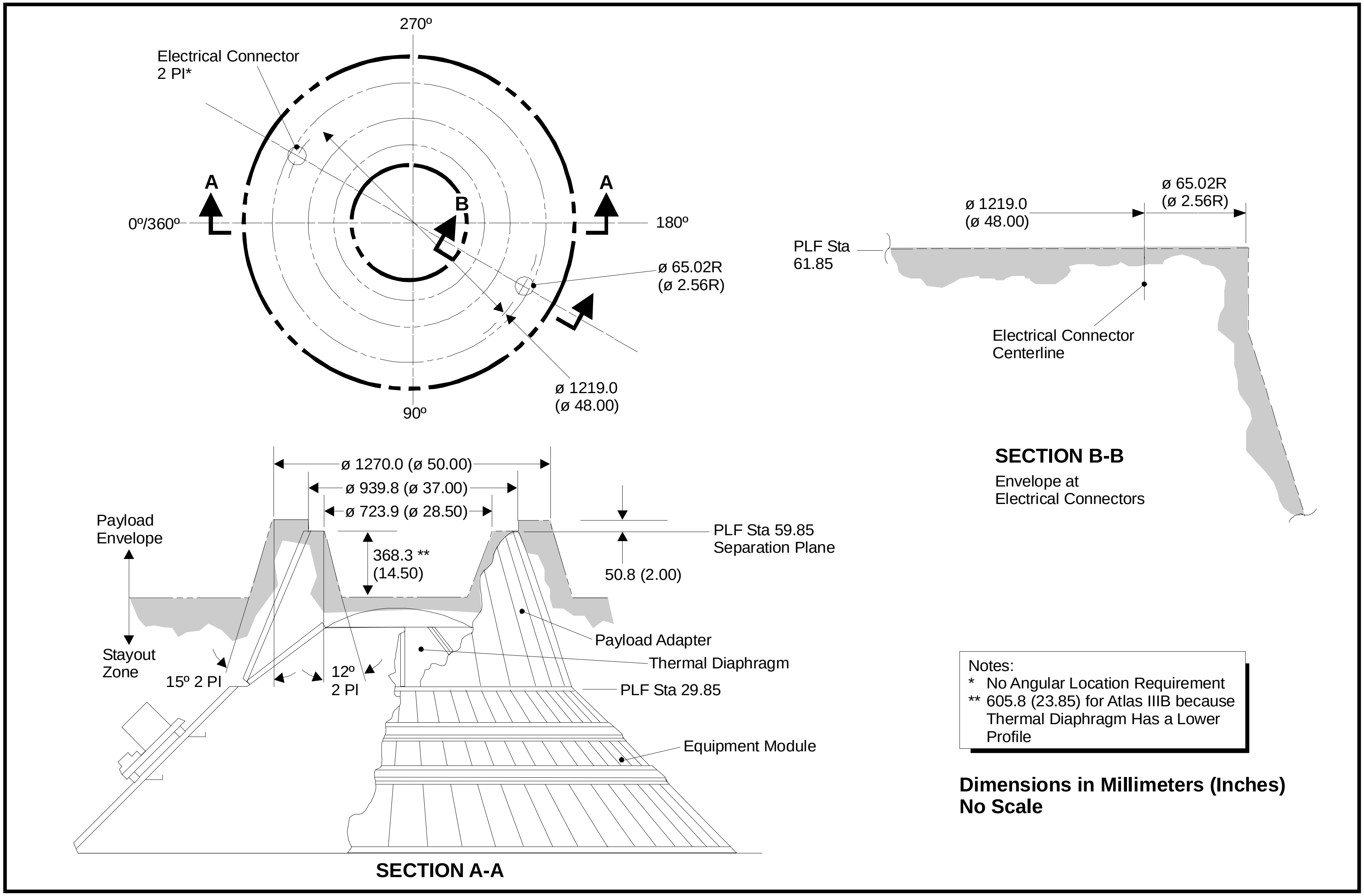


Figure 4.1.1.2-5 Type A Payload Adapter Envelope

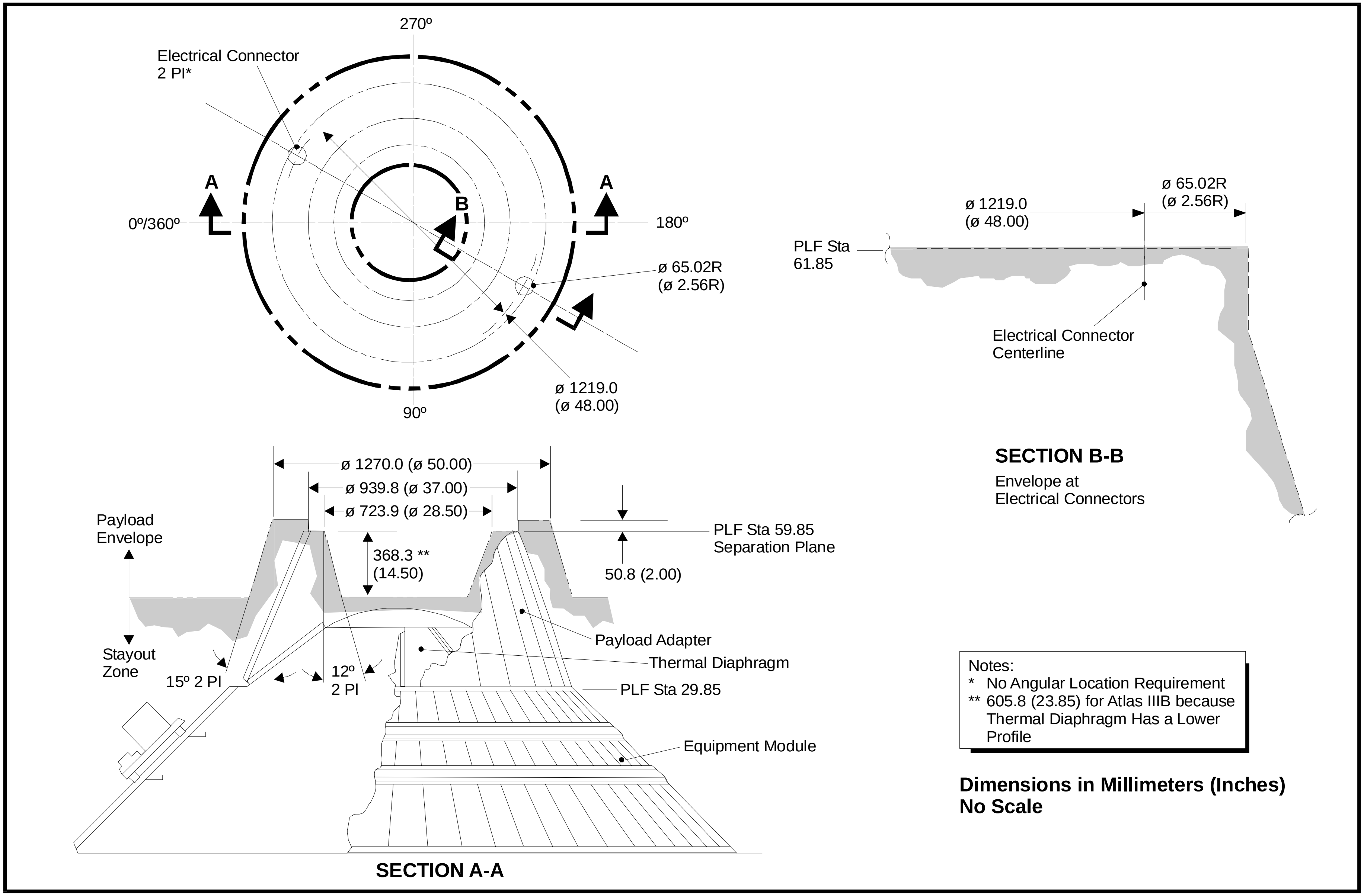


Figure 4.1.1.2-6 Type A1 Payload Adapter Envelope

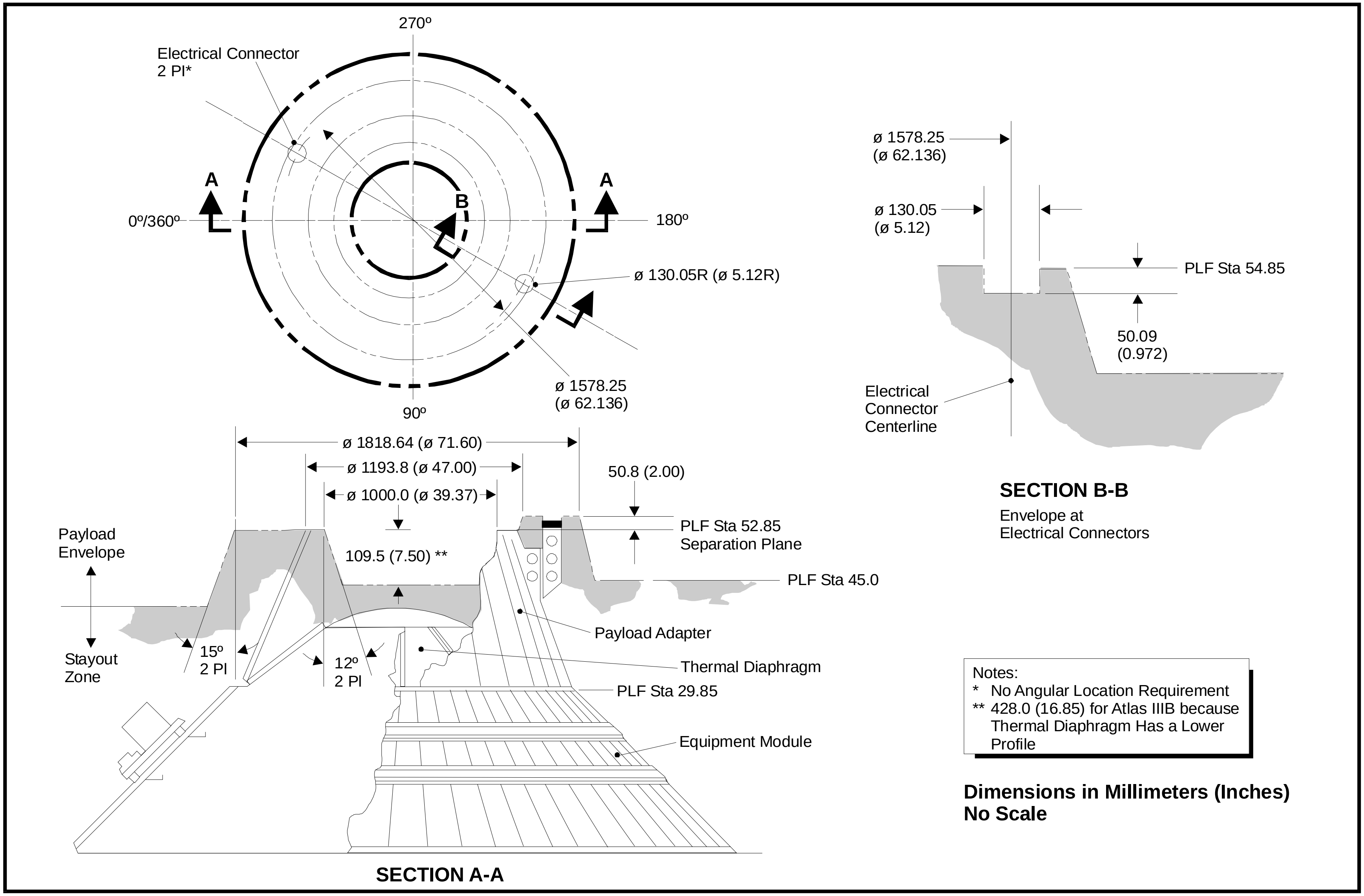


Figure 4.1.1.2-7 Type B Payload Adapter Envelope

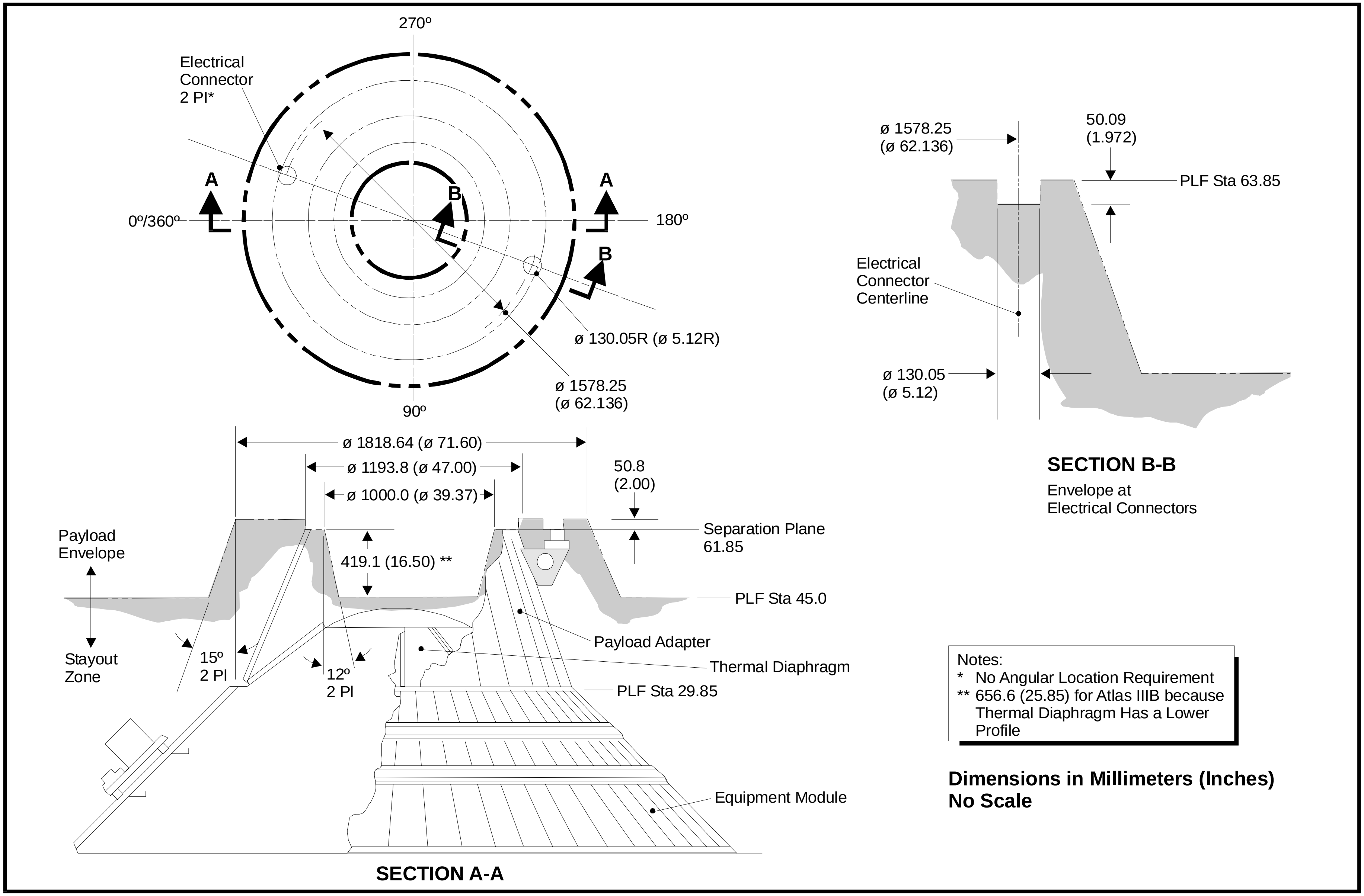


Figure 4.1.1.2-8 Type B1 Payload Adapter Envelope

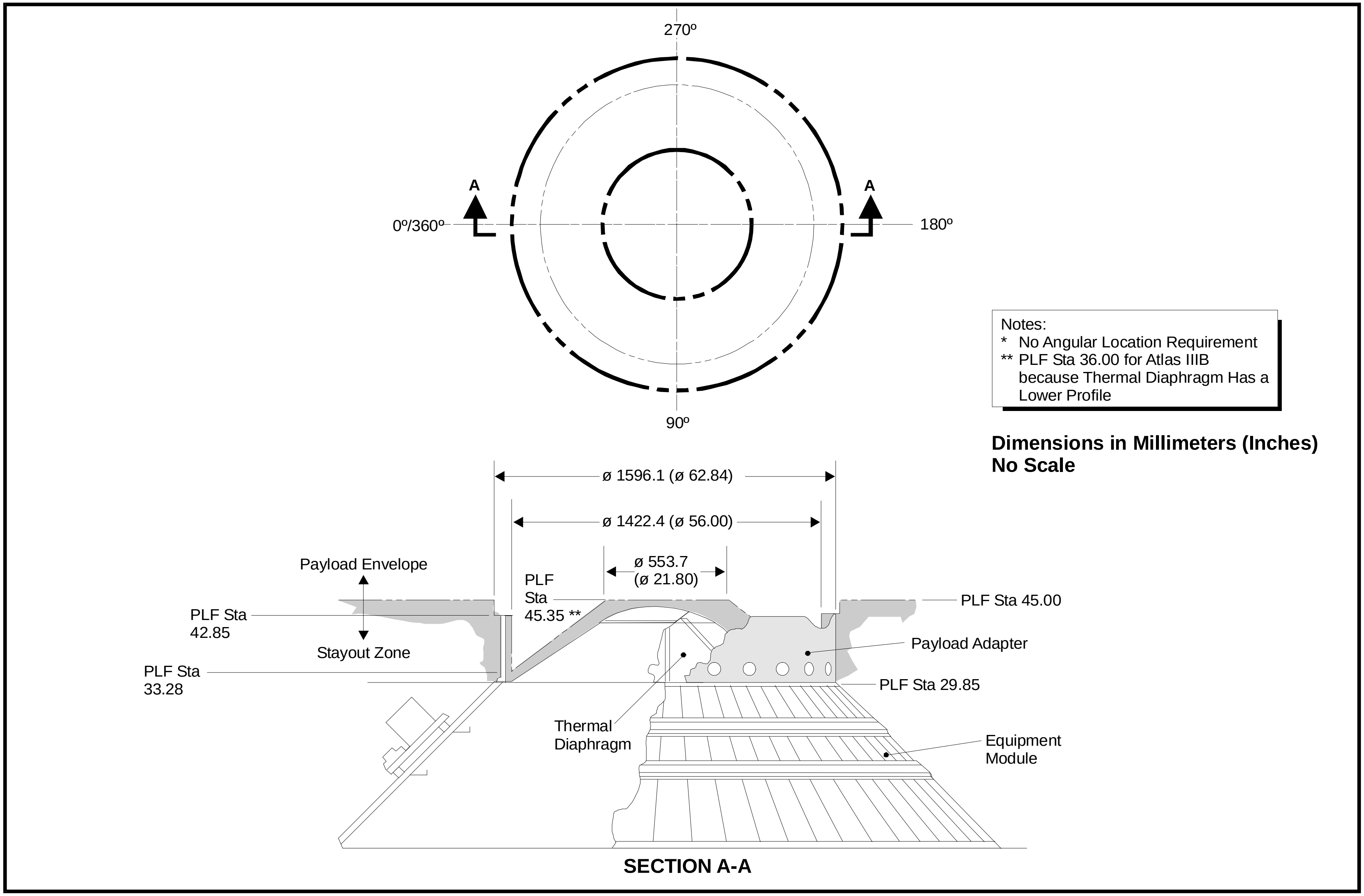


Figure 4.1.1.2-9 Type C and C1 Payload Adapter Envelope

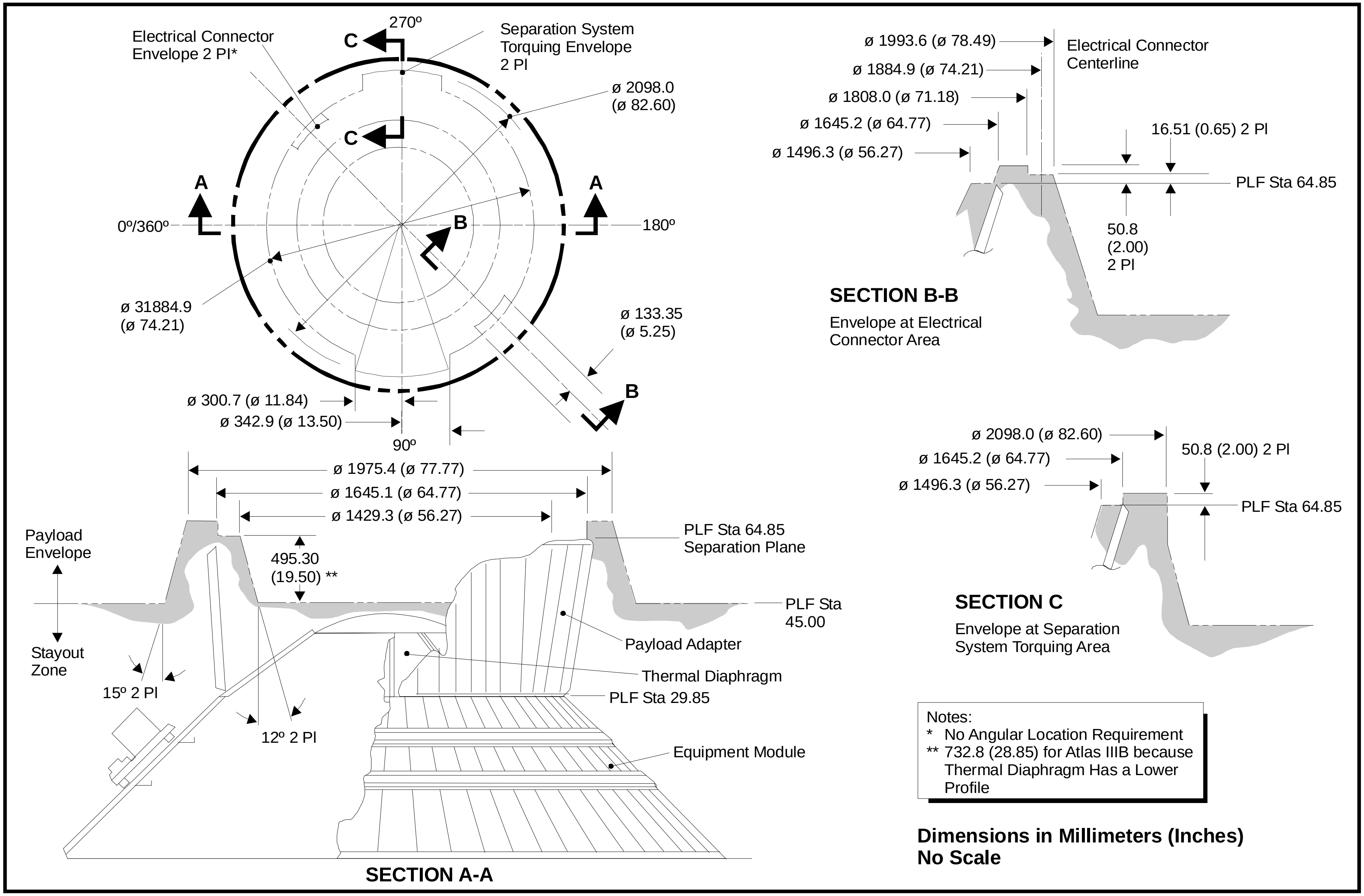


Figure 4.1.1.2-10 Type D Payload Adapter Envelope

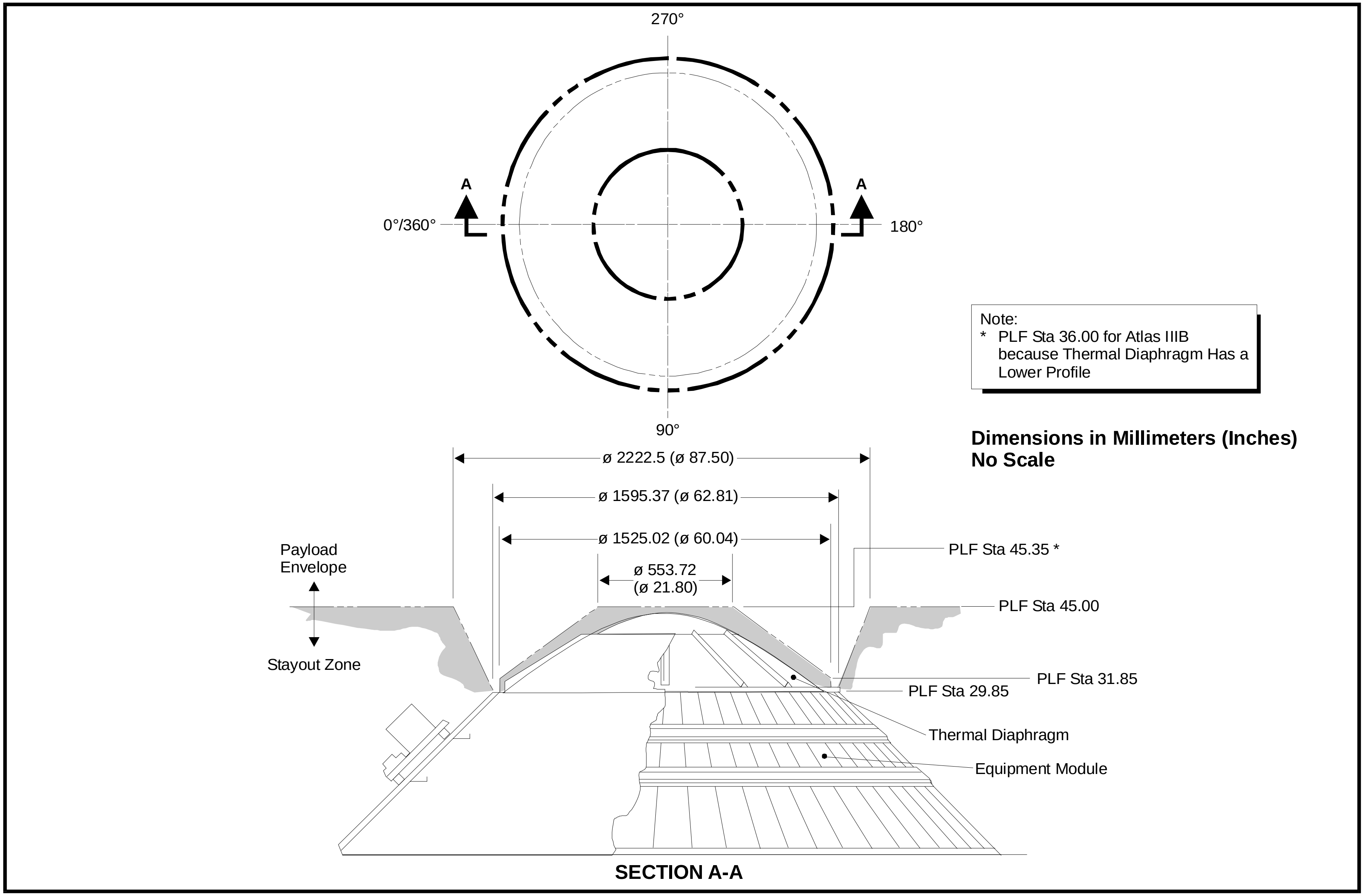


Figure 4.1.1.2-11 Equipment Module Envelope

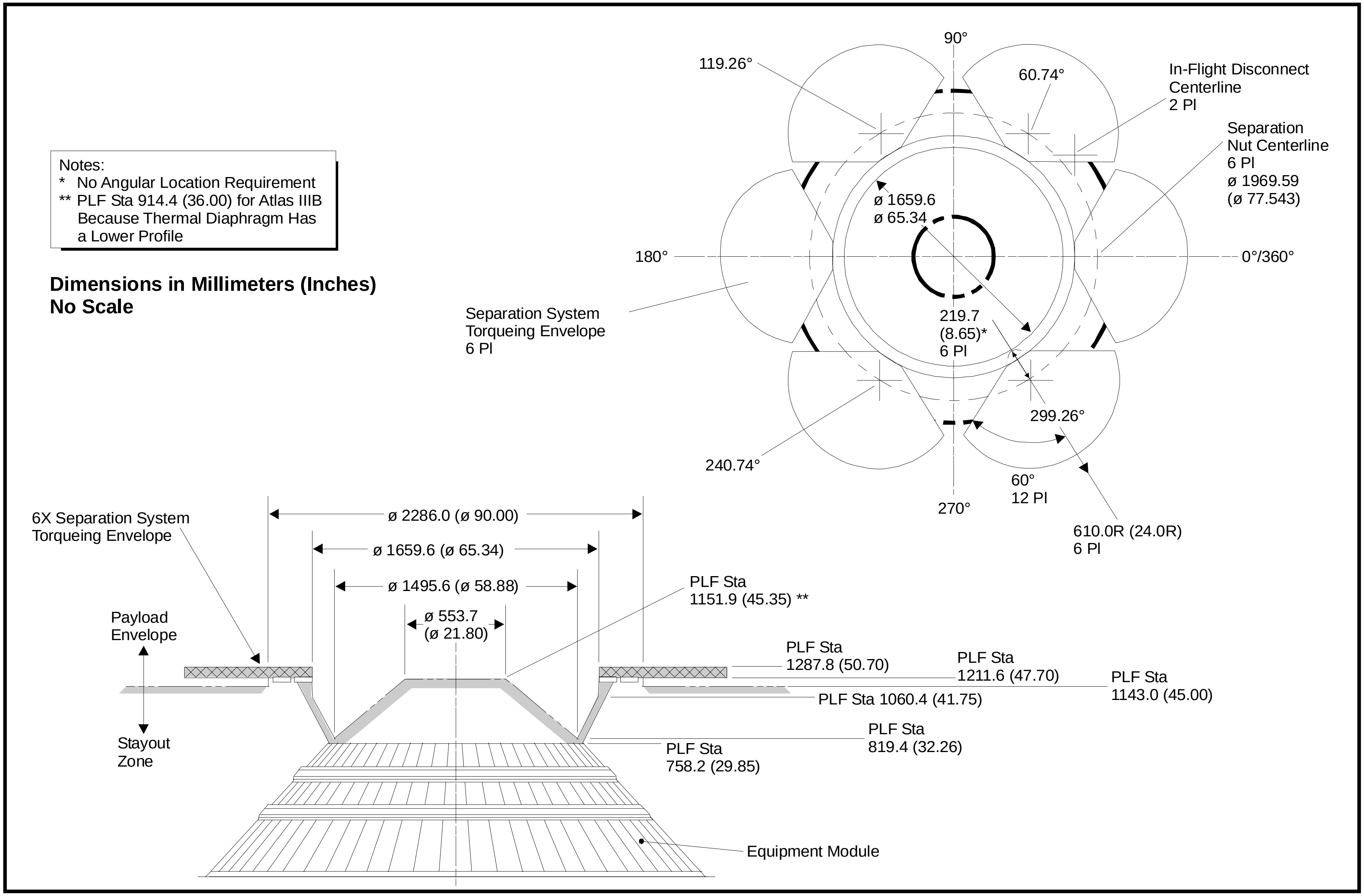


Figure 4.1.1.2-12 Type E Payload Adapter Envelope

4.1.2 Mechanical Interface—Payload Adapters

The payload adapter supports the spacecraft on top of the launch vehicle equipment module and provides the mechanical and electrical interfaces between the launch vehicle and spacecraft. The Atlas offers eight different payload adapters all of which use of an aluminum skin, stringer, and frame construction with machined forward and aft rings that mate to the spacecraft and launch vehicle equipment module (Figure 4.1.2-1). The primary structural interface between the launch vehicle and spacecraft occurs at the payload adapter forward ring. This ring interfaces with the spacecraft aft ring and a payload separation system holds the two rings together for the structural joint and provides the release mechanism for spacecraft separation. The payload adapter also provides mounting provisions for the separation springs, supports the interfacing components for electrical connectors between the launch vehicle and spacecraft, and supports additional mission-unique hardware, including range safety destruct units**,** as necessary. Electrical bonding is provided across all mechanical interface planes associated with these adapters.

The Atlas program offers a range of spacecraft interface options including Marmon-type clamp-band separation systems and discrete separation nut interfaces. There is also the option of bolting directly to the top of the equipment module or spacer style adapters for spacecraft which provide their own adapter and separation system. All of the Atlas payload adapters and separation systems are designed so that they can modified to allow the spacecraft to be oriented inside of the payload fairing to best meet mission requirements. In addition, alternate adapter designs can be developed on a mission-peculiar basis.

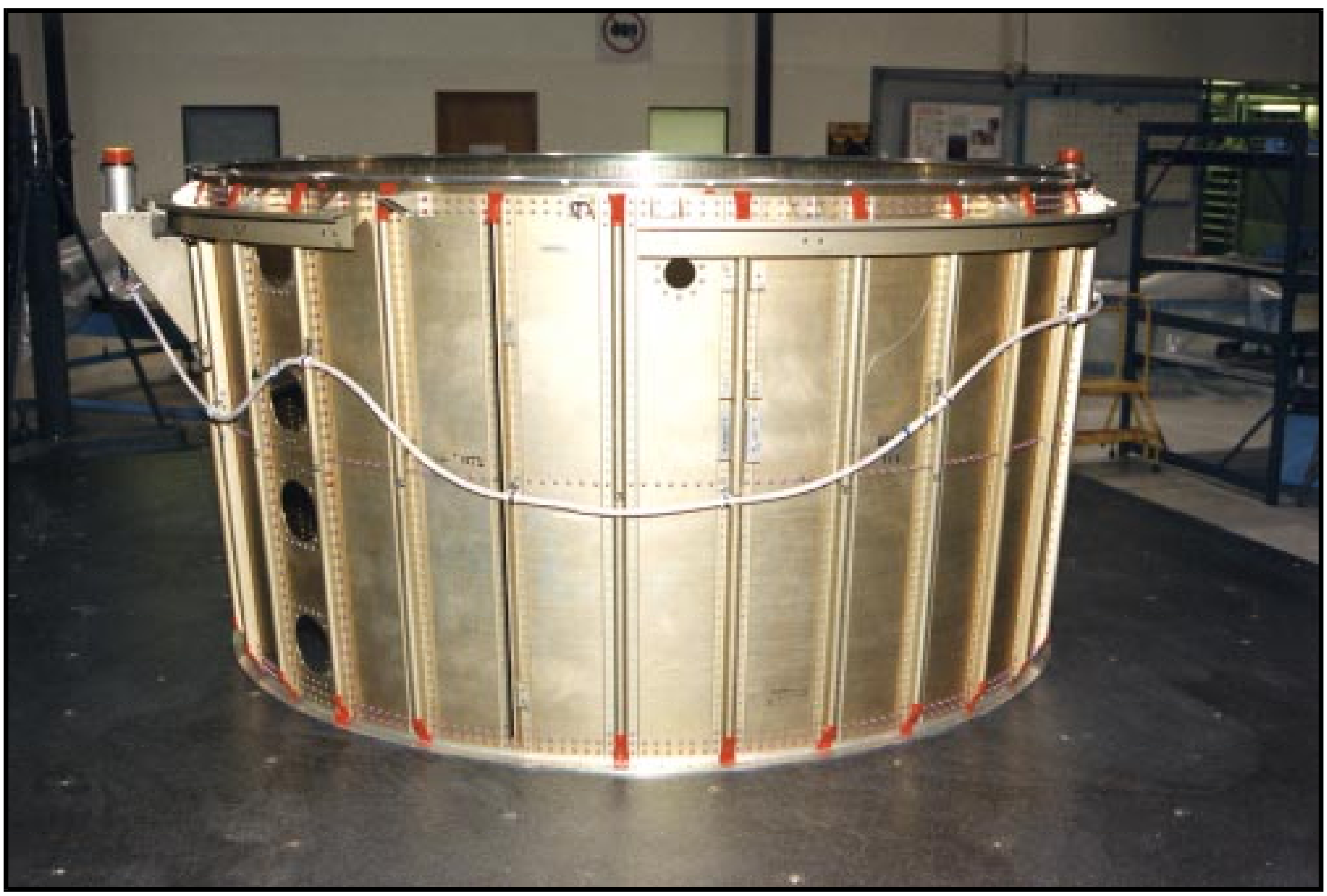


Figure 4.1.2-1 Atlas Payload Adapter

4.1.2.1 Payload Adapter Interfaces—The Atlas Type A, A1, B, B1, and D adapters use a launch vehicle provided Marmon-type clampband payload separation system. These separation systems (Fig. 4.1.2.1-1) consist of a clamp band set and separation springs to give the necessary separation energy after the clamp band is released. The clamp band set consists of a clamp band for attaching the spacecraft to the adapter structure plus devices to extract, catch, and retain the clamp band on the adapter structure after separation. The separation spring assemblies are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through continuity loops installed in the spacecraft electrical connector and wired to the upper stage instrumentation for monitoring and telemetry verification.

The Type A adapter has a 945-mm (37-in.) diameter forward interface ring and is compatible with PAM-DII/937B adapters. The Type A1 adapter has a 959-mm (38-in.) diameter forward interface ring and is compatible with the PAM-D/937A. The Type B and B1 adapters have a 1,215-mm (47-in.) diameter forward interface ring and are compatible with the 1194A adapter. The Type D adapter has a 1,666 mm (66.0 inch) diameter forward interface ring and is compatible with the 1666A adapter.

The Atlas Type E Adapter was developed to support a heavy 4,990-kg (11,000-lb) large diameter (1,956-mm /77-in.) payload using a six-hardpoint interface with the spacecraft. The 1956-mm (77-in.) payload separation system (PSS77) consists of a six-separation nut set that attaches the payload to the forward ring of the payload adapter, and a separation spring set that provides the necessary separation energy after the separation nut is actuated. The separation nut set (Fig. 4.1.2.1-2) consists of the separation nut, separation stud, and stud catcher, which captures the stud on the payload structure after separation. For spacecraft requiring a lower shock environment, there is the option of using a fast acting shockless separation nut (FASSN) with the Type E adapter. The FASSN system is designed to rapidly separate a bolted joint while producing minimal shock. A 0.5-in. diameter bolt version of this system

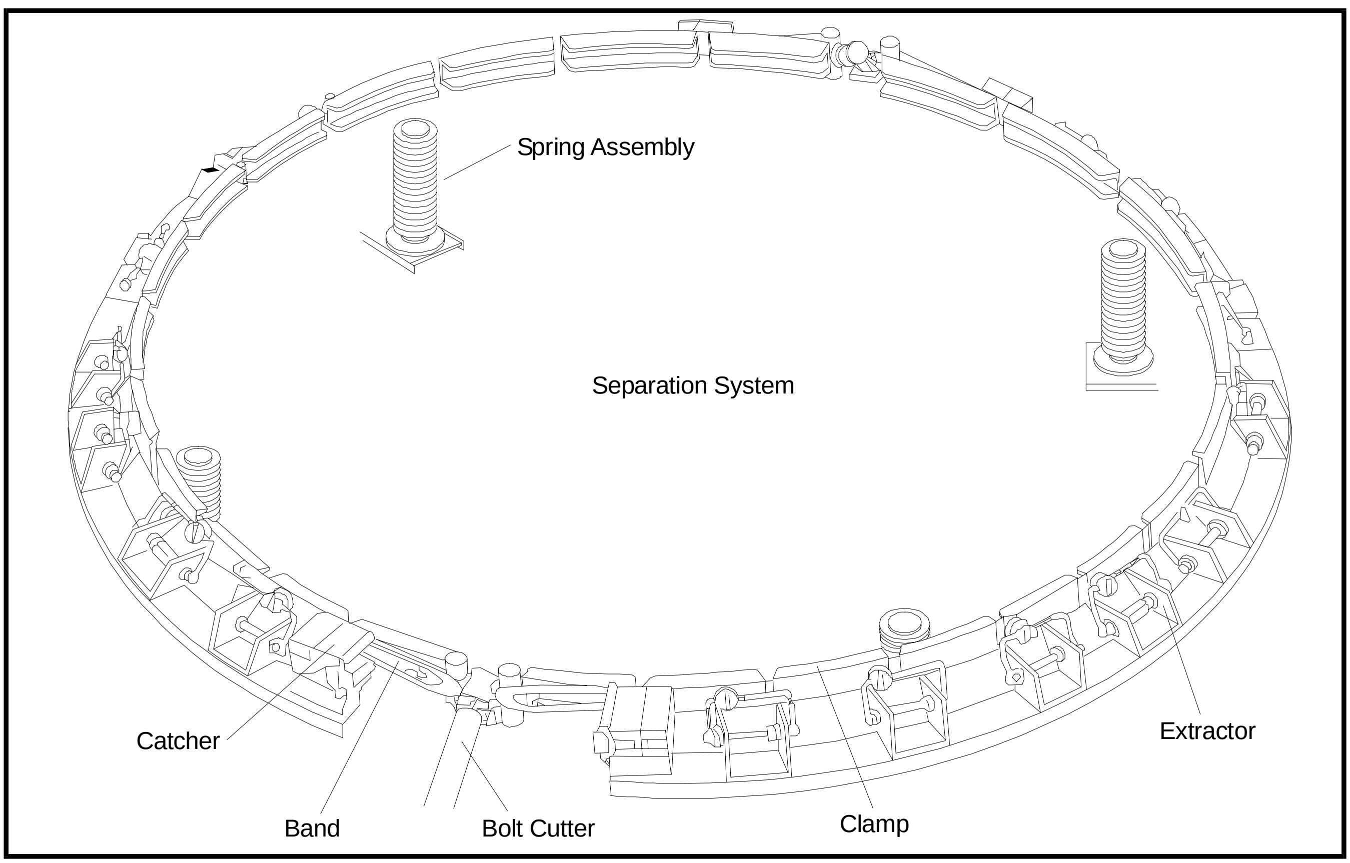


Figure 4.1.2.1-1 The separation system attaches the spacecraft to the payload adapter.

has been qualified for flight, including more than 450 successful activations, and will be used on a Lockheed Martin built spacecraft in 1998. The separation spring is mounted on the aft side of the payload adapter forward ring, and the pushrod of each spring acts through a hole in the forward ring to push against the payload aft structure. The spring set is sized appropriately to provide the proper separation velocity between launch vehicle and payload and begins acting immediately after actuation of the separation nut. The separation sequence is initiated by redundant commands from the upper-stage guidance and control to fire the redundant pyrotechnic pressure cartridges of each separation nut. Power for this operation is supplied from the main vehicle battery. Positive spacecraft separation is detected through continuity loops installed in the spacecraft interface connector and wired to the upper stage instrumentation for monitoring and telemetry verification. Coordinated tooling between the spacecraft and payload adapter is required for this system.

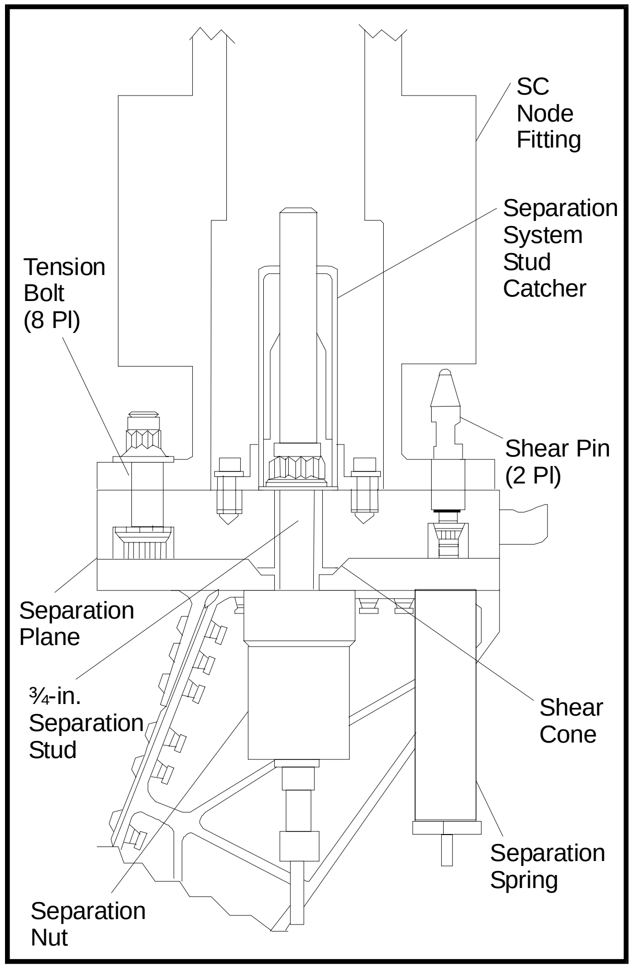


Figure 4.1.2.1-2 Separation System Design Overview

For spacecraft which provide their own adapter and separation system, a bolted interface is provided by the equipment module or Type C and C1 adapters. If a customer-provided spacecraft adapter is used, it must provide interfaces for ground handling, encapsulation and transportation equipment. In particular, there will need to be provisions for three torus arm fittings and an encapsulation diaphragm unless a launch vehicle supplied intermediate adapter is used.

Figures 4.1.2.1-3 through 4.1.2.1-8 show the interfaces for the Type A, A1, B, B1, C, C1 and D adapters. Figure 4.1.2.1-9 shows the interfaces for the equipment module. Figure 4.1.2.1-10 shows the interface for the Type E adapter.

4.1.2.2 Spacecraft Interfaces—The primary spacecraft interface with the launch vehicle occurs at the spacecraft aft ring. This ring must be compatible with the launch vehicle payload adapter forward ring and with the payload separation system including the separation springs. The location and configuration of additional interfacing components, in particular the electrical connectors, is coordinated through interface control drawings. The interface requirements for spacecraft using the Atlas Type A, A1, B, B1, and D are shown in Figures 4.1.2.2-1 through 4.2.2.2-4. For spacecraft which provide their own adapters, an bolted interface suitable for mating to the equipment module, Type C or C1 adapters as shown in Figures 4.1.2.1-5, 4.1.2.1-6, or 4.1.2.1.-8 is required.

