Revision 10a **January 2007**

ATLAS LAUNCH SYSTEM MISSION PLANNER'S GUIDE

APPROVALS

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ATLAS

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FOREWORD

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

Users of this guide are encouraged to contact the offices listed below to discuss the Atlas V launch vehicle family and how the Atlas V family satisfies user requirements.

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This guide is subject to change and will be revised periodically. Revision 10 has extensive updates from the *Atlas Launch System Mission Planner's Guide,* Revision 9. The most significant change is this document addresses only the Atlas V 400 series, Atlas V 500 series and Atlas V HLV configurations. An itemized list of updates has been documented on the revisions page. Revision 10 of this document replaces all previous Atlas Launch System Mission Planner's Guide document revisions.

Change pages to this printed document will not be provided; however, the version on the CLS website will be maintained. The most current version of this document can be found on the Internet at: *http://www.lockheedmartin.com/CLS*.

ATLAS LAUNCH SYSTEM MISSION PLANNER'S GUIDE REVISIONS

Revision Date	Rev No.	Change Description
January 2007	10a	Sections 1 and 5—Updates Include • Changed data to reflect LMCLS and ULA organizations

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1. INTRODUCTION

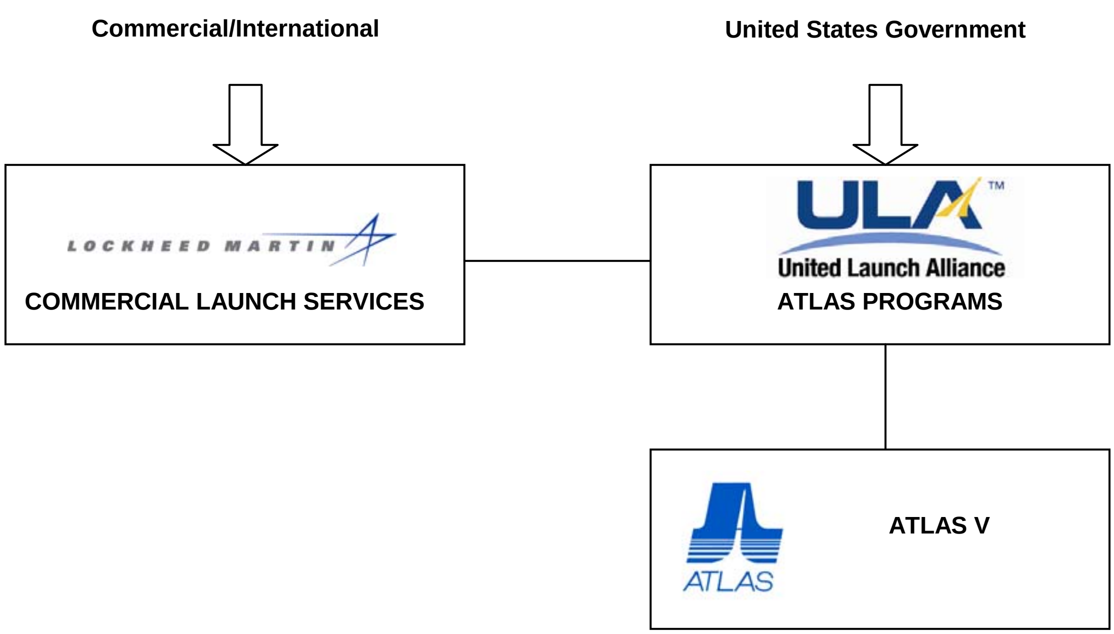
This *Atlas® Launch System Mission Planner's Guide* (AMPG) provides current and potential Atlas V launch services customers with information about the Atlas V launch vehicle family and related spacecraft services. The Atlas™ family of launch vehicles has a long established heritage and a record of unsurpassed reliability and performance. Since the publication of the previous release of this guide in 2001, the Atlas IIA, IIAS, and Atlas III configurations have completed their life cycles and have been superseded by the Atlas V 400 and 500 series. A full range of technical planning data and requirements are included in this guide to allow the user to assess the compatibility of the user's spacecraft with various interfaces that comprise the Atlas V launch vehicle system.

1.1 LAUNCH SERVICES

Atlas V launch services are offered to commercial and international launch services users by contracting with Lockheed Martin Commercial Launch Services (LMCLS), a wholly-owned subsidiary of the Lockheed Martin Corporation, and to United States government customers through United Launch Alliance (ULA) Atlas Programs. Atlas V launch services are offered through a dedicated team of technical, business management, and marketing specialists to provide the optimum and most cost-effective space transportation solutions for the launch services customer.

The structure and relationship between LMCLS and ULA Atlas Programs is shown in Figure 1.1-1.

Figure 1.1-1: The Atlas Team



1.2 LAUNCH SERVICES ORGANIZATION AND FUNCTION

Atlas V launch services addresses the full range of spacecraft integration, processing, encapsulation, launch operations and verification of orbit activities. The typical launch service includes:

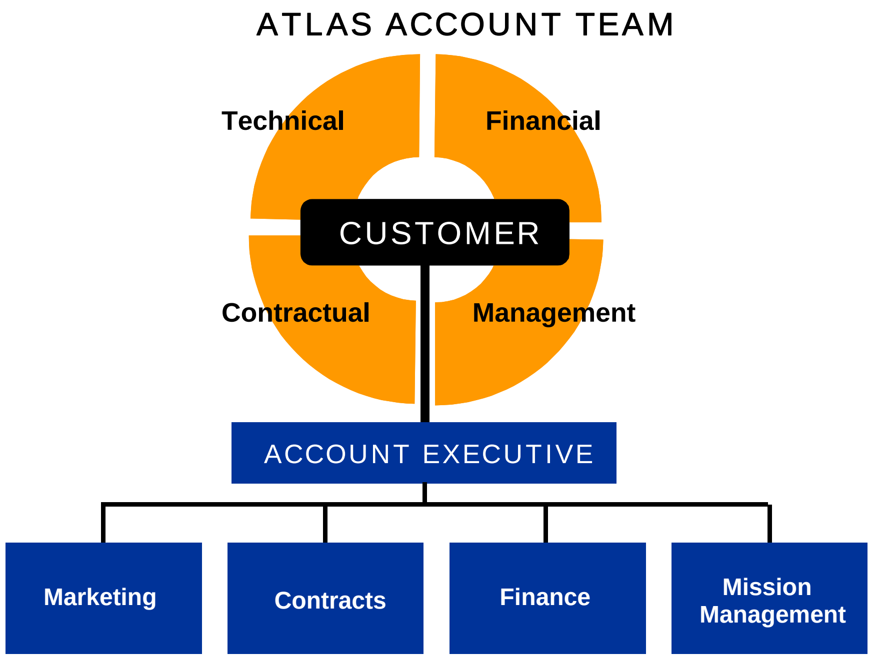
1. Complete launch services using a dedicated Atlas V launch vehicle;
2. Launch operations services;
3. Mission-peculiar hardware and software design, test, and production;
4. Launch vehicle/spacecraft integration and interface design;
5. Mission management;
6. Program management;
7. Launch facilities and support provisions;
8. Payload processing facilities;
9. Spacecraft support at the launch site;
10. Validation of spacecraft separation sequence and orbit;
11. Range Safety interface;
12. Customer launch event support;
13. Mission Success events.

For United States Department of Defense (DoD) government missions, the United Launch Alliance (ULA) Atlas Government Program Office (AGPO) organization is focused to handle unique requirements typically present with the integration and launch of these types of DoD spacecraft. The AGPO Mission Manager, the primary interface with the DoD Atlas customer, coordinates all program and mission management activities required to support a mission and ensures that all technical and contractual issues are addressed in a timely manner. In addition, the ULA Atlas Program’s Mission Integration Program Director assigns a dedicated Integration Manager, who acts as both the mission integrator and vehicle manager.

The Atlas Team manages the Atlas launch manifest for both Cape Canaveral Air Force Station (CCAFS) and Vandenberg Air Force Base (VAFB) in a single, integrated process. In this way, all requirements from all customers are viewed together according to a specific set of rules that cover delays, priorities, and the sequence of launches or firing order.

For international and commercial customers, the Atlas Team uses a focused Account Team management structure, overlaid above the functional organization. Account Teams are organized to concentrate resources and attention to a single customer. Every customer has a designated Account Team whose members are accountable for the entire customer relationship and sales cycle. Four skilled persons comprise each Account Team, one each from the following specialties: sales and marketing, program management/technical operations, contracts and risk management, and finance. Figure 1.2-1 illustrates the Account Team structure.

Figure 1.2-1: Atlas Account Team Structure



The Account Team works with each commercial customer to identify the optimal launch solution that best supports the customer's business or program objectives. Each expert on the team contributes to the overall solution and guides the process from the initial marketing consultations through the launch and post-flight activities. In this way, all specialized areas are brought to bear on the customer's requirements simultaneously—there is no transfer of responsibility or accountability from one business element to another that could disrupt continuity of service to the customer. The Account Executive has primary responsibility to be the customer's voice inside the Atlas Team organization and to be available as the key person to whom the customer can consult on any issue. Account Executives have direct access to the senior management.

The contracts and finance persons provides the necessary skills to assist commercial customers in all contractual, insurance, and financing matters for a launch service. These activities may be supplemented by contributions from legal staff as well. The Atlas Team maintains relationships with the major organizations offering and underwriting launch insurance. A variety of insurance-related services are available.

For commercial missions, the Mission Management Director's primary duties include providing technical advice during mission planning activities, arranging the necessary resources for successful implementation of the launch services mission, and acting as liaison with the major suppliers, subcontractors, and the spacecraft manufacturers. For DoD government missions, the ULA Mission Manager arranges the necessary resources for successful implementation of the launch services mission and acts as liaison with the major suppliers, subcontractors, and the spacecraft manufacturers.

1.3 LAUNCH SYSTEM CAPABILITIES AND INTERFACES

From the user's perspective, the Atlas V launch system comprises a number of hardware and software subsystems and engineering, manufacturing, and operations processes designed to integrate the spacecraft with the launch vehicle correctly and optimally. The following paragraphs summarize the major interface and process components of the Atlas V launch system. Each subject corresponds to an appropriate section of this document where more detailed information on the same subject can be found.

1.3.1 Atlas V Launch System

The Atlas V 400 and 500 series launch vehicles are the latest evolutionary versions of the Atlas launch system, and were placed into service in 2002. Atlas V uses a standard Common Core Booster™ (CCB), up to five strap-on solid rocket boosters (SRB), a Centaur in either the Single-Engine Centaur (SEC) or the Dual-Engine Centaur (DEC) configuration, and one of several Payload Fairings (PLF). Figure 1.3.1-1 illustrates key components of the Atlas V launch vehicles. A three-digit (XYZ) naming convention was developed for the Atlas V 400 and 500 series to identify its multiple configuration possibilities as illustrated in Figure 1.3.1-2. Flight-proven and operational beginning in August 2002, the Atlas V is today meeting a wide variety of commercial and U.S. government launch requirements. The Atlas V family, shown in Figure 1.3.1-3, includes the flight-proven Atlas V 400 and 500 series and the Atlas V Heavy Lift Vehicle, in development.

The Atlas V 400 series incorporates the flight proven 4-m diameter Atlas V 12.0 m (39.3 ft) Large Payload Fairing (LPF), the 12.9 m (42.3 ft) Extended Payload Fairing (EPF), or the 13.8 m (45.3 ft) Extended EPF (XEPF). Figure 1.3.1-4 summarizes characteristics of the Atlas V 400 series. The Atlas V 500 series incorporates the flight proven 5-m diameter 20.7 m (68 ft) short, the 23.5 m (77 ft) medium, or the 26.5 m (87 ft) long payload fairing. Figure 1.3.1-5 summarizes characteristics of Atlas V 500 series. The Atlas V Heavy Lift Vehicle (HLV) configuration is currently under development and incorporates the three 5-m payload fairings. Figure 1.3.1-6 summarizes characteristics of the Atlas V HLV.

1.3.2 Atlas V Launch System Environments

The Atlas V launch system provides spacecraft preflight and flight environments that are typically more benign than those available with other launch systems. All environments specified for the Atlas V launch system (e.g., shock, vibration, acoustic, thermal, electromagnetic) are based on in-depth engineering analyses of existing and evolved hardware. These have been fully validated with both test and flight telemetry data. Flight telemetry data from Atlas V vehicle configurations are continually used to update these environments as required. Verification that the customer's flight environments remained within specified levels is obtained through a combination of standard instrumentation and analysis, or with use of additional mission-unique instrumentation near the spacecraft interface. This hardware enables telemetering of high-frequency measurements near the spacecraft interface. The environment envelopes to which customer spacecraft are exposed are fully discussed in Section 3.0.

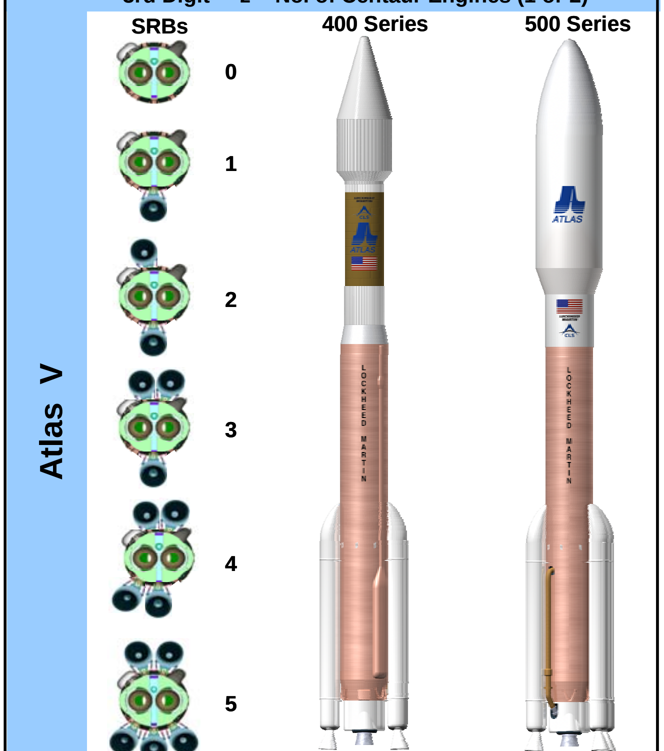
1.3.3 Spacecraft and Ground System Interfaces

The Atlas V launch system offers a broad range of launch vehicle and ground processing hardware and facility options to meet spacecraft interface requirements. Primary interfaces between the Atlas V launch vehicle and spacecraft consist of a payload adapter and a PLF. The payload adapter supports the spacecraft on top of the launch vehicle and provides mounting for the payload

Figure 1.3.1-2: Atlas V Naming Designator Definition

Vehicle Naming Convention: Atlas V xyz

1st Digit = x = PLF Diameter (meters, approx)
2nd Digit = y = No. of SRBs (0 to 5)
3rd Digit = z = No. of Centaur Engines (1 or 2)



		400 Series	500 Series
5	PLF Diameter	4 meter	5 meter
2	Number of SRBs	0 thru 3	0 thru 5
1	Number of Centaur Engines	1 or 2	1 or 2

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Figure 1.3.1-3: Atlas V Launch Vehicle Family

Atlas V 400 Series				Atlas V 500 Series						HLV
401	**411**	**421**	**431**	**501**	**511**	**521**	**531**	**541**	**551**	**–**
Performance to GTO (1,804 m/s), kg (lb)										
4,950	6,075	7,000	7,800	3,970	5,370	6,485	7,425	8,240	8,700	13,000
(10,913)	(13,393)	(15,432)	(17,196)	(8,752)	(11,839)	(14,297)	(16,369)	(18,166)	(19,180)	(28,660)
Performance to GTO (1,500 m/s), kg (lb)										
3,765	4,535	5,255	5,885	3,000	4,040	4,930	5,645	6,280	6,695	
(8,300)	(9,998)	(11,585)	(12,974)	(6,614)	(8,907)	(10,869)	(12,445)	(13,845)	(14,760)	

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separation system, spacecraft electrical interfaces to the launch vehicle, mission-unique spacecraft purge system connections, and mission unique instrumentation. The Atlas V program has a set of standard flight-proven payload adapters to meet identified spacecraft interface requirements. Specialized payload adapters can be developed to meet mission-specific requirements of certain customer spacecraft. In addition, the user has the option to provide the payload adapter and separation system. Atlas V payload adapter systems are described in Section 4.1.2 and Appendix E.

The PLF encloses and protects the spacecraft during ground operations and launch vehicle ascent. The PLF also incorporates hardware to control thermal, acoustic, electromagnetic, and cleanliness environments for the spacecraft and may be tailored to provide access and radio frequency communications to the encapsulated spacecraft. The Atlas V program offers three 4-m diameter payload fairings configurations: the LPF, the EPF, and the XEPF compatible with the Atlas V 400 series vehicles. A 5-m diameter PLF is available for Atlas V 500 vehicles in a short, medium, or long length configuration. Atlas V PLF systems are described fully in Section 4.1.1 and Appendix D.

Atlas V launch services also provide facilities, hardware, and services necessary to support the spacecraft during ground operations and processing. Standing agreements at both CCAFS and VAFB ensure the proper reception, handling, processing, integration, and storage of commercial and governmental spacecraft. These items are discussed in Section 4.2.

Figure 1.3.1-1: Atlas V Launch Vehicle Common System Elements

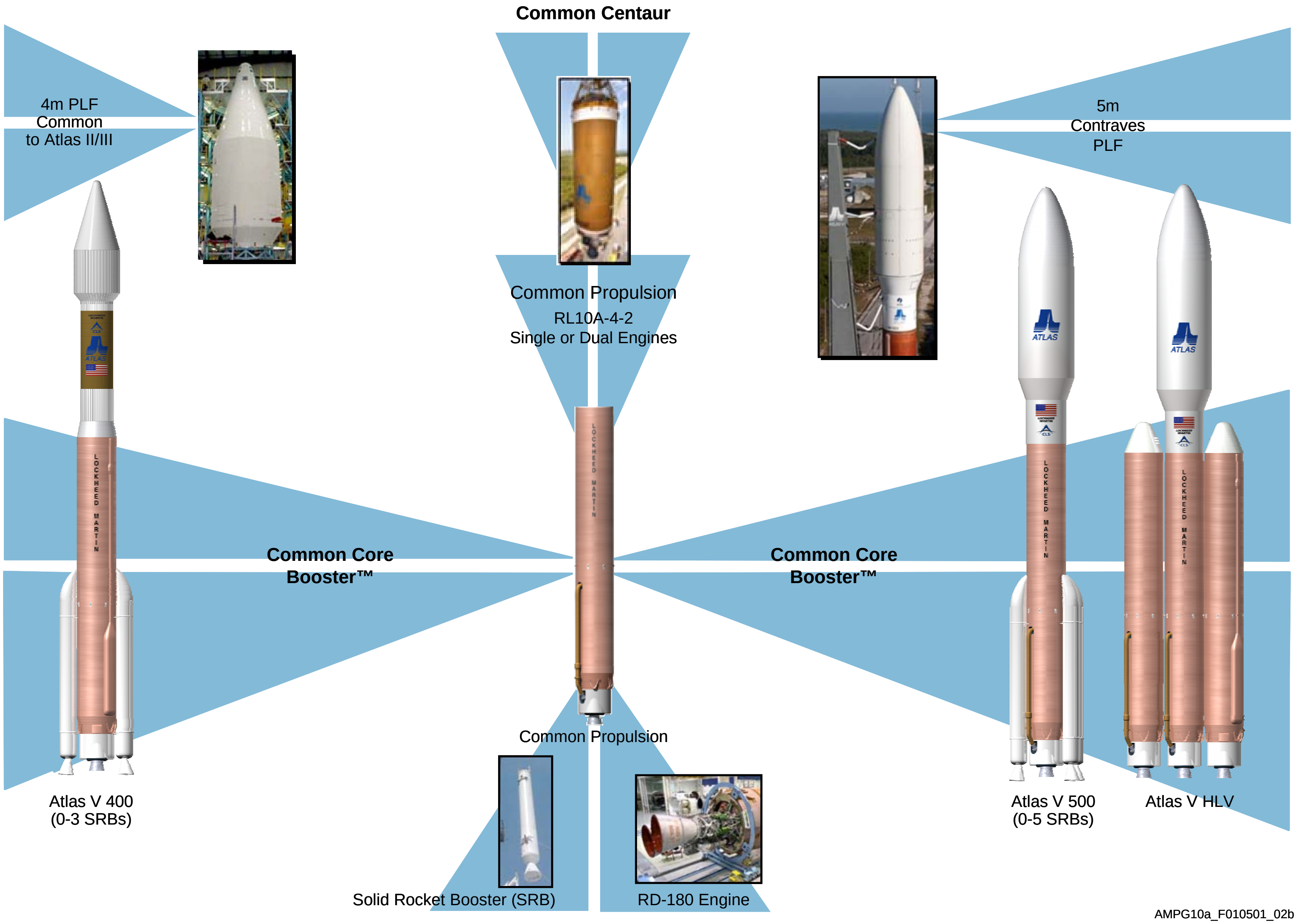
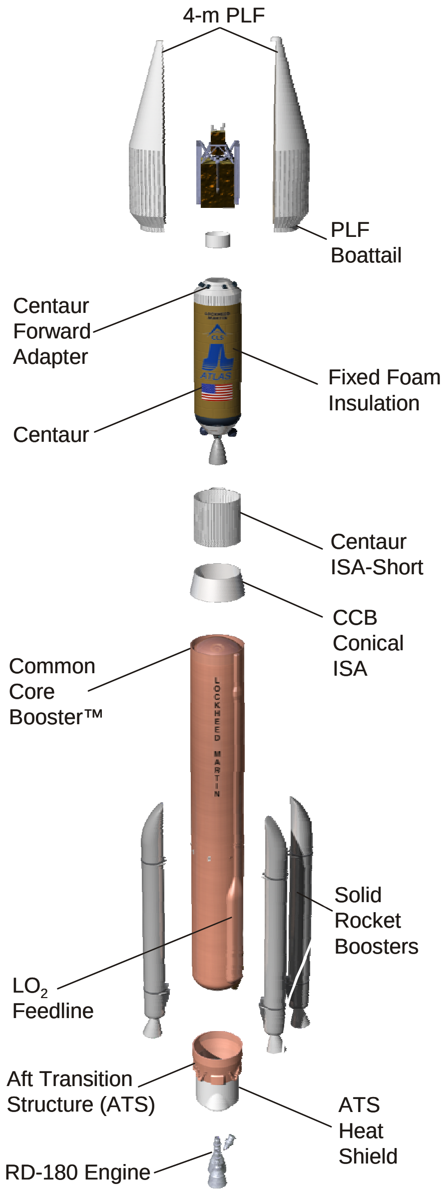


Figure 1.3.1-4: Atlas V 400 Series Launch System



PAYLOAD FAIRING (PLF)

Features	LPF	EPF	XEPF
Diameter:	4.2 m	4.2 m	4.2 m
Length:	12.0 m	12.9 m	13.8 m
Mass:	2,111 kg	2,289 kg	2,503 kg

Subsystems

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

COMMON CENTAUR

Features

Size:	3.05-m Dia x 12.68-m Length with Extended Nozzle
Inert Mass:	2,086 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 or Two Pratt & Whitney Restartable Engine(s)
— Model:	RL10A-4-2
— Thrust:	99.2 kN (SEC) 198.4 kN (DEC)
— I_{SP}:	450.5 s
(SEC)	One Electromechanically Actuated 51-cm Columbium Fixed Nozzle Four 27-N Hydrazine Thrusters Eight 40-N Lateral Hydrazine Thrusters
(DEC)	Two Hydraulically Actuated 51-cm Columbium Extendible Nozzles Eight 40-N Hydrazine Thrusters Four 27-N 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
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

SOLID ROCKET BOOSTER (SRB)

Zero-to-Three	Ground-Lit
Size:	155-cm Dia x 19.5-m Length
Mass:	46,559 kg (Each Fueled)
Thrust:	1,361 kN (Each)
I_{SP}:	275 s
Nozzle Cant:	3 deg

CENTAUR INTERSTAGE ADAPTER (C-ISA SHORT)

Features

Size:	3.05-m Dia x 3.13-m Length
Mass:	342 kg

Subsystems

Structure:	Aluminum Lithium Skin Stringer & Frame
Separation:	Low-Cost Atlas Separation System

CCB CONICAL INTERSTAGE ADAPTER

Features

Size:	3.05-m Dia (Top) x 1.65-m Length 3.81-m Dia (Bottom)
Mass:	418 kg
Structure:	Composite (Graphite Epoxy) with Aluminum Ring Frames Fwd & Aft

COMMON CORE BOOSTER™ (CCB)

Features

Size:	3.81-m Dia x 32.48-m Length
Propellant:	284,089-kg LO_2 & RP-1
Inert Mass:	21,277 kg
Guidance:	From Upper Stage

Subsystems

Structure:	Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure; Composite Heat Shield
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 for Tank Pressurization, Computer-Controlled Pressurization System
Hydraulics:	Fluid—Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Redundant Rate Gyros, Electrical Power

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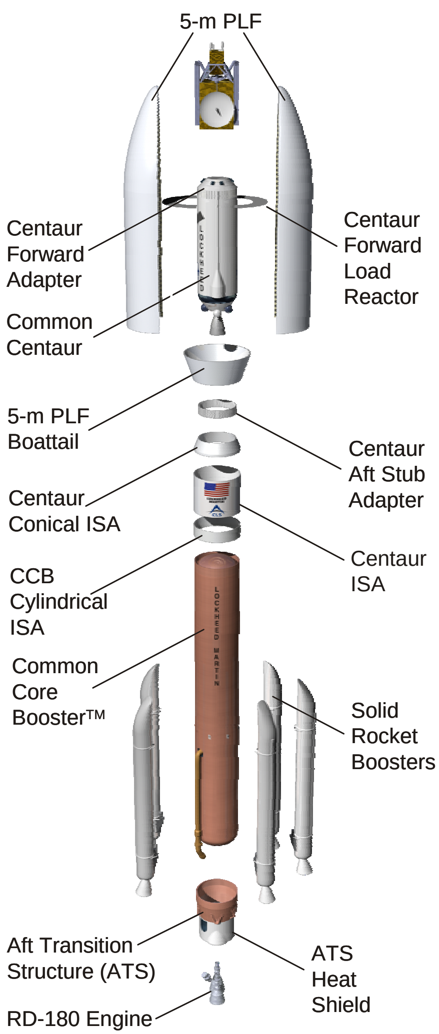
Figure 1.3.1-5: Atlas V 500 Series Launch System



PAYLOAD FAIRING (PLF)

Features	**5-m Short**	**5-m Medium**	**5-m Long**
Diameter:	5.4 m	5.4 m	5.4 m
Length:	20.7 m	23.4 m	26.5 m
Mass:	3,540 kg	4,019 kg	4,394 kg

Subsystems
- Fairing: Bisector; Sandwich Construction with Graphite Epoxy Face Sheets & an Aluminum Honeycomb Core
- Boattail: Fixed, Composite Sandwich Const
- Separation: Vertical Separation by a Linear Piston & Cylinder Activated by a Pyrotechnic Cord; Horizontal Separation by an Expanding Tube Shearing a Notched Frame, Activated by a Pyrotechnic Cord

COMMON CENTAUR

Features **All Common with Atlas 400 Series**
- Size: 3.05-m Dia x 12.68-m Length with Extended Nozzle
- Inert Mass: 2,138 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 or Two Pratt & Whitney Restartable Engine(s)
  - — Model: RL10A-4-2
  - — Thrust: 99.2 kN (SEC) 198.4 kN (DEC)
  - — I_{SP}: 450.5 s
  - (SEC) One Electromechanically Actuated 51-cm Columbium Fixed Nozzle; Four 27-N Hydrazine Thrusters; Eight 40-N Lateral Hydrazine Thrusters
  - (DEC) Two Hydraulically Actuated 51-cm Columbium Extendible Nozzles; Eight 40-N Lateral Hydrazine Thrusters; Four 27-N Hydrazine Thrusters
- Pneumatics: Common with Atlas V 400 Series
- Avionics: Common with Atlas V 400 Series
- Insulation: Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding with Optional Radiation Shields

SOLID ROCKET BOOSTERS (SRB)

Zero-to-Five Ground-Lit
- Size: 155-cm Dia x 19.5-m Length
- Mass: 46,559 kg (Each Fueled)
- Thrust: 1,361 kN (Each)
- I_{SP}: 275 s
- Nozzle Cant: 3 deg

CENTAUR INTERSTAGE ADAPTER (C-ISA LARGE)

Features
- Size: 3.81-m Dia x 4.46-m Length
- Mass: 2,292 kg (Includes ISA, Aft Stub Adapter and Boattail)

Subsystems
- Structure: Composite Sandwich (Aluminum Core/Graphite Epoxy Face Sheets)

CCB CYLINDRICAL INTERSTAGE ADAPTER

Features
- Size: 3.81-m Dia x 0.32-m Length
- Mass: 282 kg

Subsystems
- Structure: Aluminum Machined Rolled-Ring Forging

COMMON CORE BOOSTER™ (CCB)

Features **Common with Atlas V 400 Series**
- Size: 3.81-m Dia x 32.46-m Length
- Propellant: 284,089-kg LO_2 & RP-1
- Inert Mass: 21,336 kg for 55Z Configuration
- Guidance: From Upper Stage

Subsystems
- Structure: Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure; Composite Heat Shield
- 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} = 338.4 s
- Pneumatics: Helium for Tank Pressurization, Computer-Controlled Pressurization System
- Hydraulics: Integral with Engine Provides Gimbal Control
- Avionics: Flight Control, Flight Termination, Telemetry, Redundant Rate Gyros, Electrical Power

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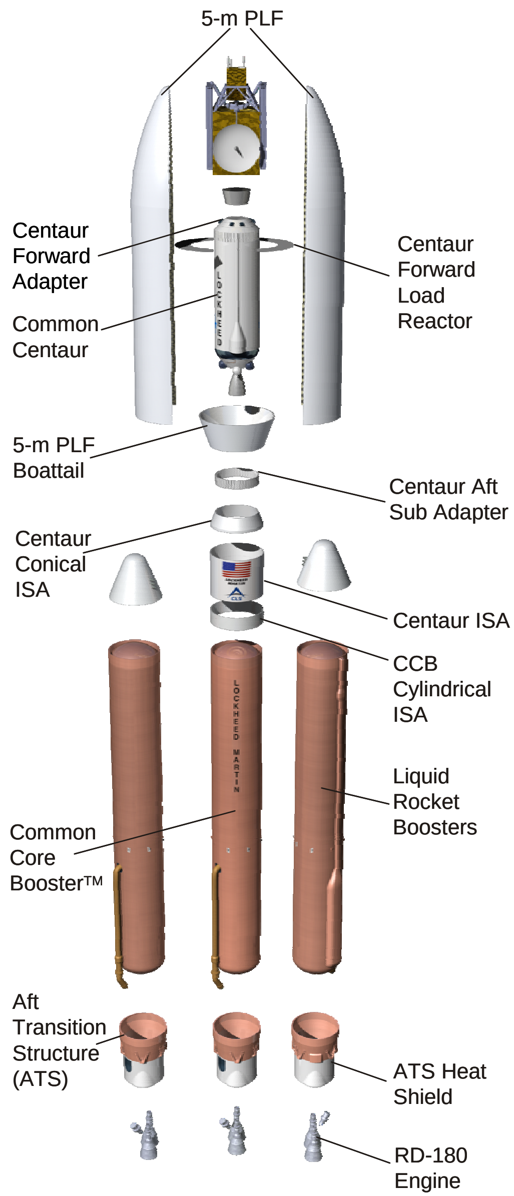
Figure 1.3.1-6: Atlas V HLV Launch System



PAYLOAD FAIRING

Features	**5-m Long**
Diameter:	5.4 m
Length:	26.5 m
Mass:	4,394 kg
Subsystems	
Fairing:	Bisector; Sandwich Construction with Graphite Epoxy Face Sheets & an Aluminum Honeycomb Core
Boattail:	Fixed, Composite Sandwich Construction
Separation:	Vertical Separation by a Linear Piston & Cylinder Activated by a Pyrotechnic Cord; Horizontal Separation by an Expanding Tube Shearing a Notched Frame, Activated by a Pyrotechnic Cord

COMMON CENTAUR

Features	
Size:	3.05-m dia x 12.68-m Length with Extended Nozzle
Inert Mass:	2,316 kg
Propellant:	20,860-kg LH_2 & LO_2
Guidance:	Inertial
Subsystems	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	One or Two Pratt & Whitney Restartable Engine(s)
— Model:	RL 10A-4-2
— Thrust:	99.2 kN (SEC) 198.4 kN (DEC)
— I_{SP}:	450.5 s
(SEC)	One Electromechanically Actuated 51-cm Columbium Fixed Nozzle Four 27-N Hydrazine Thrusters Eight 40.5-N Lateral Hydrazine Thrusters
(DEC)	Two Hydraulically Actuated 51-cm Columbium Extendible Nozzles Eight 40-N Lateral Hydrazine Thrusters Four 27-N Hydrazine Thrusters
Pneumatics:	Common with Atlas V 400/500 Series
Avionics:	Common with Atlas V 400/500 Series
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding with Optional Radiation Shields

LIQUID ROCKET BOOSTERS (LRB)

Features	
Size:	3.81-m dia x 35.99-m Length
Propellant:	286,795-kg LO_2 & RP-1 (Each)
Inert Mass:	21,902 kg
Thrust:	4,152-kN VAC 100% Thrust (Each)
I_{SP}:	338.4 s
Structure:	Same as Common Core Booster

CENTAUR INTERSTAGE ADAPTER (C-ISA LARGE)

Features	
Size:	3.81-m dia x 4.46-m Length
Mass:	2,292 kg (Includes ISA, Aft Stub Adapter and Boattail)
Subsystems	
Structure:	Composite Sandwich (Aluminum Core/Graphite Epoxy Face Sheets)

CCB CYLINDRICAL INTERSTAGE ADAPTER

Features	
Size:	3.81-m dia x 0.60-m Length
Mass:	282 kg
Subsystems	
Structure:	Aluminum Machined Rolled-Ring Forging

COMMON CORE BOOSTER™ (CCB)

Features	
Size:	3.81-m dia x 32.46-m Length
Propellant:	286,795-kg LO_2 & RP-1
Inert Mass:	23,648 kg
Guidance:	From Upper Stage
Subsystems	
Structure:	Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure; Composite Heat Shield
Separation:	8 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers) SL 100% Thrust = 3,827 kN, I_{SP} = 311.9 s Vac 100% Thrust = 4,152 kN, , I_{SP} = 338.4 s
Pneumatics:	Helium (Pressure Tanks), Computer-Controlled Pressurization System
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Redundant Rate Gyros, Electrical Power

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1.3.4 Atlas V Mission Integration and Management

The Atlas V mission integration and management process is designed to use the engineering and production talents of the Atlas Team effectively. Spacecraft contractor organizations are also engaged to integrate the customer's spacecraft with the Atlas V launch vehicle. Section 5.0 is an overview of the mission integration process and launch services management functions currently in operation for commercial and government missions. For typical communications spacecraft missions, a 12-month integration schedule is discussed. The management approach and a summary of integration analysis tasks are provided to enable the customer to understand the process fully and participate where appropriate.

1.3.5 Spacecraft and Launch Site Facilities

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

1.3.6 Atlas V Launch Operations

Atlas V launch operations processes require involvement of the launch services customer and spacecraft contractor. Section 7.0 provides an overview of our operations processes, discussing issues that an Atlas V customer may wish to consider early in the mission integration process.

1.3.7 Atlas V Enhancements

Section 8.0 is designed to provide insight to the Atlas V customer community of the Atlas V program plans for enhancing the Atlas V launch vehicle to meet launch services requirements of the 21st Century. Enhancements include:

1. Dual Spacecraft System;
2. Dual Payload Carrier;
3. Heavy Lift Payload Truss.

1.3.8 Supplemental Information

Five appendices are provided in this document to address various items in more detail:

1. Appendix A discusses the history, heritage, and evolution of the Atlas booster and Atlas/Centaur launch vehicles. A more detailed description of Atlas booster and Centaur stages and subsystems is provided. The reliability growth method for evaluating mission and vehicle reliability also is summarized.
2. Appendix B details our mission success philosophy and quality assurance process at Atlas facilities and at those of major subcontractors and suppliers.
3. Appendix C defines spacecraft technical data requirements to support the mission integration process. In addition, a discussion of the type and format of technical data required by the Atlas Program is listed to provide insight into the exchange of information between the spacecraft contractor, launch services customer, and the Atlas Program that occurs during a typical integration.
4. Appendix D describes the Atlas V payload fairings.
5. Appendix E describes the Atlas V payload adapters.

1.4 ADVANTAGES OF SELECTING ATLAS V

All commercial agreements required to conduct Atlas V launch services are maintained on behalf of all customers. Agreements are in place covering spacecraft and launch vehicle processing facilities, services, and launch site support at CCAFS in Florida. Similar agreements are nearing completion for comparable services at VAFB in California.

An Atlas V launch service provides the following key advantages to the customer:

1. Flight-proven, reliable Atlas V launch vehicles and ground system support facilities;

2. Benign and fully validated launch environments for a customer’s spacecraft (e.g., shock, vibration, acoustic, thermal) that are typically equal to or better than those of other launch vehicles;

3. Atlas V Launch facilities at CCAFS (LC-41) and VAFB (SLC-3E) to accommodate virtually any type of orbital requirements, including geostationary transfer orbits (GTO), low Earth orbits (LEO), low or high inclination orbits, heavy lift missions, and interplanetary missions.

4. A streamlined Atlas V launch processing approach and steady launch tempo to maintain schedules and -commitments;

5. An experienced organization that has launched almost 600 Atlas missions in nearly 40 years of experience;

6. A flexible and demonstrated mission design capability to maximize spacecraft on-orbit maneuver lifetime (OML) or other parameters through optimized pairing of spacecraft and the Atlas launch vehicle’s customized performance capabilities;

1.5 CONCLUSION

The Atlas team is eager to assist customers in defining and developing future Atlas V missions. New launch services customers may refer to the Foreword of this guide for information regarding the appropriate representative to contact for their mission requirements. Contact information for additional information requests can also be found in the Foreword.