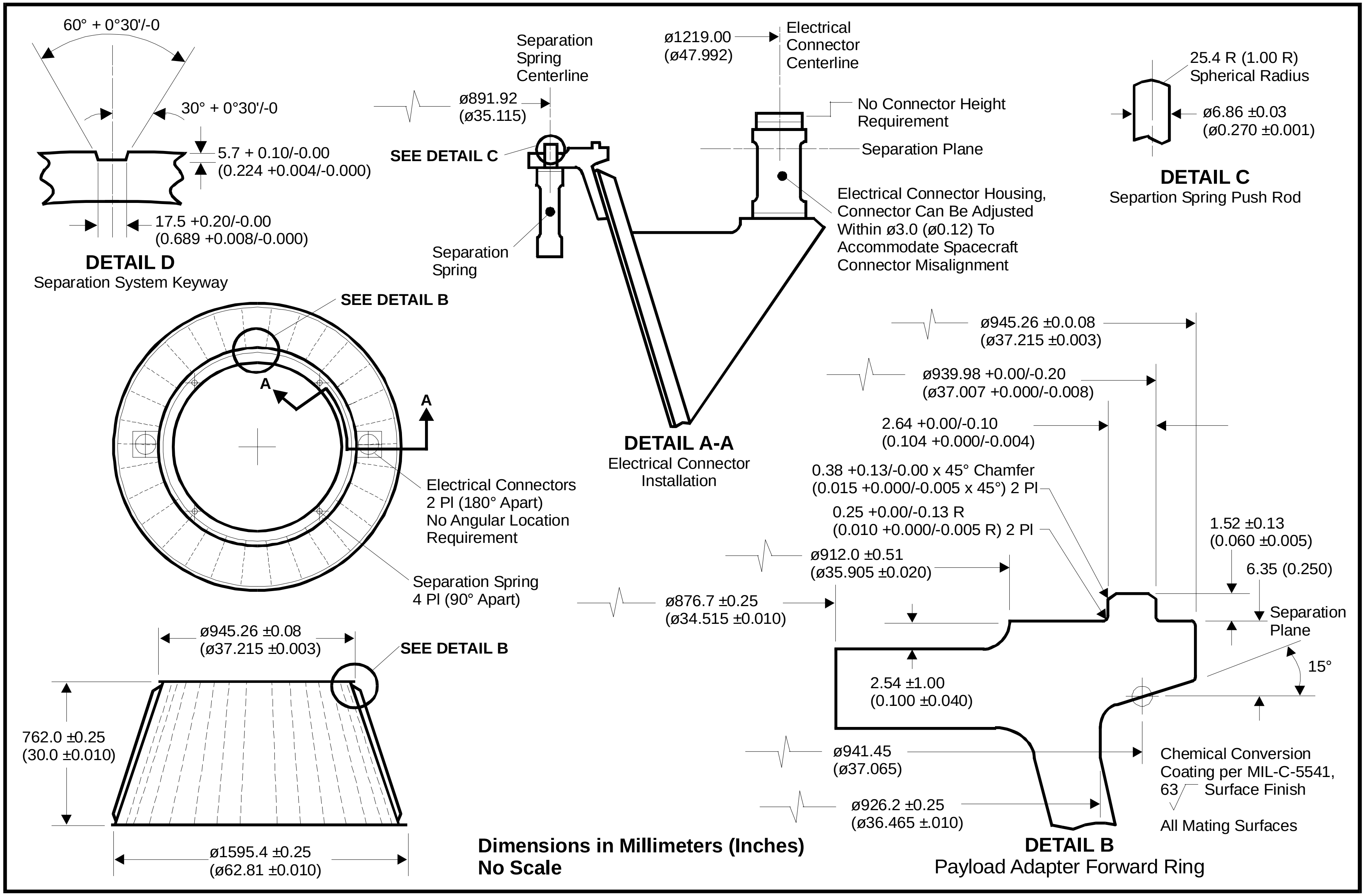


Figure 4.1.2.1-3 Type A Payload Adapter Mechanical Interfaces

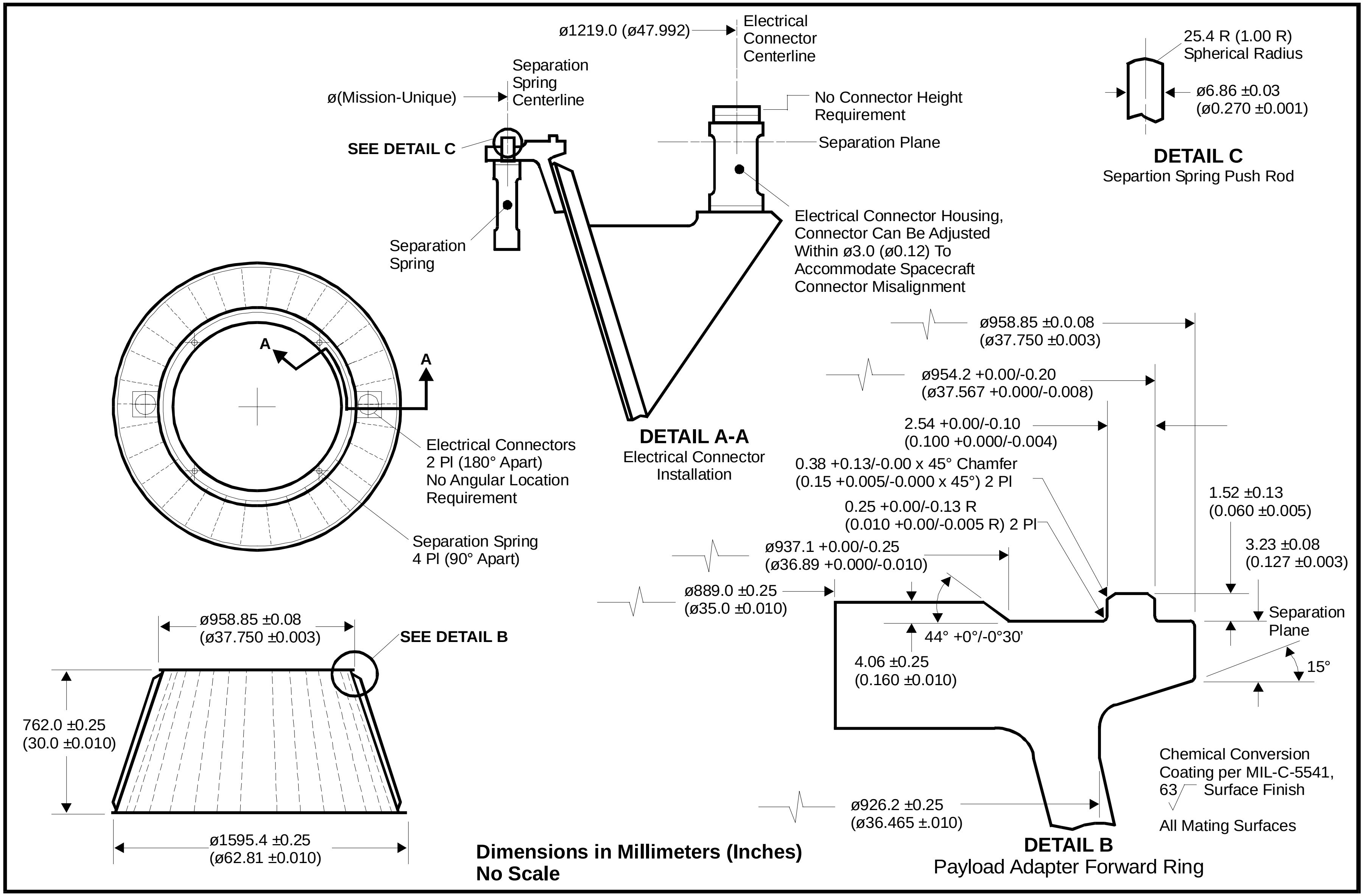


Figure 4.1.2.1-4 Type A1 Payload Adapter Mechanical Interfaces

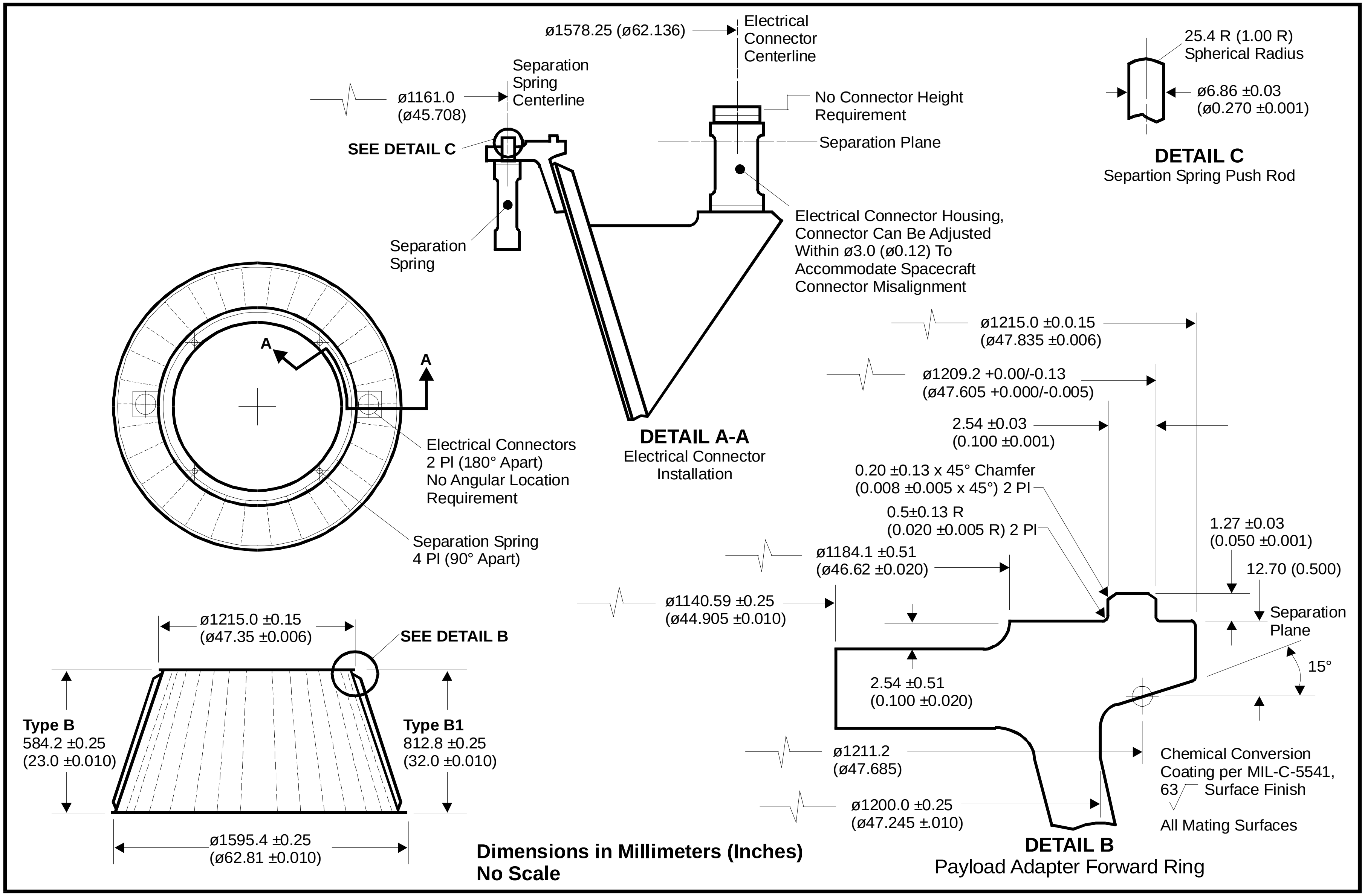


Figure 4.1.2.1-5 Type B/B1 Payload Adapter Mechanical Interfaces

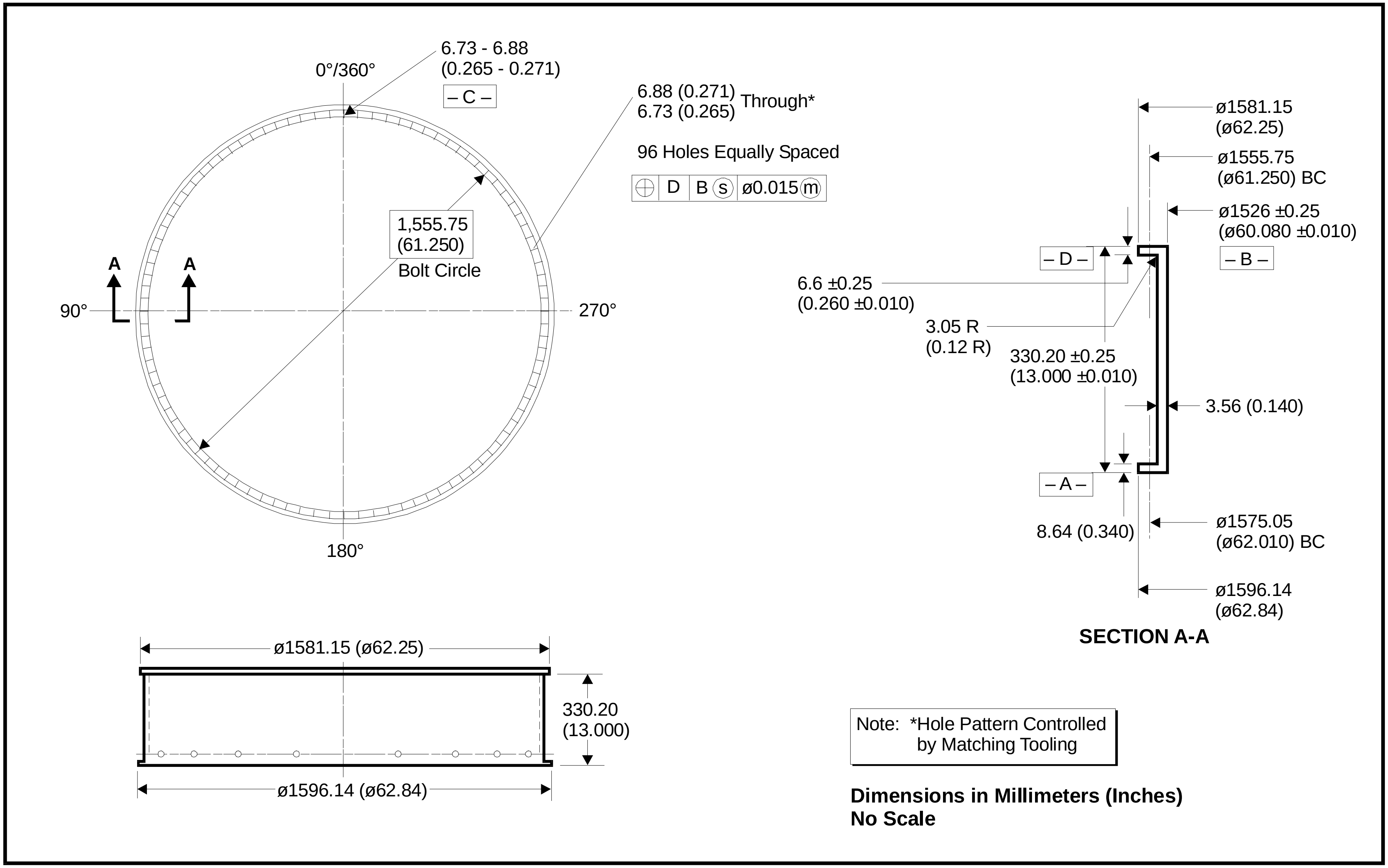


Figure 4.1.2.1-6 Type C Payload Adapter Mechanical Interfaces

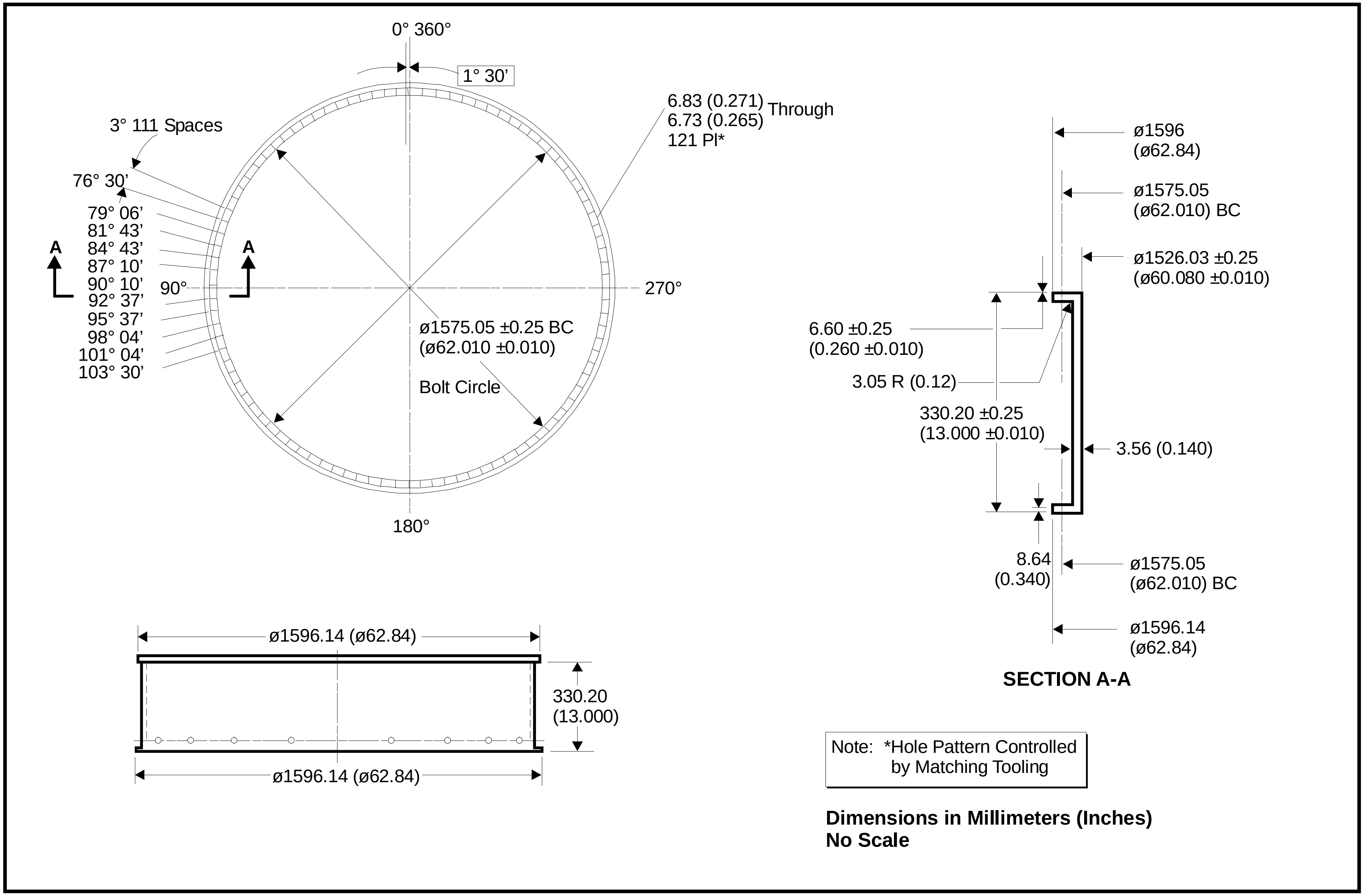


Figure 4.1.2.1-7 Type C1 Payload Adapter Mechanical Interfaces

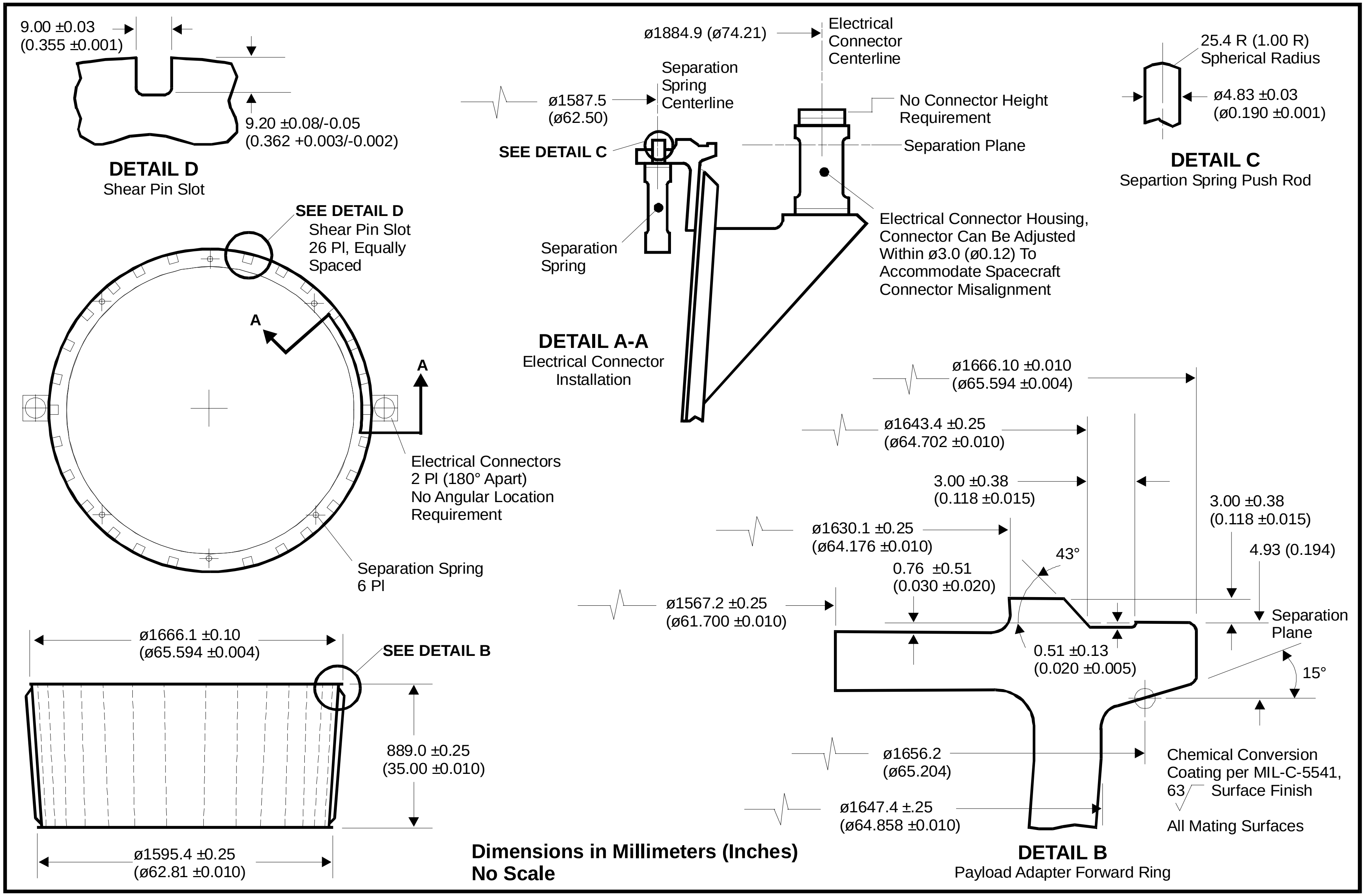


Figure 4.1.2.1-8 Type D Payload Adapter Mechanical Interfaces

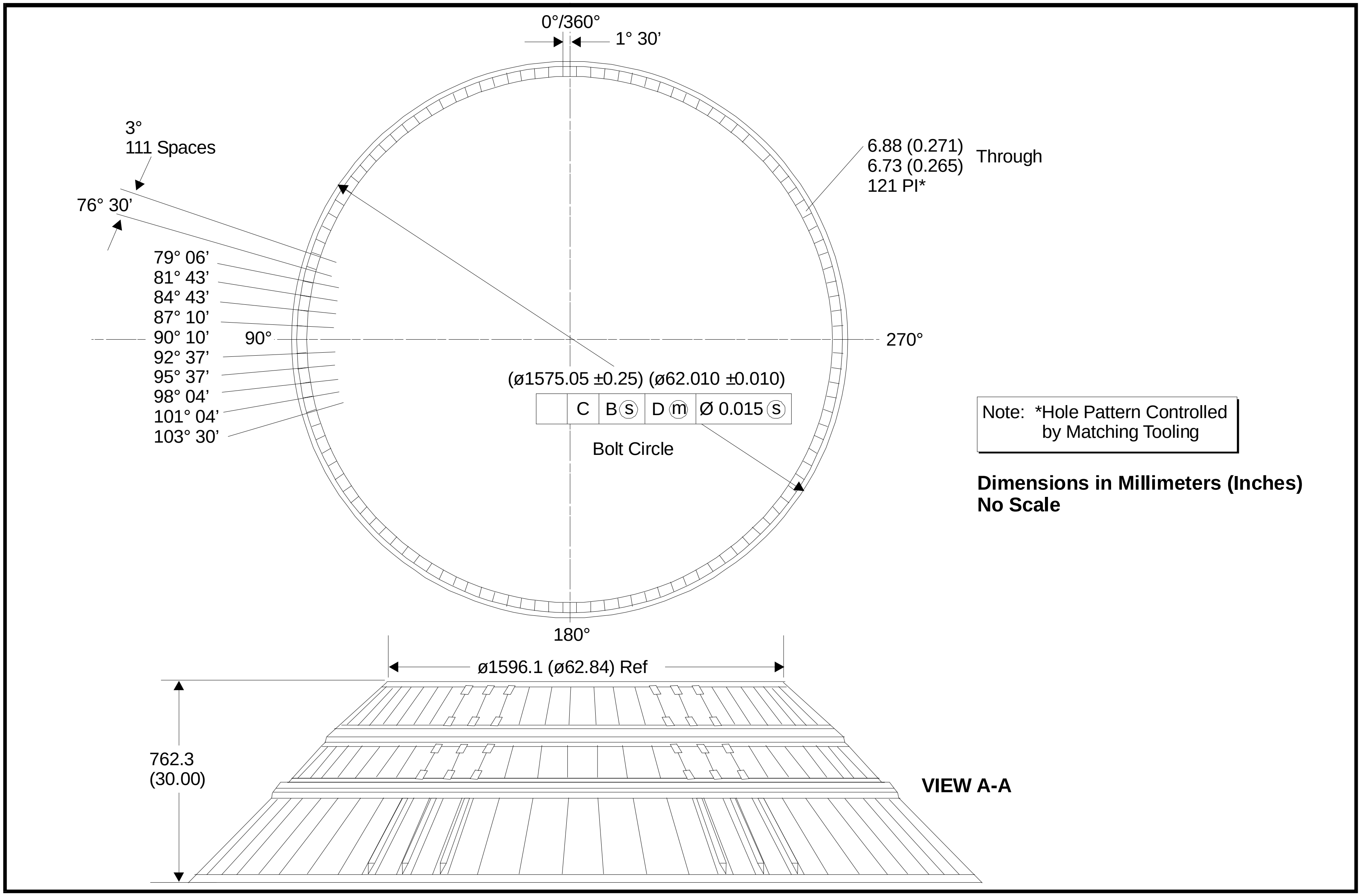


Figure 4.1.2.1-9 Equipment Module Mechanical Interfaces

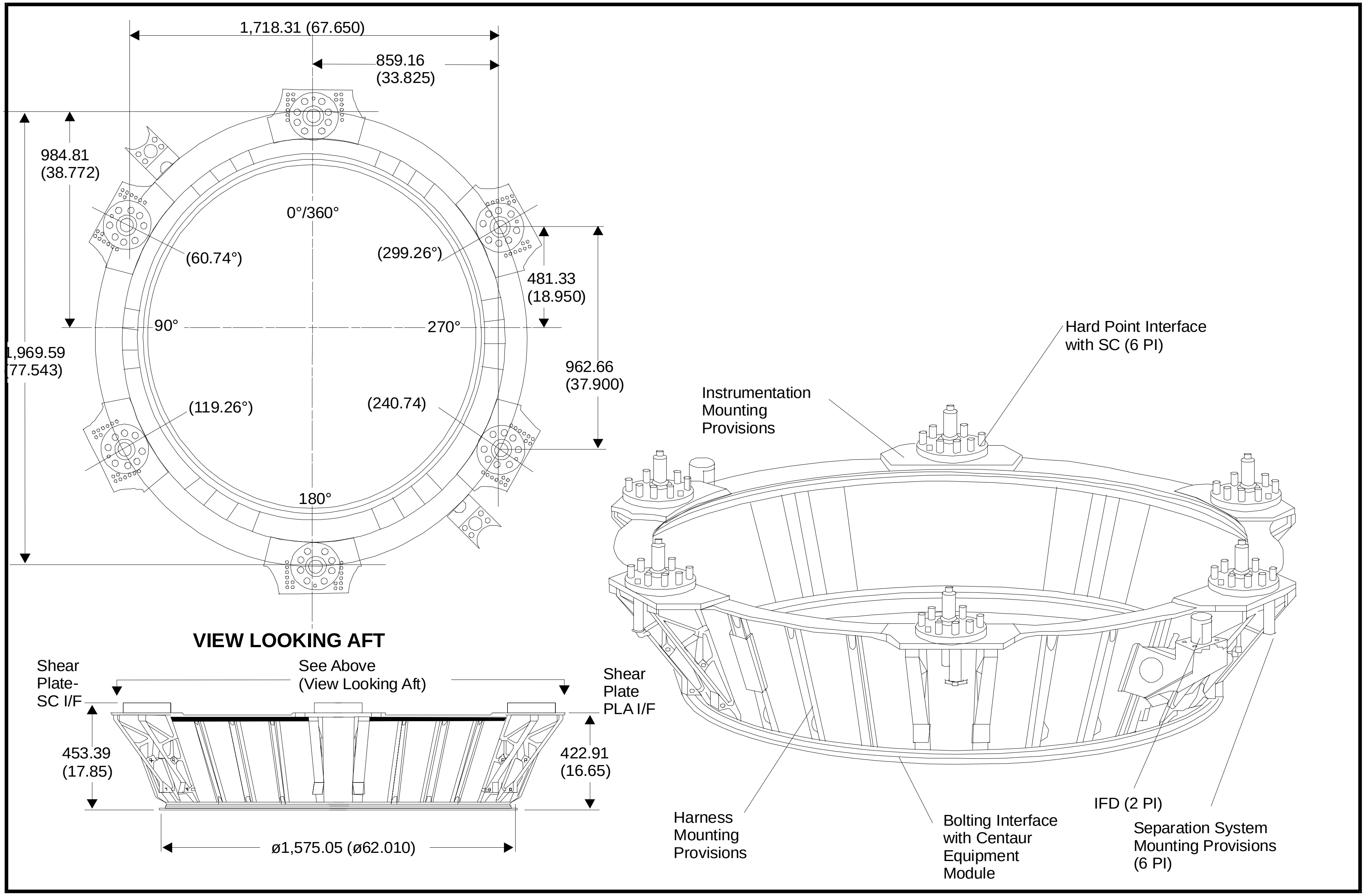


Figure 4.1.2.1-10 Type E Payload Adapter Mechanical Interface

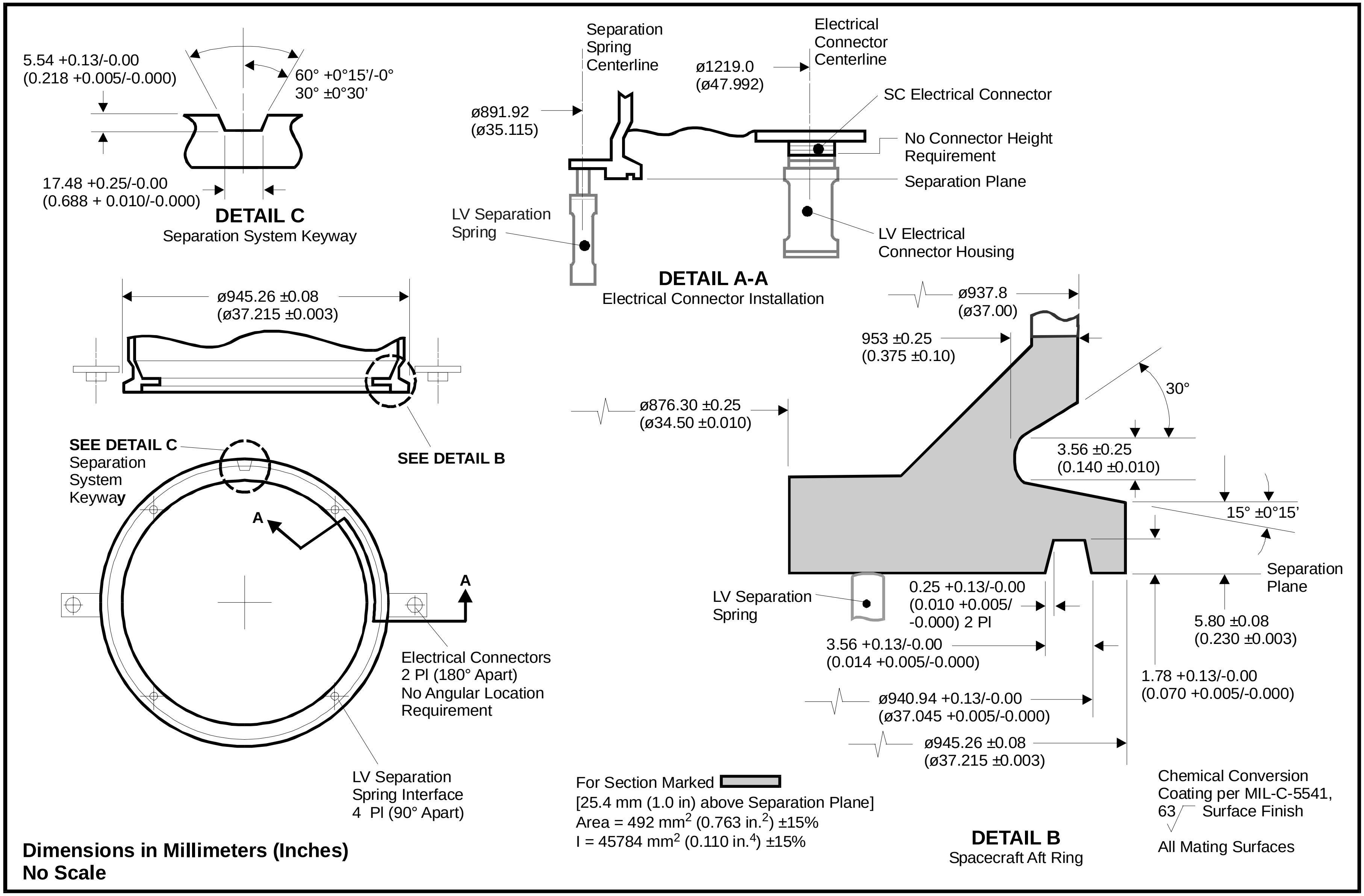


Figure 4.1.2.2-1 Type A Adapter Spacecraft Interface Requirements

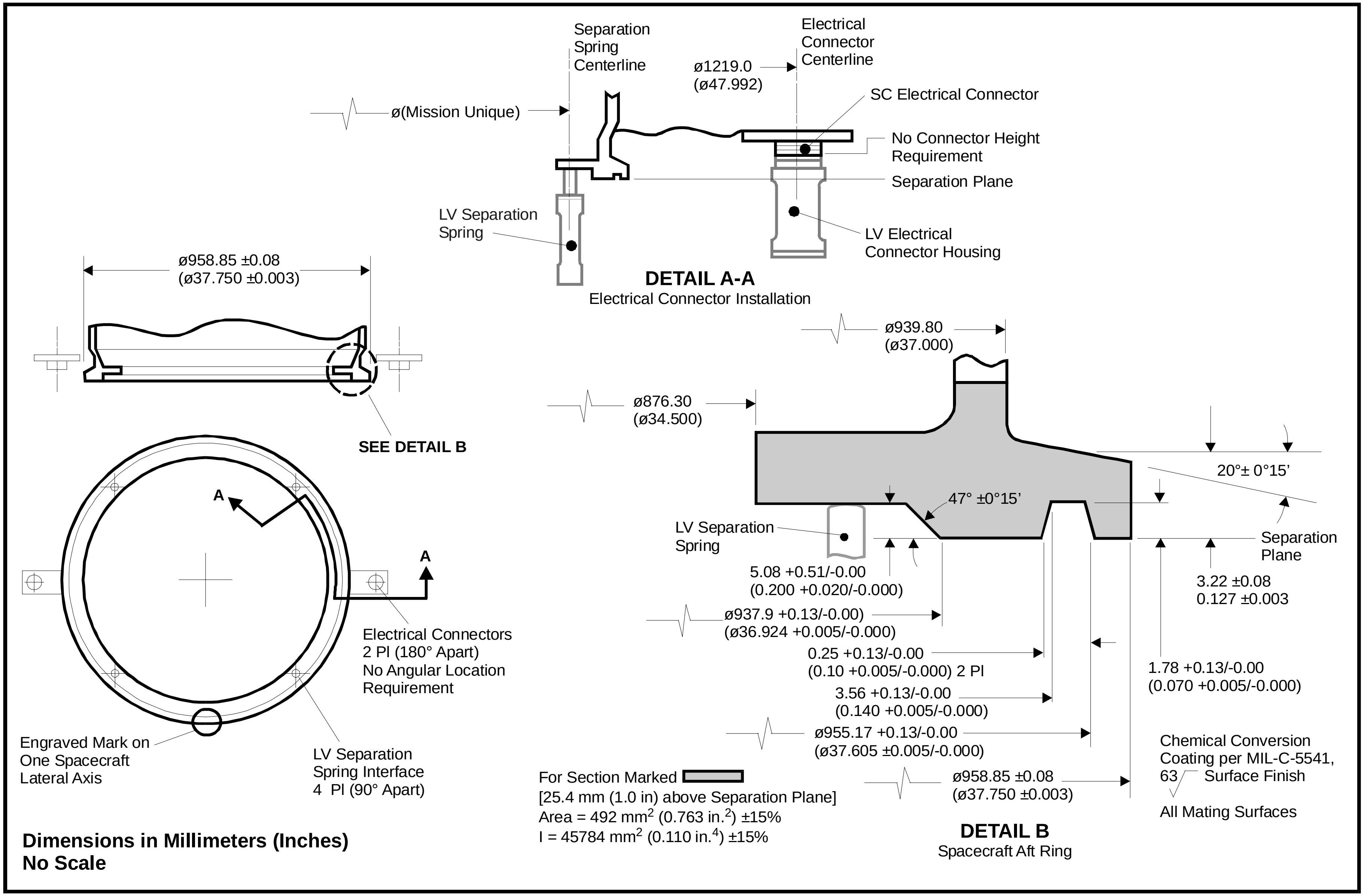


Figure 4.1.2.2-2 Type A1 Adapter Spacecraft Interface Requirements

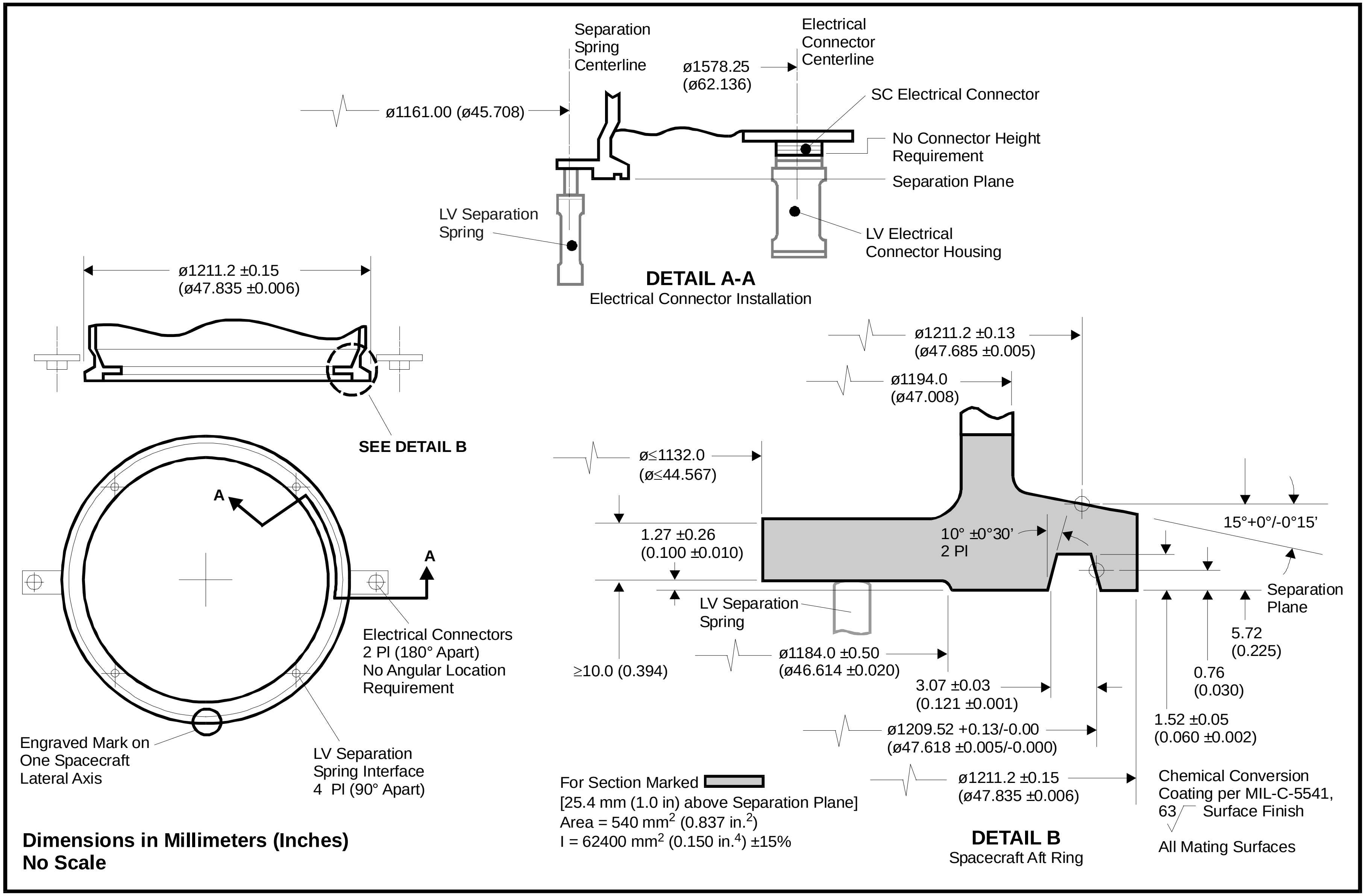


Figure 4.1.2.2-3 Types B and B1 Adapter Spacecraft Interface Requirements

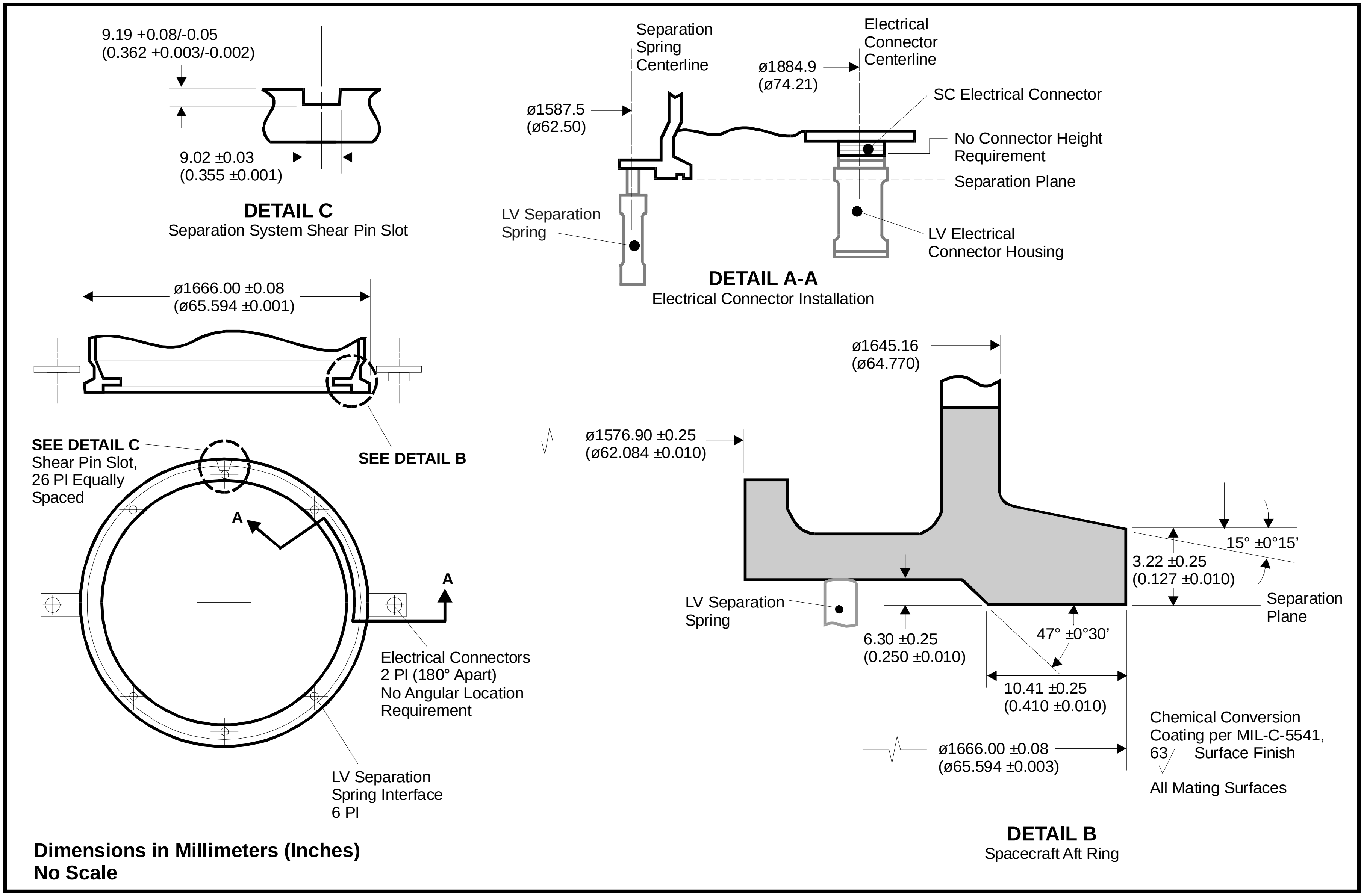


Figure 4.1.2.2-4 Type D Adapter Spacecraft Interface Requirements

4.1.2.3 Spacecraft Adapter Structural Capabilities—The allowable spacecraft weights and longitudinal centers of gravity for the Type A, A1, B, B1, D, and E adapter/separation systems are shown in Figure 4.1.2.3-1. Figure 4.1.2.3-1 also shows equipment module load capability. This curve is used to assess the structural capability of the launch vehicle when user-supplied spacecraft adapters are used.

These spacecraft mass and CG capabilities were determined using generic spacecraft interface ring geometry as shown in Figures 4.1.2.2-1 through 4.1.2.2-4, and quasi-static load factors shown in Table 3.2.1.1-1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-peculiar coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables. (See Section 8.0 for additional payload adapter and separation system capability options.)

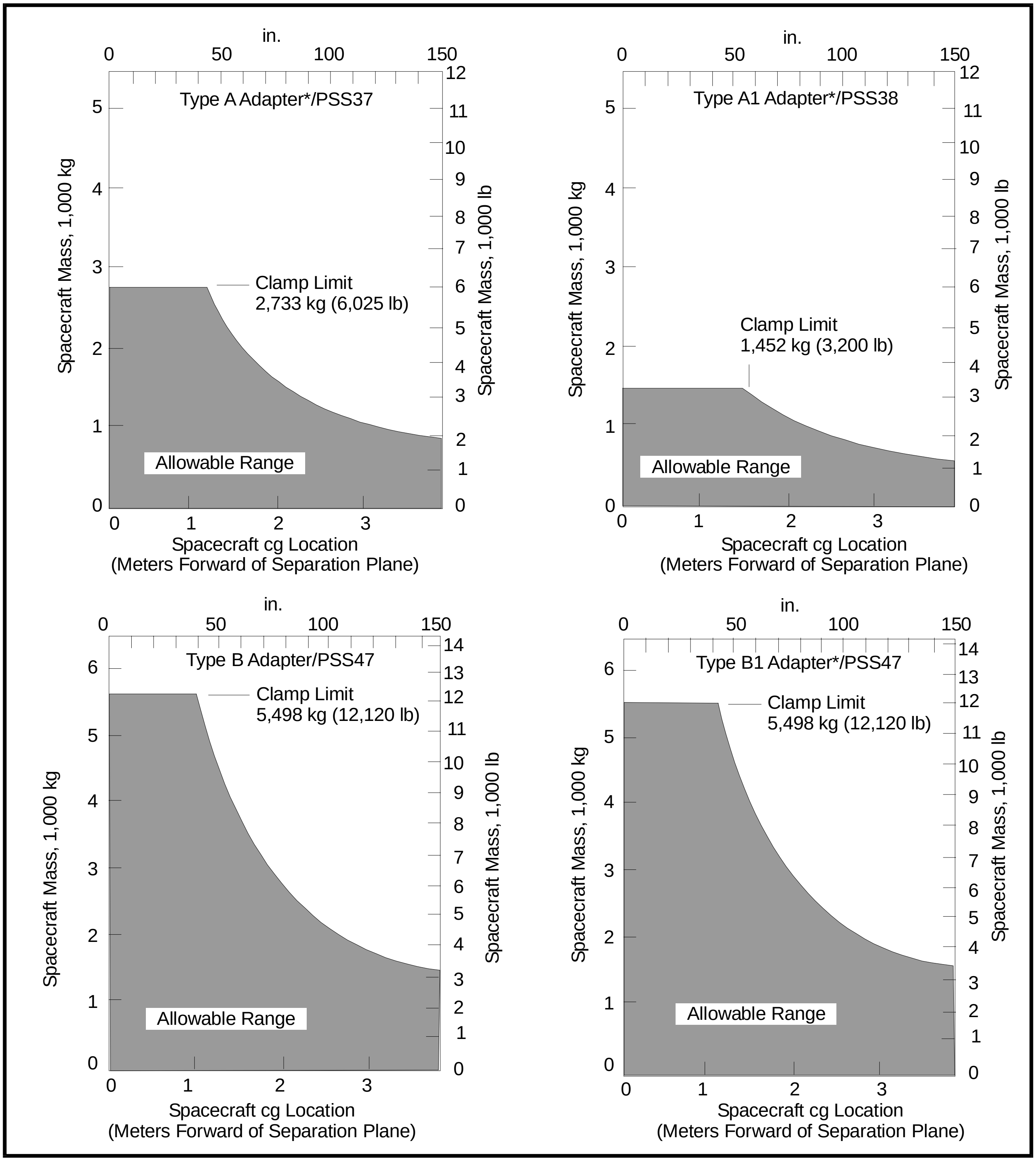
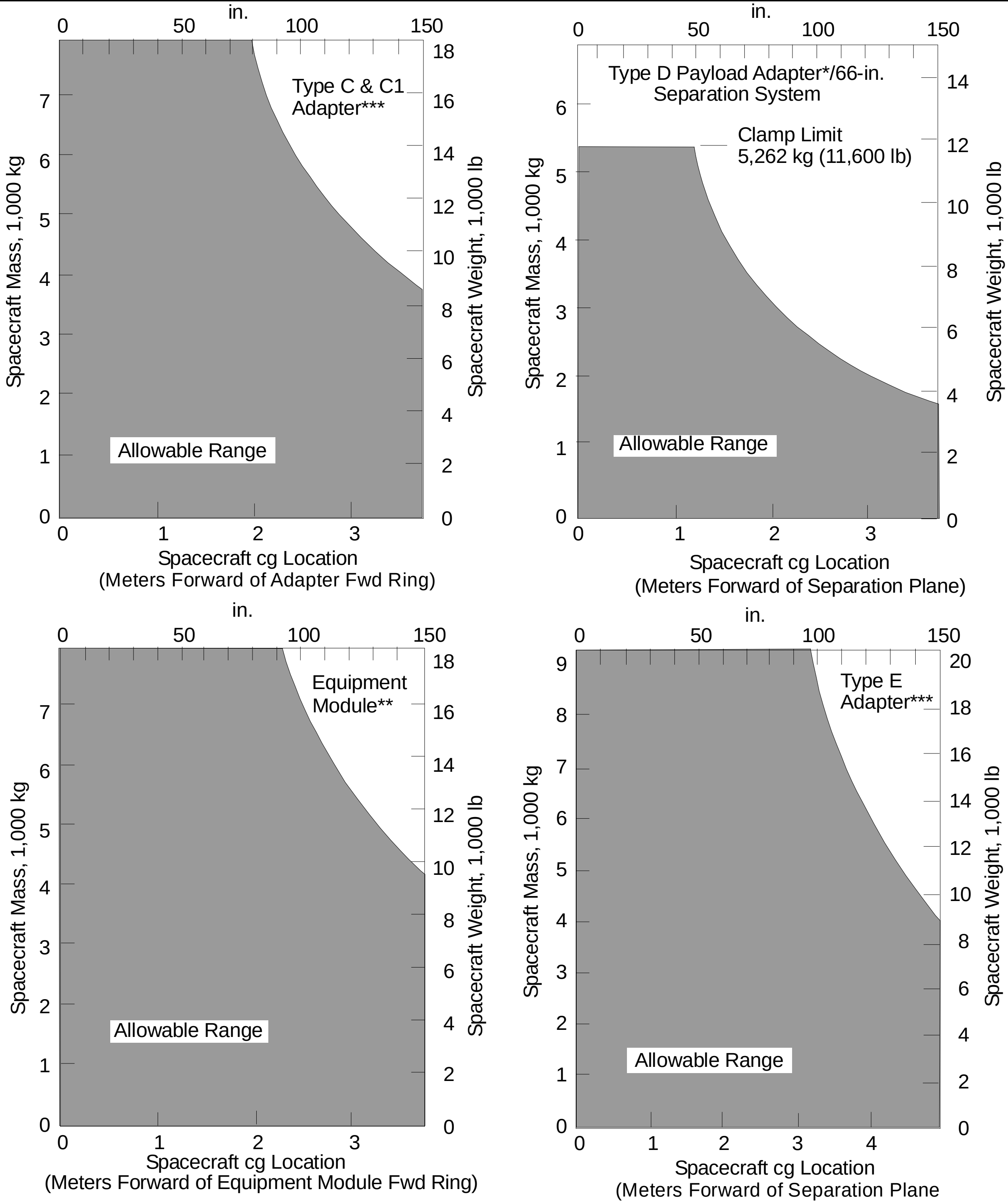


Figure 4.1.2.3-1 Equipment Module and Space Vehicle Adapter/Separation System Structural Capabilities



Note:
* Contact Lockheed Martin if Spacecraft Design Exceeds Clamp Limits

** Contact Lockheed Martin if Spacecraft Design Exceeds Equipment Module Limits

*** Structural Capability of Type C, C1 & E Adapters Limited by Equipment Module

Figure 4.1.2.3-1 (concl)

4.1.3 Electrical Interfaces

The spacecraft/launch vehicle electrical interfaces are shown in Figures 4.1.3-1 and 4.1.3-2. Typical standard interfaces include:

1) A spacecraft-dedicated umbilical interface between the umbilical disconnect located on the Centaur upper stage and rise-off disconnects at the spacecraft/launch vehicle interface;
2) Spacecraft/launch vehicle separation indicators located in the spacecraft/launch vehicle rise-off disconnects to verify separation;
3) A spacecraft destruct interface activated by the Centaur Range Safety system;
4) Standard rise-off disconnects or other connectors from MIL-C-81703 that may be required by mission-peculiar changes. A unique keying arrangement for each connector is highly recommended. For standard connectors, the following part numbers apply:

37 Pin	MS3446E37-50P	(LV-Side)
	MS3464E37-50S	(S/C-Side)
61 Pin	MS3446E61-50P	(LV-Side)
	MS3464E61-50S	(S/C-Side)

The launch vehicle can also be configured to provide electrical interfaces for various mission-peculiar requirements. The complement of signals available is as follows: two separation commands, 16 control commands that can be configured as 28-V discretes, a standard umbilical with a mix of wire configurations, and an instrumentation interface, which contains two discrete inputs for detection of spacecraft separation, four analog inputs for general use, ten command feedback discretes, and two serial data interfaces for downlinking data from the spacecraft, if desired.

The following paragraphs describe the Atlas electrical interfaces in detail.

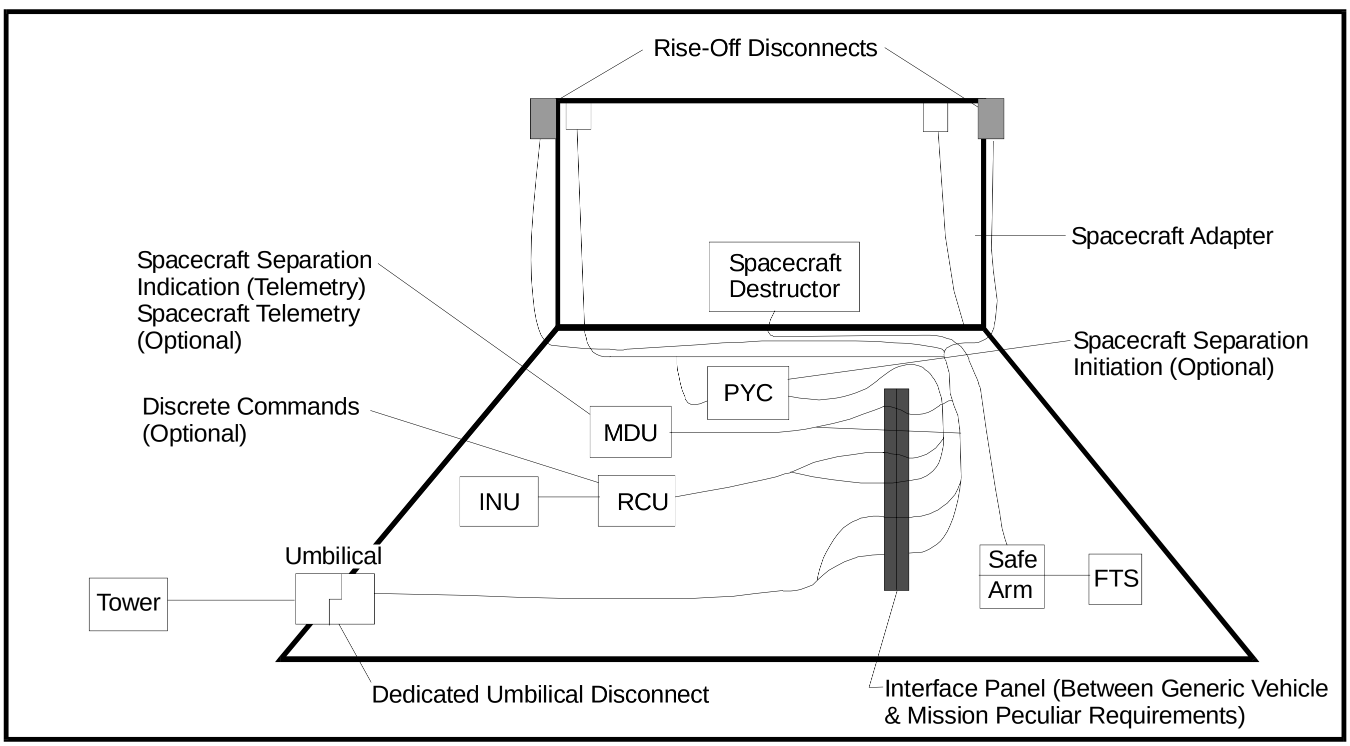


Figure 4.1.3-1 Typical Spacecraft/Launch Vehicle Electrical Interface

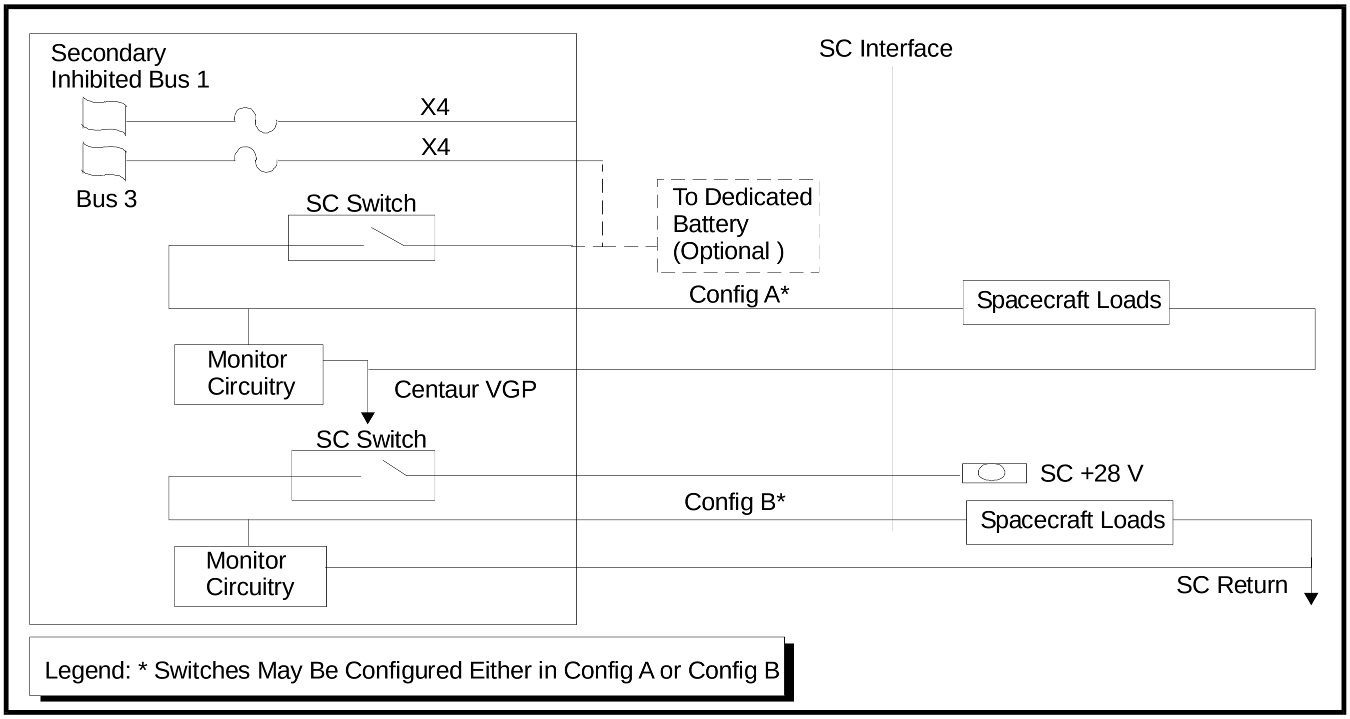


Figure 4.1.3-2 Atlas IIA, IIAS, IIIA, and IIIB Spacecraft Switch Interface

4.1.3.1 Umbilical Interface—A spacecraft-dedicated umbilical disconnect for on-pad operations is located on the Centaur forward umbilical panel. The umbilical interfaces with two spacecraft-dedicated rise-off disconnects located on the spacecraft (Fig. 4.1.3-1). This umbilical interface provides signal paths between the spacecraft and ground support equipment for spacecraft system monitoring during prelaunch and launch countdown.

The umbilical disconnect separates at liftoff. The two spacecraft rise-off disconnects separate at spacecraft/Centaur separation.

The spacecraft/ground support equipment (GSE) umbilical contains the following complement of wires from spacecraft to umbilical disconnect:

1) 41 twisted shielded wire pairs—20 american wire gage (AWG);
2) 6 twisted shielded wire triples—20 AWG;
3) 6 twisted shielded controlled impedance wire pairs—75Ω ±10%.

Additional disconnects for signal interface between the spacecraft and launch vehicle can be incorporated for mission-peculiar requirements. These disconnects also separate at spacecraft/Centaur separation.

4.1.3.2 Electrical In-Flight Disconnects—The standard adapters provide for two rise-off disconnect options of either 37 pins or 61 pins for the spacecraft interface. These rise-off disconnects typically provide a spacecraft-dedicated umbilical interface between the spacecraft and GSE. Lockheed Martin can provide more disconnects on a mission-peculiar basis if the spacecraft requires them. Interface requirements for the disconnects are shown in Figures 4.1.2.2-1, 4.1.2.2-2, and 4.1.2.2-3.

4.1.3.3 Spacecraft Separation System—The baseline separation system for spacecraft/launch vehicle separation is a pyrotechnic V-type clamp band system. The Type E PSS77 separation system uses separation nuts activated by pyrotechnic command.

The separation sequence is initiated by redundant commands from the upper-stage guidance system. Power for this is supplied from the main vehicle battery.

Positive spacecraft separation is detected via continuity loops installed in the spacecraft rise-off disconnects and wired to the Centaur instrumentation system. The separation event is then telemetered to the ground.

4.1.3.4 Control Command Interface—For the Atlas IIA and Atlas IIIA vehicles, the remote control unit (RCU) provides as many as 16 control commands to the spacecraft. For Atlas IIAS and Atlas IIIB, two switches are used for solid rocket booster ISDS resulting in up to 14 being available for spacecraft use. These commands are solid state switches configured as 28-Vdc commands.

Closure of the switches is controlled by the inertial navigation unit (INU). Parallel digital data from the INU are decoded in the RCU and the addressed relays are energized or de-energized under INU software control.

The basic switch configuration is shown in simplified form in Figure 4.1.3-2. The figure also shows a typical spacecraft interface schematic.

Command feedback provisions are also incorporated to ensure that control commands issued to the spacecraft are received through spacecraft/launch vehicle rise-off disconnects. The spacecraft is responsible for providing a feedback loop on the SC side of the interface.

4.1.3.5 Spacecraft Telemetry Interface—Lockheed Martin offers two options for the transmission of spacecraft data.

***4.1.3.5.1 RF Reradiation*—**For this option, a modification of the Atlas metal nose fairing is made to accommodate an RF reradiating antenna system to reradiate spacecraft RF telemetry and command signals.

***4.1.3.5.2 Spacecraft Serial Data Interface*—**The Atlas launch vehicle can provide transmission of two spacecraft serial data interfaces. The spacecraft data are interleaved with the launch vehicle data and serially transmitted in the pulse-code modulation (PCM) bit stream. For each data interface, as an input to the launch vehicle, the spacecraft shall provide NRZ-L coded data and a clock from dedicated drivers. Spacecraft data and clock signals shall be compliant with Electronics Industry Association (EIA) RS-422, Electrical Characteristics of Balanced Voltage Digital Interface Circuits, with a maximum data bit rate of 2 kbps. Spacecraft data will be sampled by the launch vehicle on the falling edge of the spacecraft clock signal. The clock-to-data skew shall be less than 50 microseconds and the signal and clock duty cycles shall be 50% ± 5%.

Cable from the signal driver to the launch vehicle shall have a nominal characteristic impedance of 78Ω and a maximum one-way resistance of 10Ω. Switching can be provided so that spacecraft Channels 1 and 2 can be switched in the launch vehicle data acquisition system. An example of its use might be for a single ground line to read either of the two data signals. Switching is controlled at the launch vehicle.

The data are presented as the original NRZ-L data stream in real time for those portions of prelaunch and flight for which Atlas data are received (identified by the RF link analysis report generated by Lockheed Martin for each mission). For postflight analysis, the spacecraft data can be recorded to magnetic tape.

4.1.3.6 Spacecraft Destruct Option—If required for range safety considerations, Atlas can provide a spacecraft destruct capability. A safe/arm initiator receives the destruct command from the Centaur flight termination system (FTS). The initiator ignites electrically initiated detonators, which set off a booster charge. The charge ignites a mild detonating fuse which, in turn, detonates a conically shaped explosive charge that perforates the spacecraft propulsion system.

4.2 SPACECRAFT-TO-GROUND EQUIPMENT INTERFACES

4.2.1 Spacecraft Console

Floor space is allocated on the operations level of the launch control facility (the blockhouse or Launch Service Building for SLC-36 and the Launch Service Building at SLC-3E) for installation of a spacecraft ground control console. This console is typically provided by the user, and interfaces with Lockheed Martin-provided control circuits through upper stage umbilicals to the spacecraft. The control circuits provided for spacecraft use are isolated physically and electrically from those of the launch vehicle to minimize electromagnetic interference (EMI) effects. Spacecraft that require a safe/arm function for apogee motors will also interface with the range-operated pad safety console. Lockheed Martin will provide cabling between the spacecraft blockhouse console and the pad safety console. The safe/arm command function for the spacecraft apogee motor must be inhibited by a switch contact in the pad safety console. Pad Safety will close this switch when pad evacuation has been verified.

4.2.2 Power

Several types of electrical power are available at the launch complex for spacecraft use. Commercial ac power is used for basic facility operation. Critical functions are connected to an uninterruptable power system (UPS). The dual-UPS consists of battery chargers, batteries, and a static inverter. The battery chargers are normally operated from the commercial system. However, at our Cape Canaveral Air Station (CCAS) facility one UPS may be operated on diesel generator power for major testing and launch. UPS power is available for spacecraft use in the blockhouse, launch service building, and umbilical tower.

Twenty-eight-Vdc power can be provided for spacecraft use in the blockhouse and the launch service building. The facility power supplies are operated on the UPS to provide reliable service.

4.2.3 Liquids and Gases

All chemicals used will be in compliance with the requirements restricting ozone-depleting chemicals.

Gaseous Nitrogen (GN_2)—Three pressure levels of GN_2 are available on the service tower for spacecraft use. Nominal pressure settings are 13,790 kN/m² (2,000 psi), 689.5 kN/m² (100 psi), and approximately 68.95 kN/m² (10 psi). The 10-psi system is used for purging electrical cabinets for safety and humidity control.

Gaseous Helium (GHe)—Gaseous helium is available on the mobile service towers at both CCAS and Vandenberg Air Force Base (VAFB). At CCAS, GHe at 15,169 kN/m² (2200 psi) is available. At VAFB SLC-3, GHe at 589.5 kN/m² (100 psig) and 34,475 kN/m² (5,000 psig) is available.

Liquid Nitrogen (LN_2)—LN_2 is available at the CCAS launch complex storage facility. LN_2 is used primarily by the Atlas pneumatic and LN_2 cooling systems. Small Dewars can be filled at range facilities and brought to the Atlas launch complex for spacecraft use.

At VAFB, GN_2 is provided to payload pneumatic panels at SLC-3E. GN_2 is provided at 689.5 kN/m² (100 psig), 2,758 kN/m² (400 psig), 24,822 kN/m² (3,600 psig), and 34,475 kN/m² (5,000 psig).

4.2.4 Propellant and Gas Sampling

Liquids and gases provided for spacecraft use will be sampled and analyzed by the range propellant analysis laboratory. Gases, such as helium, nitrogen, and breathing air, and liquids such as hypergolic fuels and oxidizers, water, solvents, and hypergolic decontamination fluids may be analyzed to verify that they conform to the required specification.

4.2.5 Payload Transport Canister

Lockheed Martin offers the customer the option of using our payload transport canister (PTC) for spacecraft transport from the PPF to the HPF. The PTC has a self-contained GN_2 purge system used to maintain a positive GN_2 pressure in the canister during transport. Figure 4.2.5-1 shows the payload canister.

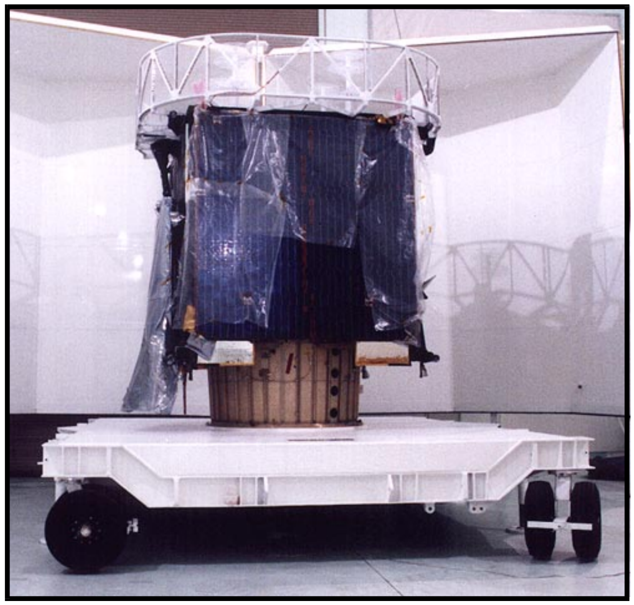


Figure 4.2.5-1 Payload Transport Canister

4.2.6 Work Platforms

The launch complex service tower provides work decks approximately 10-ft apart in the spacecraft area. Portable workstands will be provided to meet spacecraft mission requirements where the fixed work decks do not suffice. Access can be provided inside the encapsulated nose fairing. The access requirements will be developed during the planning stage of each mission.

4.3 RANGE AND SYSTEM SAFETY INTERFACES

4.3.1 Requirements

To launch from either CCAS or VAFB, launch vehicle and spacecraft design and ground operations must meet applicable launch site Range Safety requirements, U.S. Air Force instructions concerning explosives safety, occupational safety and health, and U.S. consensus safety standards.

In addition, when using spacecraft processing facilities operated by Astrotech International Corporation, the National Aeronautics and Space Administration (NASA), or the U.S. Air Force, compliance with their facility safety policies is also required.

Over the years, the CCAS and VAFB Range Safety organizations have regularly modified their safety requirements documents. The single safety document for both CCAS and VAFB (Eastern/Western Range Regulation [EWR] 127-1) was updated on 31 October 1997. Earlier versions of this Range Safety document could still apply to a given spacecraft or mission depending on when the spacecraft was designed and built and which Range is used. Earlier versions include the following:

1) Eastern Range;
    a) Eastern Range Regulation (ERR) 127-1, June 1993;
    b) Eastern Space and Missile Center Regulation (ESMCR) 127-1, July 1984.
2) Western Range;
    a) Western Range Regulation (WRR) 127-1, June 1993;
    b) Western Space and Missile Center Regulation (WSMCR) 127-1, 15 December 1989;
    c) WSMCR 127-1, 15 May 1985.
3) Both Ranges;
    a) Eastern and Western Range (EWR) 127-1, 31 March 1995;

Compliance documents are agreed upon with the Range, Lockheed Martin, and the spacecraft contractor at the outset of the mission integration process. Other documents that may apply to the launch site safety interface are as follows:

1) ESMCR 160-1, Radiation Protection Program;
2) Air Force Manual (AFM) 91-201, Explosives Safety Standard;
3) Air Force Regulation (AFR) 127-12, Air Force Occupational Safety, Fire Prevention, and Health Program;
4) MIL-STD 1522A, Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems;
5) MIL-STD 1576, Electroexplosive Subsystem Safety Requirements and Test Methods for Space Systems;
6) Atlas Launch Site Safety Manual.

At the start of mission integration (or soon after), the Range Safety documents applicable to spacecraft design and ground processing operations will be tailored with the Range. Tailoring defines the safety criteria specifically interpreted for application to a specific spacecraft.

4.3.2 Applicability

A major component of the launch site Range Safety regulations (e.g., EWR 127-1) addresses the spacecraft design and operational requirements that must be met to obtain Range Safety approval. For all spacecraft and missions, compliance with the 127-1 regulations will be specifically addressed in safety submittals.

Lockheed Martin System Safety engineers will evaluate, analyze, and provide guidance for spacecraft design to support Range Safety approval. Lockheed Martin will also assess the design and processing requirements applicable to mission-peculiar launch vehicle/spacecraft interfaces. Should areas of noncompliance be determined, Lockheed Martin will evaluate the situation and provide advice for resolution of the specific noncompliance while still meeting the intent of the safety requirement. For commercial programs, Lockheed Martin will act as the spacecraft contractor's agent for all interface activities with the launch site Range Safety office.

4.3.3 Safety Interface

The typical process used by the Atlas program to safety certify spacecraft at the range is illustrated in Figure 4.3.3-1. The figure illustrates the documents or tasks required of the spacecraft contractor. For all mission integration efforts, Lockheed Martin will provide a qualified system safety engineer to help the spacecraft contractor through the range approval process. All Range approvals will be obtained by the Atlas program-regardless of the data on which they are based. Approval to fly on the range is based on documentation of the spacecraft design that verifies the design meets the intent of the applicable Range Safety requirements document. The Atlas safety interface and approval process is summarized in the following paragraphs which describe the various safety submittals or tasks.

Mission Orientation—Lockheed Martin and the spacecraft contractor will introduce a new system or mission to the Range during a mission orientation meeting at the Range soon after contract award. Figure 4.3.3-1 shows the basic elements of the orientation. During the orientation meeting, mission-peculiar design and operational issues are reviewed so that they can be resolved early in the integration phase of the program. The mission orientation also provides the opportunity to determine which Range Safety requirements will be imposed on spacecraft design and operations and to develop program specific schedules.

Propellant Leak Contingency Plan (PLCP) Package—PLCP data is developed by the spacecraft contractor and used by the Atlas program to develop the preliminary PLCP for the spacecraft. The PLCP provides a plan for emergency offload of spacecraft propellants. The PLCP will require the development of detailed procedures by the spacecraft contractor to perform the offload.

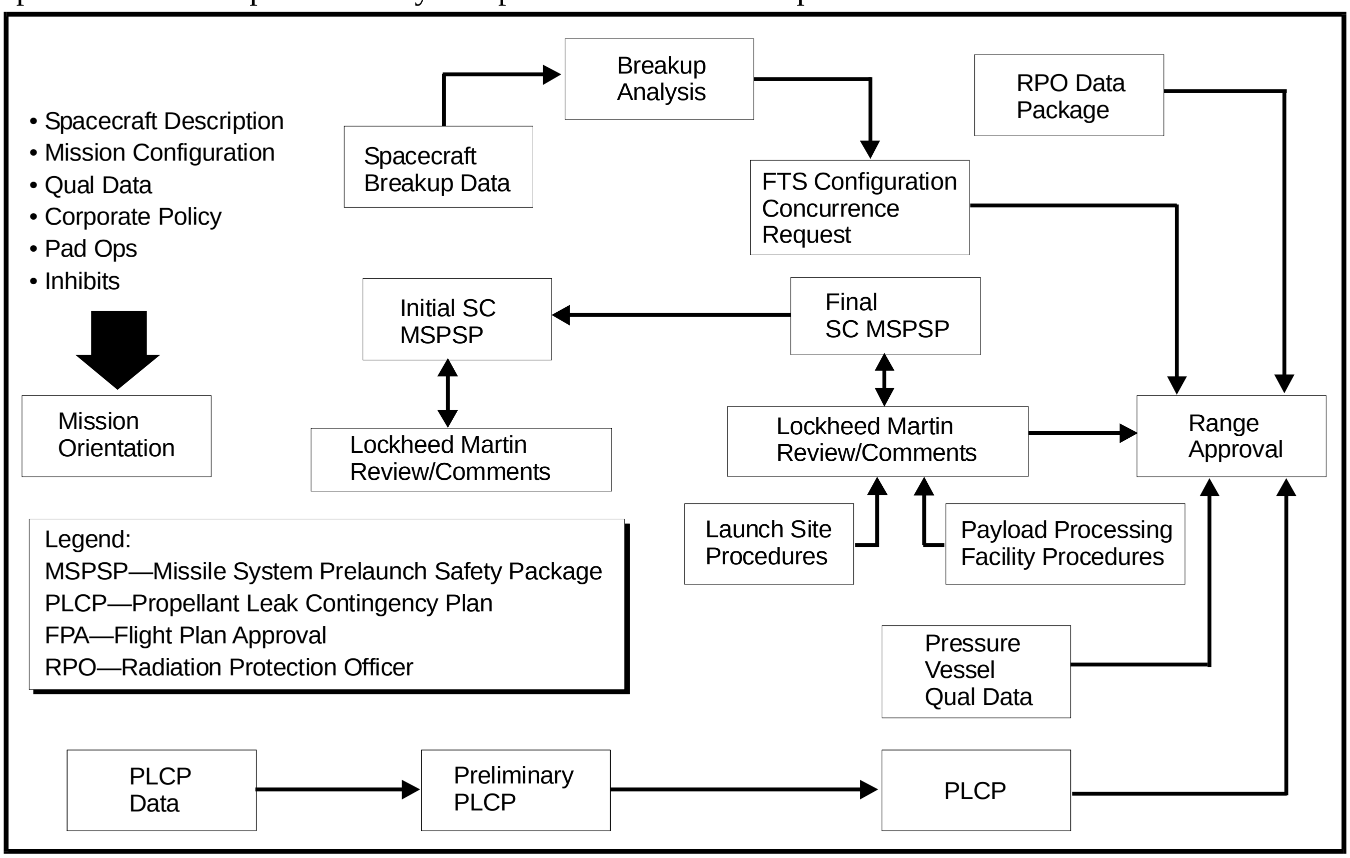


Figure 4.3.3-1 System and Range Safety Interfaces and Requirements

Spacecraft Missile System Prelaunch Safety Package (MSPSP)—The spacecraft contractor develops the initial spacecraft MSPSP to describe the spacecraft, hazards associated with the spacecraft (e.g., pressure systems, ordnance control systems, toxic hazards, access requirements, processing hazards, etc.), and the means in which each hazard is eliminated or controlled to an acceptable level. Range Safety regulations provide details on the format and contents of the MSPSP.

The initial MSPSP is typically submitted to Lockheed Martin about 9 months before initial launch capability (ILC). The Atlas program will review and the document and forward it with comments to the range for review and comment. The Atlas program will pass along comments to the spacecraft contractor for incorporation into the final spacecraft MSPSP. The final MSPSP is typically submitted to the Atlas program approximately 4 months before ILC.

Mission-Unique MSPSP—Lockheed Martin will combine data from the spacecraft MSPSP with data from the Atlas booster MSPSP and the ICD to perform and document a safety assessment of the booster-spacecraft interface. Results will be delivered to the Range as the mission-unique MSPSP, both in preliminary and final documents.

Range Safety Data Package—Lockheed Martin will develop a preliminary range safety data package that provides the basic spacecraft configuration, the preliminary flight profile, and the time of launch. Lockheed Martin will forward this package to the Range with a preliminary flight plan approval request to gain preliminary approval to fly the mission on the Range, as designed. About 60 days before ILC, the Atlas program will provide the final Range Safety data package to the Range with a request for final flight plan approval (FPA). Final FPA is usually received from the Range about 7 days before ILC.

Spacecraft Breakup Data—The spacecraft contractor will provide spacecraft breakup data to Lockheed Martin for the Range Safety Data Package. Lockheed Martin will use this breakup data to perform a breakup analysis on the spacecraft under expected mission conditions.

Flight Termination System (FTS) Configuration Concurrence—Based on the breakup analysis, the Atlas Program will submit an FTS Configuration Concurrence Request to the Range. The purpose of this concurrence request is to obtain an agreement with the Range regarding requirements for a spacecraft destruct capability. With typical communications satellites, Lockheed Martin usually pursues FTS concurrence without a spacecraft FTS destruct system, since there is no appreciable additional public safety hazard.

Pressure Vessel Qualification Data—The spacecraft contractor will provide pressure vessel qualification data, through the Atlas program, to the Range for review and acceptance.

Radiation Protection Officer (RPO) Data—The spacecraft contractor will submit data on the type and intensity of RF radiation that the spacecraft will transmit during ground testings, processing, and launch at the Range. Lockheed Martin will forward this data to the RPO for review and approval of RF-related operations to be performed at the launch site.

Spacecraft Processing Procedures—The spacecraft contractor will provide onsite processing procedures to the operator of the payload processing facility (e.g., Astrotech) through the Atlas program for review and approval. The procedures must comply with the processing facility safety policy.

5.0 MISSION INTEGRATION AND MANAGEMENT

5.1 INTEGRATION MANAGEMENT

Clear communication between spacecraft and launch vehicle contractors is vital to Mission Success® accomplishment. Procedures and interfaces have been established to delineate areas of responsibility and authority.

The mission integration and management process defined in this section has been successfully used on all recent commercial and government Atlas missions. These identified processes and interfaces have enabled mission integration in as little as 6 months, with 18 months being a more typical schedule.

As an additional information resource, Appendix C of this document details the preferred approach and format for the exchange of data required for mission integration. When necessary, deviations from these specified practices can be accommodated.

5.1.1 Launch Vehicle Responsibilities

Lockheed Martin is responsible for Atlas design, integration, checkout, and launch. This work is performed primarily at the Lockheed Martin Astronautics (LMA) Waterton plant in Denver, Colorado (Fig. 5.1.1-1). Additional facilities in San Diego, California, and Harlingen, Texas, support hardware manufactures with major tank welding and structural assembly. Major subcontractors include Pratt & Whitney (upper-stage main engines and Atlas III booster engines), Honeywell (inertial navigation unit [INU]), Rockwell International-Rocketdyne (Atlas IIA/IIAS booster engines), and Thiokol (solid rocket boosters [SRB]). As the spacecraft-to-launch vehicle integrating contractor, Lockheed Martin is responsible for payload integration (e.g., electrical, mechanical, environmental, and electromagnetic compatibilities; guidance system integration; mission analysis; software design; range safety documentation and support; launch site processing; and coordination). Lockheed Martin produces all launch vehicle-related software for Atlas launches and is responsible for the launch vehicle ascent trajectory, data acquisition, performance analysis, targeting, guidance analysis, and range safety analysis.

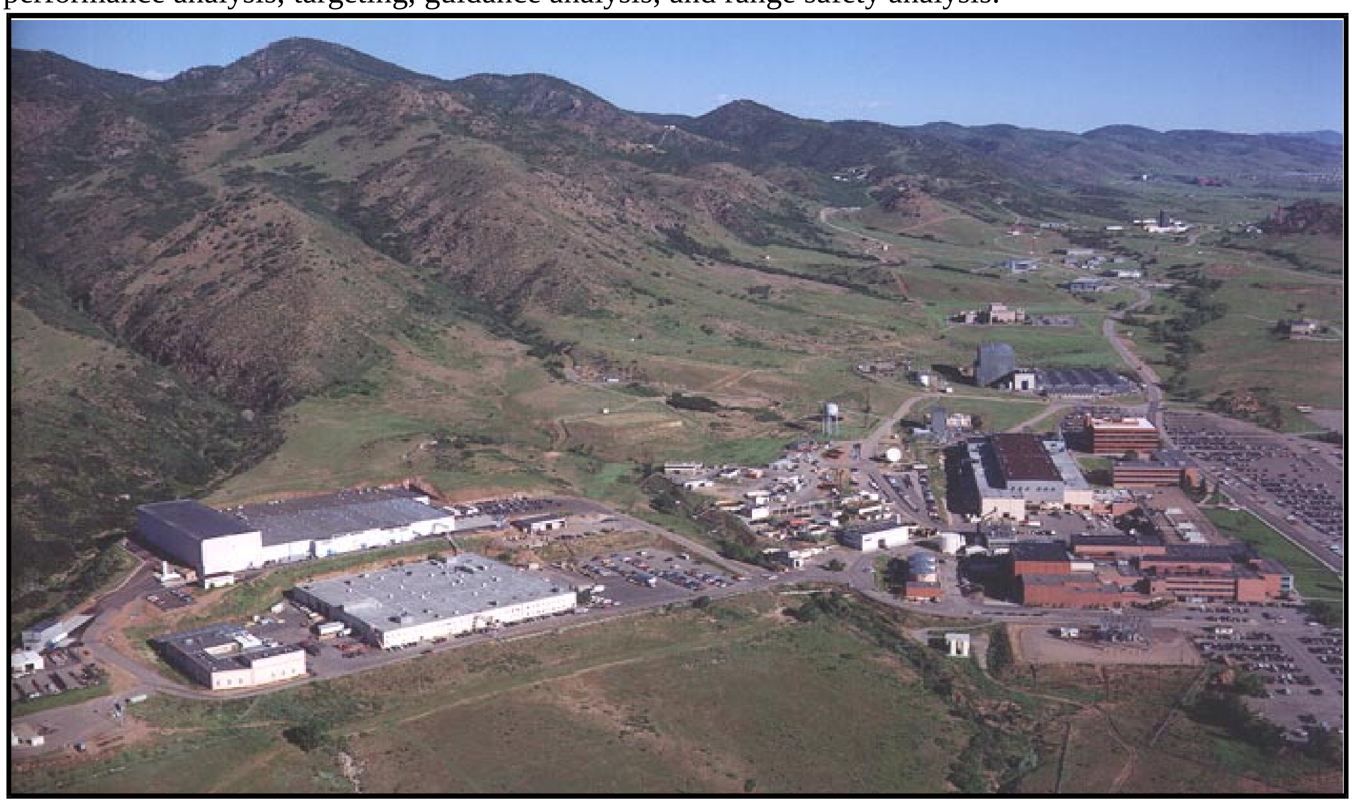


Figure 5.1.1-1 Lockheed Martin Facilities in Denver, Colorado

5.1.2 Spacecraft Responsibilities

Each spacecraft mission has unique requirements. Interested Atlas users are encouraged to discuss their particular needs with Lockheed Martin. Appendix C, Spacecraft Data Requirements, can be used as a guide to initiating dialog. Shaded items in Appendix C should be used as the basis for the first face-to-face meeting between Lockheed Martin and the potential user to assist in determining spacecraft/launch vehicle compatibility.

Customers are encouraged to contact Lockheed Martin to verify the latest launch information, including:

1) Hardware status and plans;
2) Launch and launch complex schedules;
3) Hardware production schedule and costs.

5.1.3 Integration Organization

For all Atlas missions, Lockheed Martin management assigns a mission manager. The mission manager is responsible for overall management of the particular customer activities at Lockheed Martin facilities and the launch site. This person is the principal interface with the customer for all technical and launch vehicle/satellite interface and integration matters.

The LMA Atlas program assigns a program manager and mission integrator for each Atlas mission. The program manager coordinates Astronautics' resources to ensure timely delivery of Atlas vehicle hardware through the production process. The mission integrator is responsible for coordinating the engineering integration of the spacecraft with the Atlas launch vehicle.

The Commercial Launch Services (CLS) organization is aligned to provide low-risk launch services. CLS responds to a customer order by arranging services from several organizations. LMA is the subcontractor responsible for Atlas production, launch, and mission-peculiar integration processing. CLS has contracts with the U.S. government for use of Atlas launch complexes and payload integration facilities and with the U.S. Air Force (USAF) for range and launch site services. Astrotech spacecraft payload integration facilities are contracted for most Atlas launches, with additional support from other government facilities when required.

To provide maximum efficiency in managing the many launch site operations, a launch site-based launch operations manager is assigned to each mission. The launch operations manager represents the mission manager during development, integration, and installation of all spacecraft-peculiar items at the launch site and upon arrival of the spacecraft. This program organization concept has been used successfully for all major Atlas programs.

Lockheed Martin's approach to integration management is through establishment of a formal interface control document (ICD) agreement and formal configuration control after ICD signature. Existing ICDs may be adapted to reduce development time. Coordination of the tasks required to develop and maintain the ICD is accomplished through management and technical working groups.

5.1.4 Integration Program Reviews

During the integration process, reviews focus management attention on significant milestone during the launch system design and launch preparation process. As with the working group meetings, these reviews can be tailored to user requirements; however, for a first-of-a-kind launch, they may include a preliminary and critical mission-peculiar design review (MPDR) and a launch readiness review (LRR). For typical communications satellite launches, only one design review is required.

In recognition of a valuable "check-and-balance" function provided in the past by the National Aeronautics and Space Administration (NASA), CLS has established an independent technical oversight function. This group of senior personnel participates in technical and mission readiness reviews and in formal review processes: engineering review board (ERB), preliminary design review (PDR), critical design review (CDR), and final launch readiness.

5.1.4.1 Mission-Peculiar Design Reviews (MPDR)—As Lockheed Martin has significant experience with most communications satellite contractors, we have found that one MPDR can successfully meet the goals and requirements that were, in the past, met by both the PDR and CDR. Approximately midway through the mission integration process, the MPDR is conducted to ensure that customer requirements have been correctly and completely identified and that integration analyses and designs meet these requirements. Lockheed Martin prepares and presents the reviews with participation from the spacecraft contractor, launch services customer, and launch vehicle management.

For missions in which spacecraft require unique interfaces with the launch vehicle or the launch vehicle mission is unlike other previously flown Atlas missions, two design reviews may be proposed by Lockheed Martin after discussions with the launch services customer and spacecraft contractor.

5.1.4.2 Launch Readiness Review (LRR)—This review, conducted approximately 2 days before launch, provides a final prelaunch assessment of the integrated spacecraft/launch vehicle system and launch facility readiness. The LRR provides the forum for final assessment of all launch system preparations and the contractors' individual certifications of launch readiness.

5.1.5 Integration Control Documentation

5.1.5.1 Program Master Schedule—This top-level schedule is prepared by Lockheed Martin and monitored by the Mission Integration Team. It maintains visibility and control of all major program milestone requirements, including working group meetings, major integrated reviews, design and analysis requirements, and major launch operations tests. It is developed from the tasks and schedule requirements identified during the initial integration meetings and is used by all participating organizations and working groups to develop and update subtier schedules.

5.1.5.2 Interface Requirements Documents (IRD)—The customer creates the IRD to define technical and functional requirements imposed by the spacecraft on the launch vehicle system. The document contains applicable spacecraft data identified in Appendix C. Information typically includes:

1) Mission Requirements—Including orbit parameters, launch window parameters, separation functions, and any special trajectory requirements, such as thermal maneuvers and separation over a telemetry and tracking ground station;
2) Spacecraft Characteristics—Including physical envelope, mass properties, dynamic characteristics, contamination requirements, acoustic and shock requirements, thermal requirements, and any special safety issues;
3) Mechanical and Electrical Interfaces—Including spacecraft mounting constraints, spacecraft access requirements, umbilical power, command and telemetry, electrical bonding, and electromagnetic compatibility (EMC) requirements;
4) Mechanical and Electrical Requirements for Ground Equipment and Facilities—Including spacecraft handling equipment, checkout and support services, prelaunch and launch environmental requirements, spacecraft gases and propellants, spacecraft radio frequency (RF) power, and monitor and control requirements;
5) Test Operations—Including spacecraft integrated testing, countdown operations, and checkout and launch support.

5.1.5.3 Interface Control Document (ICD)—This document defines spacecraft-to-launch vehicle and launch complex interfaces. All mission-peculiar requirements are documented in the ICD. The ICD is prepared by Lockheed Martin for the Interface Control Working Group (ICWG) and is under configuration control after formal signoff. The document contains the appropriate technical and functional requirements specified in the IRD and any additional requirements developed during the integration process. The ICD supersedes the IRD and is approved with signature by both Atlas program management and the launch service customer.

5.2 MISSION INTEGRATION ANALYSIS

In support of a given mission, Lockheed Martin will perform analyses summarized in Table 5.2-1. This table indicates the specific output of the analyses to be performed, required spacecraft data, the timing during the integration cycle that the analysis is completed, and the application of the analyses to first-of-a-kind and follow-on missions. In this context a follow-on mission is an exact copy of a previous mission, with no change to functional requirements or physical interfaces. Table 5.2-1 represents the standard integration analyses. For many missions, Lockheed Martin uses generic versions of these analyses and may not be required to perform a mission-peculiar version.

Table 5.2-1 Summary of Atlas Mission Integration Analyses

Analysis	SC Data	Analysis Products	No. of Cycles	Schedule	First-of-a-Kind	Follow-On
1) Coupled Loads	SC Dynamic Math Model	• SC Loads • Dynamic Loss of Clearance • Launch Availability • PLF Jettison Evaluation	2	Model Delivery + 4 mo	X	
2) Integrated Thermal	SC Geometric & Thermal Math Models & Power Dissipation Profile	• SC Component Temperature • Prelaunch Gas Cond & Set Points	1	Model Delivery + 6 mo	X	
3) PLF Venting	SC Venting Volume	• Press Profiles • Depressurization Rates	1	SC Data + 2 mo	X	
4) Critical Clearance	SC Geometric Model; SC Dynamic Model	• SC-to-PLF Loss of Clearance (Dynamic + Static)	2	SC Model Del + 4 mo	X	
5) PLF Jettison Clearance	SC Geometric Model	• P/L Clearance Margin During PLF Jettison Event	1	Design Review	X	
6) SC Separation & Clearance	SC Mass Properties	• SC Sep Clearance • SC Sep Att & Rate & Spin-Up Verification	1	Design Review	X	
7) Postseparation Clearance		• LV-SC Sep History	1	Design Review	X	
8) Pyro Shock	SC Interface Definition	• SC Shock Environment	1	Design Review	X	
9) Acoustics	SC Geometry Fill Factors	• SC Acoustics Environment	1	Design Review	X	
10) EMI/EMC	• SC Radiated Emissions Curve • SCRadiated Susceptibility Curve • SC Rec Op & Demise Thresholds • SC Diplexer Rejection	• Confirmation of Margins • Integrated EMI/EMC Analysis	1	Design Review	X	
11) Contamination	SC Contamination Limits for Sensitive, Critical, Vertical & Horizontal Surfaces	• Contamination Analysis • Contamination Assessment	1	Design Review	X	

Table 5.2-1 (concl)

Analysis	SC Data	Analysis Products	No. of Cycles	Schedule	First-of-a-Kind	Follow-On
12) RF Link Compatibility & Telemetry Cov (Airborne)	SC Transmitter & Receiver Characteristics	• Link Margins • ARIA Positioning Requirements	1	Design Review	X	
13) RF Link Compatibility & Telemetry Cov (Ground)	SC Transmitter & Receiver Characteristics	• Link Margins • Identify Required Hardware	1	Design Review	X	
14) Performance	SC Mass & Mission Requirements Definition	• Config, Perf & Wt Status Report • Performance Margin	3	ATP + 3-mo Dsn Rev Final Targeting	X	X
15) Stability		• Control System Margins • RCS Use	1	L-2 mo	X	
16) Mass Properties		• Mass Properties of LV		Coincident with Perf Reports	X	X
17) Trajectory Analysis	SC Mass & Mission Requirements Definition	• LV Ref Trajectory • Performance Margin	1	Design Review	X	X
18) Guidance Analysis	Mission Requirements	• Guidance S/W Algorithms • Mission Targeting Capability & Accuracies	1	Design Review	X	
19) Injection Accuracy	Mission Requirements	• LV System Orbit Injection Accuracy	1	Design Review	X	
20) Launch Window	Window Definition	• Window Durations	1	Design Review	X	X
21) Wind Placard	SC Mass Properties	• LV Ground & Flight Winds Restrictions	1	Design Review	X	
22) Range Safety	SC Breakup Data & Propulsion Characteristics	• Trajectory Data & Tapes for Range Approval	2	L-1 yr, Prelim L-7 wk, Final	X	
23) SC-to-LV FMEA	SC Electrical Schematics	• Design Weaknesses or Oversight • Reliability Test Rqmts • Critical Components	1	Design Review	X	
24) Electrical Compatibility	Electrical Interface Requirements	• End-to-End Circuit Analysis	1	Design Review	X	
25) Postflight		• Flight Eval of Mission Data, LV Perf & Environment	1	L + 60/days	X	X
26) Destruct Sys		• Confirmation of Mtg Range Safety Rqmts	1	Design Review	X	
27) Mission Targeting	Orbit Requirements	• Flight Constants Tapes • Firing Tables	1	L-1 mo	X	X
28) Flight Software	Mission Requirements	• FCS Software	1	L-3 weeks	X	X

Figure 5.2-1 is our preliminary schedule for a typical Atlas mission. The full-scale integration process begins at approximately L-12 months.

The following paragraphs describe the integration analyses.

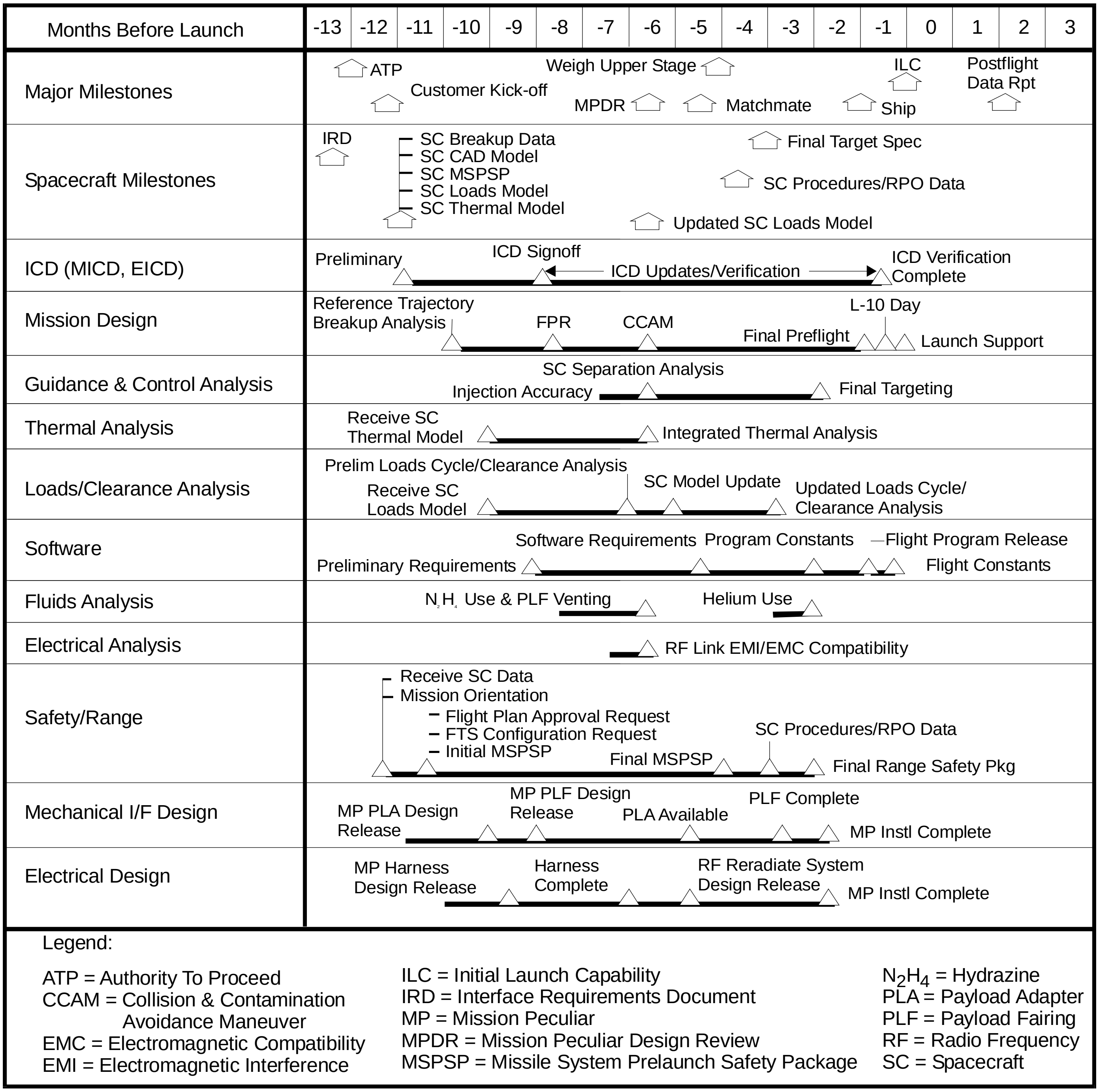


Figure 5.2-1 Twelve-month Generic Mission Integration Schedule

5.2.1 Coupled Loads Analysis (CLA)

During the course of the Atlas program, a set of test-correlated three-dimensional (3-D) analytical launch vehicle models is generated for use in the mission-peculiar dynamic CLA. Lockheed Martin will perform mission-peculiar analyses where launch vehicle and payload parameters may be affected, as discussed in Section 3.2.1.2. Included are:

1) Spacecraft loading and loss of clearance evaluation for the critical flight events of liftoff, transonic buffet, gust, SRB air-lit start, SRB burnout and SRB jettison, booster engine cutoff (BECO), booster package jettison (BPJ), and main engine cutoff (MECO);
2) Flight wind launch availability assessment;
3) Payload fairing (PLF) jettison evaluation applicable to a mission spacecraft.

Analysis of all events uses state-of-the-art finite element models of the booster coupled with a customer-supplied dynamic math model of the spacecraft. Appendix C of this document provides a description of the type and format of the dynamic model.

5.2.2 Integrated Thermal Analysis (ITA)—Preflight and Flight

Lockheed Martin performs an integrated launch vehicle-spacecraft analysis of the thermal environments imposed on the spacecraft under prelaunch conditions and for flight mission phases up to spacecraft separation. The integrated thermal analysis (ITA) is performed with customer-supplied spacecraft geometric and thermal math models and a detailed spacecraft power dissipation timeline. The results are provided to the customer for evaluation and can be used to design thermal interfaces and mission operations to maintain predicted spacecraft temperatures within allowable limits.

In addition to the ITA, Lockheed Martin performs PLF aeroheating analyses, PLF gas conditioning analyses, and free molecular heating analyses to verify compliance with customer ICD thermal requirements and thermal requirements derived from the ITA.

Thermal analyses ensure that vehicle design is compatible with and has adequate margins over proposed spacecraft thermal constraints. Analyses include assessment of vehicle aeroheating, PLF surface temperature ranges, maximum and minimum prelaunch air conditioning temperatures and velocities, and spacecraft-to-Centaur interface temperature ranges.

Prelaunch payload gas conditioning analyses verify that the thermal environment is compatible with the defined spacecraft. A worst-case analysis is performed (using the maximum air conditioning supply rate) to determine flow conditions around the air conditioning inlet and at the minimum flow area location between the spacecraft and fairing. The inlet duct configuration can be designed to minimize air impingement velocity.

The thermal analysis predicts air conditioning gas temperature variations along the spacecraft during prelaunch operations. The analysis is performed by combining the extremes of air conditioning temperature and flowrate conditions with spacecraft power dissipation levels identified for each mission.

PLF internal surface temperature ranges are predicted by analyzing flight aerodynamic heating using our thermal analyzer computer program.

PLF jettison time is selected to meet the spacecraft free molecular heating constraint. Atlas missions ensure a benign spacecraft thermal environment by selecting jettison time based on a flight program calculation of 3-sigma maximum qV during flight.

5.2.3 PLF Venting Analysis (Ascent Phase)

A venting analysis is performed on the PLF to determine mission-peculiar pressure profiles in the payload compartment during launch vehicle ascent. Existing models that have been validated with flight data are used for this analysis. The analysis incorporates the customer-provided spacecraft venting configuration and any mission-specific PLF requirements (e.g., thermal shields). Analysis outputs provided to the customer include PLF pressure profiles and depressurization rates as a function of flight time.

5.2.4 Critical Clearance Analysis (Loss of Clearance)

A spacecraft-to-launch vehicle loss of clearance analysis is performed with the coupled dynamic loads analysis (Sect. 5.2.1). The clearance loss due to dynamic response of the spacecraft and launch vehicle during major events is combined with static deflections, manufacturing stackup tolerances, and misalignments. These values are compared against the allowable payload envelope.

5.2.5 PLF Jettison and Loss of Clearance Analyses

Verification of payload clearance during PLF jettison is performed using the effects of thermal preload, disconnect forces, shear pin forces, actuator forces, and dynamic response in a fully 3-D non-linear analysis.

5.2.6 Spacecraft Separation Analysis

Extensive Monte Carlo analysis of the preseparation dynamics, using a 3-degree-of-freedom (DOF) simulation of the vehicle and attitude control system, demonstrates compliance with all spacecraft attitude pointing and angular rate and spin rate requirements under nominal and 3-sigma dispersions.

A two-body 6-DOF Monte Carlo simulation of the Centaur/spacecraft separation event is performed using finalized spacecraft mass properties to verify that Centaur will not recontact the spacecraft after separation system release. This analysis demonstrates the minimum relative separation velocity, ensuring that adequate separation distance is achieved before initiating any postseparation Centaur maneuvers.

5.2.7 Spacecraft Postseparation Clearance Analysis

After the spacecraft has separated from the Centaur vehicle, Centaur performs a collision and contamination avoidance maneuver (CCAM). The Centaur reaction control system uses 12 hydrazine thrusters rated at 27-N (6-lbf) thrust each. Four thrusters are dedicated to propellant settling (axial) control and eight are allocated to roll, pitch, and yaw control. The thrusters are located on the aft bulkhead of the liquid oxygen (LO_2) tank inboard of the 3.0-m (10-ft) tank diameter. Before spacecraft separation, this location precludes a direct line of impingement to the spacecraft. In addition, the thrust directions are either 90° or 180° away from the spacecraft.

The CCAM is designed to positively preclude physical recontact with the spacecraft and eliminate the possibility of significant impingement of Centaur effluents on the spacecraft. The CCAM consists of a series of three attitude maneuvers, combined with axial thrust from the reaction control system (RCS) settling motors and blowdown of the Centaur tanks. Shortly after spacecraft separation, the Centaur turns 4° from the separation attitude. The Centaur then typically turns 50° and activates the settling motors to impart a ΔV to move the Centaur a significant distance from the spacecraft. This maneuver minimizes any plume flux to the spacecraft. The final maneuver turns the Centaur normal to the flight plane. In this attitude the tank blowdown is executed at approximately 1.8 km (1 nmi) from the spacecraft. This CCAM sequence, which is operational on all Atlas missions, ensures adequate in-plane and out-of-plane separation between the Centaur and the spacecraft and minimizes the RCS motor plume flux at the spacecraft.

5.2.8 Pyroshock Analysis

The spacecraft pyroshock environment is maximum for the spacecraft separation event. PLF separation and Atlas/Centaur separation are also significant events, but the distances of the shock sources from the spacecraft/Centaur interface make them less severe for the spacecraft than activation of the PSS. Verification of this environment has been accomplished by ground PSS testing of our existing separation systems.

5.2.9 Acoustic Analysis

Analysis of the acoustic environment of the payload compartment includes effects of noise reduction of the PLF and payload fill factors. Verification includes flight measurements taken from several Atlas/Centaur flights and ground acoustic testing of representative PLF/payload configurations.

5.2.10 Electromagnetic Interference (EMI)/Electromagnetic Compatibility (EMC) Analysis

Lockheed Martin maintains an EMI/EMC plan to ensure compatibility between all avionics equipment. This plan covers requirements for bonding, lightning protection, wire routing and shielding, and procedures. Lockheed Martin analyzes intentional and unintentional RF sources to confirm 6-dB margins with respect to all general EMI/EMC requirements. In addition, an electro-explosive device (EED) RF susceptibility analysis is performed to range requirements. This analysis is intended to confirm 20-dB margin with respect to the EED no-fire power. Comprehensive reports are published describing the requirements and results of these analyses.

5.2.11 Contamination Analysis

Lockheed Martin provides new spacecraft with an assessment of contamination contributions from Atlas launch vehicle sources, as required. Starting from PLF encapsulation of the spacecraft through CCAM, contamination sources are identified and analyzed. This assessment provides a first-order contamination analysis to allow the spacecraft user to determine final onorbit contamination budgets. A more detailed mission-peculiar analysis can be provided to the spacecraft after identification in the ICD.

Lockheed Martin also implements a contamination control plan that ensures hardware cleanliness from the manufacturing phase through launch operations. Material control, ground support hardware cleanliness, and contamination monitoring are specified and implemented through the contamination control plan. A mission-unique appendix will supplement the contamination control plan if the mission-specific requirements identified in the ICD are more stringent than those baselined in the control plan.

5.2.12 RF Link Compatibility and Telemetry Coverage Analysis (Airborne)

Lockheed Martin conducts an airborne link analysis on all RF links between ground stations and the Atlas/Centaur vehicle. The systems analyzed are: the S-band telemetry system, the active C-band vehicle tracking system, and the flight termination system. Lockheed Martin uses a program that takes into account airborne and ground station equipment characteristics, vehicle position, and attitude. This analysis includes maximizing link margins with the Tracking and Data Relay Satellite System (TDRSS) when the TDRSS-compatible transmitter is used on Centaur. A comprehensive report is published describing the link requirements and results.

Lockheed Martin conducts an RF compatibility analysis between all airborne RF transmitters and receivers to ensure proper function of the integrated system. The transmit frequencies and their harmonics are analyzed for potential interference to each active receiver. In addition, strong site sources, such as C-band radar, are also analyzed. The spacecraft contractor provides details of the active transmitters and receivers for this analysis. A comprehensive report is published describing requirements and results of the analysis.

5.2.13 RF Link Compatibility and Telemetry Coverage Analysis (Ground)

Lockheed Martin conducts a ground link analysis on spacecraft RF systems to ensure that a positive link exists between the spacecraft and the spacecraft checkout equipment to checkout the spacecraft telemetry and command system. Information about spacecraft requirements is contained in the ICD. A technical report is published describing link requirements and implementation of the link system.

5.2.14 Performance Analysis

The capability of Atlas to place the spacecraft into the required orbit(s) is evaluated through our trajectory simulation program, TRAJEX, which is briefly discussed in Section 5.2.17. Vehicle performance capability is provided through our configuration, performance, and weight status report. This report is tailored to accommodate the needs of specific missions.

The status report shows the current launch vehicle propellant margin and flight performance reserve (FPR) for the given mission and spacecraft mass. A comprehensive listing of the vehicle configuration status, mission-peculiar groundrules and inputs, and vehicle masses for performance analysis is included. The report also provides the more commonly used payload partial derivatives (the tradeoff coefficients) with respect to the major vehicle variables (e.g., stage inert weights, propellant loads, stage propulsion parameters). The detailed trajectory simulation used for the performance assessment is provided as an appendix to the report.

5.2.15 Stability and Control Analysis

Linear stability analysis, primarily frequency response and root-locus techniques, and nonlinear time-varying 6-DOF simulation are performed to determine the Atlas and Centaur autopilot configurations; establish gain and filter requirements for satisfactory rigid body, slosh, and elastic mode stability margins; verify vehicle/launch stand clearances; and demonstrate Centaur RCS maneuver and attitude hold capabilities. Uncertainties affecting control system stability and performance are evaluated through a rigorous stability dispersion analysis. Tolerances are applied to vehicle and environmental parameters and analyzed using frequency response methods, ensuring that the Atlas autopilot maintains robust stability throughout the defined mission. Correlation of simulation results with previous postflight data has confirmed the adequacy of these techniques.

5.2.16 Mass Properties Analysis

Lockheed Martin performs mass properties analysis, reporting, and verification to support performance evaluation, structural loads analysis, control system software configuration development, ground operations planning, airborne shipping requirements, and customer reporting requirements.

5.2.17 Trajectory Analysis and Design

The Lockheed Martin trajectory design process ensures that all spacecraft, launch vehicle, and range-imposed environmental and operational constraints are met during flight, while simultaneously providing performance-efficient flight designs. This process typically provides propellant margin (PM) above the required performance reserves.

The trajectory design and simulation process provides the vehicle performance capability for the mission. It provides the basis, by the simulation of nonnominal vehicle and environmental parameters, for analyses of FPR and injection accuracy. Telemetry coverage assessment, RF link margins, PLF venting, and in-flight thermal analyses also rely on the reference mission design. The trajectory design is documented in the status report (Sect. 5.2.14). Detailed insight into the tradeoffs used for the trajectory design is provided in the trajectory design report.

Our principal analysis tool for the trajectory design process is the TRAJEX (trajectory executive) simulation program. This flexible, accurate, numeric integration simulation has a 25-year history of operational use in our trajectory design process. The detailed propulsion, mass properties, aerodynamic, and steering control modeling is embedded in this program. TRAJEX has oblate Earth and gravity capability, selectable atmospheric models, and other selectable routines, such as Sun position and tracker locations, to obtain output for these areas when they are of interest.

TRAJEX interfaces directly with actual flight computer software. This feature bypasses the need to have engineering equivalents of the flight software. Another powerful feature of TRAJEX is compatibility with 6-DOF modeling of the vehicle, which will facilitate key dynamic analyses for our vehicle family. Other features include significant flexibility in variables used for optimization, output, and simulation interrupts.

5.2.18 Guidance Analysis

Analyses are performed to demonstrate that spacecraft guidance and navigation requirements are satisfied. Analyses include targeting, standard vehicle dispersions, extreme vehicle dispersions, and guidance accuracy. The targeting analysis verifies that the guidance program achieves all mission requirements across launch windows throughout the launch opportunity. Standard vehicle dispersion analysis demonstrates that guidance algorithms are insensitive to 3-sigma vehicle dispersions by showing that the guidance program compensates for these dispersions while minimizing orbit insertion errors. Extreme vehicle dispersions (e.g., 10 sigma) and failure modes are selected to stress the guidance program and demonstrate that the guidance software capabilities far exceed the vehicle capabilities.

5.2.19 Injection Accuracy Analysis

The guidance accuracy analysis combines vehicle dispersions and guidance hardware and software error models to evaluate total guidance system injection accuracy. Hardware errors model the off-nominal effects of guidance system gyros and accelerometers. Software errors include INU computation errors and vehicle dispersion effects. Positive and negative dispersions of more than 30 independent vehicle and atmospheric parameters that perturb Atlas and Centaur performance are simulated. This accuracy analysis includes noise and twist and sway effects on guidance system alignment during gyro compassing and the covariance error analysis of the guidance hardware.

5.2.20 Launch Window Analysis

Launch window analyses are performed to ensure the proper launch vehicle and spacecraft launch constraints are understood. The Atlas/Centaur launch vehicle can accommodate up to a 2-hour launch window, any time of day, any day of the year. Customers are requested to provide the opening and closing times for the maximum launch window the spacecraft is capable of supporting. If the maximum launch window is longer than 2 hours, a 2-hour span within the total launch opportunity will be jointly chosen by Lockheed Martin and the customer. This decision can be made as late as a few days before launch. The selected span will be chosen based on operational considerations, such as preferred time of day or predicted weather.

Some missions may have more complicated window constraints requiring analysis by Lockheed Martin. For example, launch system performance capability constrains the windows for missions that require precise control of the right ascension of the ascending node. That control is achieved by varying the trajectory as a function of launch time. We have successfully analyzed a variety of window constraints for past missions, and we are prepared to accommodate required window constraints for future missions.

5.2.21 Wind Placard Analysis (Prelaunch, Flight)

Wind tunnel tests of the Atlas IIA and IIAS configurations have been performed to determine loading for both ground and flight wind conditions and have been further applied to the Atlas IIIA and IIIB configurations. This information, combined with launch site wind statistics, is used to determine the wind placards and subsequent launch availability for any given launch date. All Atlas vehicle configurations provide at least 85% annual launch availability.

5.2.22 Range Safety Analyses

Lockheed Martin performs the flight analyses required to comply with Eastern/Western Range regulations for both the request for preliminary flight plan approval and the more detailed submittal for final flight plan approval. These submittals occur approximately 1 year before launch for the initial request and approximately 45 days before for the second. Reports and magnetic tapes of the required information are provided to the range agency in the required formats and include nominal and nonnominal trajectories and impact locations of jettisoned hardware. During spacecraft integration, a range support plan is prepared documenting our planned coverage.

5.2.23 End-to-End Electrical Compatibility Analysis

Lockheed Martin conducts an end-to-end electrical circuit analysis to verify proper voltage/current parameters and any required timing/sequencing interfaces between all spacecraft and launch vehicle airborne interfaces (through to the end function). This analysis requires data from the spacecraft, such as pin assignments, wiring interfaces, and circuit detail of avionics (first level) to verify end-to-end (spacecraft-to-launch vehicle) compatibility. All "in-between" wiring/circuits are analyzed to verify proper routing, connections and functionality of the entire system interface. This analysis is documented as part of the ICD verification process and used to generate inputs for all necessary launch site interface testing.

5.2.24 Postflight Data Analysis

For Atlas missions, Lockheed Martin uses an effective analysis technique to obtain the individual stage payload performance capabilities derived from available flight test data. The main outputs of the analysis are: (1) Atlas stage performance with respect to the predicted nominal (given in terms of Centaur propellant excess), (2) Centaur stage performance with respect to its predicted nominal (given also in terms of Centaur propellant excess), and (3) the average thrust and specific impulse of the Centaur stage. In addition to these outputs, the postflight performance report presents historical data for past flights of similar family and statistics of the outputs of principal interest. The report also gives insight into the behavior of the propellant utilization (PU) system of the Centaur stage and it provides a trajectory listing of simulated Centaur flight that effectively matches observed data from the actual flight.

A principal input into the postflight analysis is flight telemetry data of the actual variables as function of time from the onboard flight computer. Telemetered outputs from the PU system are used to obtain the propellants remaining in the LO_2 and liquid hydrogen (LH_2) tanks at Centaur final cutoff. Times of key vehicle mark events are also required. The actual vector states of radius and velocity at Atlas stage shutdown, compared to the predicted nominal values, provide sufficient knowledge to obtain the Atlas stage performance. The flight propellant excess at Centaur final cutoff (from the PU system data) and the actual burn times for Centaur provide key data to determine the thrust and specific impulse for the Centaur stage.

In addition to the performance evaluation of the launch vehicle, the postflight report provides an assessment of the injection conditions in terms of orbital parameters and deviations from the target values and spacecraft separation attitude and rates.

Finally, the report documents the payload environments to the extent that the launch vehicle instrumentation permits. These environments could include interface loads, acoustics, vibration, and shock.

5.2.25 Destruct System Analysis

Lockheed Martin updates the airborne segment of the detailed description of the Atlas/Centaur range safety system, including a full analysis of system safety and reliability for each mission as required. The analysis includes the qualification status of each component and a summary of component and system-level testing. The analysis also includes worst-case component, battery, cordage, connectors, firing, squib parameters, and timing. It is designed to evaluate the tolerance to single-point failure, RF pattern capability, critical resistance, RF sensitivity, voltage standing-wave ratio (VSWR), and insertion loss. Our range safety system is evaluated with respect to meeting requirements of Eastern Space and Missile Center (ESMCR) 127-1 and/or EWR 127-1.

5.2.26 Mission Targeting

Mission targeting is conducted to define the target orbit parameters that will be used to guide the launch vehicle into the desired orbit. This process requires a target specification from the spacecraft agency and results in the publication of the flight constants tapes used to load the flight computer and the mission-peculiar firing tables.

5.2.27 Mission-Peculiar Flight Software

Our mission-peculiar software activity for mission integration is a controlled process that ensures the generation and release of the flight control subsystem (FCS) software to support the launch schedule. Our modular software design minimizes the impact of changes due to mission-peculiar requirements. The mature system of development and controls at Lockheed Martin, in place and operational for all Atlas missions, is used to customize a reliable, tested product specifically designed to satisfy new mission requirements.

The FCS software baseline encompasses all the functionality needed to fly most typical Atlas missions.

A complete software validation test program is run using the specific mission trajectory to validate the flight software for nominal, 3-sigma dispersed, severe stress, and failure mode environments before flight. Testing and validation are completed in our Systems Integration Laboratory (SIL), which includes flight-like avionics components.

5.3 POLICIES

This section is intended to provide potential and current launch services customers with information concerning some management, integration, and production policies to ensure efficient integration and launch of the customer's payload.

5.3.1 Launch Vehicle Logos

As part of our standard launch service, the Atlas program offers customers the option of placing a mission or company logo on portions of that mission's PLF hardware. The logo can be placed in standard locations on the PLF cylindrical section. To support manufacture of the mission PLF, the Atlas program typically needs to have final artwork for the logo by 10 months before launch. This timeframe allows the Lockheed Martin engineering organization to transform the artwork into a template to be used at our Harlingen facility for application of the final logo artwork onto the fairing. Delivery of the customer PLF logo design is a schedule milestone required to support nominal assembly spans for PLF fabrication. Logo changes or modifications proposed by customers after this milestone may be accommodated in the following manner:

1) Area of application is confined to the PLF nosecone;
2) All art and material for the changes design will be supplied by the requester;
3) Art should be designed to account for parallax distortions caused by the slope of the nosecone;
4) Requester-supplied materials (art, adhesives, etc) shall comply with Atlas program furnished material and process specifications (Atlas program Specifications 0-00274 and 5-70042);
5) Changes will be supplied at a time that supports scheduled PLF completion date at our Harlingen facility.

5.3.2 Launch Scheduling

Atlas launch capability is nominally 10 to 12 missions per year depending on vehicle mix. A typical launch year capability at CCAS consists of six launches on Pad 36A and six launches on Pad 36B. Annual maintenance is scheduled for Pad 36A and worked during burn-off/refurbishment period on Pad 36B. The nominal launch rate capability from SLC-3 at VAFB is projected to be four per year.

Missions are contracted and scheduled into available launch opportunities typically 12 to 18 months in advance. Missions that are reflights of an existing bus will typically be 6 to 12 months. The earlier a desired schedule position is contracted for the more likely it will be available.

Scheduling and rescheduling launches in the manifest requires the equitable treatment of all customers. Table 5.3.2-1 summarizes the scheduling and rescheduling typically followed to ensure equitable treatment. Sequential scheduling of launches in the queue, the customer's position in the queue, and vehicle processing flow time will dictate earliest launch date(s) (Fig. 5.3.2-1). Lockheed Martin endeavors to fill each position in the queue. Consequently, once in queue, close coordination is required should the customer desire rescheduling. Rescheduling requires mutual agreement on the selection of available launch opportunities offered by Lockheed Martin.

Table 5.3.2-1 Scheduling Guidelines for Original Manifesting or Manifest Changes Due to Delays

General Policy
• Scheduling Missions in the Manifest & Adjusting Schedules Due to Atlas Delays or Customer Delays; Require Consistent & Even-Handed Treatment While Minimizing Cost & Revenue Impacts to Both the Customer & the Atlas Program • Positions Are Assigned To Satisfy Contract Provisions & Maintain Customer Satisfaction, While Avoiding or Minimizing Aggregate Delays To Manifest
Groundrules
• Publish a Publicly Releasable 18-mo Look-Ahead Manifest Annually in July • Six Launches per year Are Nominally Planned on Pad 36A & Six Launches per year on Pad 36B, with Time Allocated for Pad Maintenance & Contingencies • Every Effort Is Made To Conduct Customer Launches in a Period Desired by the Customer; However, Should a Customer's Scheduled Launch Date Conflict with That of Another Customer, That Customer with the Earlier Effective Contract Date May Be Considered in Determining Which Customer Is Entitled To Launch First • If a Customer Contracts for Two or More Launches, Payloads May Be Interchanged Subject to Mutual Agreement
Delays
• Once in Queue, Customers Will Stay in Queue if Lockheed Martin Causes a Delay or if the Customer Causes a Short Delay • Customer Delays (Announced or Anticipated) May Require Resequencing to the Next Available Launch Opportunity • A Customer-Directed Delay May Be Considered Equivalent to a New Contract Award Date for Priority Determination
Exceptions
• Reflight or Replacement Launches for Satellite or Launch Vehicle Failure May Be Given Priority Within the Manifest Guidelines • Planetary Window Missions Will Be Given Special Consideration Within the Manifest Guidelines
Note: Mission Scheduling Guidelines Are General Guidelines Only & Subject To Change

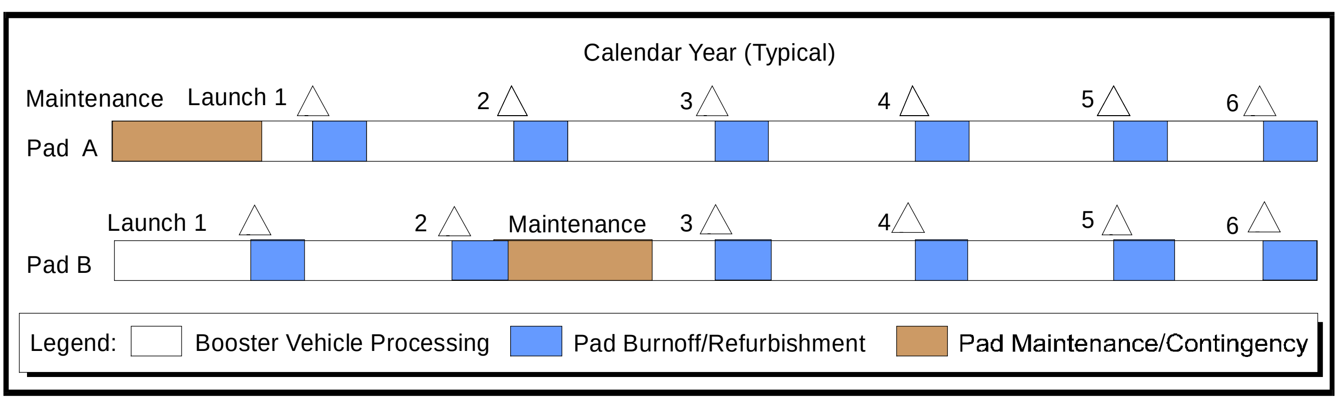


Figure 5.3.2-1 Typical Launch Schedule

5.3.3 Spacecraft Launch Window Options

Atlas can be launched at any time of the day year round. However, seasonal weather patterns should be considered in setting launch windows when possible. To ensure on-time launches and avoid cost or schedule delays, missions that may be scheduled during the months of June, July, August, and September should be planned for morning launches. Launches in the afternoon during these months have an increased probability of delays due to seasonal thunderstorm activity. Scheduling in the morning will reduce the risk of such delays and avoid cost associated with them. Options for afternoon summer launches may be available with recognition of the additional cost.

Any launch window duration can be accommodated. However, a window of between 45 minutes and 120 minutes is recommended. Shorter windows increase the risk of a launch delay if exceeded due to weather or technical problem resolution. Windows longer than 2 hours may be limited by liquid oxygen supplies or crew rest limits.

6.0 SPACECRAFT AND LAUNCH FACILITIES

Lockheed Martin has formal agreements with the United States Air Force (USAF), the National Aeronautics and Space Administration (NASA), and Astrotech for the use of payload and launch vehicle processing facilities at and near the Atlas launch sites at Cape Canaveral Air Station (CCAS), Florida. Similar agreements for sites at Vandenberg Air Force Base (VAFB) in California are being planned and/or implemented. Long-term use agreements are in place for Complex 36 at CCAS and are in work for Complex 3 at VAFB that encompass facilities, range services, and equipment.

This section summarizes the launch facilities capabilities available to Atlas users. Both CCAS and VAFB facilities are discussed. Additional CCAS facilities data can be found in the Atlas Launch Services Facilities Guide. Additional VAFB facilities data can be obtained by contacting Lockheed Martin.

6.1 CCAS SPACECRAFT FACILITIES

CCAS facilities include spacecraft processing facilities available to commercial and U.S. government users. Figures 6.1-1 and 6.1-2 illustrate the location of facilities between SLC-36 and Astrotech.

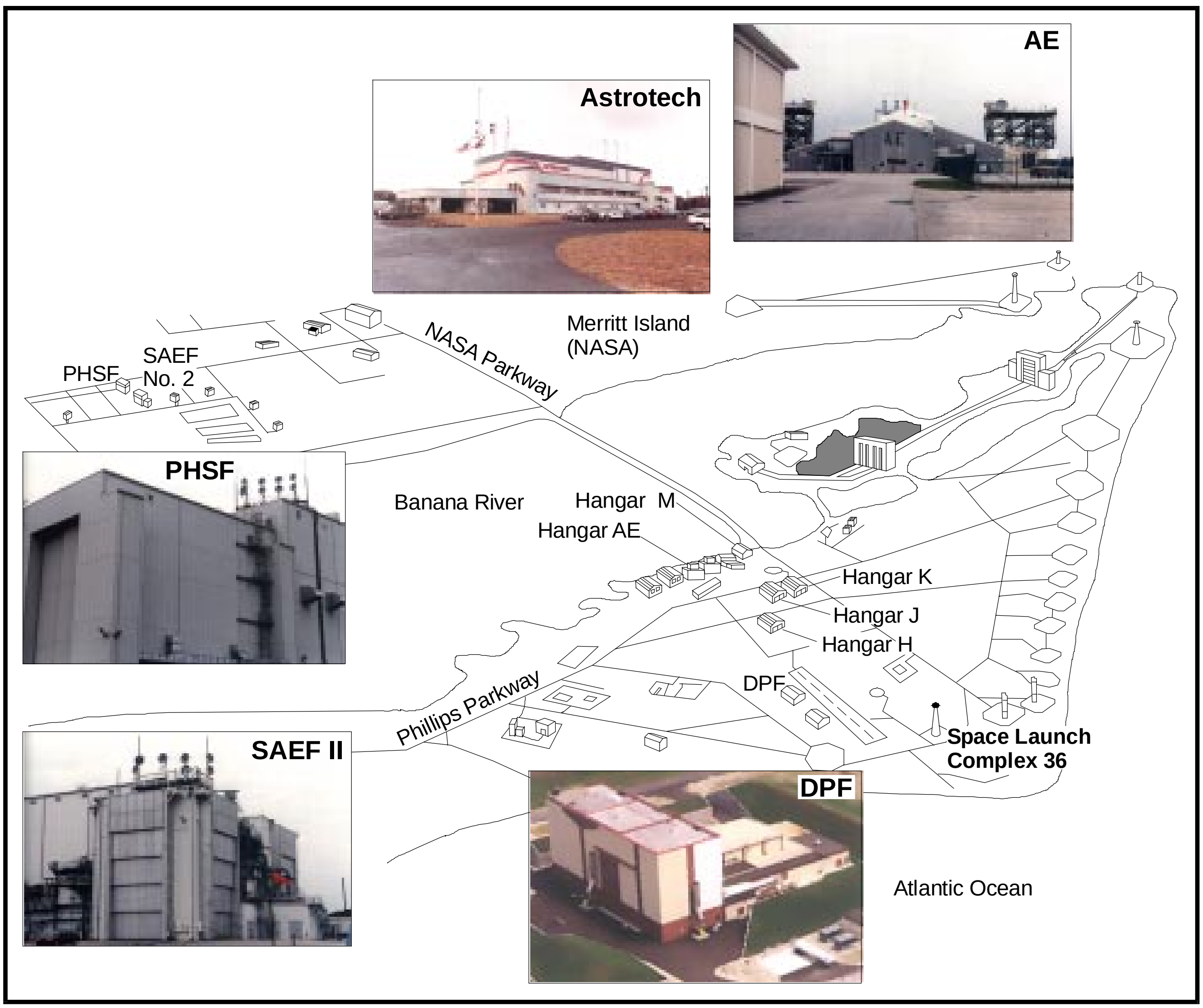


Figure 6.1-1 Facilities at Cape Canaveral Air Station, Florida

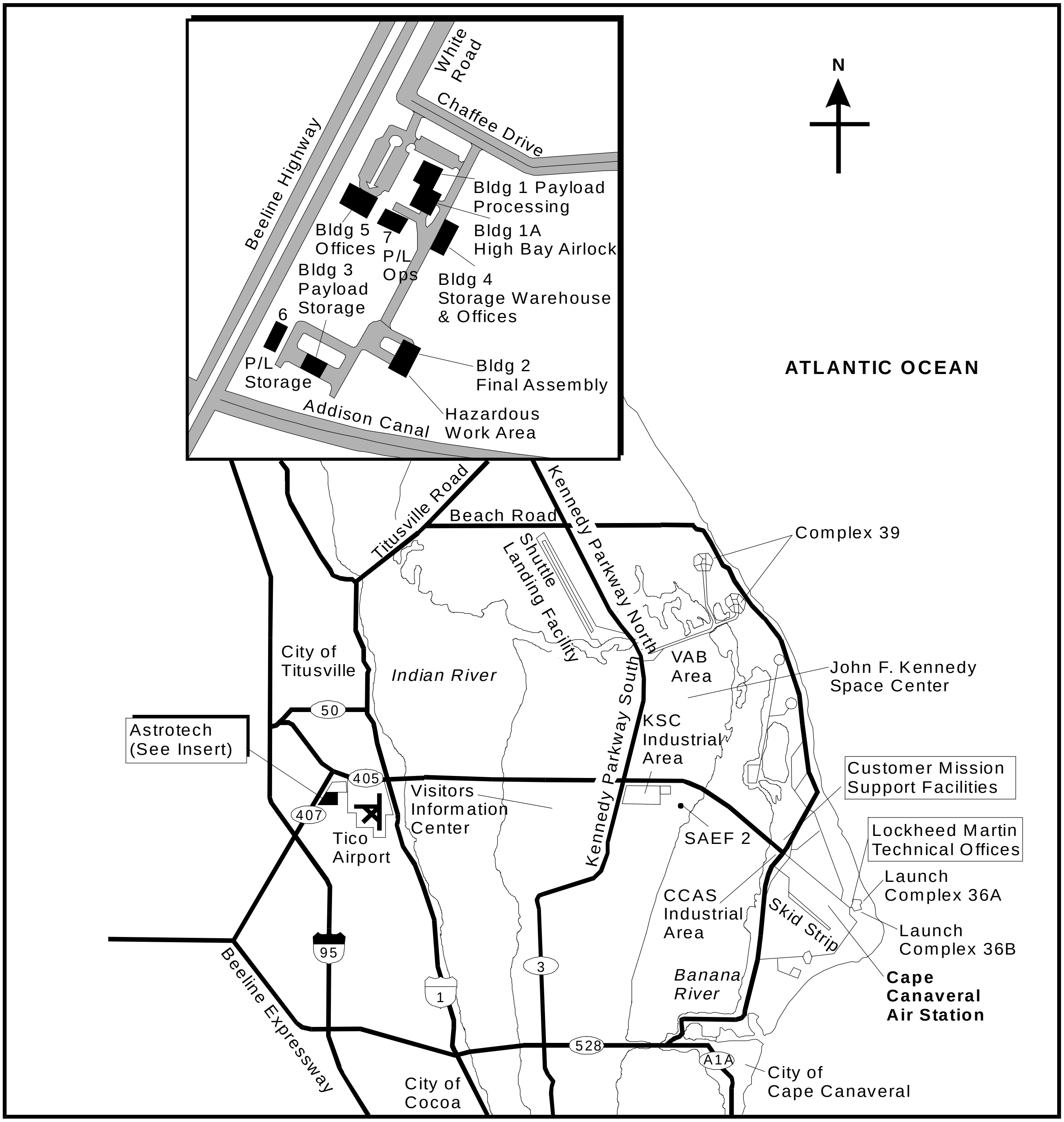


Figure 6.1-2 Facility Locations at CCAS

6.1.1 Astrotech

The Astrotech commercial payload processing facility (PPF), owned and operated by Spacehab Inc., is the primary facility for processing Atlas class civil government and commercial spacecraft. This facility (Fig. 6.1.1-1) contains separate nonhazardous and hazardous processing buildings, storage buildings, and offices. The facilities and floor plans are described in the following sections. Astrotech complies fully with all applicable federal, state, regional, and local statutes, ordinances, rules, and regulations relating to safety and environmental requirements.

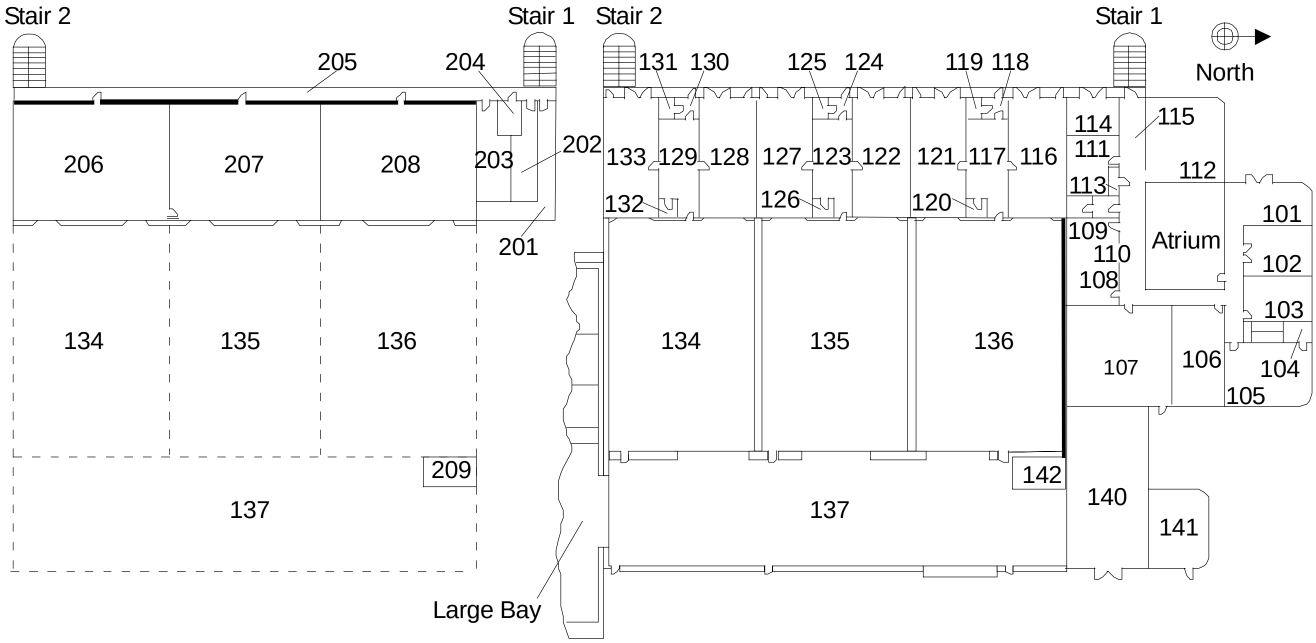
Should the Astrotech facility not adequately satisfy spacecraft processing requirements, government facilities, as outlined in Lockheed Martin/NASA and Lockheed Martin/USAF agreements, are available. These facilities also are described in the following sections.

Astrotech PPF—Astrotech Building 1, with its high bay expansion, is considered the primary PPF. Its floor plan is depicted in Figures 6.1.1-2 and 6.1.1-3. With overall dimensions of approximately 61.0 m (200 ft) by 38.1 m (125 ft) and a height of 14 m (46 ft), the building's major features are:

1) An airlock;
2) High bays (three identical and one larger);
3) Control rooms (two per high bay);
4) Office complex, administrative area, communications mezzanine, conference rooms, and support areas.

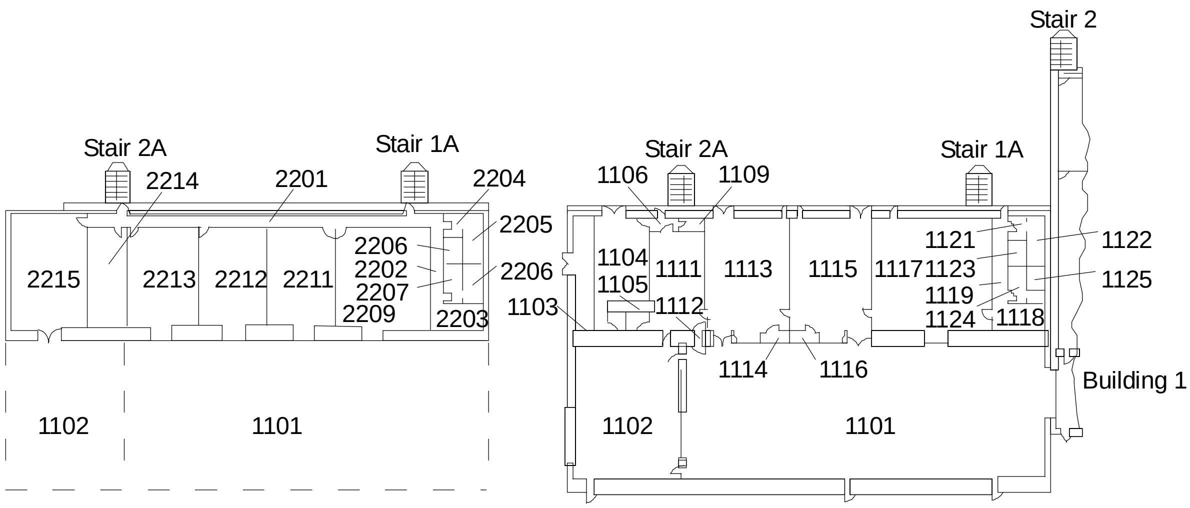
Table 6.1.1-1 lists details of room dimensions, cleanliness, and crane capabilities of this facility.

Figure 6.1.1-1 Astrotech Facility



Rm	Function	Rm	Function	Rm	Function
108	Conference Room	121	Control Room A2	133	Control Room C2
109	Women's Restroom	122	Control Room B1	134	High Bay C
110	Women's Lounge	123	Change Room B	135	High Bay B
111	Men's Restroom	124	Vestibule B	136	High Bay A
112	Break/Lunch Room	125	Storage B	137	Common Airlock
114	Machine Shop	126	Restroom B	202	Women's Restroom
115	Corridor	127	Control Room B2	203	Men's Restroom
116	Control Room A1	128	Control Room C1	205	Corridor
117	Control Room A	129	Change Room C	206	Office Area C
118	Vestibule 1	130	Vestibule C	207	Office Area B
119	Storage A	131	Storage C	208	Office Area A
120	Restroom A	132	Restroom C		

Figure 6.1.1-2 Astrotech Building 1 Detailed Floor Plan



Room	Function	Room	Function	Room	Function
1101	Large High Bay D	1117	Office Area D1	2205	Men’s Restroom
1102	Large Airlock	1118	Break Room	2207	Women’s Washroom
1104	Conference D1	1119	Corridor	2208	Women’s Restroom
1105	Closet	1121	Men’s Washroom	2209	Office Area D2
1106	Restroom	1122	Men’s Restroom	2211	Office Area D3
1107	Restroom	1124	Women’s Washroom	2212	Office Area D4
1108	Vestibule	1125	Women’s Restroom	2213	Office Area D5
1111	Change Room D	2201	Corridor	2214	Office Area D2
1112	Air Shower	2202	Corridor	2215	Office Area D6
1113	Control Room D2	2203	Breakroom		
1115	Control Room D1	2204	Men’s Washroom		

Figure 6.1.1-3 Astrotech Building 1 Large Bay Detailed Floor Plan—First Floor

Table 6.1.1-1 Astrotech's Building 1 features four high bay areas.

Building 1 High Bays (3): Class 100,000 Clean Room		
Temperature	23.8 ±2.8°C	75 ±5°F
Relative Humidity	50 ±5%	
Usable Floor Space	18x12.19 m	60x40 ft
Floor Area	223.1 m^2	2,400 ft^2
Clear Height	13.1 m	43 ft
Crane Type (Each Bay)	Bridge	
Crane Capacity	9,072 kg	10 ton
Crane Hook Height	11.3 m	37.1 ft
Door Size	6.1x7 (h) m	20x23 (h) ft
High Bay Control Rooms (2 Bay)		
Size	9.14x4.27 m	30x14 ft
Area	39.0 m^2	420 ft^2
Ceiling Height	2.67 m	8.75 ft
Door Size	2.44x2.44 m	8x8 ft
Bay Window Size	1.22x2.44 m	4x8 ft
Temperature	23.8 ±2.8°C	75 ±5°F
Airlock: Class 100,000 Clean Room		
Temperature	23.8 ±2.8°C	75 ±5°F
Relative Humidity	50 ±5%	
Usable Floor Space	36x9.14 m	120x30 ft
Clear Ceiling Height	7.32 m	24 ft
Door Size	6.1x7 m	20x23 ft
Building 1 Large Bay Control Rooms (2)		
Size	12.2x13.1 m	40x43 ft
Area	159.8 m^2	1,720 ft^2
Temperature	23.8 ±2.8°C	
Offices (Second Floor)		
Floor Size (1 Room)	12.8x7.3 m	75 ±5°F
Floor Area	93.44 m	1,008 ft
Floor Size (2 Room)	12.5x7.3 m	41x24 ft
Floor Area	92.25 m	984 ft
Acoustic Ceiling Height	2.43 m	8 ft
Bay Window Size	1.22x1.22 m	4x4 ft
Building 1 High Bay: 100,000 Clean Room		
Temperature	23.8 ±2.8°C	75 ±5°F
Relative Humidity	50 ±5%	
Usable Floor Space	15.5x38.1 m	51x125 ft
Floor Area	592.2 m^2	6,375 ft^2
Clear Height	18.3 m	60 ft
Crane Type (Each Bay)	Bridge	
Crane Capacity	27,216 kg	30 ton
Crane Hook Height	16.8 m	55 ft
Door Size	6.1x15.2 (h) m	20x60 (h) ft
Building 1 Large Bay Airlock		
Temperature	23.8 ±2.8°C	75 ±5°F
Relative Humidity	50 ±5%	
Usable Floor Space	12.2x15.2 m	40x50 ft
Clear Ceiling Height	18.3 m	60 ft
Door Size	6.1x15.2 (h)m	20x50 (h) ft

Astrotech Hazardous Processing Facility (HPF)—Astrotech's Building 2 is considered the primary HPF. With overall dimensions of approximately 48.5 m (159 ft) by 34.1 m (112 ft) and a height of 14 m (46 ft), the major features are:

1) An airlock;
2) Two spacecraft processing high bay and operations rooms;
3) Two encapsulation high bays;
4) One spin balance bay.

The Building 2 floor plan is depicted in Figure 6.1.1-4. Each high bay is built to explosion-proof or equivalent standards to support operations involving liquid propellant transfer, solid propellant motor preparations, and ordnance installation. Table 6.1.1-2 lists the facility's room dimensions, cleanliness levels, and crane capabilities.

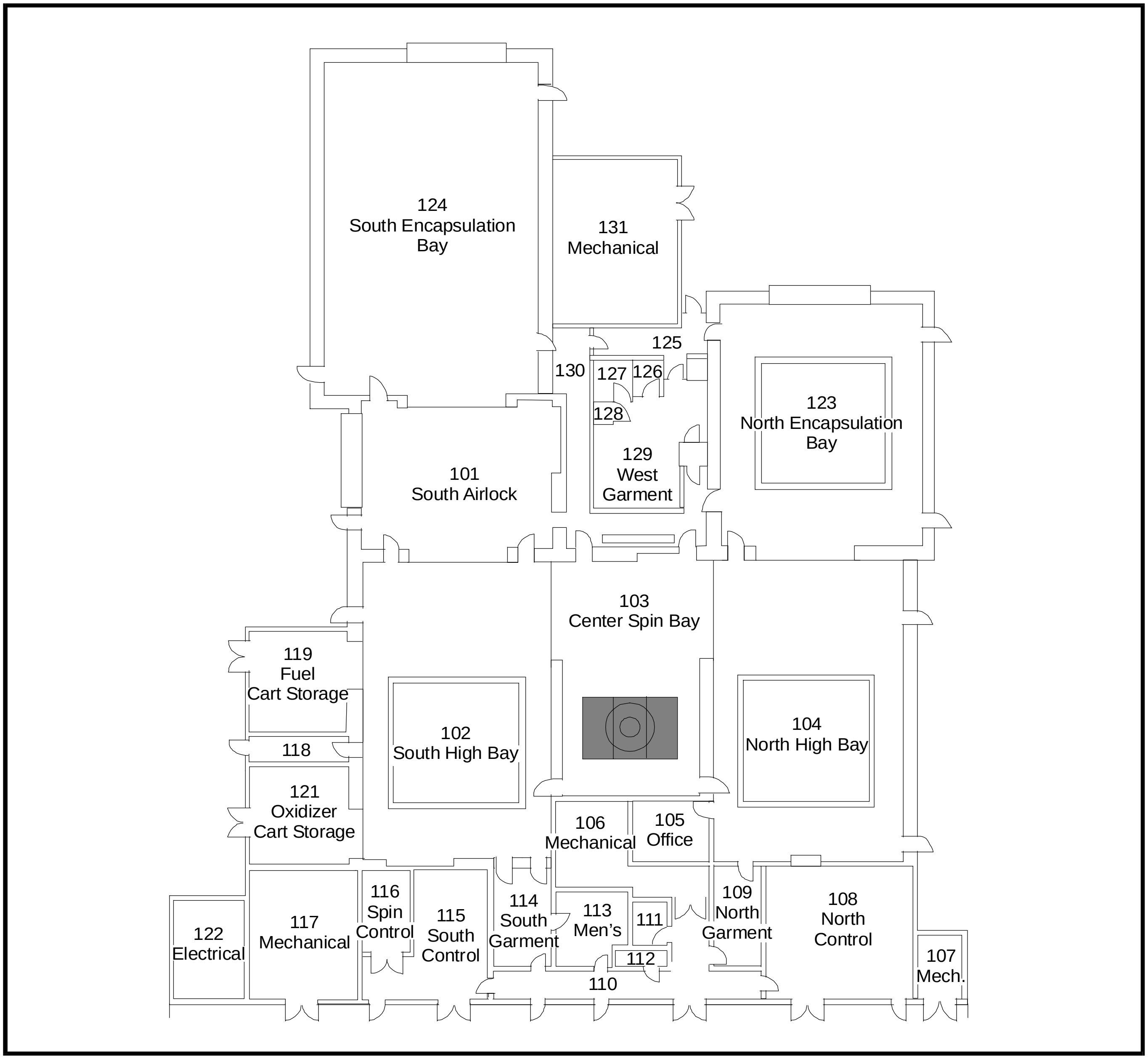


Figure 6.1.1-4 Astrotech's Building 2 provides excellent hazardous operations facilities for liquid and solid propellant-based satellites.

Astrotech Spacecraft Support Facilities—Building 3 at Astrotech is a thermally controlled, short-term payload storage area with payload processing activities or long-term satellite storage. Storage bay and door dimensions and thermal control ranges are identified in Table 6.1.1-3.

Astrotech's Building 4 is a warehouse storage area without environmental controls and is suitable for storage of shipping containers and mechanical ground support equipment (GSE). Table 6.1.1-4 details this facility's dimensions.

Astrotech's Building 5 provides 334.4 m^2 (3,600 ft^2) of customer office space divided into 17 individual offices with a reception area sufficient to accommodate up to three secretaries.

Building 6 at Astrotech provides storage primarily for Lockheed Martin payload fairing (PLF) support equipment. However, if additional nonenvironmentally controlled storage is required by customer users, this additional space can be made available.

Table 6.1.1-2 Astrotech's Building 2 features three blast-proof processing rooms.

South & North High Bays (Rooms 102, 104)			**Airlock (Room 101)**		
Cleanliness	Class 100,000		Cleanliness	Class 100,000	
Temperature	75±5°F	23.9±2.8°C	Temperature	75±5°F	23.9±2.8°C
Relative Humidity	50 ±5%		Relative Humidity	50±5%	
Floor Space	60x37 ft	18.3x11.3 m	Floor Space	38x29 ft	11.6x8.8 m
Fuel Island	25x25 ft	7.6x7.6 m	Clear Height	43 ft	13.1 m
Clear Height	43 ft	13.1 m	Door to Room 124 (wxh)	20x40 ft	6.1x12.2 m
Doors to Room 101 & 123 (wxh)	20x40 ft	6.1x12.2 m	Door to Room 102 (wxh)	20x40 ft	6.1x12.2 m
Door to Room 103 (wxh)	20x40 ft	6.1x12.2 m	Crane	Monorail	
Crane Type (Each Bay)	Bridhe		Crane Capacity	2 ton	1.8 tonne
Crane Capcity	10 ton	9.1 tonne	Crane Hook Height	37 ft	11.3 m
Crane Hook Height	37 ft	11.3 m	**Prop Cart Rooms (Rooms 119, 121)**		
North High Bay Control Room (Room 108)			Floor Space	20x20 ft	6.1x6.1 m
Temperature	70 to 75°F	21.1 to 23.9°C	Clear Height	9.33 ft	2.84 m
Floor Space	25x30 ft	7.6x9.1 m	Door to Room 102 (wxh)	10x8 ft	3.1x2.4 m
Ceiling Height	9.33 ft	2.84 m	Door to Outside (wxh)	6x6.67 ft	1.8x2.03 m
Door to Outside (wxh)	8x8 ft	2.4x2.4 m	**North Encapsulation High Bay (Room 123)**		
Bay Window (wxh)	2.67x2.50 ft	0.81x0.76 m	Cleanliness	Class 100,000	
South High Bay Control Room (Room 115)			Temperature	75±5°F	23.9±2.8°C
Temperature	70 to 75°F	21.1 to 23.9°C	Relative Humidity	50±5%	
Floor Space	25x15 ft	7.6x4.6 m	Floor Space	50x40 ft	15.2x12.2 m
Ceiling Height	9.33 ft	2.84 m	Fuel Island	25x25 ft	7.6x7.6 m
Door to Outside (wxh)	8x8 ft	2.4x2.4 m	Clear Height	65 ft	19.8 m
Bay Window (wxh)	2.67x2.50 ft	0.81x0.76 m	Door to Outside (wxh)	20x50 ft	6.1x15.2 m
Spin Balance High Bay (Room 103)			Door to Room 104 (wxh)	20x40 ft	6.1x12.2 m
Cleanliness	Class 100,000		Crane Type	Bridge	
Temperature	75±5°F	23.9±2.8°C	Crane Capacity	30 ton	27.2 tonne
Relative Humidity	50±5%		Crane Capacity	27,216 kg	30 ton
Floor Space	48x27 ft	14.6x8.2 m	Crane Hook Height	55 ft	16.8 m
Clear Height	43 ft	13.1 m	**South Encapsulation High Bay (Room 124)**		
Doors to Room 102 & 104, (wxh)	20x40 ft	6.1x12.2 m	Cleanliness	Class 100,000	
Crane Type	Bridge		Temperature	75±5°F	23.9±2.8°C
Crane Capacity	10 ton	9.1 tonne	Relative Humidity	50±5%	
Crane Hook Height	37.0 ft	11.3 m	Floor Space	45x70 ft	13.7x21.3 m
Spin Balance High Bay Control Room (Room 116)			Clear Height	65 ft	19.8 m
Temperature	75±5°F	23.9±2.8°C	Door to Outside (wxh)	23x55 ft	7.0x16.8 m
Floor Space	10x15 ft	3x4.6 m	Door to Room 101 (wxh)	20x40 ft	6.1x12.2 m
Clear Height	9.33 ft	2.84 m	Crane Type	Bridge	
Door to Room 115 (wxh)	6x6.67 ft	1.8x2.03 m	Crane Capacity	30 ton	27.2 tonne
			Crane Hook Height	55 ft	16.8 m

Spacecraft Services—Full services for payload processing and integration can be provided at Astrotech.

Electrical Power and Lighting—The Astrotech facility is served by 480-Vac/three-phase commercial 60- and 50-Hz electrical power that can be redistributed as 480-Vac/three-phase/30-A, 120/208-Vac/three-phase/60-A, or 120-Vac/single-phase, 20-A power to any location in Buildings 1 and 2. Commercial power is backed up by a diesel generator during critical testing and launch periods.

Astrotech can provide 35 kW of 230/380-Vac/three-phase 50-Hz power, which is also backed up by a diesel generator.

The high bays and airlocks in Buildings 1 and 2 are lighted by 400-W metal halide lamps to maintain 100-ft-candles illumination. Control rooms, offices, and conference areas have 35-W fluorescent lamps to maintain 70-ft-candles of illumination.

Telephone and Facsimile—Astrotech provides all telephone equipment, local telephone service, and long-distance access. A Group 3 facsimile machine is available.

Intercommunication Systems—Astrotech provides a minimum of three channels of voice communications among all work areas. The facility is connected to the NASA/USAF Operational Intercommunications System and Transistorized Operational Phone System (TOPS) to provide multiple-channel voice communications between the Astrotech facility and selected locations at CCAS.

Closed-Circuit Television (CCTV)—CCTV cameras are located in high bays of Building 2 and can be placed in high bays of Building 1, as required, to permit viewing operations in those areas. CCTV can be distributed within the Astrotech facility to any location desired. In addition, Astrotech has the capability to transmit and receive a single channel of video to and from Kennedy Space Center (KSC)/CCAS via a dedicated microwave link.

Table 6.1.1-3 Astrotech's payload storage building is used for short-term hardware storage.

Building 3: Thermally Controlled Storage Facility		
Temperature Control	70 to 78°F	21.1 to 25.6°C
Relative Humidity	50±10%	
Floor Space		
• Bays A, C, D & F	25x22 ft	7.6x6.7 m
• Bays B & E	25x24 ft	7.6x7.3 m
Clear Height (All Bays)	28 ft	8.5 m
Door Size (wxh)		
• Bays A, C, D & F	20x25 ft	6.1x7.6 m
• Bays B & E	18x25 ft	5.5x7.6 m
Crane	None	

Table 6.1.1-4 Astrotech's warehouse storage building is suitable for storage of items not requiring climate controls.

Building 4: Storage Without Environmental Control		
Floor Space	50x125 ft	15.2x38.1 m
Clear Height	28 ft	8.5 m
Door Size	20x27 ft	6.1x8.2 m
Crane	None	
Floor Space		
• Bays A, C, D & F	25x22 ft	7.6x6.7 m
• Bays B & E	25x24 ft	7.6x7.3 m
Clear Height (All Bays)	28 ft	8.5 m
Door Size (wxh)		
• Bays A, C, D & F	20x25 ft	6.1x7.6 m
• Bays B & E	18x25 ft	5.5x7.6 m
Crane	None	

Command and Data Links—Astrotech provides both wideband and narrowband data transmission capability and the KSC/CCAS cable transmission system to all locations served by the KSC/CCAS network. If a spacecraft requires a hardline transmission capability, the spacecraft is responsible for providing correct signal characteristics to interface to the KSC/CCAS cable transmission system.

Astrotech provides antennas for direct S-band, C-band, and Ku-band airlinks from the Astrotech facility to Launch Complex 36 and antennas for S-band, C-band, and Ku-band airlinks between Astrotech Buildings 1 and 2. There is also a ground connection between the PPF and HPF for hardline radio frequency (RF) transmissions up to Ku-band.

Remote Spacecraft Control Center—Astrotech has the capability to link remote ground stations (voice and data) between Astrotech and CCAS resources.

Temperature and Humidity Control—The environment of all Astrotech high bays and airlocks is maintained at a temperature of 24 ±2.8°C (75 ±5°F) and a relative humidity of 50 ±5%. The environment of all other areas is maintained by conditioned air at a temperature between 21 and 25°C (70 to 78°F) and a comfortable humidity.

Compressed Air—Regulated compressed air at 125 psi is available in Buildings 1 and 2.

Security and Emergency Support**—**Perimeter security is provided 24 hours a day. Access to the Astrotech facility is via the main gate, where a guard is posted during working hours to control access. Internal security is provided by cypher locks on all doors leading into payload processing areas. Brevard County provides emergency medical support and the City of Titusville provides emergency fire support. In an accident, personnel will be transported to Jess Parish Hospital in Titusville. Both medical and fire personnel have been trained by NASA.

Foreign Trade Zone**—**Astrotech has been designated as a foreign trade zone. Astrotech will coordinate all licensing requirements to meet governmental regulations for importing and exporting support hardware for the duration of mission support.

6.1.2 USAF and NASA Facilities

Building AE Mission Operations Facility—Building AE is a spacecraft and missions operations facility originally constructed by the USAF. With overall dimensions of approximately 36.6 m (120 ft) by 97.5 m (320 ft), it features:

1) Spacecraft checkout areas (high and low bays);
2) Mission director's center and guest observation room (Fig. 6.1.2-1);
3) Telemetry ground station and laboratory (including Astrotech's communications link with the range).

Building AE is located in the CCAS industrial area on Hangar Road. Table 6.1.2-1 provides a detailed description of Building AE facilities.

Spacecraft Assembly and Encapsulation Facility (SAEF) No. 2 PPF—SAEF 2 (Fig. 6.1.2-2) is a NASA facility located southeast of the KSC industrial area. It features:

1) A high bay;
2) Two low bays;
3) A test cell;
4) Two control rooms.

Details of the SAEF 2 facility are listed in Table 6.1.2-2.

Payload Hazardous Servicing Facility (PHSF)—The PHSF is a NASA facility located southeast of the KSC Industrial Area (next to SAEF2). Additional features of the PHSF Service Building are described in Table 6.1.2-3.

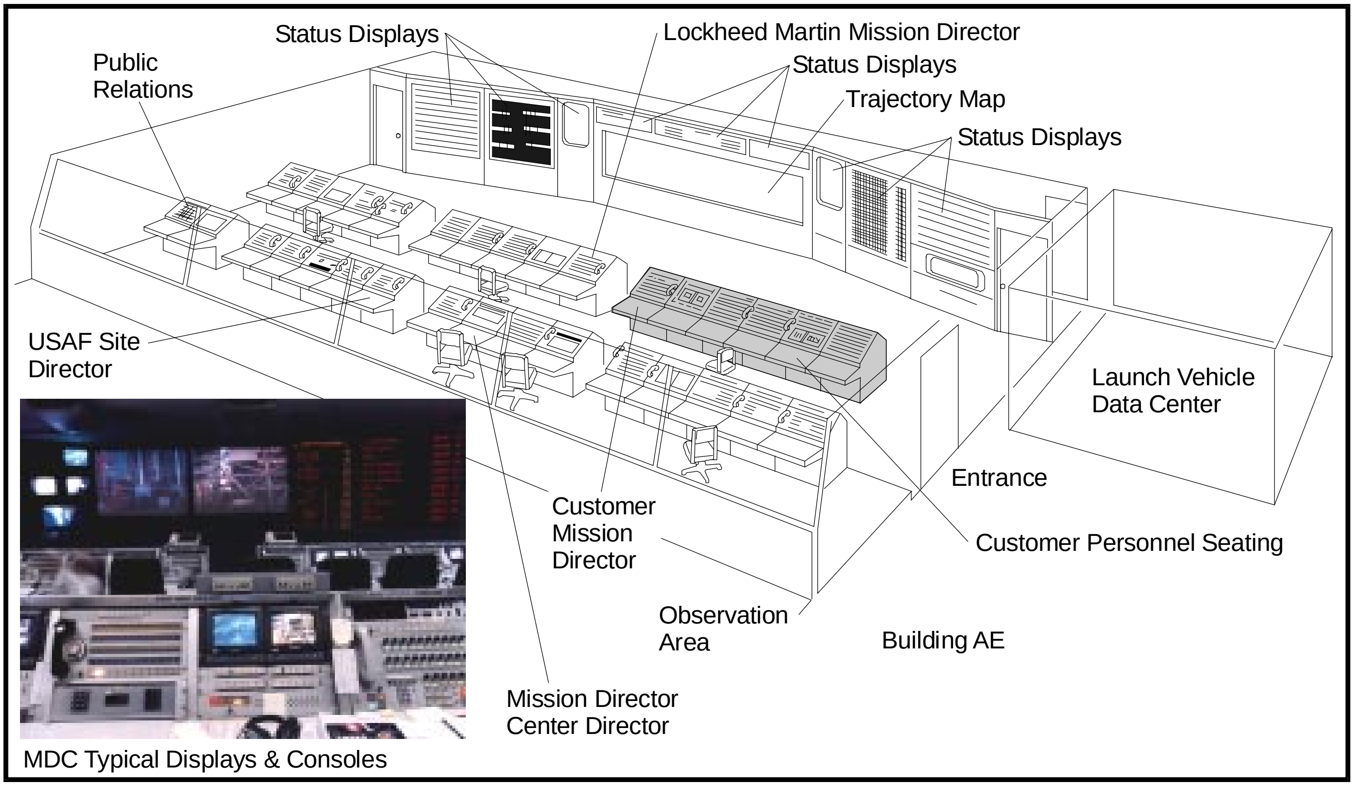


Figure 6.1.2-1 Mission Director's Center

Table 6.1.2-1 Building AE provides adequate space for smaller spacecraft processing activities.

Spacecraft Laboratory			**Airlock**		
Low Bay Floor Area	108 m^2	1,161 ft^2	Floor Space	5.5x14.6 m	18x48 ft
Height	6.1 m	20 ft	Floor Area	80.3 m^2	864 ft^2
Door Size	2.4x3 m	8x10 ft	Room Height	12.8 m	42 ft
Crane Type	Bridge		Crane Type	Monorail	
Crane Capacity	1,814 kg	2 ton	Crane Capacity	5,443 kg	6 ton
Crane Hook Height	11.3 m	37 ft	Crane Hook Height	10.3 m	33.75 ft
High Bay Clean Room (Four Rooms)			**Airlock Outside Door**		
Test Area			Width	4.9 m	16 ft
Floor Space	14.6x19.2 m	48x63 ft	Height	11.1 m	36.5 ft
Floor Area	280.32 m^2	3,024 ft^2	**Test & Storage Area Rooms**		
Room Height	12.8 m	42 ft	Southside Room		
Crane Type	Bridge	Not Clean Room Compatible	Floor Space	9.4x12.2 m	31x40 ft
Crane Capacity	2,722 kg	3 ton	Floor Area	114.68 m^2	1,240 ft^2
Crane Hook Height	10.3 m	33.75 ft	Room Height	3 m	10 ft
Crane Type	Monorail		South Door	3x3 m	10x10 ft
Crane Capacity	1,814 kg	2 ton	North Door	2.5x2 (h) m	8.5x7 (h) ft
Crane Hook Height	11.3 m	37 ft			

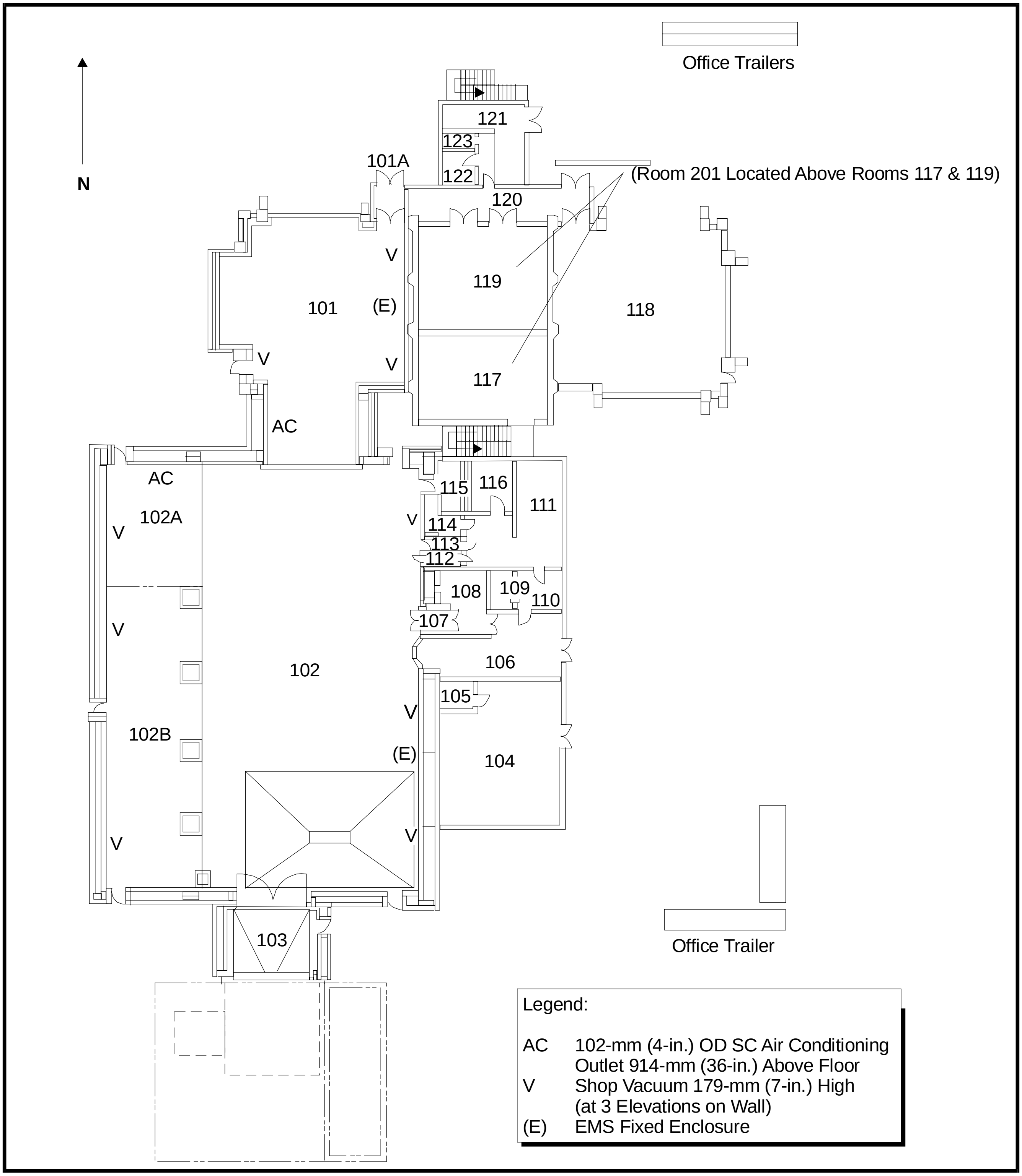


Figure 6.1.2-2 Building Floor Plan for SAEF 2

Table 6.1.2-2 SAEF 2 is the larger of the alternate hazardous processing facilities.

Control Rooms (2)		
Floor Space	9.14x10.97 m	30x36 ft
Floor Area	100.3 m^2	1,080 ft^2
Raised Flooring	0.31 m	1 ft
Ceiling Height	2.44 m	8 ft
High Bay		
Floor Size	14.94x30.18 m	49x99 ft
Floor Area	450.9 m^2	4,851 ft^2
Clear Ceiling Height	22.56 m	74 ft
Filtration	Class 100,000	
Crane Type (Ea Bay)	Bridge	
Crane Capacity	9,072 kg	10 ton
Low Bays (2)		
Floor Size (No. 1)	5.79x21.95 m	19x72 ft
Floor Area (No. 1)	127.1 m^2	1,368 ft^2
Clear Height (No. 1)	7.62 m	25 ft
Floor Size (No. 2)	5.79x8.23 m	19x27 ft
Floor Area (No. 2)	47.7 m^2	513 ft^2
Test Cell		
Floor Size	11.28x11.28 m	37x37 ft
Floor Area	127.2 m^2	1,369 ft^2
Clear Ceiling Height	15.85 m	52 ft
Door Size	6.7x12.2 m	22x40 (h) ft

Table 6.1.2-3 The PHSF Building is a recent addition to NASA's hazardous processing facility capabilities.

Service Bay		
Floor Space	18.4x32.6 m	60x107 ft
Ceiling Height	28.9 m	94 ft 10 in.
Door Dimensions	10.8x22.9 m	35x75 ft
Crane Capacity	45,400 kg	50 ton
Hook Height	25.5 m	83 ft 6 in.
Airlock		
Floor Space	15.3x25.9 m	50x80 ft
Ceiling Height	27.4 m	89 ft 10 in.
Door Dimensions	10.8x22.9 m	35x75 ft
Crane Capacity	13,600 kg	15 ton
Hook Height	22.9 m	75 ft
Equipment Airlock		
Usable Space	4.1x8.0 m	14x26 ft
Ceiling Height	3.2 m	10 ft 4 in.
Door Dimensions	3.0x3.0 m	10x10 ft
Environmental Controls		
Filtration	Class 100,000	
Air Change Rate	Four per hour Minimum	
Temperature	21.7±3.3°C	71±6°F
Relative Humidity	55% Maximum	

6.1.3 Defense System Communication Satellite (DSCS) Processing Facility

The DSCS Processing Facility (DPF) is a USAF facility accommodating both hazardous and non-hazardous payload processing operations. It provides an area in which to process and encapsulate payloads off pad. Figure 6.1.3-1 is an overview of the DPF site. The facility was designed to accommodate a DSCS III class payload consisting of a DSCS III satellite and integrated apogee boost subsystem.

The facility can accommodate 9,000 kg (20 klb) of bipropellant and/or 9,000 kg (20 klb) of solid rocket motors (SRM). The DPF is partitioned into two primary operating segments. The HPF segment consists of two high bay test cells; the nonhazardous PPF segment consists of one low bay test cell plus all other test and facility operations support areas.

Each HPF bay is a Class 100,000 clean room with approximate dimensions of 15.2x15.2x16.8-m high (50x50x55-ft high). The two cells in the HPF have been assigned the following functions:

1) East Bay (EB)—Prelaunch processing of the PLF and encapsulation of the payload within the fairing. It can also be used as a fueling cell of as an area to assemble SRMs.
2) West Bay (WB)—Bipropellant loading cell that can also be used to assemble SRMs. There is no overhead crane in this room.
3) Main Bay—Intended for nonhazardous electrical and mechanical operations and integration of payload elements before fueling. Leak testing and ordnance installation may be accomplished with prior approval from Range Safety. The main bay is a Class 100,000 clean room. Room environment is typically maintainable at 70 ±5°F with a relative humidity of 30 to 50%.

The bay is 30.5-m (100-ft) long north-south, approximately 15.2-m (50-ft) wide east-west, and 7.6-m (25-ft) high. It is equipped with a 4,500-kg (10,000-lb) crane with a hook height of 6.1 m (20 ft).

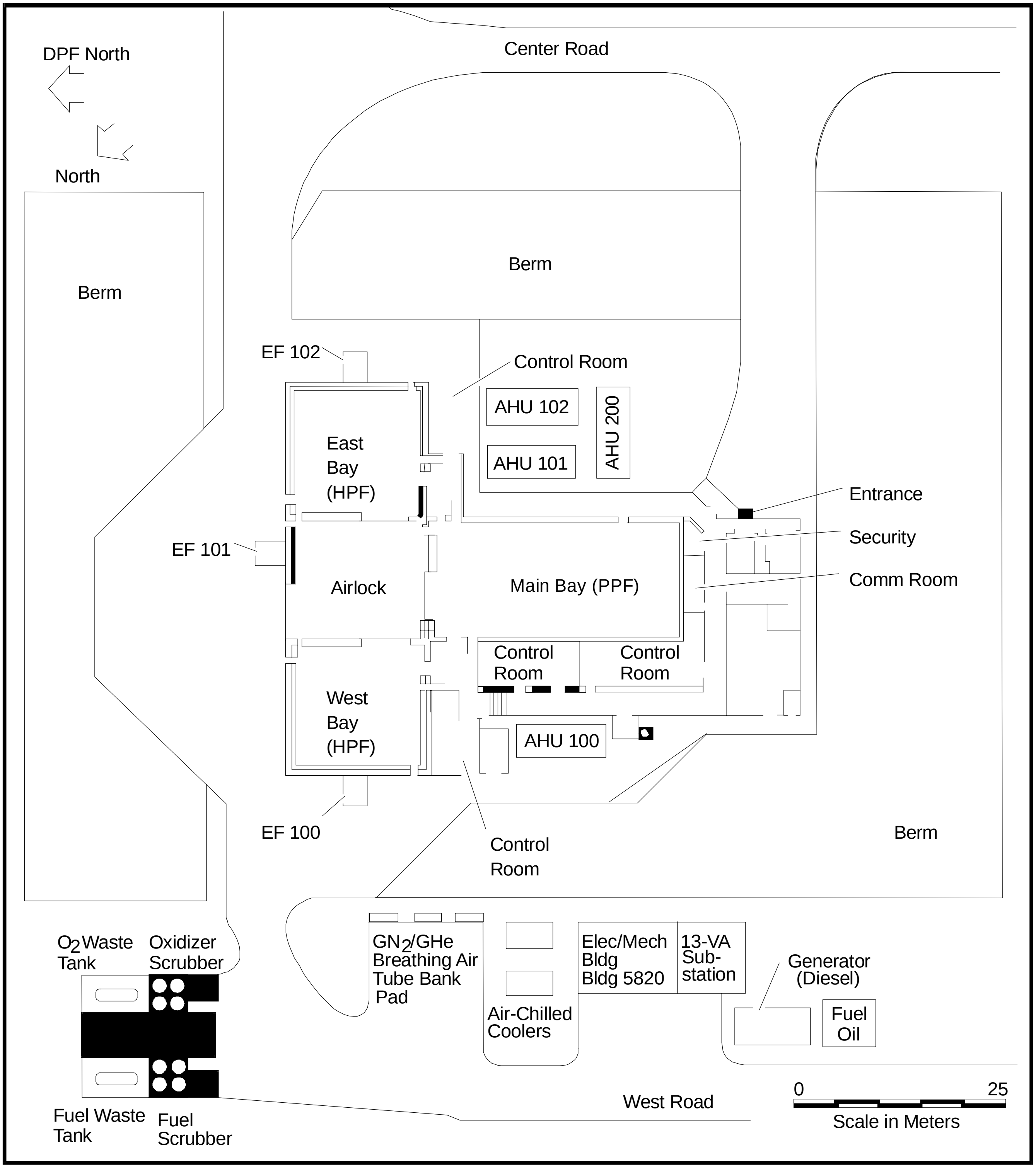


Figure 6.1.3-1 DPF Area Detail Site Plan

6.1.4 Spacecraft Instrumentation Support Facilities

CCAS area facilities described in this section can be used for spacecraft checkout as limited by compatibility to spacecraft systems. Special arrangements and funding are required to use these assets.

TEL4 Telemetry Station—The Eastern Range (ER) operates an S-band telemetry receiving, recording, and real-time relay system on Merritt Island. This system is used for prelaunch checkout of launch vehicles and spacecraft. A typical ground checkout configuration would include a reradiating antenna at the PPF, HPF, or launch pad directed toward the TEL4 antenna. Telemetry data can be recorded on magnetic tape or routed by hardline data circuits to the spacecraft ground station for analysis. TEL4 also acts as the primary terminal for telemetry data transmitted from the ER downrange stations.

Goddard Space Flight Center (GSFC)/Merritt Island Launch Area (MILA) Station—The GSFC station is also located on Merritt Island and is the ER launch area station for NASA's Ground Spaceflight Tracking and Data Network (GSTDN) Tracking and Data Relay Satellite System (TDRSS). Included are satellite ground terminals providing access to worldwide communications. Circuits from MILA to HPF, PPF, and Complex 36 are available to support checkout and network testing during prelaunch operations and spacecraft telemetry downlinking during launch and orbital operations. The MILA station can also support ground testing with TDRSS-compatible spacecraft to include TDRSS links to White Sands, New Mexico. Special arrangements and documentation are required for TDRSS testing.

JPL MIL-7.1 Station—This station is colocated at MILA on Merritt Island and is an element of the JPL Deep Space Network (DSN). This station can be configured for ground tests similar to TEL4. In addition, data from spacecraft that are compatible with the DSN can be relayed to JPL in Pasadena, California.

Eastern Vehicle Checkout Facility (EVCF)/Transportable Vehicle Checkout Facility (TVCF)—The EVCF/TVCF is an Air Force Space Control Network (AFSCN) ground station. It provides an S-band interface to AFSCN resources.

6.1.5 Payload Transport to Complex 36

The payload transport trailer is an air-ride suspension transporter dedicated to transporting the encapsulated spacecraft from the HPF to the launch complex. It has a self-contained gaseous nitrogen (GN_2) purge system and an environmental control system (ECS). One system is designated as prime and the other is designated backup depending on the anticipated ambient conditions during transport. These systems maintain a positive pressure of GN_2 or air in the payload fairing during the entire transport period. The transporter includes an environmental monitoring instrumentation system that provides continuous digital display of the payload fairing/ECS environment during transport.

6.2 ATLAS SPACE LAUNCH COMPLEX 36

The Atlas East Coast launch facility is Space Launch Complex 36 (SLC-36) (Fig. 6.2-1), located at CCAS. Major facilities include mobile service towers (MST), umbilical towers (UT), and the blockhouse. Figures 6.2-2 and 6.2-3 show diagrams of SLC-36A and SLC-36B, respectively.