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2. ATLAS V MISSION DESIGN AND PERFORMANCE

Over the past four decades, Atlas boosters and Centaur have flown together as the Atlas/Centaur launch vehicle and with other stages (e.g., Atlas/Agena and Titan/Centaur) to deliver commercial, military, and scientific spacecraft to their target orbits. Based on the Lockheed Martin experience with more than 550 Atlas launches, performance for each launch vehicle is determined by engineering analysis of existing and new hardware, emphasizing conservative performance prediction to ensure each vehicle meets design expectations. Because the Atlas V 400 and 500 series configurations are flight-proven, the performance capabilities reflect flight-qualified hardware performance characteristics. Lockheed Martin has significantly increased the performance levels available to the Atlas V launch services customer as documented in this revision of the Atlas Mission Planner's Guide. This section further describes the Atlas V family mission and performance options available with East and West Coast launches.

Atlas V launch vehicles can meet performance requirements by customizing (or standardizing) mission and trajectory designs to meet specific spacecraft mission requirements. Lockheed Martin offers performance capability levels (as opposed to explicit hardware configurations) as part of a standard launch service package to meet evolving commercial, civil government, and military spacecraft mission launch requirements. The Atlas V 400 and 500 series launch vehicles can be launched from either Cape Canaveral Air Force Station (CCAFS) in Florida or Vandenberg Air Force Base (VAFB) in California. The Atlas V Heavy Lift Vehicle (HLV) offers launch services from CCAFS LC-41.

Vehicle performance can be used in several ways. For spacecraft missions whose primary mission may require less performance than a specific Atlas V 400 or 500 series configuration, additional mission constraints can be specified that will use excess performance to benefit the launch services customer. For example, ascent trajectory designs can be shaped to improve ground coverage of major events or reduce the energy required by the spacecraft to reach its final orbit. The ascent profile can be standardized to reduce mission integration analyses and/or schedules. These options allow a more cost effective solution for cases where maximum vehicle performance is not required.

Performance capabilities quoted throughout this document are presented in terms of Payload Systems Weight. Payload Systems Weight is defined as the separated spacecraft, the spacecraft-to-launch vehicle adapter, and other mission-peculiar hardware required on the launch vehicle to support the spacecraft (e.g., spacecraft flight termination system, harnessing). See Section 2.5.1 for more details on Payload Systems Weight.

2.1 MISSION DESCRIPTIONS

Atlas V is a reliable and versatile launch system capable of delivering spacecraft to a wide range of elliptical orbits, low circular orbits, high circular orbits, and Earth-escape trajectories. All Atlas V launch vehicles are capable of launching a single large spacecraft or multiple spacecraft populating low and medium earth orbit constellations, or a mix of primary and secondary spacecraft. The Centaur is qualified for several engine restarts to facilitate placement of rideshare spacecraft into different orbits or attaining low inclination high altitude orbits such as Geosynchronous Orbit (GSO) or near-GSO. The trajectory design for each mission can be specifically tailored to optimize the mission's critical performance parameter (e.g., maximum spacecraft orbit lifetime, maximum weight to orbit) while satisfying spacecraft and launch vehicle constraints.

Atlas V mission ascent profiles are developed using one or more Centaur main engine burns. Each mission profile type is suited for a particular type of mission. The following paragraphs describe three ascent techniques with mission applications.

2.1.1 Direct Ascent Mission (1 Centaur Burn)

For this mission design, the Centaur main engine(s) are ignited just after Atlas/Centaur separation. The main engine burn continues until the Centaur and spacecraft are placed into the targeted orbit. Centaur/spacecraft separation occurs shortly after the Centaur burn is completed. Direct ascents are primarily used for Low-Earth circular orbits (LEO), Earth Escape, and for elliptic orbits with orbit geometries (i.e., arguments of perigee and inclinations) easily reached from the launch site. For most LEO missions, performance with two Centaur engines is higher than with a single engine. 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 more than 15 missions using the direct ascent design.

2.1.2 Parking Orbit Ascent Mission (2 Centaur Burns)

The parking orbit ascent, used primarily for geosynchronous transfer missions, is the most widely used Atlas V trajectory design. Performance capabilities are based on two Centaur main engine burns injecting the Centaur and the spacecraft into an orbit not achievable by direct ascent. The first Centaur main engine burn starts just after Centaur separation from the booster and is used to inject the Centaur/spacecraft into a mission performance optimal parking orbit. After a coast to the optimal location, the second Centaur main engine burn places the spacecraft into the desired orbit. If targeted to a transfer orbit, the spacecraft must then use its own propulsion system to achieve the final mission orbit. For non-transfer orbit missions where the Centaur delivers the spacecraft to the final mission orbit, the second Centaur main engine burn is used to inject the spacecraft into the desired altitude and orbit inclination. The parking orbit ascent profile is used for Geosynchronous Transfer orbits (GTO), high-altitude circular and elliptical orbits, and Earth escape trajectories requiring a coast phase to meet target conditions. More than 87 Atlas/Centaur missions have flown using the parking orbit ascent mission profile.

2.1.3 Park to Transfer Ascent Mission (3 Centaur Burns)

The GSO mission is the most common three-burn Centaur mission profile. The three-burn mission design is usually characterized by both very high altitude and substantially reduced inclination at spacecraft separation. A detailed profile is outlined in Section 2.4.5. Another typical three-burn scenario is a direct ascent to a low-Earth orbit followed by transfer and insertion into a higher circular orbit. This usually involves two (or more) spacecraft on the launch vehicle. Centaur has flown more than 14 3-burn GSO missions.

2.2 LAUNCH VEHICLE ASCENT DESCRIPTIONS

2.2.1 Booster Phase

The Common Core Booster (CCB) booster phase begins with the ignition of the RD-180 engine system followed by Solid Rocket Booster (SRB) ignition, if applicable, through booster engine throttle down and booster shutdown.

2.2.1.1 Atlas V 400 Series

The Atlas V 400 series consists of a CCB combined with zero to three SRBs and a Centaur. The booster phase begins with ignition of the RD-180 engine system. The launch vehicle is held down during booster engine start and a portion of booster throttle up. A vehicle health check is performed before achieving full throttle. After passing the health check, the vehicle is released, SRB ignitions occur (if applicable), and booster throttle up is completed. After a short vertical rise away from the pad, the 400 series launch vehicle begins its pitch-over phase, a coordinated maneuver to the prescribed ascent profile and direction. At a predetermined altitude, the Atlas V launch vehicle transitions to a nominal zero-alpha and zero-beta angle-of-attack phase to minimize aerodynamic loads and engine angles. Both of these phases are implemented through the launch-day wind-steering system, which enhances launch availability by reducing wind-induced flight loads and engine deflections.

For Atlas V 401 or 402 configurations, after reaching 24,380 m (80,000 ft) until 33,530 m (110,000 ft), an alpha-bias angle-of-attack steering technique is used to improve performance while maintaining aerodynamic loading within acceptable limits. The booster phase steering profile through the end of alpha-biased steering is implemented through our launch-day wind-steering system, which enhances launch availability by reducing wind-induced flight loads and engine deflections.

Closed-loop guidance steering is enabled at the end of alpha-biased steering. For all Atlas V 400 series launch vehicles with SRBs, the zero-alpha/zero-beta attitude is maintained until 54,860 m (180,000 ft) when closed-loop guidance steering is enabled. The SRB jettison sequence is initiated after SRB burnout. SRBs 1 and 2, if applicable, are jettisoned at a predetermined time dependent upon a dynamic pressure constraint. SRB 3 is jettisoned 1.5 seconds later, if applicable.

Near the end of the booster phase, the RD-180 engine is continuously throttled so that axial acceleration levels are not exceeded. These g-levels may be a function of payload weight and do not exceed 5.0 g steady state.

The RD-180 cutoff sequence is initiated when a propellant low-level sensor system indicates that the CCB is about to deplete available propellants. At this time, the booster engine is shut down. All Atlas V 400 series configurations retain the Payload Fairing (PLF) through the booster phase of flight.

2.2.1.2 Atlas V 500 Series

The Atlas V 500 series consists of a CCB combined with zero to five SRBs and a Centaur. The RD-180 and SRB ignition sequence is the same as the Atlas V 400 series vehicles. After a short vertical rise away from the pad, the 500 series vehicle begins its pitch-over phase, a coordinated maneuver to the prescribed ascent profile and direction. At a predetermined altitude, the Atlas/Centaur launch vehicle transitions to a nominal zero-alpha and zero-beta angle-of-attack phase to minimize aerodynamic loads and engine angles. Both of these phases are implemented through the launch-day wind-steering system, which enhances launch availability by reducing wind-induced flight loads and engine deflections.

For Atlas V 501 or 502 configurations, after reaching 24,380 m (80,000 ft) until approximately 33,530 m (110,000 ft), an alpha-bias angle-of-attack steering technique is used to improve performance while maintaining aerodynamic loading within acceptable limits. The booster phase steering profile through the end of alpha-biased steering is implemented through our launch-day wind-steering system, which enhances launch availability by reducing wind-induced flight loads and engine deflections.

Closed-loop guidance steering is enabled at the end of alpha-biased steering. For all Atlas V 500 series vehicles with SRBs, the zero-alpha/zero-beta attitude is maintained until 54,860 m (180,000 ft) when closed-loop guidance steering is enabled. The SRB jettison sequence is initiated after SRB burnout. SRBs 1 and 2, if applicable, are jettisoned at a predetermined time dependent upon the dynamic pressure constraint. SRBs 3, 4, and 5 are jettisoned 1.5 seconds later, if applicable.

For Atlas V 500 series missions, the PLF is jettisoned during the booster phase of flight. Before PLF jettison, the RD-180 engine is throttled down to maintain 2.5 g acceleration. Typically, the PLF is jettisoned when the 3-sigma free molecular heat flux falls below 1,135 W/m^2 (360 Btu/ft^2-hr). For sensitive spacecraft, PLF jettison can be delayed to reduce the heat flux with minor performance loss. After PLF jettison, the RD-180 is throttled up.

Near the end of the booster phase, the RD-180 engine is continuously throttled so that specific axial acceleration levels are not exceeded. These g-levels may be a function of payload weight and typically do not exceed 4.6 g steady state.

The RD-180 cutoff sequence is initiated when a propellant low-level sensor system indicates that the CCB is about to deplete available propellants. At this time, the booster engine is shut down.

2.2.1.3 Atlas V Heavy Lift Vehicle

The Atlas V HLV maximizes commonality between Atlas V launch vehicles by using a single CCB for the core flanked by two additional CCBs or Liquid Rocket Boosters (LRB) to make the Atlas V HLV configuration. A single RD-180 engine powers each CCB, the same engine that is used on the Atlas V 400 and 500 series launch vehicle, and commonality continues with the use of the Centaur. The HLV uses the same interstage adapters, avionics suite, and 5-m payload fairings that are common with the 500 series, which provides the remaining building hardware items that comprise the Atlas V HLV configuration.

All three RD-180 engines start simultaneously and reach full thrust shortly after liftoff. After a short vertical rise away from the pad, the HLV begins a coordinated maneuver to the prescribed ascent profile and direction. The HLV transitions to a nominal zero-alpha and zero-beta angle-of-attack phase to minimize aerodynamic loads and engine angles. Both of these phases are implemented through the launch-day wind-steering system, which enhances launch availability by reducing wind-induced flight loads and engine deflections. Shortly into this phase, the core CCB's RD-180 is throttled back to optimize performance.

After reaching 24,380 m (80,000 ft), closed-loop guidance steering is enabled. The LRB cutoff sequence is initiated when a propellant low-level sensing system indicates that the first of the two LRBs is about to deplete available propellants. At this time, both LRB RD-180 engines are shut down, and the LRBs are jettisoned.

The vehicle is now in the core solo phase. The PLF is jettisoned typically when the 3-sigma free molecular heat flux falls below 1,135 W/m^2 (360 Btu/ft^2-hr) during this phase. For sensitive spacecraft, PLF jettison can be delayed to reduce the heat flux with minor performance loss. After PLF jettison, the RD-180 is throttled up.

Near the end of the core solo phase, the RD-180 engine is continuously throttled so that specific axial acceleration levels are not exceeded. These g-levels may be a function of payload weight and do not exceed 5.0 g steady state.

The core RD-180 cutoff sequence is initiated when a propellant low-level sensor system indicates that the CCB is about to deplete available propellants. At this time, the engine is shut down.

2.2.2 Centaur Phase

The Atlas V Centaur can use either a Single Engine Configuration (SEC) or a Dual Engine Configuration (DEC). The Centaur first burn Main Engine Start (MES1) occurs approximately 10 seconds after the CCB is jettisoned. For typical Atlas V 400 series missions, the PLF is jettisoned 8 seconds after MES1, by which time the 3-sigma free molecular heat flux has typically fallen below 1,135 W/m^2 (360 Btu/ft^2-hr). For sensitive spacecraft, PLF jettison can be delayed later into the flight with minor performance loss.

For direct ascent missions, a single, long duration main engine burn injects the Centaur/spacecraft into the targeted orbit after which Centaur performs a series of pre-separation maneuvers. For parking orbit ascent missions, the Centaur first main engine burn (typically the longer of the two) injects the spacecraft into an elliptical, performance-optimized parking orbit. After first burn Main Engine Cutoff (MECO1), the Centaur and spacecraft enter a coast period, about 10 minutes for an SEC geosynchronous transfer mission, during which the Centaur slowly aligns itself to the attitude required for the Centaur second burn Main Engine Start (MES2). Should a spacecraft require attitude maneuvers during longer coast periods, the Centaur can accommodate roll axis alignment requirements and provide commanded roll rates from 0.5 to 1.5 deg/sec in either direction during significant portions of the coast period. Accommodation of larger roll rates can be evaluated on a mission-unique basis. For long coast missions, the Centaur vehicle is aligned to the ignition attitude prior to the Centaur MES2. At a guidance-calculated time, the Centaur main engine is re-ignited and the vehicle is guided to the desired orbit.

After reaching the target orbit, the Centaur main engine is shut down (MECO2), and Centaur begins its alignment to the spacecraft separation attitude. Centaur can align to any required attitude for spacecraft separation. Pre-separation spin-ups of up to 5.0 ±0.5 rpm about the roll axis can be accommodated. In addition, a pitch/yaw plane transverse spin mode can be used.

The Atlas V 500 series vehicles (with two or more SRBs) and the HLV have the capability to perform full or partial geosynchronous missions, requiring three Centaur burns. For these missions, the Centaur is equipped with an Extended Mission Kit (EMK), which includes additional battery power, additional helium pressurant gas, a full compliment of hydrazine maneuvering propellant, and radiation shielding over the hydrogen and oxygen tanks to reduce boiloff. After Centaur MECO2, the vehicle enters an approximate 5-hour coast period to geosynchronous altitude during which the Centaur performs numerous thermal conditioning maneuvers. At a guidance-calculated time, the Centaur aligns to the third burn attitude, the main engine is re-ignited (MES3), and the vehicle is guided to the desired orbit.

For all missions, after Centaur/spacecraft separation, Centaur conducts a Collision and Contamination Avoidance Maneuver (CCAM) to prevent recontact and minimize contamination of the spacecraft. A blowdown of remaining Centaur propellants follows after completion of CCAM.

2.3 MISSION DESIGN

2.3.1 Geo-transfer Trajectory and Performance Options

Based on the performance of the Atlas V family and enhanced capabilities of today's Liquid Apogee Engine (LAE) subsystems, Lockheed Martin determined that a 27-degree inclination is optimal for maximizing spacecraft beginning-of-life mass given an optimally sized spacecraft propulsion system. The 320-plus-second specific impulse of current LAEs has resulted in a shift in optimum inclination from 26.5 to 27 degrees. Lockheed Martin has identified the standard geotransfer orbit as 185 km perigee by 35,786 km apogee (100 nmi perigee by 19,323 nmi apogee) inclined 27 degrees. This is accomplished with the park orbit ascent profile. After a short coast, the transfer burn occurs across the first descending node, resulting in an argument of perigee of 180 degrees. The remaining delta-velocity to geosynchronous equatorial orbit is 1804 m/sec (5918.6 ft/sec). The performance capabilities of the Atlas V vehicle configurations to this orbit are listed in Table 2.6-1. For spacecraft weights heavier as well as much lighter than the standard GTO capability and through Centaur's flexible flight software, a number of trajectory designs are possible, each of which will minimize the spacecraft's remaining delta-velocity to geosynchronous orbit. One of the following trajectory designs is optimal for maximizing on-orbit lifetime of spacecraft that use a common source of liquid propellant for orbit insertion and on-orbit station keeping depending on mission requirements, total spacecraft mass, and dry mass-to-propellant mass ratio.

1. Subsynchronous transfer and perigee velocity augmentation

2. Short Coast Geosynchronous transfer (and reduced inclination transfers)

3. Supersynchronous transfer

4. Long coast and Extended coast Geosynchronous transfer

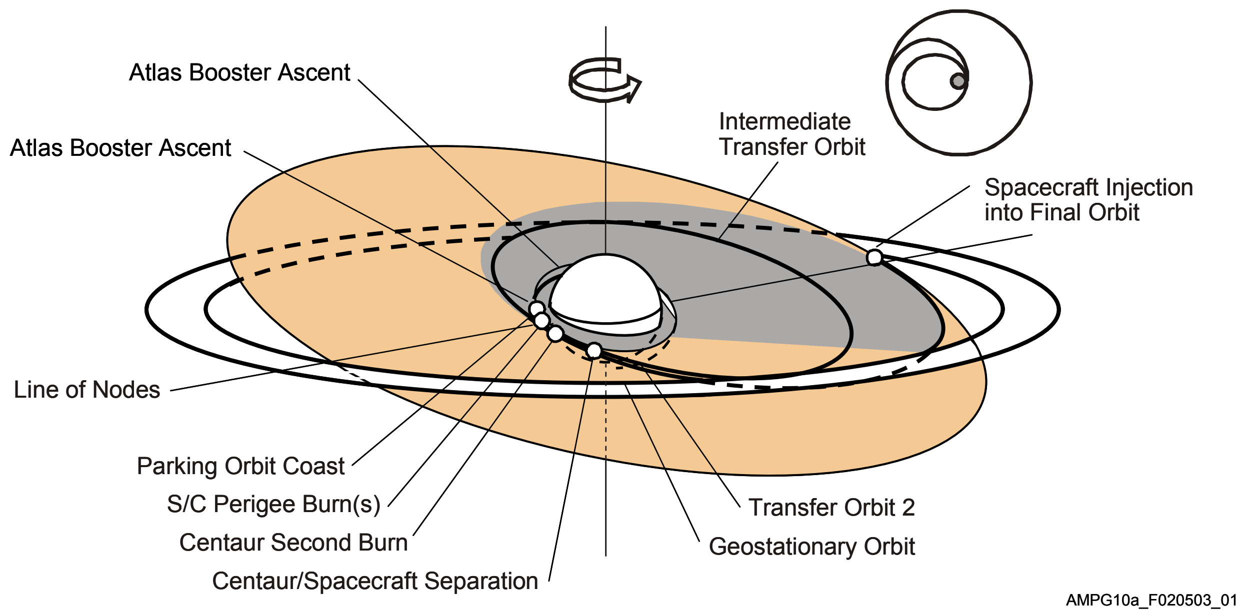
5. Three-burn Geosynchronous Orbit Injection

Since most spacecraft "qualify" for more than one of the above options, Table 2.3.1-1 at the end of this section shows example lifetime comparisons of multiple options.

2.3.1.1 Subsynchronous Transfer and Perigee Velocity Augmentation

For spacecraft heavier than the standard GTO capability for a particular Atlas V configuration, the Perigee Velocity Augmentation (PVA) trajectory design can provide increased propellant mass at beginning-of-life on GSO compared with the standard GTO design. Rather than off-load propellant to lower the launch mass to the standard GTO capability, the spacecraft is fully loaded and injected into a subsynchronous transfer orbit. This is beneficial when propellant tank capacity is large with respect to the dry mass because usable propellant on board the spacecraft is almost always more effective than higher apogee altitude. The Atlas V launch vehicle delivers the spacecraft to a subsynchronous intermediate transfer orbit (apogee less than geosynchronous) with an inclination of approximately 27 degrees since the spacecraft mass exceeds GTO launch capability. The separated spacecraft coasts to subsequent transfer orbit perigee(s), where the spacecraft supplies the required delta-velocity for insertion into geosynchronous transfer. At apogee, using one or more burns, the spacecraft lowers inclination and circularizes into GSO. As illustrated in Table 2.3.1-1, mass at beginning-of-life is enhanced. The orbit profile is shown in Figure 2.3.1.1-1. Several Atlas launch vehicle missions, including six UHF/EHF missions, have successfully used the subsynchronous transfer option.

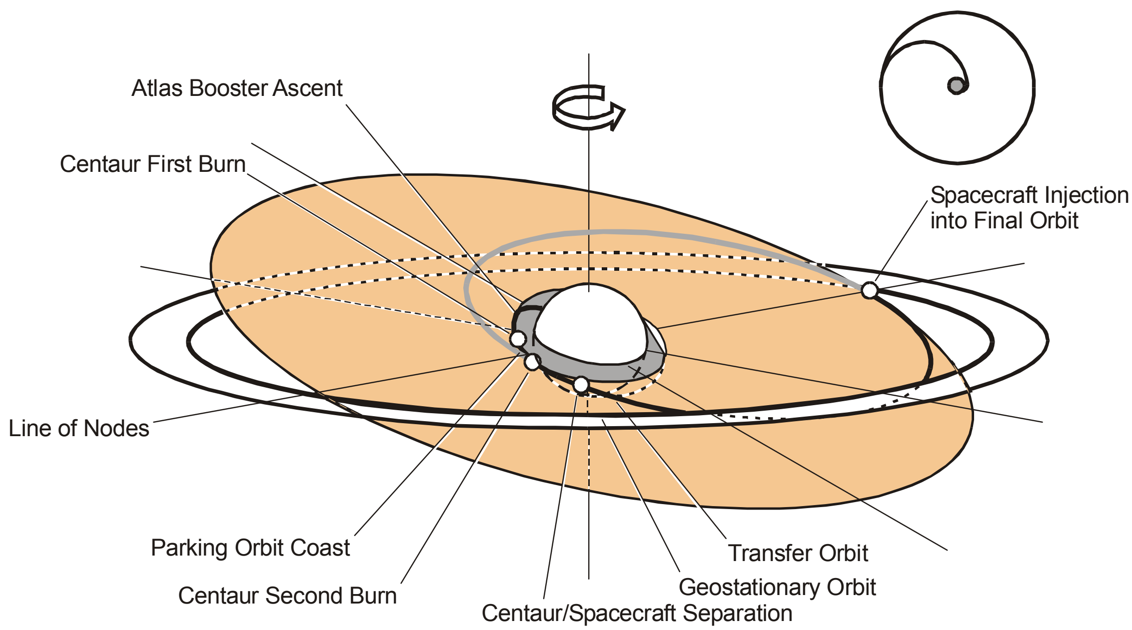
Figure 2.3.1.1-1: Subsynchronous Transfer Orbit Mission Trajectory Profile



2.3.1.2 Short Coast Geosynchronous Transfer Orbit

The short coast mission is the standard mission design for GTO launches. Figure 2.3.1.2-1 illustrates the orbital mission profile involved. For spacecraft that need to maintain geosynchronous altitude, 35,786 km (19,323 nmi), at the end of the transfer burn, the inclination will decrease from 27 degrees as the spacecraft mass decreases. The transfer orbit inclination depends on launch vehicle capability, spacecraft launch mass, and performance characteristics of both systems. With spacecraft weighing less than the GTO capability of the launch vehicle, excess performance can be used to further reduce inclination, raise perigee or both. At inclinations less than 20 degrees the launch vehicle can start increasing perigee and still maintain a relatively short coast and thus mission duration.

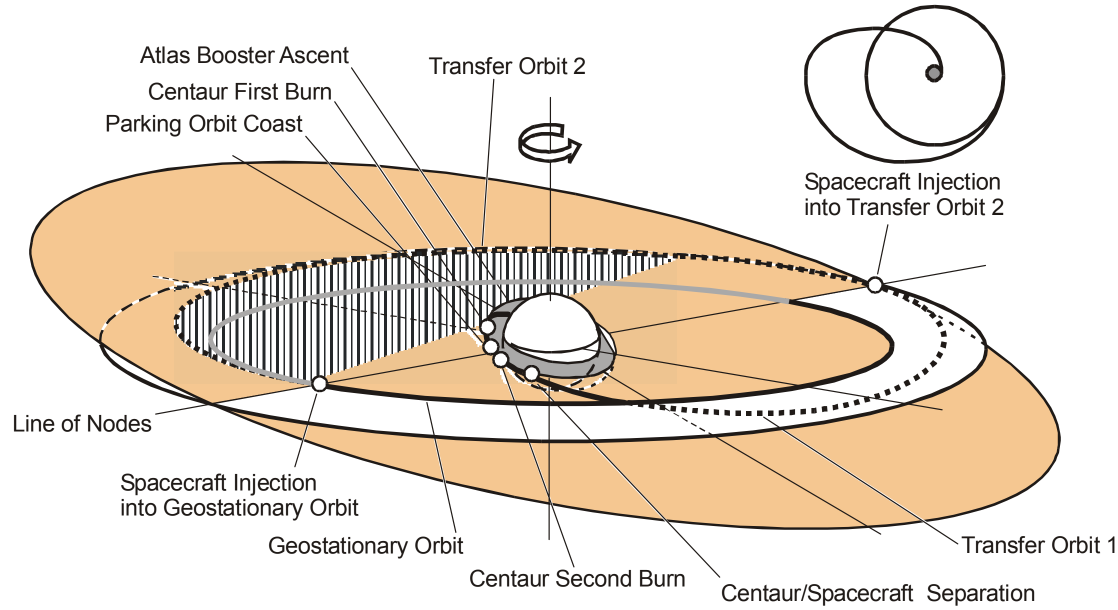
Figure 2.3.1.2-1: Short Coast Geosynchronous Transfer Orbit Mission Trajectory Profile



2.3.1.3 Supersynchronous Transfer

Many spacecraft do not need to maintain geosynchronous altitude after the transfer burn and are free to extend that altitude up to 90,000 km (48,596 nmi) or even further. The supersynchronous trajectory design offers an increase in beginning-of-life propellants by decreasing the delta-velocity required of the spacecraft for orbit insertion. This option is available if the launch vehicle capability to standard GTO is greater than the Payload Systems Weight. Initially, excess capability is used to increase apogee altitude. If the apogee altitude capability exceeds the spacecraft maximum allowable altitude, excess launch vehicle performance is used to lower orbit inclination. Optimally, when inclination reaches about 20 degrees, the perigee altitude starts to increase. The delta-velocity required of the spacecraft to reach GSO continues to decline. At supersynchronous altitudes, the decreased inertial velocity at apogee allows the spacecraft to make orbit plane changes more efficiently. The spacecraft makes the plane change and raises perigee to geosynchronous altitude in one or more apogee burns. The spacecraft then burns at perigee to continue to decrease the orbit inclination to 0 degrees and to lower the apogee so as to circularize into final geostationary orbit. 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 spacecraft propellants available for on-orbit operations. Figure 2.3.1.3-1 illustrates the supersynchronous trajectory mission profile. Table 2.3.1-1 shows an example of the advantage of using the supersynchronous transfer design over the standard short-coast option. Supersynchronous transfer trajectories have been flown on Atlas launch vehicle missions since December 1991.

Figure 2.3.1.3-1: Supersynchronous Transfer Orbit Mission Trajectory Profile

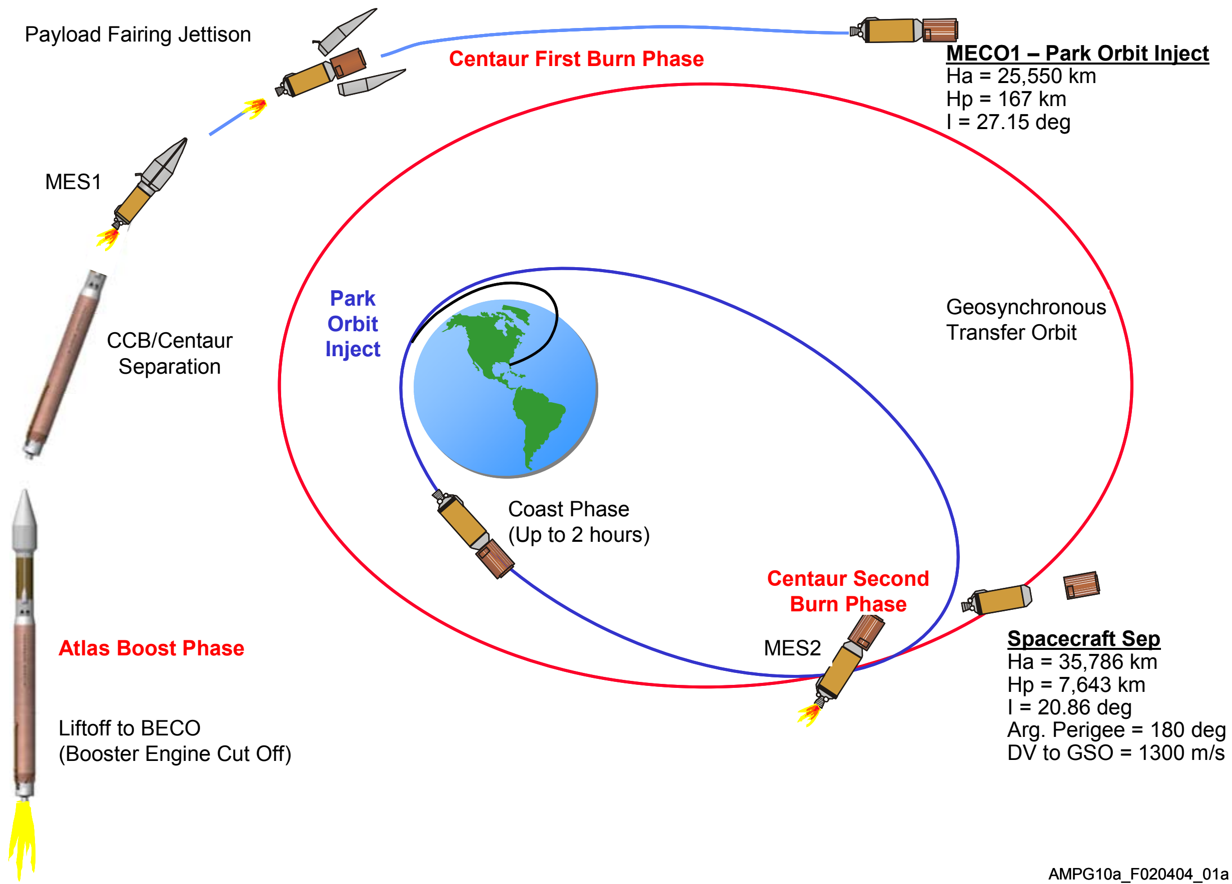


2.3.1.4 Long Coast and Extended Coast Geosynchronous Transfer

The first Atlas V long coast geotransfer was performed by AV-003 in July 2003, in which the transfer burn was delayed to occur across the first ascending node creating an apogee of 35,786 km and an argument of perigee of 0 degrees. This technique extends the park orbit coast by approximately 1 hour but can substantially reduce the spacecraft's Delta-Velocity (DV) to GSO by achieving a relatively high perigee as well as a reduced inclination.

Another long coast technique substantially reduces the DV to GSO, but the transfer burn does not occur across a node. Instead, the Centaur extends the first burn duration to achieve an elliptical park orbit. After a coast of up to 2 hours, the Centaur re-ignites its engine to extensively raise perigee and achieve an argument of perigee of 180 degrees. The extended coast allows Centaur to perform the second burn at a very high altitude thus raising perigee very efficiently. Figure 2.3.1.4-1 shows the extended coast trajectory design with example orbit parameters. Table 2.3.1-1 shows an example of the advantage of using extended coast over the supersynchronous option.

Figure 2.3.1.4-1: Extended Coast Transfer Orbit Mission Trajectory Profile



2.3.1.5 Three-Burn Geo-transfer/Geosynchronous Orbit Injection

This is the most efficient geo-transfer trajectory design option and is available to spacecraft with very low DV-to-GSO requirements. This option is currently available on the Atlas V 521, 531, 541, 551 launch vehicles and HLV. This type of profile combines the parking orbit ascent to a geosynchronous transfer burn with a long coast followed by a third Centaur burn. 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 burn places the spacecraft into transfer to geosynchronous altitude. A 5.2 hour coast follows the second burn. The Centaur engine is ignited for a third time near the apogee of the transfer orbit. This final third burn circularizes the Centaur/spacecraft at synchronous altitude and reduces the inclination to 0 degrees. For heavier spacecraft, where true geosynchronous equatorial orbit cannot be fully achieved, the third burn substantially reduces the remaining delta velocity to GSO over all 2-burn GTO flight design options. Table 2.3.1-1 shows an example of the advantage of using the 3-burn technique over the 2-burn extended-coast option.

Table 2.3.1-1: Geo-transfer Option Benefits (Example Comparison)

Parameter	Sub-synchronous		Super-synchronous		Extended Coast		Three Burn	
	GTO	**Subsynch**	**GTO**	**Super-synch**	**Super-synch**	**Extnd. Coast**	**2-Burn Extnd. Coast**	**3-Burn**
Atlas V Launch Vehicle	**401**		**521**		**521**		**541**	
S/C Mass, kg (lb)	4,859 (10,712)	**5,054 (11,142)**	6,370 (14,043)	6,370 (14,043)	5,100 (11,244)	5,100 (11,244)	5,715 (12,599)	5,715 (12,599)
S/C Offload, kg (lb) (Meet GTO Launch Capability)	195 (430)	**0**	--	**--**	--	**--**	--	**--**
S/C Dry Mass, kg (lb)	2,000 (4,409)	2,000 (4,409)	2,700 (5,952)	2,700 (5,952)	2,300 (5,071)	2,300 (5,071)	2,700 (5,952)	2,700 (5,952)
Transfer Orbit Parameters								
• Perigee Altitude, km (nmi)	185 (100)	185 (100)	185 (100)	185 (100)	686 (370)	**6,860 (3,704)**	7,685 (4,150)	**7,993 (4,316)**
• Apogee Altitude, km (nmi)	35,786 (19,323)	**30,000 (16,199)**	35,786 (19,323)	**60,000 (32,397)**	60,000 (32,397)	**35,786 (19,323)**	35,786 (19,323)	**35,786 (19,323)**
• Orbit Inclination, degree	27.0	27.0	20.9	**27.0**	19.4	**24.6**	22.5	**10.4**
• Argument of Perigee, degree	180	180	180	180	180	180	180	180
Centaur Burns - Number	2	2	2	2	2	2	2	3
Spacecraft ΔV, m/s (ft/sec) (Required for GSO Insertion)	1,804 (5,918.6)	**1,912 (6,273.0)**	1,683 (5521.7)	**1,638 (5374.0)**	1,520 (4986.9)	**1,445 (4740.8)**	1,352 (4435.7)	**1,000 (3280.8)**
S/C Mass, kg (lb) (Beginning of Life)	2,685 (5,919)	**2,698 (5,948)**	3,664 (8,078)	**3,723 (8,208)**	3,095 (6,823)	**3,168 (6,984)**	3,660 (8,069)	**4,102 (9,043)**
Mission Lifetime, years (Estimated)	13.9	**14.2**	14.9	**15.8**	14.2	**15.6**	14.8	**20.7**

2.3.2 Mission Optimization and Geo-transfer Performance Enhancement Options

Atlas V launch vehicle 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 a desired target orbit.

The SQP technique uses independent design variables chosen to maximize a performance index and satisfy specified constraints. Typical control variables include booster phase initial pitch and roll (launch azimuth) maneuvers, Atlas booster phase 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 70 equality and inequality constraints on variables, such as dynamic pressure, tracker elevation angle, and target orbit perigee, apogee, inclination, and argument of perigee.

Lockheed Martin's experience launching interplanetary and scientific spacecraft has enabled us to add an additional trajectory analysis tool 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.

Atlas V uses these trajectory analysis tools and our extensive guidance and targeting capabilities to include spacecraft characteristics and programmatic goals in our mission optimization. It is sometimes possible to improve the nominal or high performing cases of a mission design by taking advantage of one of our mission enhancement options for those spacecraft that can tolerate a variable injection state. The most widely used options are:

1. Inflight Retargeting (IFR)
2. Minimum Residual Shutdown (MRS)
3. IFR/MRS combination
4. Explicit Right Ascension of Ascending Node (RAAN) control

It should be noted that these mission enhancement options affect both the nominal injection state as well as the 99% confidence injection state. Contact Lockheed Martin for details on how each option would affect your particular mission requirements.

2.3.2.1 Inflight Retargeting

The software capability of the Centaur makes it possible to evaluate the Atlas V booster performance during ascent and to target for an optimal injection condition that is a function of the performance of the booster. Centaur can retarget transfer orbit inclinations for the standard short-coast mission and the long-coast (ascending node transfer) mission. 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 V booster dispersion is eliminated. Second, whether Atlas V performance is high, nominal, or low, the retargeting logic is calibrated to devote all remaining propellant margin to benefit the mission. While the 99% confidence injection orbit is roughly equivalent to a standard GCS, IFR does improve the nominal and positive performing cases by injecting the Centaur into a lower inclination orbit. With IFR, any desired level of confidence of a guidance shutdown can be implemented. IFR has been successfully flown on several missions, starting with the Atlas II/EUTELSAT II mission.

2.3.2.2 Minimum Residual Shutdown

Centaur propellants may be burned to minimum residual levels 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. While the 99% confidence injection orbit is roughly equivalent to a standard GCS, MRS does improve the nominal and positive performing cases by injecting the Centaur into a higher apogee orbit with the supersynchronous mission design; or into a higher perigee with the extended coast design (apogee can be slightly higher than 35,786 km). With MRS, any desired level of confidence of a guidance shutdown can be implemented. For cases designed to take full advantage of all usable propellants, the MRS targets would only be achieved, causing a guidance-commanded shutdown for a 6-sigma high performing vehicle. Alternatively, the MRS targets may be set to result in a guidance shutdown for a nominal vehicle (i.e., 50% chance to GCS, 50% chance to MRS) or any other probability of GCS. This implementation is typically used to protect a maximum apogee cap requirement.

It is practical for Centaur to burn all its propellants when the spacecraft 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 spacecraft using solid propellant (fixed impulse) Orbit Insertion Stages (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 spacecraft compensates for the effects of the actual launch vehicle performance. This primarily affects apogee altitude. The Atlas V performance variations associated with MRS are shown in Table 2.3.2.2-1. 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. MRS has been successfully executed for numerous missions and has become a common operations mode for GTO-type missions.