Figure 6.2-1 Space Launch Complex 36 (SLC-36)

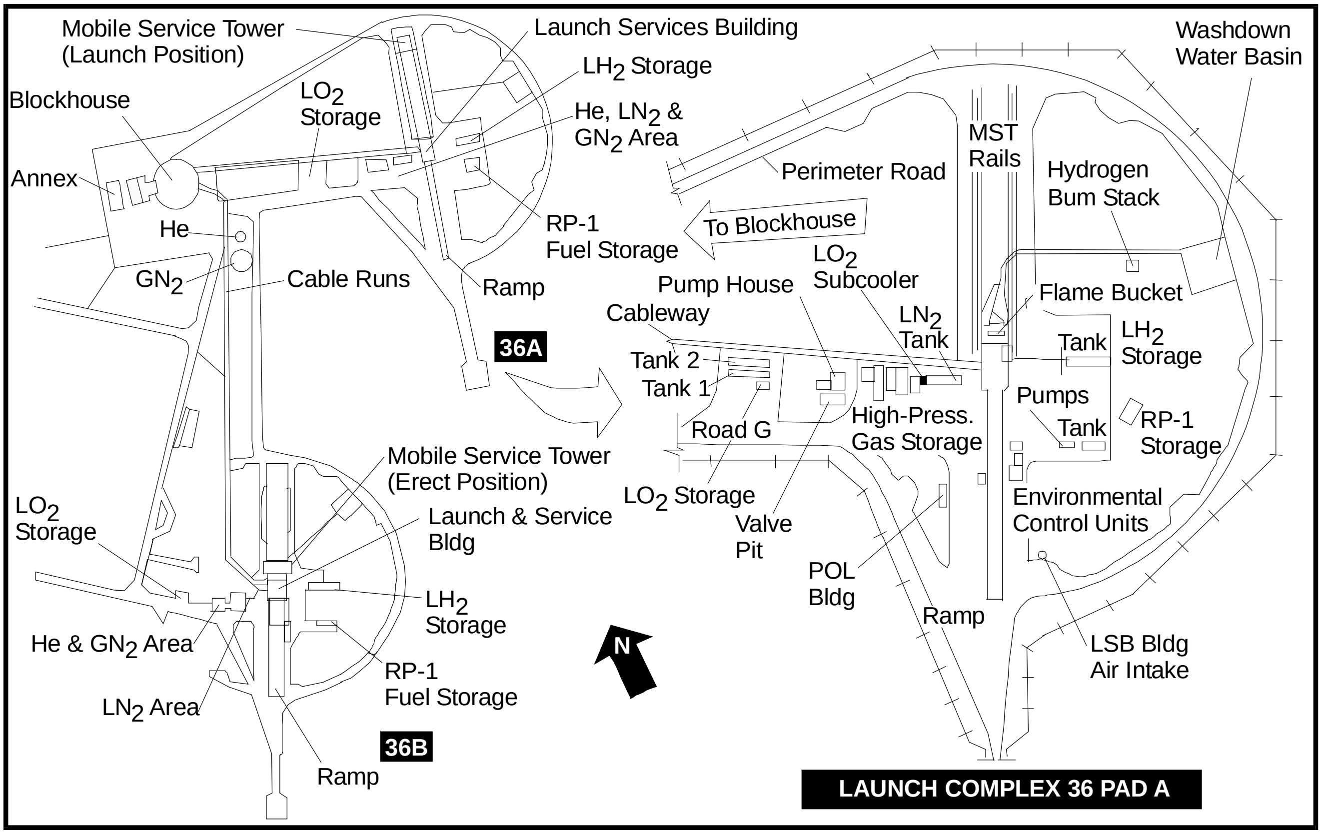


Figure 6.2-2 SLC-36A at CCAS

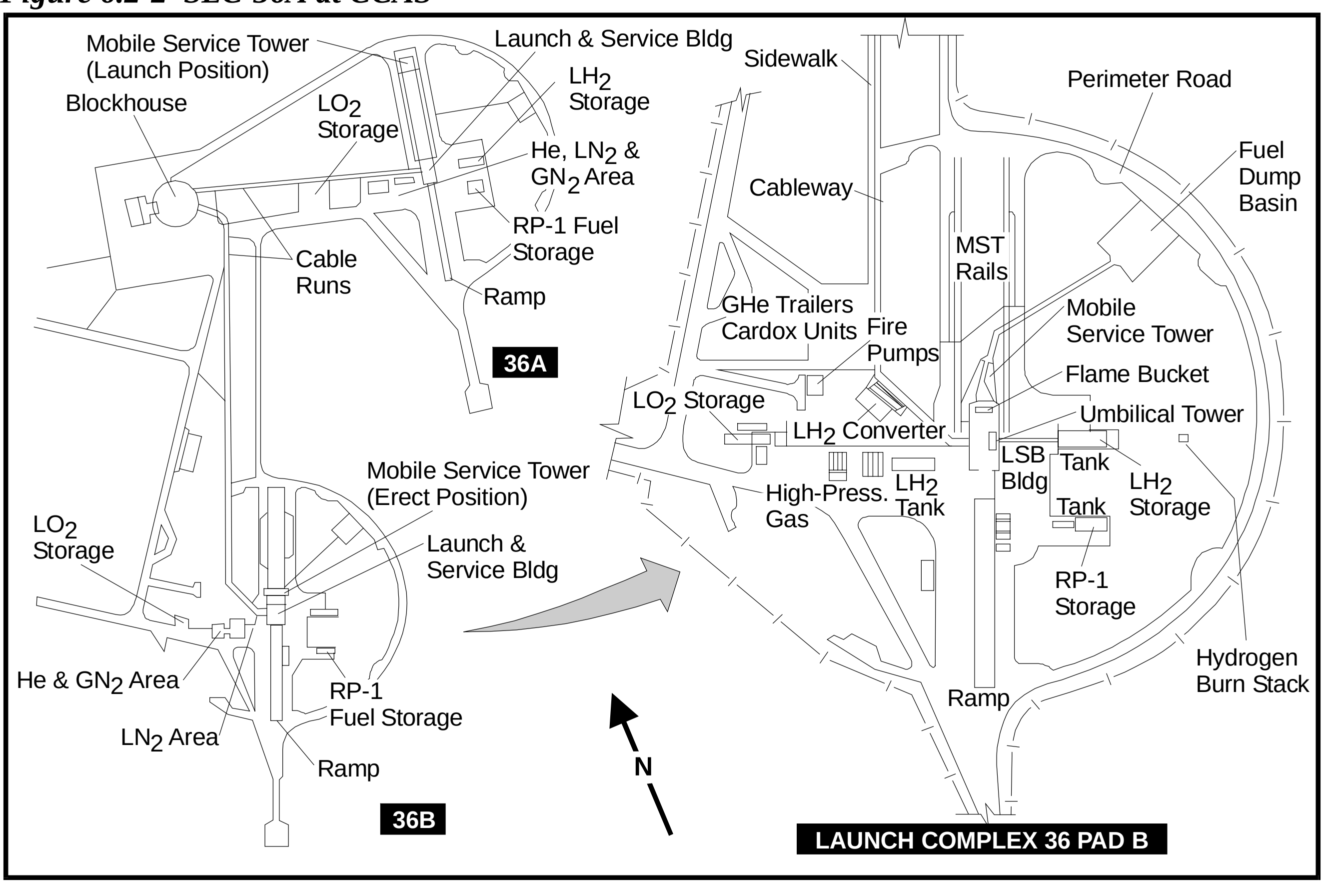


Figure 6.2-3 SLC-36B at CCAS

6.2.1 Mobile Service Towers

The SLC-36 MSTs (Figs. 6.2.1-1 and 6.2.1-2) are open steel structures with interior enclosures that contain retractable vehicle servicing and checkout levels/platforms. The primary functions of the MST are to:

1) Erect Atlas and Centaur and mate the encapsulated spacecraft;
2) Provide work areas for personnel and equipment during spacecraft mate and flight readiness checkouts;
3) Provide environmental protection to the spacecraft and launch vehicle;
4) Provide a capability for performing operational activities during the daylight, darkness, and inclement weather.

Both MSTs contain an electric, trolley-mounted overhead bridge crane used to hoist spacecraft, fairings, and the upper stage vehicle into position. The crane and the 36B MST are rated at 9,072 kg (10 tons). The 36A MST crane is rated at 18,140 kg (20 tons). Elevators serve all MST levels at 36A and 36B.

The entire MST assembly rests on four-wheel bogies that are electrically powered and provide service structure mobility along a rail system extending from the vehicle servicing (stand) position at the launch and service building to the stowed (launch) position.

The MST platforms are arranged so that the platform above acts as ceiling of the platform below.

Access to the MST is from ground level via platforms and stairways on east and west sides of the MST; from the stairs from the launch deck; or from the umbilical mast via four crossovers at MST Levels 8, 11, 13, and 17. Staircases rise to Level 4 on the east and west sides of the MST. From Level 4, a single staircase provides access to Level 20 on the north side of the structure. Two elevators, on the east and west sides of the MST, service MST Levels 1 through 18 and platform Levels 1 through 8. Emergency egress is by stair only.

The open framework of the MST is made of structural and tubular-shaped steel, which is bonded

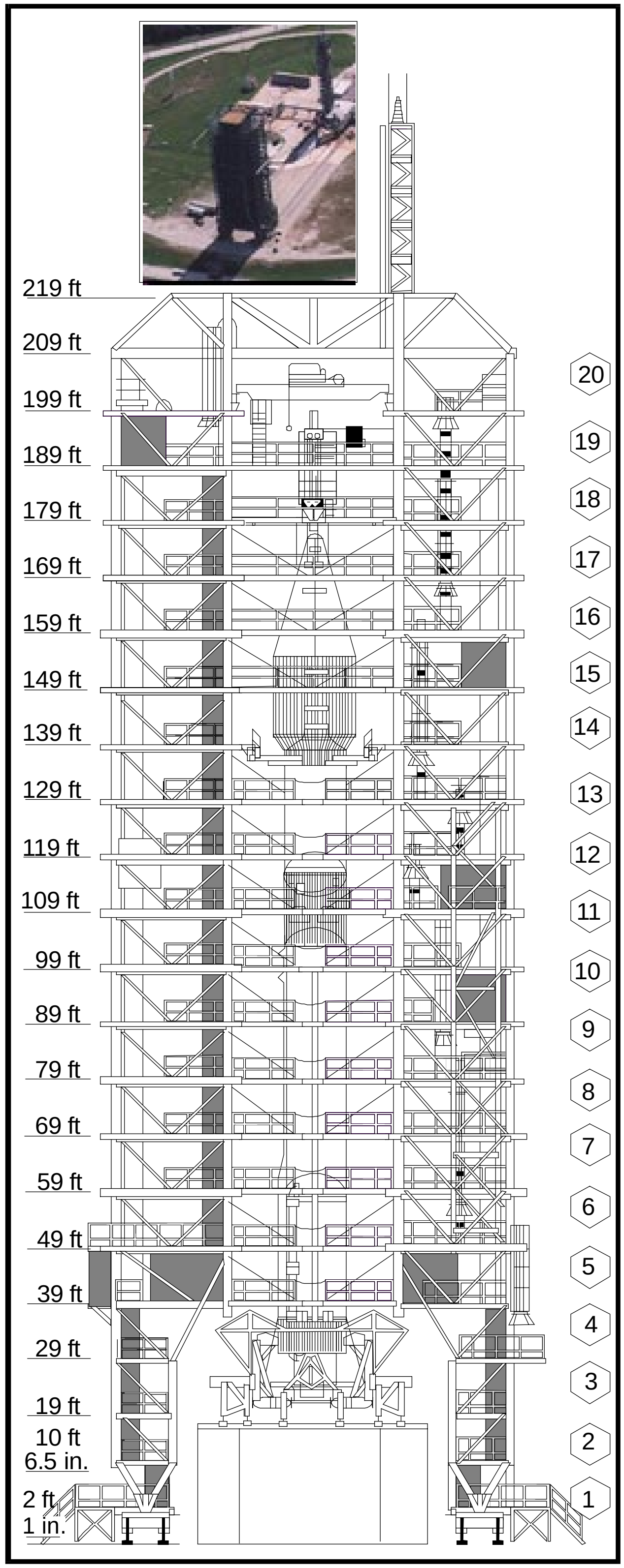


Figure 6.2.1-1 The SLC-36A MST provides access and environmental protection to the launch vehicle and spacecraft during prelaunch processing.

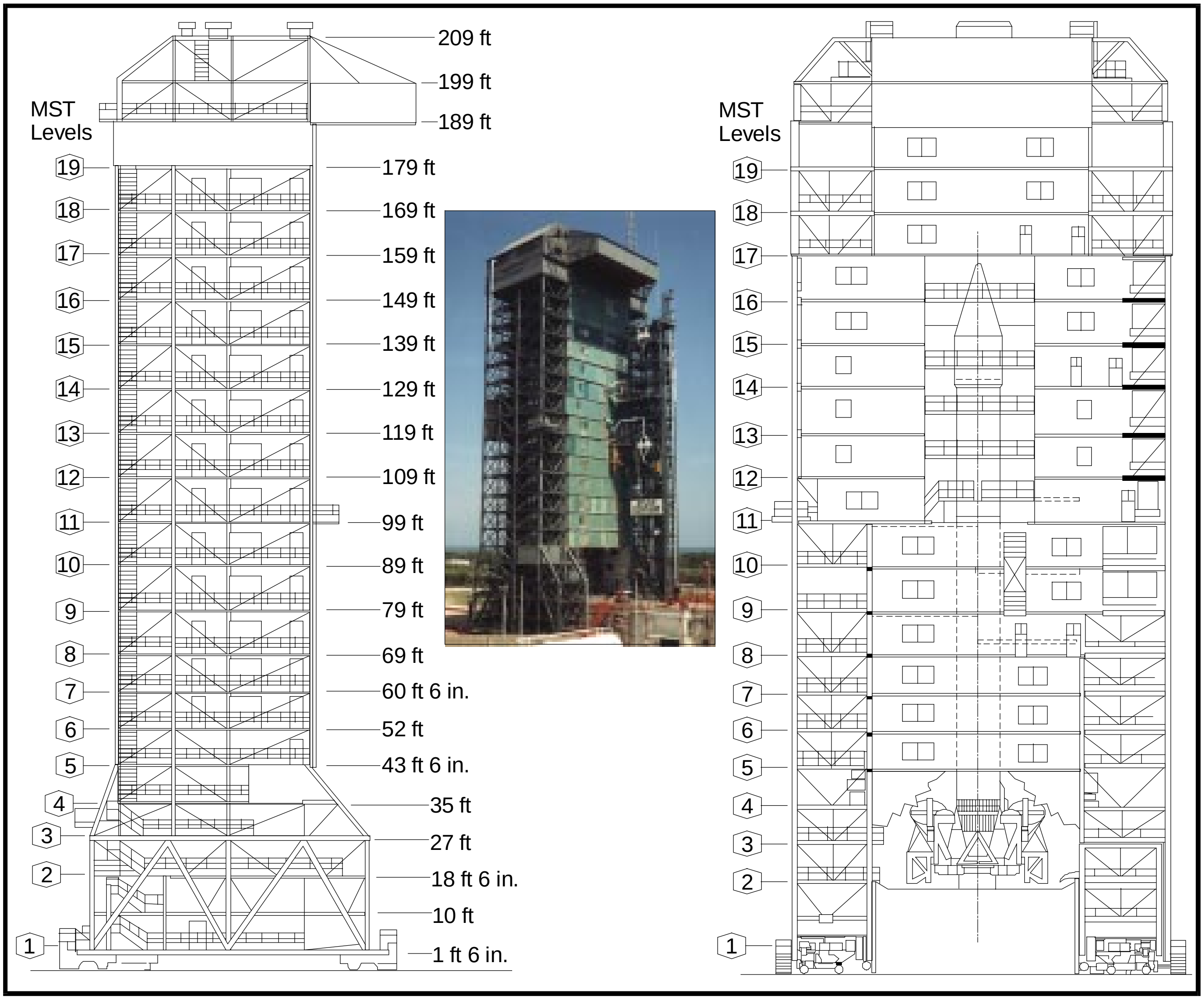


Figure 6.2.1-2 The SLC-36B MST

together and grounded. All platforms, access stands, and stairs are made of metal with nonslip surfaces. Sliding door panels protect the opening on the south side of the MSTs.

Spacecraft Work Areas—Access to the encapsulated spacecraft from the SLC-36B MST is provided at Levels 15 and 16 (Fig. 6.2.1-2) and at Levels 14 and 15 at the SLC-36A MST (Fig. 6.2.1-1). From the SLC-36B MST, Platforms 29 and 30 elevations approximately coincide with Levels 15 and 16. Platform 29 provides access to the aft area of the spacecraft via four large standard doors in medium-, large-, and extended-length large fairings. Additional access to the spacecraft can be provided from access platforms to mission-peculiar doors accessible from initial mate until fairing closeout.

Spacecraft Interface Panel—A special power panel is supplied from the critical power bus to ensure continual spacecraft support power. In case of commercial power failure, emergency power is generated at the complex by standby generator sets.

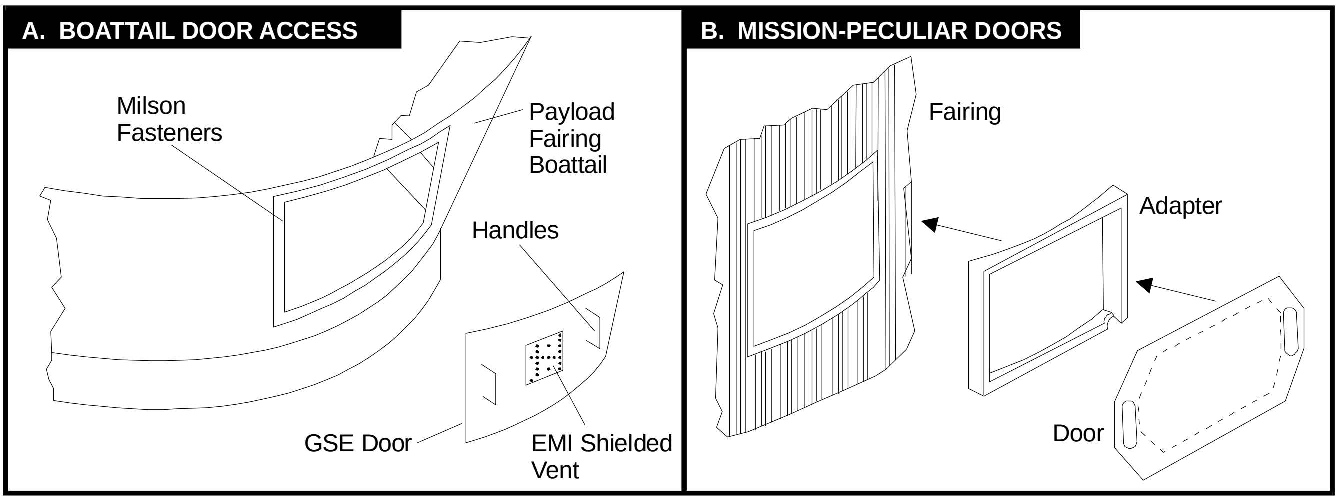
Spacecraft Access Doors—Access to the spacecraft is provided in the service tower by doors in the PLF cylinder and through standard doors in the PLF boattail. During payload integration operations, GSE doors are used when access to the spacecraft is not required. GSE doors are typically replaced with flight doors on Launch-1 day. A GSE adapter frame is used to limit use of the flight hardware. GSE doors provide contamination and electromagnetic interference (EMI) shielding (Fig. 6.2.1-3).

Spacecraft Access Platforms—Access through PLF cylinder doors may be provided by two different portable access platforms. The type used depends on the required reach from the PLF outer skin to the spacecraft item being serviced. One platform allows the technician access into the PLF cavity, are portable (Fig. 6.2.1-4), but are secured to the service tower deck when in use and during storage. Access through the PLF boattail doors is provided by the portable platforms shown in Figure 6.2.1-5.

Communications—Four types of communications equipment are installed on the MST: voice, CCTV, RF, and a public address (PA) system.

Blackphones and hazardous TOPS units comprise the voice communications system. TOPS units are located on all work levels. Blackphones are installed on spacecraft work levels.

CCTV cameras are installed on MST Level 29 to monitor launch vehicle and spacecraft processing activities.



6.2.1-3 Spacecraft access is provided through boattail and mission-peculiar doors.

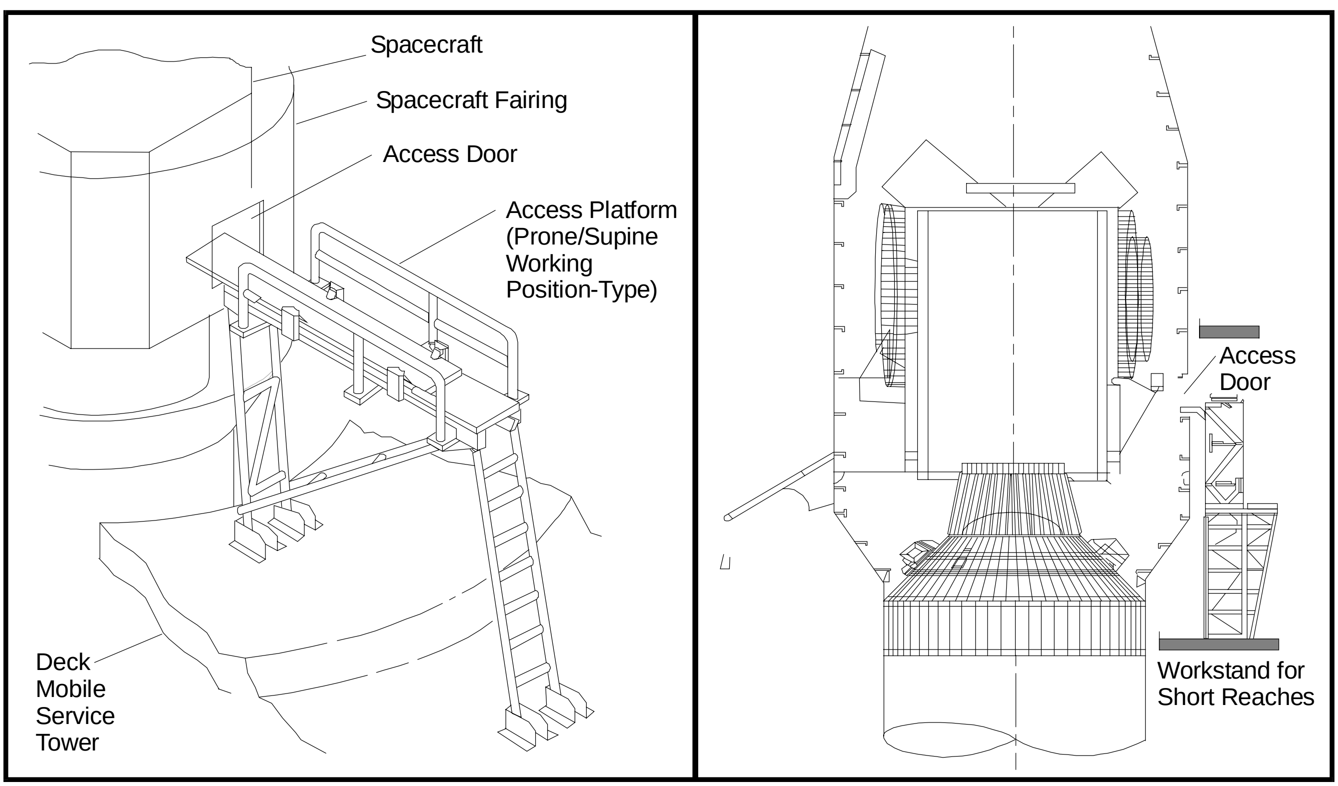


Figure 6.2.1-4 Mobile platforms provide spacecraft access if required.

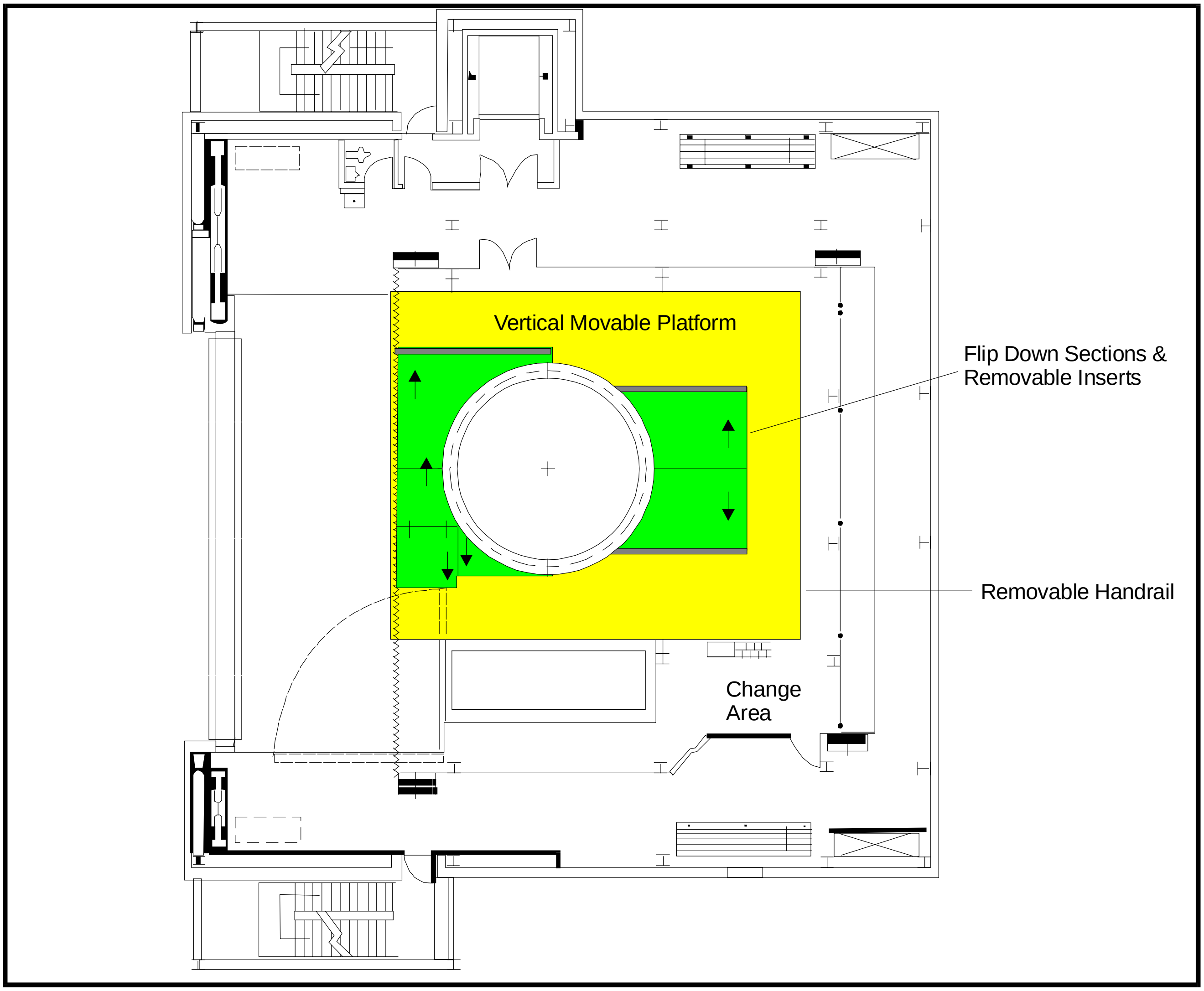


Figure 6.2.1-5 MST Movable Platforms (Typical)

S-band and Ku-band equipment may be installed on the MST per mission-peculiar requirements of the launch vehicle and/or spacecraft programs. RF cabling and reradiating antennas are available for S-band, C-band, and Ku-band telemetry.

PA speakers are installed inside and outside the enclosure. The PA system is the primary means of alerting personnel and making routine announcements. Three PA microphone stations are available. The primary station is in the blockhouse, secondary stations are in and on the deck of the Launch Services Building (LSB).

Vehicle Restraint System—SLC 36 MSTs have a vehicle restraint system that enables the mated nose fairing/launch vehicle stack to be placed into a secure "stretch" configuration during adverse weather conditions. This system is designed to withstand 240-kph (150-mph) winds.

6.2.2 Umbilical Tower (UT)

The UT (Fig. 6.2.2-1) is a fixed structural steel tower extending above the launch pad. Retractable service booms are attached to the UT. The booms provide electrical power, instrumentation, propellants, pneumatics, and conditioned air or gaseous nitrogen (GN_2) to the vehicle and spacecraft. These systems also provide quick-disconnect mechanisms at the respective vehicle interface and permit boom retraction at vehicle launch. A payload umbilical junction box interconnects the spacecraft to the electrical GSE. Limited space is available within this junction box to install spacecraft-unique electrical GSE.

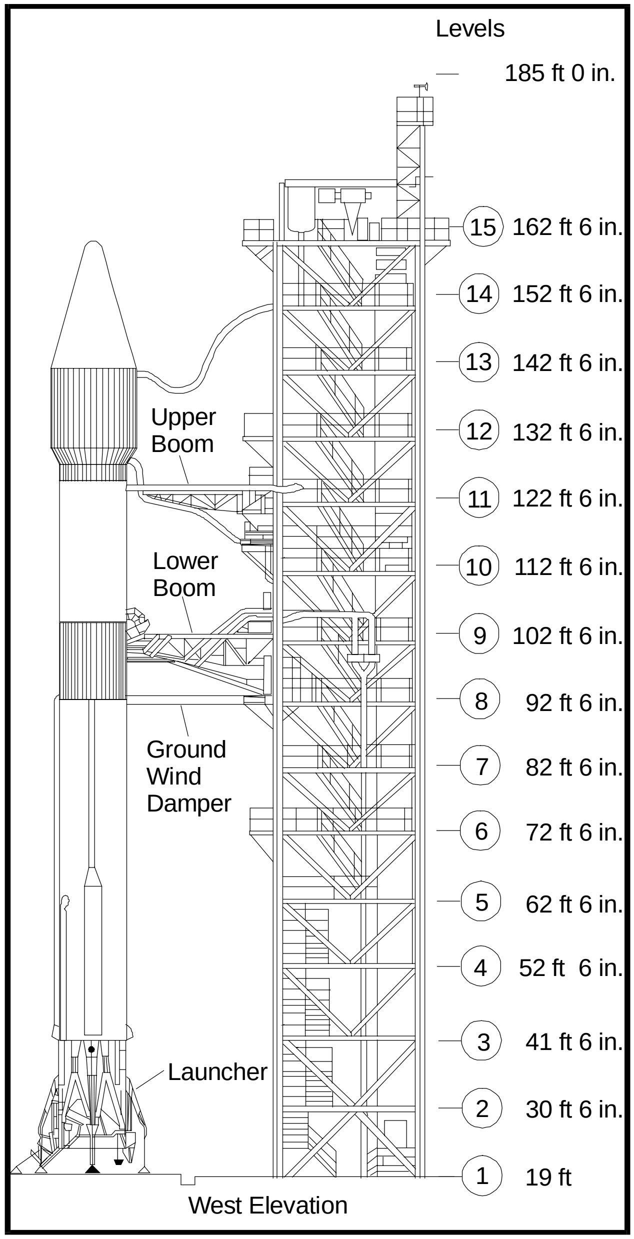


Figure 6.2.2-1 The SLC-3B umbilical tower retractable service arms allow quick disconnect and boom retraction at vehicle launch.

6.2.3 Launch Pad Ground System Elements

The launch complex is serviced by GN_2, gaseous helium (GHe), and propellant storage facilities within the complex area. ECSs exist for both the launch vehicle and the spacecraft. Detailed descriptions of the capabilities of these systems to provide spacecraft activity are in Section 4.2, Spacecraft to Ground Equipment Interfaces.

6.2.4 Blockhouse

The blockhouse (Figs. 6.2.4-1 and 6.2.4-2) is the operations and communications center for the launch complex. It contains all necessary control and monitoring equipment. The launch control, electrical, landline instrumentation, and ground computer systems are the major systems in this facility.

The launch control provides consoles and cabling for control of the launch complex systems. The landline instrumentation system (coupled with the CCTV system) monitors and records safety and performance data during test and launch operations. The ground computer system consists of redundant computer-controlled launch sets (CCLS) and a telemetry ground station. The CCLS provides control and monitoring of the vehicle guidance, navigation, and control systems and monitors vehicle instrumentation for potential anomalies during test and launch operations.

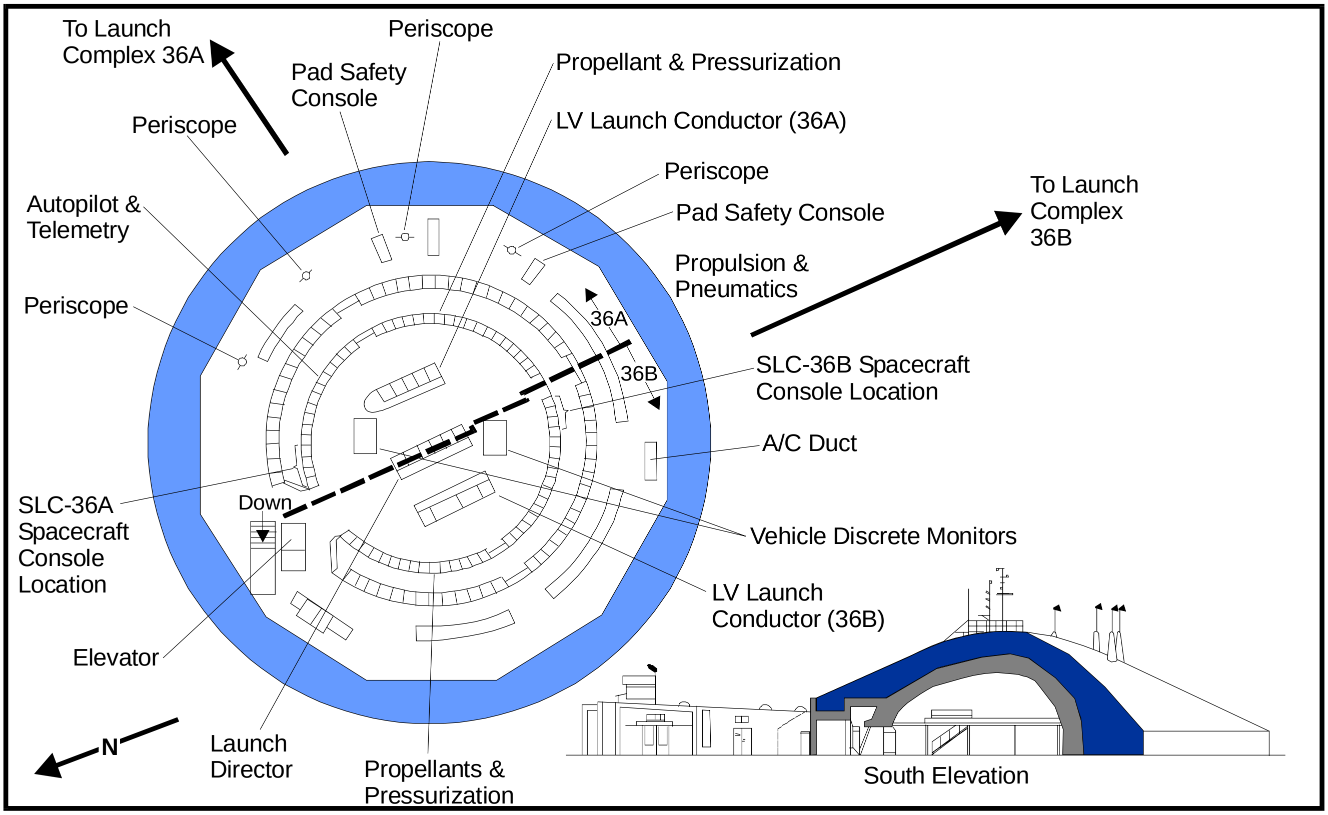


Figure 6.2.4-1 Test and launch operations are controlled and monitored from the blockhouse.

Figure 6.2.4-2 Aerial View of Common SLC-36 Blockhouse

6.2.5 Launch and Service Building (LSB)

The LSB provides the means for erecting the launch vehicle, interconnecting electrical wiring between the umbilical tower and the blockhouse, and locating spacecraft remote GSE. It also provides the launching platform. The LSB is a two-story structure (Fig. 6.2.5-1). LSBs at SLC-36A and SLC-36B are similar in function.

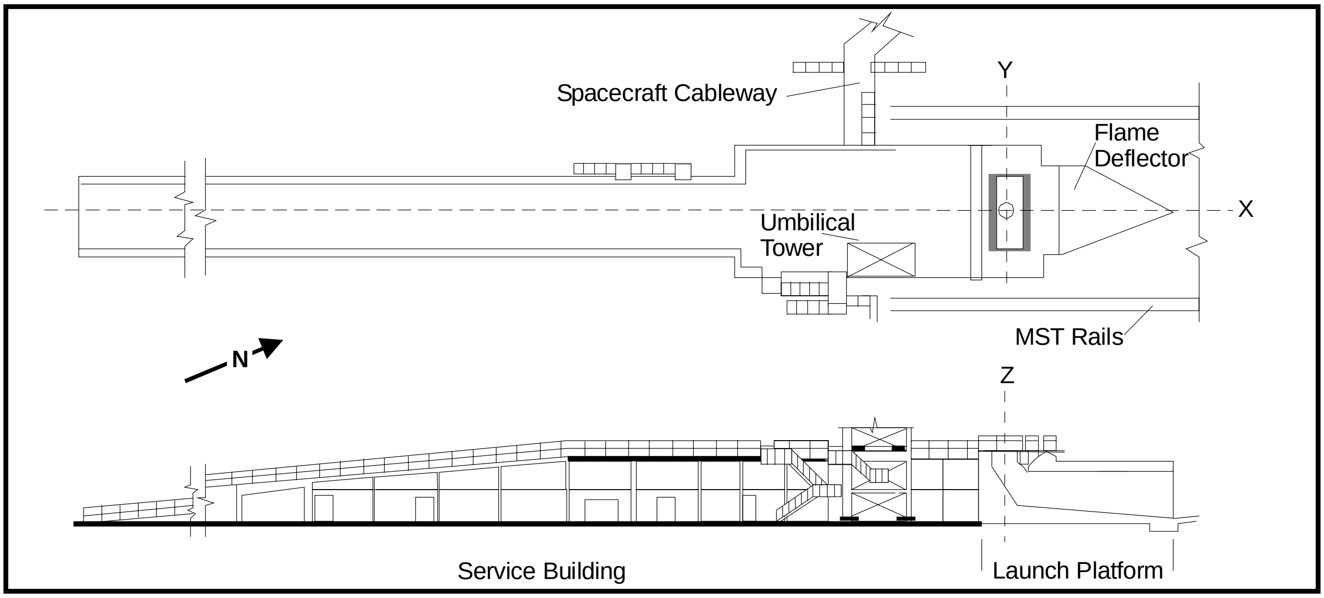


Figure 6.2.5-1 SLC-36 Launch and Service Building (LSB)

6.2.6 Customer Support Building

Lockheed Martin provides an Atlas launch services customer technical support building (Fig. 6.2.6-1). This building includes private office space, workspace, and conference rooms for customer management during spacecraft launch site processing. Reproduction machines, administrative telephones with long-distance access, and facsimile are available. The facility is conveniently located between the Astrotech and Complex 36 facilities.

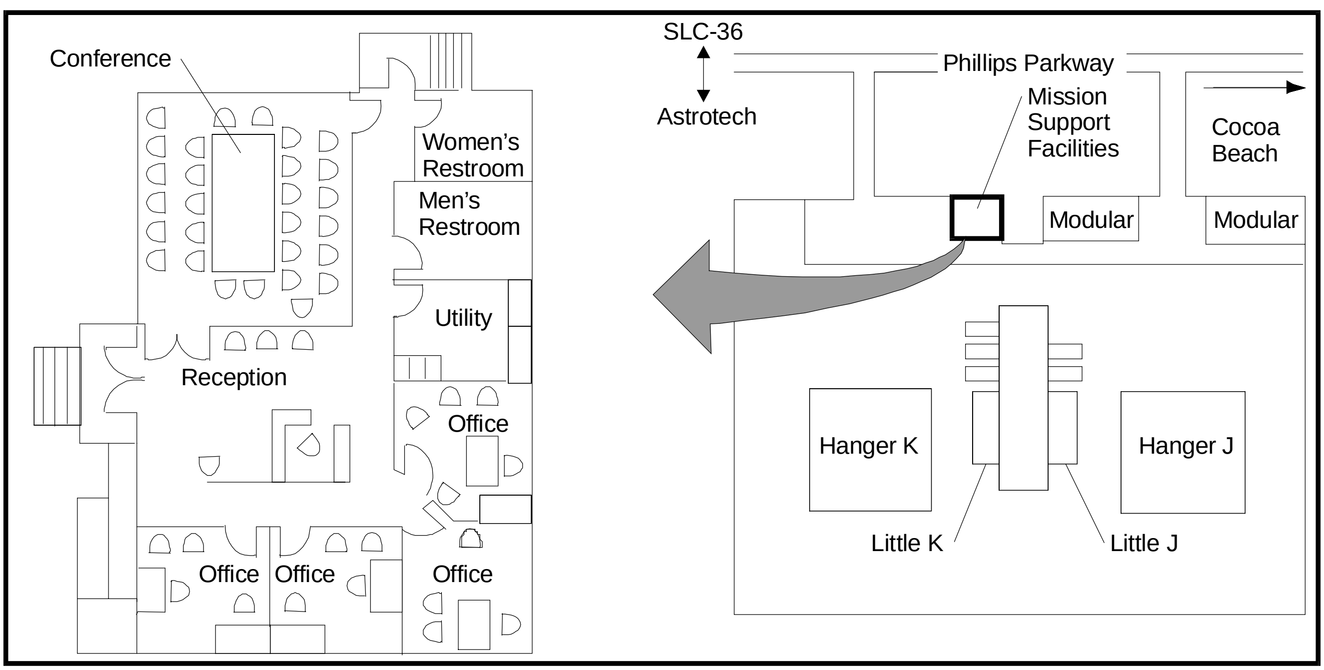


Figure 6.2.6-1 CCAS Launch Services Customer Support Facility

6.2.7 Planned SLC-36 Enhancements

SLC-36A—Launch Complex 36A will be upgraded to support Atlas IIAS launches. These modifications will be complete by mid-1997 and include a new launcher structure and an enhanced holddown release system (pyrovent valve instead of the current Barksdale valve).

SLC-36B—Launch Complex 36B will be modified to support the Atlas IIAR vehicle configuration. These modifications will be complete by late 1998 and include a MST height increase of at least 20 ft, a new 20-ton bridge crane, UT changes for a taller Atlas, and launcher modifications for the new IIAR vehicle to ground interfaces.

6.3 VAFB SPACECRAFT FACILITIES

With the expansion of Space Launch Complex 3 (SLC-3) at VAFB, California, to support the launch of the Atlas/Centaur vehicle, efforts are underway by Lockheed Martin and U.S. government Atlas launch services users on definitization and development of payload processing facilities capable of handling Atlas/Centaur class payloads.

One candidate facility in operation at VAFB is operated by Astrotech Space Operations, Inc., the operators of our primary PPF facility at CCAS. This facility is and has supported the final checkout of smaller class payloads. Depending on VAFB launch interest and U.S. government payload processing requirements for Atlas/Centaur class payloads, the Astrotech/VAFB facility may offer compatibility with spacecraft contractor and Lockheed Martin use requirements.

Figure 6.3-1 shows the location of various facilities at VAFB.

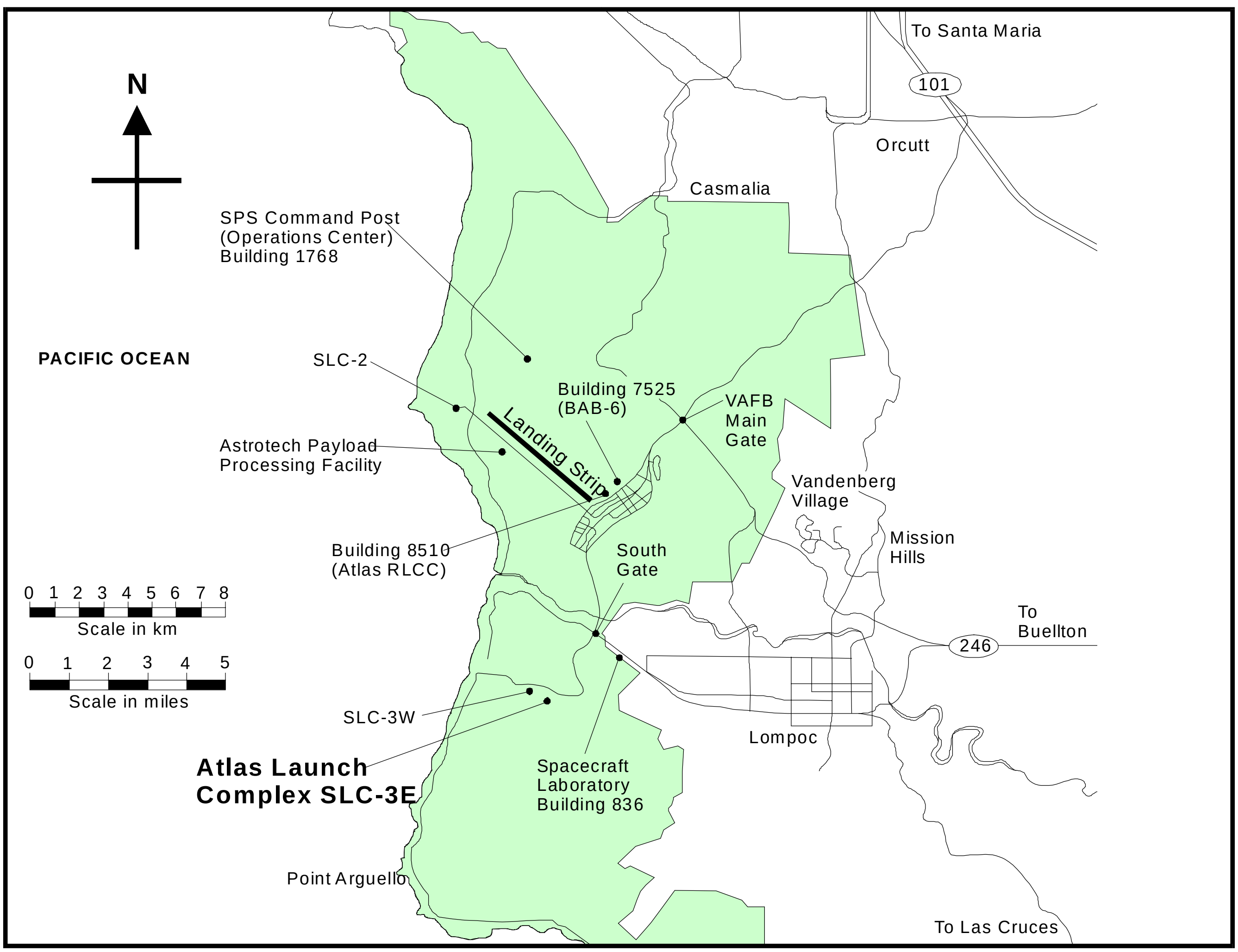


Figure 6.3-1 VAFB Facilities

6.3.1 Astrotech PPF/VAFB

The Astrotech/VAFB PPF can be used for all payload preparation operations, including liquid propellant transfer, SRM and ordnance installations, and payload fairing encapsulation. The facility is near the VAFB airfield, approximately 12 km from SLC-3E. The facility, a diagram of which is shown in Figure 6.3.1-1, contains the following:

1) One airlock;
2) Two high bays;
3) Three low bays;
4) Control rooms (one per high bay);
5) Auxiliary control room
6) Two walk-in coolers.

In addition, a facility/customer support office (Fig. 6.3.1-2) is available and is shared by Astrotech resident professional and administrative staff and customer personnel. Shared support areas include office space, a conference room, a copier, a facsimile, and amenities.

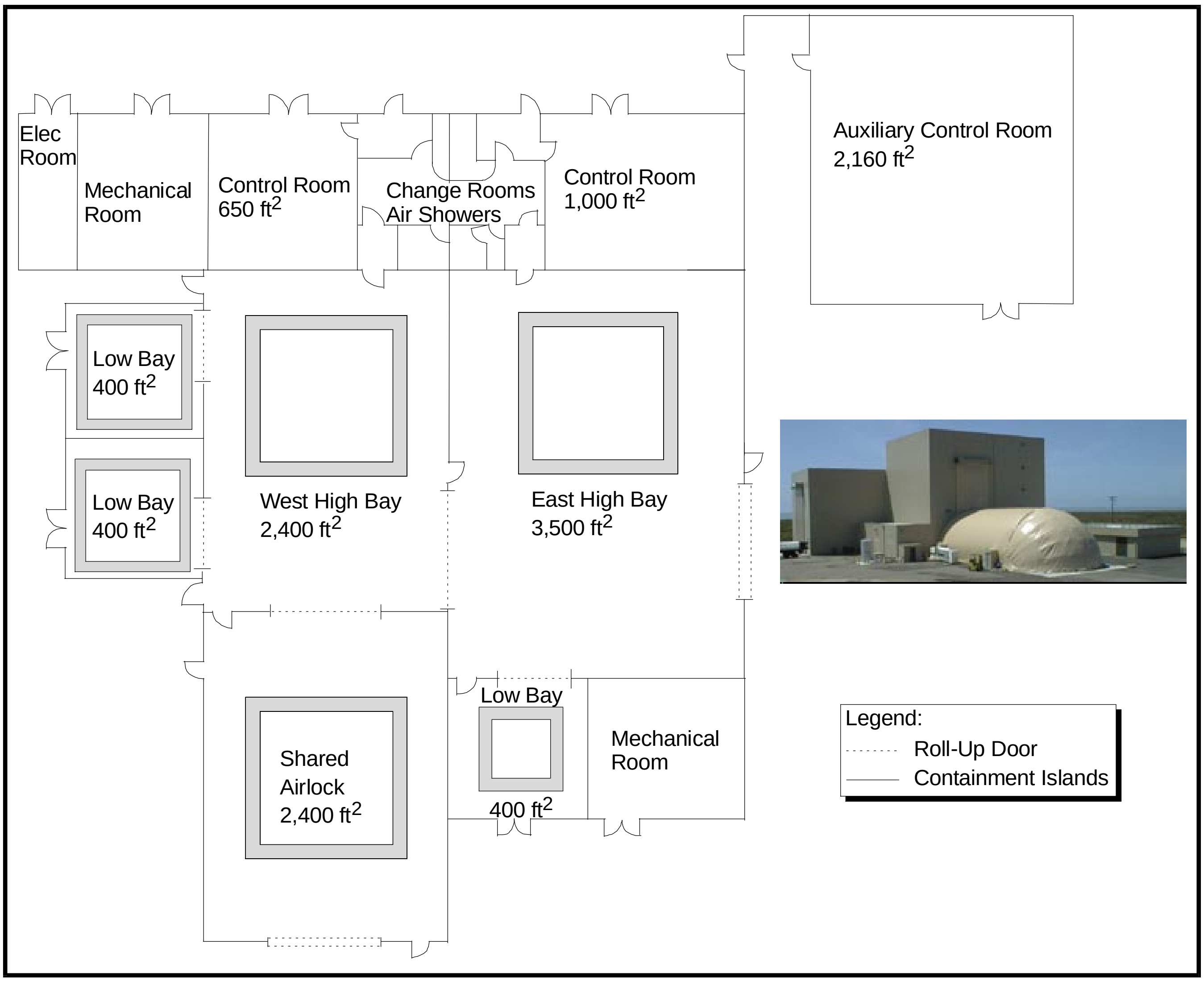


Figure 6.3.1-1 Astrotech/VAFB Payload Processing Facility Layout

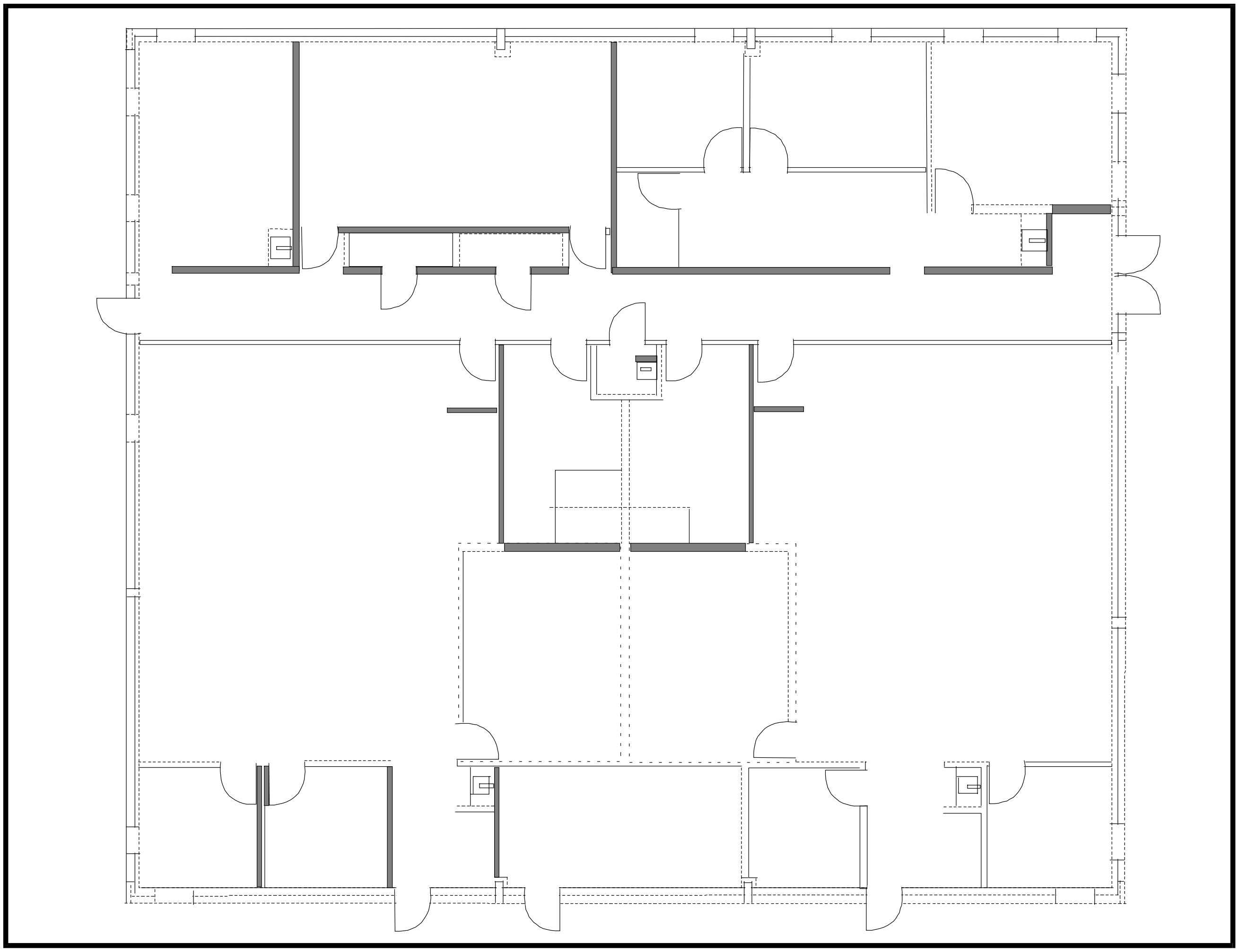


Figure 6.3.1-2 Technical Support Building

Spacecraft Services—A full complement of services can be provided at Astrotech to support payload processing and integration.

***Electrical Power*—**The Astrotech/VAFB facility is served by 480-Vac/three-phase commercial 60-Hz electrical power that can be redistributed as 480-Vac/three-phase/30-A, 120/208-Vac/three-phase/60-A, or 125-Vac/single-phase, 20-A power to all major areas within the facility. Standard power is backed up by a diesel generator during critical testing and launch periods.

***Telephone and Facsimile*—**Astrotech provides all telephone equipment, local telephone service, and long-distance access. A Group 3 facsimile machine is available and commercial telex service can be arranged.

***Intercommunication Systems*—**The Operational Voice Intercommunication System provides internal intercom and a link to other facilities as VAFB. TOPS nets are available throughout the PPF. TOPS provides operational communications to other government facilities at VAFB. TOPS allows entrance into the government voice net for direct participation during flight readiness tests and launch countdowns. A paging system is also available throughout the complex.

***Closed-Circuit Television*—**Five CCTV cameras are located in the PPF. Two are located in the processing high bay and one is in the airlock. CCTV can be distributed within the Astrotech/VAFB complex to any desired location, including the Auxiliary Control Room and Technical Support Building.

Remote Spacecraft Control Center—Astrotech has the capability to link remote ground stations (voice and data) between Astrotech and VAFB resources.

Temperature/Humidity Control—A 2,000-cubic feet per minute (cfm) humidity, ventilation, and air-conditioning (HVAC) control system provides reliable air conditioning for cleanroom operations and is capable of maintaining temperature at 21 ±1.1°C (70 ±2°F) with a relative humidity of 45 ±10%. Positive pressure is maintained in all cleanroom areas. Air is circulated through the high-efficiency particulate air (HEPA) filter bank at 3.5-4 room changes per hour. Differential pressure can be maintained between control rooms and cleanrooms to prevent toxic vapor leaks into adjacent areas.

Compressed Air—Compressed air is supplied by a stationary, two-stage, rotary tooth compressor, which delivers oil-free air for breathing air, shop air, and pallet air applications. The 30-horse power (hp) compressor provides 100 cfm at 125 pounds per in.^{2} (psi). Breathing air purifiers meet current Occupational Health and Safety Administration (OSHA), National Institute for Occupational Safety and Health (NIOSH), and Environmental Protection Agency (EPA) guidelines for production of Grade D breathing air.

Security and Emergency Support—Physical security is provided by a locked gate and two S&G locked entry doors. All doors providing access to closed areas are alarmed with remote readout at the VAFB Law Enforcement Desk. After receipt of an alarm, USAF armed security will respond within 15 minutes. The alarm system is designed to allow completely segregated operations in the two processing high bays.

6.3.2 Other Spacecraft Facilities

Other USAF, NASA, or privately operated facilities may become available for Atlas class payload processing in the future. This document will be updated as more information becomes available.

6.3.3 Payload Encapsulation and Transportation to SLC-3

During final checkout and propellant loading of the spacecraft, Lockheed Martin will require use of the PPF for approximately 30 days to receive and verify cleanliness of the fairing and to encapsulate the spacecraft.

After encapsulation, Lockheed Martin will transport the encapsulated payload to SLC-3E and mate the spacecraft/fairing assembly to the launch vehicle. Transportation hardware and procedures will be similar to those used at our East Coast SLC-36 launch site. Postmate spacecraft testing can be performed from GSE located on the MST, in the LSB payload user's room, or through connectivity to the VAFB fiber-optics transmission system (FOTS) from offsite locations.

6.4 ATLAS SPACE LAUNCH COMPLEX-3 (SLC-3)

SLC-3 at VAFB in California has supported the launch of Atlas vehicles since the 1970s. Through March 1995, the SLC-3W complex supported the launch of refurbished Atlas E space launch vehicles to deliver payloads such as the National Oceanic and Atmospheric Administration (NOAA) and Defense Meteorological Satellite Program (DMSP) polar weather satellites to low-Earth orbit (LEO). In 1992 the USAF has contracted with Lockheed Martin to convert the inactive Complex 3E site to support the launch of Atlas/Centaur to orbits not attainable from the East Coast CCAS launch site (Fig. 6.4-1). The initial operational capability of SLC-3E was late 1997. This section discusses facilities available to support spacecraft and launch vehicle integration and launch.

SLC-3 is located on South VAFB, 11 km (7 miles) from the base industrial area on North VAFB and approximately 6.5 km (4 miles) from NASA Building 836. For reference, locations of SLC-2, SLC-3W, and Buildings 8510 and 7525 are shown in Figure 6.3-1. Major facilities at SLC-3 include the MST, LSB, UT, and a launch operations building (LOB).

The upgrade of SLC-3E to an Atlas IIAS configuration included relocation of the Atlas launch control center from the existing blockhouse to a remote launch control center (RLCC) located in Building 8510 on North VAFB. Building 7525, also located on North VAFB, is used for launch vehicle receiving and inspection. High-volume, high-pressure GN_2 is supplied to the SLC-3E site.

Figure 6.4-2 provides a view of the SLC-3E launch complex. The Lockheed Martin launch site activation to supported a mid-1998 launch capability.

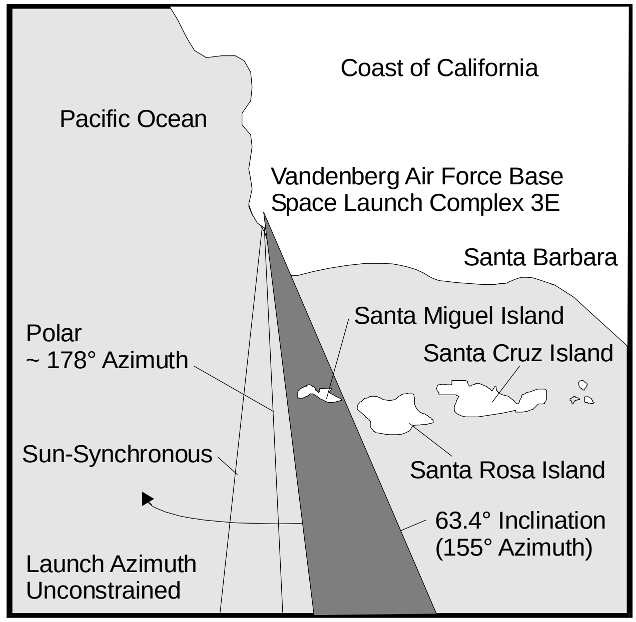


Figure 6.4-1 Available Launch Azimuths from Complex 3E

Figure 6.4-2 SLC-3E modifications for Atlas IIA/IIAS launches are complete.

6.4.1 Mobile Service Tower (MST)

The MST is a multilevel, movable, totally enclosed steel-braced frame structure for servicing of launch vehicles and payloads (Fig. 6.4.1-1). A truck system on rails is used for transporting the MST from its park position at a point approximately 76.2-m (250-ft) south of the LSB to its service position over the launcher. The tower is secured in place with a seismic tie-down system at both tower positions. The MST is normally in place over the launch pad except during major systems tests and before cryogenic tanking during the launch countdown sequence.

The MST has 19 levels. A hammerhead overhang is incorporated at the top of the structure on the north side to allow a 18,150-kg (20-ton) overhead bridge crane on Level 19 to move outside the MST for erection and mating of the Centaur and payload. A 31,750-kg (35-ton) torus crane under Level 8 is used for erecting and attaching the solid rocket boosters (SRB) to the Atlas booster. Erecting the Atlas IIAS vehicle is accomplished by an erection hoist. The MST provides access to the Atlas booster, the Centaur upper stage, and the payload. It also provides a lighted, weather-protected work area for erection, mating, and checkout of flight vehicle.

In addition to external siding, the MST incorporates an environmentally controlled area (ECA) around the vehicle on Levels 8 through 15 to protect the Centaur and payload.

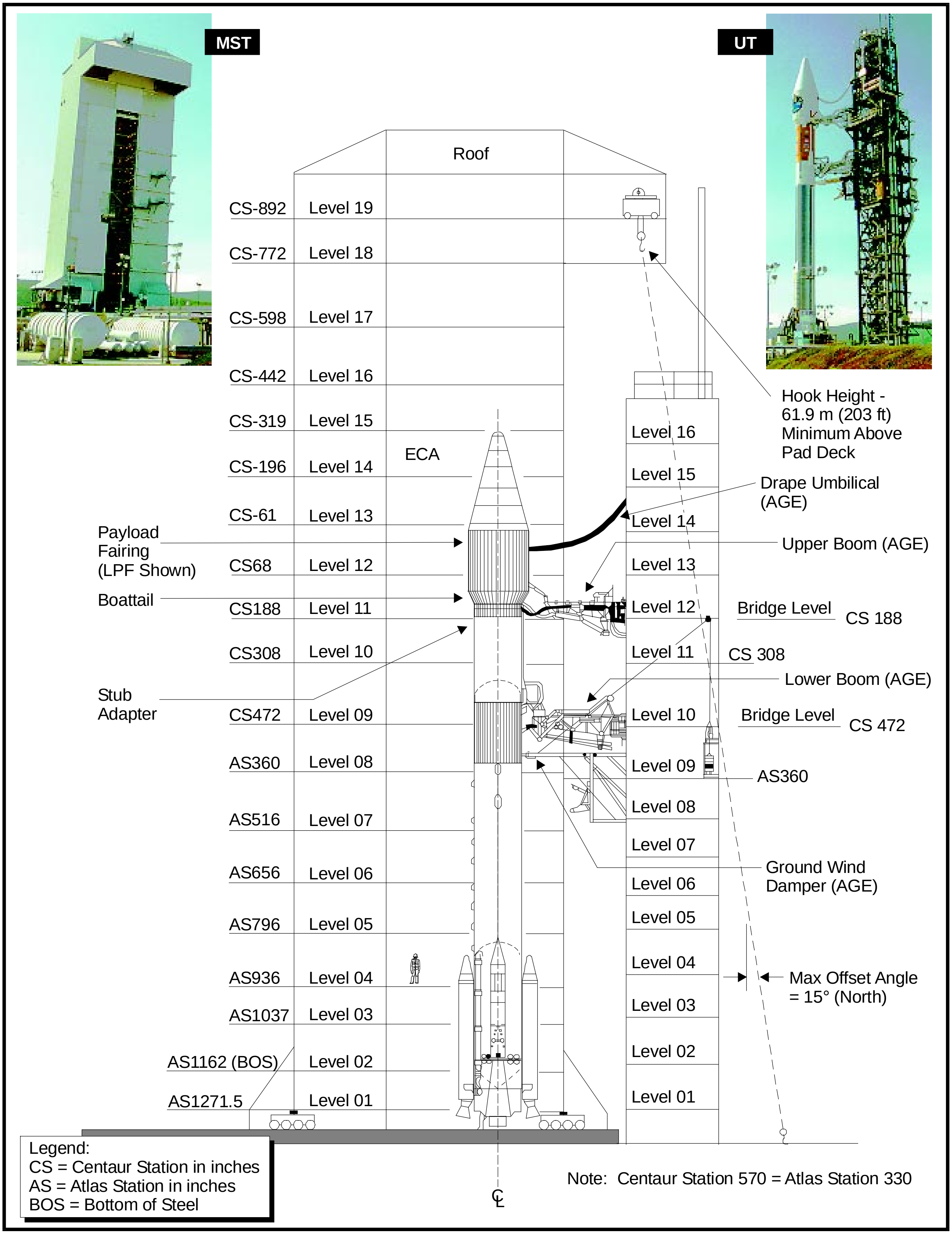


Figure 6.4.1-1 SLC-3E MST and UT

6.4.2 Umbilical Tower (UT)

The UT is a steel structure with 16 levels (Fig. 6.4.1-1). The UT supports two retractable umbilical booms, a draped umbilical, and a ground wind damper. The draped umbilical is used to supply conditioned air to the payload via its connection to the PLF. The umbilical tower supports power cables, command and control cables, propellant and gas lines, monitoring cables, and air-conditioning ducts routed from the LSB pad deck to appropriate distribution points.

6.4.3 Launch and Service Building

The LSB is a reinforced concrete and steel structure that is the platform on which the Atlas family of vehicles is assembled, tested, and launched. The top of the LSB, or LSB pad deck, provides support for the Atlas launcher and the MST while it is in the service position. The LSB pad deck also is a support structure for the UT that has supporting columns extending down through the upper-level pad deck and lower-level foundation into the ground. The LSB provides a protective shelter for shop areas, storage, locker rooms, air-conditioning equipment, electrical switch gear, instrumentation, fluid and gas transfer equipment, launch control equipment, and other launch-related service equipment.

The LSB equipment serves as a front end for all aerospace ground equipment (AGE) and vehicle control functions. This equipment issues commands as requested by operators, provides safing when operator connections are broken, and acquires data for monitoring of all pad activities.

The LSB also contains a payload user's room, which is electrically interconnected to MST Level 11 (with capability to route cabling from Level 11 to Levels 12 through 15) and the T-O umbilical. Capability also exists to connect the user's room to the FOTS for connectivity to offsite locations.

6.4.4 Launch Operations Building (LOB)

The LOB is an existing reinforced concrete and steel structure that provides 24-hour launch complex safety monitoring and control. The LOB provides 24-hour monitoring for critical systems and command and control capabilities except during hazardous operations, when responsibility is transferred to the RLCC. Systems that are monitored from the LOB include the environmental control system, the fire and vapor detection system, and the fire suppression and deluge system.

6.4.5 SLC-3E Payload Support Services

Electrical interfaces exist in the LSB payload user's room and the MST on Levels 11 and 15. This power is available in 120-V 20-A, 208-V 30-A, and 208-V 100-A technical power. Critical technical power circuits, 120-V 20-A and 208-V 30-A, are also provided and are backed up by uninterruptible power systems.

To support spacecraft testing while in the MST, GN_2 and GHe support service are supplied as part of the facility on MST Levels 11, 12, and 13. Type 1, Grade B, GN_2 per MIL-P-27401 is supplied through a 2-micron nominal 10-micron absolute filter in the pressure ranges of 0-100, 0-400, 1,500-3,600, and 2,500-5,000 psig. Type 1, Grade A, GHe per MIL-P-27407 is also supplied through a 2-micron nominal 10-micron absolute filter in the pressure ranges of 0-100 and 2,500-5,000 psig. Both clean gas and contaminated gas vent systems exist on MST Levels 11, 12, and 13. Contingency offload of spacecraft propellants is supported via the facility spacecraft propellant deservicing system. Propellant deservicing interfaces exist on MST Levels 11, 12, and 13 with ground interfaces at the propellant deservicing pad, with portable fuel and oxidizer scrubbers connected to the MST contaminated vent system. A SLC-3E breathing air system is available in the MST on MST Levels 11, 12, and 13, at the propellant deservicing pad, and at the scrubber pad to support self-contained atmospheric-protective ensemble (SCAPE) operations required for spacecraft processing.

A payload user's room is provided in the LSB to support spacecraft testing. The user's room is electrically connected to MST Level 11 by an array of payload support cables. The support cables are terminated at a connector interface panel both in the user's room and in the ECA on MST Level 11. Cable trays and cable passage ways are provided on MST Levels 11, 12, 13, 14 and 15 to route cabling from the connector interface panel on Level 11 to Levels 12-15. In addition, the LSB payload user's room can be electrically connected to the VAFB FOTS to provide connectivity to offsite locations.

6.4.6 Remote Launch Control Center (RLCC)

The RLCC (Fig. 6.4.6-1) is the focal point for launch site test monitoring and recording. The RLCC supports routine daily vehicle and AGE processing activities and total monitor and control over hazardous operations requiring launch site evacuation (i.e., wet dress rehearsal and launch). AGE systems in the RLCC (with interconnectivity between the RLCC and the LSB and LOB at SLC-3E) provide command control and monitoring of the overall launch control system. The RLCC also provides RLCC-to-SLC-3E communications, launch site control interfaces between the CCLS and launch vehicle and AGE, and, through the safe/arm and securing unit (SASU), control of safety-critical functions independent of the CCLS and RLCC-to-SLC-3E AGE interface links.

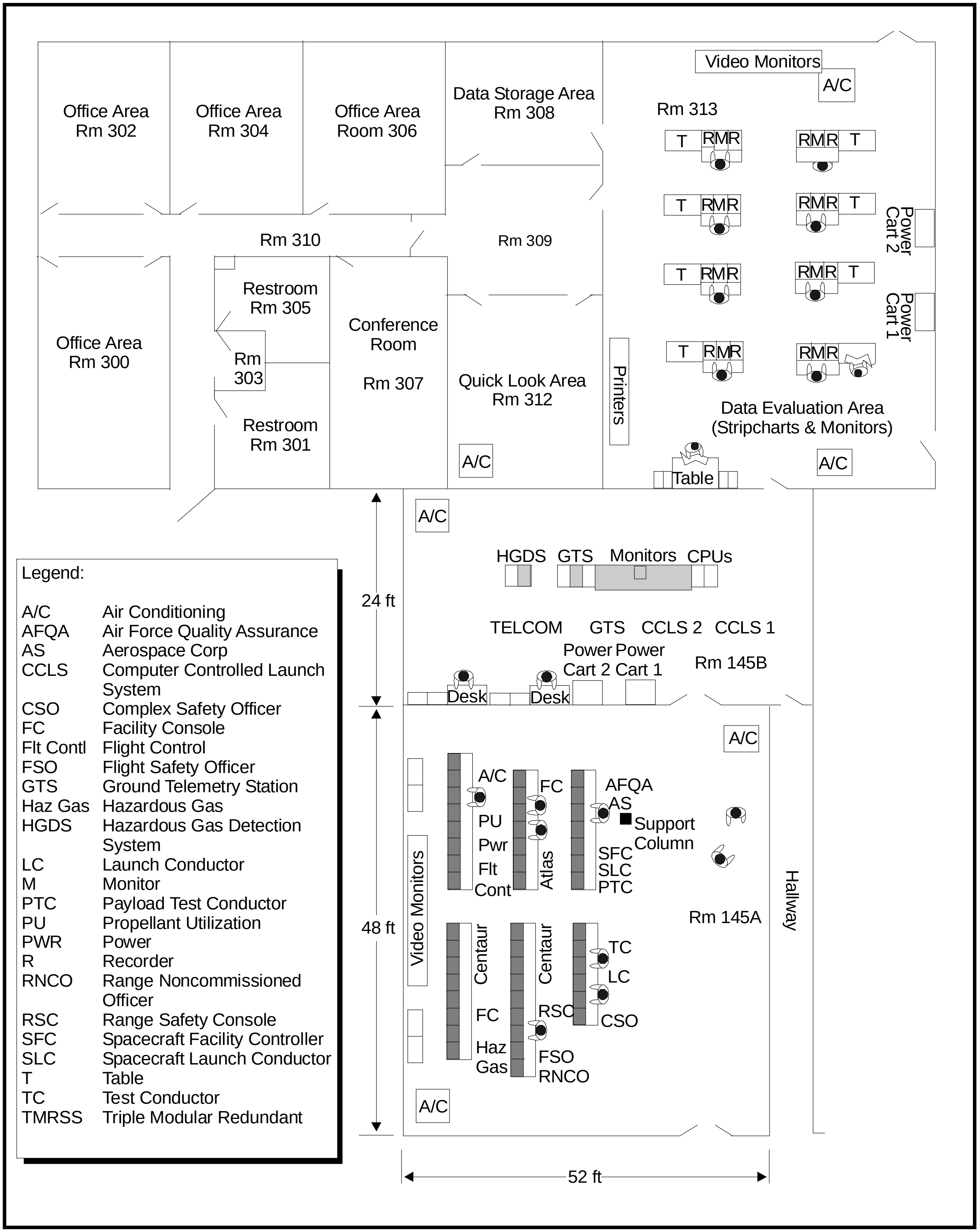


Figure 6.4.6-1 Atlas RLCC Area Within Building 8510

7.0 LAUNCH CAMPAIGN

7.1 VEHICLE INTEGRATION AND SITE PREPARATION

Lockheed Martin provides complete vehicle integration and launch services for its customers. A system of facilities, equipment, and personnel trained in launch vehicle/spacecraft integration and launch operations is in place. The following sections summarize the types of support and services available. Figure 7.1-1 shows a typical factory-to-launch operations flow.

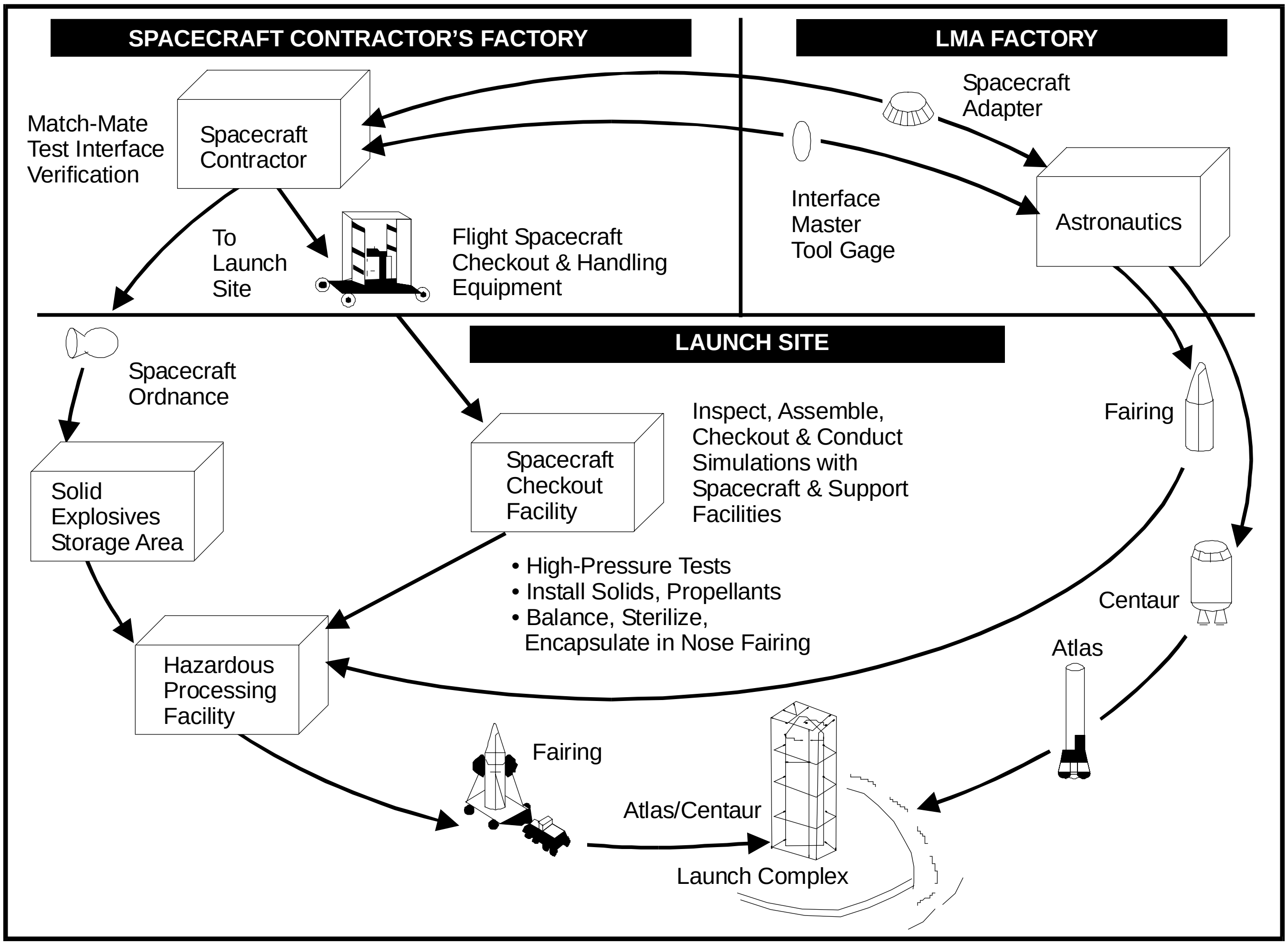


Figure 7.1-1 Typical Factory-to-Launch Operations

7.1.1 Vehicle Spacecraft Integration

Lockheed Martin performs launch vehicle/spacecraft integration and interface verification testing. Testing includes:

1) Matchmate testing of interface hardware at the spacecraft contractor's facility;
   a) Prototype items:
      i) For early verification of design;
      ii) For accessibility to install equipment;
      iii) For development of handling/installation procedures;
   b) Flight items:
      i) For verification of critical mating interfaces before hardware delivery to launch site;
      ii) Separation system installation;
      iii) Bolt hole pattern alignments and indexing;
      iv) Mating surface flatness checks;

v) Electrical conductivity checks;
vi) Electrical harness cable lengths;
vii) Electrical connector mechanical interface compatibilities.