Table 2.3.2.2-1: Atlas V Perigee Velocity Variations with MRS

Perigee Velocity Dispersions	2.33 Sigma	3 Sigma
Atlas V 400 Series	75.9 m/s (249 ft/sec)	97.7 m/s (321 ft/sec)
Atlas V 500 Series	59.1 m/s (194 ft/sec)	76.1 m/s (250 ft/sec)

2.3.2.3 IFR/MRS Combination

This mission design allows the transfer orbit inclination target to vary with Atlas V booster performance while apogee altitude varies with Centaur performance. As with MRS alone, no Centaur propellants remain to guarantee a specific transfer orbit target apogee so this design takes full advantage of flight performance. IFR/MRS can be used with an apogee cap or unconstrained apogee. Using a part of the performance for inclination reduction reduces the range of apogee altitudes at injection. This option is beneficial to spacecraft that are nominally near their maximum allowable injection apogee by using some positive performance to be used to decrease inclination while taking using some positive performance to raise injection apogee.

2.3.2.4 Right Ascension of Ascending Node Control

Some spacecraft mission objectives may require launch-on-time placement into transfer and/or final orbit. For Earth orbital missions, RAAN may be targeted to a specific 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 target parameters. With GTO missions, some spacecraft mission operational lifetimes can be enhanced by controlling RAAN of the targeted transfer orbit. A spacecraft intended to operate in a non-0 degree geosynchronous final orbit can benefit with proper RAAN placement. A drift toward a 0 degree inclination orbit can help reduce the typical north-south station keeping budget of the spacecraft thereby increasing the amount of time the spacecraft can remain in an operational orbit.

2.3.3 Injection Accuracy and Separation Control

The combination of the Atlas V precision guidance hardware with flexible guidance software provides accurate spacecraft injection conditions for a wide variety of mission types. These functional capabilities have been demonstrated on LEO and GTO 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 Centaur burn. To date, the Atlas launch vehicle has met all GTO mission injection accuracy requirements.

On more than 100 past missions, Lockheed Martin has demonstrated its capability to deliver spacecraft of various volumes and masses to precisely targeted orbits. Because the Atlas V vehicle is primarily designed for dedicated, single payload missions, a number of Atlas-unique, flight-proven mission trajectory and targeting enhancement options are offered to maximize the benefits of the Atlas V system to the spacecraft mission. Atlas V provides exceptional orbit placement capabilities as demonstrated by the flight-derived orbital injection accuracy data of the Atlas V family of launch vehicles in Tables 2.3.3-2 and 2.3.3-3. Orbital insertion accuracy for Guidance Commanded Shutdown (GCS) missions is listed in Table 2.3.3-2. Since propellant margin is reserved to accommodate launch vehicle dispersions and ensure a guidance

Table 2.3.3-1: Typical Injection Accuracies at Spacecraft Separation

Atlas V								
Orbit at Centaur Spacecraft Separation				± 3-sigma Errors				
Mission	Apogee km (nmi)	Perigee km (nmi)	Inclination deg	Apogee km (nmi)	Perigee km (nmi)	Inclination deg	Argument of Perigee deg	RAAN deg
GTO (Coast ≤ 800 sec)	35,897 (19,383)	195 (105)	25.6	168 (91)	4.6 (2.5)	0.025	0.2	0.22
GTO (Coast ~ 5400 sec)	35,765 (19,312)	4,316 (2,330)	21.7	238 (129)	12.0 (6.5)	0.025	0.37	0.39
Super-Synch	77,268 (41,722)	294.5 (159)	26.4	586 (316)	4.6 (2.5)	0.025	0.32	0.34
Legend: N/A = Not Applicable or Available								

commanded shutdown, the flight results shown in Table 2.3.3-2 reflect the precision of the Atlas launch vehicle guidance and navigation hardware and flight software algorithms. For missions that incorporate Atlas V mission targeting options of IFR, MRS, or combination IFR/MRS, injection accuracy is provided in Table 2.3.3-3 in terms of the difference between the predicted and achieved ideal velocity required to inject from the transfer orbit to Geosynchronous Orbits. This method provides a common comparison of flight results for the wide variety of Atlas V launch vehicle mission optimization and targeting options. In addition, the delta-velocity to GSO that results from the achieved transfer orbit not only reflects guidance system performance, but also includes actual launch vehicle performance. As shown in Table 2.3.3-3, the near-zero statistical mean of the Atlas launch vehicle flight results demonstrates the accuracy of the Atlas family of launch vehicle performance modeling and preflight performance predictions.

2.3.3.1 Attitude Orientation and Stabilization

The Guidance, Navigation, and Control (GN&C) system can orient the spacecraft to any desired separation 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 pre-separation stabilization, Centaur can provide a stabilized spin rate to the spacecraft about any desired axis, including arbitrary combinations of vehicle primary axes. The Centaur system can accommodate longitudinal spin rates up to 5.0 rpm, subject to some limitation due to space vehicle mass property misalignments. Spin rates about transverse axes have been demonstrated up to 7 deg/sec. A detailed analysis for each Centaur/spacecraft combination will determine the maximum achievable spin rate. Furthermore, known rates imparted to the spacecraft by the separation system (e.g., due to spacecraft center of gravity offsets) can be compensated for by imparting an equal and opposite rate before the separation event. This pre-compensation feature can improve the final spacecraft pointing, nutation, and transverse rates, subject to spin rate requirements and attitude control constraints.

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

Table 2.3.3-2: Injection Accuracy Results for GCS Missions

Vehicle	Satellite	Launch Date	Achieved –Target Apogee	3σ Apogee	Achieved Apogee Accuracy (σ)	Achieved –Target Perigee	3σ Perigee	Achieved Perigee Accuracy (σ)	Achieved –Target Inclination	3σ Inclination	Achieved Inclination Accuracy (σ)	Achieved –Target Argument of Perigee	3σ Argument of Perigee	Achieved Argument of Perigee Accuracy (σ)
AC-101	MLV 1	10-Feb-92	-12.16	41.00	-0.89	-0.22	1.30	-0.51	0.003	0.020	0.45	0.020	0.180	0.33
AC-72	Galaxy V	13-Mar-92	-0.01	49.00	0.00	0.03	1.10	0.08	0.010	0.030	1.00	-0.030	0.210	-0.43
AC-105	Intelsat K	09-Jun-92	-6.02	58.18	-0.31	0.05	1.05	0.14	-0.002	0.013	-0.46	0.001	0.128	0.02
AC-103	MLV 2	02-Jul-92	-20.58	41.00	-1.51	-0.10	1.30	-0.23	0.001	0.020	0.15	-0.020	0.180	-0.33
AC-104	MLV 3	19-Jul-93	-8.04	41.00	-0.59	0.02	1.30	0.05	-0.001	0.020	-0.15	0.015	0.180	0.25
AC-106	MLV 4	28-Nov-93	-14.71	41.00	-1.08	-0.17	1.30	-0.39	-0.001	0.020	-0.15	0.017	0.180	0.28
AC-77	GOES J	23-May-95	-33.40	58.00	-1.73	0.22	1.10	0.60	-0.007	-0.016	1.31	-0.005	0.389	-0.04
AC-118	MLV 5	31-Jul-95	12.43	41.00	0.91	0.07	1.30	0.16	-0.006	0.020	-0.90	0.010	0.180	0.17
AC-122	Inmarsat 3	03-Apr-96	0.54	68.50	0.02	0.07	1.20	0.18	0.000	0.043	0.00	0.000	0.279	0.00
AC-78	SAX	30-Apr-96	0.90	2.50	1.08	0.00	3.60	0.00	0.007	0.019	1.11	NA	NA	NA
AC-129	Inmarsat 3	17-Dec-96	10.60	68.50	0.46	0.13	1.20	0.33	-0.001	0.043	-0.07	0.010	0.279	0.11
AC-79	GOES K	25-Apr-97	11.00	57.70	0.57	0.50	1.00	1.50	-0.011	0.016	-2.06	0.030	0.250	0.36
AC-131	MLV 6	24-Oct-97	8.10	41.00	0.59	0.18	1.30	0.42	0.000	0.020	0.00	0.010	0.180	0.17
AC-151	Intelsat 806	27-Feb-98	-18.90	56.20	-1.01	0.49	1.00	1.47	0.005	0.010	1.50	0.015	0.315	0.14
AC-153	Intelsat 805	18-Jun-98	-31.10	56.20	-1.66	-0.11	1.00	-0.33	0.004	0.010	1.20	0.020	0.315	0.19
AC-141	EOS Terra	18-Dec-99	-0.59	3.80	-0.47	0.02	1.35	0.04	-0.007	0.100	-0.21	NA	NA	NA
AC-138	MLV 8	20-Jan-00	0.50	41.00	0.04	0.03	1.30	0.07	0.001	0.020	0.15	-0.002	0.180	-0.03
AC-137	GOES-L	03-May-00	28.12	80.53	1.05	0.25	0.94	0.80	-0.006	0.035	-0.51	0.018	0.236	0.23
AC-140	MLV 9	19-Oct-00	-0.05	41.00	0.00	0.22	1.30	0.51	0.003	0.020	0.45	-0.005	0.180	-0.08
AC-157	MLV 11	05-Dec-00	11.33	58.60	0.58	0.04	0.97	0.12	-0.002	0.011	-0.56	0.000	0.191	0.00
AC-156	ICO A1	19-Jun-01	7.33	32.05	0.69	-1.17	24.26	-0.14	0.001	0.060	0.05	NA	NA	NA
AC-142	GOES M	23-Jul-01	-13.39	80.52	-0.50	-0.03	0.94	-0.10	-0.003	0.035	-0.26	-0.008	0.236	-0.10
AC-160	MLV 10	8-Sep-01	-0.41	*	*	0.05	*	*	0.001	0.068	0.04	0.131	1.172	0.34
AC-162	MLV 12	11-Oct-01	2.29	58.74	0.12	-0.67	0.93	-2.16	0.000	0.0105	0.00	-0.005	0.191	-0.08
AV-002	Hellas-Sat	13-May-03	25.00	207.90	0.36	-0.03	1.08	-0.08	0.000	0.014	-0.086	0.005	0.230	0.061
AV-003	Rainbow 1	17-July-03	-1.36	54.00	-0.08	0.17	5.40	0.09	0.000	0.020	0.000	-0.003	0.400	-0.023
AC-164	MLV 14	2-Dec-03	0.19	*	*	0.35	*	*	-0.001	0.068	-0.04	0.086	1.172	0.22
AC-203	UHF-F11	17-Dec-03	3.27	46.56	0.21	0.85	2.14	1.19	0.001	0.0124	0.24	-0.001	0.1523	-0.02
AC-165	AMC-10	5-Feb-04	-1.15	104.33	-0.03	0.05	2.02	0.07	0.002	0.011	0.56	0.016	0.310	0.16
AC-166	AMC-11	19-May-04	-17.52	104.33	-0.50	-0.11	2.02	-0.16	0.002	0.011	0.56	-0.012	0.310	-0.12

Note:
--- Indicates data not yet published
* Indicates values not targeted for these missions.

Table 2.3.3-3: Injection Accuracy Results for IFR, MRS, and IFR/MRS Missions

Vehicle	Mission	ILC	Type	Predicted ΔV to GSO (m/s)	Achieved ΔV to GSO (m/s)	Delta (m/s)
AC-102	EUTELSAT II F3	7-Dec-91	IFR	1,542.4	1,539.3	-3.1
AC-75	UHF F2	3-Sept-93	MRS	2,349.0	2,409.0	60.0
AC-108	Telstar 4	15-Dec-93	IFR	1,774.4	1,740.2	-34.2
AC-73	GOES-I (8)	13-Apr-94	MRS	1,732.3	1,740.2	7.9
AC-76	UHF F3	24-June-94	MRS	2,325.3	2,375.8	50.5
AC-107	DirecTV / DBS-2	3-Aug-94	MRS	1,776.0	1,765.8	-10.2
AC-111	INTELSAT 703	6-Oct-94	MRS	1,802.2	1,764.8	-37.4
AC-110	Orion Atlantic F1	29-Nov-94	IFR	1,472.4	1,472.2	-0.2
AC-113	INTELSAT 704	10-Jan-95	MRS	1,758.2	1,740.8	-17.4
AC-112	EHF F4	28-Jan-95	MRS	1,961.6	1,959.0	-2.6
AC-115	INTELSAT 705	22-Mar-95	MRS	1,769.7	1,748.6	-21.1
AC-114	AMSC-1 / MSAT	7-Apr-95	MRS	1,770.4	1,748.4	-22.0
AC-116	EHF F5	31-May-95	MRS	1,969.6	1,983.6	14.0
AC-117	JCSAT 3	28-Aug-95	IFR/MRS	1,525.9	1,531.9	6.0
AC-119	EHF F6	22-Oct-95	MRS	1,955.6	1,958.2	2.6
AC-120	Galaxy-IIIR	14-Dec-95	MRS	1,835.2	1,828.2	-7.0
AC-126	Palapa-C1	31-Jan-96	IFR/MRS	1,507.0	1,499.4	-7.6
AC-125	EHF F7	25-July-96	MRS	1,962.9	1,966.3	3.4
AC-123	GE-1	8-Sept-96	MRS	1,638.2	1,628.9	-9.3
AC-124	Hot Bird 2	21-Nov-96	IFR	1,730.2	1,740.0	9.8
AC-127	JCSAT 4	16-Feb-97	IFR/MRS	1,511.3	1,501.4	-9.9
AC-128	TEMPO	8-Mar-97	MRS	2,132.6	2,121.7	-10.9
AC-133	Superbird-C	27-July-97	IFR/MRS	1,543.1	1,529.1	-14.0
AC-146	GE-3	4-Sept-97	IFR	1,600.6	1,599.7	-0.9
AC-135	EchoStar III	5-Oct-97	IFR/MRS	1,756.9	1,752.7	-4.1
AC-149	Galaxy VIII-i	8-Dec-97	MRS	1,712.4	1,685.7	-26.7
AC-132	UHF F/O F8	16-Mar-98	MRS	2,081.9	2,057.9	-24.0
AC-134	HOT BIRD 5	9-Oct-98	IFR	1,754.7	1,763.2	8.5
AC-130	UHF F/O F9	20-Oct-98	MRS	1,992.9	2,003.4	10.5
AC-152	JCSAT 6	15-Feb-99	IFR/MRS	1,483.7	1,492.8	9.1
AC-154	Eutelsat W3	12-Apr-99	IFR/MRS	1,607.3	1,605.6	-1.7
AC-155	EchoStar V	23-Sept-99	MRS	1,714.2	1,711.2	-3.0
AC-136	UHF F/O F10	23-Nov-99	MRS	2,000.5	1,992.1	-8.4
AC-158	Hispasat-1C	3-Feb-00	IFR	1,595.4	1,593.5	-1.9
AC-201	Eutelsat W4	24-May-00	IFR	1,585.5	1,607.6	22.1
AC-139	TDRS-H	30-June-00	MRS	1,908.0	1,957.5	49.5
AC-161	EchoStar VI	14-July-00	MRS	1,762.8	1,770.2	7.4
AC-204	EchoStar VII	21-Feb-02	MRS	1,609.9	1,597.7	-12.2
AC-143	TDRS I	8-Mar-02	MRS	1,908.5	1,925.9	17.4
AV-001	Hotbird 6	21-Aug-02	IFR	1,569.4	1,571.6	2.2
AC-159	Hispasat 1D	18-Sep-02	IFR/MRS	1,619.1	1,625.6	6.5
AC-144	TDRS J	4-Dec-02	MRS	1,913.4	1,877.9	-35.6
AC-205	AsiaSat 4	12-Apr-03	MRS	1,717.0	1,701.2	-15.8
AC-202	MBSAT	13-Mar-04	IFR/MRS	1,758.3	1759.0	0.7
					Mean	-1.2
					Standard Deviation	20.3

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 non-spinning precision pointing mode, the system can maintain attitude errors less than 0.7 degrees, and attitude rates less than 0.2, 0.2, and 0.25 deg/sec about the pitch, yaw, and roll axes, respectively (prior to spacecraft separation) as shown in Table 2.3.3.2-1. Although the attitude and rates of a non-spinning 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 degrees per axis; body axis rates are typically less than 0.6 deg/sec in the pitch or yaw axis and 0.5 deg/sec in the roll axis. The angular momentum of the spacecraft after separation is often a parameter of interest. 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.

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

Any other axis of spin can be achieved before separation. For example, the spin axis can be aligned with the principal axes of the spacecraft (as provided by the customer). The magnitude of the spin rate achievable for an arbitrary axis will depend upon spacecraft mass properties, and must be determined on a mission-peculiar basis.

For a mission requiring pre-separation spinup, conditions just before spacecraft separation combine with any tip-off effects induced by the separation system and any spacecraft principal axis misalignments to produce post-separation 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 degrees, respectively.

The Atlas V Centaur flight control systems have been generically designed to accommodate spacecraft that fall within the range of payload mass properties, correlated with specific vehicle maneuvers, as identified in Table 2.3.3.2-2. The payload mass properties identified in this table include the spacecraft(s), the payload adapter, spacecraft separation system, and the associated 3-sigma uncertainties. Spacecraft that fall outside of these generic design ranges may be accommodated on a mission-peculiar basis.

Table 2.3.3.2-1: Guidance and Control Capabilities Summary

Centaur Coast Phase Attitude Control	
• Roll Axis Pointing, Half Angle	≤ 1.6 deg
• Passive Thermal Control Commanded Rate, (Clockwise or Counterclockwise)	
− Minimum	0.5 deg/sec
− Maximum	1.5 deg/sec
Centaur Separation Parameters at Separation Command (with No Spin Requirement)	
• Pitch, Yaw, Roll Axis Pointing, Half Angle	≤ 0.7 deg
• Body Axis Rates,	
− Pitch	±0.2 deg/sec
− Yaw	±0.2 deg/sec
− Roll	±0.25 deg/sec
Spacecraft Separation Parameters at Separation Command (with Transverse Spin Requirement)	
• Transverse Rotation Rate	≤ 7.0 deg/sec
Spacecraft Separation Parameters Following Separation (with Nonspinning or Slowspinning Requirement)	
• Pitch, Yaw & Roll Axis Pointing (per Axis)	≤ 0.7 deg
• Body Axis Rates,	
− Pitch	±0.6 deg/sec
− Yaw	±0.6 deg/sec
− Roll	±0.5 deg/sec
Spacecraft Separation Parameters Following Separation (with Longitudinal Spin Requirement)	
• Nutation, Half Angle	≤ 5.0 deg
• Momentum Pointing, Half Angle	≤ 3.0 deg
• Spin Rate,	≤ 30.0 ±3.0 deg/sec
Note: Capabilities Are Subject to Spacecraft Mass Properties Limitations	

Table 2.3.3.2-2: Design Range of Payload Mass Properties

Atlas V Config.	Pre-Spacecraft Separation Maneuver	Spacecraft Mass, kg (lb)	Forward cg Location, * mm (in.)	Lateral cg Offset, mm (in.)	Moments of Inertia, kg-m^2 (slug-ft^2)	Products of Inertia, kg-m^2 (slug-ft^2)
SEC	5-rpm Longitudinal Axis Spin	910–5,670 (2,000–12,500)	1,016–4,572 (40–180) **	±12 (±0.5)	Ixx= 410–5,420 (300–4,000) Iyy= 390–9,490 (285–7,000) Izz= 39–9,490 (285–7,000)	Ixy= ± 68 (± 50) Ixz= ± 68 (± 50) Iyz= ± 340 (± 250)
DEC	5-rpm Longitudinal Axis Spin	4,080–5,670 (9,000–12,500)	1,778–4,572 (70–180) **	±12 (±0.5)	Ixx= 2,580–5,420 (1,900–4,000) Iyy= 2,030–9,490 (1,500–7,000) Izz= 2,030–9,490 (1,500–7,000)	Ixy= ± 68 (± 50) Ixz= ± 68 (± 50) Iyz= ± 340 (± 250)
All	7-deg/s Transverse Axis Spin	1,810–5,670 (4,000–12,500)	1,016–3,302 (40–130)	±76 (±3)	Ixx= 1,080–5,420 (800–4,000) Iyy= 1,360–9,490 (1,000–7,000) Izz= 1,360–9,490 (1,000–7,000)	Ixy= ± 135 (± 100) Ixz= ± 135 (± 100) Iyz= ± 135 (± 100)
400 Series	3-Axis Stabilized Attitude Hold	910–9,070 (2,000–20,000)	1,016–4,572 (40–180)	±127 (±5)	Ixx= 410–10,850 (300–8,000) Iyy= 390–27,100 (285–20,000) Izz= 390–27,100 (285–20,000)	Ixy= ± 910 (± 670) Ixz= ± 910 (± 670) Iyz= ± 910 (± 670)
500 Series	3-Axis Stabilized Attitude Hold	1,360–19,050 (3,000–42,000)	1,016–5,715 (40–225)	±127 (±5)	Ixx= 680–40,700 (500–30,000) Iyy= 1,020–258,000 (750–190,000) Izz= 1,020–258,000 (750–190,000)	Ixy= ± 2,700 (± 2,000) Ixz= ± 2,700 (± 2,000) Iyz= ± 2,700 (± 2,000)

* Longitudinal cg Location Is Tabulated as Inches Forward of the Forward Ring of the Centaur Forward Adapter.
** Longitudinal cg Position Is Constrained as a Function of Payload Mass. Payload Longitudinal cg Position Is Restricted to 2540 mm (100 in.) above the Forward Ring of the Centaur Forward Adapter for a 2630.8 kg (5,800-lbm) Spacecraft. Linear Interpolation from this Point to Extreme Points Is Used.
By Definition, the x-axis is the Centaur Longitudinal Axis with Positive Direction Measure Forward. The y-axis is the Centaur Pitch Axis, and the z-axis is the Centaur Yaw Axis.

2.3.3.3 Separation System

The relative velocity between the spacecraft and the Centaur is a function of the mass properties of the separated spacecraft and the separation mechanism. The Atlas V separation systems are designed to preclude recontact between the spacecraft and Centaur and provide adequate separation for collision and contamination avoidance. Typically, the separation system achieves at least 0.27 m/sec (0.9 ft/sec) relative separation velocities.

2.4 ATLAS V SEQUENCE OF EVENTS

2.4.1 Mission Timelines

To familiarize users with Atlas V mission sequences, information is provided in the following table and figures regarding typical Atlas V mission designs. Table 2.4.1-1 shows the mission sequence data for of a typical Atlas V 401 GTO mission, Atlas V 401 LEO mission, Atlas V 521 Standard GTO mission, Atlas V 521 Extended Coast GTO mission, Atlas V 521 3-Burn mission, and Atlas V HLV GSO mission, respectively. These data are representative, as Atlas V can be launched at any time of day to meet spacecraft mission requirements.

Table 2.4.1-1 Typical Atlas V Launch Vehicle Mission Sequence Timeline

Atlas V Mission Sequence	401 Standard GTO	401 LEO Sun-synch	521 Standard GTO	521 Extended Coast GTO	521 3-Burn GSO	HLV GSO
(Time from T-0)	(sec)	(sec)	(sec)	(sec)	(sec)	(sec)
Guidance Go-Inertial	-8	-8	-8	-8	-8	-8
RD-180 Ignition	-2.7	-2.7	-2.7	-2.7	-2.7	-2.7
SRB Ignition	--	--	0.8	0.8	0.8	--
Liftoff	1.10	1.10	1.05	1.04	1.04	0.97
CCB Throttle Down/Up for max Q	--	--	38/58	39/57	41/57	59/312 *
Strap-On SRB Jettison	--	--	118	116	117	--
LRB Engine Cutoff	--	--	--	--	--	228
LRBs Jettison	--	--	--	--	--	236
PLF Jettison (500 series and HLV)	--	--	212	206	210	298
CCB Engine Cutoff (BECO)	242	238	249	251	252	367
CCB/Centaur Separation	250	246	257	259	260	375
Centaur Main Engine Start 1 (MES1)	260	256	267	269	270	385
PLF Jettison (400 series)	268	264	--	--	--	--
Centaur Main Engine Cutoff (MECO1)	946	1,015	904	1,041	782	563
Start Turn to MES2 Attitude	1,103	--	1,032	7,871	983	818
MES2	1,473	--	1,402	8,241	1,353	1,188
MECO2	1,689	--	1,668	8,373	1,617	1,678
Start Turn to MES3 Attitude	--	--	--	--	19,893	19,957
MES3	--	--	--	--	20,263	20,327
MECO3	--	--	--	--	20,391	20,543
Start Alignment to Separation Attitude	1,691	1,017	1,670	8,375	20,393	20,545
Separate Spacecraft	1,858	1,184	1,837	8,542	20,560	20,712
Start Turn to CCAM Attitude	1,883	1,209	1,862	8,567	20,585	20,737
Centaur End of Mission	6,258	5,584	6,237	12,942	24,960	25,112
* HLV CCB throttling is for performance efficiency and not for max Q control						

2.4.2 Ascent Profiles

Figures 2.4.2-1 through 2.4.2-6 show corresponding sequences-of-events for the illustrated Atlas V 400 series, 500 series and HLV timelines.

Figure 2.4.2-1: Typical Atlas V 401 Standard Short Coast GTO Ascent Profile

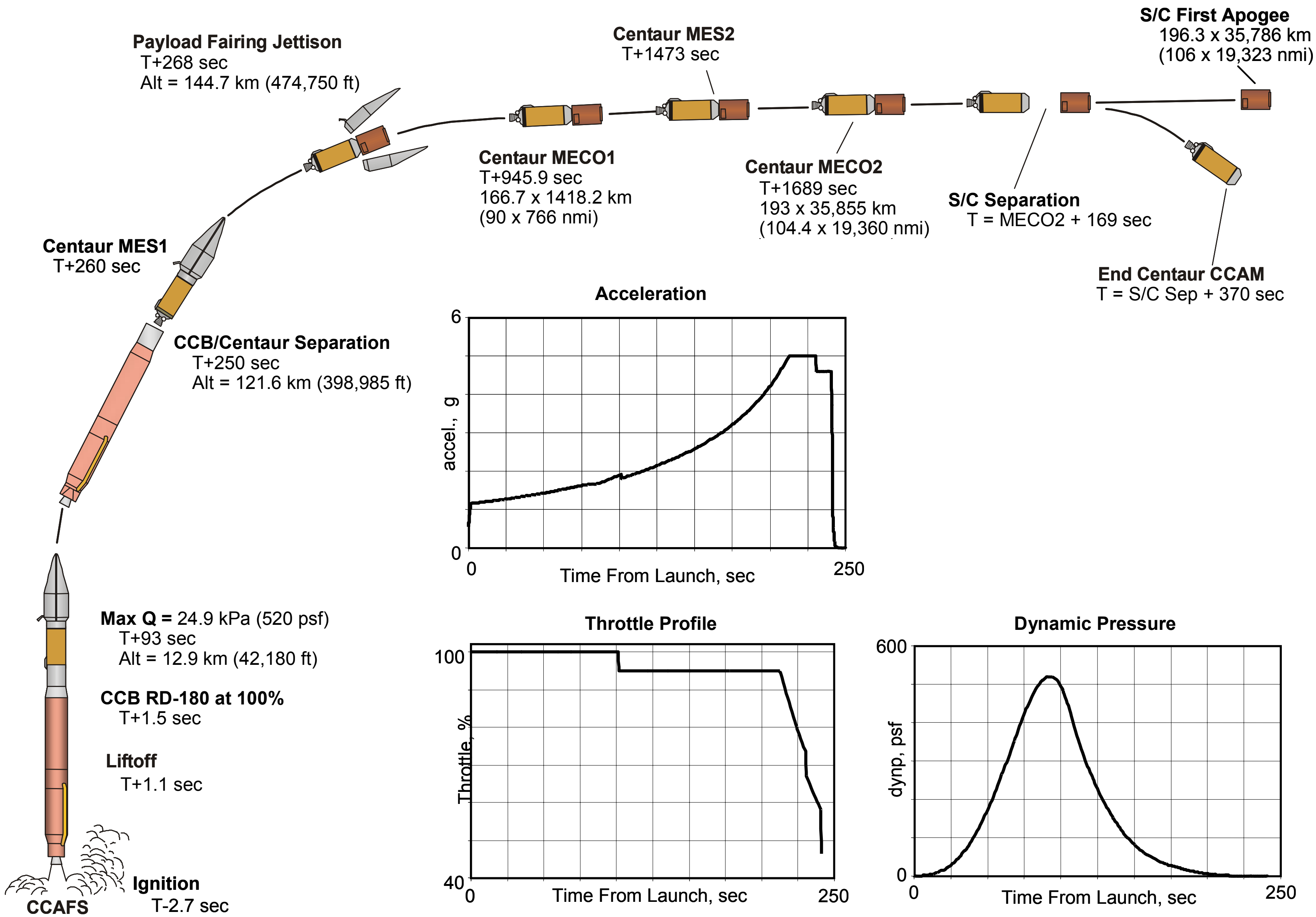


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Figure 2.4.2-2: Typical Atlas V 401 Standard LEO Sun-Synchronous Ascent Profile

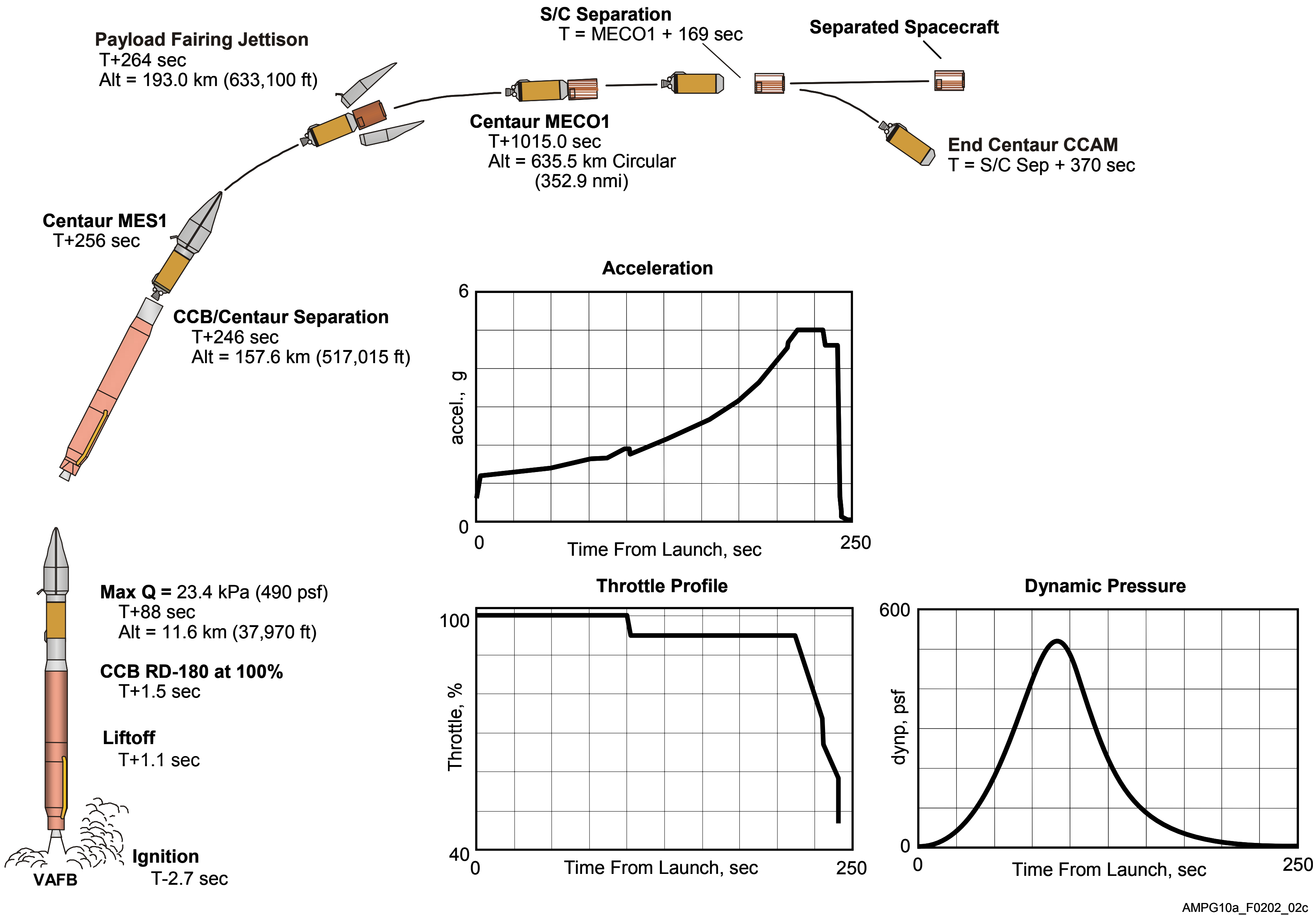

Figure 2.4.2-3: Typical Atlas V 521 Standard Short Coast GTO Ascent Profile (1 of 2)

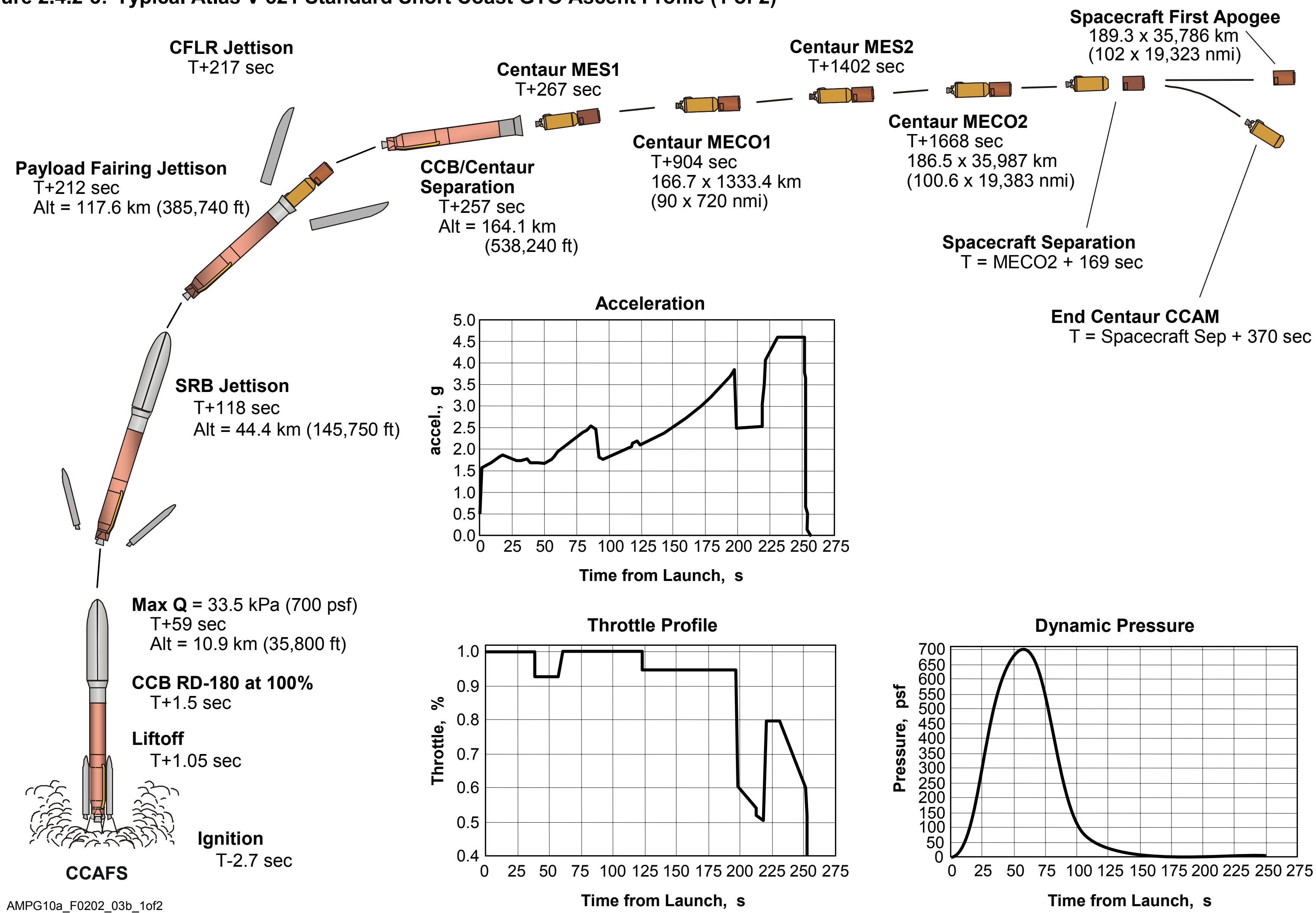

Figure 2.4.2-3: Typical Atlas V 521 Standard Short Coast GTO Ascent Profile (2 of 2)

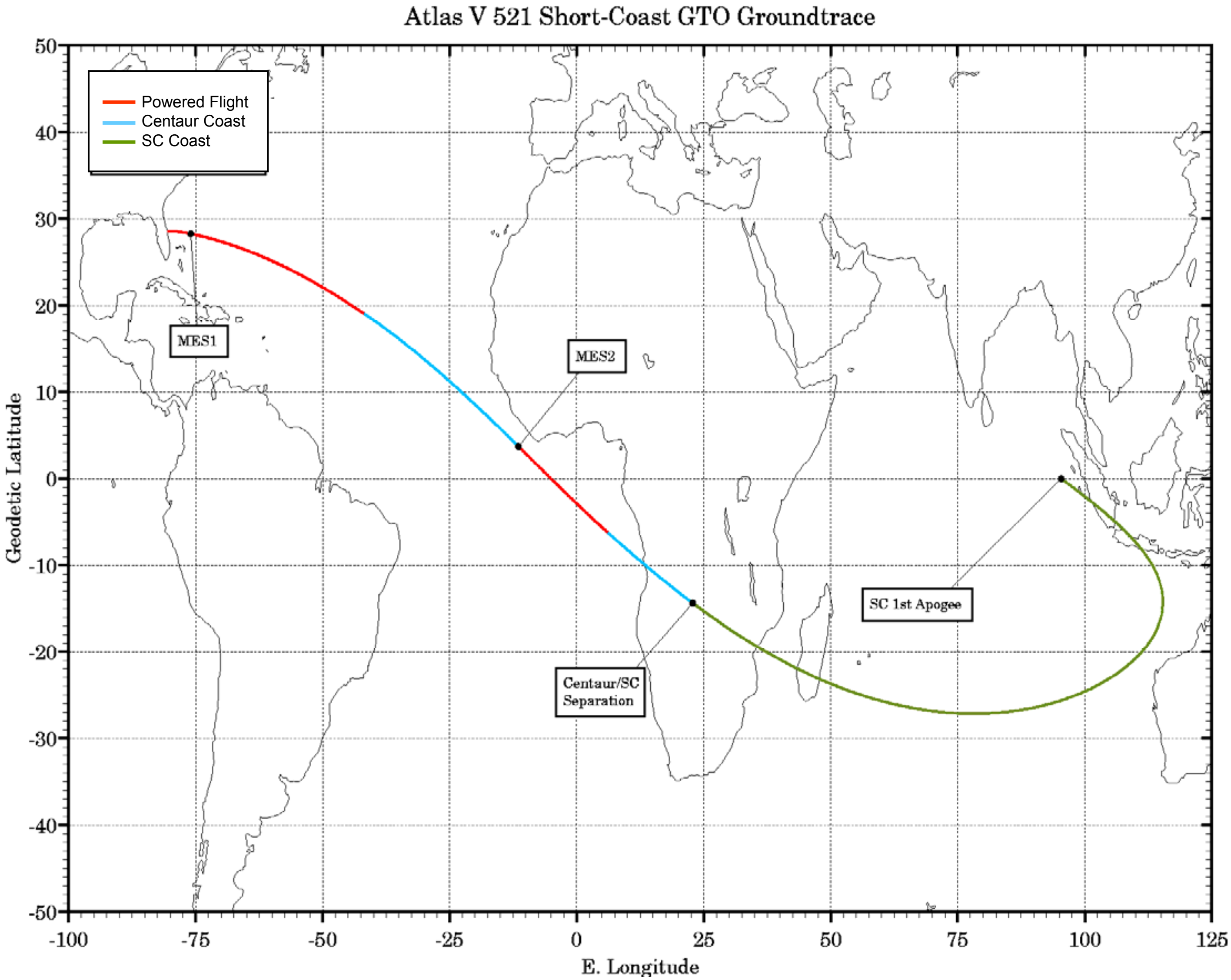

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Figure 2.4.2-4: Typical Atlas V 521 Extended Coast GTO Ascent Profile (1 of 2)