A matchmate at the spacecraft contractor's facility is required for all first-of-a-kind spacecraft. For follow-on and second-of-a-kind spacecraft, matchmates are optional based on experience with the spacecraft and may be performed at the launch site if required.

2) Avionics/electrical system interface testing in the Systems Integration Laboratory (SIL), using a spacecraft simulator or prototype test items for verifying functional compatibility:
    a) Data/instrumentation interfaces;
    b) Flight control signal interfaces;
    c) Pyrotechnic signal interfaces.
3) Special development tests at the launch site;
    a) Spacecraft data flow tests at launch pad (to verify spacecraft mission-peculiar command, control, and/or data return circuits, both hardline and/or radio frequency [RF]);
    b) Electromagnetic compatibility (EMC) testing at the launch pad (to verify spacecraft, launch vehicle, and launch pad combined EMC compatibility).

In addition to integration and interface verification test capabilities, Lockheed Martin uses test facilities to perform system development and qualification testing. Facilities include an integrated acoustic and thermal cycling test facility capable of performing tests on large space vehicles. Other test facilities include the vibration test laboratory, the hydraulic test laboratory, the pneumatic high-pressure and gas flow laboratories, and our propellant tanking test stands.

7.1.2 Launch Services

In addition to its basic responsibilities for Atlas design, manufacture, checkout, and launch, Lockheed Martin offers the following operations integration and documentation services for prelaunch and launch operations:

1) Launch site operations support;
    a) Prelaunch preparation of the Lockheed Martin-supplied payload adapter, nose fairing, and other spacecraft support hardware;
    b) Transport of the encapsulated spacecraft to the launch pad and mating of the encapsulated assembly to the launch vehicle;
    c) Support of launch vehicle/spacecraft interface tests;
    d) Support of spacecraft on-stand launch readiness tests (if requested);
    e) Prepare for and conduct the joint launch countdown.
2) Provide basic facility services and assistance in installation of spacecraft ground support equipment at the launch site:
    a) Installation of spacecraft power, instrumentation, and control equipment in the launch services building and blockhouse;
    b) Provision of electrical power, water, gaseous helium (GHe) and gaseous nitrogen (GN_2) long-run cable circuits, and on-stand communications;
    c) Supply of on-stand payload air conditioning;
    d) Provision of a spacecraft RF reradiate system in the umbilical tower (permitting on-stand spacecraft RF testing).
3) Coordination, preparation, and maintenance of required range support documents:
    a) Air Force System Command documents required whenever support by any element of the Air Force Satellite Control Facility (AFSCF) is requested (includes Operations Requirements

Document [ORD], which details all requirements for support from the AFSCF remote tracking stations [RTS] and/or satellite test center [STC] during onorbit flight operations);

b) Range ground safety and flight safety documentation as required by the launch site range safety regulation;
   i) Missile system prelaunch safety package (MSPSP), which provides detailed technical data on all launch vehicle and spacecraft hazardous items, forming the basis for launch site approval of hazardous ground operations at the launch site;
   ii) Flight data safety package, which compiles detailed trajectory and vehicle performance data (nominal and dispersed trajectories, instantaneous impact data, 3-sigma maximum turn rate data, etc), forming the basis for launch site approval of mission-unique targeted trajectory.

4) Flight status reporting during launch ascent, which is real-time data processing of upper-stage flight telemetry data:
   a) Mark event voice callouts of major flight events throughout launch ascent;
   b) Orbital parameters of attained parking and transfer orbits (from upper-stage guidance data);
   c) Confirmation of spacecraft separation, time of separation, and spacecraft altitude at separation.
5) Transmission of spacecraft data via upper-stage telemetry (an option), which interleaves a limited amount of spacecraft data into the upper-stage telemetry format and downlinks it as part of the upper-stage flight data stream (Ref Sect. 4.1.3.5.2 for requirements).
6) Postflight processing of launch vehicle flight data, which provide quick-look and final flight evaluation reports of selected flight data on a timeline and quantitative basis, as negotiated with the customer.

7.1.3 Propellants, Gases, and Ordnance

All chemicals used will be in compliance with the requirements restricting ozone-depleting chemicals. Minor quantities of GN_2, liquid nitrogen (LN_2), GHe, isopropyl alcohol, and deionized water are provided before propellant loading. A hazardous materials disposal service is also provided. Spacecraft propellants are available at the Cape Canaveral Air Station (CCAS) fuel storage depot. The U.S. national aerospace standards and U.S. military specification that they meet are described in Table 7.1.3-1. Similar services are expected at Vandenberg Air Force Base (VAFB). All propellants required by the spacecraft must comply and be handled in compliance with these standards:

1) **Sampling and Handling**—Analysis of fluid and gas samples is provided as specified in the interface control document (ICD);
2) **Propellant Handing and Storage**—Short-term storage and delivery to the Hazardous Processing Facility (HPF) of spacecraft propellants;
3) **Ordnance Storage, Handling, and Test**—Spacecraft ordnance and solid motors receiving inspection, bridge wire check, leak test, motor buildup, motor cold soak safe and arm check, x-ray, and delivery to HPF. Flight units may be stored for about 3 months and spares may be stored for up to 6 months. Other long-term storage is provided on a space-available basis and

Table 7.1.3-1 Hypergolic Propellants Available at CCAS Fuel Storage Depot

1)	Propellant, Hydrazine, Standard Grade, MIL-P-26536
2)	Propellant, Hydrazine, Monopropellant Grade, MIL-P-26536
3)	Propellant, Hydrazine/Uns-dimethylhydrazine, MIL-P-27402
4)	Monopropellant, High Purity Hydrazine, MIL-P-26536
5)	Propellant, Monomethylhydrazine, MIL-P-27404
6)	Propellant, Uns-dimethylhydrazine, MIL-P-25604
7)	Propellant, Nitrogen Tetroxide (NTO), NAS3620
8)	Propellant, Nitrogen Tetroxide (MON-1), NAS3620
9)	Propellant, Nitrogen Tetroxide (MON-3), NAS3620
10)	Propellant, Mixed Oxides of Nitrogen (MON-10), MIL-P-27408
11)	Propellant, Nitrogen Tetroxide (MON-3, Low Iron), NAS3620

must be arranged in advance. In addition, a safe facility is available for test and checkout (receiving, inspection, and lot verification testing) of ordnance devices.

7.2 INTEGRATED TEST PLAN (ITP)

All testing performed during Atlas design, development, manufacture, launch site checkout, and launch operations is planned and controlled through the Atlas ITP. This encompasses all launch vehicle testing, including spacecraft mission-peculiar equipment and launch vehicle/spacecraft integrated tests. A document titled Test and Evaluation Master Plan (TEMP), with the same function, structure, and utilization as the IPT, is used for SLC-3E missions.

The ITP/TEMP documents all phases of testing in an organized, structured format. It provides the visibility necessary to formulate an integrated test program that satisfies overall technical requirements and provides a management tool to control test program implementation.

The ITP/TEMP consists of an introductory section (defining test concepts, philosophy, and management policies), a summary section (providing a system-by-system listing of all tests, requirements, and constraints for hardware development), and seven sections designated for seven different phases of testing (e.g., design evaluation, qualification, components, flight acceptance, launch site) (Fig. 7.2-1).

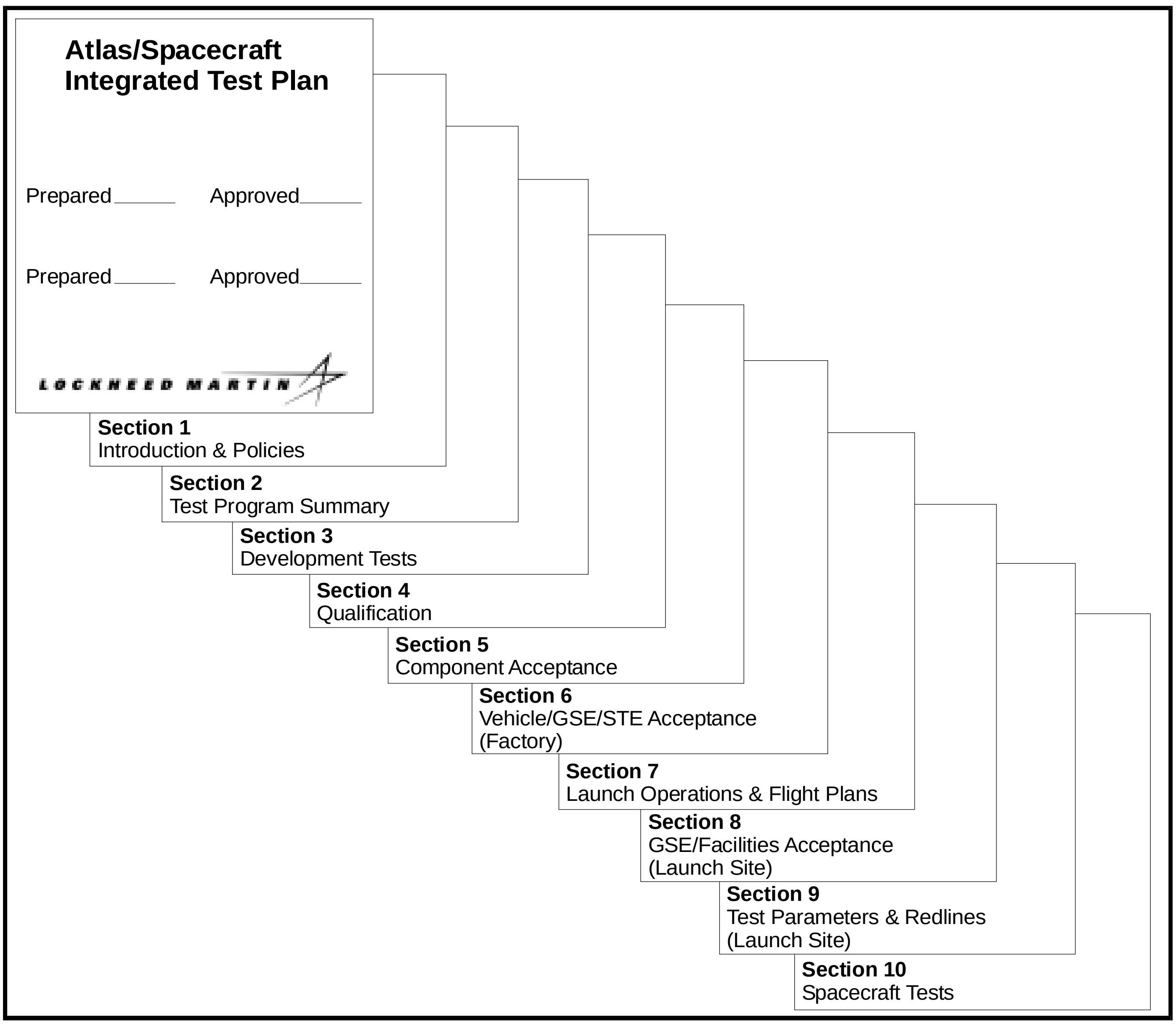


Figure 7.2-1 Integrated Test Plan Organization

Subsections within these headings consist of the individual test plans for each Atlas component, system, and integrated system, and provided detailed test requirements and parameters necessary to achieve desired test objectives. Each subsection is issued as a unique standalone document, permitting its review, approval, and implementation to be accomplished independently from the parent document. Signature approval is required from Lockheed Martin and the customer for all launch vehicle and spacecraft integrated tests.

7.3 TEST PROCEDURES

All test operations are performed according to documented test procedures prepared by test operations personnel using either the approved ITP or TEMP subsections together with engineering drawings and specifications. The procedures for testing of Atlas flight hardware are formally reviewed, approved, and released before testing. The procedures are verified as properly performed by inspection and made a part of each vehicle's permanent history file for use in determining acceptance for flight and final launch readiness.

Test procedures are documents for spacecraft mission-peculiar hardware and joint launch vehicle/spacecraft integrated tests and operations. Customers are urged to discuss their needs with Lockheed Martin early in the mission planning phase so that the various interface and hardware tests can be identified and planned. Customer personnel review and approve mission-peculiar test procedures and participate as required in launch vehicle/spacecraft integrated tests.

7.4 LAUNCH VEHICLE VERIFICATION TASKS

The following paragraphs provide an overview of the typical sequence of tests and activities performed during manufacture, prelaunch checkout, major launch readiness operations, and launch countdown of the Atlas launch vehicle. The purpose is to provide customers with an overview to the general flow and overall scope of activities typically performed.

7.4.1 Factory Tests

Flight vehicle acceptance (or factory) tests are performed after final assembly is complete. Functional testing is typically performed at the system level: low-pressure and leak checks of propellant tanks and intermediate bulkhead, checkout of propellant-level sensing probes, verification of electrical harnesses, and high-pressure pneumatic checks.

7.4.2 Launch Site Prelaunch Operations

Figure 7.4.2-1 shows a typical Atlas checkout and launch operations sequence. After arrival at the launch site, all launch vehicle items are inspected before erection on the launch pad.

After erection of the Atlas and connection of ground umbilical lines, subsystem and system-level tests are performed to verify compatibility between airborne systems and associated ground support equipment in preparation for subsequent integrated system tests.

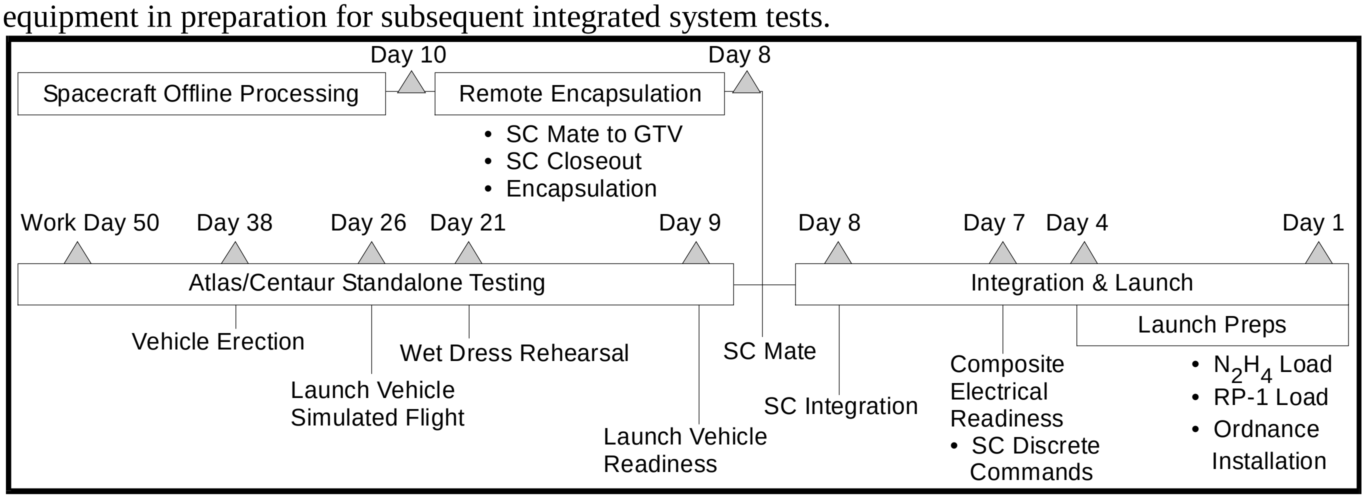


Figure 7.4.2-1 Typical Launch Operations Sequence

The payload fairing (PLF) halves and payload adapter are prepared for spacecraft encapsulation in the HPF for CCAS operations and in the Payload Processing Facility (PPF) for VAFB operations (Fig. 7.4.2-2). Two major tests are performed before the launch vehicle and launch pad are prepared to accept the spacecraft and start integrated operations.

Launch Vehicle Simulated Flight—This first major launch vehicle test verifies that all integrated Atlas/Centaur ground and airborne electrical systems are compatible and capable of proper integrated system operation throughout a simulated launch countdown and plus-count flight sequence.

Wet Dress Rehearsal (WDR)—The WDR is a tanking test to verify readiness of all ground/airborne hardware, support functions, the launch countdown procedure, and Atlas and spacecraft system launch operations personnel assigned launch countdown responsibilities. Although pad operations and selected system responses are simulated, the WDR demonstrates that the integrated Atlas ground, airborne, and associated launch support functions, including range operations, are ready to support launch operations.

The customer supports the launch vehicle simulated flight and WDR operations. Although spacecraft and launch vehicle activities are simulated, participation enhances efficiency of personnel and procedures in subsequent integrated tests and the launch operation.

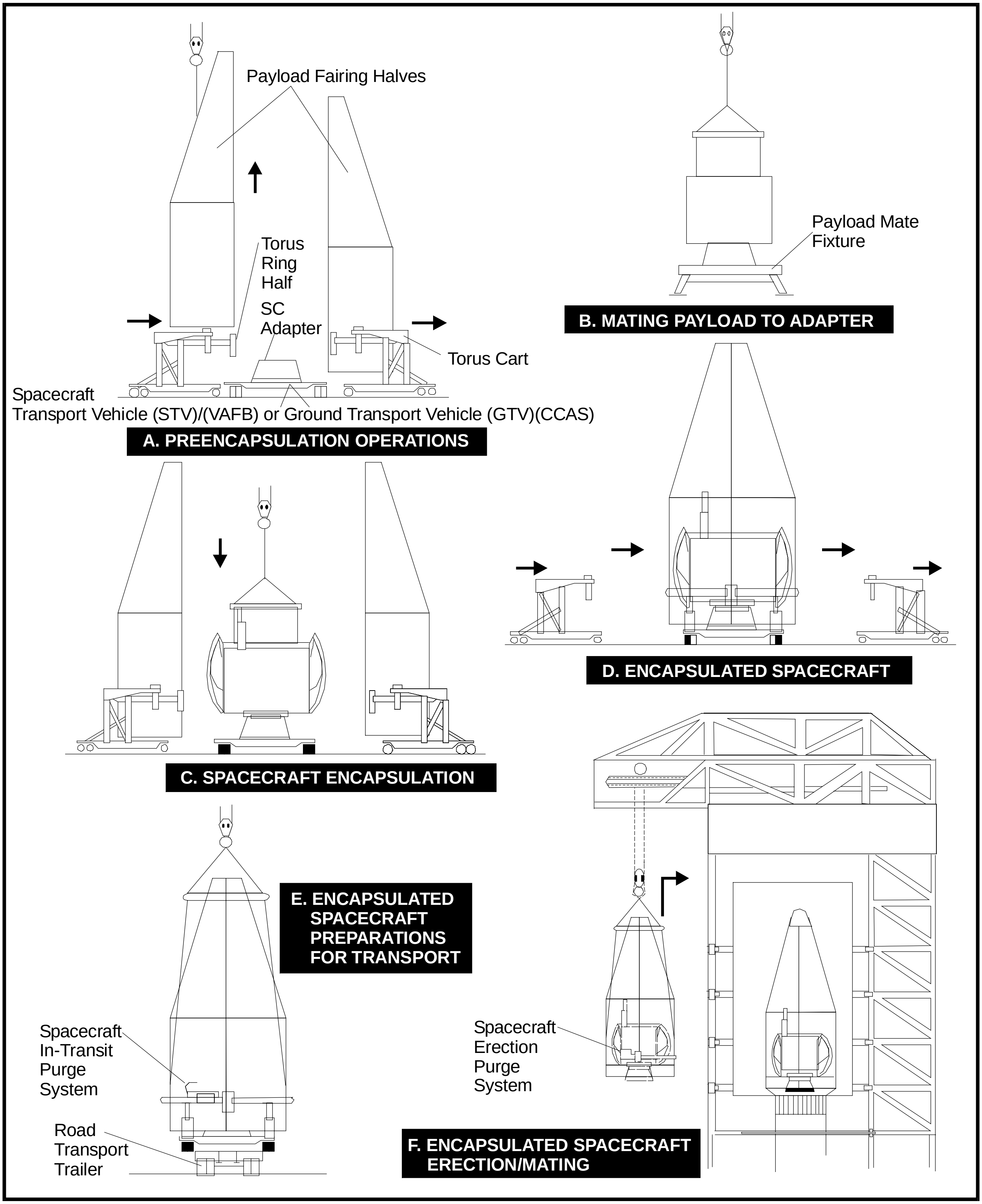


Figure 7.4.2-2 Payload fairing and spacecraft processing are identical to previous Atlas/Centaur operations.

7.4.3 Integrated Operations

After successful WDR, the launch vehicle and launch pad are prepared to accept the spacecraft and commencement of integrated operations. Major prelaunch integrated test and operations are discussed in the following paragraphs.

Erection and Mating of Spacecraft to Centaur—Figure 7.4.3-1 depicts the integrated launch operations tasks to be performed. After arrival at the launch pad, the encapsulated spacecraft assembly is positioned atop the launch pad ramp in front of the erected Atlas/Centaur vehicle. Gas conditioning is available for the encapsulated spacecraft on the ramp and during hoisting. Next, a hoisting sling is fastened to the encapsulated assembly lifting fixture (torus ring) and tie-downs to the ground transport vehicle are released.

The assembly is then hoisted into the service tower by the gantry crane (Fig. 7.4.3-2) and lowered onto Centaur. During final lowering, the payload gas conditioning launch umbilical is attached to the PLF, air flow is initiated, and the payload adapter protective cover is removed. After mechanical attachment of the spacecraft PLF to the Centaur equipment module, the torus ring is removed, necessitating a temporary detachment of the gas conditioning duct. The nose cap contamination cover is removed and the flight upper cone/cap assembly is installed.

Spacecraft Functional Tests—Spacecraft functional tests are performed by the spacecraft contractor shortly after spacecraft mating. These tests verify spacecraft/launch vehicle/launch complex/RF interfaces before initiation of more extensive spacecraft on-stand testing. Included as an integral part of these checks is a verification of the spacecraft launch umbilical and the spacecraft flight harness routed through the Centaur vehicle. The main operations performed during spacecraft testing at the service tower are:

1) Umbilical and RF S-band, C-band, and/or K-band link checks (without spacecraft);
2) Spacecraft batteries trickle charge;

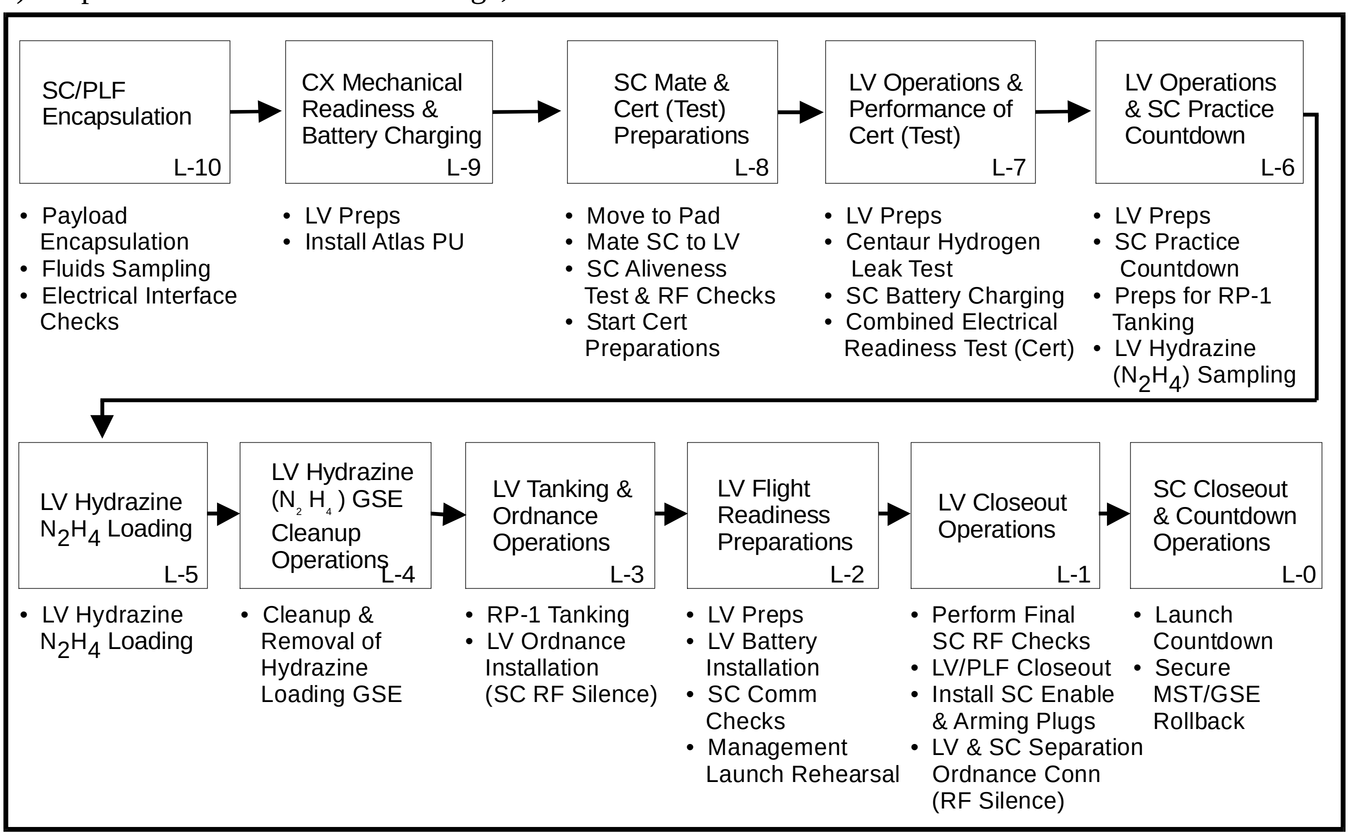


Figure 7.4.3-1 Atlas Launch Site Integrated Launch Operations

3) TM/TC operations in video configuration;
4) TM/TC operations in RF configuration (via reradiation system);
5) Spacecraft flight configuration verifications;
6) Spacecraft/ground stations end-to-end test;
7) Stray voltage test with launch vehicle.

Test Equipment:

1) Spacecraft mechanical and electrical ground support equipment (GSE) at the launch complex;
2) Command and telemetry hardlines among launch pad, blockhouse/RLCC, and PPF;
3) RF consoles equipment at the blockhouse;
4) Spacecraft electrical simulator (as required).

Test Description—These checks are performed with the spacecraft mechanically and electrically mated to the Centaur as for launch. The spacecraft contractor performs a spacecraft-to-blockhouse interface test to verify the total integrated hardline links (i.e., facility hardlines, umbilical cabling, and Centaur flight harness). From the time the spacecraft is connected to the blockhouse test setup to the liftoff, routine operations are conducted, such as battery charging, pressures monitoring, launch configuration check, and temperatures monitoring.

Figure 7.4.3-2 Encapsulated spacecraft is hoisted into the MST.

Special Spacecraft Tests—Special spacecraft tests may be necessary to investigate anomalies in planned tests or operations; reverify equipment operation or performance following changes that were made to correct anomalies; or accommodate spacecraft launch site schedules or operations planning. All special tests are conducted according to written and approved test procedures. For example, an operational spacecraft practice countdown may be performed by the spacecraft contractor to verify spacecraft timelines and to coordinate Lockheed Martin support during final launch preparations. In some instances a spacecraft simulator may be required to validate electrical interfaces, spacecraft communications, and spacecraft GSE before mating the encapsulated payload with the launch vehicle.

Composite Electrical Readiness Test (CERT)—Lockheed Martin performs this test as the final integrated launch vehicle readiness test before start of the launch countdown. The spacecraft contractor provides an input to the test procedure and participates in the test. The test typically is performed on L-7 day.

1) Purpose—To provide a launch readiness verification of the Atlas/Centaur ground and airborne electrical systems (with a minimum of systems violation) after reconnection of the ground umbilicals, after mating of the flight spacecraft assembly, and before pyrotechnic installation for launch. This is the final EMI compatibility test between the spacecraft and launch vehicle.
2) Test Configuration—The Atlas vehicle is in a flight configuration except for pyrotechnics, propellants, and batteries. All batteries are simulated by the gantry test rack (GTR) battery simulator system. The umbilicals remain connected for this test with the service tower in place and the GTR monitoring vehicle loads. The spacecraft is mated in the launch configuration, including pyrotechnics, propellants, and batteries.

3) Test Conditions—All Atlas and Centaur umbilicals remain connected throughout the test. Vehicle power to Atlas and Centaur is provided by the battery simulator. Additional battery simulators are used for Centaur pyrotechnic batteries. The vehicle is armed and on internal power. The flight termination system (FTS) is not tested. All launch vehicle electroexplosive devices (EED) will be simulated by squib simulators. PLF pyrotechnic wiring is used and the spacecraft pyrotechnic circuits are simulated. Atlas/Centaur telemetry and C-band radiate. Landline instrumentation, launch control GSE, the computer-controlled launch set (CCLS), telemetry, and the GTR are used for event monitoring. The spacecraft will be powered and a monitor mode established.
4) Test Description—Atlas electronic systems operate through an abbreviated launch countdown, which includes a vehicle flight control end-to-end steering test. A simulated flight sequence test is performed. All pyrotechnic signals are generated and each associated airborne pyrotechnic circuit is monitored in the low current mode for proper response. Centaur tank pressurization, N_2H_4 engine valve actuation, prestart, and start phases are monitored for proper vehicle responses. A post-test critique is performed at the completion of the test.

L-4 Day Operations—Between CERT and the start of L-3 day final launch preparations, spacecraft and launch vehicle readiness operations are performed as described in the following paragraph.

Atlas operations consist of removing CERT test equipment and performing ground and airborne systems readiness and early vehicle closeout tasks. Typically, these Lockheed Martin activities are performed on L-4 day, but final scheduling may vary at the discretion of the Lockheed Martin test conductor. Tasks include:

1) Complex electrical readiness;
2) Airborne electrical readiness;
3) Propellant loading control unit (PLCU) readiness;
4) Fluids sampling;
5) Complex mechanical readiness;
6) Atlas propulsion readiness;
7) Centaur propulsion readiness;
8) Boom lanyards installation;
9) Telemetry system readiness;
10) Landline instrumentation readiness;
11) Atlas hydraulic readiness;
12) Centaur hydraulic readiness;
13) RP-1 tanking preparation;
14) Closeout tasks (to be scheduled by launch site).

Hydrazine Tanking Operations—Hydrazine tanking operations of the Centaur and Atlas roll control module (ARCM) storage spheres occur between L-5 and L-4 days' operations. Because of the safety requirements for N_2H_4 tanking, the launch pad is cleared. No further testing is required for the N_2H_4 system.

L-3 Day Operations—Activities to be performed on L-3 day (and continuing through L-1 day) consist of final preparations necessary to ready the launch vehicle, spacecraft, and launch complex for start of the L-0 day launch countdown. Because many tasks are hazardous (e.g., limiting pad access, RF transmissions) and/or are prerequisites to others, they are organized on an integrated basis with their sequence and timeliness controlled by a launch precountdown operations procedure. Major operations planned for L-3 day are discussed below; those for L-2 and L-1 days are discussed in subsequent paragraphs. This division of tasks versus L-day is typical, but may be varied if required.

Atlas RP-1 Tanking—RP-1 fuel is tanked aboard the first stage. Together with associated preparations and securing, the task requires approximately 4.5 hours to accomplish and requires limited personnel access in the pad area during the period RP-1 is being transferred.

Installation of Atlas/Centaur Pyrotechnics—After completion of the above, an RF-silence period will be imposed during which mechanical installation of launch vehicle pyrotechnics will be performed. This takes approximately 8 hours.

The electrical harnesses are connected to the pyrotechnic devices and shielding caps installed on the pyrotechnic initiator end of the harness. In the period from L-3 through L-1 days, additional EEDs are installed and all EEDs are connected to the appropriate wiring. At L-1 day, the EEDs' connections are in flight configuration.

L-2 Day Operations—L-2 day operations involve a continuation of launch vehicle readiness activities consisting of approximately 8 hours of launch preparations followed by 7 hours of selected closeout tasks. Included are:

1) Atlas propulsion launch preparations (to include trichloroethylene flush [CCAS] or hot GN_2 purge [VAFB] and hypergolic igniters installation);
2) Umbilical boom lanyard connections;
3) Spacecraft separation pyrotechnic battery installation;
4) Atlas and Centaur battery installations;
5) Closeout tasks (to be scheduled by launch site).

Launch Countdown Rehearsal—A joint spacecraft/launch vehicle countdown rehearsal is typically performed on L-2 day. This procedure uses key elements from the respective countdowns arranged on an abbreviated timeline. The objective of this integration test is to acquaint launch team operations personnel with communications systems, reporting, and status procedures that are used in the launch countdown. Simulated "holds" are included to rehearse hold-and-recycle procedures. The operation is critiqued and recommendations are incorporated as required to improve overall communications procedures. Spacecraft personnel participating in this rehearsal use the actual operating stations they will use for the countdown operation.

L-1 Day Operations—L-1 day operations consist of approximately 15 hours of integrated testing and final readiness tasks. Final scheduling of these activities will be formulated at the launch site and documented in the L-1 day section of the launch precountdown operations procedure. Major operations are summarized in the following paragraphs.

Spacecraft Ordnance Connect—45 SW/30 SW Safety may approve spacecraft ordnance connection on L-1 day. The task is performed by the spacecraft contractor with Lockheed Martin support. Supporting data must be submitted with the request. During the ordnance connect procedure, power-on and power-off measurements are required on ordnance circuits.

Launch Vehicle Final Readiness Tasks—Early on L-1 day, approximately 9.5 hours of launch vehicle operations are performed:

1) Centaur propulsion final preparations;
2) Centaur hydraulic readiness tests;
3) Atlas and Centaur battery connections;
4) Centaur transfer line purge;
5) Atlas RP-l and liquid oxygen ($L0_2$) final connections;
6) Centaur C-band and telemetry early tests;
7) Atlas RP-1 system securing.

Launch Vehicle Ordnance Tasks—After completion of the above, launch vehicle final ordnance tasks, lasting approximately 3 hours, are performed by Lockheed Martin. Due to the hazardous nature of these tasks, RF silence is required throughout most of this period. One exception is the period during FTS command tests with the range (before FTS destructor installations), during which the launch pad is in an “area red” condition. Specific ordnance tasks include:

1) Mechanical installation and electrical connection;
    a) Atlas destructor charge, Centaur destructor charge, and Centaur safe/arm initiator;
    b) Atlas/Centaur shaped charge detonators and staging systems.
2) Electrical connection of all previously installed Atlas, Centaur, and PLF pyrotechnics.

Launch Vehicle Closeout Tasks—After completion of ordnance operations, RF silence conditions are lifted. Final L-1 day operations consist of approximately 3.5 hours of PLF, umbilical lanyard, and ground wind damper closeout tasks, and the PLF isolation diaphragm is removed. PLF access doors for the equipment module are closed for flight.

Test and Checkout Responsibilities—Lockheed Martin operations personnel are responsible for launch vehicle standalone operations and for the overall conduct of integrated spacecraft/launch vehicle tests and operations. The spacecraft contractor is responsible for the conduct of spacecraft standalone tests and operations.

Specifically, the Lockheed Martin launch site function performs the following major tasks in the test conduct arena:

1) Launch vehicle integration with the spacecraft contractor activities associated with the launch vehicle and the launch complex;
2) Provide overall launch pad operation integration and conduct the final countdown and launch;
3) Organize and chair ground operations working groups (GOWG) and technical interchange meetings between contractors, as necessary, to provide an integrated spacecraft/launch vehicle operation;
4) Represent the launch vehicle operations at the spacecraft/launch vehicle launch base-related working group meetings, joint procedure critiques, and readiness reviews;
5) Prepare and control all launch vehicle-prepared integrated spacecraft/launch vehicle test procedures and launch vehicle-unique test procedures;
6) Provide integrated safety documentation and monitoring for compliance with the applicable Range Safety requirements;
7) Provide integrated launch complex schedule;
8) Support spacecraft prelaunch activities;
9) Provide launch complex security;
10) Provide launch complex safety;
11) Conduct readiness reviews at key points during launch vehicle processing.

The spacecraft contractor is responsible for providing spacecraft-associated test equipment and the logistical and technical support required to support spacecraft launch operations at the launch site. The spacecraft contractor is responsible for the following major tasks in the test conduct arena:

1) Prepare and control spacecraft standalone procedures;
2) Prepare, input, and approve spacecraft/launch vehicle integrated procedures;
3) Attend all launch base-related working group meetings, joint procedure critiques, and readiness reviews;
4) Provide safety documentation (e.g., procedures plans) and monitor compliance with the applicable range safety requirements;

5) Plan and, if necessary, implement spacecraft abort activities.

The Lockheed Martin mission integrator continues to manage the mission-peculiar engineering aspects of the program by performing the following major tasks during the launch campaign:

1) Provide technical management overview of the launch vehicle/spacecraft launch campaign to ensure compliance to mission requirements and resolution of requirements issues;
2) Ensure that launch vehicle/spacecraft interface requirements to be verified during launch operations are flowed down into appropriate site checkout and test procedures and parameter/redline documents;
3) Act as engineering focal point, responsible for timely and thorough communication of all issues and decisions within Lockheed Martin engineering and spacecraft community and initiate engineering review board (ERB) activity as required;
4) Integrate ICD waiver/deviation requests and disposition with spacecraft community and integrate ICD change activity as appropriate;
5) Manage mission integration schedule to ensure timely completion of Lockheed Martin engineering milestones for the launch campaign and review and approve mission-peculiar technical documentation;
6) Ensure closeout of all mission-peculiar action items.

Tasks versus their approximate launch-day schedule (launch days are calendar days before launch) are listed in Table 7.4.3-1.

Table 7.4.3-1 Launch-Day Task Schedule

Tasks	**L-Day**
1) Ground & Airborne Systems Readiness Tests	L-4
2) Hydrazine Loading of Centaur Reaction Control System	L-4/L-3
3) Atlas RP-1 Tanking	L-3
4) Pyrotechnic Installations	L-3
5) Atlas & Centaur Battery Installations	L-2
6) Ordnance Final Installations & Hookups	L-1
7) Launch Vehicle Closeout Tasks	L-1

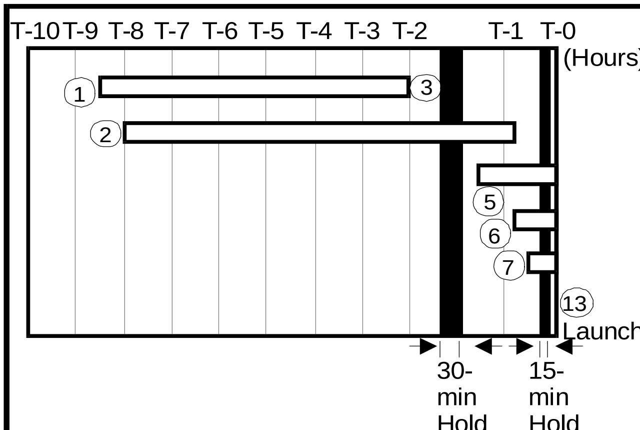
7.4.4 Launch Countdown Operations

The Atlas launch countdown consists of an approximate 9- to 10-hour count, which includes two built-in holds (one at T-105 minutes [for 30 minutes] and the second at T-5 minutes [for 15 minutes]) to enhance the launch-on-time capability (Fig. 7.4.4-1).

Lockheed Martin's launch conductor performs the overall launch countdown for the total vehicle. The launch management is designed for customers and Lockheed Martin efficiencies and control elements (Fig. 7.4.4-2).

Spacecraft operations during the countdown should be controlled by a spacecraft test conductor located either in the launch control facility or at some other spacecraft control center (e.g., in the spacecraft checkout facility) at the option of the spacecraft customer.

As launch pad integrator, Lockheed Martin prepares the overall countdown procedure for launch of the vehicle. However, the spacecraft agency prepares its own launch countdown



1.	Start Launch Complex Mechanical & Electrical Preps	T-515
2.	Start SC Initialization Preps	T-480
3.	Start Tower Removal	T-135
4.	Built-In Hold for 30 min	T-105
5.	Start Centaur LO_2 Tanking	T-91
6.	Start Atlas LO_2 Tanking	T-65
7.	Start Centaur LH_2 Tanking	T-51
8.	T-5 & Holding for 15 min	T-5
9.	Atlas to Internal Power	T-4:00
10.	Centaur to Internal Power	T-2:00
11.	Pressurization to Internal	T-70
12.	Start Launch Automatic Sequence	T-31

Figure 7.4.4-1 Launch Countdown Summary

procedure for controlling spacecraft operations. The two procedures are then integrated in a manner that satisfies the operations and safety requirements of both and permits a synchronization of tasks through periodic status checks at predetermined times early in the count and a complete mesh of operations during the final steps leading to final committal to launch.

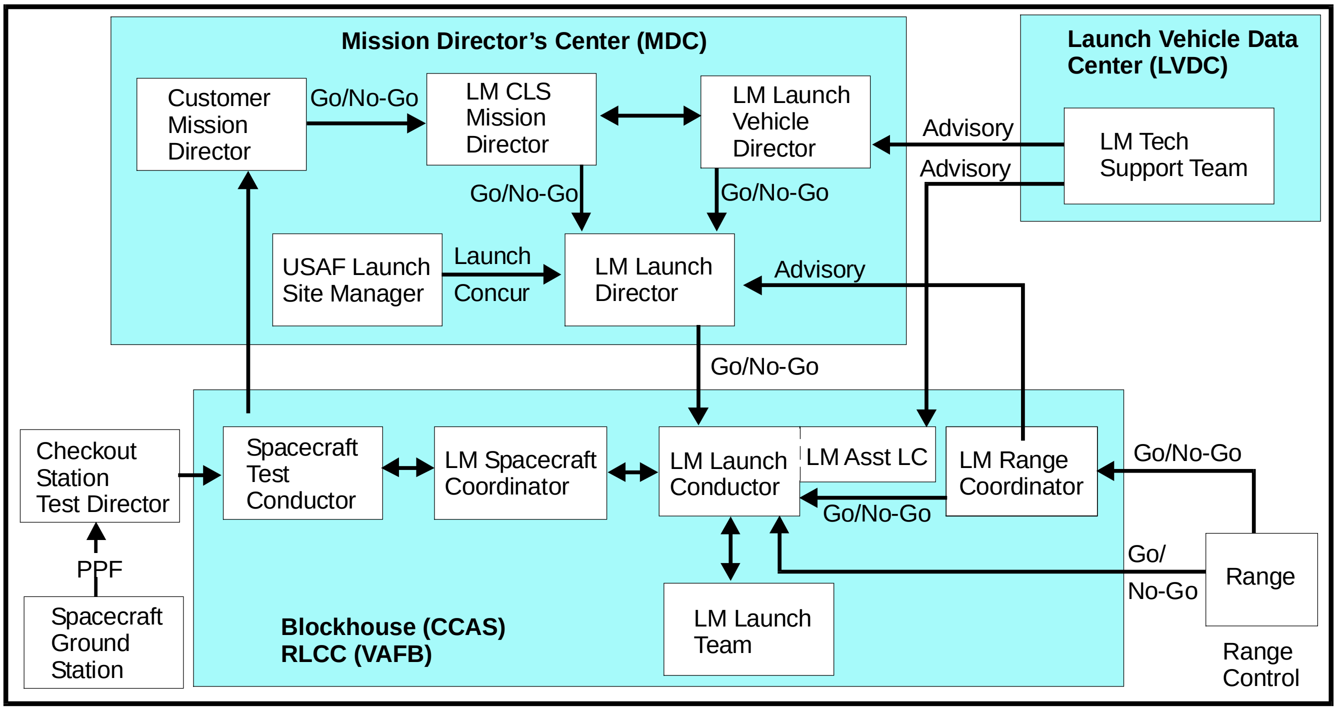


Figure 7.4.4-2 Typical Launch Day Management Flow Diagram

7.5 LAUNCH CAPABILITY

In addition to the scheduled 30-minute and 15-minute countdown holds, additional hold time can be scheduled for up to 2 hours under normal environmental conditions or until end of the scheduled launch window, whichever comes first.

Launch window restrictions have typically been determined by the spacecraft mission requirements. The Atlas launch vehicle essentially does not have launch window constraints beyond those of the mission.

7.6 WEATHER LAUNCH CONSTRAINTS

In addition to mission-dependent launch window restrictions, the decision to launch depends on weather launch constraints. Weather launch constraints include cloud conditions, lightning, thunderstorms, and ground and upper atmosphere winds. Excessive winds during launch may cause overloading of the vehicle structure and control system. Limiting conditions have been well defined and operational approaches developed to ensure launch within safe limits.

The decision to launch may be constrained if significant weather and/or thunderstorm conditions exist in the proximity of the launch site or the planned vehicle flight path at the time of liftoff. A go/no-go decision to launch is made by the Lockheed Martin launch director based on the following information.

7.6.1 Avoidance of Lightning

The presence of lightning or equivalent weather conditions within a certain distance from the launch complex requires a halt to all pad operations. A complete list of launch vehicle lightning constraints is contained in the appropriate generic launch vehicle PRD. Several conditions, as identified below, are used in determining operations criteria:

1) Do not launch for 30 minutes after any type of lightning occurs in a thunderstorm if the flight path will carry the vehicle within 18 km (10 nmi) of that thunderstorm, unless the cloud that produced the lightning has moved more than 18 km (10 nmi) away from the planned flight path.
2) Do not launch if the planned flight path will carry the vehicle:
   a) Within 18 km (10 nmi) of cumulus clouds with tops higher than the -20°C level;
   b) Within 9 km (5 nmi) of cumulus clouds with tops higher than the -10°C level;
   c) Through cumulus clouds with tops higher than the -5°C level;
   d) Within 18 km (10 nmi) of the nearest edge of any thunderstorm cloud, including its associated anvil;
   e) Through any cumulus cloud that has developed from a smoke plume while the cloud is attached to the smoke plume, or for the first 60 minutes after the cumulus cloud is observed to have detached from the smoke plume.
3) Do not launch if at any time during the 15 minutes before launch time, the absolute electric field intensity at the ground exceeds 1 kV/m within 9 km (5 nmi) of the planned flight path, unless there are no clouds within 18 km (10 nmi) of the launch site. This rule applies for ranges equipped with a surface electric field mill network.
4) Do not launch if the planned flight path is through a vertically continuous layer of clouds with an overall depth of 1,370 m (4,500 ft) or greater, where any part of the clouds is located between the 0 and -20°C temperature levels.
5) Do not launch if the planned flight path is through any cloud types that extend to altitudes at or above the 0°C temperature level and are associated with the disturbed weather within 9 km (5 nmi) of the flight path.
6) Do not launch if the planned flight path will carry the vehicle through thunderstorm debris clouds or within 9 km (5 nmi) of thunderstorm debris clouds not monitored by a field network or producing radar returns greater than or equal to l0 dBz.
7) Good Sense Rule—Even when constraints are not violated, if any other hazardous conditions exist, the launch weather team reports the threat to the launch director. The launch director may hold at any time based on the instability of the weather.

7.6.2 Ground Winds Monitoring

The Atlas launch vehicle is subject to ground wind restrictions during vehicle erection and assembly, after tower rollback up to the time of launch, and at launch. Lockheed Martin has an established ground winds restriction procedure that provides for limiting wind speeds for all ground winds critical conditions. In addition, the document provides insight into the nature of ground winds loadings and possible courses of action should the wind speed limits be attained. The ground winds restriction procedure also contains limiting wind speeds during Atlas/Centaur erection, hoisting, and payload hoisting.

The ground winds monitoring system is designed to monitor vehicle loads after mobile service tower (MST) rollback and before launch. This is accomplished by sampling flight rate gyro rotational velocities (pitch and yaw signals), ground winds anemometer speed, ground wind directional azimuth, tanking levels, and tank ullage pressures. Data are processed, providing the ground winds monitor with a ground winds load ratio (LR) that represents the maximum load-to-limit allowable ratio in the vehicle or launcher at any given time. In addition, the LR is presented from the computer-controlled launch set (CCLS) using a present ground winds monitoring station strip chart. This system requires the presence of a ground winds monitor (one person) to evaluate the plotted and printed output data and immediately inform the launch conductor whenever the LR is approaching an out-of-tolerance condition.

7.6.3 Flight Wind Restrictions

Most loads experienced by the Atlas vehicle in flight can be calculated well in advance of the vehicle's launch date. However, one major loading condition induced by the prevailing atmospheric winds (called flight wind profile) must be accounted for just before launch if maximum launch availability and mission success are to be ensured during marginal weather conditions.

On each mission, the pitch and yaw program is designed on launch day based on the actual launch day winds as determined from launch site weather balloon soundings. This capability is provided by a computer software program called Automatic Determination and Dissemination of Just Updated Steering Terms (ADDJUST) performed on Lockheed Martin Denver-based computer systems. Specifically, ADDJUST makes it possible to accomplish the following automatically:

1) Design an Atlas booster phase pitch/yaw program pair based on wind data measured at the launch site during the launch countdown;
2) Determine whether the wind profile loads and engine angles violate the vehicle's structural and control constraints;
3) Transmit the designed programs to the CCLS computer at the launch site launch control facility (for subsequent loading into the flight computer) with verification of correct transmittal of data.

Wind Sounding Procedure—Operations begin with the release of weather balloons from the launch range at specific intervals before launch. Raw wind data obtained from each balloon sounding are computer-reduced by range weather personnel to wind speed and direction data.

ADDJUST Program Procedure—Launch site wind data are received by computers at Lockheed Martin in Denver, CO, via data phone and automatically verified. The ADDJUST design and verification sequence is then executed. The resulting pitch and yaw program pair designed by ADDJUST is available for transmission back to the launch site approximately 10 minutes after Denver completes reception of the wind data. Pitch and yaw data transfer occurs directly from the Denver computer to the launch site backup CCLS computer via standard telephone lines. Simultaneously with this transmission, ADDJUST will proceed with loads validation computations, checking predicted loads and engine angles resulting from the pitch and yaw program design versus vehicle structural and control allowables. This will be followed by an engineering trajectory simulation run to check all trajectory-related parameters.

Launch Recommendations—With the ADDJUST-generated programs, all constraints must be satisfied before a "go" recommendation for launch may be made. The ADDJUST designer was developed so that the trajectory related constraints resulting from the engineering trajectory simulation would be satisfied. While the designer minimizes angle of attack, it cannot design pitch and yaw programs for a specific chosen set of loads.

7.7 LAUNCH POSTPONEMENTS

7.7.1 Launch Abort and Launch Vehicle 24-hour Recycle Capability

Before T-4 seconds (when the upper stage aft panel is ejected), the launch vehicle has a 24-hour turnaround capability after a launch abort due to a nonlaunch vehicle/GSE problem. If the abort occurs after securing the interstage adapter area, access to it will be required to service the N_2H_4 system.

7.7.2 Launch Abort and Vehicle 48-hour Recycle Requirements

A launch abort after T-4 seconds and before T-0.7 seconds requires a 48-hour recycle. The principal reason for a 48-hour recycle versus a 24-hour recycle is the added time requirement for replacing the upper stage aft panel (ejected at T-4 seconds) and removal and replacement of the propellant pressurization lines pyrovalves (fired at T-2 seconds).

8.0 ATLAS SYSTEM ENHANCEMENTS

Maximizing on the combined strengths of the engineering, development, and production functions of the Lockheed Martin Corporation, several Atlas launch systems enhancements are being implemented or planned to maintain the competitiveness of Atlas in the launch services market. Several initiatives are being supported to expand the operational capability of the Atlas/Centaur vehicle. And finally, looking toward 21st century requirements, several major Atlas modifications are planned and initial engineering is beginning. This section describes, and Table 8-1 summarizes, the initiatives being pursued by the Atlas program.

8.1 MISSION-UNIQUE ENHANCEMENTS

8.1.1 Spacecraft Dispensers and Universal Spacecraft Separation Node

To support multiple payload launches on Atlas, the program has predesigned several different configurations of payload "dispensers." Dispensers for both radial and longitudinal spacecraft separation have been investigated. Figure 8.1.1-1 illustrates a radial release dispenser that accommodates four spacecraft. In addition to primary structure design, significant emphasis has been placed on the design of an efficient spacecraft attach point. The result of these efforts is illustrated in Figure 8.1.1-2. The Universal Spacecraft Separation Node (USSN) is designed to maximize the capabilities of a given separation device through the use of unique load carrying features. It has the flexibility to be adapted to a variety of spacecraft and dispenser configurations and can accommodate a fast acting shockless separation nut (FASSN).

The FASSN provides virtually shockless and instantaneous release of the spacecraft by converting separation bolt strain energy into rotational kinetic energy and by the elimination of pyrotechnic devices. A ½-in. diameter version of this system has been qualified for flight, including more than 450 successful activations, and will be used on a Lockheed Martin-built spacecraft in 1998. A ¾-in. diameter version of the FASSN is also being considered for development. Although the spacecraft interface shown in these figures is the baseline used

Table 8-1 Atlas Launch System Enhancement Initiatives

Initiative	Initial Launch Capability
Mission-Unique Enhancements	
• Centaur Extended Mission Kit	Operational
• Heavy Payload (14,000-lb Class) Uprate	Operational
• Centaur Short Burn Capability	Operational
• Centaur Type F Adapter Interface	ATP + 24 mo
• Extended-Length Large Fairing	Operational
Evolutionary Enhancements	
• Heavy Payload (18,000-lb+) Uprate	ATP + 30 mo
• Reengined Atlas (Atlas IIIA)	4th Qtr 1998
• Reengined Atlas w/Stretched Centaur (Atlas IIIB)	3rd Qtr 2000

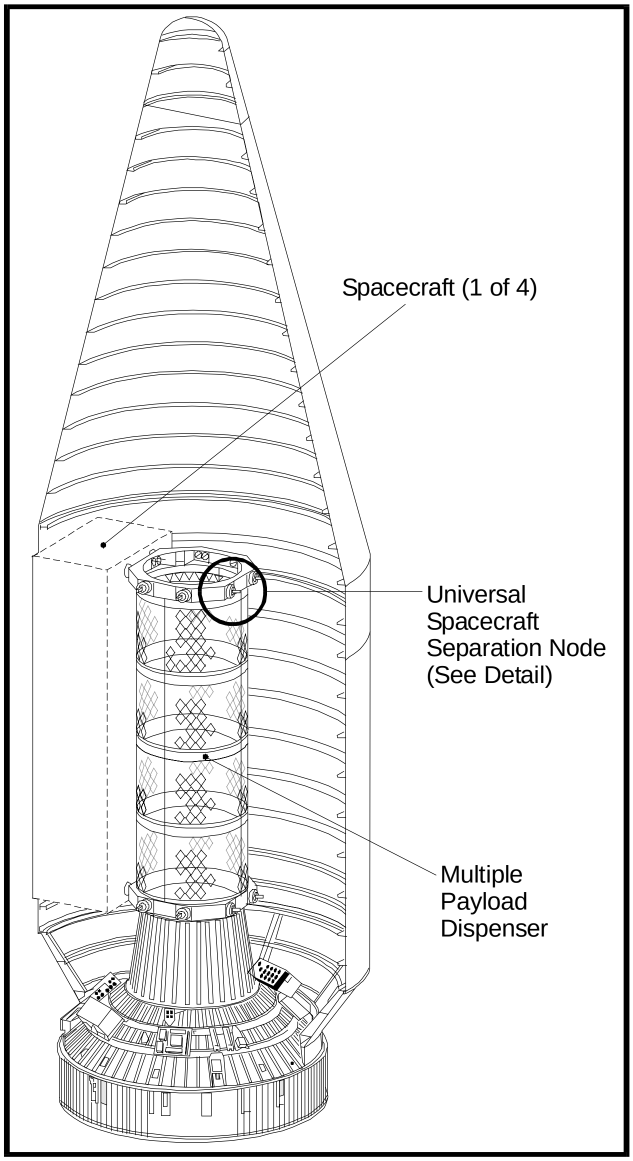


Figure 8.1.1-1 Multiple Payload Dispenser for Four Spacecraft—Shown in the Atlas EPF

in prototype design, the node can be modified to optimize individual spacecraft needs. The USSN allows a simple bolt interface between the launch vehicle and spacecraft; this significantly simplifies mechanical interface integration efforts.

8.1.2 Centaur Type F Adapter Interface

A Type F adapter has been conceptually developed for the Hughes HS702 spacecraft bus. It has a 1663.70-mm (65.50-in.) diameter, four hardpoint forward interface ring (Fig. 8.1.1-1). The preliminary structural capability of the Type F adapter is shown in Figure 8.1.1-2. The separation nuts under consideration for use with the Type F adapter include a standard pyrotechnic-activated nut and the FASSN discussed in Section 8.1.1. The pyrotechnic nut was developed by Lockheed Martin and is qualified for flight on the Type E adapter for NASA's EOS AM-1 Mission.

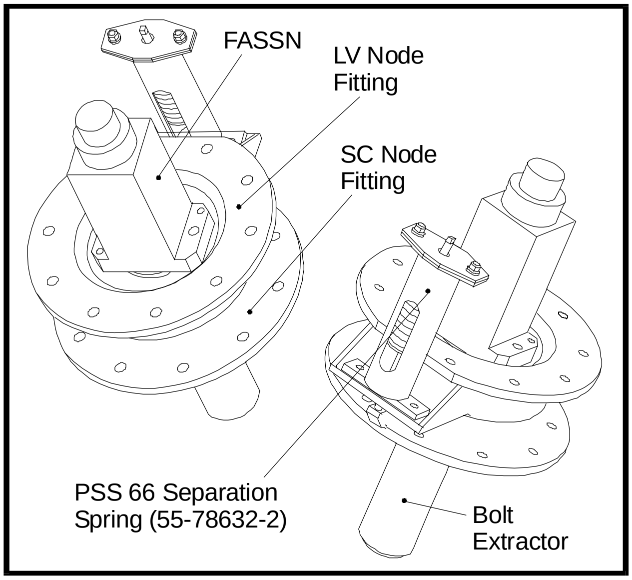


Figure 8.1.1-2 The Universal Spacecraft Separation Node (USSN)—for Use with "Hard-Point" Interface Spacecraft

8.1.3 Payload Adapter/Payload Separation System Upgrades for IIIB Class Spacecraft

The goal of the Atlas program is to provide the increased strength required for the heavier class of spacecraft that can be flown on the IIIB vehicle and the Atlas V family of vehicles while maintaining the standard spacecraft interface diameters. Both the payload adapters and payload separation systems will be strengthened to meet the anticipated increases in tension and compression line load requirements.

Table 8.1.3-1 summarizes the planned limit line load capabilities for each of the standard interface diameters and includes a brief description of the anticipated hardware modifications. The projected line load capabilities were determined as to envelope the maximum expected spacecraft masses and center-of-gravity locations as well as potential peaking affects.

Table 8.1.3-1 Payload Adapter/Payload Separation System Upgrades

Interface Diameter, mm (in.)	Limited Line Load Allowed	Current Interface Capabilities		Targeted Interface Capabilites		Targeted Spacecraft Mass, cg	Planned Hardware Modifications
		Comp	Tension	Comp	Tension		
937 (37)	N/mm (lb/in.)	180 (1,030)	88 (500)	180 (1,060)	133 (760)	4,140 kg (9,130 lb) 1.40 m (55.12 in.)	Upgraded PSS37
1194 (47)	N/mm (lb/in.)	186 (1,060)	103 (590)	266 (1,520)	210 (1,190)	5,860 kg (12,920 lb) 2.50 m (98.43 in.)	• Upgraded PSS47 • Type B1 PLA to 2090 & Stringer Gage Increase
1,666 (66)	N/mm (lb/in.)	151 (860)	65 (370)	151 (819)	88 (500)	5,980 kg (13,180 lb) 2.40 m (94.49 in.)	Upgraded PSS66

8.2 ATLAS EVOLUTIONARY ENHANCEMENTS

8.2.1 Heavy Payload (18-klb+) Uprates

For the largest Atlas class payloads that may require a large-diameter payload interface, a 2,667-mm (105-in.) diameter interface has been conceptually defined. It offers greater stiffness and structural capability than the strengthened equipment module interface. For requirements in the 6,800-kg (15,000-lb) to 13,600-kg (30,000-lb) payload class, a truss adapter concept is proposed. It is similar in design to the truss flown with Centaur on NASA's Viking Mars lander program in the 1970s (Fig. 8.2.1-1) and is fully encapsulated within the PLF.

The design incorporates an aft torque box that incorporates multiple hard points in the Centaur equipment module aft ring (Fig. 8.2.1-2). A series of struts attaches to the hard points and supports a forward torque box. The forward torque box provides a bolted interface. The structural capability of the truss is shown in Figure 8.2.1-3. It should be noted that if a mission application exceeds this curve, appropriate modifications can be made to this design concept to match a particular requirement. A mission-unique payload adapter and separation system that mates with the top of the truss can be provided by either the Atlas program or the spacecraft manufacturer.

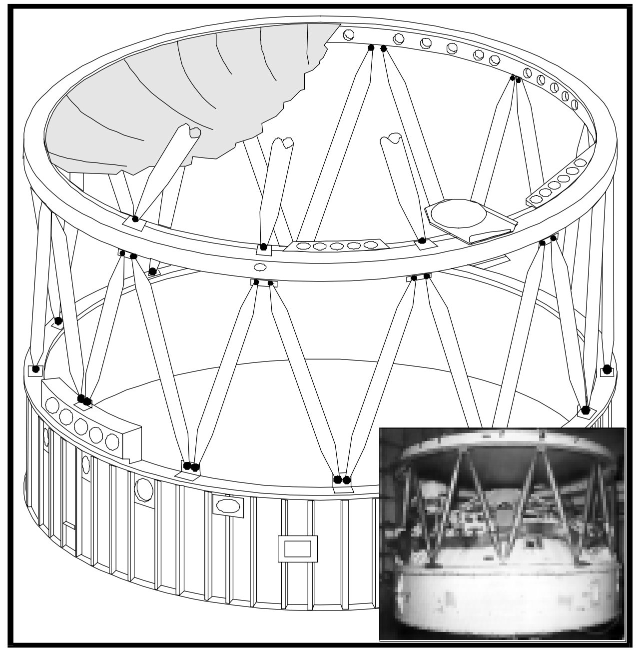


Figure 8.2.1-1 Centaur/Viking Spacecraft Truss Adapter for Heavy Spacecraft

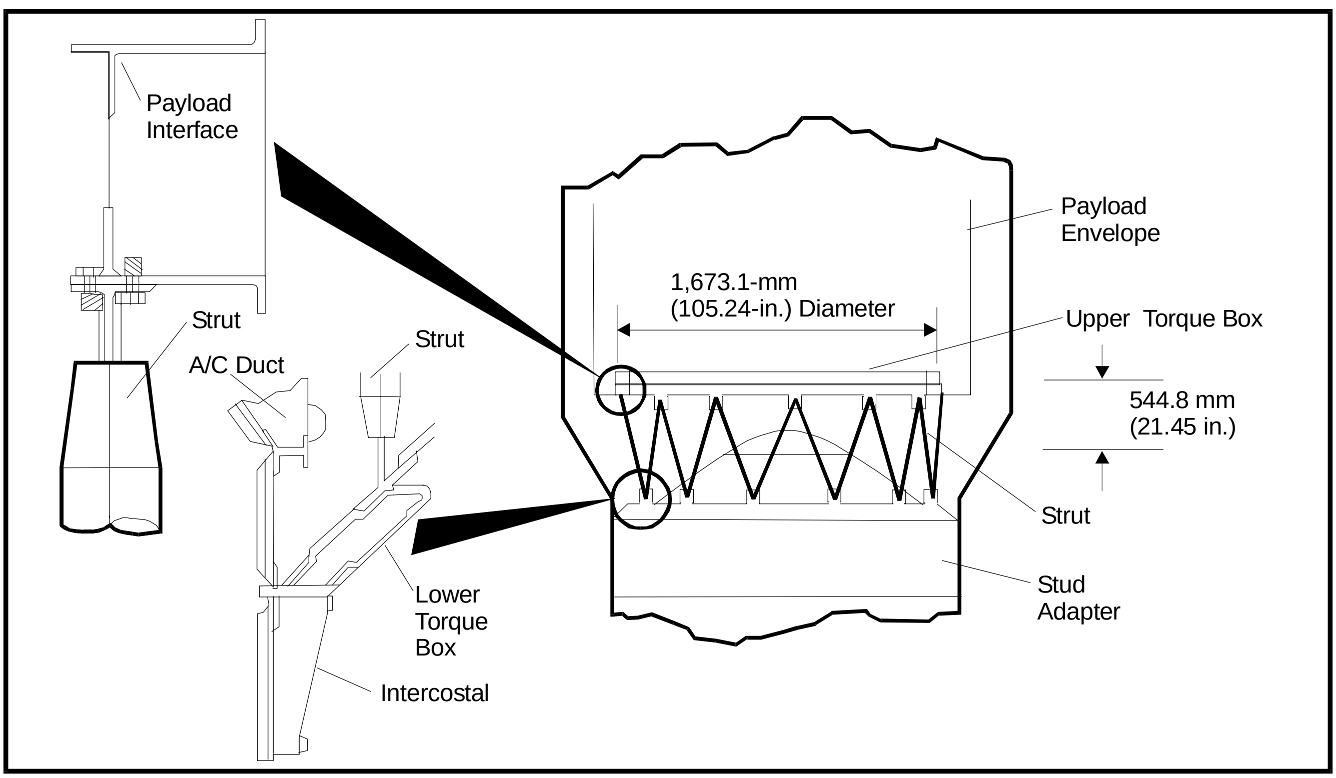


Figure 8.2.1-2 Truss Adapter Interface Concept

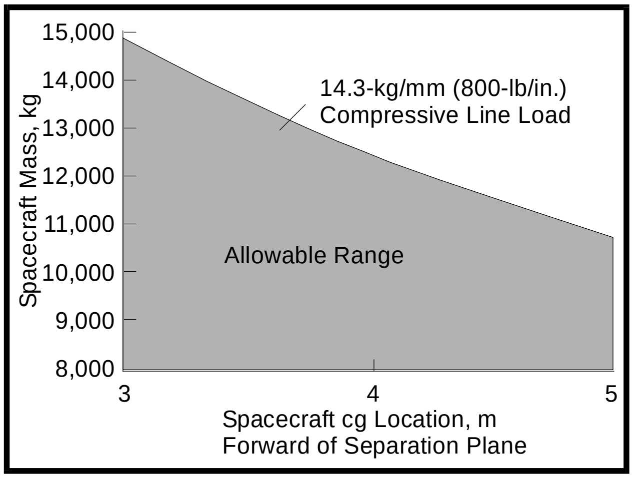


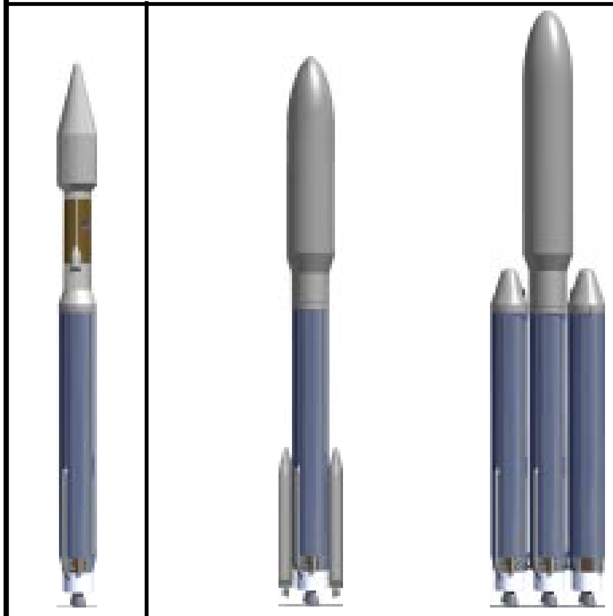
Figure 8.2.1-3 Truss Adapter Structural Capability

8.2.2 Atlas V Family of Vehicles

With the Air Force EELV development program, Lockheed Martin is developing the Atlas V family of vehicles to meet the increasing demand by both Government and commercial payloads for lower recurring costs and the demands to lift heavier intermediate class payloads to orbit. The primary objectives for this development are to reduce recurring costs, improve standardization within the launch fleet and support operations, increase operational efficiencies, increase lift capability margin, and increase system reliability and availability. Lockheed Martin has defined a number of design goals, including a performance specification of delivering approximately 5,000-kg (11,000-lb) payload systems weight to a typical geosynchronous transfer orbit (GTO) and up to 6,350 kg (14,000 lb) of payload systems weight to a geostationary orbit (GSO). Options and upgrades are also under development to increase GTO payload up to 8,200 kg (18,000 lb).

The Atlas V family configurations are shown in Figure 8.2.2-1. The Atlas V uses a Common Core Booster™ concept that maximizes commonality between launch vehicles by using a single booster for the core and as strap-on boosters to make the Atlas V HLV configuration. Each booster is powered by a single RD-180 engine, the same engine that will fly on the Atlas IIIA. Commonality continues with a single-engine Centaur (SEC) upper stage for the Atlas V 400, 500 and HLV configurations. Common interstage adapters, a common avionics suite, and payload fairings provide the remaining building blocks needed for our Atlas V family configurations.

ATLAS V



Atlas V 400	Atlas V 500			Atlas V HLV
	Number of Solids			
	3	4	5	
Performance kg (lb)				
5,000 (11,000) to GTO	6,900 (15,200) to GTO	7,600 (16,700) to GTO	8,200 (18,000) to GTO	
	3,000 (6,600) to GSO	3,400 (7,450) to GSO	3,750 (8,250) to GSO	6,350 (14,000) to GSO
8,000 (17,600) to LEO Polar			17,000 (37,450)to LEO Polar	19,000 (42,000) to LEO Polar
	17,250 (38,000) to LEO 28.5	18,750 (41,300) to LEO 28.5	20,050 (44,200) to LEO 28.5	

Note:
GTO & GSO Performance Uses Single Engine Centaur; Atlas V 400 & Atlas V Heavy LEO Performance Is With Single Engine Centaur. Atlas V 500 LEO Performance is With Dual Engine Centaur

Figure 8.2.2-1 The Atlas V Launch Vehicle Family

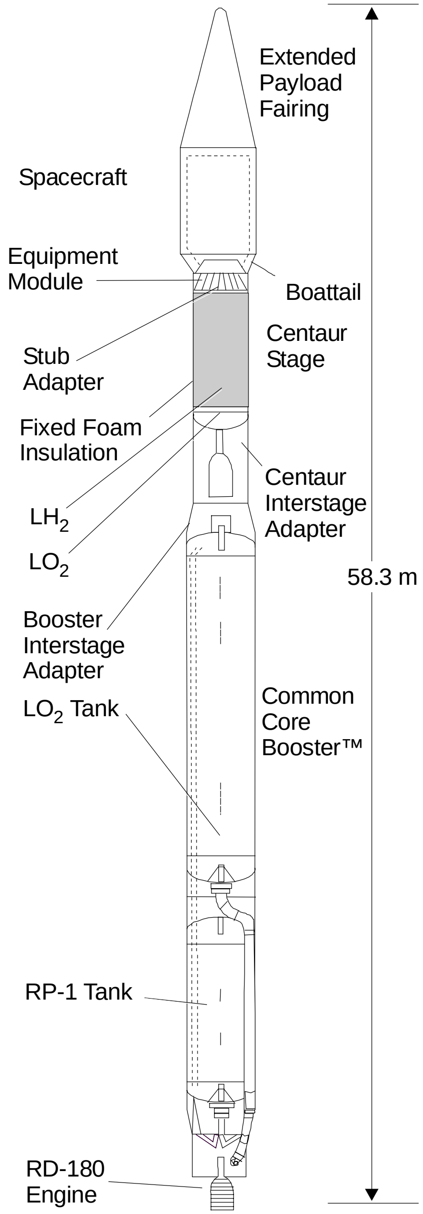
The Atlas V 400 (4m EPF) configuration delivers the medium lift Delta and Atlas class missions and an optional 3m MPF is available for smaller spacecraft. The Atlas V 500 and HLV delivers large intermediate and Titan class missions to both low earth and high-energy orbits. Figures 8.2.2-2 and 8.2.2-3 summarize the characteristics of Atlas V 400 and Atlas V HLV vehicle systems.

Atlas V 500 can fly with any increment of SRMs from zero to five. A 5-m class payload fairing is also provided by the Atlas V 500. The vehicle can fly with either a single-engine or dual-engine Centaur, depending on mission needs. Atlas V 500 can be available for launch as early as 2001, given appropriate market demand.

Although the Atlas V family involves a number of new development and changes to previous Atlas flight and ground systems, the electrical and mechanical payload interfaces of the Atlas IIIA, as defined in this document will be maintained and enhanced. Additionally, the environments experienced by the spacecraft during flight operations, including random vibration, pyrotechnic shock, load factors, and acoustics levels, will be maintained at or below Atlas IIIA levels for the Atlas V 400 configuration, and at or below Titan IV levels for the Atlas V HLV configuration.

Figure 8.2.2-4 shows how our Atlas V 400 configuration is evolved from our highly successful Atlas launch vehicle program. This design approach emphasizes using Atlas IIIA and IIIB components that will be flight verified before Atlas V use. Modifications include a new structurally stable Common Core Booster™ sized for the most stressing Atlas V mission and a 66-in. stretch to the single-engine Centaur that will be flight-proven on Atlas IIIA before Atlas V. Operability is enhanced by using a concept that ships fully checked-out, flight-ready vehicles elements to the launch site, including elimination of cryogenic propellant load testing (Wet Dress Rehearsal) at the launch site.

8.2.2.1 Atlas V 400 Performance Data—Detailed Atlas V 400 performance data are provided in Figures 8.2.2.1-1 through 8.2.2.1-4. Data shown for several types of launch missions are based on our estimate of the vehicles lift capability. Tabular performance data are provided in Tables 8.2.2.1-1 and 8.2.2.1-2. The performance groundrules are on each curve and additional information is provided in Section 2.0, Paragraphs 2.7.1 and 2.7.2.