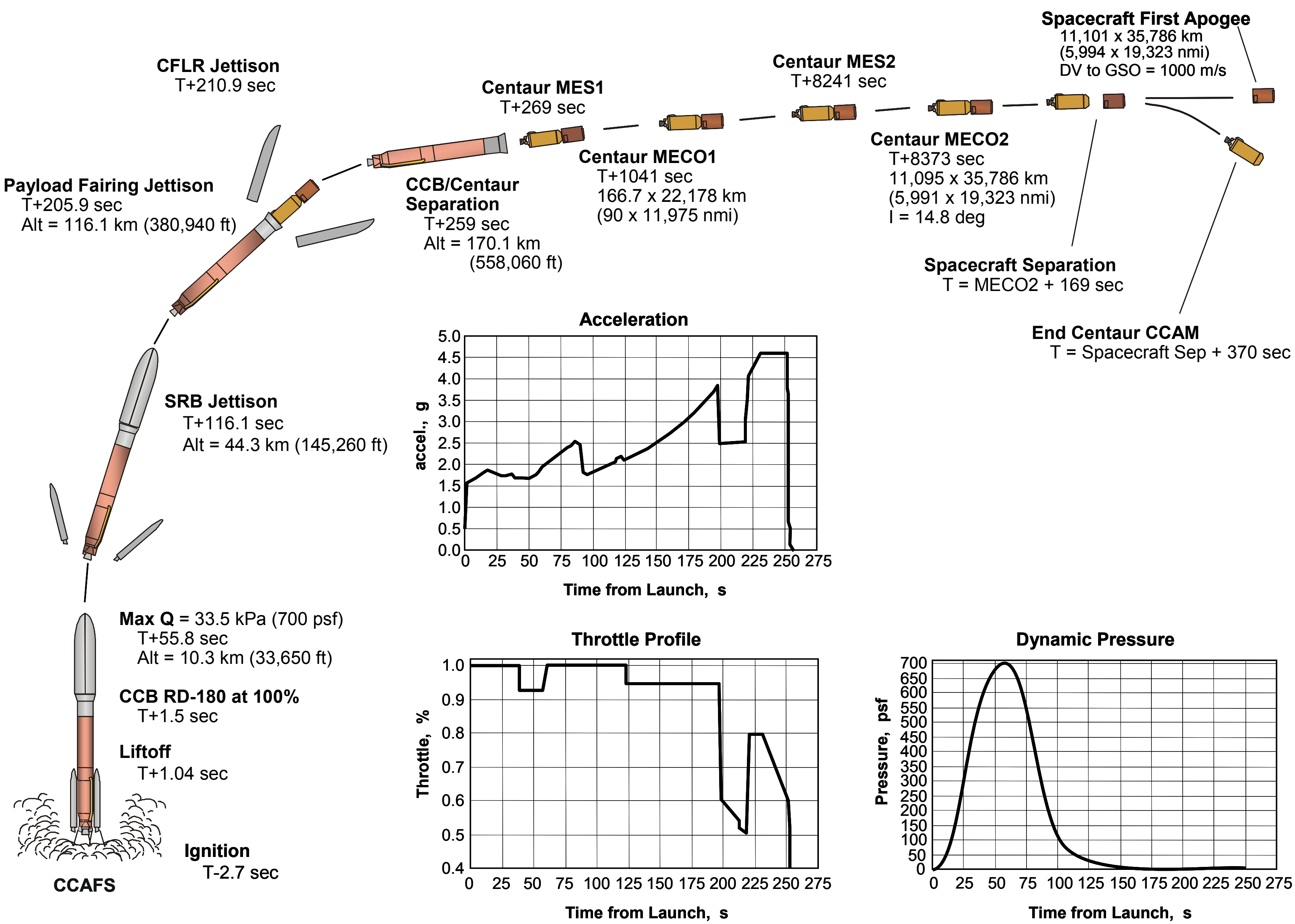

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Figure 2.4.2-4: Typical Atlas V 521 Extended Coast GTO Ascent Profile (2 of 2)

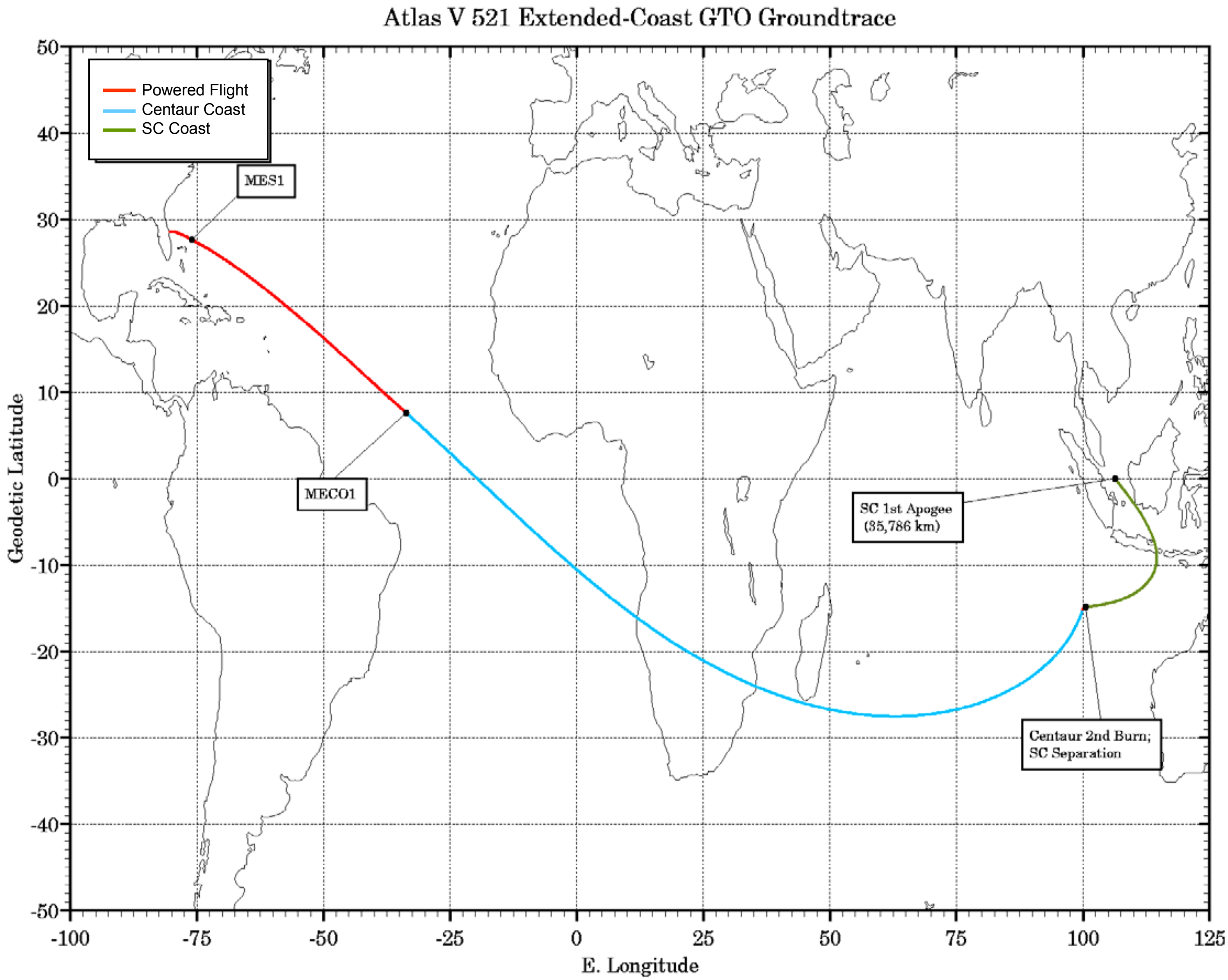

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Figure 2.4.2-5: Typical Atlas V 521 3-Burn GSO Ascent Profile (1 of 2)

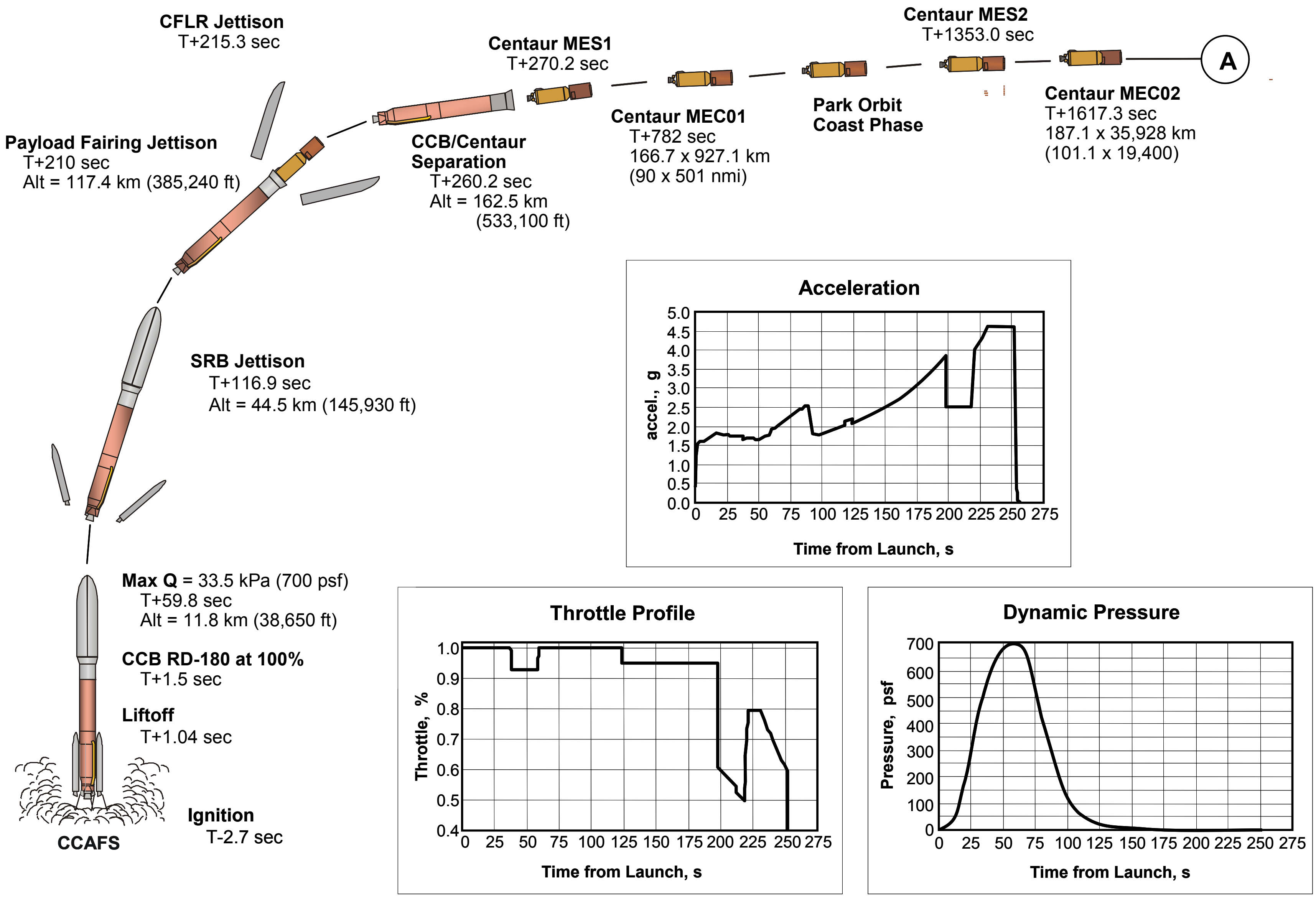

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Figure 2.4.2-5: Typical Atlas V 521 3-Burn GSO Ascent Profile (2 of 2)

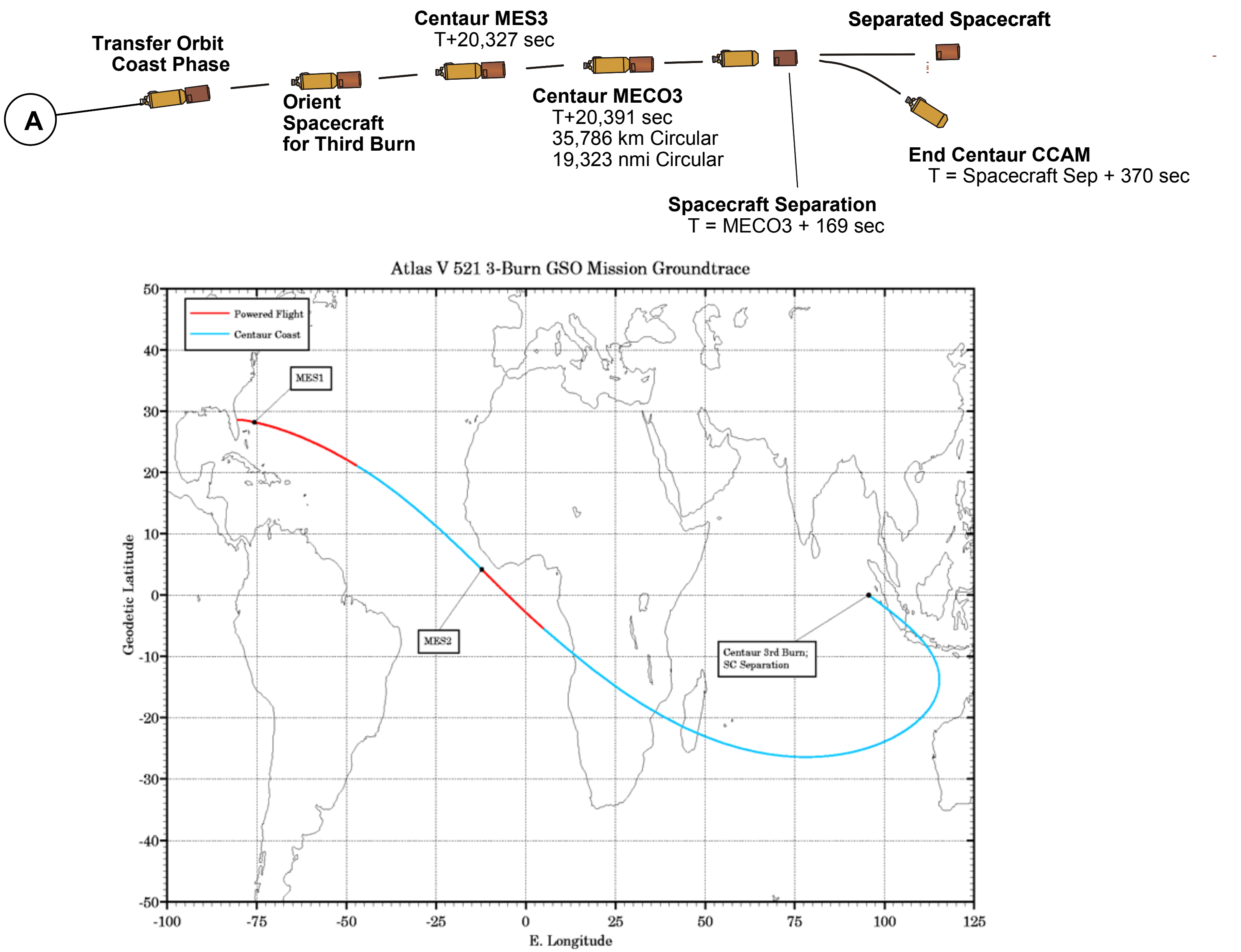

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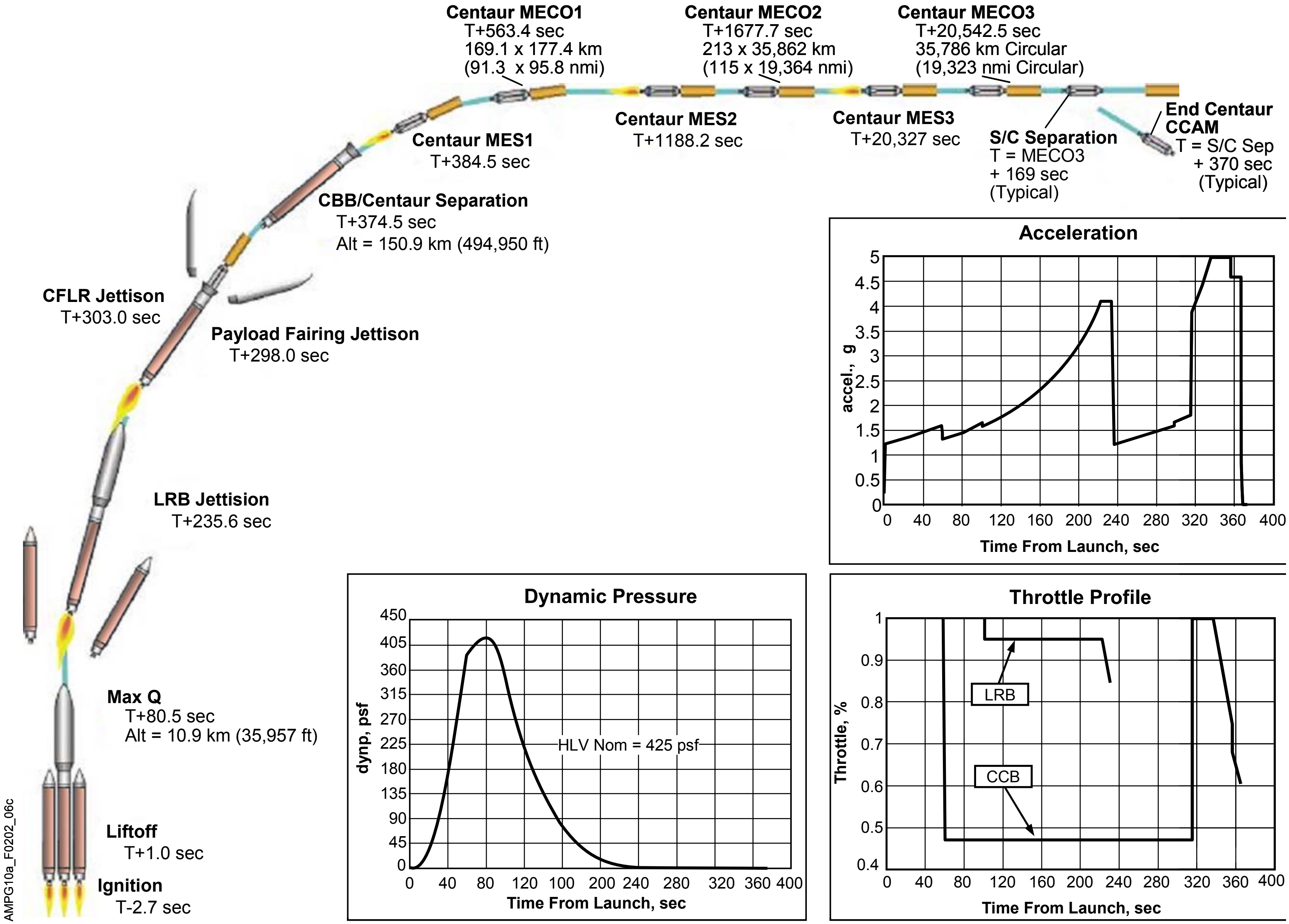
Figure 2.4.2-6: Typical Atlas HLV GSO Ascent Profile



2.5 PERFORMANCE GROUND RULES

Atlas V performance ground rules for various missions that launch from CCAFS in Florida or VAFB in California are described in this section.

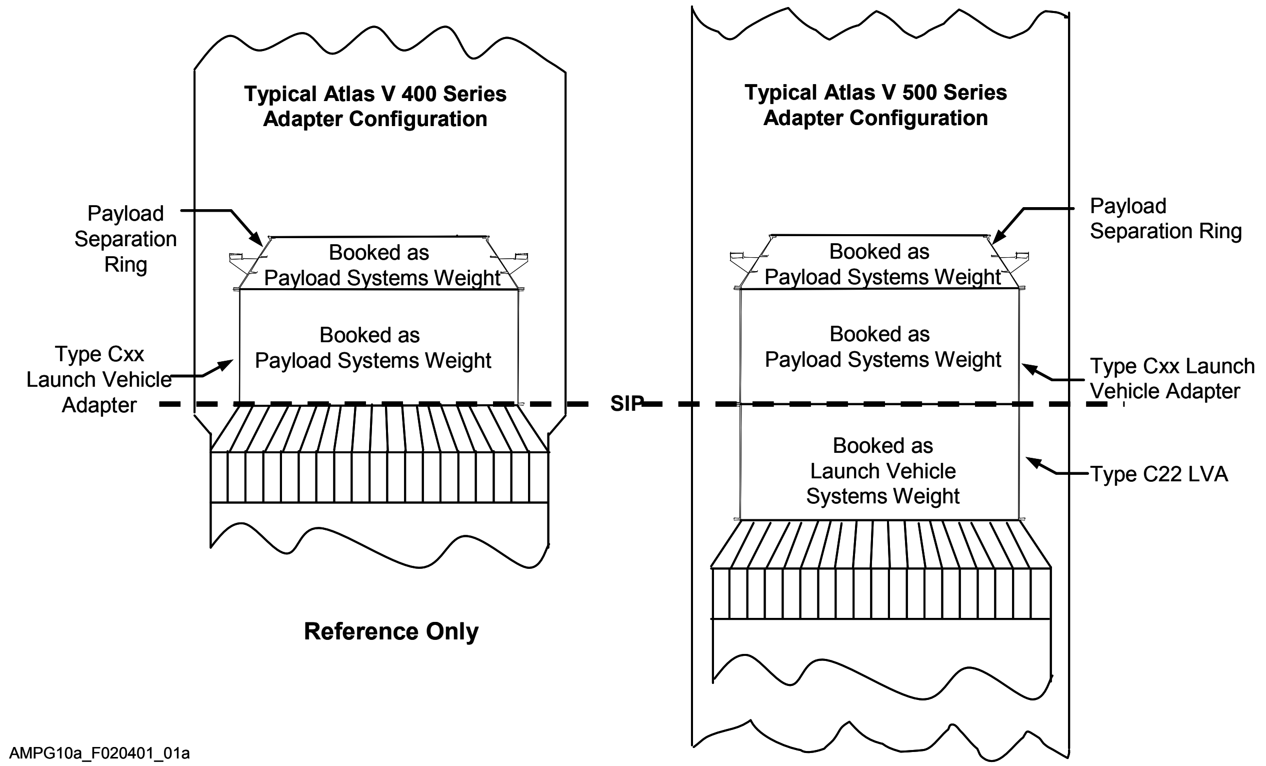
2.5.1 Payload Systems Weight Definition

Performance capabilities quoted throughout this document are presented in terms of Payload Systems Weight (PSW). PSW is defined as the separated spacecraft, the spacecraft-to-launch vehicle adapter, and other mission-peculiar hardware required on the launch vehicle to support the spacecraft. Table 2.5.1-1 provides masses for Atlas V standard payload adapters (reference Section 4.1.2 and Appendix E for payload adapter details). Data are also provided estimating performance effects of various Launch Vehicle Mission Peculiar (LVMP) hardware requirements. As a note, PLF performance effects shown (spacecraft mass-equivalent) are approximate. The launch vehicle trajectory, spacecraft mass, and mission target orbit can affect the performance contributions of each mission-peculiar item. Figure 2.5.1-1 illustrates the PSW definition.

Table 2.5.1-1: Performance Effects of Spacecraft Required Hardware

Item	Configuration	Mass, kg (lb)
Payload Adapter Mass		
Type A	Adapter	44 (97)
Type B1194	PSR + Launch Vehicle Adapter C13 C15 C22 C25 C29	 71.2 (157) 73.5 (162) 84.4 (186) 87.5 (193) 92.5 (204)
Type D1666	PSR + Launch Vehicle Adapter C13 C15 C22 C25 C29	 72.6 (160) 75.3 (166) 85.7 (189) 89.4 (197) 93.9 (207)
Other Spacecraft Required Hardware		
PLF Acoustic Panels	LPF	11 (25)
PLF Thermal Shield	LPF	4 (9)
Environmental Verification Package	Telepak, Instruments	9 (20)
Centaur Standard Package	Flight Termination System Airborne Harness	8 (18)
PLF Standard Package	Two Standard Access Doors Reradiating Antenna PLF Customer Logo	Included in PLF effect
LVMP Performance Effects		
	Performance Ratio	
Centaur Hardware	100 %	
PLF Hardware	15 %	

Figure 2.5.1-1: Payload Systems Weight Definition



2.5.2 Payload Fairing Performance Effects

2.5.2.1 Atlas V 400 Series

Atlas V 400 series performance is based on use of the 4-m EPF. GTO performance with the LPF is approximately 35 kg (77.2 lb) more than the EPF with the 401 configuration and 44 kg (97 lb) more with the 431 configuration. GTO performance with the XEPF is approximately 35 kg (77.2 lb) less than the EPF with the 401 configuration and 44 kg (97 lb) less with the 431 configuration.

2.5.2.2 Atlas V 500 Series

Atlas V 500 series performance is based on use of the 5-m short PLF. For spacecraft that require greater volume (height), the 5-m medium and long PLFs are available. For Atlas V 500 series configurations that use the 5-m medium PLF, the typical range of performance degradation is 59 kg (131 lb) for the Atlas V 511 and 521, to 67 kg (147 lb) for the Atlas V 501 GTO missions. For Atlas V 500 series configurations that use the 5-m long PLF, the typical range of performance degradation is 118 kg (260 lb) for the Atlas V 511 and 521, to 134 kg (295 lb) for the Atlas V 501 GTO missions. Additional payload fairing information can be found in Section 4.1.

2.5.2.3 Atlas V HLV

Atlas V HLV performance is based on use of the 5-m long PLF, although both short and medium payload fairings are available for spacecraft that require less volume (height).

2.5.3 Launch Vehicle Performance Confidence Levels

Atlas V missions are targeted to meet the requirements of each user. Historically, Atlas missions have been designed with a 2.33-sigma performance confidence level (99% probability of achieving target orbit elements within guidance injection accuracy). With the flexibility of Atlas/Centaur hardware and flight software, confidence levels can be set based on each mission's requirements. The MRS performance option, discussed later in this section, takes full advantage of this concept. All Earth escape-related performance and VAFB data in this document are based on the 3-sigma (99.87%) confidence level of GCS.

2.5.4 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. 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.

2.5.5 Centaur Coast Capability

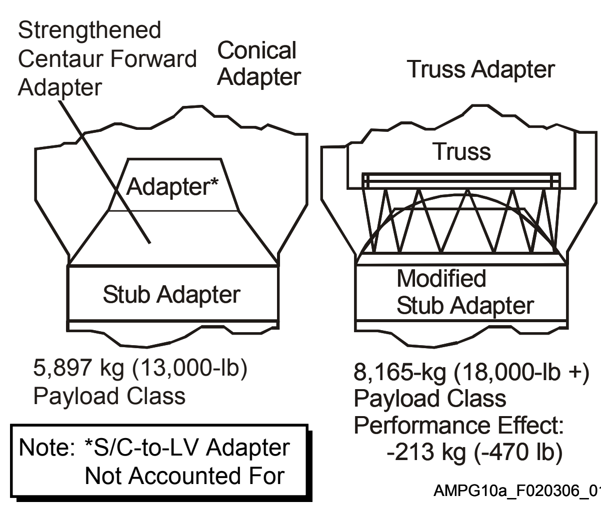
The standard Centaur incorporates park orbit coasts as short as 8 minutes and as long as 2 hours. The minimum coast time is constrained by Centaur propellant settling and conditioning requirements. A shorter coast time could be implemented on a mission-peculiar basis, if required. No additional hardware would be required, but vehicle turn angles may be limited. Longer coast times can also be accommodated although hardware changes are required. A Centaur EMK is required to support Centaur long coast durations. For the Atlas V 400 series this includes substitution of two 250 amp-hour main vehicle batteries for the standard 150 amp-hour battery. If additional helium pressurant is required, the Centaur can accommodate a fourth helium bottle.

For the Atlas V 500 series vehicles, the EMK consists of the same additional battery power, Centaur LH2 tank sidewall radiation shield, and the additional helium bottle, if required. Coasts of up to 6 hours in duration and/or Centaur three-burn missions are achievable with these EMK items. Performance estimates using long parking or transfer orbit coasts include the effect of an EMK.

2.5.6 Heavy Spacecraft Lift Capability

The Centaur forward adapter and payload adapters have been optimized for geosynchronous transfer missions. To manage the larger spacecraft masses (typically greater than 5,890 kg [13,000 lb]), two heavy spacecraft interfaces have been identified. Figure 2.5.6-1 illustrates these interfaces. The strengthened Centaur forward adapter is standard with Atlas V with a capability of 907.2 kg (2000 lbs). See Section 4.1.2.4 for truss capabilities. The user must account for the mass of a spacecraft-to-launch vehicle adapter in both cases. In addition, the stated performance penalty must be accounted for if the truss adapter is exercised.

Figure 2.5.6-1: Heavy Spacecraft Interfaces



2.6 ATLAS V PERFORMANCE CAPABILITY

Lockheed Martin offers a broad range of performance levels for the Atlas V series of launch vehicles as illustrated in Table 2.6-1.

Table 2.6-1: Atlas V 400/500 Series and HLV Performance Capabilities Summary

Orbit Type (DV to GSO)	400 Series				500 Series						HLV
	Number of Solid Rocket Boosters										
	0	1	2	3	0	1	2	3	4	5	N/A
	Payload Systems Weight (PSW), kg (lb)										
GTO (1804 m/s)	4,950 (10,913)	6,075 (13,393)	7,000 (15,432)	7,800 (17,196)	3,970 (8,752)	5,370 (11,839)	6,485 (14,297)	7,425 (16,369)	8,240 (18,166)	8,700 (19,180)	13,000 (28,660)
GTO (1500 m/s)	3,765 (8,300)	4,535 (9,998)	5,255 (11,585)	5,885 (12,974)	3,000 (6,614)	4,040 (8,907)	4,930 (10,869)	5,645 (12,445)	6,280 (13,845)	6,695 (14,760)	--
GSO	--	--	--	--	--	--	2,760 (6,085)	3,255 (7,176)	3,730 (8,223)	3,960 (8,730)	6,454 (14,229)
LEO I =28.5 deg	12,500* (27,558)*	--	--	--	10,300 (22,707)	12,590 (27,756)	15,080 (33,245)	17,250 (38,029)	18,955 (41,788)	20,520* (45,238)*	29,400* (64,816)*
LEO Sun-sync	7,095 (15,642)	8,763 (19,320)	10,168 * (22,416)	11,547 * (25,458)	6,319 (13,931)	8,310 (18,320)	10,161 (22,401)	11,803 (26,021)	13,049 (28,768)	14,096 (31,076)	--

Atlas V 400 Series:
- GTO and Sun-sync Performance is SEC
- LEO 28.5 deg Performance is DEC (402 Only)
- Quoted Performance is with 4-m EPF

Atlas V 500 Series and HLV:
- GTO, GSO and Sun-sync Performance is SEC
- 500 Series LEO 28.5 deg Performance is DEC
- 500 Series Quoted Performance is with 5-m Short PLF
- HLV LEO Performance is DEC
- HLV Quoted Performance is with 5-m Long PLF

* For 400 series, PSW above 9,072 kg (20,000 lb) may require mission-unique accommodations.
For 500 series and HLV, PSW above 19,051 kg (42,000 lb) may require mission-unique accommodations.

Note:
- GTO (1804 m/s): ≥ 185 x 35,786 km (≥ 100 x 19,323 nmi), Inclination = 27.0 deg, Argument of Perigee = 180 deg, CCAFS
- GTO (1500 m/s): Apogee Height = 35,786 km (19,323 nmi), Argument of Perigee = 180 deg, CCAFS
- GSO: 35,786 km Circular (19,323 nmi Circular), Inclination = 0 deg, CCAFS
- LEO 28.5 deg: 185 km (100 nmi) Circular, CCAFS
- LEO Sun-sync: 200 km (108 nmi) Circular, VAFB

- GCS: Guidance Commanded Shutdown, 2.33 sigma for CCAFS, 3-sigma for VAFB

2.6.1 Short-Coast Subsynchronous and Supersynchronous Geo-Transfer Capability

The optimum trajectory profile for achieving these transfer orbits is the parking orbit ascent. The 27 degree inclined orbit missions are launched at flight azimuths that have been approved and flown. The Centaur second burn is executed near the first descending node of the parking orbit (near the equator). Performance data are shown for a 2.33 σ GCS for a transfer orbit apogee inclination of 27 degrees.

Figure 2.6.1-1 with Table 2.6.1-1 illustrates the performance of the Atlas V 401 through 431 launch vehicles to geo-transfer orbits of various apogees. Figure 2.6.1-2 with Table 2.6.1-2 illustrates the performance of the Atlas V 501 through 551 launch vehicles to geo-transfer orbits of various apogees. Figure 2.6.1-3 with Table 2.6.1-3 illustrates the performance of the Atlas V HLV launch vehicle to geotransfer orbits of various apogees.

2.6.2 Short-Coast Standard Geo-transfer with Reduced Inclination Capability

Performance to orbit degrades on Atlas V vehicles for inclinations other than 28.5 degrees mostly due to the out of plane steering required to inject the vehicle into the desired final inclination. Performance data are shown for a 2.33 σ GCS for a transfer orbit apogee altitude of 35,786 km (19,323 nmi).

Figure 2.6.2-1 with Table 2.6.2-1 illustrates the performance of the Atlas V 401 through 431 launch vehicles to geo-transfer orbits of various inclinations. Figure 2.6.2-2 with Table 2.6.2-2 illustrates the performance of the Atlas V 501 through 551 launch vehicles to geo-transfer orbits of various inclinations. Figure 2.6.2-3 with Table 2.6.2-3 illustrates the performance of the Atlas V HLV launch vehicle to geo-transfer orbits of various inclinations.

2.6.3 Minimum Delta-V to Geosynchronous Orbit Performance Data

For many missions, spacecraft have limited propellant capability and need to be injected closer to GSO than our standard transfer orbit to achieve the required on-orbit lifetimes. For those missions, a mission design that minimizes the Delta-V to GSO remaining for a given PSW will provide the best mission design/maximum on-orbit lifetime. A large number of cases are presented for Delta-Vs from 1,000 m/s to 1,800 m/s for the Atlas V 400 series and 0 m/s to 1,800 m/s for the Atlas V 500 series vehicles. Depending on the maximum apogee capability of the spacecraft, varying performance can be achieved. Performance data shown in this section embrace the supersynchronous (with inclination reduction) and extended-coast mission design options.

Table 2.6.3-1 summarizes the Atlas V 401-431 launch vehicles minimum delta-velocity to geosynchronous orbit with various PSW. Figures 2.6.3-1a through 2.6.3-1d with Tables 2.6.3-1a through 2.6.3-1d depicts the Atlas V 401-431 launch vehicles minimum delta-velocity to geosynchronous orbit with various apogee caps. Table 2.6.3-2 summarizes the Atlas V 501-551 launch vehicles minimum delta-velocity to geosynchronous orbit with various PSW. Figures 2.6.3-2a through 2.6.3-2f with Tables 2.6.3-2a through 2.6.3-2f depict the Atlas V 501 - 551 launch vehicles minimum delta-velocity to geosynchronous orbit with various apogee caps.

Using the three-burn geo-transfer injection design, Table 2.6.3-3 summarizes the Atlas V 521-551 launch vehicles minimum delta-velocity to geosynchronous orbit with various PSW. Figure 2.6.3-3 with Tables 2.6.3-3a through 2.6.3-3d depicts the Atlas V 521-551 launch vehicles minimum delta-velocity to geosynchronous orbit.

2.6.4 Earth-Escape Performance Capability

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 1-hour coast time between the Centaur burns. The actual coast time necessary to achieve the desired departure asymptote will be determined by specific mission requirements and will affect actual performance. Our quoted performance assumes that 3-sigma (99.87% confidence level) flight performance reserves are held.

Figures 2.6.4-1a through 2.6.4-1d illustrate Earth Escape (C3) performance for the Atlas V 401 – 431 launch vehicles.

For the Atlas V 500 series launch vehicles, 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 48V (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.

Figures 2.6.4-2a through 2.6.4-2f show Earth Escape performance for the 501 through 551 vehicles with and without the Star 48V OIS.

2.6.5 CCAFS Low-Earth Orbit (LEO) Capability

Atlas V can launch spacecraft into a wide range of LEOs from CCAFS using direct ascent or parking orbit ascent mission profiles. LEO capabilities typically require Atlas V heavy payload modifications.

Direct Ascent to Circular Orbit — Circular orbit PSW capability to LEO is accomplished using the one Centaur burn mission profile. The maximum capability is available with planar ascent to a 28.5 degree inclination orbit. Inclinations from 28.5 degrees up to 55 degrees or even higher are possible with the direct ascent. Direct ascent performance to inclinations greater than 55 degrees may have significant performance impacts due to Range Safety constraints. Direct ascent to reduced inclination orbits is also possible, but at the expense of substantial performance.

Figure 2.6.5-1 shows LEO performance for Atlas V 402 from CCAFS from 200 km to 600 km circular. Figures 2.6.5-2a through 2.6.5-2c illustrate LEO performance from ~200 km to ~600 km for the 531, 532 and 552 configuration from CCAFS.

Direct Ascent to Elliptical Orbit — Elliptical orbit performance capability is accomplished using the direct ascent with perigee altitude at or above 185 km (100 nmi). Similar Range Safety and orbital mechanics constraints limit inclinations available with a CCAFS launch.

Parking Orbit Ascent to Circular Orbit — Spacecraft delivery to low-altitude circular orbit can be accomplished using the two Centaur burn mission profile. The first Centaur burn is used to inject the Centaur and spacecraft into an elliptic parking orbit. A park orbit perigee altitude of 167 km (90 nmi) is assumed for our reference cases. Expected parking orbit coast durations may require use of the Centaur EMK. The second Centaur burn will circularize the spacecraft into the desired orbit altitude.

High inclination orbits (inclinations greater than 55 degrees) impose Range Safety restrictions that require the Atlas V launch vehicle to meet Instantaneous Impact Point (IIP) constraints along the Eastern seaboard of the United States and Canada. Additional inclination is added in the later stages of the Centaur first burn and the Centaur second burn. As desired orbit inclination increases, performance degradations become more pronounced. High inclination orbit performance capabilities (≥63.4 deg) are more optimally achieved with launch from VAFB.

Circular orbit performance capabilities for altitudes between 600 km (324 nmi) and 2,000 km (~1,000 nmi) are shown in Figure 2.6.5-1 for 28.5, 55, and 63.4 degree inclinations for the Atlas V 402 launch vehicle.

2.6.6 VAFB LEO Capability

Atlas V can launch spacecraft into a wide range of LEOs from VAFB with inclinations ranging from 63.4 degrees through Sun-synchronous to retrograde using direct ascent or parking orbit ascent mission profiles. LEO capabilities typically require Atlas V heavy payload modifications.

Direct Ascent to Circular Orbit — Circular orbit PSW capability to LEO is accomplished using the one Centaur burn mission profile. The maximum capability is available with a 63.4 degree inclination orbit. Inclinations from 63.4 degrees and higher are possible with the direct ascent from VAFB.

Figures 2.6.6-1a through 2.6.6-1d with Tables 2.6.6-1a through 2.6.6-1d show LEO performance for Atlas V 400 series from VAFB from 200 km to 600 km circular. Figures 2.6.6-2a through 2.6.6-2g with Tables 2.6.6-2a through 2.6.6-2g show LEO performance from 200 km to 600 km for the Atlas V 500 series configuration.

Direct Ascent to Elliptical Orbit — Elliptical orbit performance capability is accomplished using the direct ascent with perigee altitude at or above 185 km (100 nmi).

Parking Orbit Ascent to Circular Orbit — Spacecraft delivery to low-altitude circular orbit can be accomplished using the two Centaur burn mission profile. The first Centaur burn is used to inject the Centaur and spacecraft into an elliptic parking orbit. A park orbit perigee altitude of 167 km (90 nmi) is assumed for our reference cases. Expected parking orbit coast durations may require use of the Centaur EMK. The second Centaur burn will circularize the spacecraft into the desired orbit altitude.

Figures 2.6.6-1a through 2.6.6-1d with Tables 2.6.6-1a through 2.6.6-1d show LEO performance for Atlas V 400 series from VAFB from 600 km to 2000 km circular. Figures 2.6.6-2a through 2.6.6-2g with Tables 2.6.6-2a through 2.6.6-2g show LEO performance from 600 km to 2,000 km for the Atlas V 500 series configuration.

2.6.7 Intermediate Circular Orbit Capability

Similar ground rules apply to intermediate circular orbit data (altitudes between ~500 km (270 nmi) and >9,000 km (>4,860 nmi)) as to LEO circular orbit data with the exception of heavy spacecraft requirements. The lower performance capabilities associated with the higher energy circular missions allow the use of standard spacecraft interfaces.

Figures 2.6.7-1a through 2 6.7-1c depict Intermediate Circular Orbit performance for the 531, 532 and 552 launch vehicles.

2.6.8 VAFB Elliptical Orbit Transfer Capability

Atlas V can launch spacecraft into high-inclination elliptical transfer orbits from SLC-3E at VAFB. Transfer orbits at 63.4 degrees 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 Centaur second burn executed near the first antinode (argument of perigee equaling 270 degrees).

2.6.9 VAFB High-Inclination, High-Eccentricity Orbit Capability

Atlas V can insert spacecraft into orbits with 12-hour or 24-hour periods at an inclination of 63.4 degrees by launching from VAFB. 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 2.6.6.

The following sections contain the figures and tables described herein and are arranged by Mission Category. We believe this is more convenient for the user.

Figure 2.6.1-1: Atlas V 401 - 431 Apogee Variation Performance to Geotransfer Orbit - CCAFS

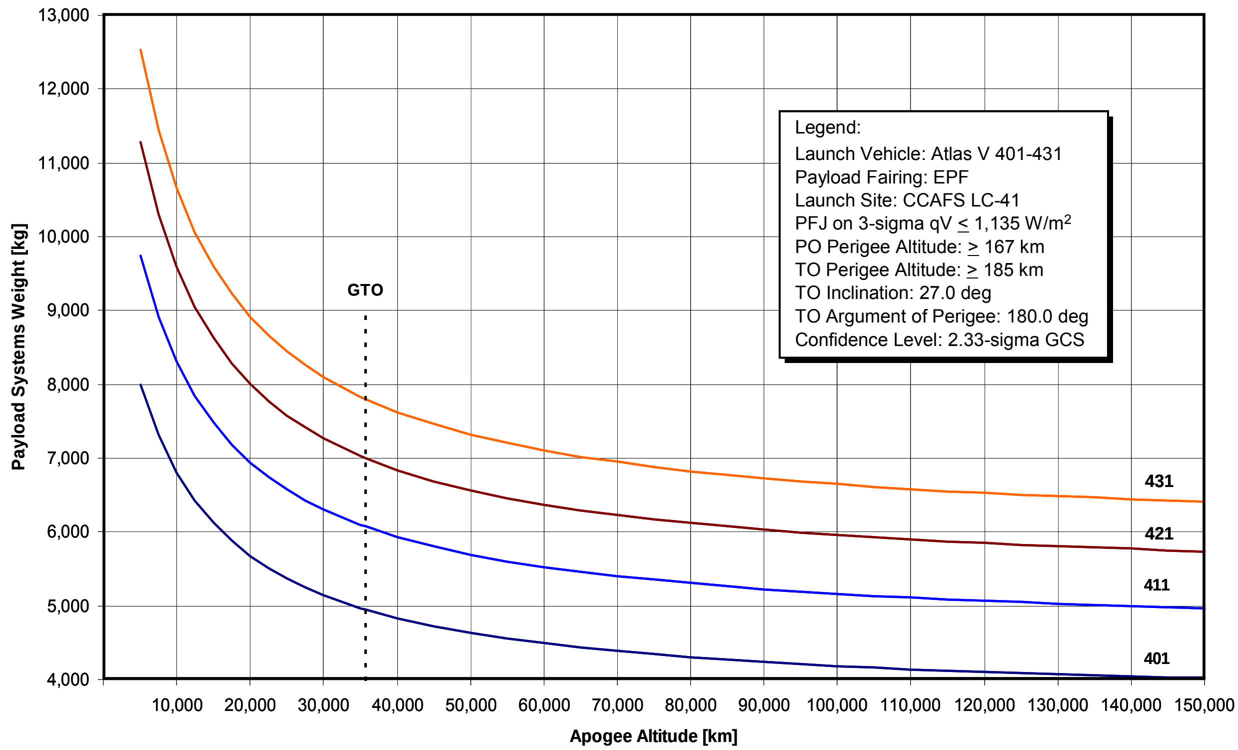


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Table 2.6.1-1: Atlas V 401 - 431 Geotransfer Orbit Performance – PSW vs Apogee Altitude (1 of 2)

Atlas V 401 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	7,990	17,615
7,500	4,050	7,310	16,116
10,000	5,400	6,805	15,002
12,500	6,749	6,420	14,153
15,000	8,099	6,120	13,492
17,500	9,449	5,880	12,963
20,000	10,799	5,680	12,522
22,500	12,149	5,510	12,147
25,000	13,499	5,370	11,839
27,500	14,849	5,250	11,574
30,000	16,199	5,145	11,343
35,000	18,898	4,970	10,957
35,786	**19,323**	**4,950**	**10,913**
40,000	21,598	4,835	10,659
45,000	24,298	4,725	10,417
50,000	26,998	4,630	10,207
55,000	29,698	4,555	10,042
60,000	32,397	4,490	9,899
65,000	35,097	4,435	9,777
70,000	37,797	4,385	9,667
75,000	40,497	4,340	9,568
80,000	43,197	4,305	9,491
85,000	45,896	4,270	9,414
90,000	48,596	4,240	9,347
95,000	51,296	4,210	9,281
100,000	53,996	4,185	9,226
105,000	56,695	4,165	9,182
110,000	59,395	4,140	9,127
115,000	62,095	4,125	9,094
120,000	64,795	4,105	9,050
125,000	67,495	4,090	9,017
130,000	70,194	4,075	8,984
135,000	72,894	4,060	8,951
140,000	75,594	4,045	8,918
145,000	78,294	4,035	8,896
150,000	80,994	4,025	8,873

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 411 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	9,750	21,495
7,500	4,050	8,920	19,665
10,000	5,400	8,310	18,320
12,500	6,749	7,845	17,295
15,000	8,099	7,480	16,490
17,500	9,449	7,185	15,840
20,000	10,799	6,945	15,311
22,500	12,149	6,745	14,870
25,000	13,499	6,575	14,495
27,500	14,849	6,430	14,175
30,000	16,199	6,310	13,911
35,000	18,898	6,100	13,448
35,786	**19,323**	**6,075**	**13,393**
40,000	21,598	5,935	13,084
45,000	24,298	5,805	12,798
50,000	26,998	5,695	12,555
55,000	29,698	5,605	12,357
60,000	32,397	5,525	12,180
65,000	35,097	5,460	12,037
70,000	37,797	5,400	11,905
75,000	40,497	5,350	11,795
80,000	43,197	5,305	11,695
85,000	45,896	5,265	11,607
90,000	48,596	5,225	11,519
95,000	51,296	5,195	11,453
100,000	53,996	5,165	11,387
105,000	56,695	5,135	11,321
110,000	59,395	5,110	11,265
115,000	62,095	5,090	11,221
120,000	64,795	5,065	11,166
125,000	67,495	5,050	11,133
130,000	70,194	5,030	11,089
135,000	72,894	5,015	11,056
140,000	75,594	4,995	11,012
145,000	78,294	4,985	10,990
150,000	80,994	4,970	10,957

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.1-1: Atlas V 401 - 431 Geotransfer Orbit Performance – PSW vs Apogee Altitude (2 of 2)

Atlas V 421 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	11,275	24,857
7,500	4,050	10,300	22,707
10,000	5,400	9,590	21,142
12,500	6,749	9,050	19,951
15,000	8,099	8,625	19,015
17,500	9,449	8,285	18,265
20,000	10,799	8,005	17,648
22,500	12,149	7,775	17,141
25,000	13,499	7,580	16,711
27,500	14,849	7,415	16,347
30,000	16,199	7,270	16,027
35,000	18,898	7,030	15,498
35,786	**19,323**	**7,000**	**15,432**
40,000	21,598	6,840	15,079
45,000	24,298	6,690	14,749
50,000	26,998	6,565	14,473
55,000	29,698	6,460	14,242
60,000	32,397	6,370	14,043
65,000	35,097	6,295	13,878
70,000	37,797	6,230	13,735
75,000	40,497	6,170	13,602
80,000	43,197	6,120	13,492
85,000	45,896	6,075	13,393
90,000	48,596	6,035	13,305
95,000	51,296	5,995	13,216
100,000	53,996	5,960	13,139
105,000	56,695	5,930	13,073
110,000	59,395	5,900	13,007
115,000	62,095	5,875	12,952
120,000	64,795	5,855	12,908
125,000	67,495	5,830	12,853
130,000	70,194	5,810	12,809
135,000	72,894	5,790	12,765
140,000	75,594	5,775	12,731
145,000	78,294	5,755	12,687
150,000	80,994	5,740	12,654

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 431 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	12,530	27,623
7,500	4,050	11,450	25,243
10,000	5,400	10,665	23,512
12,500	6,749	10,065	22,189
15,000	8,099	9,600	21,164
17,500	9,449	9,225	20,337
20,000	10,799	8,915	19,654
22,500	12,149	8,660	19,092
25,000	13,499	8,445	18,618
27,500	14,849	8,260	18,210
30,000	16,199	8,095	17,846
35,000	18,898	7,835	17,273
35,786	**19,323**	**7,800**	**17,196**
40,000	21,598	7,625	16,810
45,000	24,298	7,460	16,446
50,000	26,998	7,320	16,138
55,000	29,698	7,205	15,884
60,000	32,397	7,105	15,664
65,000	35,097	7,020	15,476
70,000	37,797	6,950	15,322
75,000	40,497	6,885	15,179
80,000	43,197	6,825	15,046
85,000	45,896	6,775	14,936
90,000	48,596	6,730	14,837
95,000	51,296	6,690	14,749
100,000	53,996	6,650	14,660
105,000	56,695	6,615	14,583
110,000	59,395	6,585	14,517
115,000	62,095	6,555	14,451
120,000	64,795	6,530	14,396
125,000	67,495	6,505	14,341
130,000	70,194	6,485	14,297
135,000	72,894	6,465	14,253
140,000	75,594	6,445	14,209
145,000	78,294	6,425	14,164
150,000	80,994	6,410	14,131

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.1-2: Atlas V 501-551 Apogee Variation Performance to Geotransfer Orbit

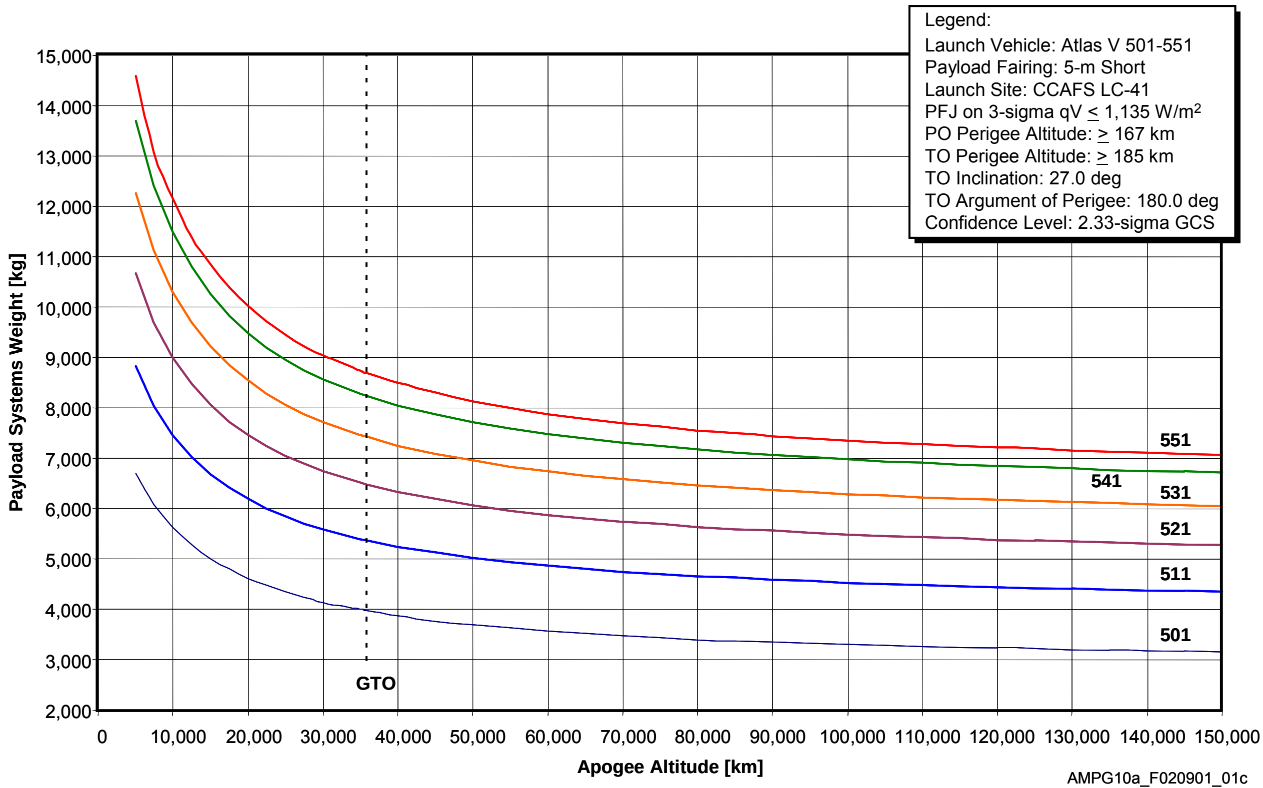

Table 2.6.1-2: Atlas V 501 – 551 Geotransfer Orbit Performance - PSW vs Apogee Altitude (1 of 3)

Atlas V 501 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	6,705	14,782
7,500	4,050	6,080	13,404
10,000	5,400	5,625	12,401
12,500	6,749	5,280	11,640
15,000	8,099	5,010	11,045
17,500	9,449	4,795	10,571
20,000	10,799	4,615	10,174
22,500	12,149	4,470	9,854
25,000	13,499	4,345	9,579
27,500	14,849	4,235	9,336
30,000	16,199	4,140	9,127
35,000	18,898	3,990	8,796
35,786	**19,323**	**3,970**	**8,752**
40,000	21,598	3,870	8,532
45,000	24,298	3,770	8,311
50,000	26,998	3,690	8,135
55,000	29,698	3,625	7,992
60,000	32,397	3,565	7,859
65,000	35,097	3,515	7,749
70,000	37,797	3,470	7,650
75,000	40,497	3,435	7,573
80,000	43,197	3,400	7,496
85,000	45,896	3,370	7,429
90,000	48,596	3,345	7,374
95,000	51,296	3,320	7,319
100,000	53,996	3,300	7,275
105,000	56,695	3,280	7,231
110,000	59,395	3,260	7,187
115,000	62,095	3,245	7,154
120,000	64,795	3,230	7,121
125,000	67,495	3,215	7,088
130,000	70,194	3,200	7,055
135,000	72,894	3,190	7,033
140,000	75,594	3,175	7,000
145,000	78,294	3,165	6,978
150,000	80,994	3,155	6,955

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 511 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	8,825	19,455
7,500	4,050	8,035	17,714
10,000	5,400	7,460	16,446
12,500	6,749	7,025	15,487
15,000	8,099	6,680	14,727
17,500	9,449	6,410	14,131
20,000	10,799	6,185	13,635
22,500	12,149	5,995	13,216
25,000	13,499	5,840	12,875
27,500	14,849	5,705	12,577
30,000	16,199	5,585	12,313
35,000	18,898	5,395	11,894
35,786	**19,323**	**5,370**	**11,839**
40,000	21,598	5,240	11,552
45,000	24,298	5,120	11,287
50,000	26,998	5,015	11,056
55,000	29,698	4,935	10,880
60,000	32,397	4,860	10,714
65,000	35,097	4,800	10,582
70,000	37,797	4,745	10,461
75,000	40,497	4,695	10,351
80,000	43,197	4,655	10,262
85,000	45,896	4,620	10,185
90,000	48,596	4,585	10,108
95,000	51,296	4,555	10,042
100,000	53,996	4,525	9,976
105,000	56,695	4,500	9,921
110,000	59,395	4,480	9,877
115,000	62,095	4,460	9,832
120,000	64,795	4,440	9,788
125,000	67,495	4,420	9,744
130,000	70,194	4,405	9,711
135,000	72,894	4,390	9,678
140,000	75,594	4,375	9,645
145,000	78,294	4,360	9,612
150,000	80,994	4,350	9,590

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.1-2: Atlas V 501 – 551 Geotransfer Orbit Performance - PSW vs Apogee Altitude (2 of 3)

Atlas V 521 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	10,670	23,523
7,500	4,050	9,700	21,384
10,000	5,400	9,000	19,841
12,500	6,749	8,470	18,673
15,000	8,099	8,055	17,758
17,500	9,449	7,715	17,008
20,000	10,799	7,460	16,446
22,500	12,149	7,235	15,950
25,000	13,499	7,045	15,531
27,500	14,849	6,885	15,179
30,000	16,199	6,745	14,870
35,000	18,898	6,515	14,363
35,786	**19,323**	**6,485**	**14,297**
40,000	21,598	6,335	13,966
45,000	24,298	6,185	13,635
50,000	26,998	6,065	13,371
55,000	29,698	5,965	13,150
60,000	32,397	5,880	12,963
65,000	35,097	5,805	12,798
70,000	37,797	5,745	12,665
75,000	40,497	5,685	12,533
80,000	43,197	5,635	12,423
85,000	45,896	5,595	12,335
90,000	48,596	5,555	12,246
95,000	51,296	5,515	12,158
100,000	53,996	5,485	12,092
105,000	56,695	5,455	12,026
110,000	59,395	5,430	11,971
115,000	62,095	5,405	11,916
120,000	64,795	5,380	11,861
125,000	67,495	5,360	11,817
130,000	70,194	5,340	11,772
135,000	72,894	5,320	11,728
140,000	75,594	5,305	11,695
145,000	78,294	5,290	11,662
150,000	80,994	5,275	11,629

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 531 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	12,260	27,028
7,500	4,050	11,130	24,537
10,000	5,400	10,315	22,740
12,500	6,749	9,700	21,384
15,000	8,099	9,225	20,337
17,500	9,449	8,845	19,500
20,000	10,799	8,535	18,816
22,500	12,149	8,280	18,254
25,000	13,499	8,060	17,769
27,500	14,849	7,875	17,361
30,000	16,199	7,715	17,008
35,000	18,898	7,460	16,446
35,786	**19,323**	**7,425**	**16,369**
40,000	21,598	7,250	15,983
45,000	24,298	7,085	15,619
50,000	26,998	6,950	15,322
55,000	29,698	6,835	15,068
60,000	32,397	6,740	14,859
65,000	35,097	6,655	14,672
70,000	37,797	6,580	14,506
75,000	40,497	6,520	14,374
80,000	43,197	6,460	14,242
85,000	45,896	6,410	14,131
90,000	48,596	6,365	14,032
95,000	51,296	6,325	13,944
100,000	53,996	6,290	13,867
105,000	56,695	6,255	13,790
110,000	59,395	6,225	13,724
115,000	62,095	6,195	13,657
120,000	64,795	6,170	13,602
125,000	67,495	6,145	13,547
130,000	70,194	6,125	13,503
135,000	72,894	6,105	13,459
140,000	75,594	6,085	13,415
145,000	78,294	6,065	13,371
150,000	80,994	6,050	13,338

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.1-2: Atlas V 501 – 551 Geotransfer Orbit Performance - PSW vs Apogee Altitude (3 of 3)

Atlas V 541 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	13,700	30,203
7,500	4,050	12,415	27,370
10,000	5,400	11,490	25,331
12,500	6,749	10,795	23,799
15,000	8,099	10,260	22,619
17,500	9,449	9,835	21,682
20,000	10,799	9,485	20,910
22,500	12,149	9,200	20,282
25,000	13,499	8,955	19,742
27,500	14,849	8,750	19,290
30,000	16,199	8,570	18,893
35,000	18,898	8,275	18,243
35,786	**19,323**	**8,240**	**18,166**
40,000	21,598	8,045	17,736
45,000	24,298	7,860	17,328
50,000	26,998	7,710	16,997
55,000	29,698	7,580	16,711
60,000	32,397	7,470	16,468
65,000	35,097	7,385	16,281
70,000	37,797	7,300	16,093
75,000	40,497	7,230	15,939
80,000	43,197	7,170	15,807
85,000	45,896	7,110	15,675
90,000	48,596	7,060	15,564
95,000	51,296	7,015	15,465
100,000	53,996	6,975	15,377
105,000	56,695	6,940	15,300
110,000	59,395	6,905	15,223
115,000	62,095	6,875	15,157
120,000	64,795	6,845	15,090
125,000	67,495	6,820	15,035
130,000	70,194	6,795	14,980
135,000	72,894	6,770	14,925
140,000	75,594	6,750	14,881
145,000	78,294	6,730	14,837
150,000	80,994	6,710	14,793

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 551 GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
5,000	2,700	14,580	32,143
7,500	4,050	13,085	28,847
10,000	5,400	12,175	26,841
12,500	6,749	11,400	25,132
15,000	8,099	10,855	23,931
17,500	9,449	10,385	22,895
20,000	10,799	10,030	22,112
22,500	12,149	9,710	21,407
25,000	13,499	9,460	20,855
27,500	14,849	9,225	20,337
30,000	16,199	9,045	19,940
35,000	18,898	8,740	19,268
35,786	**19,323**	**8,700**	**19,180**
40,000	21,598	8,490	18,717
45,000	24,298	8,295	18,287
50,000	26,998	8,130	17,923
55,000	29,698	7,995	17,626
60,000	32,397	7,875	17,361
65,000	35,097	7,780	17,152
70,000	37,797	7,695	16,964
75,000	40,497	7,620	16,799
80,000	43,197	7,545	16,634
85,000	45,896	7,495	16,523
90,000	48,596	7,435	16,391
95,000	51,296	7,395	16,303
100,000	53,996	7,350	16,204
105,000	56,695	7,310	16,116
110,000	59,395	7,275	16,038
115,000	62,095	7,240	15,961
120,000	64,795	7,210	15,895
125,000	67,495	7,185	15,840
130,000	70,194	7,155	15,774
135,000	72,894	7,135	15,730
140,000	75,594	7,110	15,675
145,000	78,294	7,090	15,631
150,000	80,994	7,070	15,586

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.1-3: Atlas V HLV Performance to Geotransfer Orbit

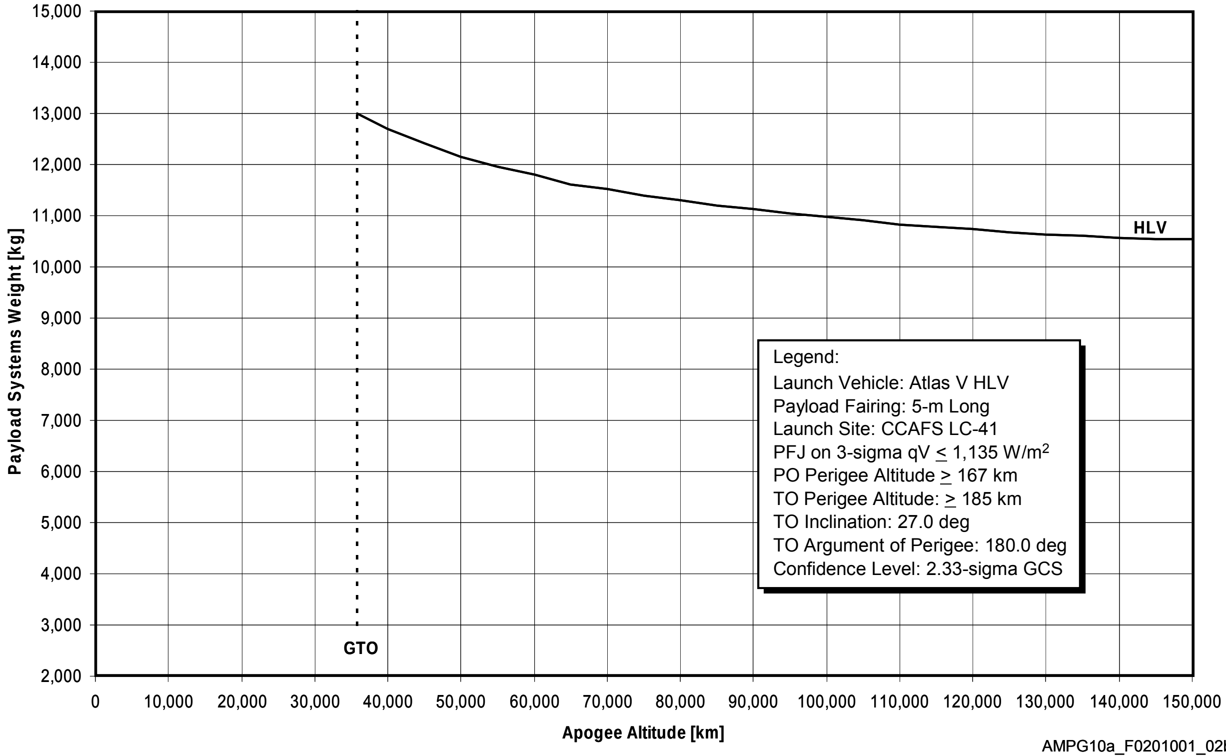

Table 2.6.1-3: Atlas V HLV Geotransfer Orbit Performance PSW vs Apogee Altitude

Atlas V HLV GTO - PSW vs. Apogee Altitude

Apogee Altitude		Payload Systems Weight	
[km]	[nmi]	[kg]	[lb]
35,786	**19,323**	**13,000**	**28,660**
40,000	21,598	12,705	28,009
45,000	24,298	12,410	27,359
50,000	26,998	12,155	26,797
55,000	29,698	11,960	26,367
60,000	32,397	11,795	26,003
65,000	35,097	11,615	25,606
70,000	37,797	11,515	25,386
75,000	40,497	11,400	25,132
80,000	43,197	11,295	24,901
85,000	45,896	11,205	24,702
90,000	48,596	11,125	24,526
95,000	51,296	11,045	24,350
100,000	53,996	10,985	24,217
105,000	56,695	10,905	24,041
110,000	59,395	10,830	23,876
115,000	62,095	10,780	23,765
120,000	64,795	10,740	23,677
125,000	67,495	10,665	23,512
130,000	70,194	10,635	23,446
135,000	72,894	10,600	23,369
140,000	75,594	10,560	23,280
145,000	78,294	10,550	23,258
150,000	80,994	10,540	23,236

Notes:
Launch Site: CCAFS LC-41
5-m Long PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Inclination = 27.0 deg
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.2-1: Atlas V 401 - 431 Reduced Inclination Performance to Geotransfer Orbit - CCAFS

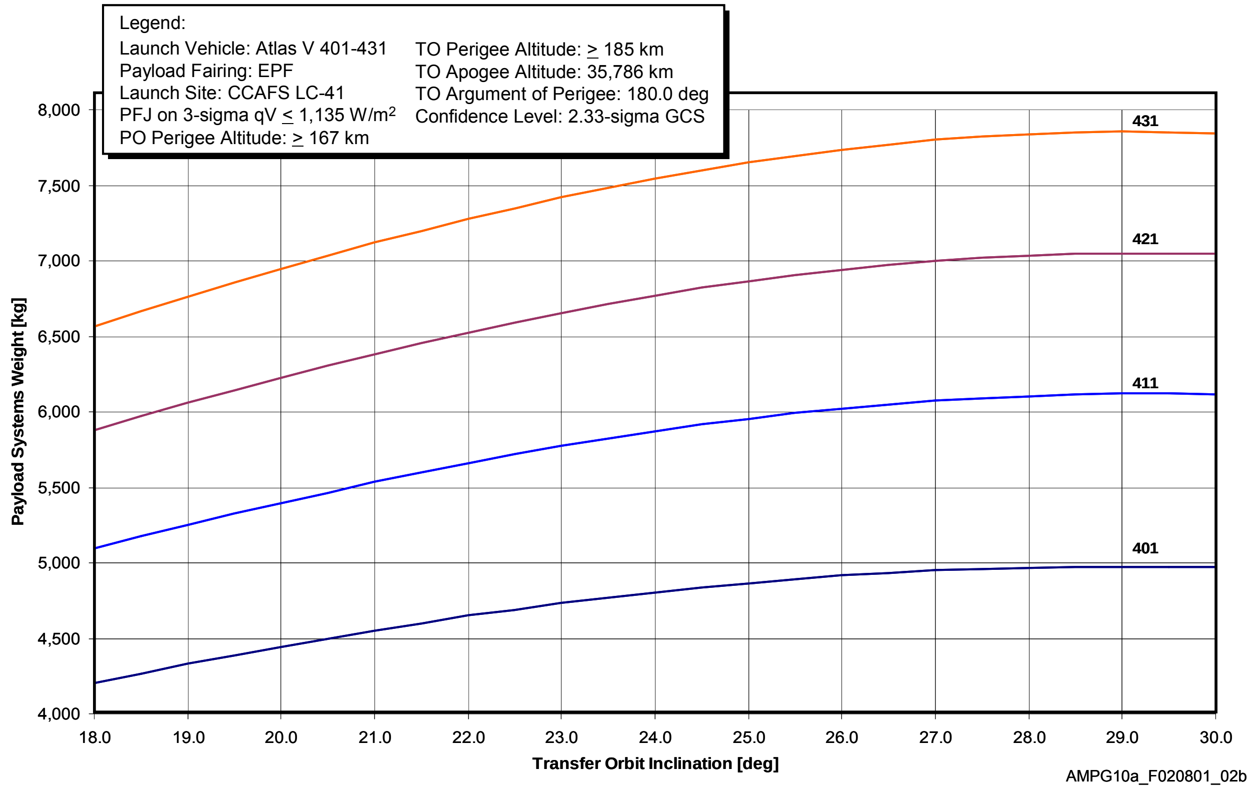

Table 2.6.2-1: Atlas V 401 - 431 Geotransfer Orbit Performance – PSW vs Orbit Inclination (1 of 2)

Atlas V 401 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight	
	[kg]	[lb]
30.0	4,975	10,968
29.5	4,975	10,968
29.0	4,975	10,968
28.5	4,970	10,957
28.0	4,965	10,946
27.5	4,960	10,935
27.0	**4,950**	**10,913**
26.5	4,935	10,880
26.0	4,915	10,836
25.5	4,890	10,780
25.0	4,865	10,725
24.5	4,840	10,670
24.0	4,805	10,593
23.5	4,770	10,516
23.0	4,735	10,439
22.5	4,690	10,340
22.0	4,650	10,251
21.5	4,600	10,141
21.0	4,550	10,031
20.5	4,500	9,921
20.0	4,445	9,799
19.5	4,390	9,678
19.0	4,330	9,546
18.5	4,265	9,403
18.0	4,205	9,270

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 411 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight	
	[kg]	[lb]
30.0	6,115	13,481
29.5	6,120	13,492
29.0	6,120	13,492
28.5	6,115	13,481
28.0	6,105	13,459
27.5	6,090	13,426
27.0	**6,075**	**13,393**
26.5	6,050	13,338
26.0	6,020	13,272
25.5	5,990	13,205
25.0	5,955	13,128
24.5	5,915	13,040
24.0	5,870	12,941
23.5	5,825	12,842
23.0	5,775	12,731
22.5	5,720	12,610
22.0	5,660	12,478
21.5	5,600	12,346
21.0	5,535	12,202
20.5	5,465	12,048
20.0	5,395	11,894
19.5	5,325	11,739
19.0	5,250	11,574
18.5	5,175	11,409
18.0	5,095	11,232

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.2-1: Atlas V 401 - 431 Geotransfer Orbit Performance – PSW vs Orbit Inclination (2 of 2)

Atlas V 421 GTO - PSW vs. Inclination

Inclination	Payload Systems Weight	
[deg]	[kg]	[lb]
30.0	7,045	15,531
29.5	7,050	15,542
29.0	7,050	15,542
28.5	7,045	15,531
28.0	7,035	15,509
27.5	7,020	15,476
27.0	**7,000**	**15,432**
26.5	6,970	15,366
26.0	6,940	15,300
25.5	6,905	15,223
25.0	6,865	15,134
24.5	6,820	15,035
24.0	6,770	14,925
23.5	6,715	14,804
23.0	6,655	14,672
22.5	6,590	14,528
22.0	6,525	14,385
21.5	6,455	14,231
21.0	6,380	14,065
20.5	6,305	13,900
20.0	6,225	13,724
19.5	6,145	13,547
19.0	6,060	13,360
18.5	5,970	13,161
18.0	5,880	12,963

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 431 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight	
	[kg]	[lb]
30.0	7,845	17,295
29.5	7,850	17,306
29.0	7,855	17,317
28.5	7,850	17,306
28.0	7,835	17,273
27.5	7,820	17,240
27.0	**7,800**	**17,196**
26.5	7,770	17,130
26.0	7,735	17,052
25.5	7,695	16,964
25.0	7,650	16,865
24.5	7,600	16,755
24.0	7,545	16,634
23.5	7,485	16,501
23.0	7,420	16,358
22.5	7,350	16,204
22.0	7,280	16,049
21.5	7,200	15,873
21.0	7,120	15,697
20.5	7,035	15,509
20.0	6,945	15,311
19.5	6,855	15,112
19.0	6,760	14,903
18.5	6,665	14,694
18.0	6,565	14,473

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.2-2: Atlas V 501-551 Reduced Inclination Performance to Geotransfer Orbit - CCAFS

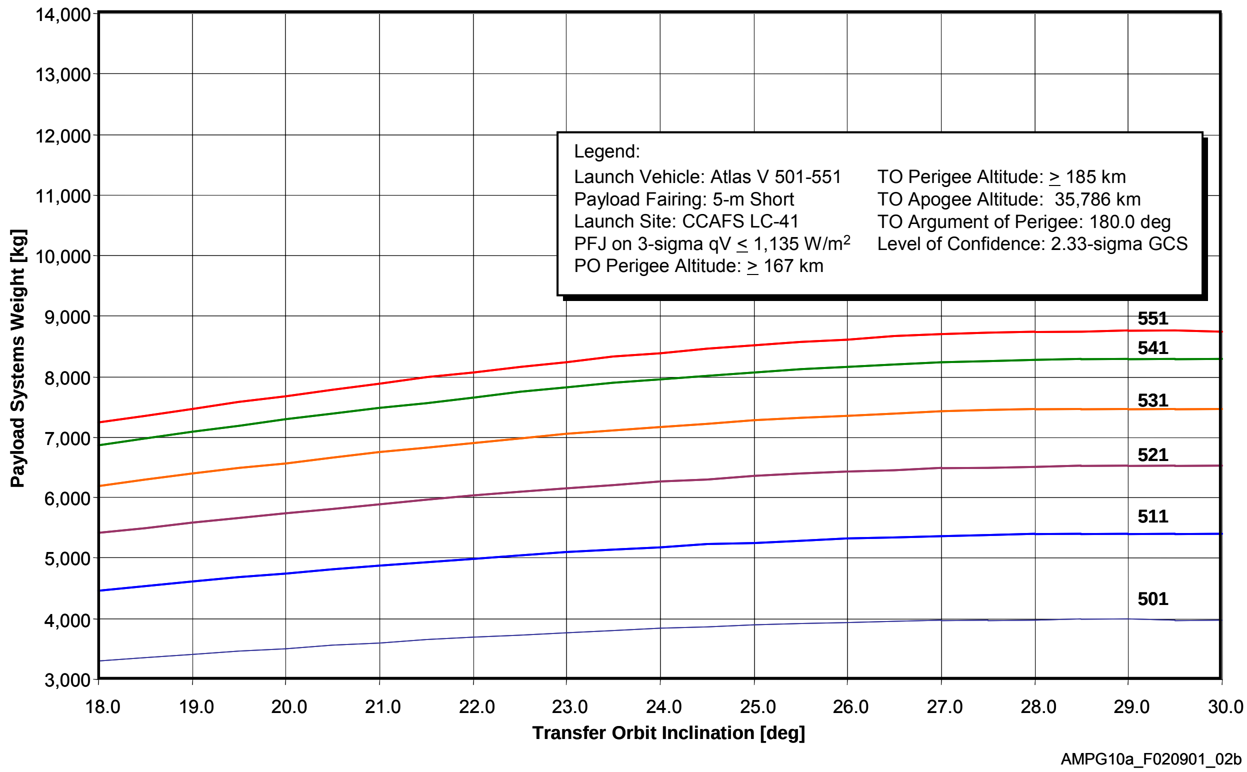

Table 2.6.2-2: Atlas V 501 - 551 Geotransfer Orbit Performance - PSW vs Orbit Inclination (1 of 3)