PAYLOAD FAIRING

Features:	Medium	Extended (Shown)
Diameter:	3.3 m	4.2 m
Length:	10.4 m	12.9 m
Mass:	1,409 kg	2,255 kg
Subsystems:		
Fairing:	Aluminum Skin Stringer & Frame Bisector	
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Bisector	
Separation:	Pyro Bolts & Spring Thrusters	

CENTAUR UPPER STAGE—SINGLE-ENGINE CENTAUR

Features:	
Size:	3.05-m Dia x 12.68-m Length with Extended Nozzle
Inert Mass:	2,026 kg
Propellant:	20,799 kg LH_2 & LO_2
Guidance:	Inertial
Subsystems:	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	One Pratt & Whitney Restartable Engine
— Model:	RL10A-4-2
— Thrust:	99.2 kN
— I_{SP}:	450.5 s
	One Fixed 51-cm Columbium Nozzle Four 27-N Hydrazine Thrusters Eight 40.5-N Lateral Hydrazine Thrusters
Pneumatics:	Helium & Hydrogen Autogenous (Tank Pressurization)
Avionics:	Guidance, Navigation & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, Tracking, Range Safety Command, Electrical Power, Redundant EMA Actuators for Engine Gimbal Control
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

CENTAUR INTERSTAGE ADAPTER (ISA) (SHORT)

Features:	
Size:	3.05-m Dia x 3.13-m Length
Mass:	374 kg
Subsystems:	
Structure:	Aluminum Lithium Skin Stringer & Frame
Separation:	Low-Cost Atlas Separation System

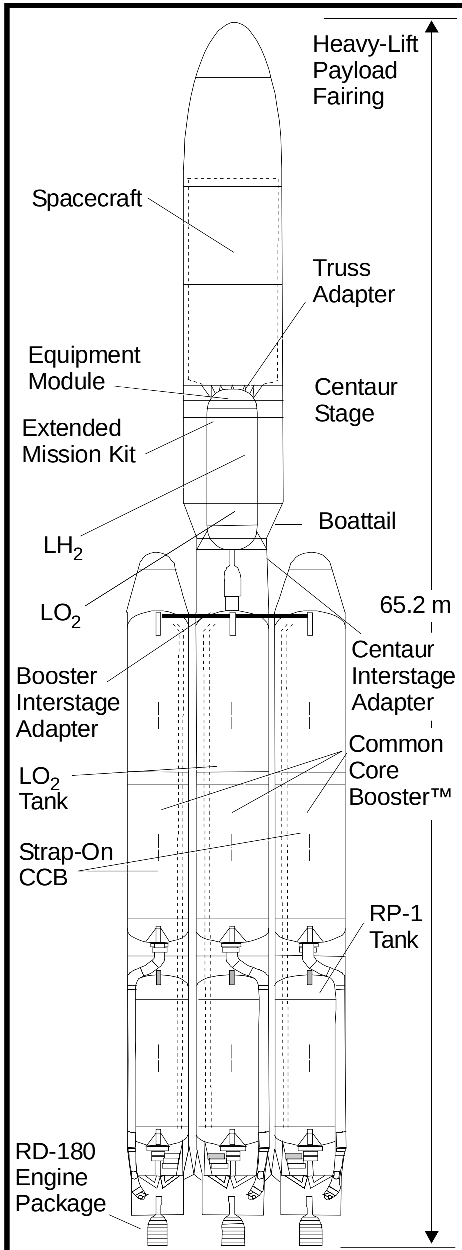
BOOSTER INTERSTAGE ADAPTER

Features:	
Size:	3.05-m Dia (Top) x 1.65-m Length 3.83-m Dia (Bottom)
Mass:	420 kg
Structure:	Composite (Graphite Epoxy) with Aluminum Ring Frames Fwd & Aft

COMMON CORE BOOSTER™

Features:	
Size:	3.81-m Dia x 32.46-m Length
Propellant:	284,453 kg LO_2 & RP-1
Guidance:	From Upper Stage
Subsystems:	
Structure:	Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure
Separation:	8 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers) SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
Pneumatics:	Helium (Pressure Tanks), Computer-Controlled Pressurization System
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Redundant Rate Gyro System, Electrical Power

Figure 8.2.2-2 The Atlas V 400 launch system is derived from flight-proven Atlas systems and is capable of meeting a wide variety of mission requirements.



PAYLOAD FAIRING

Features:
- Diameter: 5.4 m
- Length: 26.4 m Without Boattail
- Mass: 5,088 kg Without Boattail

Subsystems:
- Fairing: Bisector. Sandwich construction with graphite epoxy face sheets and an aluminum honeycomb core.
- Separation: Vertical Separation by means of a linear piston & cylinder activated by a pyrotechnic cord. Horizontal separation by means of an expanding tube shearing, a notched doubler, activated by a pyrotechnic cord.
- Boattail: Fixed, Composite Sandwich Const

CENTAUR UPPER STAGE—SINGLE-ENGINE CENTAUR

Features:
- Size: 3.05-m Dia x 12.68-m Length with Extended Nozzle
- Inert Mass: 2,026 kg
- Propellant: 20,799 kg LH_2 & LO_2
- Guidance: Inertial

Subsystems:
- Structure: Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
- Propulsion: One Pratt & Whitney Restartable Engine
  - — Model: RL10A-4-2
  - — Thrust: 99.2 kN
  - — I_{SP}: 450.5 s
  - One Fixed 51-cm Columbium Nozzle
  - Four 27-N Hydrazine Thrusters
  - Eight 40.5-N Lateral Hydrazine Thrusters
- Pneumatics: Helium & Hydrogen Autogenous (Tank Pressurization)
- Avionics: Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, Tracking, Range Safety Command, Electrical Power, Redundant EMA Actuators for Engine Gimbal Control
- Insulation: Polyvinyl Chloride Foam (1.6-cm Thick)

CENTAUR INTERSTAGE ADAPTER (ISA) (HLV-Upper)

Features:
- Size: 3.83-m Dia x 3.81-m Length
- Mass: 1,297kg

Subsystems:
- Structure: Composite (Nomex Core/Graphite Epoxy Face Sheets)
- Separation: Low-Cost Atlas Separation System

BOOSTER ISA (HLV-Lower)

Features:
- Size: 3.83-m Dia x 0.32-m Length
- Mass: 272 kg

Subsystems:
- Structure: Aluminum Machined Rolled-Ring Forging

COMMON CORE BOOSTER™

Features:
- Size: 3.81-m Dia x 32.46-m Length
- Propellant: 284,453 kg LO_2 & RP-1
- Guidance: From Upper Stage

Subsystems:
- Structure: Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure

Separation: 8 Retro Rockets

Propulsion: Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers)
SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s
Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s

Pneumatics: Helium (Pressure Tanks), Computer-Controlled Pressurization System

Hydraulics: Integral with Engine Provides Gimbal Control

Avionics: Flight Control, Flight Termination, Telemetry, Redundant Rate Gyro System, Electrical Power

STRAP-ON CCB™

Two Common Core Boosters™ with Nosecones & Strap-on Attach Hardware
- Size: 3.81-m Dia x 36.34-m Length
- Propellant: 284,453 Kg (Each)
- Thrust: SL 100% = 3,827 kN, I_{SP} 311.3 s (Each)
  Vac 100% = 4,152 kN, I_{SP} 337.8 s (Each)

EXTENDED MISSION KIT

Centaur Tank Sidewall & Component Radiation Shields
Additional Hydrazine Bottle
Additional Helium Bottle

Figure 8.2.2-3 The Atlas V HLV launch system is derived from flight-proven Atlas systems and provides heavy lift capability.

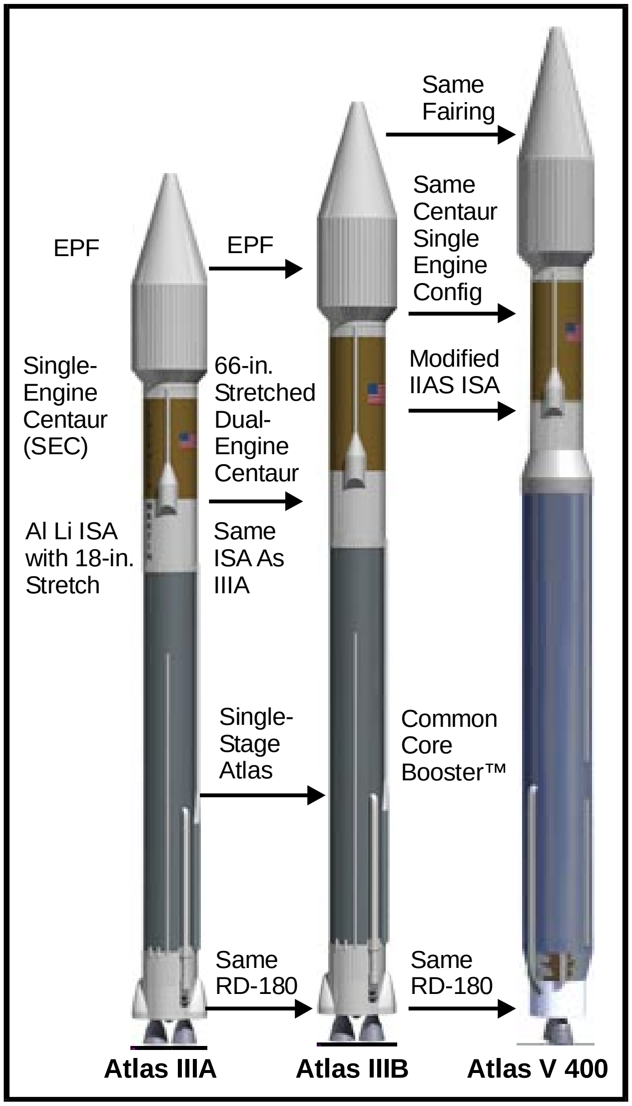


Figure 8.2.2-4 Atlas III to Atlas V 400 Evolution

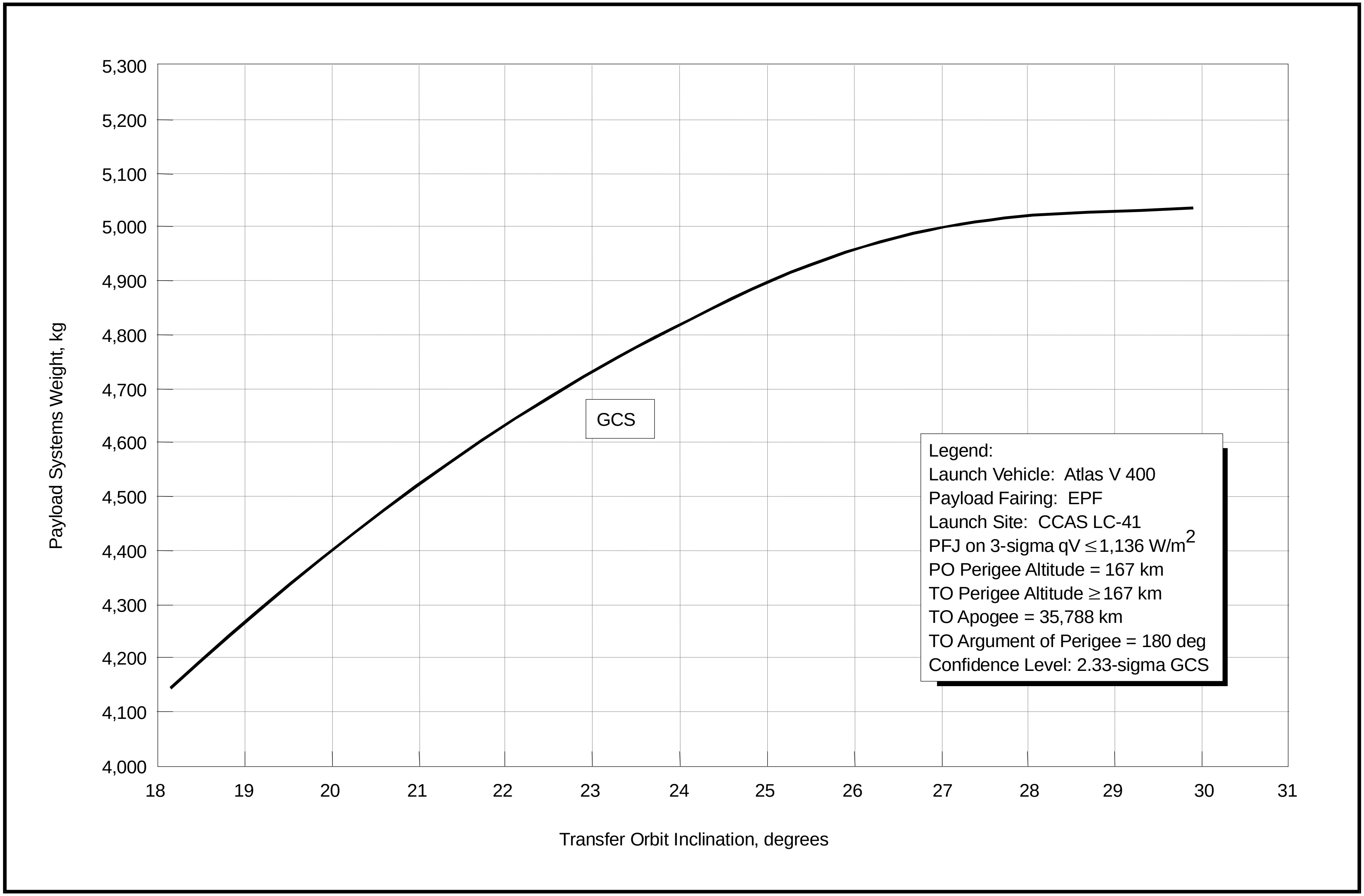


Figure 8.2.2.1-1 Atlas V 400 Reduced Inclination Elliptical Orbit Performance (Metric)

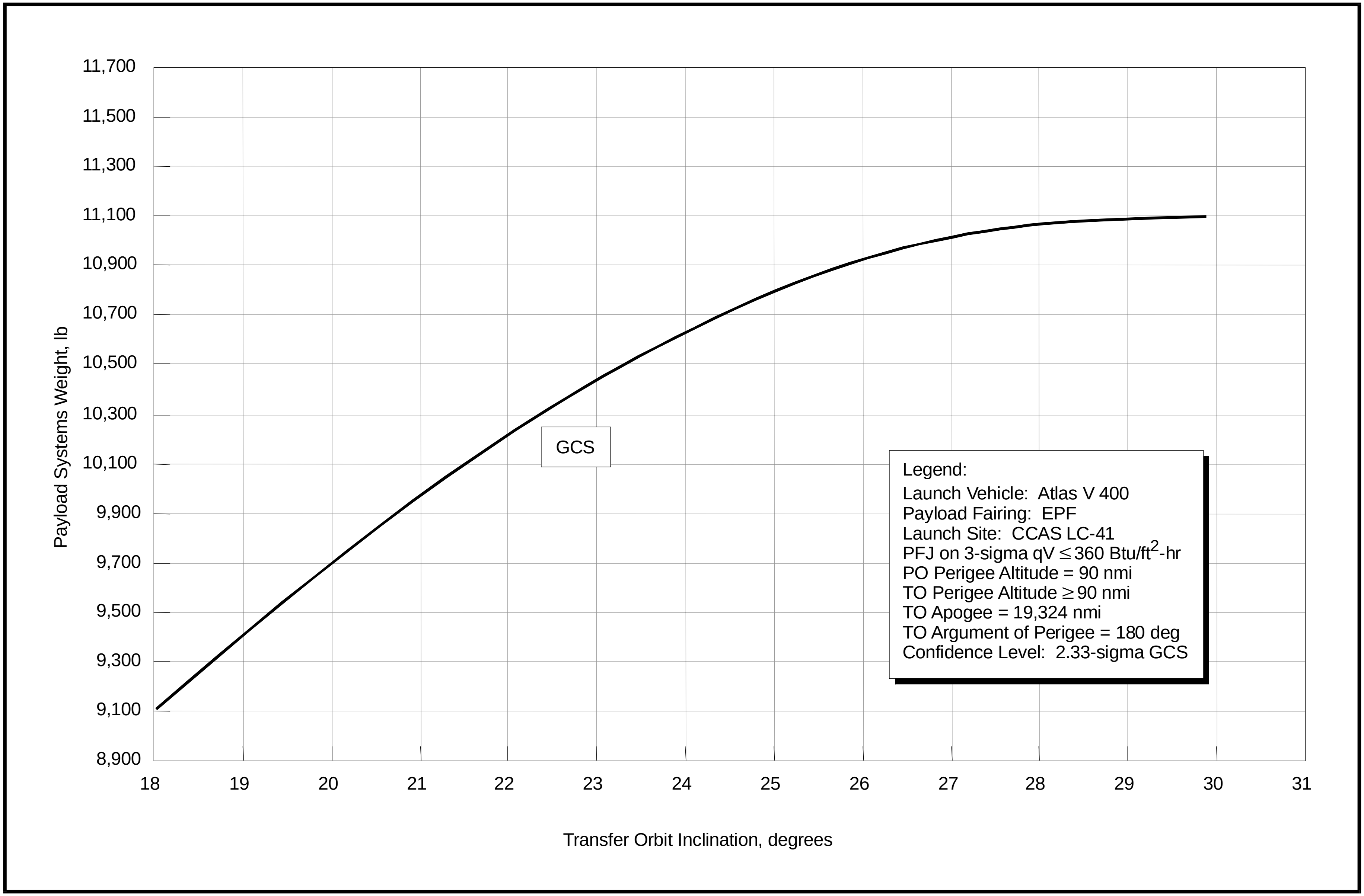


Figure 8.2.2.1-2 Atlas V 400 Reduced Inclination Elliptical Orbit Performance

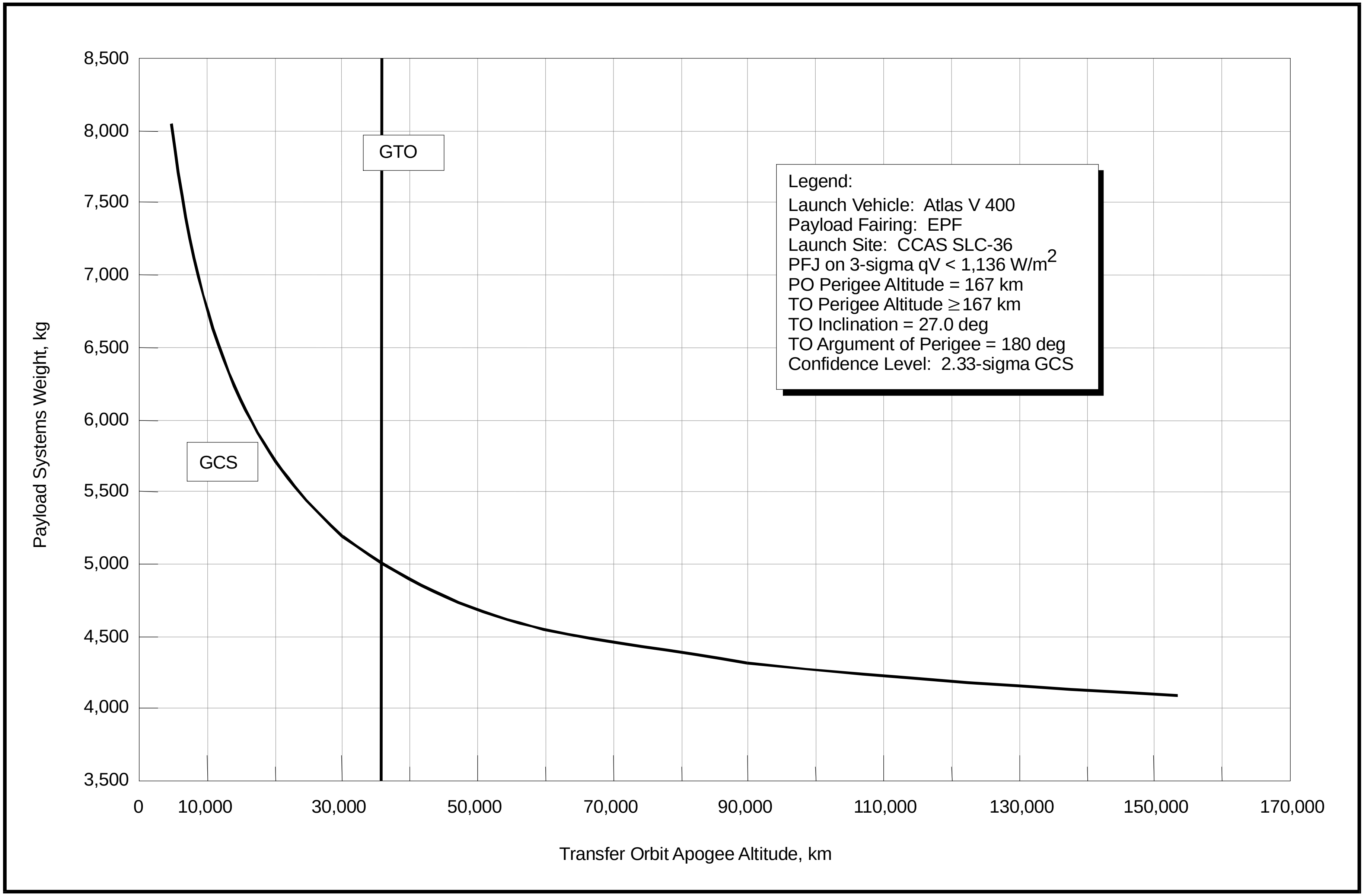


Figure 8.2.2.1-3 Atlas V 400 Performance to Elliptical Transfer Orbit (Metric)

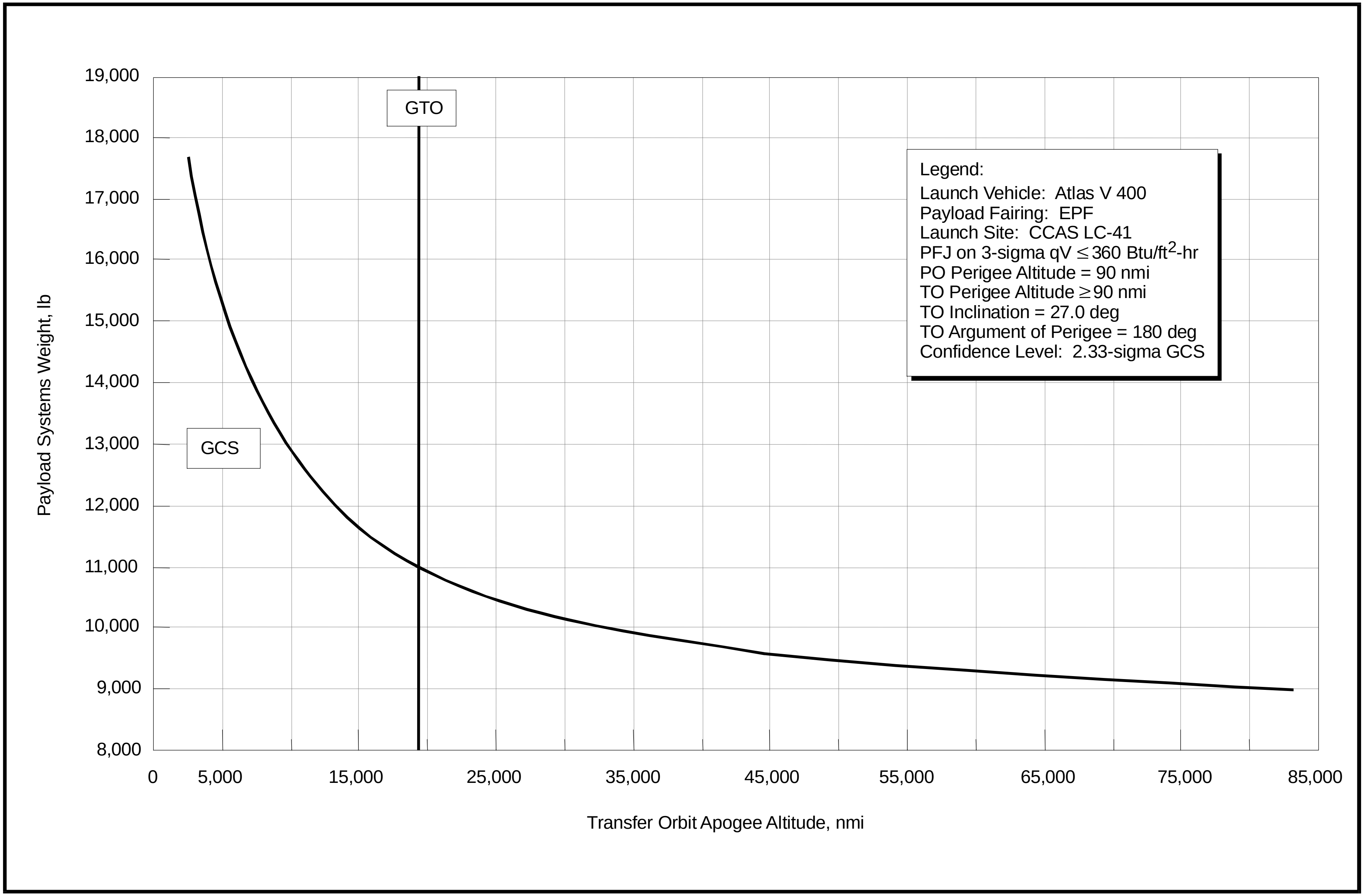


Figure 8.2.2.1-4 Atlas V 400 Performance to Elliptical Transfer Orbit

Table 8.2.2.1-1 Atlas V 400 Elliptical Transfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload Systems Weight, kg (lb)	
km	(nmi)	GCS	
4,630	(2,500)	8,026	(17,695)
5,556	(3,000)	7,754	(17,095)
6,482	(3,500)	7,510	(16,557)
7,408	(4,000)	7,290	(16,072)
8,334	(4,500)	7,094	(15,640)
9,260	(5,000)	6,919	(15,254)
10,186	(5,500)	6,759	(14,900)
11,112	(6,000)	6,615	(14,583)
12,038	(6,500)	6,482	(14,289)
13,427	(7,250)	6,306	(13,902)
14,816	(8,000)	6,150	(13,557)
16,205	(8,750)	6,012	(13,255)
17,594	(9,500)	5,891	(12,988)
18,983	(10,250)	5,779	(12,741)
20,372	(11,000)	5,680	(12,522)
23,150	(12,500)	5,506	(12,139)
25,928	(14,000)	5,362	(11,822)
28,706	(15,500)	5,241	(11,554)
31,484	(17,000)	5,135	(11,322)
34,262	(18,500)	5,045	(11,123)
35,788	(19,324)	5,000	(11,023)
37,966	(20,500)	4,940	(10,892)
40,744	(22,000)	4,872	(10,740)
43,522	(23,500)	4,814	(10,612)
46,300	(25,000)	4,759	(10,492)
49,078	(26,500)	4,710	(10,385)
51,856	(28,000)	4,667	(10,288)
54,634	(29,500)	4,624	(10,194)
57,412	(31,000)	4,588	(10,115)
62,968	(34,000)	4,525	(9,975)
68,524	(37,000)	4,469	(9,852)
74,080	(40,000)	4,421	(9,746)
79,636	(43,000)	4,380	(9,655)
87,044	(47,000)	4,332	(9,549)
92,600	(50,000)	4,299	(9,478)
98,156	(53,000)	4,272	(9,418)
109,268	(59,000)	4,224	(9,312)
120,380	(65,000)	4,185	(9,226)
131,492	(71,000)	4,150	(9,149)
142,604	(77,000)	4,122	(9,088)
153,716	(83,000)	4,097	(9,031)

Note: Extended Payload Fairing (4.2-m Diameter, 12.9-m Length); Jettison at 3-sigma qV≤1,009 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude = 166.7 km (90 nmi); Transfer Orbit Perigee Altitude ≥ 166.7 (90 nmi); Transfer Orbit Inclination = 27°; Argument of Perigee = 180°

Table 8.2.2.1-2 Atlas V 400 Performance to Reduced Inclination Transfer Orbit—PSW vs Orbit Inclination

Inclination, °	Payload Systems Weight, kg (lb)	
	GCS	
18.0	4,124	(9,091)
18.5	4,195	(9,248)
19.0	4,264	(9,401)
19.5	4,331	(9,549)
20.0	4,397	(9,693)
20.5	4,459	(9,831)
21.0	4,520	(9,964)
21.5	4,577	(10,091)
22.0	4,632	(10,213)
22.5	4,684	(10,327)
23.0	4,733	(10,436)
23.5	4,779	(10,537)
24.0	4,822	(10,630)
24.5	4,861	(10,717)
25.0	4,897	(10,795)
25.5	4,928	(10,865)
26.0	4,956	(10,927)
26.5	4,980	(10,979)
27.0	5,000	(11,023)
27.5	5,013	(11,052)
28.0	5,022	(11,071)
28.5	5,028	(11,085)
29.0	5,031	(11,091)
29.5	5,033	(11,095)
30.0	5,033	(11,096)

Note: Extended Payload Fairing (4.2-m Dia, 12.9-m Length); Jettison at 3-sigma qv≤1,009 W/m^2); (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude = 166.7 km (90 nmi); Transfer Orbit Perigee Altitude ≥ 166.7 km (90 nmi); Transfer Orbit Apogee Altitude = 35,788 km (19,324 nmi); Argument of Perigee = 180°

APPENDIX A—ATLAS HISTORY, VEHICLE DESIGN, AND PRODUCTION

The Atlas/Centaur launch vehicle is manufactured and operated by Lockheed Martin to meet commercial and government medium and intermediate space lift needs.

A.1 VEHICLE DEVELOPMENT

Atlas—The Atlas vehicle has evolved through various United States Air Force (USAF) and National Aeronautics and Space Administration (NASA) programs over the years. The result is an efficient, highly reliable launch system. The Atlas team's development experience dates from the mid-1940s, when initial studies began to explore the feasibility of long-range ballistic missiles (Fig. A.1-1). Since the first research and development (R&D) launch in 1957, more than 600 Atlas vehicles have been manufactured, and more than 500 have flown. Versions of Atlas were built specifically for manned and unmanned space missions and as a booster for Centaur, the high-energy upper stage initially designed to launch NASA space probes on lunar and planetary missions. As payload weights increased, requirements were met with Atlas improvements, including increased propellant tank sizes and upgraded engine performance (Fig. A.1-2).

In 1981, the Atlas G booster was developed to improve Atlas/Centaur performance by increasing propellant capacity and upgrading engine thrust. The General Dynamics Space Systems Division (now Lockheed Martin Astronautics [LMA]) developed this baseline into today's Atlas I, II, IIA, and IIAS launch vehicle family. Atlas IIIA, with an initial launch capability (ILC) of 1998, continues the evolution of the Atlas launch vehicle family.

Centaur—Development began on the Centaur high-energy upper stage in 1958 as a means to launch NASA spacecraft on lunar and planetary missions. NASA's ambitious planetary exploration goals also required the development of improved avionics capable of guiding the Surveyor soft-lander space probes on lunar missions. Throughout the operational history of Centaur, systematic upgrades to the avionics have provided outstanding orbital insertion accuracy with balanced cost and weight considerations. Centaur's evolution to reach the current Centaur IIA/IIAS and planned IIIA single-engine (SEC)/Centaur, IIIB dual engine (DEC) and SEC configurations is depicted in Figure A.1-3.

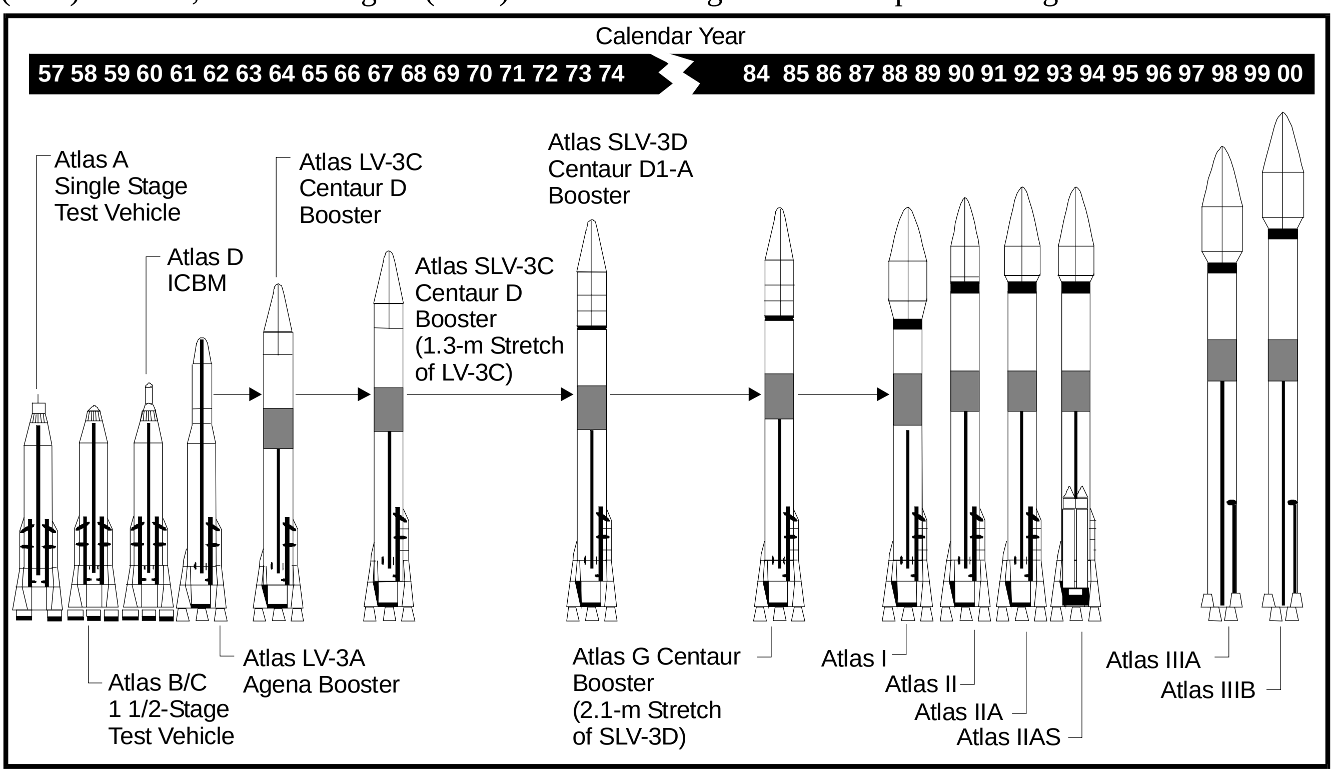


Figure A.1-1 Atlas has successfully evolved to satisfy requirements of many missions.

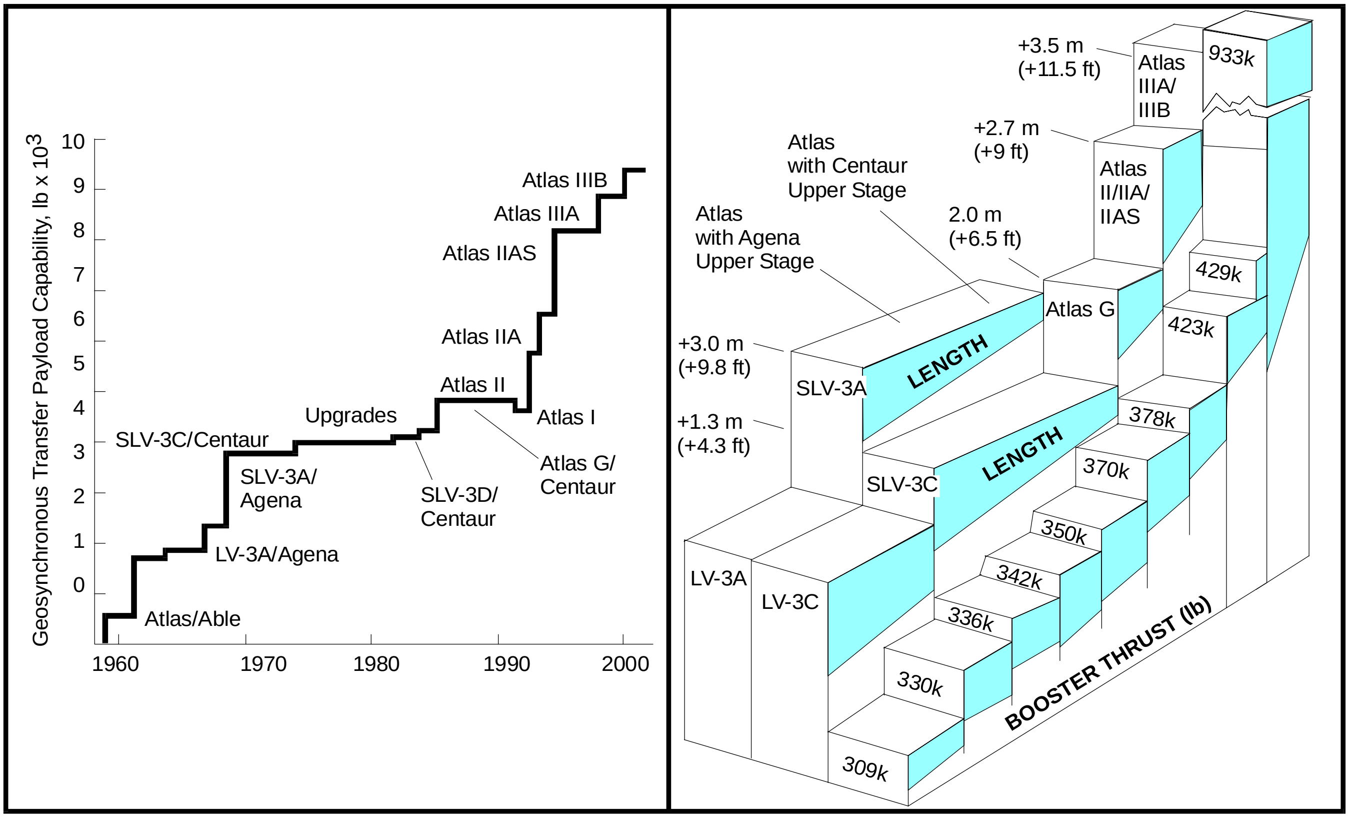


Figure A.1-2 Atlas performance has increased with spacecraft requirements.

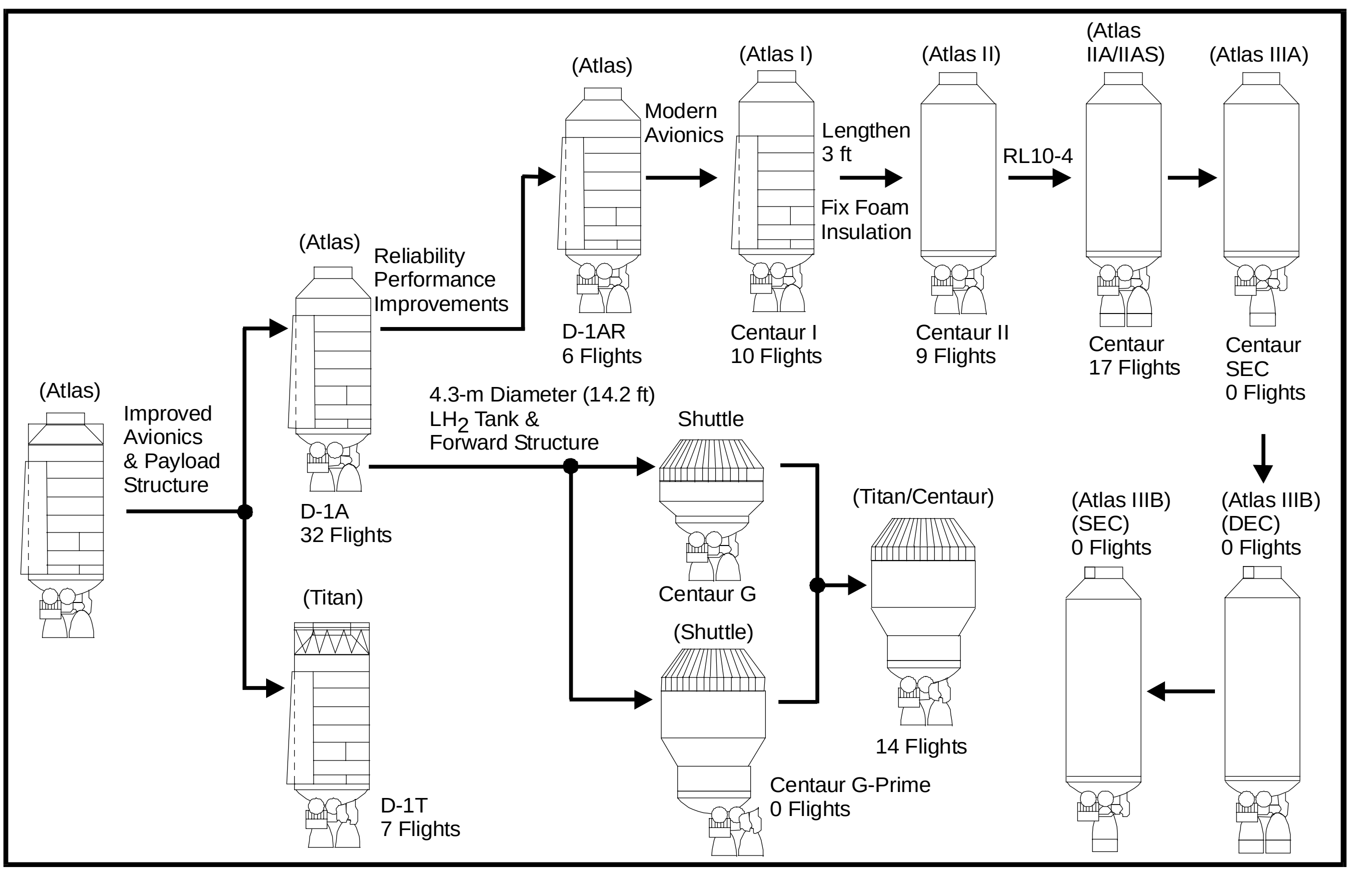


Figure A.1-3 Centaur evolution reflects changes to accommodate other boost vehicles and performance and reliability improvements.

The first successful flight of Centaur atop an Atlas launch vehicle occurred in November 1963. This was the world's first in-flight ignition of a hydrogen-powered vehicle. Three years later, Centaur performed the first successful space restart of liquid hydrogen (LH_2) engines in October 1966. With this flight, the Centaur R&D phase was completed, and Centaur became fully operational. The Centaur upper stage has now flown more than 100 missions on both the Atlas and Titan boosters. Throughout this history, the design has been refined and enhanced, enabling Centaur to remain the most efficient upper stage in the world today.

Atlas/Centaur—More than 90 spacecraft of diverse types and applications have been integrated and launched on the Atlas/Centaur space launch system during the past 30-plus years. The delivery of lunar and planetary missions to precise orbits has been most noteworthy. In addition, a wide range of Earth-orbiting satellite platforms have also been accommodated. Figure A.1-4 illustrates the diverse range of Atlas/Centaur missions and launch services users. Table A.1-1 chronologically lists each flight with additional details concerning mission type and mission status.

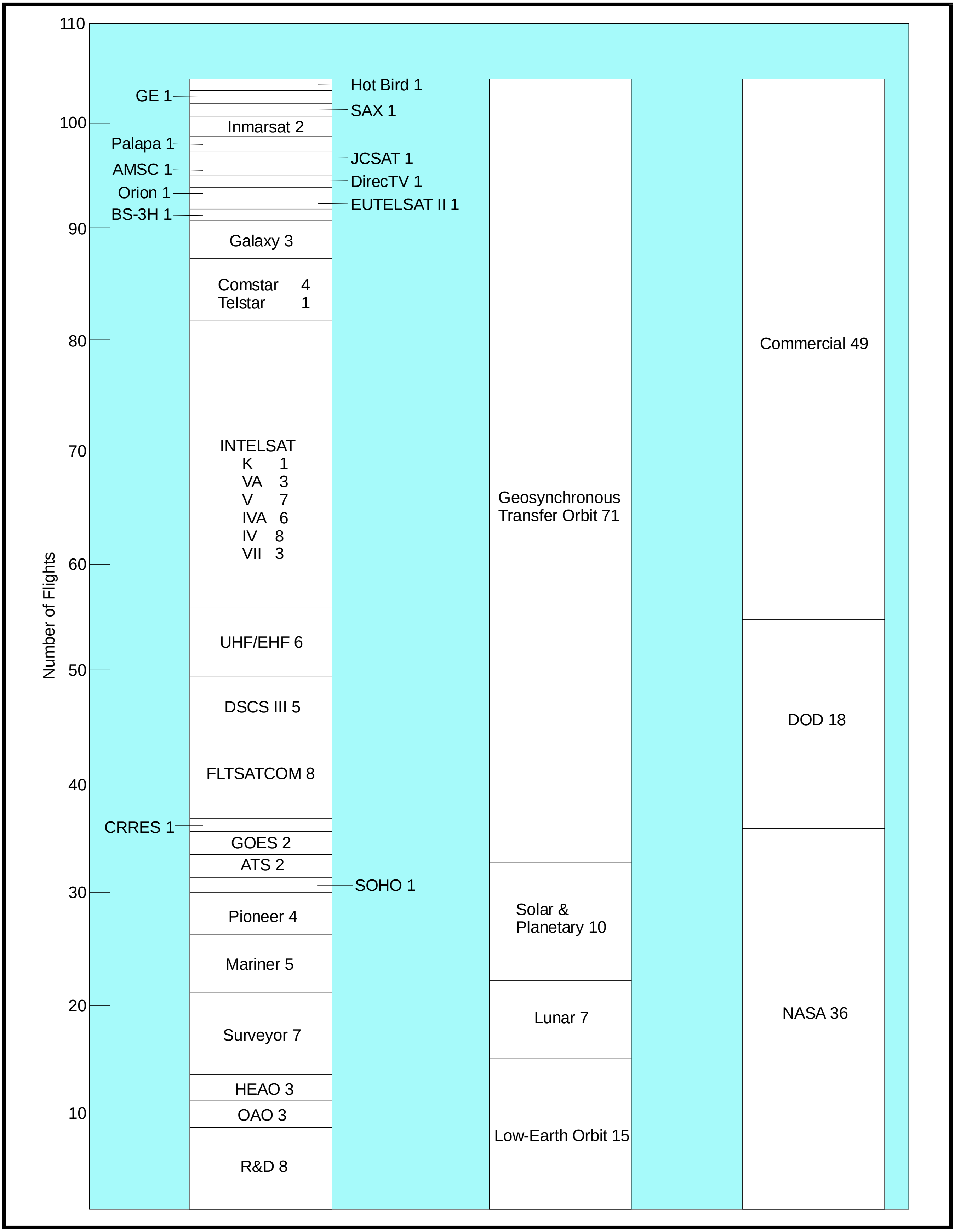


Figure A.1-4 Diverse payloads demonstrate Atlas versatility and adaptability.

Table A.1-1 Atlas/Centaur Launch History

Date	Mission	Vehicle	LV Type	Mission Type	LV Results
1962					
May 8	R&D	AC-1	LV-3C/A	R&D	Failure
1963					
November 27	R&D	AC-2	LV-3C/B	R&D	Success
1964					
June 30	R&D	AC-3	LV-3C/C	R&D	Failure
December 11	R&D	AC-4	LV-3C/C	R&D	Success
1965					
March 2	R&D	AC-5	LV-3C/C	R&D	Failure
August 11	R&D	AC-6	LV-3C/D	R&D	Success
1966					
April 7	R&D	AC-8	LV-3C/D	R&D	Failure
May 30	Surveyor	AC-10	LV-3C/D	Lunar Intercept	Success
September 20	Surveyor	AC-7	LV-3C/D	Lunar Intercept	Success
October 26	Mariner Mars	AC-9	LV-3C/D	Interplanetary	Success
1967					
April 17	Surveyor	AC-12	LV-3C/D	Lunar Intercept	Success
July 14	Surveyor	AC-11	LV-3C/D	Lunar Intercept	Success
September 8	Surveyor	AC-13	SLV-3C/D	Lunar Intercept	Success
November 7	Surveyor	AC-14	SLV-3C/D	Lunar Intercept	Success
1968					
January 7	Surveyor	AC-15	SLV-3C/D	Lunar Intercept	Success
August 10	ATS-D	AC-17	SLV-3C/D	GTO	Failure
December 7	OAO-A	AC-16	SLV-3C/D	LEO	Success
1969					
February 24	Mariner Mars	AC-20	SLV-3C/D	Interplanetary	Success
March 27	Mariner Mars	AC-19	SLV-3C/D	Interplanetary	Success
August 12	ATS-E	AC-18	SLV-3C/D	GTO	Success
1970					
November 30	OAO-B	AC-21	SLV-3C/D	LEO	Failure
1971					
January 25	INTELSAT IV	AC-25	SLV-3C/D	GTO	Success
May 8	Mariner Mars	AC-24	SLV-3C/D	Interplanetary	Failure
May 30	Mariner Mars	AC-23	SLV-3C/D	Interplanetary	Success
December 19	INTELSAT IV	AC-26	SLV-3C/D	GTO	Success
1972					
January 22	INTELSAT IV	AC-28	SLV-3C/D	GTO	Success
March 2	Pioneer F	AC-27	SLV-3C/D	Interplanetary	Success
June 13	INTELSAT IV	AC-29	SLV-3C/D	GTO	Success

Table A.1-1 (cont)

	Mission	Vehicle	LV Type	Mission Type	LV Results
1972 (cont)					
August 21	OAO-C	AC-22	SLV-3C/D	LEO	Success
1973					
April 5	Pioneer G	AC-30	SLV-3D/D-1A	Interplanetary	Success
August 23	INTELSAT IV	AC-31	SLV-3D/D-1A	GTO	Success
November 3	Mariner Venus/ Mercury	AC-34	SLV-3D/D-1A	Interplanetary	Success
1974					
November 21	INTELSAT IV	AC-32	SLV-3D/D-1A	GTO	Success
1975					
February 20	INTELSAT IV	AC-33	SLV-3D/D-1A	GTO	Failure
May 22	INTELSAT IV	AC-35	SLV-3D/D-1A	GTO	Success
September25	INTELSAT IVA	AC-36	SLV-3D/D-1AR	GTO	Success
1976					
January 29	INTELSAT IVA	AC-37	SLV-3D/D-1AR	GTO	Success
May 13	COMSTAR	AC-38	SLV-3D/D-1AR	GTO	Success
July 22	COMSTAR	AC-40	SLV-3D/D-1AR	GTO	Success
1977					
May 26	INTELSAT IVA	AC-39	SLV-3D/D-1AR	GTO	Success
August 12	HEAO-A	AC-45	SLV-3D/D-1AR	LEO	Success
September 29	INTELSAT IVA	AC-43	SLV-3D/D-1AR	GTO	Failure
1978					
January 6	INTELSAT IVA	AC-46	SLV-3D/D-1AR	GTO	Success
February 9	FLTSATCOM	AC-44	SLV-3D/D-1AR	GTO	Success
March 31	INTELSAT IVA	AC-48	SLV-3D/D-1AR	GTO	Success
May 20	Pioneer Venus	AC-50	SLV-3D/D-1AR	Interplanetary	Success
June 29	COMSTAR	AC-41	SLV-3D/D-1AR	GTO	Success
August 8	Pioneer Venus	AC-51	SLV-3D/D-1AR	Interplanetary	Success
November 13	HEAO-B	AC-52	SLV-3D/D-1AR	LEO	Success
1979					
May 4	FLTSATCOM	AC-47	SLV-3D/D-1AR	GTO	Success
September 20	HEAO-C	AC-53	SLV-3D/D-1AR	LEO	Success
1980					
January 17	FLTSATCOM	AC-49	SLV-3D/D-1AR	GTO	Success
October 30	FLTSATCOM	AC-57	SLV-3D/D-1AR	GTO	Success
December 6	INTELSAT V	AC-54	SLV-3D/D-1AR	GTO	Success
1981					
February 21	COMSTAR	AC-42	SLV-3D/D-1AR	GTO	Success
May 23	INTELSAT V	AC-56	SLV-3D/D-1AR	GTO	Success
August 5	FLTSATCOM	AC-59	SLV-3D/D-1AR	GTO	Success

Table A.1-1 (cont)

Date	Mission	Vehicle	LV Type	Mission Type	LV Results
1981 (cont)					
December 15	INTELSAT V	AC-55	SLV-3D/D-1AR	GTO	Success
1982					
March 4	INTELSAT V	AC-58	SLV-3D/D-1AR	GTO	Success
September 28	INTELSAT V	AC-60	SLV-3D/D-1AR	GTO	Success
1983					
May 19	INTELSAT V	AC-61	SLV-3D/D-1AR	GTO	Success
1984					
June 9	INTELSAT VA	AC-62	G/D-1AR	GTO	Failure
1985					
March 22	INTELSAT VA	AC-63	G/D-1AR	GTO	Success
June 29	INTELSAT VA	AC-64	G/D-1AR	GTO	Success
September 28	INTELSAT VA	AC-65	G/D-1AR	GTO	Success
1986					
December 4	FLTSATCOM	AC-66	G/D-1AR	GTO	Success
1987					
March 26	FLTSATCOM	AC-67	G/D-1AR	GTO	No Trial
1989					
September 25	FLTSATCOM	AC-68	G/D-1AR	GTO	Success
1990					
July 25	CRRES	AC-69	Atlas I	Elliptical	Success
1991					
April 18	BS-3H	AC-70	Atlas I	GTO	Failure
December 7	EUTELSAT II	AC-102	Atlas II	Supersynchronous	Success
1992					
February 10	DSCS III B14	AC-101	Atlas II	GTO	Success
March 13	Galaxy V	AC-72	Atlas I	GTO	Success
June 10	INTELSAT-K	AC-105	Atlas IIA	GTO	Success
July 2	DSCS III B12	AC-103	Atlas II	GTO	Success
August 22	Galaxy I-R	AC-71	Atlas I	GTO	Failure
1993					
March 25	UHF F/O F1	AC-74	Atlas I	Subsynchronous	Anomaly
July 19	DSCS III B9	AC-104	Atlas II	GTO	Success
September 3	UHF F/O F2	AC-75	Atlas I	Subsynchronous	Success
November 28	DSCS III B10	AC-106	Atlas II	GTO	Success
December 15	Telstar 401	AC-108	Atlas IIAS	GTO	Success
1994					
April 13	GOES I	AC-73	Atlas I	Supersynchronous	Success
June 24	UHF F/O F3	AC-76	Atlas I	Subsynchronous	Success

Table A.1-1 (concl)

Date	Mission	Vehicle	LV Type	Mission Type	LV Results
1994 (cont)					
August 3	DirecTV D2	AC-107	Atlas IIA	Supersynchronous	Success
October 6	INTELSAT 703	AC-111	Atlas IIAS	Supersynchronous	Success
November 29	Orion F1	AC-110	Atlas IIA	Supersynchronous	Success
1995					
January 10	INTELSAT 704	AC-113	Atlas IIAS	Supersynchronous	Success
January 28	EHF F4	AC-112	Atlas II	Subsynchronous	Success
March 23	INTELSAT 705	AC-115	Atlas IIAS	Supersynchronous	Success
April 7	AMSC-1	AC-114	Atlas IIA	Supersynchronous	Success
May 23	GOES-J	AC-77	Atlas I	Supersynchronous	Success
May 31	EHF F5	AC-116	Atlas II	Subsynchronous	Success
July 31	DSCS III B7	AC-118	Atlas IIA	GTO	Success
August 28	JCSAT-3	AC-117	Atlas IIAS	Supersynchronous	Success
October 22	EHF F6	AC-119	Atlas II	Subsynchronous	Success
December 2	SOHO	AC-121	Atlas IIAS	Libration Point	Success
December 14	Galaxy IIIR	AC-120	Atlas IIA	Subsynchronous	Success
1996					
January 31	Palapa	AC-126	Atlas IIAS	Supersynchronous	Success
April 3	Inmarsat 1	AC-122	Atlas IIA	GTO	Success
April 30	SAX	AC-78	Atlas I	LEO	Success
July 25	EHF F7	AC-125	Atlas II	Subsynchronous	Success
September 8	GE-1	AC-123	Atlas IIA	Supersynchronous	Success
November 21	Hotbird	AC-124	Atlas IIA	GTO	Success
December 17	Inmarsat 3	AC-129	Atlas IIA	GTO	Success
1997					
February 16	JCSAT 4	AC-127	Atlas IIAS	Supersynchronous	Success
March 8	DBS/TEMPO	AC-128	Atlas IIA	Subsynchronous	Success
April 25	GOES K	AC-79	Atlas I	Supersynchronous	Success
July 27	Superbird C	AC-133	Atlas IIAS	Supersynchronous	Success
September 4	GE-3	AC-146	Atlas IIAS	Supersynchronous	Success
October 5	Echostar III	AC-135	Atlas IIAS	GTO	Success
October 24	DSCS III	AC-131	Atlas IIA	Subsynchronous	Success
December 8	Galaxy 8i	AC-149	Atlas IIAS	Supersynchronous	Success
1998					
January 29	Capricorn MLV #7	AC-109	Atlas IIA	GTO	Success
February 27	Intelsat 806	AC-151	Atlas IIAS	GTO	Success
March 16	UHF-FO F8	AC-132	Atlas II	Subsynchronous	Success
June 18	Intelsat 805	AC-153	Atlas IIAS	GTO	Success
October 9	HotBird 5	AC-134	Atlas IIA	GTO	Success
October 20	UHF-FO F9	AC-130	Atlas IIA	Subsynchronous	Success

A.2 VEHICLE DESIGN

The Atlas and Centaur stages for Atlas IIA, and Atlas IIAS are manufactured using common hardware components and production procedures. All hardware on each stage is fully qualified and flight-proven.

A.2.1 Atlas Major Characteristics

The Atlas booster is a 1-1/2-stage liquid oxygen (LO_2)/RP-1 powered vehicle having a constant 3.05-m (10-ft) diameter and a total length of 24.9 m (81.7 ft). The half stage is the booster section, which is jettisoned during the Atlas powered phase. The full stage is the sustainer section, which remains powered from liftoff to propellant depletion.

Atlas stage avionics are integrated with the Centaur avionics system for guidance, flight control, instrumentation, and sequencing functions. An external equipment pod houses such Atlas systems as flight termination, data acquisition, pneumatics, and instrumentation.

Atlas IIA/IIAS propulsion is provided by the Rocketdyne MA-5A engine system, which includes one sustainer and one two-chamber booster engine. All engines are ignited before liftoff and develop a total rated thrust of 2,180 kN (490,000 lbf). Nominal propulsion system operation is confirmed before committing the vehicle to flight. The Atlas IIAS booster augments core booster thrust with the addition of four Thiokol Castor IVA solid rocket boosters (SRB), each generating 433.7 kN (97,500 lbf) of thrust. The SRBs burn two at a time, augmenting thrust for the first ~ 110 seconds of flight.

The Atlas IIIA, with its single two-chamber Pratt & Whitney/NPO Energomash RD-180 engine, will operate as a single stage. Stage length is increased by 4.4 m (14.5 ft).

A.2.1.1 Structure—The Atlas structure consists of three basic elements: the sustainer section, the thrust section, and the interstage adapter (ISA). The entire structure is protected for corrosion control.

Sustainer Section—The pressure-stabilized fuel and oxidizer tanks are the major elements of the sustainer section (Fig. A.2.1.1-1). The propellant tanks are of thin-wall, fully monocoque, welded, stainless-steel construction. The RP-1 fuel tank and the LO_2 tank are separated by an ellipsoidal intermediate bulkhead. Structural integrity of the tanks is maintained in flight by a fault-tolerant pressurization system and on the ground by either internal tank pressure or by the application of mechanical stretch. The sustainer section supports the sustainer engine, equipment pods, and booster jettison tracks.

Figure A.2.1.1-1 The time-tested Atlas sustainer tank provides the primary interface for the booster section and Centaur upper stage.

The Atlas IIAS sustainer section is functionally similar to the Atlas II section, but includes an increase in tank skin gauges, the addition of the SRB load ring and forward cableway, and a revision to the booster staging system to accommodate the SRBs. (Fig. A.2.1.1-2).

The Atlas IIIA/IIIB tank is functionally similar to the Atlas IIAS tanks, but includes a 10-ft LO_2 tank stretch, new skin gauges, and an elliptical aft fuel bulkhead. Atlas IIIA/IIIB carries 12 retrorockets of the same Atlas IIA/IIAS design to compensate for the higher staging mass of the Atlas IIIA/IIIB booster.

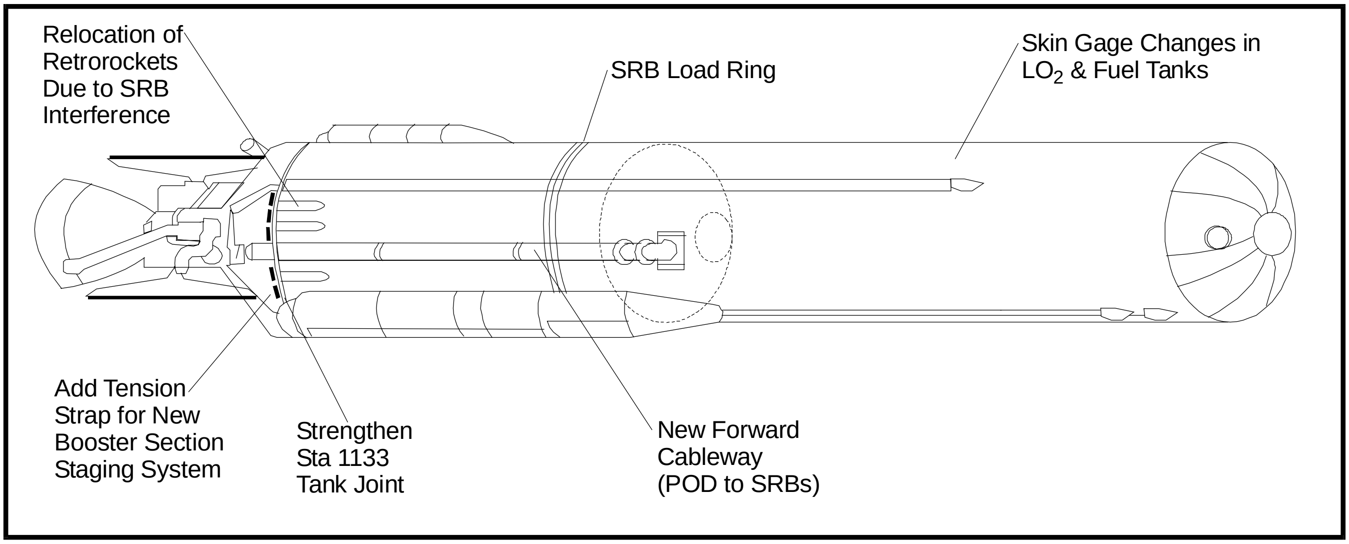


Figure A.2.1.1-2 Atlas Tank Design Updates

Figure A.2.1.1-3 The robust Atlas booster section is flight proven.

Thrust Section—The thrust section provides the structural support and an aerodynamic enclosure for the booster engine (Fig. A.2.1.1-3). The booster package is jettisoned about 165 seconds after launch. The Atlas IIA thrust section is a riveted skin and stringer construction. Cork heat shields and engine boots serve to limit the exposure of internal hardware to the base region heat loads. The thrust section is attached to the sustainer with 10 pneumatic latches.

For Atlas IIAS, integrally stiffened machined panels replace the II/IIA skin/stringer design to accommodate the SRBs (Fig. A.2.1.1-4). This thrust section is designed to manage the increased base region heat loads and is sealed to keep out fuel-rich combustibles. All Atlas IIAS booster staging system components are designed to react Castor IVA SRB loads, and the Atlas IIAS booster staging system design is functionally redundant.

The Atlas IIIA/IIIB booster section has a similar integrally machined panel design as Atlas IIAS, but with all provisions for booster jettison and airlit SRB attachment eliminated, such as separation latches, jettison tracks, and the second pair of SRB attach fittings. The 1206 frame is also upgraded to support the new RD-180 engine truss attachment interface.

The Atlas IIAS composite nacelle structures have been completely redesigned for Atlas IIIA/IIIB to enclose the larger RD-180 engine. The new cylindrical aluminum engine fairing design includes two externally mounted bottle fairings that contain six of 13 helium bottles used for tank pressurization. The IIIA/IIIB booster section, with both the thrust cylinder and engine/bottle fairings, is shown in Figure A.2.1.1-5.

Interstage Adapter (ISA)—The ISA provides a physical connection between the Atlas and the Centaur stages (Fig. A.2.1.1-6). The 3.96-m (13-ft) long cylindrical adapter is of aluminum skin/stringer construction. The Atlas IIA/IIAS common adapter bolts to the Centaur aft ring and the

Atlas forward ring. The ISA houses the hydrazine-based Atlas roll-control module (ARCM) and contains the Atlas/Centaur separation system.

The Atlas IIIA/IIIB ISA is 18-in. longer to accommodate the longer single-engine Centaur installation, but retains the same Atlas and Centaur structural interfaces. The IIIA/IIIB ISA used aluminum-lithium alloy skin and stringers for reduced weight compared to the aluminum of the IIA/IIAS design. The ARCM installation has been eliminated from Atlas IIIA/IIIB.

Atlas/Centaur Separation System—A pyrotechnically actuated flexible linear-shaped charge (LSC) separation system is used to sever the attachment of the Centaur vehicle from the Atlas stage. This system is fault-tolerant and ensures reliable severance of the ISA about 1.3 cm (0.5 in.) aft of the ISA/Centaur interface.

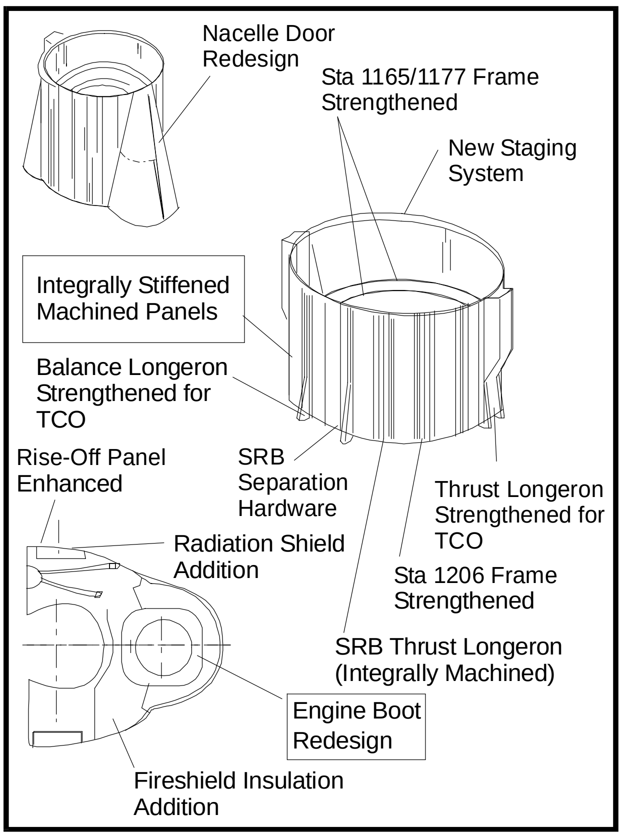


Figure A.2.1.1-4 The Atlas Booster IIAS Booster Thrust Structure

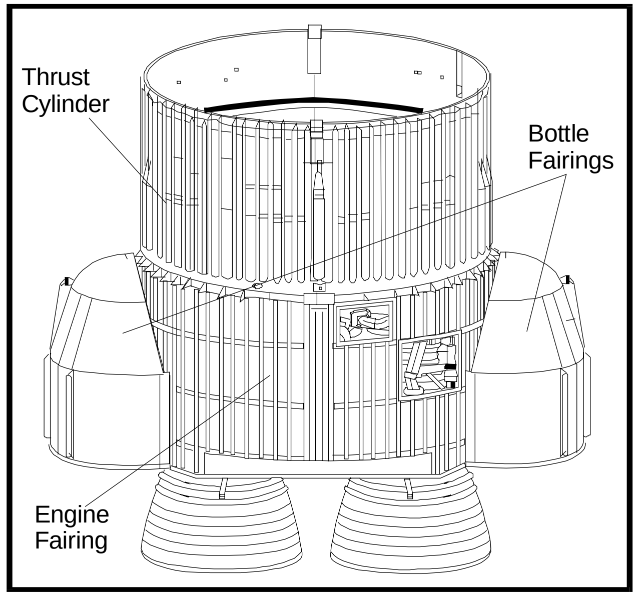


Figure A.2.1.1-5 IIIA Thrust Structure

Figure A.2.1.1-6 The interstage adapter provides the attachment between Atlas and Centaur stages.

A.2.1.2 Pneumatics—Tank pressures are controlled to maintain structural integrity and provide a positive suction head pressure to the engines.

Computer-Controlled Atlas Pressurization System (CCAPS)—CCAPS is a closed-loop tank pressure control system using flight software logic and is active before liftoff through booster engine cutoff (BECO). It controls ullage pressure in the launch vehicle RP-1 and LO_2 tanks by adding helium as required. Atlas operates under continually decaying residual pressure throughout the sustainer phase of flight. Redundant pressurization solenoid and pyrovalves control tank pressures within the desired limits. Before liftoff, the ground system pressure control unit and independent ground system relief valves provide tank pressurization and venting control.

Helium Supply System—Ten storage bottles supply helium for the pneumatics system and for tank pressure control on Atlas IIA/IIAS. This system is jettisoned with the booster section.

The Atlas IIIA/IIIB system carries 13 of the same bottles through the entire Atlas flight. The bottle arrangement has been reconfigured for IIIA/IIIB with seven bottles located in a ring in the base of the thrust section and three bottles in each of the two bottle fairings.

Engine Control System—This system consists of a single titanium bottle that supplies helium pressurant to the booster engine control package. A dedicated high-pressure helium bottle supplies the pneumatically operated booster section staging latches.

The RD-180 engine on Atlas IIIA/IIIB carries a similar engine controls bottle integral with the engine.

LO_2 Propellant Tank Vent System—An LO_2 boiloff valve controls the LO_2 propellant saturation pressure during cryogenic tanking. Just before launch, the valve is locked and remains locked throughout flight (because venting is not required).

A.2.1.3 Propulsion—The Atlas IIAS propulsion system consists of five flight-proven subsystems: MA-5A engine system; SRBs; SRB attach, disconnect, and jettison (ADJ) system; ARCM; and ground start system.

The Atlas IIIA/IIIB carries the single RD-180 engine system and a simplified ground start system. IIIA/IIIB eliminates the need for the ARCM.

MA-5A Engine System—Main propulsion is provided by the Rocketdyne MA-5A engine system. It is a calibrated, fixed-thrust, pump-fed system that uses a propellant combination of LO_2 (oxidizer) and RP-1 fuel for both the booster and sustainer engines (Fig. A.2.1.3-1). The MA-5A engine system uses two RS-27 engines packaged into one booster engine system. This engine system is jettisoned during the Atlas boost phase after the prescribed optimum axial acceleration is achieved. A single sustainer engine continues the boost phase until propellant depletion of the Atlas. All engines undergo hot-firing tests during individual engine acceptance tests.

RD-180 Engine System—The RD-180 engine, to be provided by a joint venture of Pratt & Whitney and NPO Energomash for use on Atlas IIIA/IIIB, is a two-chamber design fed by a common turbopump assembly. The RD-180 engine is throttleable from 40% to 100%, with a rated vacuum thrust of 933 klb. The two-chamber RD-180, shown in Figure A.2.1.3-2, is a derivative of the four-chamber RD-170 engine used on the Russian Zenit and Energia boosters. The RD-180 operates on a staged combustion cycle using the same LO_2 (oxidizer) and RP-1 (fuel) propellants as the MA-5A. It contains integral start systems and thrust vector control systems. Initial test firings of the RD-180 occurred in July 1997. Each RD-180 will undergo individual acceptance hot-fire tests with 3 tests planned for 1998.

Solid Rocket Booster (SRB)—The Atlas IIAS uses four Thiokol Castor IVA SRBs. Two are ignited at launch, and two are ignited in flight after burnout of the ground-lit pair.

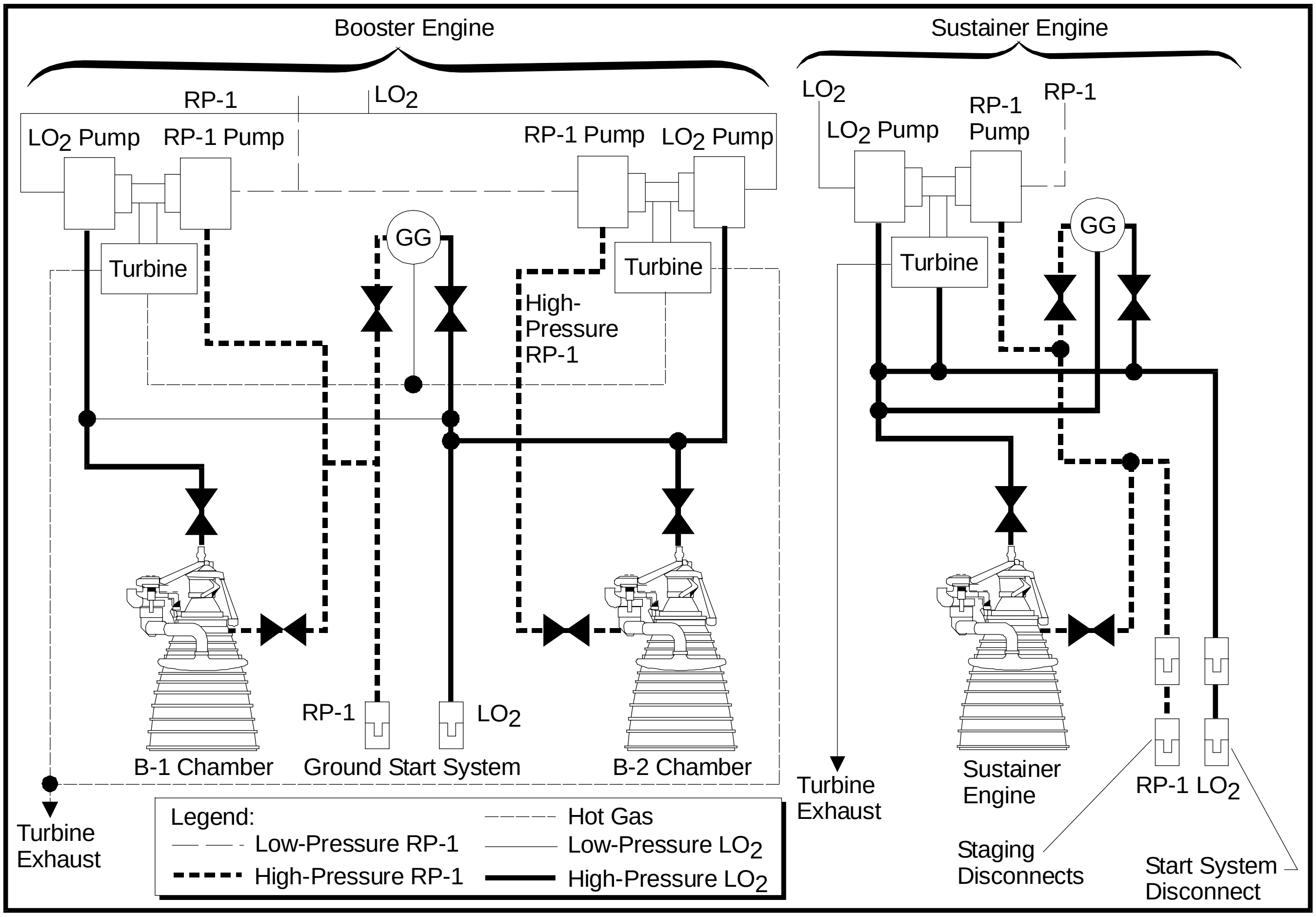


Figure A.2.1.3-1 The reliable flight-proven MA-5A booster system has demonstrated 100% Mission Success® accomplishment.

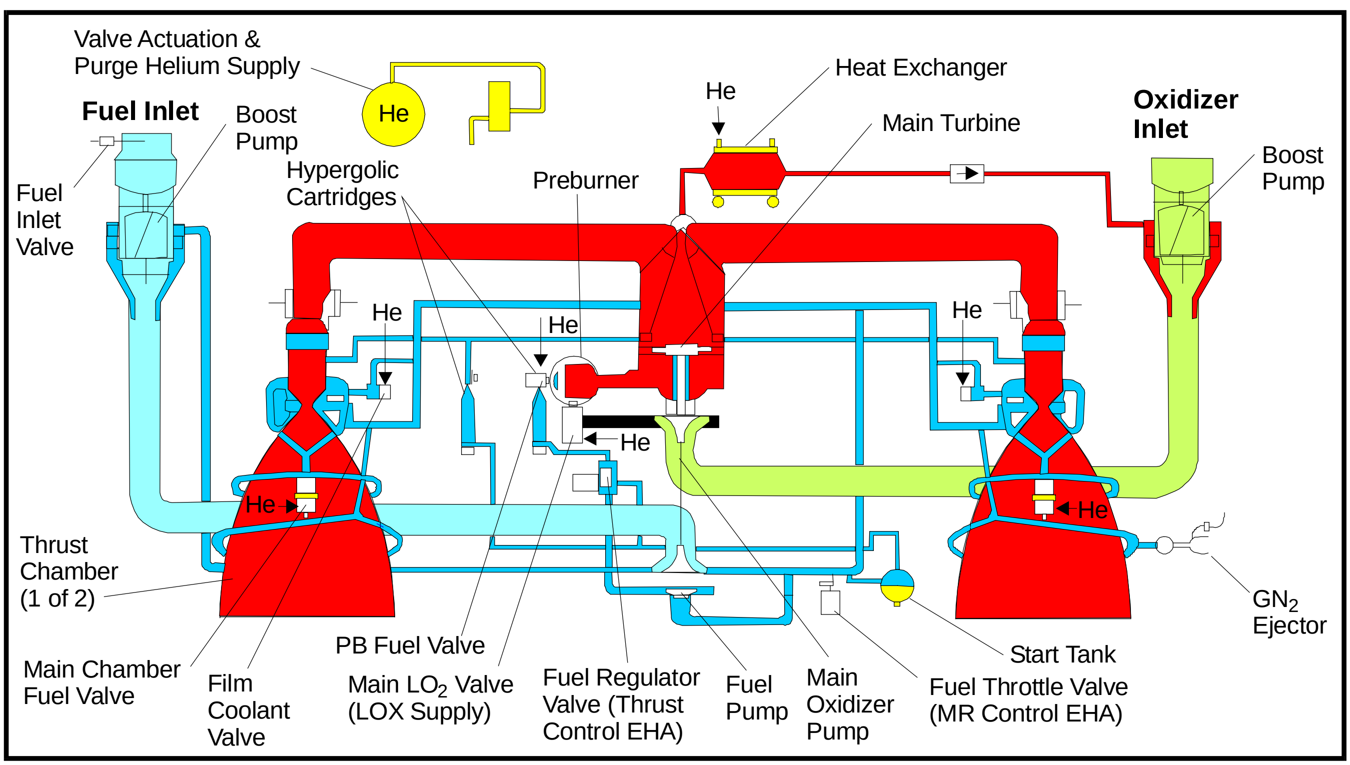


Figure A.2.1.3-2 RD-180 Engine Schematic

Attach, Disconnect, and Jettison (ADJ) System—The SRB ADJ system is designed to react SRB flight loads and safely jettison SRBs with significant outboard rotation.