Atlas V 501 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight	
	[kg]	[lb]
30.0	3,980	8,774
29.5	3,985	8,785
29.0	3,990	8,796
28.5	3,990	8,796
28.0	3,985	8,785
27.5	3,975	8,763
27.0	**3,970**	**8,752**
26.5	3,955	8,719
26.0	3,940	8,686
25.5	3,920	8,642
25.0	3,895	8,587
24.5	3,870	8,532
24.0	3,840	8,466
23.5	3,810	8,399
23.0	3,775	8,322
22.5	3,740	8,245
22.0	3,700	8,157
21.5	3,660	8,069
21.0	3,600	7,937
20.5	3,570	7,870
20.0	3,515	7,749
19.5	3,470	7,650
19.0	3,415	7,529
18.5	3,360	7,407
18.0	3,305	7,286

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 511 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight	
	[kg]	[lb]
30.0	5,400	11,905
29.5	5,405	11,916
29.0	5,405	11,916
28.5	5,405	11,916
28.0	5,395	11,894
27.5	5,385	11,872
27.0	**5,370**	**11,839**
26.5	5,345	11,784
26.0	5,325	11,739
25.5	5,295	11,673
25.0	5,260	11,596
24.5	5,225	11,519
24.0	5,185	11,431
23.5	5,140	11,332
23.0	5,095	11,232
22.5	5,045	11,122
22.0	4,990	11,001
21.5	4,935	10,880
21.0	4,875	10,747
20.5	4,815	10,615
20.0	4,750	10,472
19.5	4,685	10,328
19.0	4,615	10,174
18.5	4,545	10,020
18.0	4,470	9,854

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.2-2: Atlas V 501 - 551 Geotransfer Orbit Performance - PSW vs Orbit Inclination (2 of 3)

Atlas V 521 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight	
	[kg]	[lb]
30.0	6,520	14,374
29.5	6,530	14,396
29.0	6,530	14,396
28.5	6,525	14,385
28.0	6,515	14,363
27.5	6,500	14,330
27.0	**6,485**	**14,297**
26.5	6,455	14,231
26.0	6,430	14,175
25.5	6,395	14,098
25.0	6,355	14,010
24.5	6,310	13,911
24.0	6,265	13,812
23.5	6,210	13,690
23.0	6,155	13,569
22.5	6,095	13,437
22.0	6,035	13,305
21.5	5,965	13,150
21.0	5,895	12,996
20.5	5,820	12,831
20.0	5,745	12,665
19.5	5,665	12,489
19.0	5,585	12,313
18.5	5,505	12,136
18.0	5,415	11,938

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 531 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight	
	[kg]	[lb]
30.0	7,470	16,468
29.5	7,475	16,479
29.0	7,475	16,479
28.5	7,470	16,468
28.0	7,460	16,446
27.5	7,445	16,413
27.0	**7,425**	**16,369**
26.5	7,395	16,303
26.0	7,360	16,226
25.5	7,320	16,138
25.0	7,275	16,038
24.5	7,225	15,928
24.0	7,170	15,807
23.5	7,110	15,675
23.0	7,050	15,542
22.5	6,980	15,388
22.0	6,905	15,223
21.5	6,830	15,057
21.0	6,750	14,881
20.5	6,665	14,694
20.0	6,575	14,495
19.5	6,485	14,297
19.0	6,395	14,098
18.5	6,300	13,889
18.0	6,200	13,668

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.2-2: Atlas V 501 - 551 Geotransfer Orbit Performance - PSW vs Orbit Inclination (3 of 3)

Atlas V 541 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight [kg]	Payload Systems Weight [lb]
30.0	8,290	18,276
29.5	8,295	18,287
29.0	8,300	18,298
28.5	8,295	18,287
28.0	8,280	18,254
27.5	8,260	18,210
27.0	**8,240**	**18,166**
26.5	8,205	18,089
26.0	8,170	18,011
25.5	8,125	17,912
25.0	8,075	17,802
24.5	8,015	17,670
24.0	7,955	17,537
23.5	7,890	17,394
23.0	7,815	17,229
22.5	7,740	17,063
22.0	7,655	16,876
21.5	7,570	16,689
21.0	7,480	16,490
20.5	7,385	16,281
20.0	7,290	16,071
19.5	7,190	15,851
19.0	7,085	15,619
18.5	6,975	15,377
18.0	6,865	15,134

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 551 GTO - PSW vs. Inclination

Inclination [deg]	Payload Systems Weight [kg]	Payload Systems Weight [lb]
30.0	8,745	19,279
29.5	8,755	19,301
29.0	8,755	19,301
28.5	8,750	19,290
28.0	8,740	19,268
27.5	8,720	19,224
27.0	**8,700**	**19,180**
26.5	8,660	19,092
26.0	8,620	19,004
25.5	8,575	18,904
25.0	8,520	18,783
24.5	8,460	18,651
24.0	8,395	18,507
23.5	8,325	18,353
23.0	8,245	18,177
22.5	8,165	18,000
22.0	8,075	17,802
21.5	7,985	17,604
21.0	7,885	17,383
20.5	7,785	17,163
20.0	7,680	16,931
19.5	7,575	16,700
19.0	7,465	16,457
18.5	7,350	16,204
18.0	7,235	15,950

Notes:
Launch Site: CCAFS LC-41
PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.2-3: Atlas V HLV Reduced Inclination Performance to Geotransfer Orbit - CCAFS

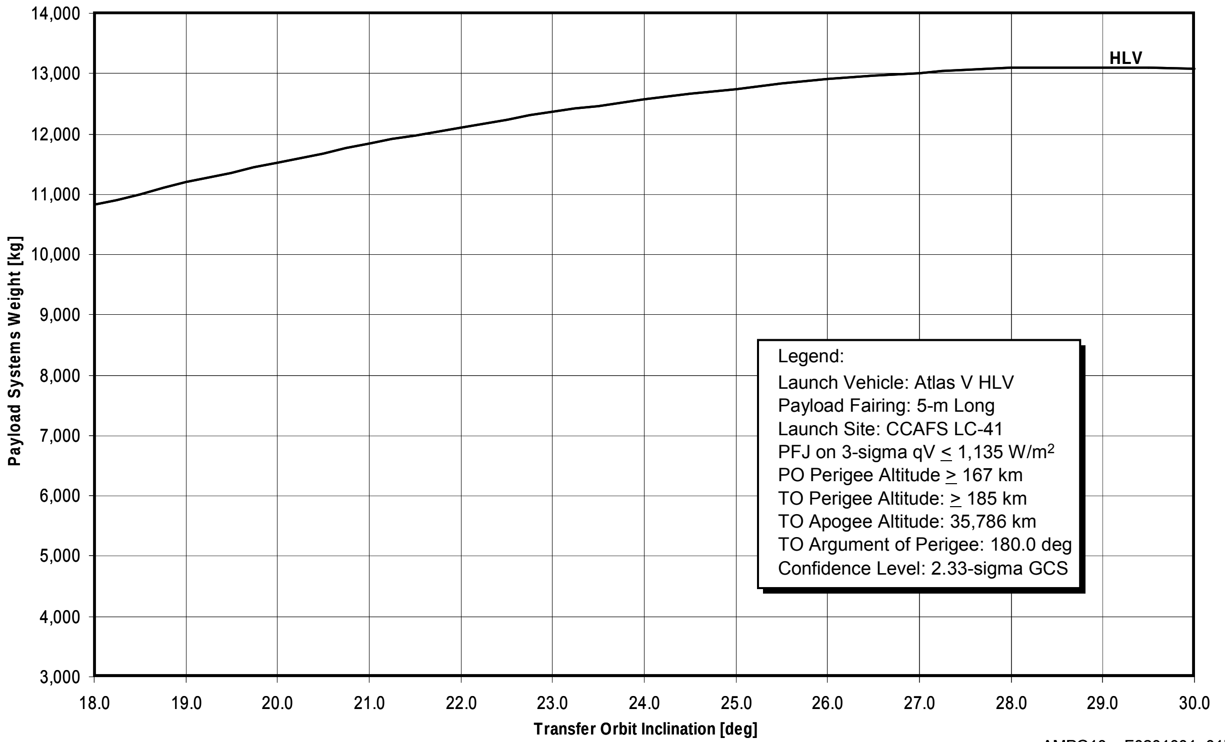

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Table 2.6.2-3: Atlas V HLV Geotransfer Orbit Performance PSW vs Orbit Inclination

Atlas V HLV GTO - PSW vs. Inclination		
Inclination [deg]	**Payload Systems Weight**	
	[kg]	**[lb]**
30.0	13,080	28,836
29.5	13,095	28,869
29.0	13,105	28,891
28.5	13,100	28,880
28.0	13,090	28,858
27.5	13,055	28,781
27.0	**13,000**	**28,660**
26.5	12,965	28,582
26.0	12,905	28,450
25.5	12,830	28,285
25.0	12,750	28,108
24.5	12,665	27,921
24.0	12,580	27,734
23.5	12,465	27,480
23.0	12,370	27,271
22.5	12,240	26,984
22.0	12,105	26,687
21.5	11,980	26,411
21.0	11,835	26,091
20.5	11,680	25,750
20.0	11,520	25,397
19.5	11,350	25,022
19.0	11,195	24,680
18.5	11,005	24,261
18.0	10,825	23,865

Notes:
Launch Site: CCAFS LC-41
5-m Long PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee.
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1: Atlas V 400 Series Minimum Delta-V to Geosynchronous Orbit Summary

Atlas V 400 Series – Delta-V to GSO at 35,786 km (19,323 nmi)								
	Payload Systems Weight							
ΔV to GSO	401 (Table 2.6.3-1a)		411 (Table 2.6.3-1b)		421 (Table 2.6.3-1c)		431 (Table 2.6.3-1d)	
[m/s]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
1,800	4,940	10,891	6,065	13,371	6,990	15,410	7,785	17,163
1,775	4,890	10,781	6,000	13,228	6,915	15,245	7,705	16,987
1,750	4,830	10,648	5,910	13,029	6,810	15,013	7,590	16,733
1,725	4,745	10,461	5,790	12,765	6,675	14,716	7,445	16,413
1,700	4,635	10,218	5,645	12,445	6,505	14,341	7,260	16,006
1,675	4,500	9,921	5,470	12,059	6,310	13,911	7,040	15,521
1,650	4,340	9,568	5,260	11,596	6,075	13,393	6,780	14,947
1,625	4,165	9,182	5,040	11,111	5,825	12,842	6,510	14,352
1,600	3,965	8,741	4,865	10,725	5,585	12,313	6,240	13,757
1,575	3,920	8,642	4,745	10,461	5,490	12,103	6,135	13,525
1,550	3,870	8,532	4,670	10,296	5,405	11,916	6,045	13,327
1,525	3,820	8,422	4,600	10,141	5,330	11,751	5,965	13,151
1,500	3,765	8,300	4,535	9,998	5,255	11,585	5,885	12,974
1,475	3,710	8,179	4,475	9,866	5,185	11,431	5,810	12,809
1,450	3,650	8,047	4,410	9,722	5,115	11,277	5,725	12,621
1,425	3,590	7,915	4,340	9,568	5,035	11,100	5,645	12,445
1,400	3,530	7,782	4,270	9,414	4,960	10,935	5,555	12,247
1,375	3,465	7,639	4,200	9,259	4,875	10,748	5,465	12,048
1,350	3,400	7,496	4,125	9,094	4,790	10,560	5,370	11,839
1,325	3,335	7,352	4,045	8,918	4,700	10,362	5,270	11,618
1,300	3,265	7,198	3,960	8,730	4,605	10,152	5,170	11,398
1,275	3,195	7,044	3,875	8,543	4,510	9,943	5,065	11,166
1,250	3,120	6,878	3,785	8,344	4,410	9,722	4,950	10,913
1,225	3,050	6,724	3,695	8,146	4,305	9,491	4,835	10,659
1,200	2,975	6,559	3,600	7,937	4,200	9,259	4,720	10,406
1,175	2,900	6,393	3,500	7,716	4,090	9,017	4,600	10,141
1,150	2,820	6,217	3,400	7,496	3,975	8,763	4,475	9,866
1,125	2,745	6,052	3,295	7,264	3,860	8,510	4,355	9,601
1,100	2,665	5,875	3,195	7,044	3,745	8,256	4,225	9,315
1,075	2,585	5,699	3,090	6,812	3,630	8,003	4,100	9,039
1,050	2,505	5,523	2,980	6,570	3,510	7,738	3,970	8,752
1,025	2,420	5,335	2,875	6,338	3,395	7,485	3,845	8,477
1,000	2,340	5,159	2,770	6,107	3,275	7,220	3,715	8,190

Figure 2.6.3-1a: Atlas V 401 Minimum Delta-V to Geosynchronous Orbit

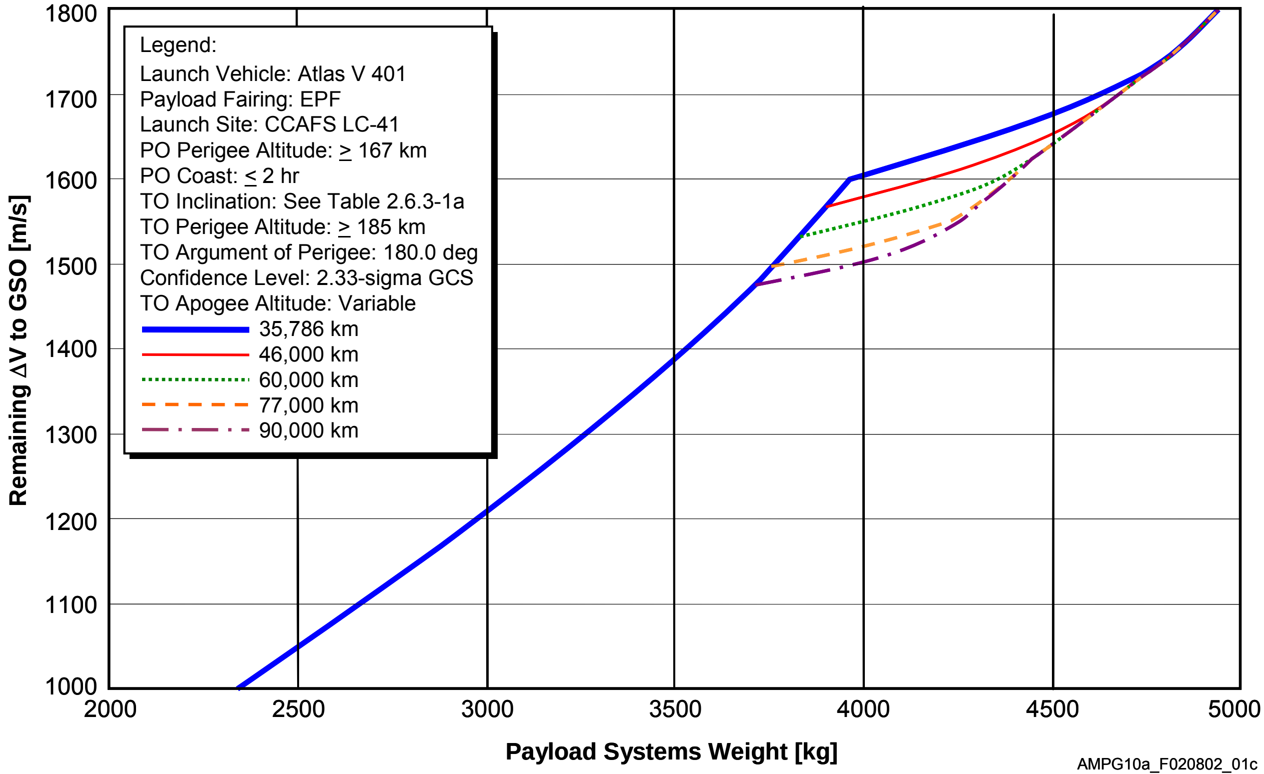

Table 2.6.3-1a: Atlas V 401 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 401 - DV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	190	103	26.8	26.3	4,940	10,891
1,775	214	116	25.7	26.4	4,890	10,781
1,750	217	117	24.5	26.7	4,830	10,648
1,725	230	124	23.3	26.9	4,745	10,461
1,700	233	126	22.0	27.3	4,635	10,218
1,675	240	130	20.7	28.0	4,500	9,921
1,650	291	157	19.6	30.0	4,340	9,568
1,625	501	271	19.1	37.0	4,165	9,182
1,600	4,089	2,208	26.7	135.1	3,965	8,741
1,575	4,624	2,497	26.5	134.0	3,920	8,642
1,550	4,989	2,694	26.1	133.0	3,870	8,532
1,525	5,158	2,785	25.5	131.7	3,820	8,422
1,500	5,146	2,779	24.6	130.4	3,765	8,300
1,475	5,487	2,963	24.2	129.0	3,710	8,179
1,450	5,823	3,144	23.8	127.8	3,650	8,047
1,425	6,143	3,317	23.3	126.3	3,590	7,915
1,400	6,456	3,486	22.8	125.0	3,530	7,782
1,375	6,771	3,656	22.4	123.4	3,465	7,639
1,350	7,068	3,816	21.9	122.0	3,400	7,496
1,325	7,355	3,971	21.4	120.5	3,335	7,352
1,300	7,643	4,127	20.9	119.0	3,265	7,198
1,275	7,922	4,278	20.4	117.6	3,195	7,044
1,250	8,190	4,422	19.8	116.1	3,120	6,878
1,225	8,458	4,567	19.3	114.6	3,050	6,724
1,200	8,727	4,712	18.8	113.0	2,975	6,559
1,175	8,986	4,852	18.3	111.5	2,900	6,393
1,150	9,246	4,992	17.7	110.0	2,820	6,217
1,125	9,496	5,127	17.2	108.5	2,745	6,052
1,100	9,758	5,269	16.7	107.0	2,665	5,875
1,075	10,021	5,411	16.1	105.5	2,585	5,699
1,050	10,275	5,548	15.6	103.9	2,505	5,523
1,025	10,541	5,692	15.1	102.5	2,420	5,335
1,000	10,810	5,837	14.5	100.9	2,340	5,159

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1a: Atlas V 401 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 401 - DV to GSO at 46,000 km (24,838 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.3	4,940	10,891
1,775	35,786	19,323	214	116	25.7	26.4	4,890	10,781
1,750	35,786	19,323	217	117	24.5	26.7	4,830	10,648
1,725	39,025	21,072	208	112	24.7	26.7	4,755	10,483
1,700	43,040	23,240	207	112	25.0	26.9	4,675	10,307
1,675	46,000	24,838	209	113	24.7	27.1	4,600	10,141
1,650	46,000	24,838	219	118	23.1	27.5	4,495	9,910
1,625	46,000	24,838	253	136	21.6	28.9	4,355	9,601
1,600	46,000	24,838	300	162	20.0	30.4	4,175	9,204
1,575	46,000	24,838	719	388	20.2	43.8	3,975	8,763

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 401 - DV to GSO at 60,000 km (32,397 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.3	4,940	10,891
1,775	35,786	19,323	214	116	25.7	26.4	4,890	10,781
1,750	35,786	19,323	217	117	24.5	26.7	4,830	10,648
1,725	39,025	21,072	208	112	24.7	26.7	4,755	10,483
1,700	43,040	23,240	207	112	25.0	26.9	4,675	10,307
1,675	47,671	25,740	207	112	25.3	27.0	4,605	10,152
1,650	53,124	28,685	206	111	25.7	27.1	4,530	9,987
1,625	59,263	32,000	212	115	26.0	27.5	4,455	9,822
1,600	60,000	32,397	220	119	24.3	27.8	4,365	9,623
1,575	60,000	32,397	232	125	22.2	28.4	4,215	9,292
1,550	60,000	32,397	358	193	20.5	32.5	4,005	8,830
1,525	60,000	32,397	977	528	21.3	51.0	3,780	8,333

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1a: Atlas V 401 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 401 - DV to GSO at 77,000 km (41,577 nmi) Cap								
∆V to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.3	4,940	10,891
1,775	35,786	19,323	214	116	25.7	26.4	4,890	10,781
1,750	35,786	19,323	217	117	24.5	26.7	4,830	10,648
1,725	39,025	21,072	208	112	24.7	26.7	4,755	10,483
1,700	43,040	23,240	207	112	25.0	26.9	4,675	10,307
1,675	47,671	25,740	207	112	25.3	27.0	4,605	10,152
1,650	53,124	28,685	206	111	25.7	27.1	4,530	9,987
1,625	59,263	32,000	212	115	26.0	27.5	4,455	9,822
1,600	66,662	35,995	204	110	26.3	27.2	4,390	9,678
1,575	75,137	40,571	203	109	26.6	27.3	4,320	9,524
1,550	77,000	41,577	211	114	24.7	27.7	4,230	9,326
1,525	77,000	41,577	239	129	22.1	28.7	4,055	8,940
1,500	77,000	41,577	488	264	20.5	36.3	3,795	8,367
1,475	77,000	41,577	521	281	17.6	32.1	3,450	7,606

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 401 - DV to GSO at 90,000 km (48,596 nmi) Cap								
∆V to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.3	4,940	10,891
1,775	35,786	19,323	214	116	25.7	26.4	4,890	10,781
1,750	35,786	19,323	217	117	24.5	26.7	4,830	10,648
1,725	39,025	21,072	208	112	24.7	26.7	4,755	10,483
1,700	43,040	23,240	207	112	25.0	26.9	4,675	10,307
1,675	47,671	25,740	207	112	25.3	27.0	4,605	10,152
1,650	53,124	28,685	206	111	25.7	27.1	4,530	9,987
1,625	59,263	32,000	212	115	26.0	27.5	4,455	9,822
1,600	66,662	35,995	204	110	26.3	27.2	4,390	9,678
1,575	75,137	40,571	203	109	26.6	27.3	4,320	9,524
1,550	85,107	45,954	202	109	26.9	27.4	4,255	9,381
1,525	90,000	48,596	208	112	25.4	27.7	4,175	9,204
1,500	90,000	48,596	235	127	22.4	28.7	3,995	8,807
1,475	90,000	48,596	537	290	20.7	37.6	3,705	8,168

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.3-1b: Atlas V 411 Minimum Delta-V to Geosynchronous Orbit

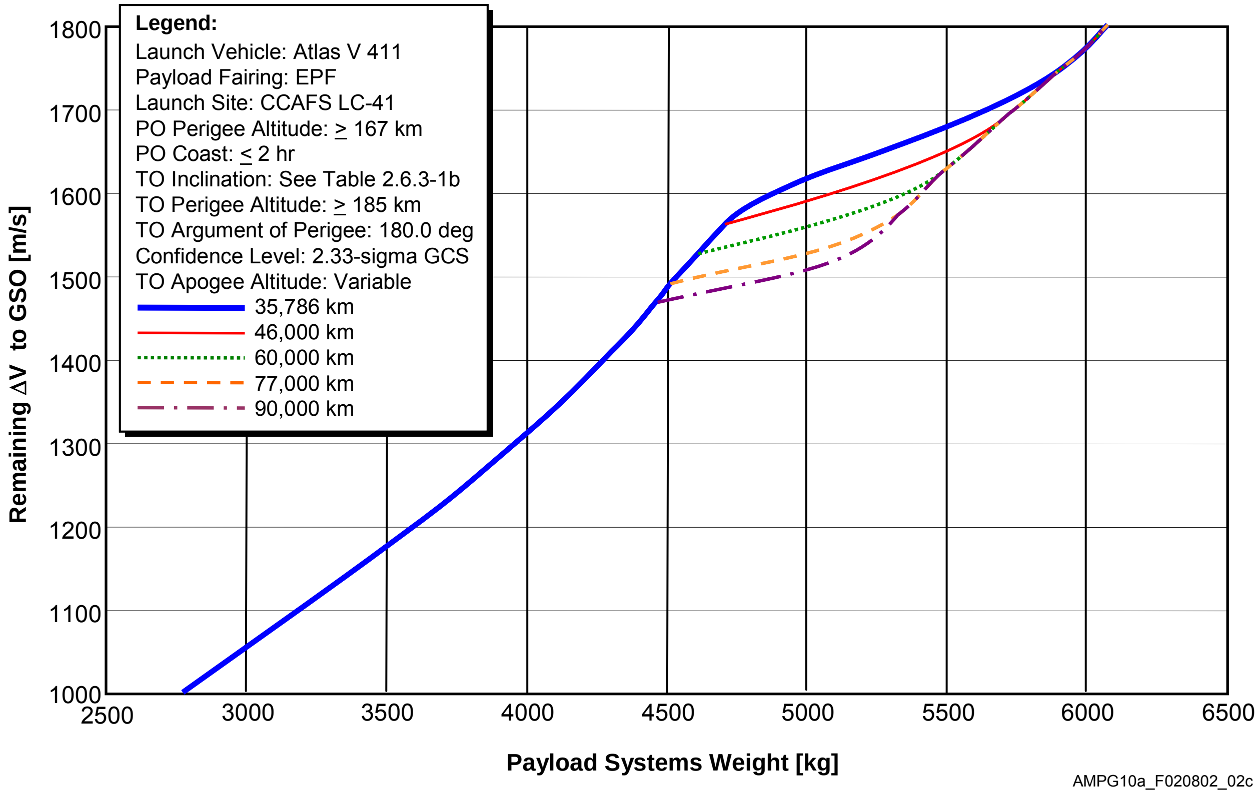

Table 2.6.3-1b: Atlas V 411 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 411 - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	190	103	26.8	27.1	6,065	13,371
1,775	214	116	25.7	27.2	6,000	13,228
1,750	217	117	24.5	27.4	5,910	13,029
1,725	230	124	23.3	27.8	5,790	12,765
1,700	233	126	22.0	28.0	5,645	12,445
1,675	240	130	20.7	28.0	5,470	12,059
1,650	248	134	19.4	28.8	5,260	11,596
1,625	741	400	20.1	44.6	5,040	11,111
1,600	3,544	1,913	25.9	102.1	4,865	10,725
1,575	4,328	2,337	26.1	108.5	4,745	10,461
1,550	5,034	2,718	26.1	116.8	4,670	10,296
1,525	5,550	2,997	25.9	120.1	4,600	10,141
1,500	6,082	3,284	25.6	123.7	4,535	9,998
1,475	6,477	3,497	25.2	123.9	4,475	9,866
1,450	6,809	3,676	24.7	122.4	4,410	9,722
1,425	7,155	3,863	24.3	120.8	4,340	9,568
1,400	7,479	4,039	23.8	119.2	4,270	9,414
1,375	7,794	4,209	23.3	117.8	4,200	9,259
1,350	8,105	4,376	22.8	116.2	4,125	9,094
1,325	8,395	4,533	22.3	114.8	4,045	8,918
1,300	8,669	4,681	21.8	113.4	3,960	8,730
1,275	8,942	4,828	21.2	111.9	3,875	8,543
1,250	9,193	4,964	20.7	110.6	3,785	8,344
1,225	9,427	5,090	20.1	109.3	3,695	8,146
1,200	9,651	5,211	19.6	108.1	3,600	7,937
1,175	9,855	5,321	19.0	106.9	3,500	7,716
1,150	10,048	5,425	18.4	105.6	3,400	7,496
1,125	10,263	5,542	17.9	104.3	3,295	7,264
1,100	10,474	5,655	17.3	103.0	3,195	7,044
1,075	10,620	5,734	16.7	101.9	3,090	6,812
1,050	10,806	5,835	16.1	100.7	2,980	6,570
1,025	10,977	5,927	15.5	99.3	2,875	6,338
1,000	11,150	6,021	14.8	98.1	2,770	6,107

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1b: Atlas V 411 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 411 - ΔV to GSO at 46,000 km (24,838 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	27.1	6,065	13,371
1,775	35,786	19,323	214	116	25.7	27.2	6,000	13,228
1,750	35,786	19,323	217	117	24.5	27.4	5,910	13,029
1,725	41,238	22,267	202	109	25.7	27.6	5,825	12,842
1,700	45,287	24,453	201	109	25.9	27.7	5,740	12,655
1,675	46,000	24,838	208	112	24.7	28.0	5,640	12,434
1,650	46,000	24,838	211	114	23.1	28.2	5,495	12,114
1,625	46,000	24,838	229	124	21.5	28.9	5,310	11,707
1,600	46,000	24,838	266	144	19.8	30.0	5,075	11,188
1,575	46,000	24,838	880	475	20.9	48.1	4,835	10,659
1,550	46,000	24,838	862	466	19.1	40.9	4,585	10,108

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours — Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 411 - ΔV to GSO at 60,000 km (32,397 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	27.1	6,065	13,371
1,775	35,786	19,323	214	116	25.7	27.2	6,000	13,228
1,750	35,786	19,323	217	117	24.5	27.4	5,910	13,029
1,725	41,238	22,267	202	109	25.7	27.6	5,825	12,842
1,700	45,287	24,453	201	109	25.9	27.7	5,740	12,655
1,675	49,931	26,961	202	109	26.2	28.0	5,655	12,467
1,650	55,392	29,909	201	108	26.5	28.0	5,570	12,280
1,625	60,000	32,397	204	110	26.2	28.3	5,485	12,092
1,600	60,000	32,397	203	109	24.2	28.6	5,355	11,806
1,575	60,000	32,397	229	124	22.2	29.2	5,155	11,365
1,550	60,000	32,397	266	143	20.0	30.6	4,880	10,759
1,525	60,000	32,397	583	315	19.4	36.5	4,585	10,108

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours — Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1b: Atlas V 411 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 411 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	27.1	6,065	13,371
1,775	35,786	19,323	214	116	25.7	27.2	6,000	13,228
1,750	35,786	19,323	217	117	24.5	27.4	5,910	13,029
1,725	41,238	22,267	202	109	25.7	27.6	5,825	12,842
1,700	45,287	24,453	201	109	25.9	27.7	5,740	12,655
1,675	49,931	26,961	202	109	26.2	28.0	5,655	12,467
1,650	55,392	29,909	201	108	26.5	28.0	5,570	12,280
1,625	60,000	32,397	204	110	26.2	28.3	5,485	12,092
1,600	68,784	37,140	200	108	27.0	28.2	5,410	11,927
1,575	77,000	41,577	202	109	27.2	28.5	5,330	11,751
1,550	77,000	41,577	208	112	24.7	28.7	5,200	11,464
1,525	77,000	41,577	225	122	22.0	29.3	4,965	10,946
1,500	77,000	41,577	587	317	21.1	39.2	4,635	10,218

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 411 - ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	27.1	6,065	13,371
1,775	35,786	19,323	214	116	25.7	27.2	6,000	13,228
1,750	35,786	19,323	217	117	24.5	27.4	5,910	13,029
1,725	41,238	22,267	202	109	25.7	27.6	5,825	12,842
1,700	45,287	24,453	201	109	25.9	27.7	5,740	12,655
1,675	49,931	26,961	202	109	26.2	28.0	5,655	12,467
1,650	55,392	29,909	201	108	26.5	28.0	5,570	12,280
1,625	60,000	32,397	204	110	26.2	28.3	5,485	12,092
1,600	68,784	37,140	200	108	27.0	28.2	5,410	11,927
1,575	76,998	41,576	202	109	27.2	28.5	5,330	11,751
1,550	87,308	47,142	199	107	27.5	28.5	5,255	11,585
1,525	90,000	48,596	206	111	25.4	28.8	5,145	11,343
1,500	90,000	48,596	229	124	22.4	29.5	4,900	10,803
1,475	90,000	48,596	543	293	20.7	37.7	4,530	9,987

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.3-1c: Atlas V 421 Minimum Delta-V to Geosynchronous Orbit

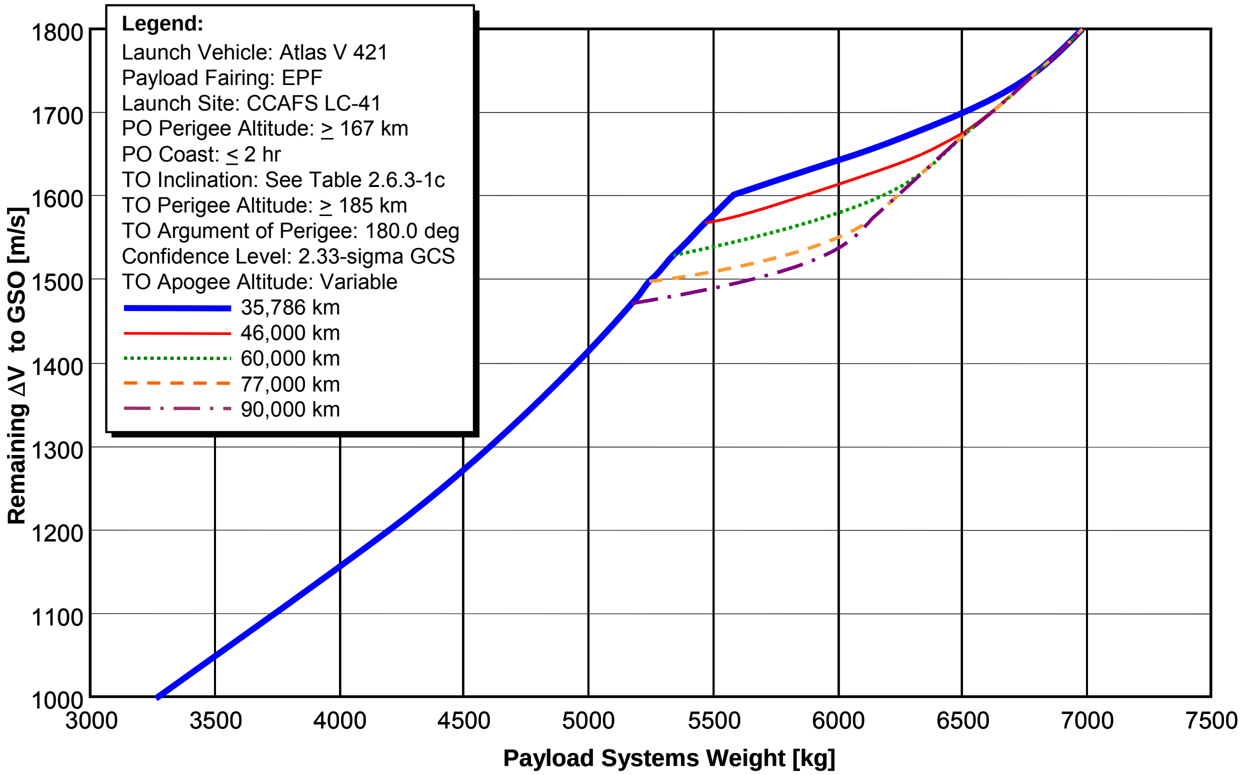

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Table 2.6.3-1c: Atlas V 421 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 421 - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	188	102	26.8	27.7	6,990	15,410
1,775	214	116	25.7	28.4	6,915	15,245
1,750	217	117	24.5	28.2	6,810	15,013
1,725	230	124	23.3	28.3	6,675	14,716
1,700	233	126	22.0	28.8	6,505	14,341
1,675	240	130	20.7	29.2	6,310	13,911
1,650	264	142	19.5	30.0	6,075	13,393
1,625	397	214	18.6	31.7	5,825	12,842
1,600	3,575	1,930	26.0	102.4	5,585	12,313
1,575	4,414	2,383	26.2	111.3	5,490	12,103
1,550	5,051	2,727	26.2	117.3	5,405	11,916
1,525	5,561	3,002	25.9	121.8	5,330	11,751
1,500	6,074	3,280	25.6	124.8	5,255	11,585
1,475	6,446	3,481	25.2	124.3	5,185	11,431
1,450	6,790	3,666	24.7	122.7	5,115	11,277
1,425	7,162	3,867	24.3	121.0	5,035	11,100
1,400	7,493	4,046	23.8	119.5	4,960	10,935
1,375	7,822	4,224	23.3	117.9	4,875	10,748
1,350	8,202	4,429	22.9	116.0	4,790	10,560
1,325	8,441	4,558	22.3	114.9	4,700	10,362
1,300	8,695	4,695	21.8	113.6	4,605	10,152
1,275	8,975	4,846	21.3	112.2	4,510	9,943
1,250	9,195	4,965	20.7	111.0	4,410	9,722
1,225	9,447	5,101	20.2	109.6	4,305	9,491
1,200	9,668	5,220	19.6	108.3	4,200	9,259
1,175	9,861	5,325	19.0	107.3	4,090	9,017
1,150	10,045	5,424	18.4	106.2	3,975	8,763
1,125	10,248	5,534	17.9	104.9	3,860	8,510
1,100	10,406	5,619	17.2	103.9	3,745	8,256
1,075	10,603	5,725	16.7	102.7	3,630	8,003
1,050	10,763	5,811	16.0	101.6	3,510	7,738
1,025	10,950	5,913	15.4	100.3	3,395	7,485
1,000	11,131	6,010	14.8	99.0	3,275	7,220