ARCM—The MA-5A engine system provides roll control during the initial boost phase. The ARCM provides fault-tolerant roll control during the Atlas IIA/IIAS sustainer solo phase. This system includes four thrusters (222.4 N [50 lbf] of initial thrust per unit) with two thrusters aligned in each roll direction, a blow down hydrazine storage sphere, control valves, and associated plumbing.

Ground Start System—The ground start system supports ignition of the booster and sustainer engines. This system consists of pressure-rated bottles for RP-1 and LO_2 with the controls necessary to fill the bottles, pressurize them, feed RP-1 and LO_2 to the Atlas gas generators, and vent the system after the MA-5A engine start and before launch. The ground start system provides the propellants needed to start the gas generators that drive the engine propellant pumps until their output flow and pressure is sufficient to feed the gas generators and main engines (booster and sustainer).

The RD-180 engine carries integral start pressurization bottle and hypergol ampoules. The ground system provides hydraulic pressure to operate the start valve.

A.2.1.4 Hydraulic Thrust Vector Control (TVC)—The Atlas MA-5A hydraulic system is composed of two independent systems, one for the booster and one for the sustainer. Hydraulic pumps, driven from the engine turbopump accessory drive pads, provide hydraulic pressure and fluid flow to the servo actuator assemblies for gimbal control of the engine thrust chambers. Pitch, yaw, and roll control of the vehicle during the booster phase of flight and pitch and yaw control during the sustainer phase of flight are provided. The sustainer system also provides fluid power to the MA-5A control package to allow propellant flow control. Hydraulic capability loss is prevented by using RP-1 fuel, which can be directed into the hydraulic system.

The RD-180 engine used on the Atlas IIIA/IIIB carries an integral TVC system that is powered by high-pressure RP-1 tapped-off from the main fuel pump. This eliminates the need for a dedicated hydraulic pump. Two servoactuators on each thrust chamber provide pitch, yaw, and roll control throughout booster flight.

A.2.1.5 Avionics—The avionics system consists of four flight-proven subsystems: flight control, telemetry, flight termination, and electrical power. All avionics packages are line-replaceable in the field.

Flight Control Hardware (Atlas IIA/IIAS)—Atlas IIA/IIAS flight control hardware consists of the Atlas servo inverter unit (SIU) and two rate gyro units (RGU). It is integrated with Centaur avionics. All aspects of guidance, flight control, and vehicle sequencing are provided by the Centaur inertial navigation unit (INU) (Fig. A.2.1.5-1). Attitude control is accomplished by a closed-loop hydraulic servo positioning system for the booster engine. The RCU provides alternating current power (synchronized by a Centaur reference signal) to the RGUs. Two RGUs, one forward and one aft, provide pitch and yaw rate signals to the Centaur INU for vehicle stabilization. The RGUs provide true body rates to the INU. The inertial measurement subsystem provides vehicle roll rates and attitude for roll control. The flight control system provides the capability to ignite two SRBs during flight. This is accomplished with two initiation units and a pyrotechnic battery. The flight control subsystem (FCS) has the capability to provide two SRB jettison events. This is accomplished with four pyrotechnic control units (two for each separation event).

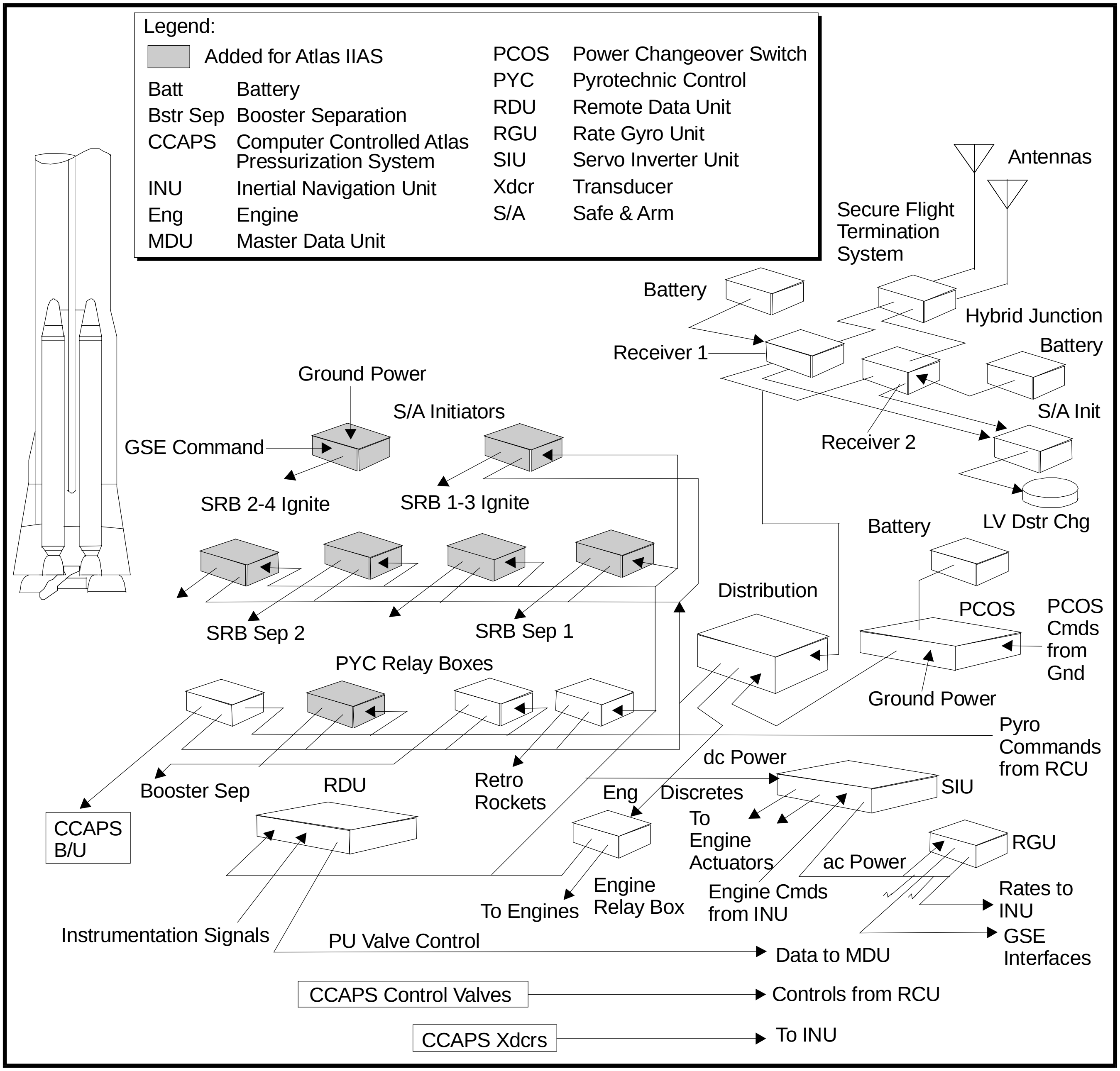


Figure A.2.1.5-1 State-of-the-art avionics integrates the Atlas stage with the Centaur stage.

Flight Control Hardware (Atlas IIIA/IIIB)—Atlas IIIA/IIIB flight control hardware consists of the Atlas remote control unit (RCU) and two RGUs. It is integrated with Centaur avionics. All aspects of guidance, flight control, and vehicle sequencing are provided by the Centaur INU (Fig. A.2.1.5-2). Attitude control is accomplished by a hydraulic servo positioning system for the booster engine. The RCU provides alternating current power (synchronized by a Centaur reference signal) to the RGUs. Two RGUs, one forward and one aft, provide pitch and yaw rate signals to the Centaur INU for vehicle stabilization. The RGUs provide true body rates to the INU. The inertial measurement subsystem provides vehicle roll rates and attitude for vehicle control. The FCS has the capability to provide for the Atlas/Centaur Separation event. This is accomplished with two pyrotechnic control units.

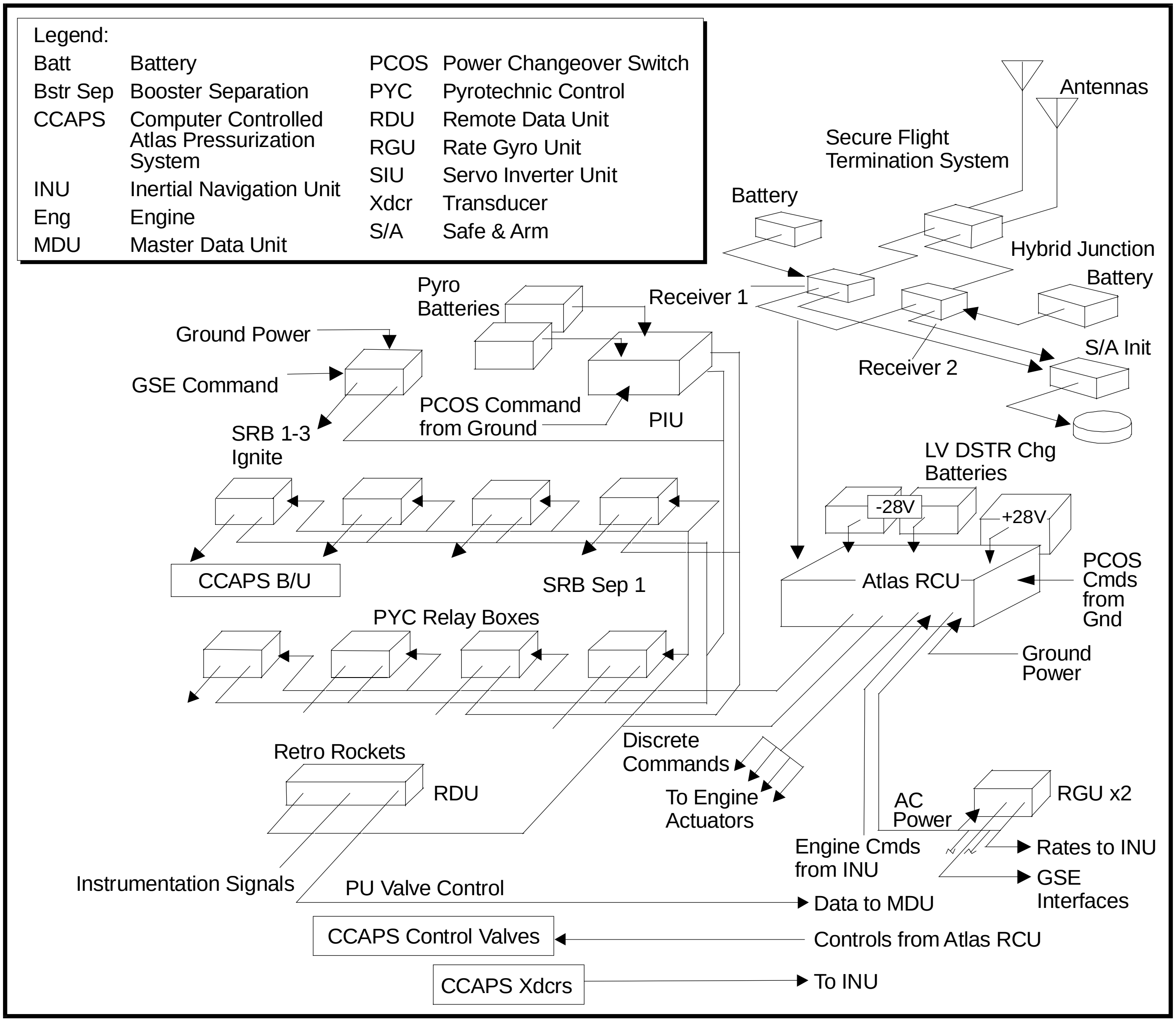


Figure A.2.1.5-2 The Atlas RCU combines several functions on the Atlas IIIA and IIIB avionics suite.

Telemetry Subsystem—The Atlas portion of the data acquisition system (DAS) consists of a remote data unit (RDU), vehicle transducers, and instrumentation harnessing. The RDU provides transducer excitation and signal conditioning for all Atlas measurements. They are digitally encoded and transmitted to the master data unit (MDU) located on the Centaur vehicle.

Secure Flight Termination System (FTS)—The Atlas stage FTS provides the capability for range safety termination of Atlas flight if there is a flight malfunction. The Atlas FTS is completely independent from the controls of other vehicle systems and from the Centaur upper stage FTS. The secure receiver responds to high-alphabet code word messages for engine shutdown, destruct, and system disable. Engine shutdown is handled by the Centaur SFTS for Atlas IIIA/IIIB (Fig. A.2.1.5-3).

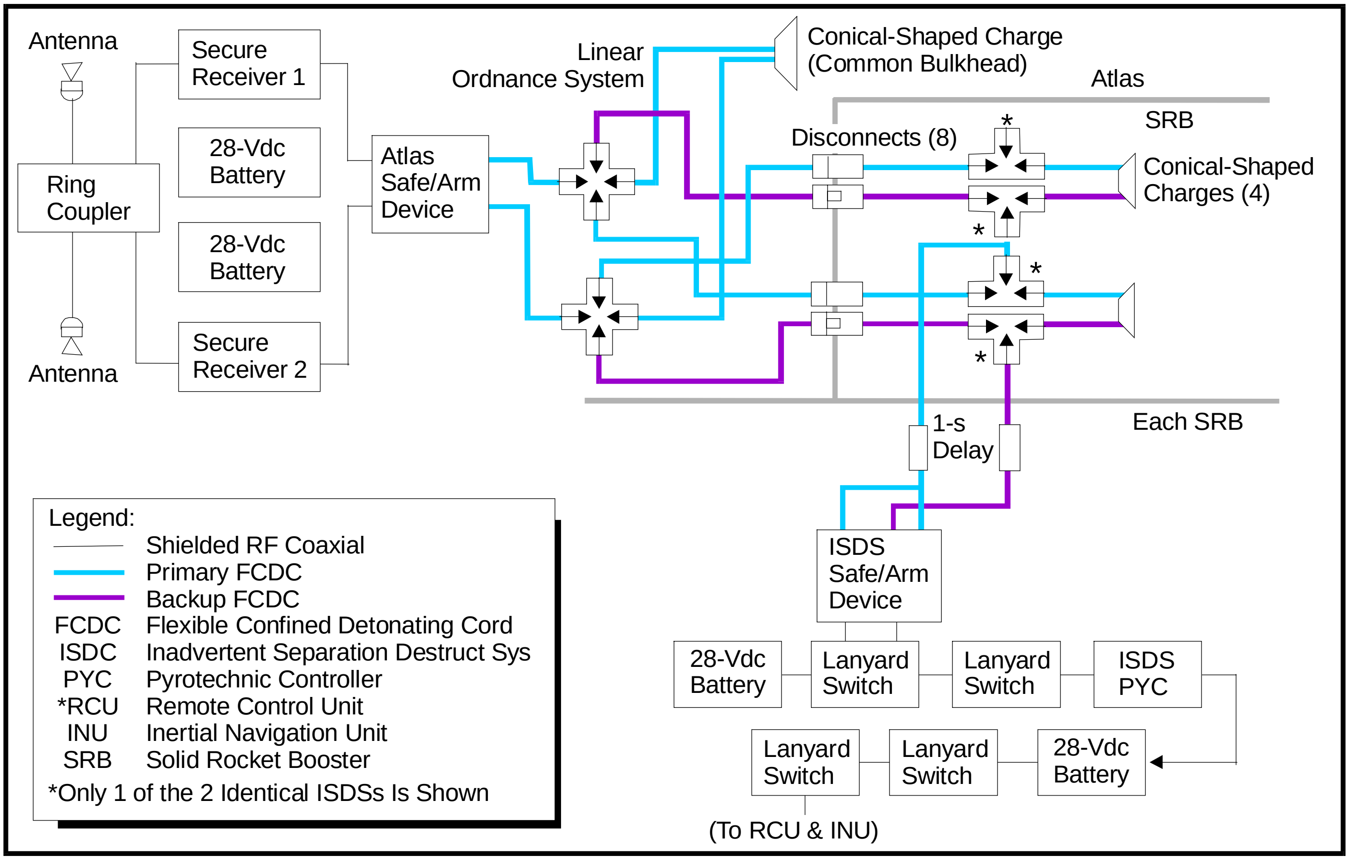


Figure A.2.1.5-3 The Atlas IIIA and IIIB FTS uses a simplified version of the Atlas IIAS.

For Atlas IIAS the FTS also includes a commanded SRB destruct capability. Figure A.2.1.5-4 illustrates the SFTS for Atlas IIAS. Each SRB is equipped with an inadvertent separation destruct system (ISDS) to terminate thrust capability in the unlikely event of an unplanned SRB jettison. With unplanned jettison, lanyard pull switches initiate a destruct signal that fires a conical-shaped charge located on the forward bulkhead of the SRB. Thrust capability is negated within 1 second after premature separation from Atlas. The system is single-fault tolerant to destruct and single-fault tolerant against inadvertent destruct. With normal Atlas IIAS flight, the system is safed before SRB jettison. Single-fault tolerance to safe and inadvertent safe is also included. Atlas IIA and Atlas III use this system, albeit without the SRB ISDS or SRB conical-shaped charge components.

Electrical Power Subsystem—The electrical power system consists of a 28-Vdc main vehicle battery, two smaller batteries used for the -28-Vdc Atlas RCU, and associated electrical harnesses. The Atlas RCU contains electrical distribution, a power changeover switch (PCOS), and a single-point grounding system. The PCOS transfers from ground power before liftoff. The main vehicle battery supplies power to three separate buses. The other batteries supply -28-Vdc power redundantly to the one -28-Vdc bus.

The pyro power system is separate from the main vehicle power system. Two pyro batteries supply power to the pyro controllers through a separate PCOS. The PCOS is housed in the pyro inhibit unit (PIU). The retro rocket and SRB separation pyros primary and backup controllers are powered from separate pyro batteries, making them completely redundant. The other pyros are fed from one pyro battery.

Propellant Utilization (PU)—The Atlas IIA/IIAS PU system measures and controls the fuel and oxidizer mixture ratio to the sustainer engine. The system is software-based and consists of differential pressure transducers, associated sense lines, INU control logic, Atlas SIU-based servo loop circuitry,

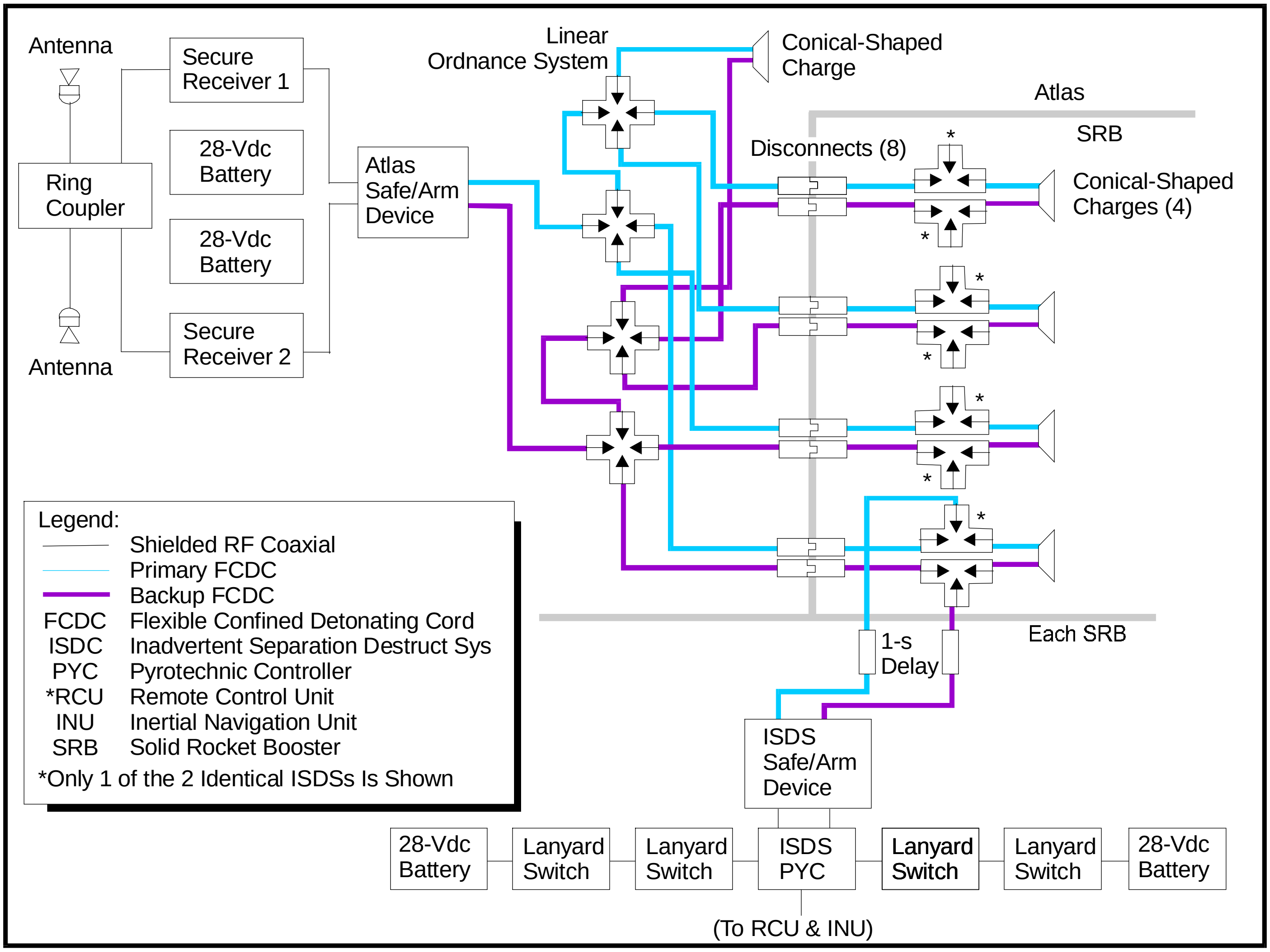


Figure A.2.1.5-4 The Atlas IIA/IIAS FTS is autonomous, secure, and approved for use at both ranges.

and a sustainer engine flow control, or PU valve. With the PU valve, sustainer engine propellant mixture ratio is controlled to uniformly deplete all usable tank propellants at sustainer engine cutoff (SECO). The PU system is used on all Atlas IIA/IIAS booster stages. The Atlas IIIA/IIIB uses similar PU sensing system, but with mixture ratio adjustments sent to the RD-180 engine from the INU via the Atlas RCU.

Atlas Propellant-Level Indicating System (PLIS)—The Atlas PLIS is used during tanking operations to indicate the level of oxidizer in the propellant tanks. The system consists of hot-wire sensors near the top of the oxygen tank and associated hardware for ground control and display. The sensors are used for two primary reasons: (1) to indicate when a minimum tanked propellant level has been reached to ensure sufficient loading to meet mission requirements, and (2) to indicate when a maximum level has been reached to ensure that sufficient ullage volume is present in the tanks for safe tank venting. Atlas fuel tank propellant levels are indicated using a site gauge temporarily installed on the vehicle during fuel tanking operations.

Atlas Propellant Depletion System—The Atlas IIA/IIAS (MA-5A) propellant depletion system is used to sense the onset of propellant outage and to initiate SECO. The system consists of two types of sensors: (1) redundant pressure switches in the sustainer engine that close when LO_2 manifold pressure drops, indicating LO_2 depletion, and (2) redundant optical sensors in the sustainer thrust cone that sense the absence of fuel as the level of fuel passes below the optical sensors. The output of these sensors is sent to the Centaur INU. When either fuel or oxidizer depletion is indicated, the INU immediately commands SECO and begins the post-SECO sequence. The primary purpose of the pressure switches is to indicate to the flight computer as quickly as possible that the engine is running out of oxidizer so the INU can command subsequent events, thereby maximizing vehicle performance. The primary purpose of the optical fuel depletion sensors is to prevent the engine from operating on only oxidizer.

Atlas IIIA/IIIB uses redundant RP and LO_2 depletion sensors to trigger a commanded BECO shutdown with depletion of either fuel or LO_2. Fuel depletion sensors are the same redundant optical sensors used on Atlas IIA/IIAS. Pressure sensors are used to detect LO_2 depletion on Atlas IIIA/IIIB.

A.2.2 Centaur Major Characteristics

The Centaur vehicle is 3.06 m (10 ft) in diameter and 10.0-m (33-ft) long. It uses liquid hydrogen and LO_2 propellants. The propulsion system uses two regeneratively cooled and turbopump-fed RL10A-4 or RL10A-4-1 engines, manufactured by Pratt & Whitney. Centaur avionics packages mounted on the forward equipment module control and monitor all vehicle functions. The Centaur IIA and IIAS stages are identical in design. Variation exists between configurations only in the propulsion system options and the avionics harnessing that connects the Centaur flight computer to the Atlas stage.

The Atlas IIIA SEC tank design is the same as the other Centaur vehicles. The overall length of the SEC is 10.5 m (34.3 ft). The additional length is due to the relocation of the single RL10A-4-1 engine onto an engine support beam spanning the centerline of the vehicle. The Atlas IIIB SEC has an extended tank that is 11.74-m (38.52 ft long). The SEC also incorporates electromechanical actuators to perform thrust vector control, in place of the hydraulic actuation system. Avionics harnessing has been updated to accommodate these vehicle changes.

A.2.2.1 Structure—The Centaur structural system consists of four major structural elements: the propellant tank, stub adapter, equipment module, and tank insulation. The entire structure is protected for corrosion control.

Propellant Tank—The propellant tank structure provides primary structural integrity for the Centaur vehicle and support for all upper-stage airborne systems and components (Fig. A.2.2.1-1). A double-wall, vacuum-insulated intermediate bulkhead separates the propellants. The tanks are constructed of thin-wall fully monocoque corrosion-resistant steel. Tank stabilization is maintained by internal pressurization.

In a dual-engine (IIA, IIAS, IIIB DEC) Centaur configuration, the engines are mounted directly to the propellant tank aft bulkhead. The single-engine IIIA and IIIB (SEC) Centaur design incorporates an engine support beam that attaches to the aft bulkhead in the existing engine mount locations, but provides for centerline mounting of the single engine 17.5-in. aft of the dual-engine location. Redesigned thrust vector control actuator supports have also been incorporated.

Stub Adapter—The stub adapter, installed on the forward tank ring, supports the equipment module, spacecraft adapter, and payload fairing (PLF). It is an aluminum skin/stringer cylindrical structure 3.05 m (10 ft) in diameter and 63.5-cm (25-in.) long (Fig. A.2.2.1-2).

Equipment Module—The equipment module provides mounting for avionics packages, electrical harnesses, and the forward umbilical panel (Fig. A.2.2.1-3). The spacecraft adapter bolts to the forward ring of the module. The load carrying capability of the equipment module has been enhanced so it can react the loads of a 5,900- to 6,350-kg (13,000- to 14,000-lb) payload. Localized structure improvements and use of aluminum-lithium stringers provide a cost and mass-efficient structural

Figure A.2.2.1-1 The Centaur tank is structurally efficient.

enhancement without requiring requalification of the avionics hardware mounted on the equipment module.

Tank Insulation—Centaur tank insulation consists of a fixed foam, whose function is to minimize ice formation and LO_2 and LH_2 boiloff on the ground and during atmospheric ascent. It consists of closed-cell polyvinyl chloride (PVC) foam panels adhesively bonded to the exterior surface of the LH_2 and LO_2 tanks (Fig. A.2.2.1-4).

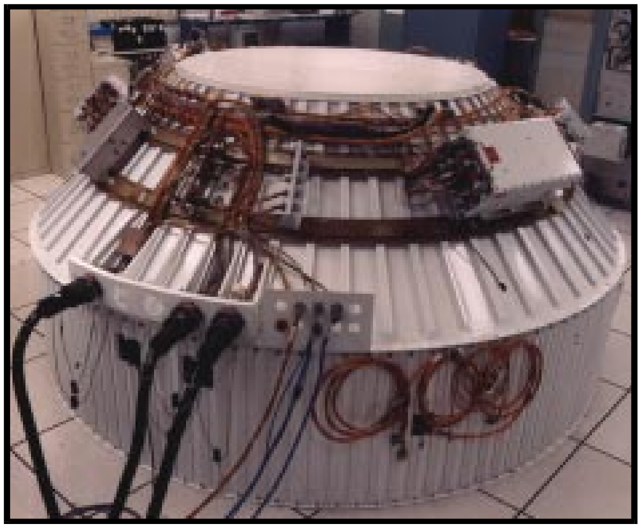


Figure A.2.2.1-2 The stub adapter and equipment module combination provides a robust, stable interface for launch vehicle avionics and the attached payload adapter and spacecraft.

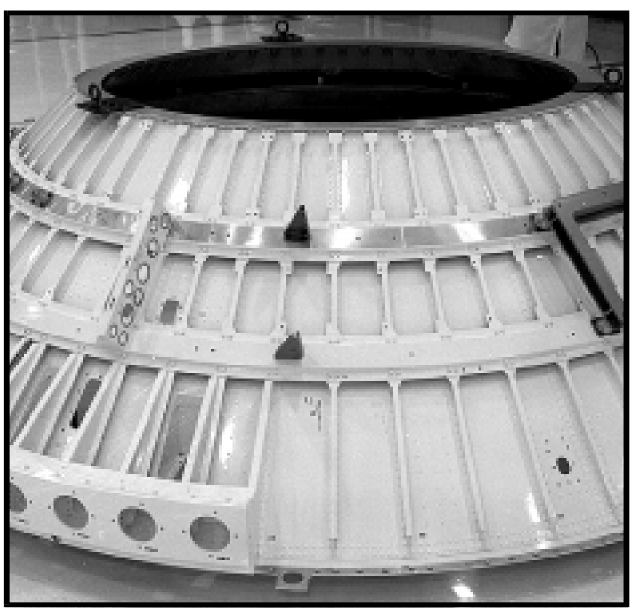


Figure A.2.2.1-3 The Centaur equipment module allows line replacement of all avionics in the field.

Figure A.2.2.1-4 Fixed foam bonded directly to the Centaur tank is a simple and efficient insulator.

A.2.2.2 Pneumatics—The pneumatic system controls tank pressure, provides reaction control and engine control bottle pressure, and provides purges. The pneumatic system consists of the computer-controlled vent and pressurization system (CCVAPS), helium supply system, and purge supply system.

CCVAPS—The CCVAPS maintains the ullage tank pressures to ensure structural integrity. It prevents tank underpressure or overpressure, maintains the structural integrity of the intermediate bulkhead, and ensures positive suction pressure to the Centaur engines. The CCVAPS is a software-based fault-tolerant, closed-loop tank pressure control system. Three redundant ullage transducers in each tank measure pressure for the INU. The INU commands either pressurization or venting to achieve the desired tank pressure profile. Pressurization control is affected by locking the LO_2 and LH_2 vent valves and cycling the fault-tolerant pressurization solenoid valves. Vent control is active during boost phase and Centaur coast phases by cycling the LH_2 solenoid vent valve or unlocking the LH_2 or LO_2 self-regulating vent valves.

Helium Supply System—The baseline configuration for short-duration coast missions (<25 minutes) requires two 66.0-cm (26-in.) composite helium storage spheres charged to 27,580 kPa (4,000 psi). The spheres are graphite-overwrapped and reinforced with a 301 CRES metallic liner that provides leak-before-burst capability. A pressure regulator provides 3,450-kPa (500-psig) helium for engine and reaction control system (RCS) controls. The two helium storage bottles provide ample helium for launch of direct ascent and first descending node geosynchronous transfer trajectory missions. This complement of bottles is used on all Atlas geosynchronous transfer orbit (GTO) class missions. For longer coast duration missions, provisions exist for addition of two more helium bottles as required.

Purge Supply System—This dedicated airborne system provides helium purge gas to critical components to prevent air/moisture injection and freezing through the Atlas boost phase.

A.2.2.3 Propulsion—The Centaur propulsion system uses LH_2 and LO_2 propellants. Primary propulsion is provided by two Pratt and Whitney RL10A-4 or RL10A-4-1 engines. The single-engine Centaur uses the single RL10A-4-1 engine with an extendible nozzle.

Main Engines—The RL10 engines are gimballed, turbopump-fed, and regeneratively cooled and consist of a fixed primary nozzle and an optional secondary extendible nozzle (Fig. A.2.2.3-1). The 51-cm (20-in.) long columbium extendible nozzle provides enhanced engine performance through an increase in nozzle expansion ratio. Engine prestart and start functions are supported by supplying pressure-fed propellants to the engine pumps (Fig. A.2.2.3-2 for IIA/IIAS/IIIB dual engine Centaur (DEC), Fig. A.2.2.3-3 for IIIA/IIIB

Figure A.2.2.3-1 The RL10A-4 engine family can use an extendible nozzle for enhanced performance.

single engine Centaur (SEC). The engines provide pitch, yaw, and roll control during powered phases of flight. The RL10A-4 engine develops a thrust of 91.19 kN (20,500 lbf) at a specific impulse of 442.5 seconds. The RL10A-4-1 engine variant develops a thrust of 97.86 kN (22,000 lbf) at a specific impulse of 444.0 seconds. Extendible nozzles increase engine thrust by 1.4 kN (300 lbf) per engine and specific impulse by 6.5 seconds. Both the RL10A-4 and RL10A-4-1 engines are fully qualified and flight-proven.

Reaction Control System (RCS)—The hydrazine-based RCS provides pitch, yaw, and roll control for Centaur during coast phases of flight and provides propellant settling during the coast up through main engine start (MES). The single-engine Centaur also uses the RCS during powered phases of flight to perform roll control. The RCS is located on the Centaur aft bulkhead providing fault-tolerant control and avoiding contamination of the spacecraft when used.

Hydraulic TVC System—The dual-engine Centaur hydraulic system consists of one hydraulic power unit with two hydraulic servo actuators for each engine. The servo actuators provide engine gimballing for steering and vehicle control during the burn phases.

The single-engine Centaur does not have a hydraulic TVC system. The hydraulic power unit and servo actuators have been removed and replaced by an electronic control unit (ECU) and two electromechanical actuators (EMA) for thrust vector control of its engine.

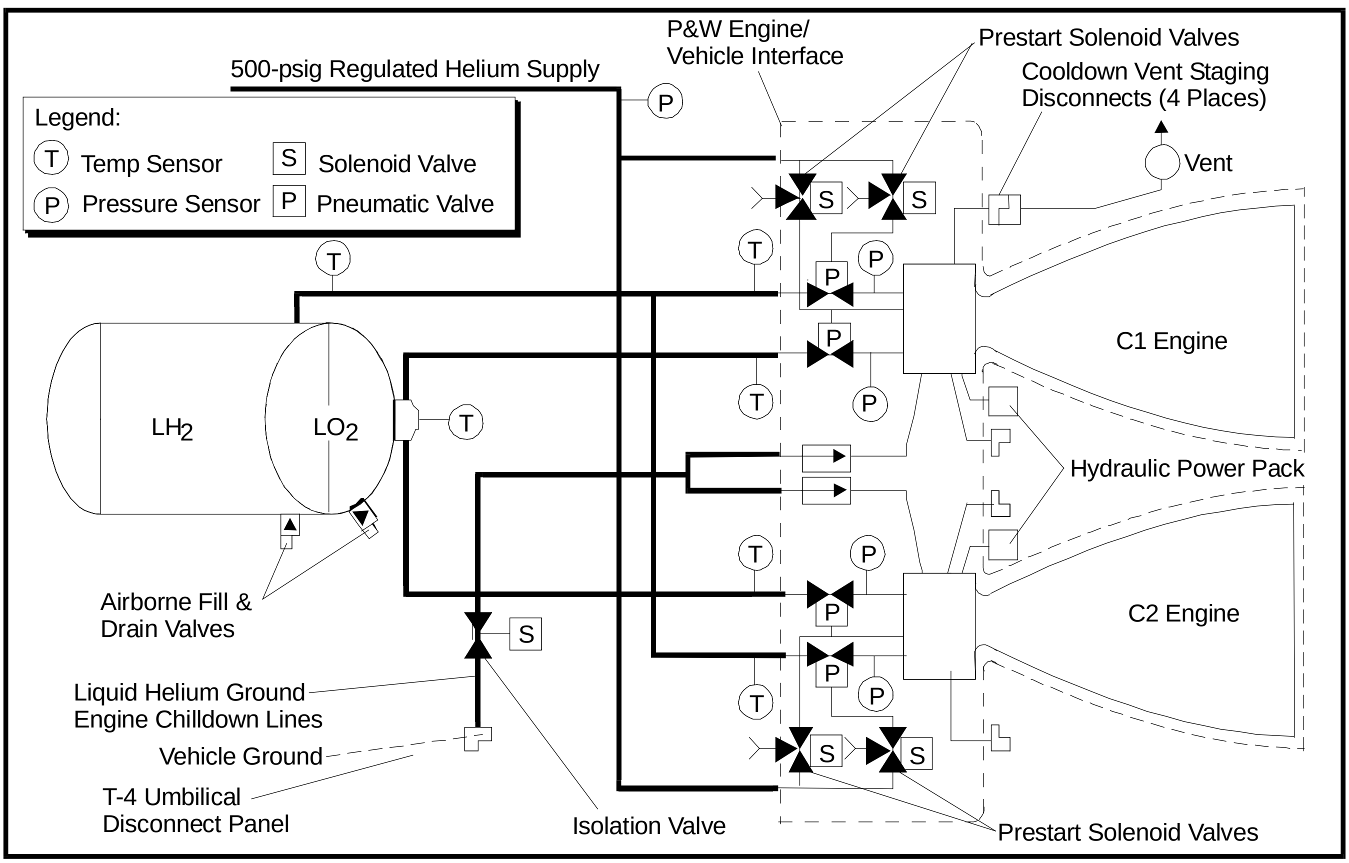


Figure A.2.2.3-2 The Atlas IIA/IIAS/IIIB (DEC) Centaur propulsion system consists of two liquid hydrogen/oxygen main engines.

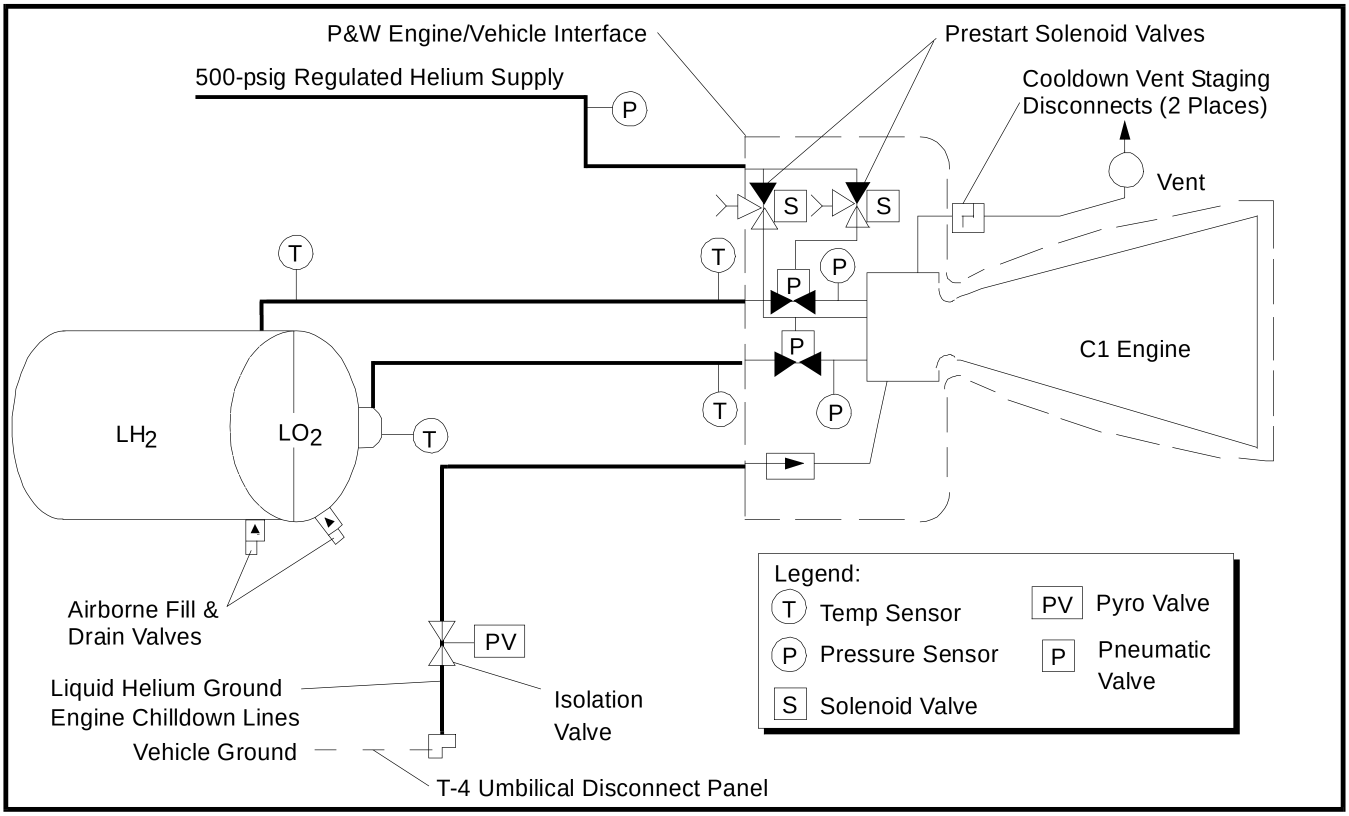


Figure A.2.2.3-3 The Atlas IIIA/IIIB (SEC) Centaur propulsion system consists of a single liquid hydrogen/oxygen main engine.

A.2.2.4 Avionics—Centaur avionics are modern, flight-proven subsystems consisting of guidance, navigation and flight control, telemetry and tracking, secure flight termination capability, and electrical power distribution (Fig. A.2.2.4-1 for IIA/IIAS, Fig. A.2.2.4-2 for IIIA/IIIB). These systems control and monitor all vehicle functions during launch and flight and are line-replaceable in the field.

Flight Control Subsystem (FCS)—The Centaur uses the most accurate, advanced guidance system available in the expendable launch vehicle industry. It is based on a standard 1750A processor and ring laser gyros that are mounted on the Centaur equipment module. The FCS performs all vehicle-required computations, including guidance, navigation, attitude control, sequencing and separation fire commands, propellant utilization control, and tank pressurization control. Input to the FCS includes incremental velocities and time, quaternion information, Atlas rate data, Atlas and Centaur tank pressures and Atlas and Centaur PU data. Output from the FCS includes command of 128 solid-state switches in the Centaur remote control unit (C-RCU) and 64 solid-state switches in the Atlas RCU (A-RCU). The RCUs control all vehicle sequencing events: RCS, engines, pressure control, TVC and propellant depletion.

The FCS performs all attitude control, guidance, and navigation computations for both the Atlas and Centaur phases of flight. FCS-commanded open-loop pitch and yaw steering occurs during the early Atlas phase based on winds-aloft measurements taken shortly before launch. Closed-loop guidance steering is then initiated to provide guidance steering based on the mission trajectory requirements.

The ring laser gyro system provides high-accuracy rate measurements for processing by the flight computer. The strapped-down inertial measurement subsystem (IMS) provides the FCS incremental timing pulses from a precision timing reference, incremental velocity pulses from the accelerometers, and body-to-inertial attitude quaternion data.

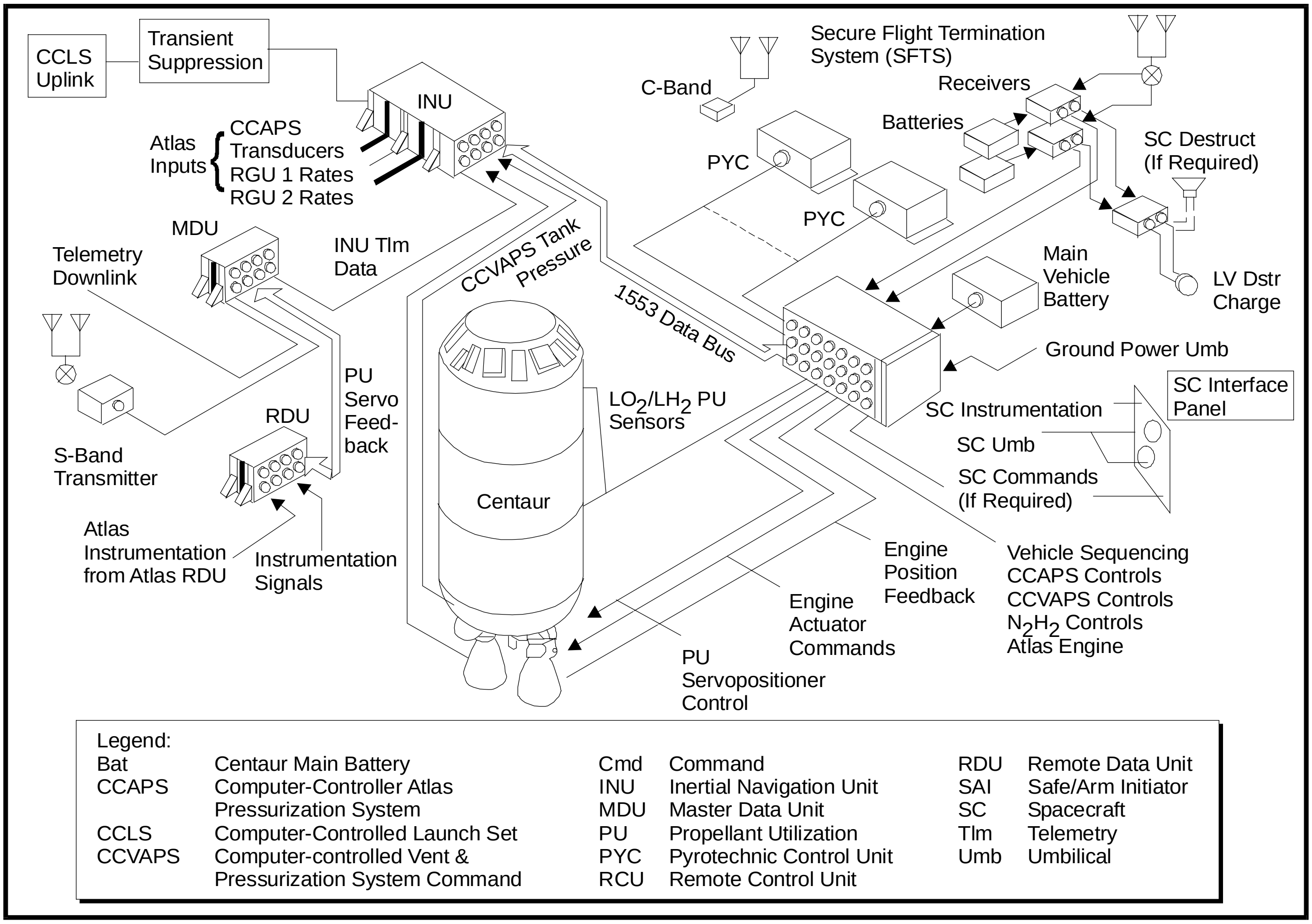
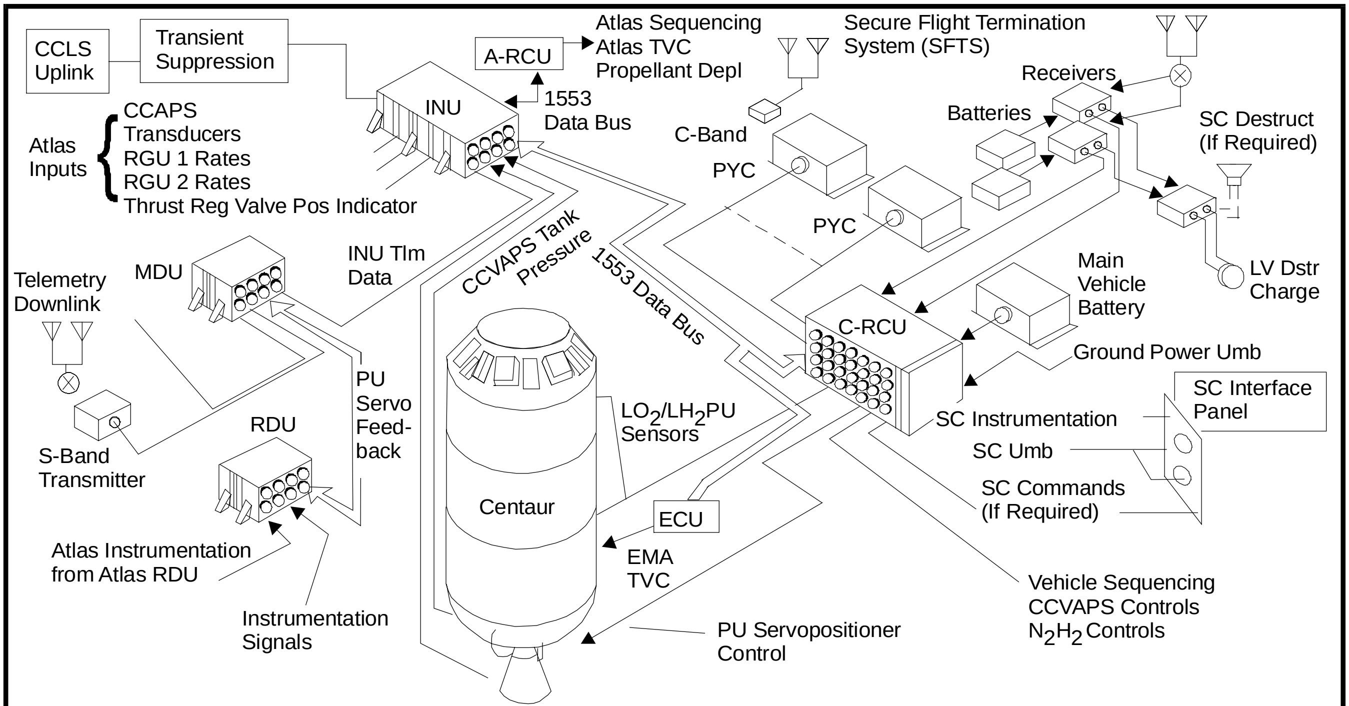


Figure A.2.2.4-1 The Centaur avionics system uses flight-proven hardware to delivery unmatched flight capability.

Telemetry and Tracking—The Atlas telemetry DAS (Fig. A.2.2.4-3) monitors several hundred vehicle parameters in addition to guidance and navigation (G&N) data before launch and throughout all phases of flight. Critical vehicle parameters are also monitored via hardwired landline instrumentation. Measurements include acceleration, vibration, temperatures, pressures, displacement, currents and voltages, engine pump speeds, and discretes. The programmable data acquisition telemetry system consists of an MDU and two RDUs, one on Atlas and one on the aft end of Centaur. The MDU, located on the Centaur equipment module, provides transducer excitation, signal conditioning, and encoding for all Centaur front-end measurements in addition to receiving and formatting data from the INU and two RDUs. The INU provides data to the MDU over the 1553 data bus. The MDU provides two pulse-code modulation (PCM) outputs; one is connected to the radio frequency (RF) transmitter, the other is used to provide a hardline link. The C-band tracking system aids in determining the real-time position of the Centaur vehicle for launch site range safety tracking requirements. The airborne transponder returns an amplified RF signal when it detects a tracking radar's interrogation. The baseline system consists of a non coherent transponder, a power divider, and two antennas. The Centaur C-band tracking system meets the requirements of Eastern and Western Ranges.

FTS—The Centaur flight termination system (FTS) provides the capability for termination of Centaur flight in case of an in-flight malfunction. The Centaur FTS is independent of controls of other vehicle systems and from the Atlas FTS. This system has redundancy and is approved for launches at Cape Canaveral. Approval for baseline use on the west coast is still under review. The secure receiver



Legend:

A-RCU	Atlas RCU	Cmd	Command	PYC	Pyrotechnic Control Unit
Bat	Centaur Main Battery	C-RCU	Centaur RCU	RCU	Remote Control Unit
CCAPS	Computer-Controller Atlas Pressurization System	ECU	Electronic Control Unit	RDU	Remote Data Unit
		EMA	Electro-Mechanical Actuator	SAI	Safe/arm Initiator
CCLS	Computer-Controlled Launch Set	INU	Inertial Navigation Unit	SC	Spacecraft
CCVAPS	Computer-Controlled Vent & Pressurization System	MDU	Master Data Unit	Tlm	Telemetry
		PU	Propellant Utilization	TVC	Thrust Vector Control
				Umb	Umbilical

Figure A.2.2.4-2 The single-engine Centaur avionics system retains flight-proven hardware.

responds only to high-alphabet code word messages. These messages command engine shutdown, vehicle destruct, and system disable. A payload destruct system can be added, if required, to disable the propulsive capability of the spacecraft. A conical-shaped destruct charge in the payload adapter can be directed at the payload propulsion system. Figure A.2.2.4-4 illustrates the Centaur FTS.

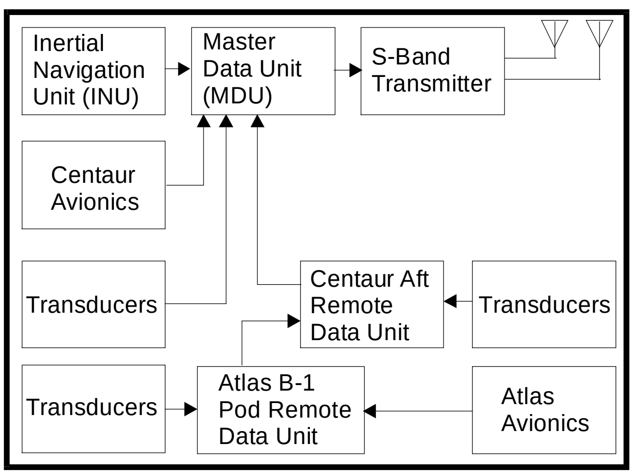


Figure A.2.2.4-3 The telemetry data acquisition system monitors the state and performance of all Atlas systems.

Electrical Power System—The electrical power system consists of a 28-Vdc main vehicle battery, two pyrotechnic batteries, the RCU, a single-point grounding system, and associated electrical harnesses. The RCU provides 128 channels of solid-state switching under INU software control via a MIL-STD-1553B data bus. Arm-safing of critical sequence functions is included. The RCU also provides power changeover capabilities from ground power to internal main vehicle battery power to meet power distribution requirements. The electrical harnesses provide interconnect wiring for avionics equipment with other Centaur systems, interconnections between Atlas and Centaur, power distribution, command, and telemetry signal paths.

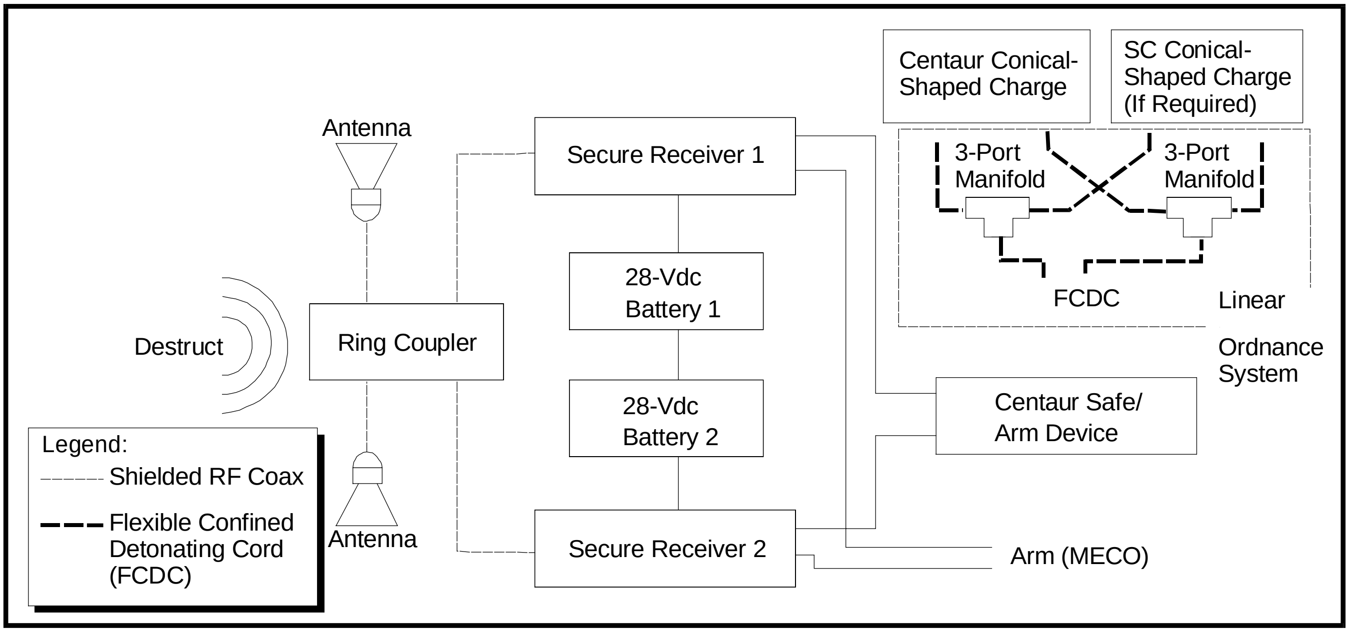


Figure A.2.2.4-4 The Centaur FTS responds to launch site-generated high-alphabet code words.

Propellant Utilization (PU)—The Centaur PU system ensures that an optimum mixture of LH_2 and LO_2 residuals remain in the Centaur propellant tanks at the end of the mission. The PU system continuously measures the mass of the remaining propellants using one of three selected externally mounted differential pressure transducers in each of the LH_2 and LO_2 tanks that measures propellant head pressure. INU software converts this head pressure to mass and calculates the sensed mass imbalance between the two tanks. The INU then commands a drive motor on the RL10 engine oxygen flow control valve to correct the sensed error by changing the engine mixture ratio. Drive motor electronic controls are contained in the RCU.

Propellant Level Indication System (PLIS)—The Centaur PLIS is used during tanking operations to indicate levels of fuel and oxidizer in the propellant tanks. The system consists of hot wire sensors near the top of the propellant tanks and associated hardware for ground control and display. The sensors prevent underloading and overloading of propellants.

A.2.2.5 Centaur Extended Mission Kit (EMK)—As Centaur's primary mission role in recent years has been the launch of geosynchronous communications satellites, power, thermal control, and fluids management hardware on Centaur has been mass-optimized to enable maximum performance for these types of missions. To meet the requirements of several unique payloads, an EMK has been designed and manufactured to provide Centaur the extensions in power and fluid management capabilities required to extend mission duration by up to two hours. The EMK adds additional helium bottles, and radiation shielding on the LO_2 tank side wall (Fig. A.2.2.5-1). Centaur's avionics and electrical components are covered with special thermal paints, tapes, and additional radiation shielding to maintain their operating temperatures (Fig. A.2.2.5-2). Several minor modifications to the Centaur aft bulkhead were incorporated across the fleet to allow "kit-able" installation of the EMK. First flight of the EMK occurred on December 2, 1995, with launch of the NASA/Solar and Heliospheric Observatory (SOHO) on Atlas IIAS.

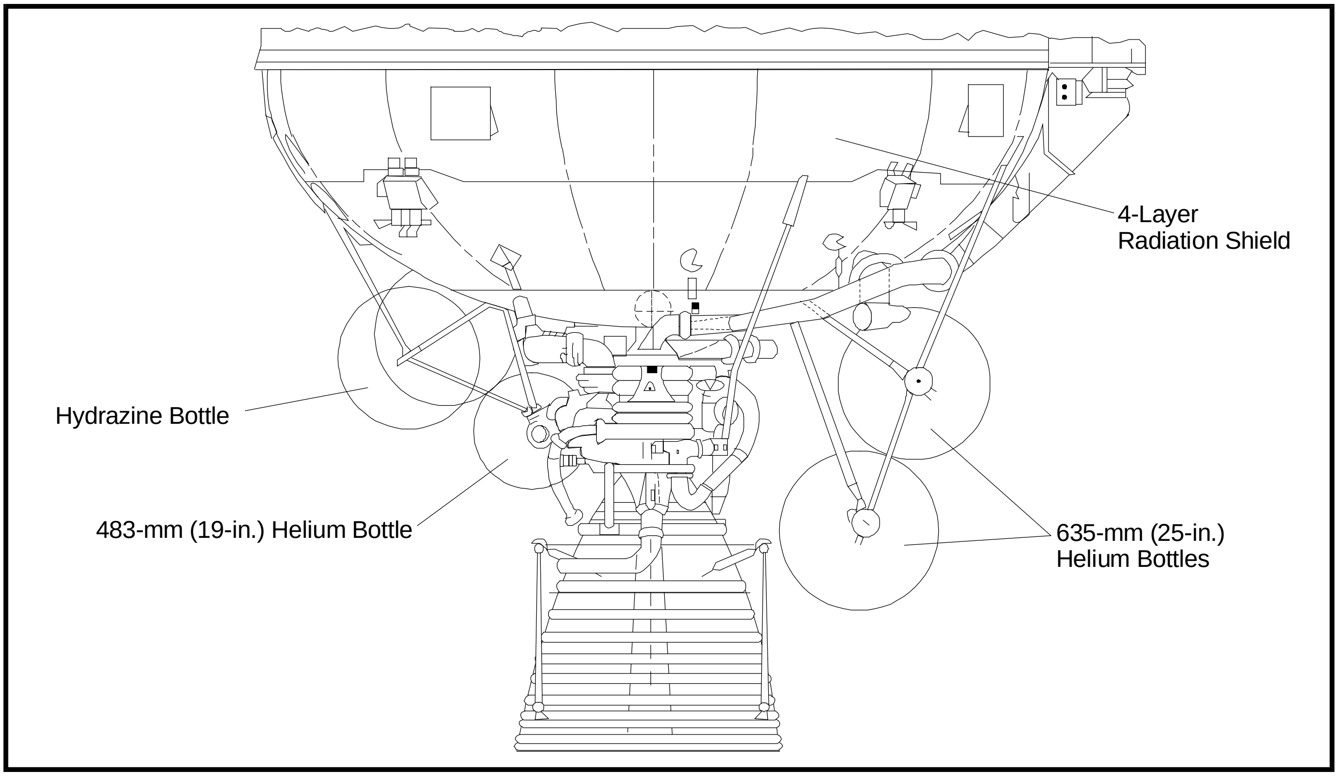


Figure A.2.2.5-1 Centaur Aft Bulkhead for Extended Coasts

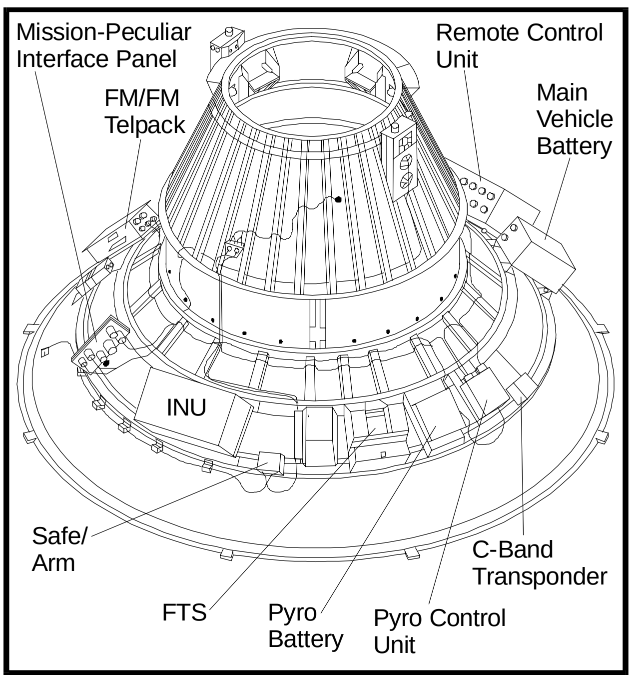


Figure A.2.2.5-2 Typical Centaur Equipment for Extended Coasts

A.2.3 Payload Fairing (PLF)

The PLF protects all components forward of the Centaur from liftoff through atmospheric ascent. Three PLF options are available (Fig. A.2.3-1).

The PLF is a two-half-shell structure. The structure is an aluminum skin/stringer/frame construction with vertical, split-line longerons. The fairing consists of a cylindrical section topped by a conical nose cone and a spherical cap. The 14.5° angle of the cone minimizes combined aerodynamic drag and weight losses.

The PLF provides thermal and acoustic enclosures for the spacecraft systems and launch vehicle electronic compartments during prelaunch and ascent. Portions of the external surfaces of the fairing are insulated with cork to limit temperatures to acceptable ranges. Noncontaminating thermal control coatings are used on internal surfaces to reduce incident heat fluxes to the spacecraft.

The fairing also provides mounting provisions for various other systems. Payload compartment cooling system provisions are in the cylindrical portion of the fairing. Electrical packages required for the fairing separation system are mounted on the internal surface of the fairing. In addition, the PLF interior may include acoustic blankets or thermal shields in the cylindrical and/or conical sections to ensure that the spacecraft environment is within acceptable limits. Optional blankets are used to reduce the acoustics environment in the fairing. Thermal shields reduce the thermal environment in the fairing.

Access to the avionics on the Centaur equipment module is obtained through either doors in the boattail assembly for the large PLF (LPF) and extended-length LPF (EPF) or the split barrel for the medium PLF (MPF). Access to the payload is through these doors or through mission-peculiar doors in the cylindrical section of the fairing. A reradiating antenna assembly may also be provided to allow spacecraft communications when the PLF is in place.

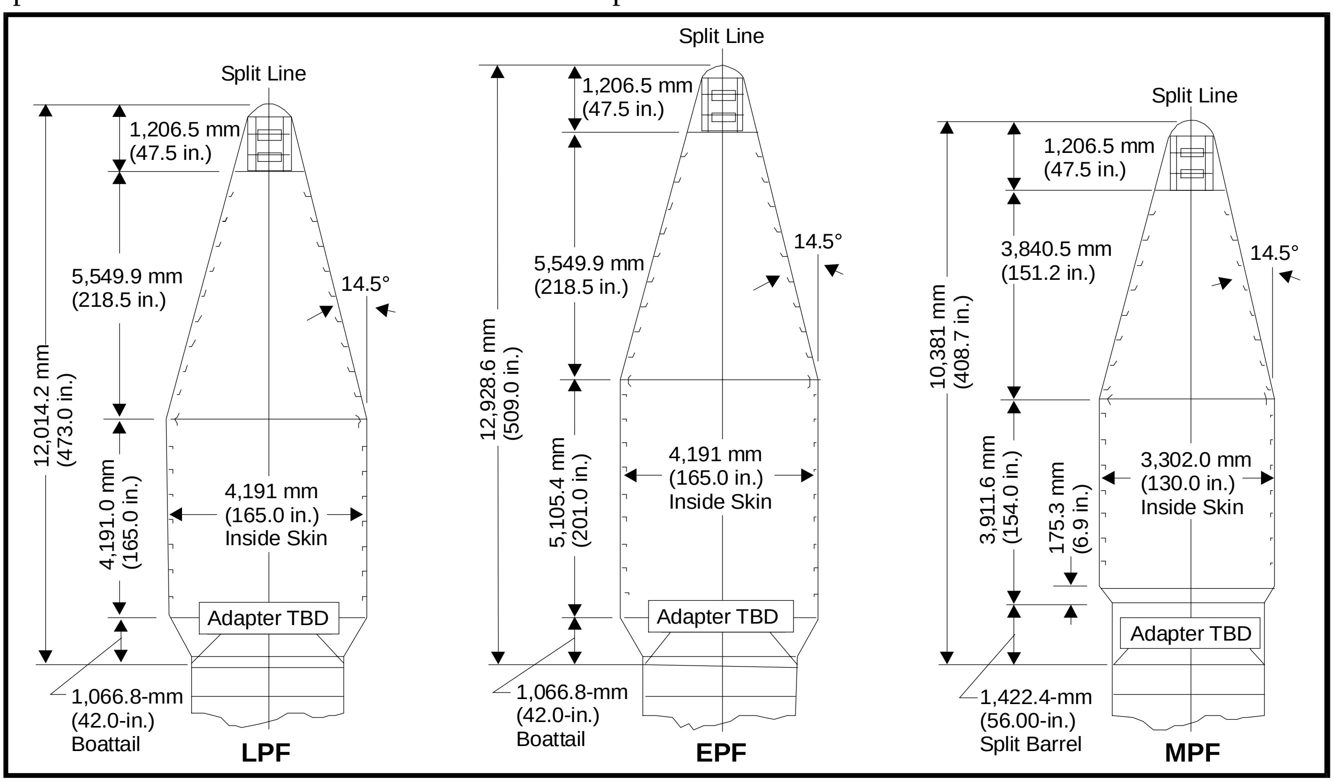


Figure A.2.3-1 Atlas PLF options provide mission-development flexibility.

A.2.4 Spacecraft and Launch Vehicle Interface Accommodations

Lockheed Martin offers a choice of eight payload adapters to meet launch vehicle and spacecraft interface requirements. These accommodations are fully discussed in earlier sections of this document.

A.3 ATLAS AND CENTAUR PRODUCTION AND INTEGRATION

Lockheed Martin provides a combination of experienced production leadership and a production organization dedicated to continuous process improvement supported by state-of-the-art computerized management systems. Our site management concept encourages personnel at each location to concentrate on attaining their respective goals to achieve overall program milestones.

A.3.1 Organization

Overall accountability for Atlas production quality, schedule, and cost lies with LMA Production Operations (Fig. A.3.1-1). Within this organization are all production sites plus core support functions of material planning, manufacturing engineering, acquisition management, environmental management, and business management.

Each site is fully empowered to produce quality products for the integrated master schedule. Each site focuses on operations of fabrication, assembly, processing, testing, and launch best suited to their capabilities (Fig. A.3.1-2).

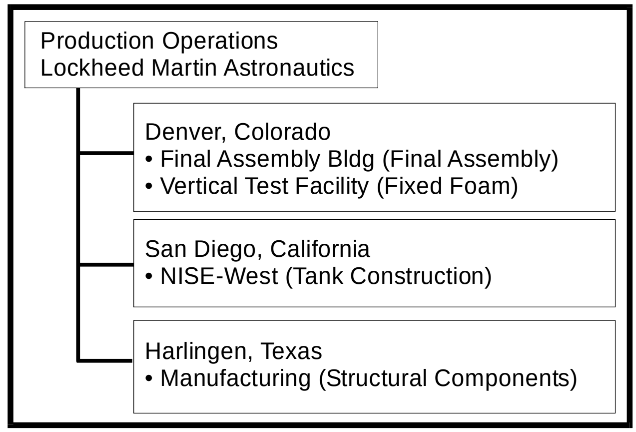


Figure A.3.1-1 Atlas Production Operations Organization

Final Assembly Building (FAB) Denver, Colorado—The FAB is a newly constructed final assembly and storage building encompassing 18,580 m^2 (200,000 ft^2), of which 3,715 m^2 (40,000 ft^2) operates as a Class 100,000 clean room for Centaur final assembly and test (Figs. A.3.1-3 and A.3.1-4). This facility ensures the cleanest possible environment for upper stage production. In addition to a modernized Atlas first stage final assembly area, the FAB houses subassembly manufacturing, completed vehicle and work-in-process storage, support areas such as tool and print cribs, production test laboratories, and office spaces. Completed early in the second quarter of 1995, the FAB (Fig. A.3.1-5) is the most comprehensive, state-of-the-art launch vehicle manufacturing, integration, and test facility in the world. A pressure testing facility located next to the FAB is used to perform final checkout of the completed launch vehicle system.

Vertical Test Facility (VTF) Denver, Colorado—A modernized fixed-foam application cell within the existing VTF (Fig. A.3.1-6) is designed to apply fixed foam to a flight vehicle.

Naval In-Service Engineering (NISE) West Facility San Diego, California—The NISE-West (formerly Plant 19) facility focuses on tank manufacturing for the Atlas and Centaur vehicles. Welding facilities at this site are used for producing not only the pressurized tank structures, but many of the fuel, LO_2, and pneumatic systems.

Harlingen Facility, Harlingen, Texas—This facility, located at the tip of southern Texas, builds major structural sections for the Atlas and Centaur vehicles. Harlingen ships its products to San Diego, California; Denver, Colorado; and the launch sites to undergo final assembly with the specific launch vehicles. Each site is fully staffed with all the disciplines needed to support the program goals. Having all functions reporting to the site director ensures a uniform emphasis on program goals and customer satisfaction.

Cape Canaveral Air Station (CCAS), Florida—CCAS is the final destination of Atlas and Centaur hardware for East Coast launches. CCAS concentrates on the final vehicle and system checks before launching the vehicles.

Vandenberg Air Force Base (VAFB), California—VAFB is the final destination for Atlas and Centaur hardware for West Coast launches. VAFB concentrates on final vehicle and system checks before launching vehicles.

	CCA	NISE-West (San Diego)	FAB/VTF Denver	Harlingen, Texas	VAFB
Fabrication	None	• Skins • Bulkhead Gores	• Insulation • Radiation Shielding	None	• Small Machined Parts
Assembly	None	• Tanks • Ducting	• Centaur RCS • Insulation • Radiation Shielding	• Atlas Thrust Structure • P/L Fairings • Atlas Pod Doors • Adapters	None
Final Assy & Installation Launch Operations	• Interstage Adapter • Atlas RCS • Payload Adapter • Payload Fairing CCA Launches	None	• Atlas & Centaur Tanks • Atlas & Centaur Thrust Structure • Engines • Avionics • Harnessing • Eqpt Module • Stub Adapter	None	• Interstage Adapter • Atlas RCS • Payload Adapter • Payload Fairing VAFB Launches

Major Subcontractors
Atlas Engines *Rocketdyne*
Centaur Engines *Pratt & Whitney*
Data Acquisition Sys *Gulton*
Inertial Navigation Unit *Honeywell*
Solid Rocket Motors *Thiokol*

Valves
Rings
Machined Parts
•
•
•

Figure A.3.1-2 Our production approach is mature and proven.