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1c: Atlas V 421 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 421 - ΔV to GSO at 46,000 km (24,838 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	189	102	26.8	27.7	6,990	15,410
1,775	35,786	19,323	214	116	25.7	28.4	6,915	15,245
1,750	37,738	20,377	200	108	25.4	28.2	6,815	15,024
1,725	41,157	22,223	201	109	25.6	28.4	6,715	14,804
1,700	45,164	24,386	201	108	25.9	28.6	6,615	14,584
1,675	46,000	24,838	204	110	24.7	28.7	6,505	14,341
1,650	46,000	24,838	209	113	23.1	28.9	6,340	13,977
1,625	46,000	24,838	225	121	21.4	29.5	6,125	13,503
1,600	46,000	24,838	347	188	20.3	32.9	5,865	12,930
1,575	46,000	24,838	551	297	19.4	36.8	5,585	12,313

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 421 - ΔV to GSO at 60,000 km (32,397 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	189	102	26.8	27.7	6,990	15,410
1,775	35,786	19,323	214	116	25.7	28.4	6,915	15,245
1,750	37,738	20,377	200	108	25.4	28.2	6,815	15,024
1,725	41,157	22,223	201	109	25.6	28.4	6,715	14,804
1,700	45,164	24,386	201	108	25.9	28.6	6,615	14,584
1,675	49,884	26,935	201	109	26.2	28.7	6,520	14,374
1,650	55,266	29,841	201	109	26.4	28.9	6,425	14,165
1,625	59,998	32,397	203	110	26.2	29.1	6,330	13,955
1,600	60,000	32,397	225	122	24.3	30.0	6,180	13,625
1,575	60,000	32,397	224	121	22.1	29.8	5,955	13,129
1,550	60,000	32,397	396	214	20.7	34.3	5,650	12,456

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1c: Atlas V 421 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 421 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	189	102	26.8	27.7	6,990	15,410
1,775	35,786	19,323	214	116	25.7	28.4	6,915	15,245
1,750	37,738	20,377	200	108	25.4	28.2	6,815	15,024
1,725	41,157	22,223	201	109	25.6	28.4	6,715	14,804
1,700	45,164	24,386	201	108	25.9	28.6	6,615	14,584
1,675	49,884	26,935	201	109	26.2	28.7	6,520	14,374
1,650	55,266	29,841	201	109	26.4	28.9	6,425	14,165
1,625	59,998	32,397	203	110	26.2	29.1	6,330	13,955
1,600	68,663	37,075	201	109	26.9	29.2	6,240	13,757
1,575	77,000	41,577	202	109	27.2	29.4	6,150	13,558
1,550	77,000	41,577	209	113	24.7	29.6	6,005	13,239
1,525	77,000	41,577	253	136	22.2	31.0	5,735	12,643
1,500	77,000	41,577	307	166	19.5	31.6	5,345	11,784
1,475	77,000	41,577	691	373	18.7	34.7	4,940	10,891

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 421 - ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	189	102	26.8	27.7	6,990	15,410
1,775	35,786	19,323	214	116	25.7	28.4	6,915	15,245
1,750	37,738	20,377	200	108	25.4	28.2	6,815	15,024
1,725	41,157	22,223	201	109	25.6	28.4	6,715	14,804
1,700	45,164	24,386	201	108	25.9	28.6	6,615	14,584
1,675	49,884	26,935	201	109	26.2	28.7	6,520	14,374
1,650	55,266	29,841	201	109	26.4	28.9	6,425	14,165
1,625	59,998	32,397	203	110	26.2	29.1	6,330	13,955
1,600	68,663	37,075	201	109	26.9	29.2	6,240	13,757
1,575	77,013	41,584	202	109	27.2	29.4	6,150	13,558
1,550	87,065	47,011	209	113	27.3	29.6	6,005	13,239
1,525	90,000	48,596	207	112	25.4	29.7	5,940	13,095
1,500	90,000	48,596	235	127	22.4	30.6	5,665	12,489
1,475	90,000	48,596	637	344	21.2	40.1	5,255	11,585

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.3-1d: Atlas V 431 Minimum Delta-V to Geosynchronous Orbit

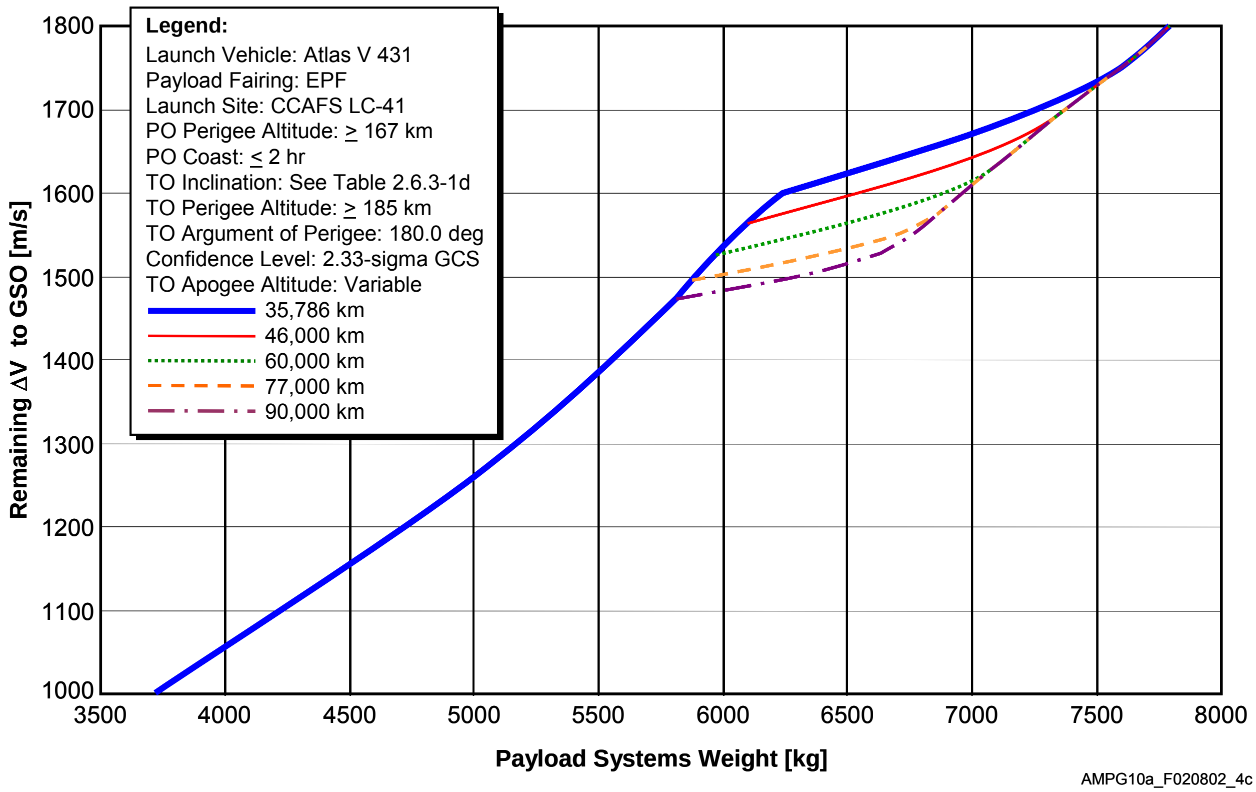


Table 2.6.3-1d: Atlas V 431 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 431 - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	191	103	26.8	28.7	7,785	17,163
1,775	214	116	25.7	28.9	7,705	16,987
1,750	217	117	24.5	28.9	7,590	16,733
1,725	230	124	23.3	29.1	7,445	16,413
1,700	233	126	22.0	29.5	7,260	16,006
1,675	240	130	20.7	29.7	7,040	15,521
1,650	237	128	19.3	30.1	6,780	14,947
1,625	579	313	19.5	37.6	6,510	14,352
1,600	3,649	1,970	26.1	103.2	6,240	13,757
1,575	4,521	2,441	26.4	115.3	6,135	13,525
1,550	5,090	2,748	26.2	120.2	6,045	13,327
1,525	5,629	3,040	26.0	125.0	5,965	13,151
1,500	6,033	3,258	25.6	126.0	5,885	12,974
1,475	6,436	3,475	25.2	124.5	5,810	12,809
1,450	6,770	3,656	24.7	123.1	5,725	12,621
1,425	7,122	3,845	24.2	121.5	5,645	12,445
1,400	7,448	4,022	23.7	120.0	5,555	12,247
1,375	7,780	4,201	23.3	118.4	5,465	12,048
1,350	8,083	4,364	22.8	116.9	5,370	11,839
1,325	8,381	4,526	22.3	115.4	5,270	11,618
1,300	8,663	4,678	21.7	114.0	5,170	11,398
1,275	8,955	4,835	21.2	112.6	5,065	11,166
1,250	9,149	4,940	20.7	111.6	4,950	10,913
1,225	9,377	5,063	20.1	110.3	4,835	10,659
1,200	9,619	5,194	19.6	109.0	4,720	10,406
1,175	9,821	5,303	19.0	107.8	4,600	10,141
1,150	10,012	5,406	18.4	106.8	4,475	9,866
1,125	10,229	5,523	17.8	105.4	4,355	9,601
1,100	10,391	5,610	17.2	104.4	4,225	9,315
1,075	10,610	5,729	16.7	103.0	4,100	9,039
1,050	10,798	5,830	16.1	101.5	3,970	8,752
1,025	10,968	5,922	15.5	100.6	3,845	8,477
1,000	11,133	6,011	14.8	99.5	3,715	8,190

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours — Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1d: Atlas V 431 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 431 - ΔV to GSO at 46,000 km (24,838 nmi) Cap

ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	191	103	26.8	28.7	7,785	17,163
1,775	35,786	19,323	214	116	25.7	28.9	7,705	16,987
1,750	37,666	20,338	201	109	25.4	28.9	7,600	16,755
1,725	41,081	22,182	205	111	25.6	29.3	7,490	16,513
1,700	45,131	24,369	202	109	25.9	29.4	7,380	16,270
1,675	46,000	24,838	205	111	24.7	29.5	7,255	15,995
1,650	46,000	24,838	211	114	23.1	29.7	7,075	15,598
1,625	46,000	24,838	226	122	21.4	30.3	6,835	15,069
1,600	46,000	24,838	257	139	19.8	31.0	6,545	14,429
1,575	46,000	24,838	840	454	20.8	46.0	6,250	13,779

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 431 - ΔV to GSO at 60,000 km (32,397 nmi) Cap

ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	191	103	26.8	28.7	7,785	17,163
1,775	35,786	19,323	214	116	25.7	28.9	7,705	16,987
1,750	37,666	20,338	201	109	25.4	28.9	7,600	16,755
1,725	41,081	22,182	205	111	25.6	29.3	7,490	16,513
1,700	45,131	24,369	202	109	25.9	29.4	7,380	16,270
1,675	49,767	26,872	202	109	26.1	29.5	7,270	16,028
1,650	55,144	29,775	203	109	26.4	29.7	7,165	15,796
1,625	60,000	32,397	204	110	26.2	29.9	7,060	15,565
1,600	60,000	32,397	209	113	24.2	30.1	6,895	15,201
1,575	60,000	32,397	234	126	22.2	30.9	6,645	14,650
1,550	60,000	32,397	399	216	20.7	34.8	6,310	13,911
1,525	60,000	32,397	1,196	646	22.2	55.6	5,975	13,173

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-1d: Atlas V 431 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 431 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	191	103	26.8	28.7	7,785	17,163
1,775	35,786	19,323	214	116	25.7	28.9	7,705	16,987
1,750	37,666	20,338	201	109	25.4	28.9	7,600	16,755
1,725	41,081	22,182	205	111	25.6	29.3	7,490	16,513
1,700	45,131	24,369	202	109	25.9	29.4	7,380	16,270
1,675	49,767	26,872	202	109	26.1	29.5	7,270	16,028
1,650	55,144	29,775	203	109	26.4	29.7	7,165	15,796
1,625	61,390	33,148	204	110	26.6	29.9	7,060	15,565
1,600	68,709	37,100	203	110	27.0	30.1	6,960	15,344
1,575	77,000	41,577	203	110	27.2	30.2	6,860	15,124
1,550	77,000	41,577	221	120	24.7	30.8	6,700	14,771
1,525	77,000	41,577	227	122	22.0	30.9	6,405	14,121
1,500	77,000	41,577	370	200	19.9	32.9	5,985	13,195

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 431 - ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	191	103	26.8	28.7	7,785	17,163
1,775	35,786	19,323	214	116	25.7	28.9	7,705	16,987
1,750	37,666	20,338	201	109	25.4	28.9	7,600	16,755
1,725	41,081	22,182	205	111	25.6	29.3	7,490	16,513
1,700	45,131	24,369	202	109	25.9	29.4	7,380	16,270
1,675	49,767	26,872	202	109	26.1	29.5	7,270	16,028
1,650	55,144	29,775	203	109	26.4	29.7	7,165	15,796
1,625	61,390	33,148	204	110	26.6	29.9	7,060	15,565
1,600	68,709	37,100	203	110	27.0	30.1	6,960	15,344
1,575	77,000	41,577	203	110	27.2	30.2	6,860	15,124
1,550	87,034	46,994	204	110	27.4	30.4	6,770	14,925
1,525	90,000	48,596	209	113	25.4	30.6	6,630	14,617
1,500	90,000	48,596	293	158	22.7	32.8	6,320	13,933
1,475	90,000	48,596	405	218	20.0	33.3	5,860	12,919

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2: Atlas V 500 Series Minimum Delta-V to Geosynchronous Orbit Summary (1 of 2)

Atlas V 500 Series – Delta-V to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Payload Systems Weight					
	501 (Table 2.6.3-2a)		511 (Table 2.6.3-2b)		521 (Table 2.6.3-2c)	
[m/s]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
1,800	3,960	8,730	5,360	11,817	6,475	14,275
1,775	3,920	8,642	5,300	11,684	6,405	14,121
1,750	3,860	8,510	5,220	11,508	6,305	13,900
1,725	3,785	8,344	5,110	11,266	6,175	13,614
1,700	3,685	8,124	4,975	10,968	6,015	13,261
1,675	3,565	7,859	4,810	10,604	5,820	12,831
1,650	3,420	7,540	4,620	10,185	5,595	12,335
1,625	3,270	7,209	4,420	9,744	5,365	11,828
1,600	3,190	7,033	4,265	9,403	5,195	11,453
1,575	3,135	6,911	4,210	9,281	5,130	11,310
1,550	3,090	6,812	4,155	9,160	5,065	11,166
1,525	3,040	6,702	4,095	9,028	4,995	11,012
1,500	3,000	6,614	4,040	8,907	4,930	10,869
1,475	2,960	6,526	3,980	8,774	4,860	10,714
1,450	2,915	6,426	3,920	8,642	4,785	10,549
1,425	2,870	6,327	3,855	8,499	4,715	10,395
1,400	2,820	6,217	3,790	8,356	4,635	10,218
1,375	2,765	6,096	3,725	8,212	4,560	10,053
1,350	2,710	5,975	3,655	8,058	4,475	9,866
1,325	2,650	5,842	3,585	7,904	4,395	9,689
1,300	2,590	5,710	3,510	7,738	4,310	9,502
1,275	2,530	5,578	3,435	7,573	4,220	9,303
1,250	2,465	5,434	3,360	7,408	4,125	9,094
1,225	2,405	5,302	3,280	7,231	4,035	8,896
1,200	2,340	5,159	3,195	7,044	3,935	8,675
1,175	2,275	5,016	3,110	6,856	3,840	8,466
1,150	2,205	4,861	3,025	6,669	3,735	8,234
1,125	2,140	4,718	2,935	6,471	3,635	8,014
1,100	2,070	4,564	2,850	6,283	3,530	7,782
1,075	2,000	4,409	2,755	6,074	3,425	7,551
1,050	1,925	4,244	2,665	5,875	3,320	7,319
1,025	1,850	4,079	2,575	5,677	3,210	7,077
1,000	1,775	3,913	2,480	5,467	3,100	6,834

Table 2.6.3-2: Atlas V 500 Series Minimum Delta-V to Geosynchronous Orbit Summary (2 of 2)

Atlas V 500 Series – Delta-V to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Payload Systems Weight					
	531 (Table 2.6.3-2d)		541 (Table 2.6.3-2e)		551 (Table 2.6.3-2f12)	
[m/s]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
1,800	7,410	16,336	8,225	18,133	8,685	19,147
1,775	7,330	16,160	8,135	17,935	8,585	18,927
1,750	7,215	15,906	8,005	17,648	8,455	18,640
1,725	7,070	15,587	7,840	17,284	8,275	18,243
1,700	6,885	15,179	7,635	16,832	8,055	17,758
1,675	6,665	14,694	7,390	16,292	7,795	17,185
1,650	6,415	14,143	7,125	15,708	7,515	16,568
1,625	6,165	13,591	6,855	15,113	7,245	15,972
1,600	5,945	13,106	6,615	14,584	7,055	15,554
1,575	5,870	12,941	6,535	14,407	6,965	15,355
1,550	5,800	12,787	6,450	14,220	6,875	15,157
1,525	5,725	12,621	6,365	14,032	6,785	14,958
1,500	5,645	12,445	6,280	13,845	6,695	14,760
1,475	5,565	12,269	6,190	13,647	6,600	14,550
1,450	5,485	12,092	6,100	13,448	6,505	14,341
1,425	5,400	11,905	6,005	13,239	6,410	14,132
1,400	5,315	11,718	5,910	13,029	6,305	13,900
1,375	5,230	11,530	5,810	12,809	6,200	13,669
1,350	5,135	11,321	5,705	12,577	6,095	13,437
1,325	5,040	11,111	5,600	12,346	5,985	13,195
1,300	4,945	10,902	5,490	12,103	5,865	12,930
1,275	4,845	10,681	5,380	11,861	5,750	12,677
1,250	4,740	10,450	5,265	11,607	5,630	12,412
1,225	4,635	10,218	5,145	11,343	5,505	12,136
1,200	4,525	9,976	5,025	11,078	5,375	11,850
1,175	4,410	9,722	4,900	10,803	5,245	11,563
1,150	4,295	9,469	4,775	10,527	5,115	11,277
1,125	4,180	9,215	4,650	10,251	4,980	10,979
1,100	4,065	8,962	4,515	9,954	4,845	10,681
1,075	3,945	8,697	4,385	9,667	4,705	10,373
1,050	3,825	8,433	4,250	9,370	4,565	10,064
1,025	3,705	8,168	4,120	9,083	4,430	9,766
1,000	3,585	7,904	3,980	8,774	4,295	9,469

Figure 2.6.3-2a: Atlas V 501 Minimum Delta-V to Geosynchronous Orbit

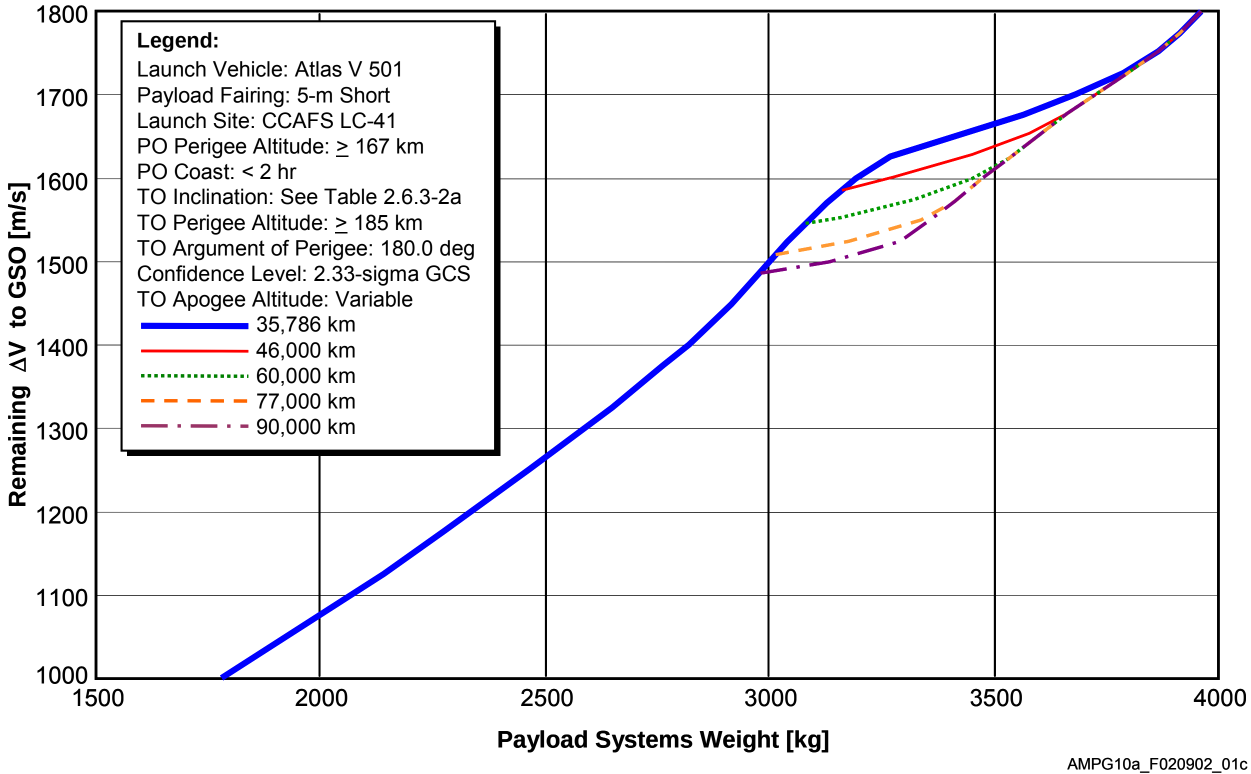


Table 2.6.3-2a: Atlas V 501 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 501 - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	190	103	26.8	25.6	3,960	8,730
1,775	214	116	25.7	25.8	3,920	8,642
1,750	217	117	24.5	25.7	3,860	8,510
1,725	230	124	23.3	26.1	3,785	8,344
1,700	233	126	22.0	26.3	3,685	8,124
1,675	240	130	20.7	28.0	3,565	7,859
1,650	248	134	19.4	28.9	3,420	7,540
1,625	2,839	1,533	25.7	39.6	3,270	7,209
1,600	3,590	1,938	26.0	118.2	3,190	7,033
1,575	4,296	2,320	26.1	117.6	3,135	6,911
1,550	4,909	2,651	26.0	117.3	3,090	6,812
1,525	5,407	2,920	25.7	116.5	3,040	6,702
1,500	5,717	3,087	25.3	122.1	3,000	6,614
1,475	5,882	3,176	24.6	121.8	2,960	6,526
1,450	5,960	3,218	23.9	121.1	2,915	6,426
1,425	5,944	3,209	23.1	119.8	2,870	6,327
1,400	6,316	3,410	22.7	118.4	2,820	6,217
1,375	6,666	3,599	22.2	116.8	2,765	6,096
1,350	7,009	3,785	21.8	115.3	2,710	5,975
1,325	7,335	3,961	21.3	113.6	2,650	5,842
1,300	7,643	4,127	20.9	111.8	2,590	5,710
1,275	7,942	4,288	20.4	110.1	2,530	5,578
1,250	8,232	4,445	19.9	108.2	2,465	5,434
1,225	8,511	4,596	19.4	106.1	2,405	5,302
1,200	8,769	4,735	18.8	103.9	2,340	5,159
1,175	9,029	4,875	18.3	102.0	2,275	5,016
1,150	9,278	5,010	17.8	99.6	2,205	4,861
1,125	9,517	5,139	17.2	97.6	2,140	4,718
1,100	9,758	5,269	16.7	95.3	2,070	4,564
1,075	10,000	5,400	16.1	92.9	2,000	4,409
1,050	10,234	5,526	15.6	90.3	1,925	4,244
1,025	10,480	5,659	15.0	91.7	1,850	4,079
1,000	10,729	5,793	14.5	87.2	1,775	3,913

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2a: Atlas V 501 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 501 - ΔV to GSO at 46,000 km (24,838 nmi) Cap

ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	25.6	3,960	8,730
1,775	35,786	19,323	214	116	25.7	25.8	3,920	8,642
1,750	35,786	19,323	217	117	24.5	25.7	3,860	8,510
1,725	39,303	21,222	201	109	24.8	25.9	3,795	8,367
1,700	43,339	23,401	202	109	25.1	26.0	3,730	8,223
1,675	46,000	24,838	201	108	24.7	26.0	3,660	8,069
1,650	46,000	24,838	209	113	23.1	26.5	3,565	7,859
1,625	46,000	24,838	222	120	21.4	27.3	3,440	7,584
1,600	46,000	24,838	306	165	20.0	31.0	3,275	7,220
1,575	46,000	24,838	492	266	19.1	33.8	3,095	6,823
1,550	46,000	24,838	735	397	18.5	35.6	2,905	6,404

Notes:
Launch Site: CCAFS LC-41 | Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) | Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours | Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 501 - ΔV to GSO at 60,000 km (32,397 nmi) Cap

ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	25.6	3,960	8,730
1,775	35,786	19,323	214	116	25.7	25.8	3,920	8,642
1,750	35,786	19,323	217	117	24.5	25.7	3,860	8,510
1,725	39,303	21,222	201	109	24.8	25.9	3,795	8,367
1,700	43,339	23,401	202	109	25.1	26.0	3,730	8,223
1,675	46,000	24,838	201	108	24.7	26.0	3,660	8,069
1,650	53,394	28,830	201	108	25.8	26.2	3,600	7,937
1,625	59,616	32,190	198	107	26.1	26.1	3,535	7,793
1,600	60,000	32,397	208	112	24.2	26.7	3,455	7,617
1,575	60,000	32,397	220	119	22.1	27.3	3,320	7,319
1,550	60,000	32,397	304	164	20.3	31.1	3,130	6,900
1,525	60,000	32,397	1,362	735	22.8	65.7	2,935	6,471

Notes:
Launch Site: CCAFS LC-41 | Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) | Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours | Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2a: Atlas V 501 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 501 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	25.6	3,960	8,730
1,775	35,786	19,323	214	116	25.7	25.8	3,920	8,642
1,750	35,786	19,323	217	117	24.5	25.7	3,860	8,510
1,725	39,303	21,222	201	109	24.8	25.9	3,795	8,367
1,700	43,339	23,401	202	109	25.1	26.0	3,730	8,223
1,675	46,000	24,838	201	108	24.7	26.0	3,660	8,069
1,650	53,394	28,830	201	108	25.8	26.2	3,600	7,937
1,625	59,616	32,190	198	107	26.1	26.1	3,535	7,793
1,600	66,719	36,025	204	110	26.3	26.7	3,475	7,661
1,575	75,086	40,543	204	110	26.6	26.7	3,420	7,540
1,550	77,000	41,577	215	116	24.7	27.3	3,335	7,352
1,525	77,000	41,577	273	148	22.3	30.0	3,175	7,000
1,500	77,000	41,577	493	266	20.6	37.9	2,940	6,482
1,475	77,000	41,577	1,627	878	23.0	70.1	2,745	6,052

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 501 - ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	25.6	3,960	8,730
1,775	35,786	19,323	214	116	25.7	25.8	3,920	8,642
1,750	35,786	19,323	217	117	24.5	25.7	3,860	8,510
1,725	39,303	21,222	201	109	24.8	25.9	3,795	8,367
1,700	43,339	23,401	202	109	25.1	26.0	3,730	8,223
1,675	46,000	24,838	201	108	24.7	26.0	3,660	8,069
1,650	53,394	28,830	201	108	25.8	26.2	3,600	7,937
1,625	59,616	32,190	198	107	26.1	26.1	3,535	7,793
1,600	66,719	36,025	204	110	26.3	26.7	3,475	7,661
1,575	75,086	40,543	204	110	26.6	26.7	3,420	7,540
1,550	85,114	45,958	195	105	26.9	26.2	3,360	7,408
1,525	90,000	48,596	201	108	25.4	26.6	3,290	7,253
1,500	90,000	48,596	221	119	22.3	27.6	3,130	6,900
1,475	90,000	48,596	354	191	19.7	31.7	2,860	6,305

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.3-2b: Atlas V 511 Minimum Delta-V to Geosynchronous Orbit

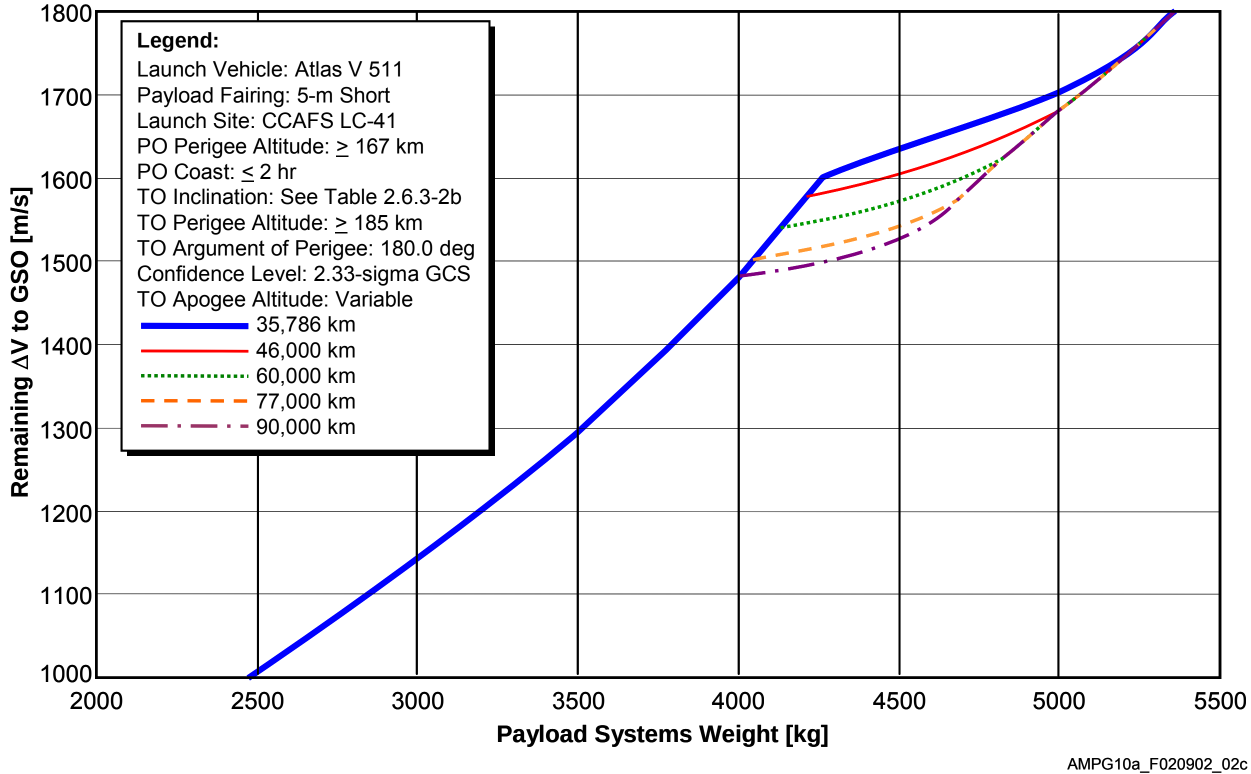


Table 2.6.3-2b: Atlas V 511 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 511 - ∆V to GSO at 35,786 km (19,323 nmi)						
∆V to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	190	103	26.8	26.3	5,360	11,817
1,775	214	116	25.7	27.2	5,300	11,684
1,750	217	117	24.5	26.5	5,220	11,508
1,725	230	124	23.3	26.6	5,110	11,266
1,700	233	126	22.0	26.9	4,975	10,968
1,675	240	130	20.7	26.7	4,810	10,604
1,650	248	134	19.4	27.7	4,620	10,185
1,625	394	213	18.6	29.0	4,420	9,744
1,600	4,183	2,259	26.8	132.3	4,265	9,403
1,575	4,516	2,438	26.4	132.1	4,210	9,281
1,550	4,873	2,631	26.0	130.6	4,155	9,160
1,525	5,183	2,799	25.5	129.3	4,095	9,028
1,500	5,526	2,984	25.0	127.9	4,040	8,907
1,475	5,863	3,165	24.6	126.4	3,980	8,774
1,450	6,207	3,351	24.2	125.0	3,920	8,642
1,425	6,514	3,517	23.7	123.6	3,855	8,499
1,400	6,866	3,708	23.2	121.9	3,790	8,356
1,375	7,160	3,866	22.7	120.6	3,725	8,212
1,350	7,477	4,037	22.3	119.1	3,655	8,058
1,325	7,795	4,209	21.8	117.6	3,585	7,904
1,300	8,050	4,347	21.2	116.3	3,510	7,738
1,275	8,327	4,496	20.7	114.8	3,435	7,573
1,250	8,640	4,665	20.2	113.4	3,360	7,408
1,225	8,882	4,796	19.7	112.0	3,280	7,231
1,200	9,146	4,938	19.2	110.6	3,195	7,044
1,175	9,397	5,074	18.6	109.3	3,110	6,856
1,150	9,619	5,194	18.1	107.9	3,025	6,669
1,125	9,866	5,327	17.5	106.6	2,935	6,471
1,100	10,052	5,428	16.9	105.2	2,850	6,283
1,075	10,271	5,546	16.4	103.9	2,755	6,074
1,050	10,491	5,665	15.8	102.5	2,665	5,875
1,025	10,681	5,767	15.2	101.2	2,575	5,677
1,000	10,879	5,874	14.6	99.8	2,480	5,467

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2b: Atlas V 511 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 511 - ΔV to GSO at 46,000 km (24,838 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.3	5,360	11,817
1,775	35,786	19,323	214	116	25.7	27.2	5,300	11,684
1,750	35,786	19,323	217	117	24.5	26.5	5,220	11,508
1,725	40,818	22,040	196	106	25.5	26.7	5,140	11,332
1,700	44,961	24,277	195	106	25.8	26.8	5,060	11,155
1,675	46,000	24,838	198	107	24.7	27.0	4,970	10,957
1,650	46,000	24,838	202	109	23.0	27.3	4,835	10,659
1,625	46,000	24,838	214	116	21.4	27.9	4,665	10,285
1,600	46,000	24,838	234	127	19.7	28.3	4,440	9,789
1,575	46,000	24,838	428	231	18.8	30.3	4,200	9,259

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 511 - ΔV to GSO at 60,000 km (32,397 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.3	5,360	11,817
1,775	35,786	19,323	214	116	25.7	27.2	5,300	11,684
1,750	35,786	19,323	217	117	24.5	26.5	5,220	11,508
1,725	40,818	22,040	196	106	25.5	26.7	5,140	11,332
1,700	44,961	24,277	195	106	25.8	26.8	5,060	11,155
1,675	49,656	26,812	196	106	26.1	27.0	4,980	10,979
1,650	55,181	29,795	195	105	26.4	27.1	4,900	10,803
1,625	60,000	32,397	196	106	26.2	27.2	4,825	10,637
1,600	60,000	32,397	202	109	24.2	27.7	4,705	10,373
1,575	60,000	32,397	212	115	22.1	28.4	4,520	9,965
1,550	60,000	32,397	250	135	20.0	29.2	4,265	9,403
1,525	60,000	32,397	506	273	19.0	30.1	3,980	8,774

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2b: Atlas V 511 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 511 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.3	5,360	11,817
1,775	35,786	19,323	214	116	25.7	27.2	5,300	11,684
1,750	35,786	19,323	217	117	24.5	26.5	5,220	11,508
1,725	40,818	22,040	196	106	25.5	26.7	5,140	11,332
1,700	44,961	24,277	195	106	25.8	26.8	5,060	11,155
1,675	49,656	26,812	196	106	26.1	27.0	4,980	10,979
1,650	55,181	29,795	195	105	26.4	27.1	4,900	10,803
1,625	60,000	32,397	196	106	26.2	27.2	4,825	10,637
1,600	68,577	37,028	195	105	26.9	27.3	4,750	10,472
1,575	77,000	41,577	194	105	27.1	27.4	4,680	10,318
1,550	77,000	41,577	214	115	24.7	28.4	4,560	10,053
1,525	77,000	41,577	231	124	22.1	29.4	4,345	9,579
1,500	77,000	41,577	380	205	19.9	31.4	4,025	8,874

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 511 - ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.3	5,360	11,817
1,775	35,786	19,323	214	116	25.7	27.2	5,300	11,684
1,750	35,786	19,323	217	117	24.5	26.5	5,220	11,508
1,725	40,818	22,040	196	106	25.5	26.7	5,140	11,332
1,700	44,961	24,277	195	106	25.8	26.8	5,060	11,155
1,675	49,656	26,812	196	106	26.1	27.0	4,980	10,979
1,650	55,181	29,795	195	105	26.4	27.1	4,900	10,803
1,625	60,000	32,397	196	106	26.2	27.2	4,825	10,637
1,600	68,577	37,028	195	105	26.9	27.3	4,750	10,472
1,575	77,000	41,577	194	105	27.1	27.4	4,680	10,318
1,550	86,990	46,971	197	106	27.4	27.8	4,610	10,163
1,525	90,000	48,596	211	114	25.4	28.4	4,510	9,943
1,500	90,000	48,596	231	125	22.4	29.2	4,280	9,436
1,475	90,000	48,596	359	194	19.7	30.3	3,925	8,653

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.3-2c: Atlas V 521 Minimum Delta-V to Geosynchronous Orbit