Figure A.3.1-3 Final Assembly Building

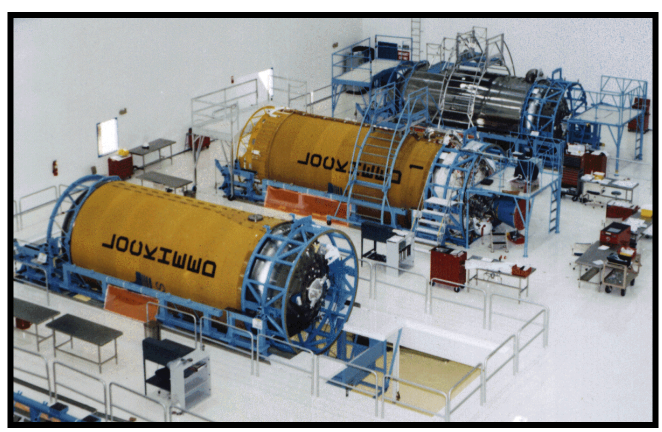


Figure A.3.1-4 Environmentally Controlled Assembly Area (Class 100,000)

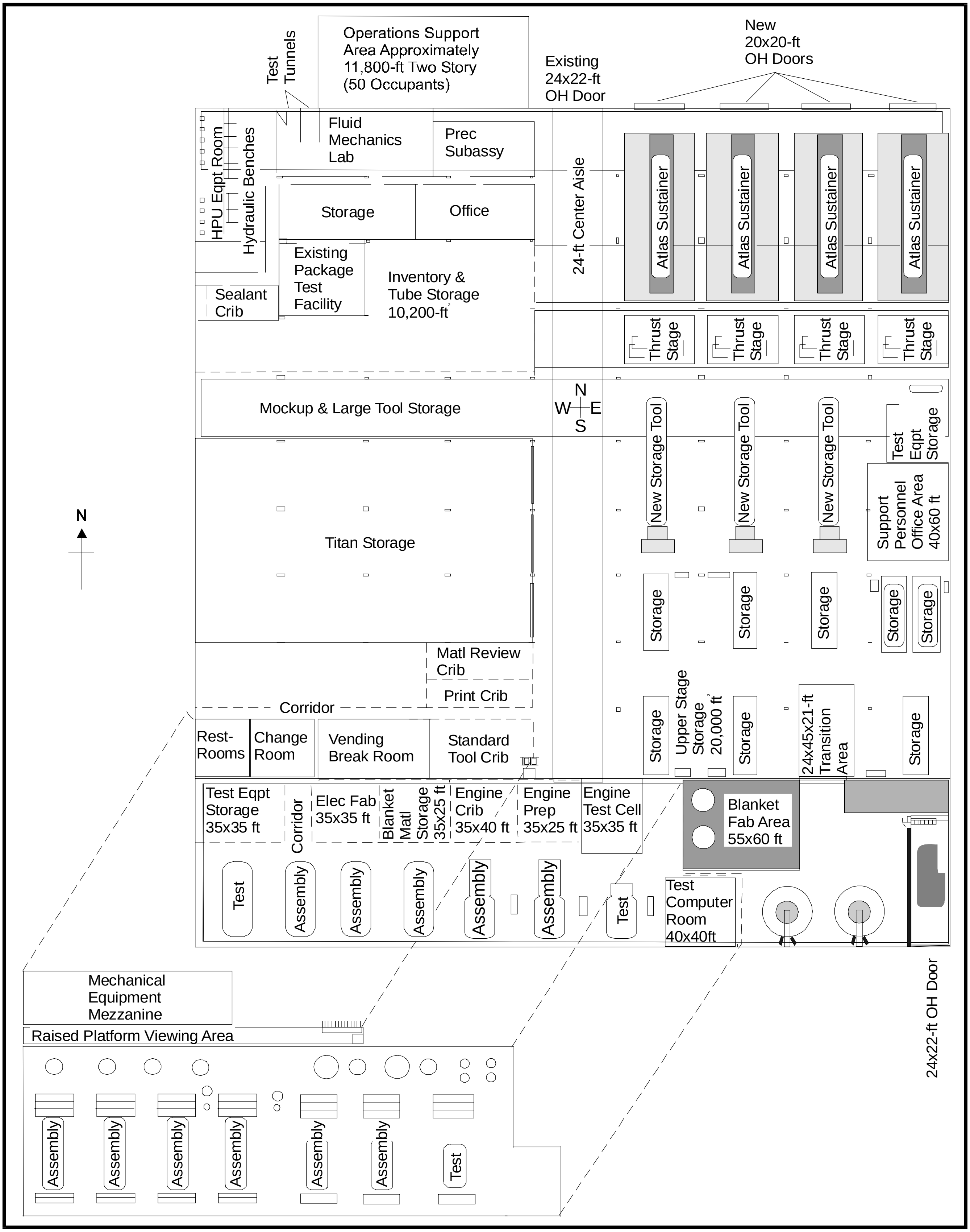


Figure A.3.1-5 Final Assembly Building: State-of-the-Art Launch Vehicle Production Facility

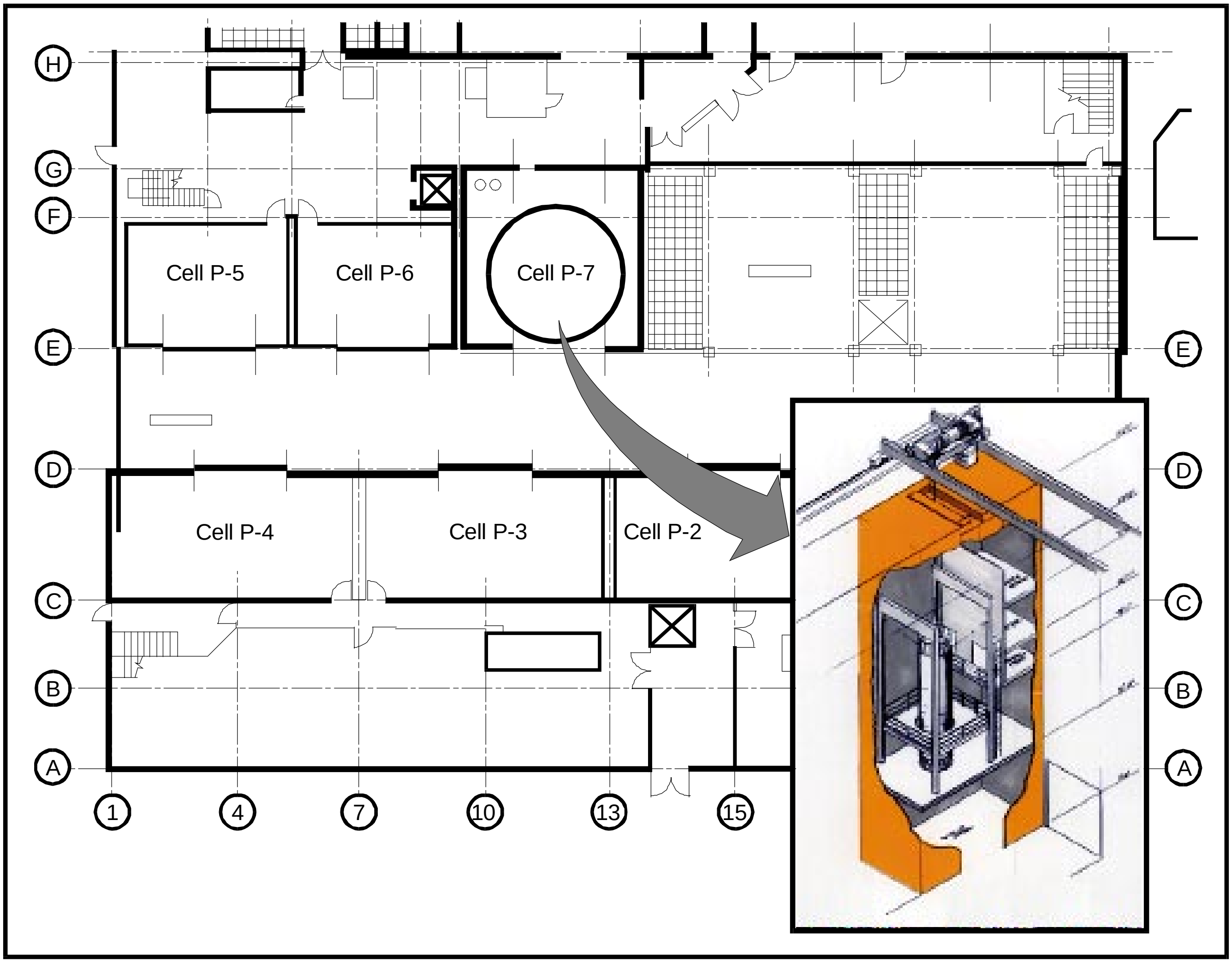


Figure A.3.1-6 A modern fixed-foam facility supports Centaur assembly in Denver.

A.3.2 Manufacturing Sequence and Identification

Our manufacturing control point product structure presents a directly identifiable assembly sequence and flow of the deliverable end item, which becomes an important management tool and a major element in the production plan.

Atlas Booster Flow—The control point product flow for the Atlas booster (including activity locations) is shown in Figure A.3.2-1. Components are delivered to the FAB from Harlingen, NISE-West, and various subcontractors. The booster is assembled, tested, and prepared for shipment to the launch site. Use of a pictorial display (Fig. A.3.2-2) provides piece-part clarity and maximizes understanding across all levels of the booster assembly process.

Centaur Upper Stage Flow—The control point product flow for the Centaur upper stage (including activity locations) is shown in Figure A.3.2-3. As with the booster, components are delivered from Harlingen, NISE-West, and subcontractors. The fixed foam is applied in the VTF. The Centaur then is assembled, tested, and prepared for shipment with the booster to the launch site. As with the booster, a pictorial display (Fig. A.3.2-4) provides piece-part and jettison component identification, facilitating assembly process clarity.

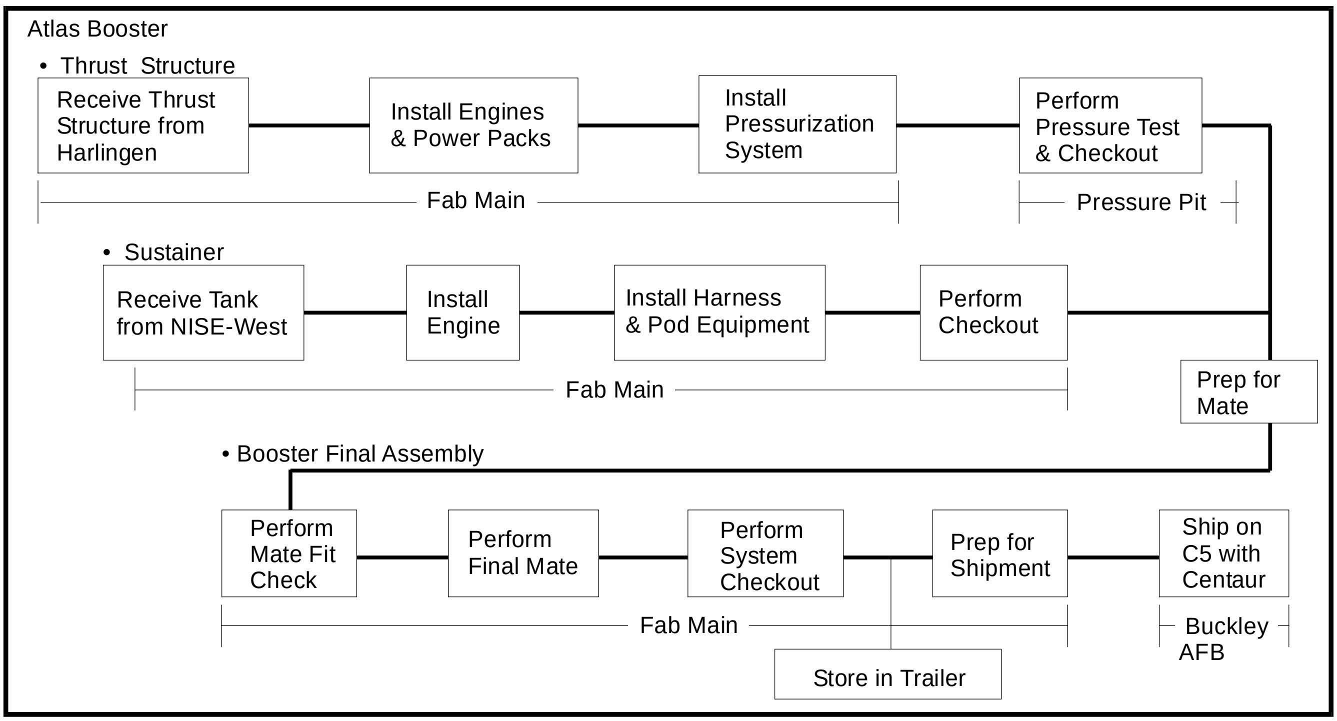


Figure A.3.2-1 Denver Product Flow for the Atlas IIA/IIAS Booster

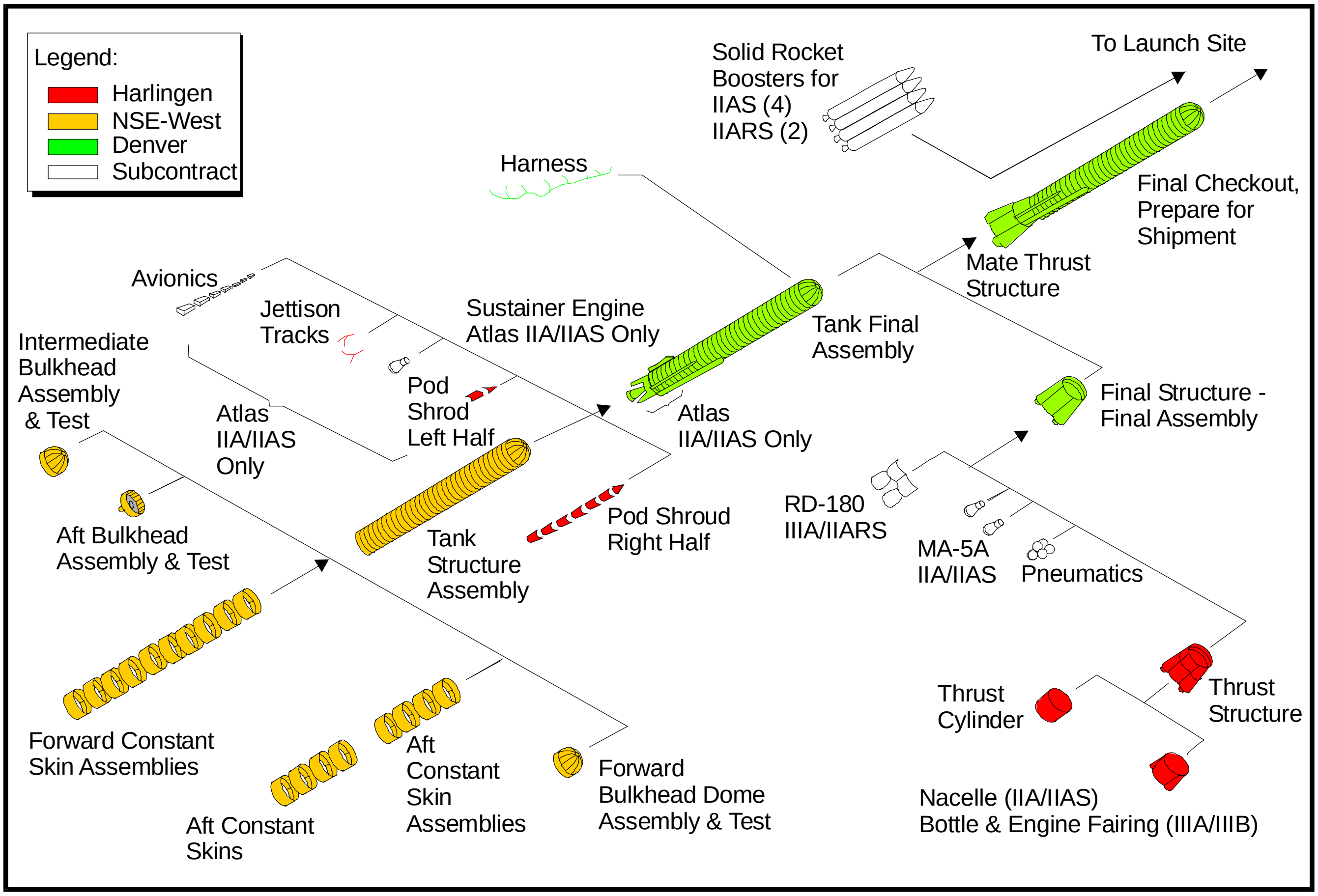


Figure A.3.2-2 Comprehensive Atlas Booster Assembly Flow

Atlas Centaur

Receive Tank from NISE-West	Apply Fixed Foam	Install Engines & Equipment Module	Perform Test & Checkout	Perform Pressure Test	Store in Dock	Prep for Shipment	Ship on C5 with Booster
Fixed-Foam Facility in VTF		FAB Clean Room		Pressure Pit	FAB Main	FAB Clean Room	Buckley AFB

Figure A.3.2-3 Denver Product Flow for the Centaur Upper Stage

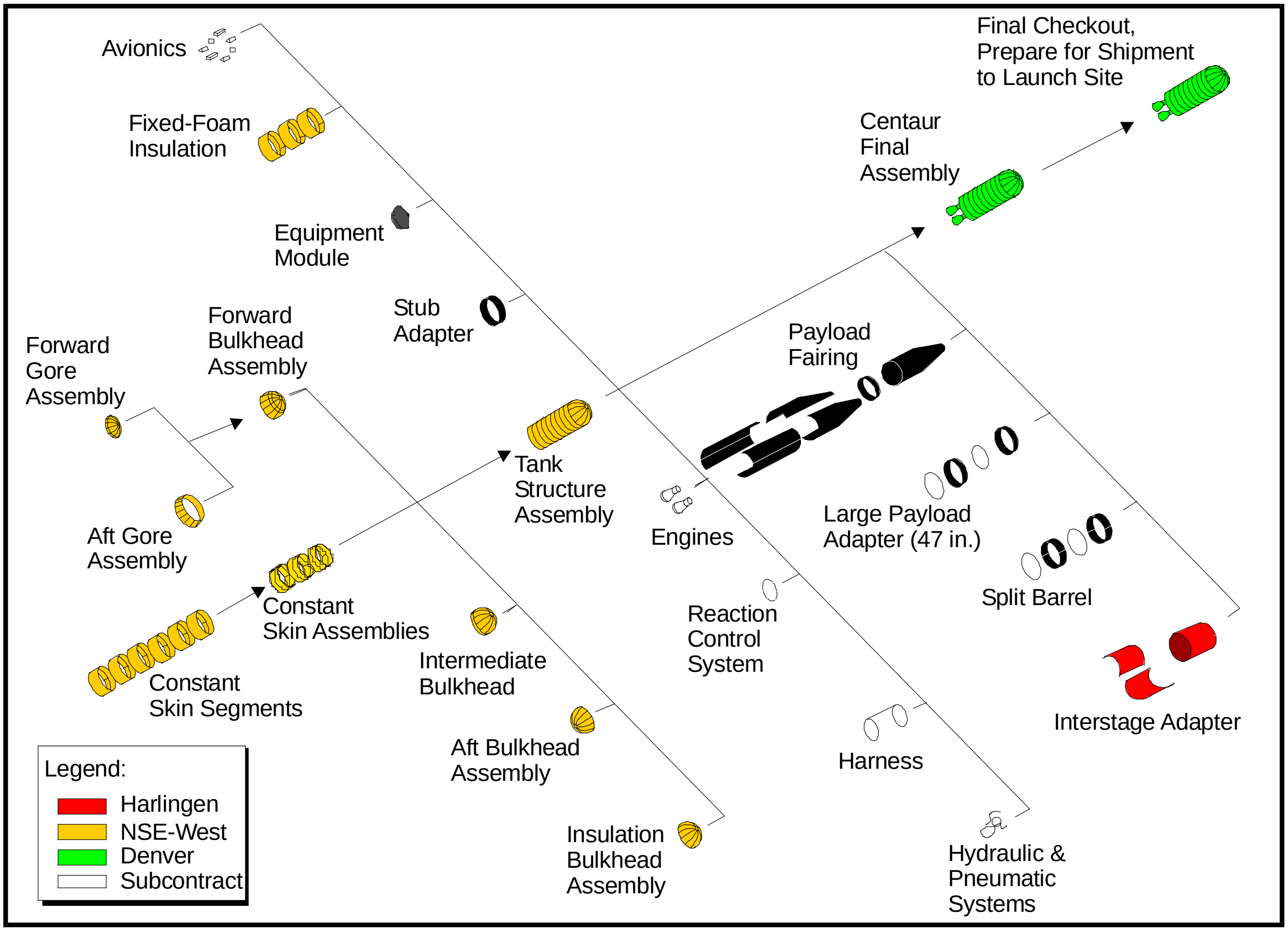


Figure A.3.2-4 Comprehensive Centaur and Jettisonable Component Assembly Flow

A.4 VEHICLE RELIABILITY

Many vehicle and process enhancements that have increased performance and operability over the history of the Atlas program have also increased reliability. The analysis method used by the Atlas program to calculate vehicle reliability is the reliability growth model developed by J.T. Duane. This method uses demonstrated flight data to predict the future performance of the Atlas launch system (Fig. A.4-1).The conservative Reliability Growth Model

is based on observations of the reduction in failure rate as the cumulative number of missions increases. When mean missions between flight failures are plotted on a log-log scale against the cumulative missions flown, the data points fall approximately in a straight line. The steepness of the slope of this line give a measure of the growth rate of improvements in reliability over time. In our case, the high value achieved is associated with performance upgrades of our operational launch vehicles. This growth rate is the result of our vigorous failure analysis and corrective action system, and our focus on controlling processes. We expect a high probability of Mission Success® occurrence by using only proven hardware of high inherent design reliability and processes that are controlled and stable.

Today's demonstrated reliability record of 0.944 is based on flight history (Fig. A.4-1). This record is based on timely, successful launches and satellite system support. For the Atlas program, it represents pride in playing an integral part in successful space program for more than 35 years.

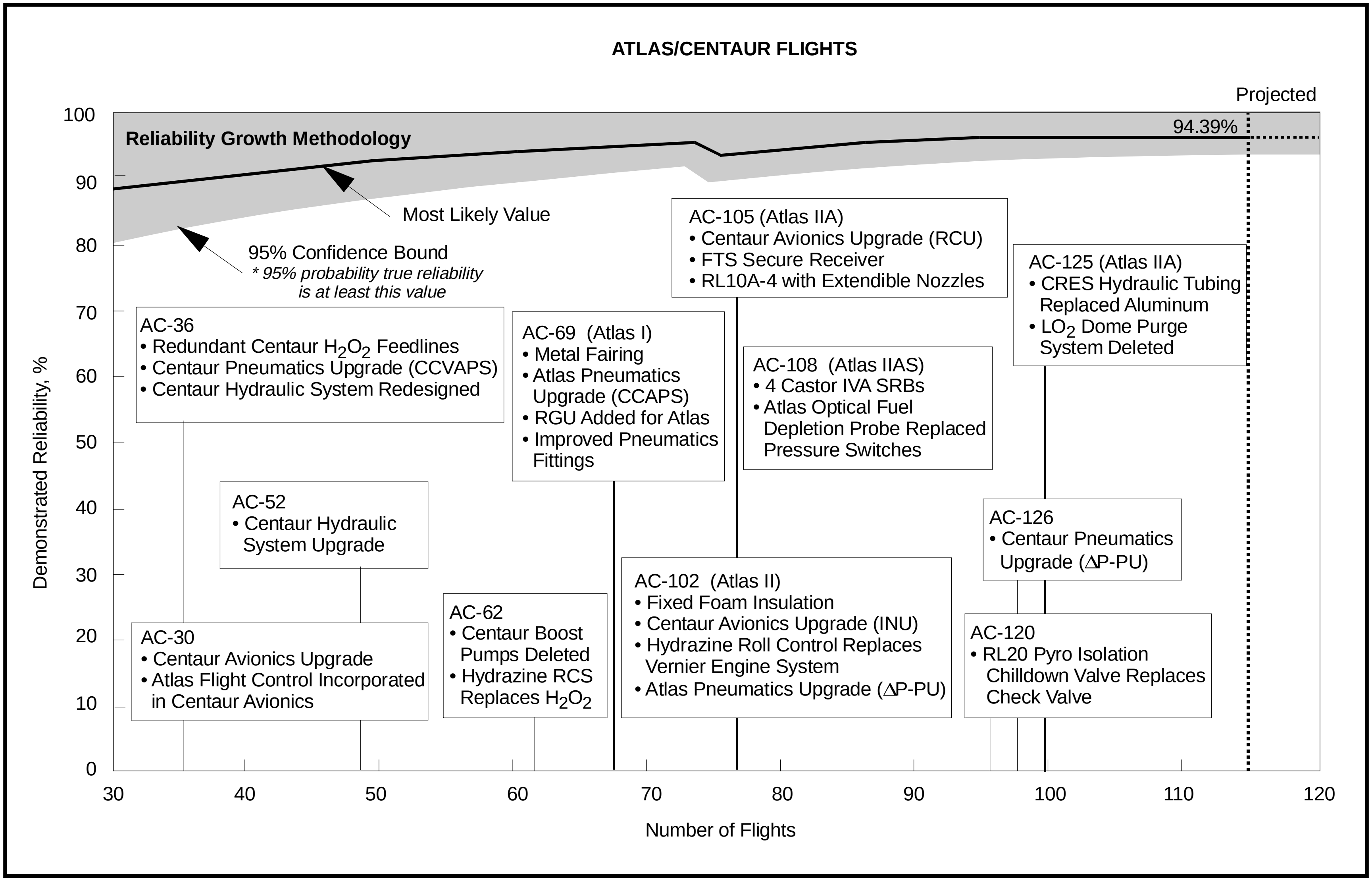


Figure A.4-1 Atlas/Centaur Reliability Using Duane Reliability Growth Method—Through AC-153

APPENDIX B—MISSION SUCCESS® AND PRODUCT ASSURANCE

B.1 ATLAS PRODUCT ASSURANCE SYSTEM

Lockheed Martin Astronautics (LMA) operates an ISO 9001 (International Organization for Standardization 9001:1994, Quality Systems, Model for quality assurance in design, development, production installation, and servicing) registered quality management system. LMA is internationally accredited through British Standards Institute (BSI) under registration number FM 35743. The registration was conferred on December 13, 1996, and includes LMA's Denver, CO; San Diego, CA; Harlingen, TX; Cape Canaveral Air Station; and Vandenberg Air Force Base facilities. Scope of the registration includes the design, development, test, manufacture, and assembly of advanced technology systems for space and defense, including space systems, launch systems, and ground systems.

Adherence to the ISO 9001 quality standard is revalidated at 6-month intervals by BSI, an independent, third-party registrar. In addition, ISO 9001 compliance is monitored and certified onsite by the U.S. Government's Defense Contract Management Command (DCMC), which also maintains insight into LMA processes.

ISO 9001 is executed through LMA internal Policies, Practices, and Procedures, which are described in our Product Delivery System Manual (PDSM), and the Atlas System Effectiveness Program Plan (SEPP), which supplements the PDSM. ISO 9001 is a basic quality management system that provides a framework of 20 major elements. The PDSM and SEPP address each element and provide an overview of the element and flowdown references to the individual procedures. LMA also includes Mission Success® accomplishment as a 21st element because of our deep focus on and commitment to Mission Success® principles.

The PDSM elements and section numbers are:

4.1 Management Responsibility
4.2 Quality System
4.3 Contract Review
4.4 Design Control
4.5 Document and Data Control
4.6 Purchasing
4.7 Control of Customer-Supplied Product
4.8 Product Identification and Traceability
4.9 Process Control
4.10 Inspection and Testing
4.11 Control of Inspection, Measuring, and Test Equipment
4.12 Inspection and Test Status
4.13 Control of Nonconforming Product
4.14 Corrective and Preventive Action
4.15 Handling, Storage, Packaging, Preservation, and Delivery
4.16 Control of Quality Records
4.17 Internal Quality Audits
4.18 Training
4.19 Servicing
4.20 Statistical Techniques
4.21 Mission Success®

B.1.1 Independent Oversight

The Mission Success® organization provides an evaluation for program and Astronautics management that is independent from Engineering and Product Assurance, to ensure all mission impacts at Lockheed Martin or at suppliers are resolved before launch. The Mission Success® organization provides proper visibility of these issues to management and customers. The Mission Success® group is the co-chair of the Atlas Space Program Reliability Board (SPRB) and is responsible for problem resolution. The SPRB is chaired by the program vice president and co-chaired by the Mission Success® organization and made up of technical experts to ensure complete investigation and resolution of all hardware concerns potentially affecting Mission Success® accomplishment. This is accomplished through the reporting of all functional failures to the Mission Success® organization, which in turn coordinates a technical evaluation with the appropriate subject-matter expert to establish mission impact. Impact failures are presented to the SPRB to ensure complete analysis and effective corrective action is taken to mitigate the mission impact. The Mission Success® organization is tasked to ensure all applicable flight constraints are resolved before launch. The Mission Success® organization provides technical concurrence with the processes used to resolve nonconformances and preclude recurrence.

Corrective/Preventative Action—The corrective and preventative action process ensures visibility and resolution of anomalous conditions affecting or potentially affecting product, process, or systems. This process encompasses customer concerns, internal activities, and supplier or subcontractor issues. The corrective and preventative action process ensures that problems are identified and documented, root cause is determined and recorded, corrective action is identified and reviewed for appropriateness, and corrective action is implemented and verified for effectiveness.

Alerts—The Government-Industry Data Exchange Program (GIDEP) review process includes evaluating GIDEP alerts for effects to Atlas hardware. All known alerts will be reviewed for effects before each flight.

Reviews—The Mission Success® organization participates in readiness reviews and data reviews to support program management in determining hardware flight worthiness and readiness. Figure B.1.1-1 is a flow of the Mission Success® acceptance process, including progressive reviews and acceptance throughout design, production, and launch site operations.

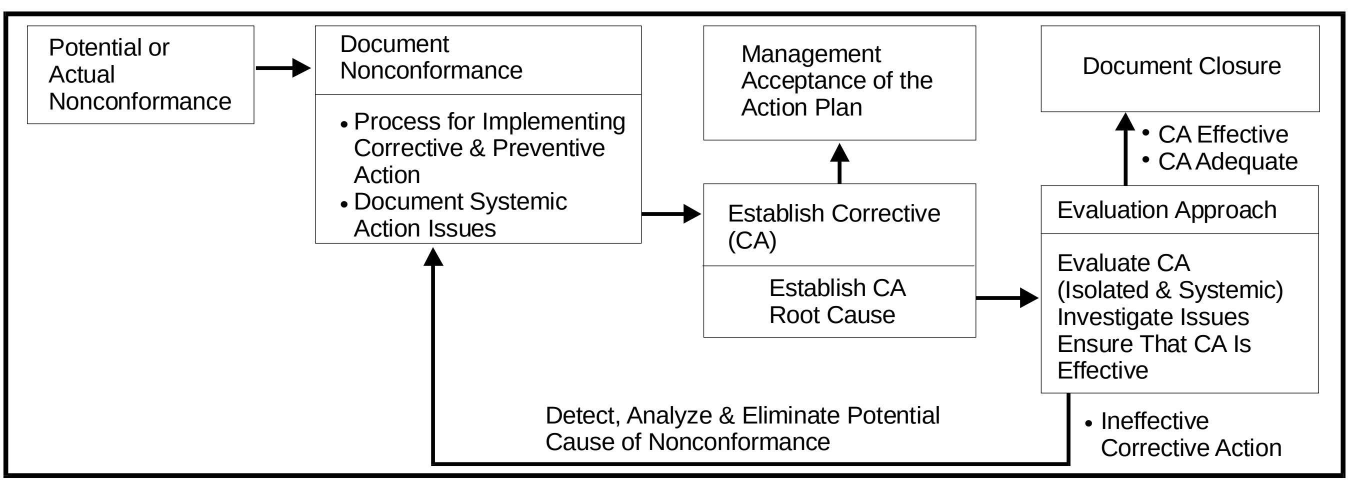


Figure B.1.1-1 Corrective and Preventive Action Process

B.1.2 Product Assurance

The Product Assurance branch of MS&PA ensures the quality of Atlas products, fabrication, and testing processes to achieve maximum effectiveness and continued customer satisfaction.

Quality is ensured through physical examination, measuring, test, process monitoring, and/or other methods as required to determine and control the quality of all deliverable airborne and ground equipment products and services. Independent verification includes mandatory inspection points; witnessing, monitoring, and surveillance; statistical methods; data analysis; trending; sampling plans; work instruction, build, and test documentation; associated data reviews; process control; audit; acceptance tooling; and/or other techniques suited to the products or processes being verified. Responsibilities are defined in the following paragraphs.

Audit—Documented procedures are established and maintained for planning, performing, reporting, and follow-up of internal audits. The Astronautics Internal Audit Program coordinates with other audit organizations, including corporate internal audit and customer audit teams, to maximize effectiveness.

The Management System Assessment organization of MS&PA has responsibility for the management and oversight of the Internal Audit Program. Auditors are independent of the person responsible for taking corrective action.

Design Reviews—Product Assurance participates at an appropriate level in conceptual, preliminary, critical, and drawing-level design reviews. The design review activity provides requirements flowdown, critical characteristics identification, acceptance strategy development, and method analysis. This ensures inspectability and acceptability of detailed drawings and specifications and promotes producibility.

Parts, Materials, and Processes Control Board (PMPCB)—The PMPCB provides the method for ensuring the use of proven parts, materials, and processes across the Atlas program. The board is the primary channel of communication for the interchange of PMP information analyses. The PMPCB provides direction for procurement activities to support program schedules. This includes the direction to order parts, assign priorities, and the providence of device lot failure mitigation.

Change Control—All engineering changes are managed and approved by an Engineering Review Board (ERB) and implemented by the Change Control Board. All preliminary changes are presented to a combined Product Support Team (PST) and Change Integration Board (CIB). This multidiscipline team develops the detailed scope of the change and provides inputs for scheduling, process planning, material planning, and configuration management. The PST and CIB provide status of each change through final engineering release and coordinate release with the product schedule and delivery needs. Product Assurance Engineering participates in this process, ensuring each change yields inspectable and acceptable products of known and controlled configuration.

Work Instructions—All work affecting quality is prescribed in documented instructions of a type appropriate to the circumstances. Work instructions encompass purchasing, handling, machining, assembling, fabricating, processing, inspection, testing, modification, installation, and any other treatment of product, facilities, standards, or equipment. The preparations and maintenance of work instructions and manufacturing processes are monitored as a function of the quality program.

Supplier Quality—A supplier rating system is maintained to monitor the performance of suppliers. Atlas hardware is procured from approved suppliers with acceptable ratings. An assessment of hardware criticality and supplier performance is performed to ensure the quality of supplier-built hardware. Based on this assessment, quality source representatives are assigned to certain suppliers and critical hardware to ensure hardware conformance to requirements during the supplier's operation through shipment.

Acceptance Status—Objective evidence of compliance to requirements of contract, specifications, configuration, and process control is maintained and made available to hardware acceptance teams at all manufacturing and test facilities. Product Assurance provides a positive system for identification of the inspection and acceptance status of products. This is accomplished by stamps, tags, routing cards, move tickets, build records, and/or other control devices.

Identification and Stamp Control—Inspection, fabrication, workmanship, including physical or electronic stamps, signatures, and acceptance and status markings are controlled and traceable to the individuals performing those functions.

Nonconforming Hardware—When material is initially found to be nonconforming, it is examined by preliminary material review (PMR) certified personnel. Assistance by inspection, manufacturing, and engineering personnel is sometimes necessary to determine if the nonconformance can be eliminated through rework, scrapping, or by returning to the supplier. If none of these criteria can be met, the material is referred to the Material Review Board (MRB) for disposition. Product Assurance ensures that all nonconforming material is identified and controlled to preclude its subsequent use in deliverable items without proper disposition. Quality Engineering chairs the MRB to determine appropriate disposition of nonconforming material. The board includes a representative from the Atlas Engineering organization, who is responsible for product design. All MRB members are required to be certified and approved by Atlas Product Assurance.

Software Quality—The Software Quality Evaluation Plan (SQEP) is designed to provide the methods necessary to:

1) Preclude the incorporation of deficiencies;
2) To detect introduced deficiencies;
3) To effect corrective action when anomalies occur.

The Atlas software development process is designed to build quality into the software and documentation, and to maintain levels of quality throughout the life cycle of the software. This includes normal technical evaluations and management reviews necessary to achieve this goal.

Record Retention—All Product Assurance records for the Atlas program are retained as required by Contract, Policy and Procedures, and specific program direction.

Training and Certification—The Certification Board is responsible for ensuring integrity in product development, test, and operations. The board ensures that personnel requiring special skills in fabrication, handling, test, maintenance, operations, and inspection of products have been trained and are qualified to ensure their capability to perform critical functions. Board responsibilities include certification of individuals and crews and oversight of offsite certification boards.

Metrics—Metrics are maintained to provide a continuous assessment of program performance, to control/reduce program costs, to ensure continued Mission Success® accomplishment, and to drive the program toward improvement. Examples of metrics maintained are:

1) Nonconformances per 1,000 hours;
2) MRBs per 1,000 hours;
3) Escapements;
4) Recurring nonconformances;
5) Supplier liability;
6) Foreign object incidents;
7) Aging of nonconformance documents.

Calibration—A calibration is maintained and documented to ensure that supplies and services presented for acceptance conform with prescribed technical requirements. This system applies to adequacy of standards, environmental control, intervals of calibration, procedures, out-of-tolerance evaluation, statuses, sources, application and records, control of subcontractor calibrations, and storage and handling.

Acceptance Tooling—When production jigs, fixtures, tooling masters, templates, patterns, test software, and other devices are used as a method for acceptance, they are proven for accuracy before release and at subsequently periodic intervals to ensure that their accuracies meet or exceed product requirements.

B.1.3 Launch Site Operations

Launch Operations Product Assurance supports launch sites during launch vehicle processing, ground systems maintenance and installations, and checkout of modifications.

Atlas Product Assurance Engineering is the focal point for the coordination of facility issues with Vandenberg Air Force Base and Cape Canaveral Air Station Product Assurance personnel. In cases where products are to be shipped with open engineering or open work, Product Assurance provides the necessary coordination with site personnel. This single-point contact between Denver and the sites ensures clear communication and provides a filter to minimize open items.

Launch processing oversight is conducted by the assignment of a vehicle-peculiar Mission Success® engineering specialist. The engineering specialist ensures that vehicle noncompliance issues have resolution and acts as the customer coordinator on vehicle certificates of conformance and flight constraint close-outs. This specialist follows his assigned vehicle throughout the production and launch sequence and reports on vehicle readiness in a series of program reviews (Fig. B.1.1-1).

All launch site operations are an extension of factory operations and are covered by the same requirements for reporting, control, and problem resolution.

B.1.3.1 Payload Operations—Payload Operations is responsible for the coordination and implementation of integrated spacecraft and launch vehicle tasks. The system engineer is responsible for integration activities, ensuring compliance to specifications and customer direction.

For launch vehicle activities, Payload Operations responds as the customer's agent in coordinating integrated activities and providing support as requested by the customer. Lockheed Martin personnel at the launch site participate in the total quality system process (oversight, engineering, and inspection) during payload integration at commercial and government processing facilities. This effort is inclusive of mating to the payload adapters, encapsulation, radio frequency (RF) checkout, integration testing, and transportation to the launch pad.

B.1.3.2 Safety Operations—Safety Operations provides onsite support from hardware receipt through launch operations to ensure the safety of personnel and hardware through enforcement of safety policies and procedures.

Safety operations performs inspections for unsafe conditions or unsafe work practices relating to industrial, material handling, fire prevention, and environmental/chemical safety.

System Safety is responsible for coordination of technical issues with the 45th Space Wing/30th Space Wing. They perform hazards analyses (i.e., preliminary hazard analysis, system hazard analysis). They are responsible for coordination of the review and check listing of all new "basic" hazardous procedures. They identify hazard controls or document requirements that will control hazards. Where necessary, they develop appropriate hazard controls. System Safety documents analyses and develops written reports documenting compliance with procedures.

APPENDIX C—SPACECRAFT DATA REQUIREMENTS

The items listed in this appendix are representative of the information required for spacecraft integration and launch activities. Additional information may be required for specific spacecraft.

C.1 INTERFACE CONTROL DOCUMENT INPUTS

Table C.1-1 indicates the spacecraft information required to assess the compatibility of the spacecraft with the Atlas launch vehicle. Data usually are provided by the customer in the form of an interface requirements document (IRD) and are the basis for preparing the interface control document (ICD). Shaded items should be provided for a preliminary compatibility assessment, while all items should be completed for a detailed assessment. The shaded items are typically supplied for the spacecraft before a proposal is offered for Atlas Launch Services. These lists are generalized and apply to any candidate mission. In cases in which Lockheed Martin has experience with the satellite bus or satellite contractor, less information can be initially provided, (assuming the spacecraft contractor is willing to use a "same as mission _____" designation for purposes of assessing preliminary compatibility. A complete IRD is typically supplied within 30 days of contract signing.

Tables C.1-2 through C.1-8 indicate spacecraft data required after contract signature to start integration of the spacecraft. The asterisks in these tables indicate data desired at an initial meeting between Lockheed Martin and the customer. These data will provide the detailed information required to fully integrate the spacecraft and determine such items as optimum mission trajectory, and verify compatibility of the launch vehicle environments and interfaces.

Table C.1-1 Spacecraft Information Worksheet

For a Preliminary Compatibility Assessment, All Shaded Items Should Be Completed. For a Detailed Compatibility Assessment, All Items Should Be Completed.		
Spacecraft Name: Spacecraft Owner: Name of Principal Contact: Telephone Number: Date:	Spacecraft Manufacturer: Spacecraft Model No.: Number of Launches: Date of Launches:	
Spacecraft Design Parameter	**SI Units**	**English Units**
TRAJECTORY REQUIREMENTS		
• Satellite Mass	______ kg	______ lbm
• Minimum Satellite Lifetime	______ yr	______ yr
• Final Orbit Apogee	______ km	______ km
• Final Orbit Perigee	______ km	______ km
• Final Orbit Inclination	______ °	______ °
• Propulsion-Propellant Type, Orbit Insertion		
• Propulsion-Propellant Type, Stationkeeping		
• Propulsion-Multiple Burn Capability (Y/N)		
• Propulsion-Propellant Mass	______ kg	______ lbm
• Propulsion-Effective I_{sp}	______ s	______ s
• Maximum Apogee Allowable	______ km	______ nmi
• Minimum Perigee Allowable	______ km	______ nmi
• Argument of Perigee Requirement	______ °	______ °
• Right Ascension of Ascending Node Requirement	______ °	______ °
• Apogee Accuracy Requirement	______ km	______ km
• Perigee Accuracy Requirement	______ km	______ km
• Inclination Accuracy Requirement	______ °	______ °
• Argument of Perigee Accuracy Requirement	______ °	______ °
• Right Ascension of Ascending Node Accuracy Requirement	______ °	______ °

Table C.1-1 (cont)

Spacecraft Design Parameter	SI Units	English Units
MECHANICAL INTERFACE		
• Spacecraft Mechanical Drawing (Launch Configuration)		
• Spacecraft Effective Diameter	______ mm	______ in.
• Spacecraft Height	______ mm	______ in.
• Spacecraft/Launch Vehicle Interface Diameter	______ mm	______ in.
• Payload Sep System Supplier (Spacecraft or Launch Veh)		
• Payload Adapter Supplier (Spacecraft or Launch Vehicle)		
• Maximum Spacecraft Cross-Sectional Area	______ m^2	______ ft^2
• Number & Size Payload Fairing Access Doors	______ mm x mm	______ in. x in.
• Preseparation RF Transmission Requirement	______ band	______ band
ELECTRICAL INTERFACE		
• Spacecraft Electrical Drawing	•	•
• Number of Launch Vehicle Signals Required	•	•
• Number of Separation Discretes Required	•	•
• Number of Umbilicals & Pins/Umbilical	•	•
• Curve of Spacecraft-Induced Electric Field Radiated Emissions	• ______ dBµ V/m	• ______ MHz
• Curve of Spacecraft-Radiated Susceptibility	• ______ dBµ V/m	• ______ MHz
• Number of Instrumentation Analogs Required	•	•
THERMAL ENVIRONMENT		
• Prelaunch Ground Transport Temperature Range	______ °C	______°F
• Prelaunch Launch Pad Temperature Range	______ °C	______ °F
• Maximum Prelaunch Gas Impingement Velocity	______ m/s	______ ft/s
• Maximum Ascent Heat Flux	______W/m^2	______$Btu/hr\text{-}ft^2$
• Maximum Free-Molecular Heat Flux	______W/m^2	______$Btu/hr\text{-}ft^2$
• Maximum Fairing Ascent Depressurization Rate	______ mbar/s	______ psi/s
• Spacecraft Vented Volume(s)	______ m^3	______ ft^3
• Spacecraft Vent Area(s)	______ cm^2	______ $in.^2$
• Prelaunch Relative Humidity Range	______ %	______ %
• Preseparation Spacecraft Power Dissipation	______ W	______ Btu/hr
• Maximum Free-Stream Dynamic Pressure	______ mbar	______ psi
DYNAMIC ENVIRONMENT		
• Maximum Allowable Flight Acoustics	______ dB OA	______ dB OA
• Allowable Acoustics Curve		
• Maximum Allowable Sine Vibration	______ G_{RMS}	______ G_{RMS}
• Allowable Sine Vibration Curve	______ G_{RMS}	______ G_{RMS}
• Maximum Allowable Shock	______ g	______ g
• Allowable Shock Curve		
• Maximum Acceleration (Static + Dynamic) Lateral	______ g	______ g
• Maximum Acceleration (Static + Dynamic) Longitudinal	______ g	______ g
• Fundamental Natural Frequency—Lateral	______ Hz	______ Hz
• Fundamental Natural Frequency—Longitudinal	______ Hz	______ Hz
• cg—Thrust Axis (Origin at Separation Plane)	______ mm	______ in.
• cg—Y Axis	______ mm	______ in.
• cg—Z Axis	______ mm	______ in.
• cg Tolerance—Thrust Axis	______ ±mm	______ ±in.
• cg Tolerance—Y Axis	______ ±mm	______ ±in.
• cg Tolerance—Z Axis	______ ±mm	______ ±in.

Table C.1-1 (concl)

Spacecraft Design Parameter	**SI Units**	**English Units**
CONTAMINATION REQUIREMENTS		
• Fairing Air Cleanliness	______ Class	______ Class
• Maximum Deposition on Spacecraft Surfaces	______ mg/m^2	______ mg/m^2
• Outgassing—Total Weight Loss	______ %	______ %
• Outgassing—Volatile Condensable Material Weight Loss	______ %	______ %
SPACECRAFT DESIGN SAFETY FACTORS		
• Airborne Pressure Vessel Burst Safety Factor		
• Airborne Pressure System Burst Safety Factor		
• Structural Limit (Yield) Safety Factor		
• Structural Ultimate Safety Factor		
• Battery Burst Safety Factor		
SPACECRAFT QUALIFICATION TEST PROGRAM		
• Acoustic Qualification	______ +dB	______ +dB
• Sine Vibration Qualification Safety Factor		
• Shock Qualification Safety Factor		
• Loads Qualification Safety Factor		
ORBIT INJECTION CONDITIONS		
• Range of Separation Velocity	______ m/s	______ ft/s
• Max Angular Rate at Separation-Roll	______ rpm	______ rpm
• Max Angular Rate Uncertainty-Roll	______ ±rpm	______ ±rpm
• Max Angular Rate at Separation—Pitch & Yaw	______ rpm	______ rpm
• Max Angular Rate Uncertainty—Pitch & Yaw	______ ±rpm	______ ±rpm
• Max Angular Acceleration	______ rad/s^2	______ rad/s^2
• Max Pointing Error Requirement	______ °	______ °
• Max Allowable Tip-Off Rate	______ °/s	______ °/s
• Coefficients of Inertia—Ixx (x=Thrust Axis)	______ $kg\ m^2$	______ $slug\ ft^2$
• Coefficients of Inertia—Ixx Tolerance	______ $\pm kg\ m^2$	______ $\pm slug\ ft^2$
• Coefficients of Inertia—Iyy	______ $kg\ m^2$	______ $slug\ ft^2$
• Coefficients of Inertia—Iyy Tolerance	______ $\pm kg\ m^2$	______ $\pm slug\ ft^2$
• Coefficients of Inertia—Izz	______ $kg\ m^2$	______ $slug\ ft^2$
• Coefficients of Inertia—Izz Tolerance	______ $\pm kg\ m^2$	______ $\pm slug\ ft^2$
• Coefficients of Inertia—Ixy	______ $kg\ m^2$	______ $slug\ ft^2$
• Coefficients of Inertia—Ixy Tolerance	______ $\pm kg\ m^2$	______ $\pm slug\ ft^2$
• Coefficients of Inertia—Iyz	______ $kg\ m^2$	______ $slug\ ft^2$
• Coefficients of Inertia—Iyz Tolerance	______ $\pm kg\ m^2$	______ $\pm slug\ ft^2$
• Coefficients of Inertia—Ixz	______ $kg\ m^2$	______ $slug\ ft^2$
• Coefficients of Inertia—Ixz Tolerance	______ $\pm kg\ m^2$	______ $\pm slug\ ft^2$

Table C.1-2 Mission Requirements

Type of Data	Scope of Data
Number of Launches*	
Frequency of Launches*	
Spacecraft Orbit Parameters Including Tolerances (Park Orbit, Transfer Orbit)*	• Apogee Altitude • Perigee Altitude • Inclination • Argument of Perigee • RAAN
Launch Window Constraints • Preseparation Function*	• Prearm • Arm • Spacecraft Equipment Deployment Timing & Constraints • Acceleration Constraints (Pitch, Yaw, Roll) • Attitude Constraints • Spinup Requirements
Separation Parameters (Including Tolerances)*	• Desired Spin Axis • Angular Rate of Spacecraft • Orientation (Pitch, Yaw & Roll Axis) • Acceleration Constraints
Any Special Trajectory Requirements	• Boost Phase • Coast Phase • Free Molecular Heating Constraints • Thermal Maneuvers • Separation Within View of Tlm & Tracking Ground Station • Telemetry Dipout Maneuvers
Note: * Information Desired at Initial Meeting Between Lockheed Martin & Customer After Contract Award	

Table C.1-3 Spacecraft Characteristics

Type of Data	Scope of Data
Configuration Drawings*	• Drawings Showing the Configuration, Shape, Dimensions & Protrusions into the Mounting Adapter (Ground Launch & Deployment Configurations) • Coordinates (Spacecraft Relative to Launch Vehicle) • Special Clearance Requirements
Apogee Kick Motor*	• Manufacturer's Designation • Thrust • Specific Impulse • Burn Action Time • Propellant Offload Limit
Mass Properties (Launch & Orbit Configurations)*	• Weight—Specify Total, Separable & Retained Masses • Center of Gravity—Specify in 3 Orthogonal Coordinates Parallel to the Booster Roll, Pitch & Yaw Axes for Total, Separable & Retained Masses • Changes in cg Due to Deployment of Appendages • Propellant Slosh Models
Moments of Inertia (Launch & Orbit Configurations)	• Specify About the Axes Through the Spacecraft cg That Are Parallel to the Atlas Roll, Pitch & Yaw Axes for Total, Separable, & Retained Masses
Structural Characteristics	• Spring Ratio of Structure, Elastic Deflection Constants, Shear Stiffness, Dynamic Model, Bending Moments & Shear Loads at Atlas/Centaur/Spacecraft Interface & Limitations To Include Acoustic, Shock, Acceleration, Temperature & Bending Moments

Table C.1-3 (cont)

Type of Data	Scope of Data
Dynamic Model for 3-D Loads Analysis	• Generalized Stiffness Matrix (Ref Paragraph C.3.2 for Details) • Generalized Mass Matrix • Description of the Model, Geometry & Coordinate System • Loads Transformation Matrix Note: Models Must Include Rigid Body & Normal Modes
Handling Constraints	• Spacecraft Orientation During Ground Transport • Spacecraft Handling Limits (e.g., Acceleration Constraints)
Spacecraft Critical Orientation During:	• Location & Direction of Antennas Checkout, Prelaunch & Orbit • Location, Look Angle & Frequency of Sensors • Location & Size of Solar Arrays
Safety Items	• General Systems Description • Basic Spacecraft Mission • Prelaunch Through Launch Configuration • Orbital Parameters • Functional Subsystems • Hazardous Subsystems • Ground Operations Flow • Flight Hardware Descriptions (Safety-Oriented) • Structural/Mechanical Subsystems • Propellant/Propulsion Subsystems • Pressurized Subsystems • Ordnance Subsystems • Electrical & Electronic Subsystems • Nonionizing Radiation Subsystems (RF/Laser) • Ionizing Radiation Subsystems • Hazardous Materials • Thermal Control Subsystems • Acoustical Subsystems Note: Hazard Identification/Controls/Verification Method Summaries for Each Subsystem GSE Descriptions • Mechanical GSE • Propellant/Propulsion GSE • Pressure GSE • Ordnance GSE • Electrical GSE • RF/Laser GSE • Ionizing Radiation GSE • Hazardous Materials GSE Note: Hazard Identification/Controls/Verification Method Summaries for Each Item Ground Operations • Hazardous Ground Operations • Procedures • Transport Configuration
Note: For Each Data Submittal—Identify Each Item/Operation Applicable to PPF, HPF, or Launch Site	
Thermal Characteristics	• Spacecraft Thermal Math Model (Ref Sect. C.3.3) • Emissivity • Conductivity • Resistivity • Thermal Constraints (Maximum & Minimum Allowable Temperature) • Heat Generation (e.g., Sources, Heat, Time of Operation)

Table C.1-3 (concl)

Type of Data	Scope of Data
Contamination Control	• Requirements for Ground-Supplied Services • In-Flight Conditions (e.g., During Ascent & After PLF Jettison) • Surface Sensitivity (e.g., Susceptibility to Propellants, Gases, & Exhaust Products)
RF Radiation	• Characteristics (e.g., Power Levels, Frequency, & Duration for Checkout & Flight Configuration) • Locations (e.g., Location of Receivers & Transmitters on Spacecraft) • Checkout Requirements (e.g., Open-Loop, Closed-Loop, Prelaunch, Ascent Phase)
Note: * Information Desired at Initial Meeting Between Lockheed Martin & Customer After Contract Award	

Table C.1-4 Aerospace Vehicle Equipment (AVE) Requirements (Mechanical)

Type of Data	Scope of Data
Mechanical Interfaces*	• Base Diameter of Spacecraft Interface* • Structural Attachments at SC Interface* • Required Accessibility to Spacecraft in Mated Condition* • Extent of Equipment Remaining with Adapter After Spacecraft Separation* • Degree of Environmental Control Required • Spacecraft Pressurization, Fueling System Connector Type & Location; Timeline for Pressure/Fuel System Operation • Spacecraft/Adapter Venting Requirements
PLF Requirements	• Heating Constraints • Venting Characteristics (e.g., Quantity, Timing & Nature of Gases Vented from P/L) • RF Reradiation System (RF Band, SC Antenna Location, etc) • PLF Separation (e.g., Altitude, Cleanliness, Shock, Aeroheating & Airload Constraints) • Acoustic Environment Constraints • Special Environmental Requirements
Preflight Environment	• PLF Separation (e.g., Altitude, Cleanliness, Shock, Aeroheating & Airload Constraints) • Acoustic Environment Constraints • Special Environmental Requirements • Requirements – Cleanliness – Temperature & Relative Humidity – Air Conditioning – Air Impingement Limits • Monitoring & Verification Requirements
Umbilical Requirements*	• Separation from Launch Vehicle • Flyaway at Launch • Manual Disconnect (Including When)
Materials	• Special Compatibility Requirements • Outgassing Requirements
Note: * Information Desired at Initial Meeting Between Lockheed Martin & Customer After Contract Award	

Table C.1-5 Aerospace Vehicle Equipment (AVE) Requirements (Electrical)

Type of Data	Scope of Data
Power Rqmts (Current, Duration, Function Time & Tolerances)*	• 28-Vdc Power • Other Power • Overcurrent Protection
Command Discrete Signals*	• Number* • Sequence • Timing (Including Duration, Tolerance, Repetition Rate, etc) • Voltage (Nominal & Tolerance) • Frequency (Nominal & Tolerance) • Current (Nominal & Tolerance) • When Discretes Are for EED Activation, Specify Minimum, Maximum & Nominal Fire Current; Minimum & Maximum Resistance; Minimum Fire Time; Operating Temperature Range & Manufacturer's Identification of Device
Other Command & Status Signals	• Status Displays • Abort Signals • Range Safety Destruct • Inadvertent Separation Destruct
Ordnance Circuits	• Safe/Arm Requirements
Telemetry Requirements*	• Quantity of Spacecraft Measurements Required To Be Transmitted by Atlas Telemetry & Type (e.g., Temperature, Vibration, Pressure, etc); Details Concerned with Related System Including Operating Characteristics (Response Definition of System) & Locations & Anticipated Time of Operation • Impedance, Capacitance, Operating Range & Full-Scale Range of Each Measurement • Signal Conditioning Requirements (e.g., Input Impedance, Impedance Circuit Load Limits, • Overcurrent Protection & Signal-to-Noise Ratio) • Discrete Events (Bilevel) • Analog Measurements • Transducers Required To Be Furnished by LV Contractor • Minimum Acceptable Frequency Response for Each Measurement • Minimum Acceptable System Error for Each Measurement (Sampling Rate Is Also Governedby This Requirement) • Period of Flight for Which Data from Each Measurement Are of Interest (e.g., from Liftoff to Space Vehicle Separation)* • Atlas Flight Data Required by Spacecraft Contractor
Bonding	• Bonding Requirements at Interface (MIL-B-5087, Class R for LV) • Material & Finishes at Interface (for Compatibility with LV Adapter)
EMC	• Test or Analyze Spacecraft Emissions & Susceptibility • EMC Protection Philosophy for Low-Power, High-Power & Pyrotechnic Circuits • LV & Site Emissions (Provided by Lockheed Martin)
Grounding Philosophy	• Structure (e.g., Use of Structural As Ground & Current Levels) • Electrical Equipment (e.g., Grounding Method for Signals & Power Supplies) • Single-Point Ground (e.g., Location & Related Equipment)
Interface Connectors*	• *Connector Item (e.g., Location & Function) • Connector Details • Electrical Characteristics of Signal on Each Pin
Shielding Requirements	• Each Conductor or Pair • Overall • Grounding Locations for Termination
Note: * Desired for Initial Integration Meeting with Lockheed Martin After Contract Award	

Table C.1-6 Test Operations

Type of Data	Scope of Data
Spacecraft Launch Vehicle Integration	• Sequence from Spacecraft Delivery Through Mating with the LV • Handling Equipment Required • Lockheed Martin-Provided Protective Covers or Work Shields Required • Identify the Space Envelope, Installation, Clearance & Work Area Requirements • Any Special Encapsulation Requirements • Support Services Required
Spacecraft Checkout AGE & Cabinet Data	• List of All AGE & Location Where Used (e.g., Storage Requirements on the Launch Pad) • Installation Criteria for AGE Items: – Size & Weight – Mounting Provisions – Grounding & Bonding Requirements – Proximity to the Spacecraft When In Use – Period of Use – Environmental Requirements – Compatibility with Range Safety Requirements & LV Propellants – Access Space to Cabinets Required for Work Area, Door Swing, Slideout Panels, etc – Cable Entry Provisions & Terminal Board Types in Cabinets &/or Interface Receptacle Locations & Types – Power Requirements & Characteristics of Power for Each Cabinet
Spacecraft Environmental Protection (Preflight)	• Environmental Protection Requirements by Area, Including Cleanliness Requirements: – Spacecraft Room – Transport to Launch Pad – Mating – inside PLF – During Countdown • Air-Conditioning Requirements for Applicable Area (Pad Area) by: – Temperature Range – Humidity Range – Particle Limitation – Impingement Velocity Limit – Flow Rate • Indicate if Space Vehicle Is Not Compatible with LV Propellants & What Safety Measures Will Be Required • Environmental Monitoring & Verification Requirements
Space Access Requirements	• Access for Space Vehicle Mating & Checkout • Access During Transportation to the Launch Pad & Erection Onto the Atlas • Access for Checkout & Achieving Readiness Prior to Fairing Installation • Access After Fairing Installation; State Location, Size of Opening & Inside Reach Required • Access During the Final Countdown, if Any • AGE Requirements for Emergency Removal
Umbilicals	• Ground Servicing Umbilicals by Function & Location in Excess of Atlas/Centaur Baseline • Structural Support Requirements & Retraction Mechanisms • Installation (e.g., When & by Whom Supplied & Installed)
Commodities Required for Both Spacecraft, AGE & Personnel	• Gases, Propellants, Chilled Water & Cryogenics in Compliance with Ozone-Depleting Chemicals Requirements • Source (e.g., Spacecraft or LV) • Commodities for Personnel (e.g., Work Areas, Desks, Phones)
Miscellaneous	Spacecraft Guidance Alignment Requirements

Table C.1-7 AGE/Facility Requirements (Electrical)

Type of Data	Scope of Data
Space Vehicle Electrical Conductor Data	SC System Schematic Showing All Connectors Required Between SC Equipment, & SC Terminal Board Position or Receptacle Pin Assigned to Each Conductor; Electrical Characteristics of Each Connector Including Maximum End-to-End Resistance, Shielding, Capacitance & Spare Conductors
Electrical Power (AGE & Facility)	• Frequency, Voltage, Watts, Tolerance, Source • Isolation Requirements • Identify if Values Are Steady or Peak Loads • High-Voltage Transient Susceptibility
RF Transmission	• Antenna Reqts (e.g., Function, Location, Physical Characteristics, Beam Width & Direction & Line-of-Sight) • Frequency & Power Transmission • Operation
Cabling	• Any Cabling, Ducting, or Conduits To Be Installed in the Mobile Service Tower; Who Will Supply, Install, Checkout & Remove
Monitors & Controls	• Specify Which Signals from SC Are To Be Monitored During Readiness & Countdown; Power Source (SC, Atlas, Centaur) • Transmission Method (e.g., SC Tlm, LV Tlm, Landline, or LV Readiness Monitor) • Location of Data Evaluation Center, Evaluation Responsibility, Measurement Limits & Go/No-Go Constraints; Identify Where in the Operational Sequence Measurements Are To Be Monitored & Evaluated; Specify Frequency & Duration of Measurements • Video Output Characteristics of Telepaks (if Available) for Closed-Loop Prelaunch Checkout at the Launch Pad; Data To Include Location & Type of Interface Connector(s), & Characteristics of Signal at Source; This Includes Voltage Level, Output Impedance, Output Current Limitation, Maximum Frequency of Data Train & Output Loading Requirements

Table C.1-8 Test Operations

Type of Data	Scope of Data
Hardware Needs (Including Dates)*	• Electrical Simulators • Structural Simulators • *Master Drill Gage
Interface Test Requirements	• Structural Test • Fit Test • Compatibility Testing of Interfaces (Functional) • EMC Demonstration • LV/Spacecraft RF Interference Test • Environmental Demonstration Test
Launch Operations	• Detailed Sequence & Time Span of All Spacecraft-Related Launch Site Activities Including: AGE Installation, Facility Installation & Activities, Spacecraft Testing & Spacecraft Servicing • Recycle Requirements • Restrictions To Include Launch Site Activity Limitations, Constraints on Launch Vehicle Operations, Security Requirements & Personnel Access Limitations & Safety Precautions • Special Requirements Include Handling of Radioactive Materials, Security & Access Control • Support Requirements To Include Personnel, Communications & Data Reduction • Launch & Flight Requirements for Real-Time Data Readout, Postflight Data Analysis, Data Distribution, Postflight Facilities
Note: * Desired for Initial Integration Meeting with Lockheed Martin After Contract Award	

C.2 SPACECRAFT DESIGN REQUIREMENTS

Table C.2-1 lists specific requirements that should be certified by analysis and/or test by the spacecraft agency to be compatible for launch with Atlas/Centaur. Should the spacecraft not meet any of these requirements, Lockheed Martin will work with the customer to resolve the incompatibility.

Table C.2-1 Spacecraft Design Requirement

Spacecraft Design Requirements	Comment
MECHANICAL	
PLF Envelope (Figs. 4.1.1.4-1 Through 4.1.1.4-3)	
P/L Adapter Envelope (Figs. 4.1.1.4-4 Through 4.1.1.4-9)	
P/L Adapter Interface (Figs. 4.1.2.1-1 Through 4.1.2.1-9 & Figs. 4.1.2.2-1 Through 4.1.2.2-4)	If P/L Adapter Is SC Provided, I/F Can Be a Field Joint at the Equipment Module Ring or on the Top of the Spacer Adapter (Type C)
ELECTRICAL	
• Two or Fewer Separation Commands (Sect. 4.1.3)	
• 16 or Fewer Control Commands (28-V Discretes or Dry Loop) (Sect. 4.1.3)	
• Instrumentation I/F, 2 or Fewer Inputs for SC Separation Detection, 4 or Fewer Analog Inputs for General Use; 10 or Fewer Cmd Feedback Discretes, 2 or Fewer Serial Data I/F for Downlinking SC Data (Sect. 4.1.3)	
• Two Umbilical Connectors at SC Interface (Fig. 4.1.3-1)	
STRUCTURE & LOADS	
• Design Load Factors (Tables 3.2.1.1-1)	
• First Lateral Modes Above 10 Hz & First Axial Mode Above 15 Hz (Sect. 3.2.1.1)	
• SC Mass vs cg Range (Fig. 4.1.2.3-1)	
• Design FS per Applicable Range Safety Documentation & MIL-STD-1522 (or Submit Deviations for Review)	
ENVIRONMENT	
• SC Test Requirements (Sect. 3.3)	
• Quasi-Sinusoidal Vibration (Fig. 3.2.3-1)	
• Acoustic Levels in the PLF (Figs. 3.2.2-1 Through 3.2.2-4)	
• Shock Induced by PLF Jettison & P/L Separation (Fig. 3.2.4-1)	
• P/L Compartment Pressures & Depressurization Rates (Figs. 3.2.6-1 Through 3.2.6-4)	
• Gas Velocity Across SC Components < 4.9 m/s (16 ft/s)	For Medium Fill SC
• Gas Velocity Across SC Components 9.1 m/s (30 ft/s)	For Large Fill SC
• Electric Fields (Figs. 3.1.2.1-1 Through 3.1.2.2-3)	
• SC Radiation (Fig. 3.1.2.4-1) Limit	
• EM Environment at Launch Range TOR-95(5663)-1 & Section 3.1.2.3	
SAFETY	
• All SC Propellant Fill & Drain Valves & All Pressurant Fill & Vent Valves Readily Accessible When SC Is Fully Assembled & Serviced in Launch Configuration (Encapsulated & on Pad)	It Is Advisable To Accommodate Normal Servicing/Deservicing & Potential Emergency Backout Situation for New Spacecraft Design
• Requirements in Range Safety Regulation	
MISCELLANEOUS	
• See Atlas Launch Services Facilities Guide for SC Propellants & Specifications Available at Launch Site Fuel Storage Depot	
Note: Compliance with Ozone-Depleting Chemicals Regulation Is Required.	

C.3 SPACECRAFT INTEGRATION INPUTS

Table C.3-1 provides a list of typical spacecraft inputs required for the integration process, the approximate need date, and a brief description of the contents. Further details on some items are provided in the following sections.

C.3.1 Computer-Aided Design (CAD) Data Transfer Requirements

CAD data must be provided according to specified software formats. The Atlas program supports three UNIX-based CAD systems-Parametric Technology Corporation (PTC) Pro/Engineer (Pro/E), PTC CADDS 5, and Structural Dynamics Research Corporation (SDRC) I-DEAS Master Series. When CAD data do not come from a supported software platform, Lockheed Martin prefers to receive solid model data translated through the Standard for the Exchange of Product Model Data (STEP) converter. If this is not available, an Initial Graphics Exchange Specification (IGES) 4.0 or higher file from a three-dimensional (3-D) wireframe system or wireframe extracted from a solid model will suffice.

Prerequisites to Data Transfer—The following criteria should be met before transferring CAD data:

1) All entities (parts, assemblies, etc) should be visible (unblanked) and fontless (e.g. no center lines, phantom lines, thick lines);
2) Single part or model files containing 30,000 entities or more should be divided up into smaller files, preferably by assemblies, all having the same origin;
3) Single CADDS 5 models (containing _pd, _fd, _td, vp_links, draw files and execute routines) should be contained in the same subdirectory structure. The entire directory should then be transferred as a single compressed file;
4) Subfigures should not be included in the model, unless the subfigures are included in the data transfer.

Data Transfer—Tape media (8mm, 1/4-inch data cartridge, and 4mm DAT) is the preferred transfer method for all database files. Files should be compressed and transferred to tape using the UNIX operating system. The following tape archive retrieval (TAR) command syntax should be used: "tar cvf/dev/rmt0 partfilenames". DOS and Microsoft Windows based 3-1/2 inch disks can also be accommodated. The spacecraft contractor should verify that the tape files contain the desired results by

Table C.3-1 Spacecraft Inputs to Integration Process

SC Data Input	Typical Need Date	Comments
Interface Rqmts Doc	Program Kickoff	See Sect. C.1
Initial Target Specification	Program Kickoff	SC Weight, Target Orbit, Separation Attitude; See Sect. C.3.4
Range Safety Mission Orientation Briefing Input	6 wk Before Meeting Date	Top-Level Description of Spacecraft & Mission Design
CAD Model	30 day After Program Kickoff	See Sect. C.3.1
Coupled Loads Model	6 mo Before Design Review	See Sect. C.3.2
Thermal Models	5 mo Before Design Review	See Sect. C.3.3
Preliminary Launch Windows	5 mo Before Design Review	Support Thermal Analysis; See Sect. C.3.3
In-Flight Breakup Data	14 mo Before Launch	See Sect. C.3.6.4.3
Intact Impact Breakup Data	14 mo Before Launch	See Sect. C.3.6.4.2
Prelim SC MSPSP	14 mo Before Launch	See Sect. C.3.6.1
SC EMI/EMC Cert Letter	6 mo Before Launch	See Sect. C.3.5
SC EED Analysis	6 mo Before Launch	See Sect. C.3.5
Procedures Used on CCAS	5 mo Before Launch	See Sect. C.3.6.2
Procedures Used at Astrotech	2 mo Before SC Arrival at Astrotech	See Sect. C.3.6.2
Final Target Specification	90 day Before Launch	Date Depends on Mission Design; See Sect. C.3.4
SC Envir Qual Test Reports	As Available	See Table C.2-1 for Envir Qual Rqmts

reading the files back onto the originating CAD system from tape before transmittal to Lockheed Martin.

File transfer protocol (FTP) method of data transfer is also feasible. An account can be established on a Lockheed Martin "firewall" server for this purpose. Once the account is set-up and a password is provided for access, up to 600 MB of data can be transmitted at one time. An alternative would involve the contractor providing similar access to one of their systems via a temporary account. In either case, the transfer type should be set to "binary."

The following information must be sent with the CAD data file regardless of the transfer method:

1) Name and phone number of the computer system administrator/operator;
2) Name and phone number of the contact person who is familiar with the model if problems or questions arise;
3) Spacecraft axis and coordinate system;
4) Spacecraft access requirements for structure that is not defined on CAD model (i.e., fill and drain valve locations);
5) Multiview plot of the model;
6) Size of the file (MB);
7) uudecode (UNIX-based) information if applicable.

C.3.2 Coupled Loads Analysis Model Requirements

The customer-supplied dynamic mathematical model of the spacecraft should consist of generalized mass and stiffness matrices and a recommended modal damping schedule. The desired format is Craig-Bampton, constrained at the Centaur interface in terms of spacecraft modal coordinates and discrete Centaur interface points. The dynamics model should have an upper frequency cutoff of 50 to 60 Hz. The output transformation matrices (OTM) should be in the form that, when multiplied by the spacecraft modal and interface generalized coordinate responses, will recover the desired accelerations, displacements, or internal loads. One of the OTMs should contain data that will allow calculation of loss of clearance between the payload fairing (PLF) and critical points on the spacecraft. Typically, the size of the OTMs is 200 to 500 rows for accelerations, 50 to 200 rows for displacements, and 300 to 1,000 rows for internal loads.

C.3.3 Spacecraft Thermal Analysis Input Requirements

Spacecraft geometric and thermal mathematical models are required to perform the integrated thermal analysis. These models should be delivered on a Macintosh computer diskette with printed listings of all the files. The geometric mathematical model (GMM) and thermal mathematical model (TMM) size should be less than 400 nodes each.

The preferred GMM format is Vector Sweep input format. Alternate formats are Thermal Radiation Analysis System (TRASYS), ESABASE, or NEVADA input formats. The documentation of the GMM should include illustrations of all surfaces at both the vehicle and component levels, descriptions of the surface properties, and the correspondences between GMM and TMM nodes.

The preferred TMM format is Thermal Analyzer input format. The alternative TMM input format is System-Improved Numerical Differencing Analyzer (SINDA). The TMM documentation should include illustrations of all thermal modeling; detailed component power dissipation's for prelaunch, ascent, and onorbit mission phases; steady-state and transient test case boundary conditions, and output to verify proper conversion of the input format to Lockheed Martin analysis codes; maximum and minimum allowable component temperature limits; and internal spacecraft convection and radiation modeling.

In addition to the TMM and GMM, launch window open and close times for the entire year are required inputs to the integrated thermal analysis.

C.3.4 Target Specifications

Target specifications normally include the final mission transfer orbit (apogee and perigee radius, argument of perigee, and inclination), spacecraft weight, and launch windows. The final target specification is due to Lockheed Martin 90 days before launch for missions incorporating minimum residual shutdown (MRS) or in-flight retargeting (IFR), and 60 days before launch for guidance-commanded-shutdown geosynchronous transfer orbit (GTO) missions.

C.3.5 Spacecraft Electromagnetic Interference (EMI) and Electromagnetic Compatibility (EMC) Certification Letter and Electroexplosive Device (EED) Analysis

A final confirmation of spacecraft transmitter and receiver parameters and emission and susceptibility levels of electronic systems is required 6 months before launch. This includes consideration of emissions from such electronic equipment as internal clocks, oscillators, and signal or data generators and susceptibility of electronics and items such as electroexplosive devices (EED) to cause upset, damage, or inadvertent activation. These characteristics are to be considered according to MIL-STD-1541 requirements to assure that appropriate margins are available during launch operations. Lockheed Martin will use the spacecraft data to develop a final analysis and certification for the combined spacecraft/launch vehicle and site environment.

C.3.6 Safety Data

The spacecraft must show compliance with Eastern Range Regulation (ERR) 127-1 to receive Range Safety approval to launch from Cape Canaveral Air Station (CCAS). Compliance with Eastern/Western Range Regulation (EWR) 127-1 is required for payloads to be launched from Vandenberg Air Force Base (VAFB). Spacecraft documentation required for submittal to Range Safety is briefly described below. Details of the spacecraft submittals will be coordinated in safety working group meetings and other channels during the integration process.

C.3.6.1 Missile System Prelaunch Safety Package (MSPSP)—The MSPSP is the data package that describes in detail all hazardous and safety-critical systems or subsystems and their interfaces in the spacecraft and its ground support equipment (GSE). In addition, the MSPSP provides verification of compliance with Chapters 3, 5, and 7 of ERR 127-1. The MSPSP must be approved by Range Safety before the arrival of any spacecraft elements on the launch site.

C.3.6.2 Spacecraft Launch Site Procedures—All hazardous spacecraft procedures must be approved by Range Safety or by the Astrotech Safety Officer, depending on where the procedures will be used, before they are run. Because the approving authority must also concur in the nonhazardous designation of procedures, all spacecraft launch site procedures must be submitted for review. The Lockheed Martin System Safety group is the customer point of contact for launch site safety authorities.

C.3.6.3 Radiation Protection Officer (RPO) Data—Permission must be secured from the Range RPO before a spacecraft is allowed to radiate radio frequency (RF) emissions on pad. The required data include descriptions of the equipment involved, the procedures that will be used, and data forms on the personnel who will be running the procedures.

C.3.6.4 Spacecraft Breakup Data Requirements—The data described in the following three subsections are required to perform analyses that satisfy 45th Space Wing/SEY requirements for submitting a request for Range Safety flight plan approval.

***C.3.6.4.1 Inadvertent Spacecraft Separation and Propulsion Hazard Analysis*—**This set of data is related to inadvertent separation of the spacecraft during early ascent and the potential for launch area hazard that could exist in the event the spacecraft engine(s) fire. Typical spacecraft propulsion system data provided by the customer include the maximum tanked weight, maximum loaded propellant weight, maximum axial thrust (all motors), and maximum resultant specific impulse.

C.3.6.4.2 Intact Impact Analysis**—**This set of data is related to the ground impact of the spacecraft. The intact impact analysis assumes ground impact of a fully loaded, fueled, intact spacecraft and assumes the propellants combine and explode. Typical spacecraft data provided by the customer include the types and weights of explosive propellants; estimates of the number of pieces of the spacecraft that could break off resulting from an explosion; and locations, size, weight, and shape of each piece.

C.3.6.4.3 Destruct Action Analysis**—**This set of data is related to the flight termination system (FTS) destruction of the launch vehicle. The destruct action analysis assumes in-flight destruction of the vehicle by detonation of the Range Safety charge. Typical spacecraft data provided by the customer include an estimate of the number of spacecraft pieces that could break off as a result of commanded vehicle destruction and estimates of their size, weight, shape, and mounting location on the spacecraft.

C.3.7 SPACECRAFT PROPELLANT SLOSH MODELING

Lockheed Martin performs a spacecraft and Centaur propellant slosh analysis as part of the launch vehicle's autopilot design. The spacecraft propellant tank geometry, propellant densities, and maximum and minimum tank fill levels are required to perform this analysis. The data other than the tank fill levels should be available as part of the ICD or MSPSP inputs.