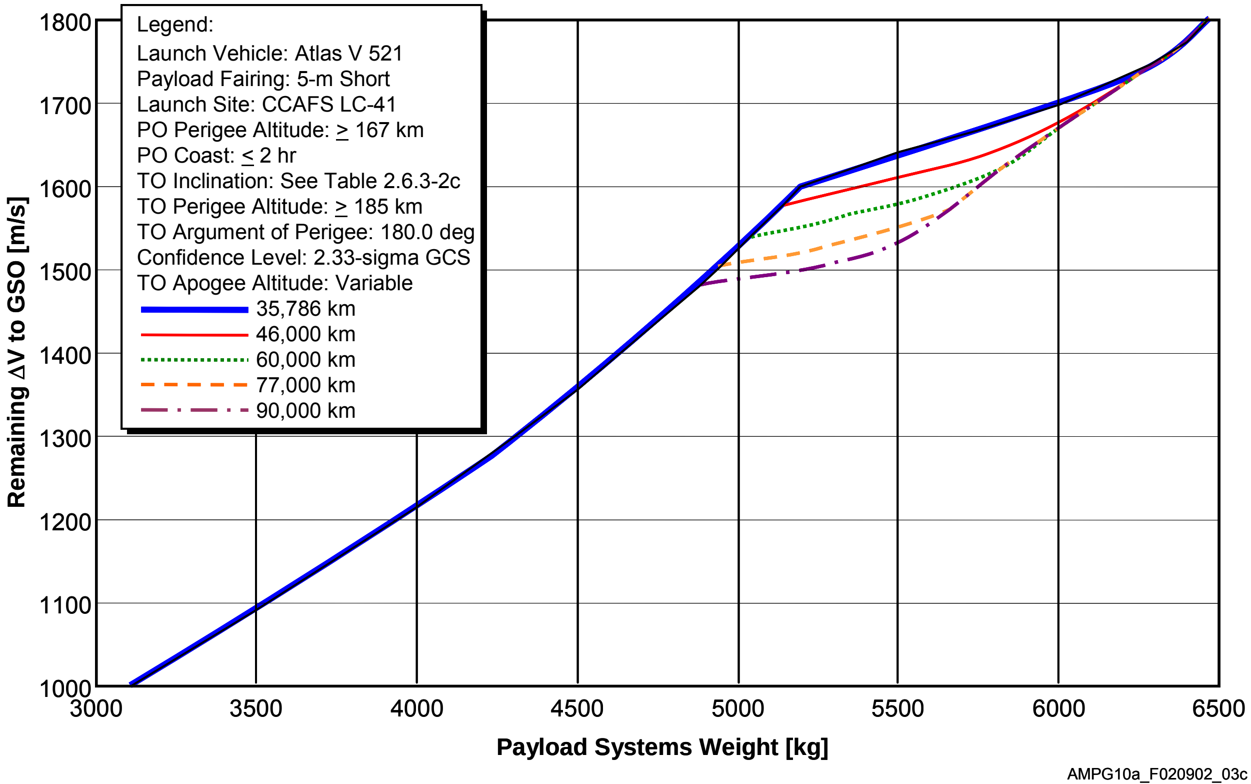


Table 2.6.3-2c: Atlas V 521 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 521 - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	190	103	26.8	26.8	6,475	14,275
1,775	214	116	25.7	27.8	6,405	14,121
1,750	217	117	24.5	27.3	6,305	13,900
1,725	230	124	23.3	27.5	6,175	13,614
1,700	233	126	22.0	27.8	6,015	13,261
1,675	240	130	20.7	27.5	5,820	12,831
1,650	248	134	19.4	26.6	5,595	12,335
1,625	408	220	18.7	28.3	5,365	11,828
1,600	4,174	2,254	26.8	133.2	5,195	11,453
1,575	4,515	2,438	26.4	132.4	5,130	11,310
1,550	4,844	2,615	25.9	131.0	5,065	11,166
1,525	5,189	2,802	25.5	129.6	4,995	11,012
1,500	5,509	2,974	25.0	128.2	4,930	10,869
1,475	5,874	3,172	24.6	126.7	4,860	10,714
1,450	6,172	3,332	24.1	125.3	4,785	10,549
1,425	6,545	3,534	23.7	123.7	4,715	10,395
1,400	6,868	3,709	23.2	122.2	4,635	10,218
1,375	7,224	3,901	22.8	120.7	4,560	10,053
1,350	7,510	4,055	22.3	119.3	4,475	9,866
1,325	7,814	4,219	21.8	117.8	4,395	9,689
1,300	8,138	4,394	21.3	116.3	4,310	9,502
1,275	8,416	4,544	20.8	114.9	4,220	9,303
1,250	8,709	4,703	20.3	113.4	4,125	9,094
1,225	8,972	4,845	19.8	112.0	4,035	8,896
1,200	9,208	4,972	19.2	110.8	3,935	8,675
1,175	9,465	5,111	18.7	109.3	3,840	8,466
1,150	9,685	5,230	18.1	108.1	3,735	8,234
1,125	9,852	5,320	17.5	107.0	3,635	8,014
1,100	10,119	5,464	17.0	105.5	3,530	7,782
1,075	10,336	5,581	16.4	104.2	3,425	7,551
1,050	10,510	5,675	15.8	103.0	3,320	7,319
1,025	10,734	5,796	15.2	101.5	3,210	7,077
1,000	10,955	5,915	14.7	100.1	3,100	6,834

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Confidence Level: 2.33 Sigma GCS
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2c: Atlas V 521 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 521 - ΔV to GSO at 46,000 km (24,838 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.8	6,475	14,275
1,775	35,786	19,323	214	116	25.7	27.8	6,405	14,121
1,750	35,786	19,323	217	117	24.5	27.3	6,305	13,900
1,725	40,980	22,128	199	107	25.5	27.9	6,215	13,702
1,700	45,011	24,304	195	105	25.8	27.8	6,115	13,481
1,675	46,000	24,838	198	107	24.6	28.0	6,010	13,250
1,650	46,000	24,838	204	110	23.1	28.5	5,850	12,897
1,625	46,000	24,838	207	112	21.3	28.2	5,645	12,445
1,600	46,000	24,838	243	131	19.7	28.1	5,385	11,872
1,575	46,000	24,838	482	260	19.1	29.8	5,120	11,288

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 521 - ΔV to GSO at 60,000 km (32,397 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.8	6,475	14,275
1,775	35,786	19,323	214	116	25.7	27.8	6,405	14,121
1,750	35,786	19,323	217	117	24.5	27.3	6,305	13,900
1,725	40,980	22,128	199	107	25.5	27.9	6,215	13,702
1,700	45,011	24,304	195	105	25.8	27.8	6,115	13,481
1,675	49,770	26,874	194	105	26.1	28.0	6,025	13,283
1,650	55,047	29,723	199	107	26.3	28.4	5,930	13,073
1,625	60,000	32,397	195	106	26.2	28.3	5,840	12,875
1,600	60,000	32,397	205	110	24.2	28.8	5,695	12,555
1,575	60,000	32,397	208	112	22.0	28.8	5,475	12,070
1,550	60,000	32,397	282	152	20.1	29.8	5,175	11,409
1,525	60,000	32,397	678	366	19.9	35.9	4,870	10,736

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2c: Atlas V 521 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 521 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.8	6,475	14,275
1,775	35,786	19,323	214	116	25.7	27.8	6,405	14,121
1,750	35,786	19,323	217	117	24.5	27.3	6,305	13,900
1,725	40,980	22,128	199	107	25.5	27.9	6,215	13,702
1,700	45,011	24,304	195	105	25.8	27.8	6,115	13,481
1,675	49,770	26,874	194	105	26.1	28.0	6,025	13,283
1,650	55,047	29,723	199	107	26.3	28.4	5,930	13,073
1,625	60,000	32,397	195	106	26.2	28.3	5,840	12,875
1,600	68,445	36,957	195	105	26.9	28.4	5,750	12,677
1,575	76,900	41,522	194	105	27.1	28.5	5,665	12,489
1,550	77,000	41,577	209	113	24.7	29.7	5,525	12,181
1,525	77,000	41,577	221	119	22.0	29.6	5,270	11,618
1,500	77,000	41,577	368	199	19.9	30.2	4,895	10,792

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 521- ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	26.8	6,475	14,275
1,775	35,786	19,323	214	116	25.7	27.8	6,405	14,121
1,750	35,786	19,323	217	117	24.5	27.3	6,305	13,900
1,725	40,980	22,128	199	107	25.5	27.9	6,215	13,702
1,700	45,011	24,304	195	105	25.8	27.8	6,115	13,481
1,675	49,770	26,874	194	105	26.1	28.0	6,025	13,283
1,650	55,047	29,723	199	107	26.3	28.4	5,930	13,073
1,625	60,000	32,397	195	106	26.2	28.3	5,840	12,875
1,600	68,445	36,957	195	105	26.9	28.4	5,750	12,677
1,575	76,900	41,522	194	105	27.1	28.5	5,665	12,489
1,550	86,972	46,961	195	105	27.4	28.7	5,585	12,313
1,525	90,000	48,596	202	109	25.4	29.1	5,465	12,048
1,500	90,000	48,596	258	139	22.5	31.1	5,195	11,453
1,475	90,000	48,596	429	232	20.1	31.7	4,790	10,560

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.3-2d: Atlas V 531 Minimum Delta-V to Geosynchronous Orbit

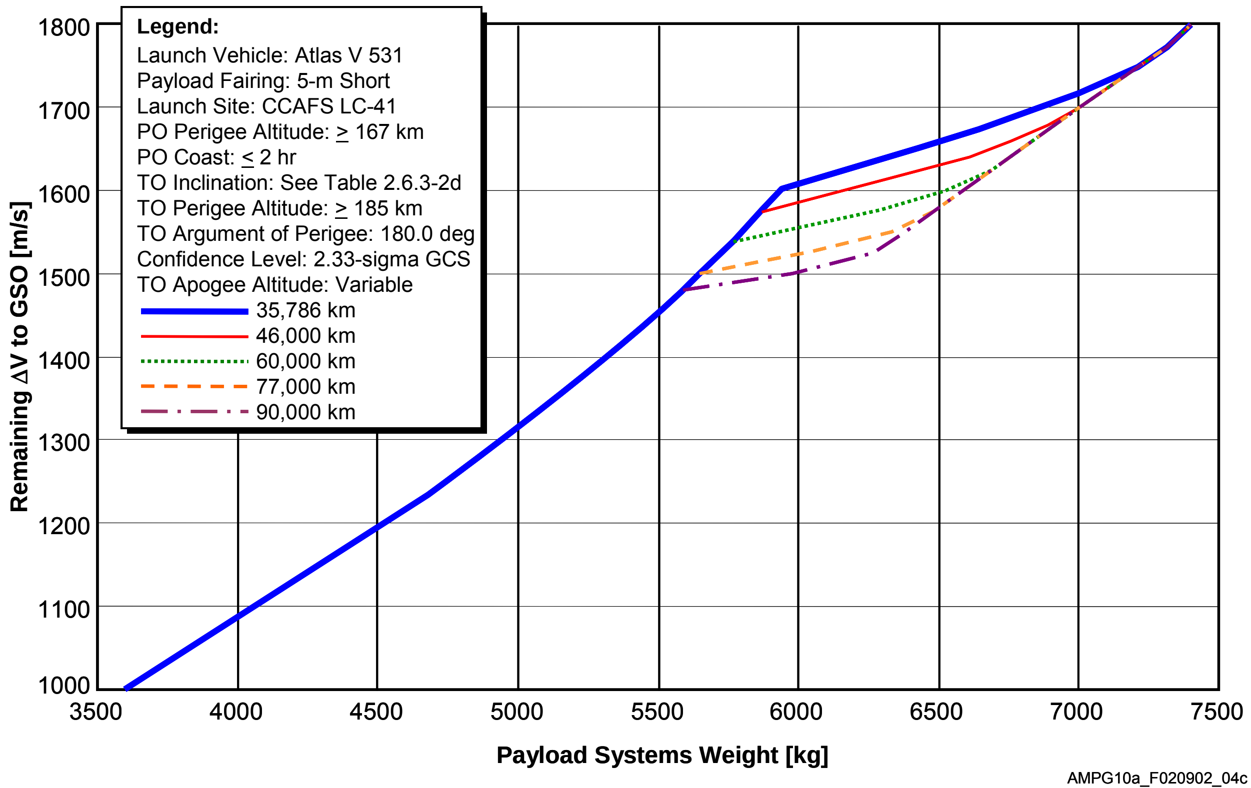


Table 2.6.3-2d: Atlas V 531 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 531 - ∆V to GSO at 35,786 km (19,323 nmi)						
∆V to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	190	103	26.8	28.0	7,410	16,336
1,775	214	116	25.7	28.2	7,330	16,160
1,750	217	117	24.5	28.2	7,215	15,906
1,725	230	124	23.3	28.3	7,070	15,587
1,700	233	126	22.0	28.2	6,885	15,179
1,675	240	130	20.7	28.2	6,665	14,694
1,650	274	148	19.5	27.4	6,415	14,143
1,625	449	242	18.9	28.3	6,165	13,591
1,600	4,197	2,266	26.8	133.2	5,945	13,106
1,575	4,543	2,453	26.4	132.4	5,870	12,941
1,550	4,887	2,639	26.0	131.0	5,800	12,787
1,525	5,232	2,825	25.5	129.6	5,725	12,621
1,500	5,584	3,015	25.1	128.1	5,645	12,445
1,475	5,973	3,225	24.7	126.5	5,565	12,269
1,450	6,292	3,397	24.2	125.1	5,485	12,092
1,425	6,636	3,583	23.8	123.6	5,400	11,905
1,400	6,969	3,763	23.3	122.1	5,315	11,718
1,375	7,319	3,952	22.9	120.4	5,230	11,530
1,350	7,593	4,100	22.4	119.1	5,135	11,321
1,325	7,921	4,277	21.9	117.5	5,040	11,111
1,300	8,254	4,457	21.4	116.1	4,945	10,902
1,275	8,496	4,588	20.9	114.8	4,845	10,681
1,250	8,766	4,733	20.3	113.4	4,740	10,450
1,225	9,028	4,874	19.8	112.0	4,635	10,218
1,200	9,293	5,018	19.3	110.7	4,525	9,976
1,175	9,500	5,130	18.7	109.4	4,410	9,722
1,150	9,750	5,265	18.2	108.0	4,295	9,469
1,125	9,971	5,384	17.6	106.8	4,180	9,215
1,100	10,199	5,507	17.1	105.4	4,065	8,962
1,075	10,404	5,618	16.5	104.1	3,945	8,697
1,050	10,612	5,730	15.9	102.8	3,825	8,433
1,025	10,762	5,811	15.3	101.7	3,705	8,168
1,000	10,990	5,934	14.7	100.1	3,585	7,904

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2d: Atlas V 531 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 531 - ΔV to GSO at 46,000 km (24,838 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	28.0	7,410	16,336
1,775	35,786	19,323	214	116	25.7	28.2	7,330	16,160
1,750	35,786	19,323	217	117	24.5	28.2	7,215	15,906
1,725	41,189	22,240	194	105	25.6	28.5	7,115	15,686
1,700	45,170	24,390	197	106	25.9	28.8	7,005	15,443
1,675	46,000	24,838	197	106	24.6	28.8	6,880	15,168
1,650	46,000	24,838	198	107	23.0	28.8	6,700	14,771
1,625	46,000	24,838	220	119	21.4	29.9	6,460	14,242
1,600	46,000	24,838	359	194	20.3	31.0	6,175	13,614
1,575	46,000	24,838	519	280	19.3	30.5	5,890	12,985

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 531 - ΔV to GSO at 60,000 km (32,397 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	28.0	7,410	16,336
1,775	35,786	19,323	214	116	25.7	28.2	7,330	16,160
1,750	35,786	19,323	217	117	24.5	28.2	7,215	15,906
1,725	41,189	22,240	194	105	25.6	28.5	7,115	15,686
1,700	45,170	24,390	197	106	25.9	28.8	7,005	15,443
1,675	49,685	26,828	195	105	26.1	29.0	6,900	15,212
1,650	55,214	29,813	195	106	26.4	29.0	6,795	14,980
1,625	60,000	32,397	203	110	26.2	29.6	6,690	14,749
1,600	60,000	32,397	214	116	24.2	30.0	6,525	14,385
1,575	60,000	32,397	228	123	22.1	30.6	6,275	13,834
1,550	60,000	32,397	325	176	20.4	32.0	5,940	13,095
1,525	60,000	32,397	542	292	19.2	32.8	5,585	12,313

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2d: Atlas V 531 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 531 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	28.0	7,410	16,336
1,775	35,786	19,323	214	116	25.7	28.2	7,330	16,160
1,750	35,786	19,323	217	117	24.5	28.2	7,215	15,906
1,725	41,189	22,240	194	105	25.6	28.5	7,115	15,686
1,700	45,170	24,390	197	106	25.9	28.8	7,005	15,443
1,675	49,685	26,828	195	105	26.1	29.0	6,900	15,212
1,650	55,214	29,813	195	106	26.4	29.0	6,795	14,980
1,625	60,000	32,397	203	110	26.2	29.6	6,690	14,749
1,600	68,580	37,030	201	109	26.9	29.7	6,590	14,528
1,575	76,596	41,358	221	119	27.1	30.7	6,485	14,297
1,550	77,000	41,577	208	113	24.7	30.0	6,335	13,966
1,525	77,000	41,577	249	134	22.1	31.5	6,040	13,316
1,500	77,000	41,577	436	235	20.3	32.1	5,635	12,423

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 531 - ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	28.0	7,410	16,336
1,775	35,786	19,323	214	116	25.7	28.2	7,330	16,160
1,750	35,786	19,323	217	117	24.5	28.2	7,215	15,906
1,725	41,189	22,240	194	105	25.6	28.5	7,115	15,686
1,700	45,170	24,390	197	106	25.9	28.8	7,005	15,443
1,675	49,685	26,828	195	105	26.1	29.0	6,900	15,212
1,650	55,214	29,813	195	106	26.4	29.0	6,795	14,980
1,625	60,000	32,397	203	110	26.2	29.6	6,690	14,749
1,600	68,580	37,030	201	109	26.9	29.7	6,590	14,528
1,575	76,596	41,358	221	119	27.1	30.7	6,485	14,297
1,550	86,978	46,965	197	106	27.4	29.8	6,405	14,121
1,525	90,000	48,596	210	114	25.4	30.4	6,265	13,812
1,500	90,000	48,596	240	130	22.4	31.7	5,965	13,151
1,475	90,000	48,596	442	239	20.2	31.5	5,510	12,147

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.3-2e: Atlas V 541 Minimum Delta-V to Geosynchronous Orbit

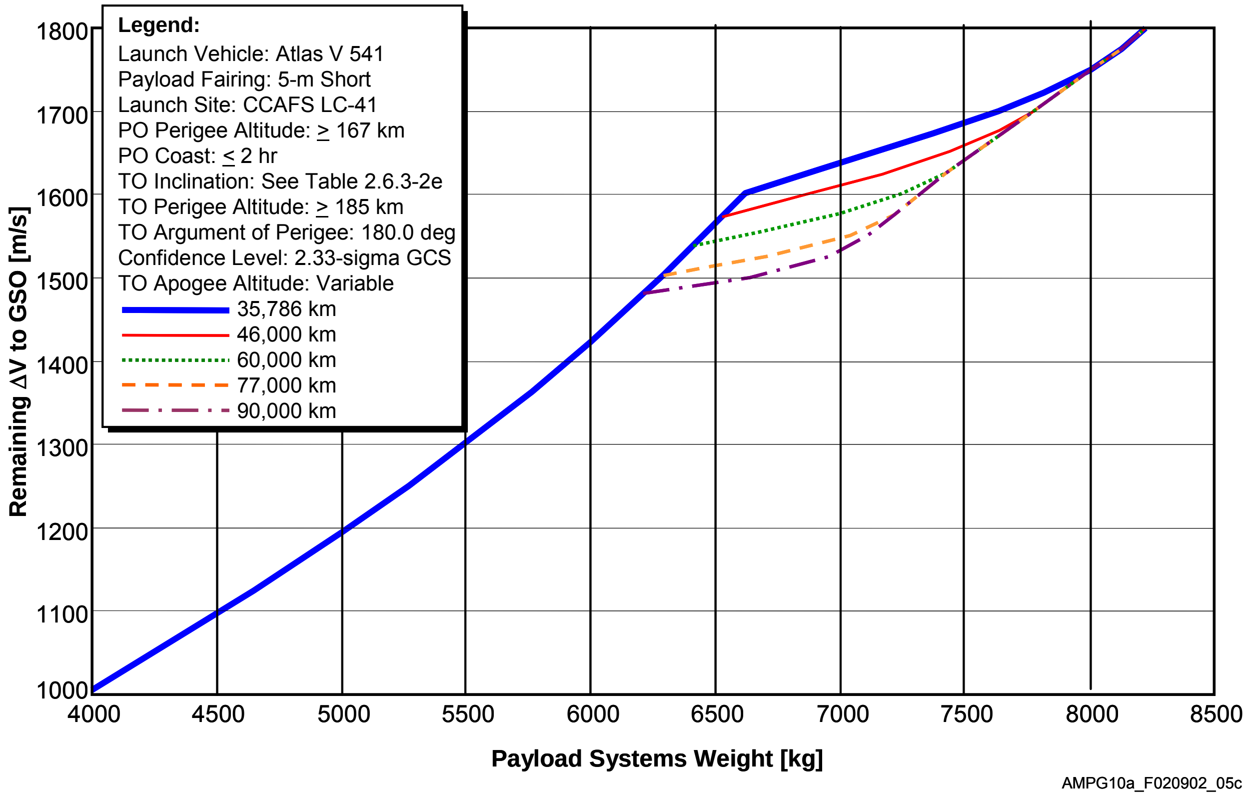


Table 2.6.3-2e: Atlas V 541 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 541 - ∆V to GSO at 35,786 km (19,323 nmi)						
∆V to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	190	103	26.8	28.9	8,225	18,133
1,775	214	116	25.7	28.9	8,135	17,935
1,750	217	117	24.5	29.0	8,005	17,648
1,725	230	124	23.3	28.9	7,840	17,284
1,700	233	126	22.0	28.9	7,635	16,832
1,675	240	130	20.7	28.1	7,390	16,292
1,650	298	161	19.6	27.7	7,125	15,708
1,625	475	257	19.0	28.2	6,855	15,113
1,600	4,228	2,283	26.9	133.9	6,615	14,584
1,575	4,601	2,484	26.5	132.4	6,535	14,407
1,550	4,942	2,668	26.0	130.9	6,450	14,220
1,525	5,256	2,838	25.6	129.6	6,365	14,032
1,500	5,634	3,042	25.2	128.1	6,280	13,845
1,475	6,005	3,242	24.7	126.5	6,190	13,647
1,450	6,333	3,419	24.3	125.1	6,100	13,448
1,425	6,653	3,592	23.8	123.7	6,005	13,239
1,400	7,005	3,782	23.4	122.1	5,910	13,029
1,375	7,364	3,976	22.9	120.4	5,810	12,809
1,350	7,698	4,157	22.5	118.8	5,705	12,577
1,325	7,977	4,307	21.9	117.6	5,600	12,346
1,300	8,188	4,421	21.4	116.7	5,490	12,103
1,275	8,473	4,575	20.9	115.2	5,380	11,861
1,250	8,842	4,774	20.4	113.3	5,265	11,607
1,225	9,066	4,895	19.9	112.1	5,145	11,343
1,200	9,304	5,024	19.3	110.8	5,025	11,078
1,175	9,537	5,150	18.8	109.8	4,900	10,803
1,150	9,825	5,305	18.3	108.1	4,775	10,527
1,125	9,925	5,359	17.6	107.2	4,650	10,251
1,100	10,206	5,511	17.1	105.4	4,515	9,954
1,075	10,423	5,628	16.5	104.3	4,385	9,667
1,050	10,676	5,765	16.0	102.6	4,250	9,370
1,025	10,911	5,891	15.4	101.3	4,120	9,083
1,000	11,071	5,978	14.8	99.9	3,980	8,774

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2e: Atlas V 541 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 541 - ΔV to GSO at 46,000 km (24,838 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	28.9	8,225	18,133
1,775	35,786	19,323	214	116	25.7	28.9	8,135	17,935
1,750	37,832	20,428	194	105	25.4	29.1	8,015	17,670
1,725	41,219	22,256	197	106	25.6	29.4	7,895	17,405
1,700	45,240	24,428	196	106	25.9	29.6	7,770	17,130
1,675	46,000	24,838	206	111	24.7	30.2	7,630	16,821
1,650	46,000	24,838	201	108	23.0	29.7	7,430	16,380
1,625	46,000	24,838	202	109	21.3	29.5	7,170	15,807
1,600	46,000	24,838	319	172	20.1	29.2	6,860	15,124
1,575	46,000	24,838	561	303	19.5	30.6	6,550	14,440
1,550	46,000	24,838	882	476	19.2	33.9	6,260	13,801

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours — Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 541 - ΔV to GSO at 60,000 km (32,397 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	28.9	8,225	18,133
1,775	35,786	19,323	214	116	25.7	28.9	8,135	17,935
1,750	37,832	20,428	194	105	25.4	29.1	8,015	17,670
1,725	41,219	22,256	197	106	25.6	29.4	7,895	17,405
1,700	45,240	24,428	196	106	25.9	29.6	7,770	17,130
1,675	49,840	26,911	198	107	26.1	29.9	7,655	16,876
1,650	55,289	29,854	198	107	26.4	30.1	7,535	16,612
1,625	60,000	32,397	200	108	26.2	30.3	7,420	16,358
1,600	60,000	32,397	203	109	24.2	30.4	7,245	15,972
1,575	60,000	32,397	205	111	22.0	30.3	6,965	15,355
1,550	60,000	32,397	325	176	20.4	29.9	6,595	14,539
1,525	60,000	32,397	697	376	20.0	33.7	6,235	13,746

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours — Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2e: Atlas V 541 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 541 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	28.9	8,225	18,133
1,775	35,786	19,323	214	116	25.7	28.9	8,135	17,935
1,750	37,832	20,428	194	105	25.4	29.1	8,015	17,670
1,725	41,219	22,256	197	106	25.6	29.4	7,895	17,405
1,700	45,240	24,428	196	106	25.9	29.6	7,770	17,130
1,675	49,840	26,911	198	107	26.1	29.9	7,655	16,876
1,650	55,289	29,854	198	107	26.4	30.1	7,535	16,612
1,625	60,000	32,397	200	108	26.2	30.3	7,420	16,358
1,600	68,688	37,089	200	108	26.9	30.5	7,315	16,127
1,575	77,000	41,577	200	108	27.1	30.6	7,210	15,895
1,550	77,000	41,577	210	113	24.7	30.9	7,030	15,498
1,525	77,000	41,577	231	125	22.1	31.8	6,700	14,771
1,500	77,000	41,577	454	245	20.4	31.7	6,260	13,801

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours — Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 541 - ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	28.9	8,225	18,133
1,775	35,786	19,323	214	116	25.7	28.9	8,135	17,935
1,750	37,832	20,428	194	105	25.4	29.1	8,015	17,670
1,725	41,219	22,256	197	106	25.6	29.4	7,895	17,405
1,700	45,240	24,428	196	106	25.9	29.6	7,770	17,130
1,675	49,840	26,911	198	107	26.1	29.9	7,655	16,876
1,650	55,289	29,854	198	107	26.4	30.1	7,535	16,612
1,625	60,000	32,397	200	108	26.2	30.3	7,420	16,358
1,600	68,688	37,089	200	108	26.9	30.5	7,315	16,127
1,575	77,000	41,577	200	108	27.1	30.6	7,210	15,895
1,550	87,048	47,002	200	108	27.4	30.8	7,105	15,664
1,525	90,000	48,596	205	111	25.4	30.9	6,955	15,333
1,500	90,000	48,596	210	114	22.3	31.0	6,620	14,595
1,475	90,000	48,596	456	246	20.3	31.7	6,125	13,503

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours — Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.3-2f: Atlas V 551 Minimum Delta-V to Geosynchronous Orbit

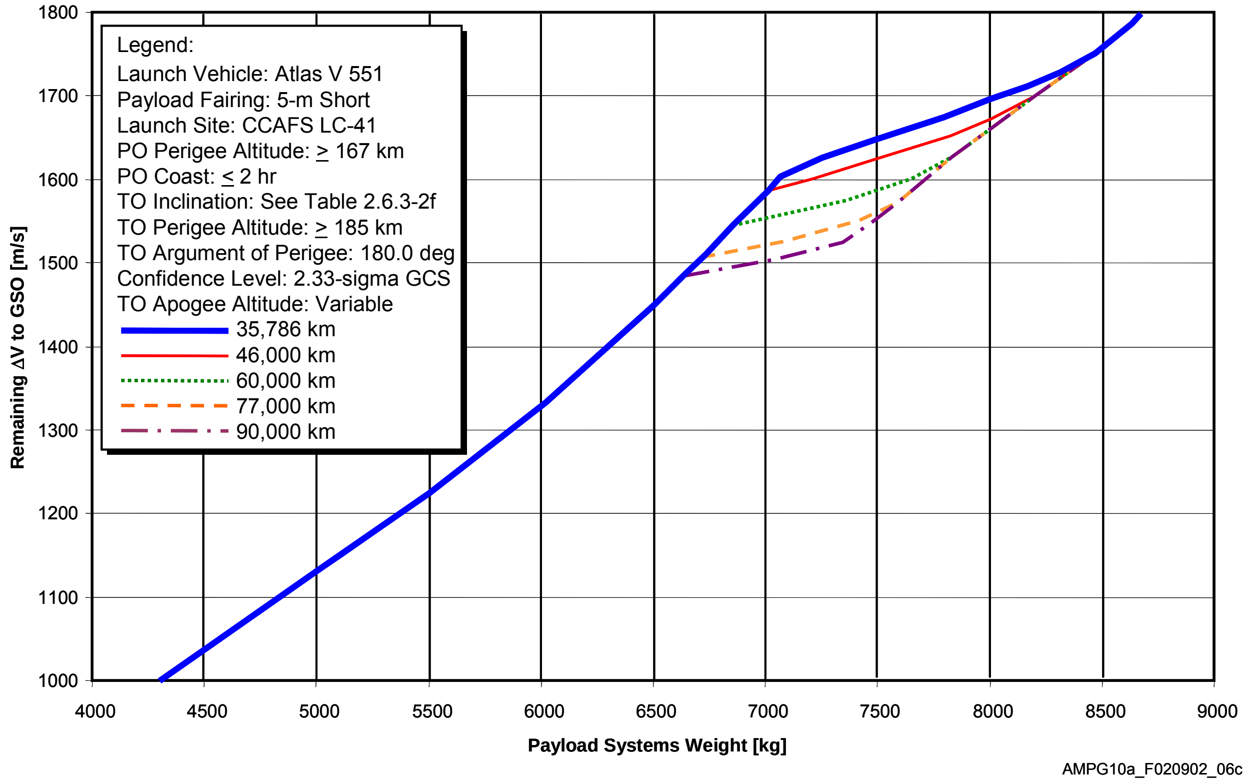


Table 2.6.3-2f: Atlas V 551 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 3)

Atlas V 551 - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	190	103	26.8	29.6	8,685	19,147
1,775	214	116	25.7	30.0	8,585	18,927
1,750	217	117	24.5	30.0	8,455	18,640
1,725	230	124	23.3	29.7	8,275	18,243
1,700	233	126	22.0	29.6	8,055	17,758
1,675	240	130	20.7	29.5	7,795	17,185
1,650	325	175	19.7	28.8	7,515	16,568
1,625	540	291	19.3	29.9	7,245	15,972
1,600	4,154	2,243	26.8	132.6	7,055	15,554
1,575	4,522	2,442	26.4	132.3	6,965	15,355
1,550	4,832	2,609	25.9	131.6	6,875	15,157
1,525	5,228	2,823	25.5	130.0	6,785	14,958
1,500	5,591	3,019	25.1	128.5	6,695	14,760
1,475	5,917	3,195	24.7	127.1	6,600	14,550
1,450	6,281	3,391	24.2	125.5	6,505	14,341
1,425	6,606	3,567	23.8	124.1	6,410	14,132
1,400	6,946	3,751	23.3	122.6	6,305	13,900
1,375	7,328	3,957	22.9	120.8	6,200	13,669
1,350	7,696	4,155	22.4	119.1	6,095	13,437
1,325	7,949	4,292	21.9	117.9	5,985	13,195
1,300	8,365	4,517	21.5	115.8	5,865	12,930
1,275	8,572	4,628	20.9	114.9	5,750	12,677
1,250	8,817	4,761	20.4	113.6	5,630	12,412
1,225	9,071	4,898	19.9	112.3	5,505	12,136
1,200	9,449	5,102	19.4	110.1	5,375	11,850
1,175	9,545	5,154	18.8	109.7	5,245	11,563
1,150	9,754	5,267	18.2	108.5	5,115	11,277
1,125	9,981	5,389	17.6	107.3	4,980	10,979
1,100	10,183	5,499	17.0	106.0	4,845	10,681
1,075	10,390	5,610	16.5	104.8	4,705	10,373
1,050	10,616	5,732	15.9	103.1	4,565	10,064
1,025	10,822	5,843	15.3	101.7	4,430	9,766
1,000	10,987	5,932	14.7	100.8	4,295	9,469

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2f: Atlas V 551 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 3)

Atlas V 551 - ΔV to GSO at 46,000 km (24,838 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	29.6	8,685	19,147
1,775	35,786	19,323	214	116	25.7	30.0	8,585	18,927
1,750	37,710	20,362	196	106	25.4	29.8	8,460	18,651
1,725	40,938	22,105	202	109	25.5	30.3	8,330	18,364
1,700	45,099	24,352	200	108	25.8	30.4	8,205	18,089
1,675	46,000	24,838	201	109	24.7	30.9	8,025	17,692
1,650	46,000	24,838	202	109	23.0	30.7	7,790	17,174
1,625	46,000	24,838	206	111	21.3	31.0	7,495	16,524
1,600	46,000	24,838	319	172	20.1	30.1	7,180	15,829
1,575	46,000	24,838	561	303	19.5	30.7	6,865	15,135

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 551 - ΔV to GSO at 60,000 km (32,397 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	29.6	8,685	19,147
1,775	35,786	19,323	214	116	25.7	30.0	8,585	18,927
1,750	37,710	20,362	196	106	25.4	29.8	8,460	18,651
1,725	40,938	22,105	202	109	25.5	30.3	8,330	18,364
1,700	45,099	24,352	200	108	25.8	30.4	8,205	18,089
1,675	49,745	26,860	201	109	26.1	30.6	8,075	17,802
1,650	55,167	29,788	202	109	26.4	30.8	7,950	17,527
1,625	60,000	32,397	202	109	26.2	31.0	7,825	17,251
1,600	60,000	32,397	214	115	24.2	31.4	7,635	16,832
1,575	60,000	32,397	220	119	22.1	31.6	7,345	16,193
1,550	60,000	32,397	338	183	20.4	30.7	6,960	15,344
1,525	60,000	32,397	586	316	19.4	30.7	6,570	14,484

Notes:
Launch Site: CCAFS LC-41
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Park Orbit Coast ≤ 2 Hours
Confidence Level: 2.33 Sigma GCS
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-2f: Atlas V 551 Minimum Delta-V to Geosynchronous Orbit – Apogee Cap (3 of 3)

Atlas V 551 - ΔV to GSO at 77,000 km (41,577 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	29.6	8,685	19,147
1,775	35,786	19,323	214	116	25.7	30.0	8,585	18,927
1,750	37,710	20,362	196	106	25.4	29.8	8,460	18,651
1,725	40,938	22,105	202	109	25.5	30.3	8,330	18,364
1,700	45,099	24,352	200	108	25.8	30.4	8,205	18,089
1,675	49,745	26,860	201	109	26.1	30.6	8,075	17,802
1,650	55,167	29,788	202	109	26.4	30.8	7,950	17,527
1,625	60,000	32,397	202	109	26.2	31.0	7,825	17,251
1,600	68,478	36,975	203	110	26.9	31.2	7,710	16,998
1,575	77,000	41,577	205	111	27.2	31.6	7,595	16,744
1,550	77,000	41,577	215	116	24.7	31.8	7,415	16,347
1,525	77,000	41,577	230	124	22.0	32.2	7,070	15,587
1,500	77,000	41,577	475	256	20.5	32.4	6,605	14,562

Notes:
Launch Site: CCAFS LC-41 Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Atlas V 551 - ΔV to GSO at 90,000 km (48,596 nmi) Cap								
ΔV to GSO	Apogee		Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,800	35,786	19,323	190	103	26.8	29.6	8,685	19,147
1,775	35,786	19,323	214	116	25.7	30.0	8,585	18,927
1,750	37,710	20,362	196	106	25.4	29.8	8,460	18,651
1,725	40,938	22,105	202	109	25.5	30.3	8,330	18,364
1,700	45,099	24,352	200	108	25.8	30.4	8,205	18,089
1,675	49,745	26,860	201	109	26.1	30.6	8,075	17,802
1,650	55,167	29,788	202	109	26.4	30.8	7,950	17,527
1,625	60,000	32,397	202	109	26.2	31.0	7,825	17,251
1,600	68,478	36,975	203	110	26.9	31.2	7,710	16,998
1,575	77,000	41,577	205	111	27.2	31.6	7,595	16,744
1,550	86,866	46,904	204	110	27.4	31.6	7,490	16,513
1,525	90,000	48,595	207	112	25.4	31.6	7,335	16,171
1,500	90,000	48,595	279	151	22.6	33.6	6,965	15,355
1,475	90,000	48,595	499	270	20.5	32.7	6,460	14,242

Notes:
Launch Site: CCAFS LC-41 Confidence Level: 2.33 Sigma GCS
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Park Orbit Coast ≤ 2 Hours Transfer Orbit Argument Perigee = 180 deg
Apogee is at 1st Spacecraft Apogee, Perigee and Inclination are at Spacecraft Separation, True Anomaly is at Injection
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-3: Atlas V 5Y1 3-Burn Delta-V to Geosynchronous Orbit Summary

Atlas V 5Y1 3-Burn - ΔV to GSO at 35,786 km (19,323 nmi)								
ΔV to GSO	Payload Systems Weight							
	521 (Table 2.6.3-3a)		531 (Table 2.6.3-3b)		541 (Table 2.6.3-3c)		551 (Table 2.6.3-3d)	
[m/s]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
1,000	4,295	9,469	5,055	11,144	5,715	12,599	6,090	13,426
975	4,255	9,381	5,005	11,034	5,655	12,467	6,030	13,294
950	4,210	9,281	4,950	10,913	5,600	12,346	5,970	13,162
925	4,170	9,193	4,900	10,803	5,545	12,225	5,905	13,018
900	4,130	9,105	4,850	10,692	5,485	12,092	5,845	12,886
875	4,090	9,017	4,800	10,582	5,430	11,971	5,785	12,754
850	4,050	8,929	4,750	10,472	5,375	11,850	5,725	12,621
825	4,010	8,841	4,700	10,362	5,320	11,729	5,665	12,489
800	3,965	8,741	4,650	10,251	5,265	11,607	5,610	12,368
775	3,925	8,653	4,600	10,141	5,210	11,486	5,550	12,236
750	3,885	8,565	4,550	10,031	5,160	11,376	5,495	12,114
725	3,845	8,477	4,505	9,932	5,105	11,255	5,435	11,982
700	3,805	8,389	4,455	9,822	5,050	11,133	5,380	11,861
675	3,770	8,311	4,410	9,722	5,000	11,023	5,325	11,740
650	3,730	8,223	4,360	9,612	4,950	10,913	5,265	11,607
625	3,690	8,135	4,315	9,513	4,895	10,792	5,210	11,486
600	3,650	8,047	4,265	9,403	4,845	10,681	5,155	11,365
575	3,610	7,959	4,220	9,303	4,795	10,571	5,100	11,244
550	3,575	7,882	4,175	9,204	4,745	10,461	5,050	11,133
525	3,535	7,793	4,130	9,105	4,695	10,351	4,995	11,012
500	3,495	7,705	4,085	9,006	4,645	10,240	4,940	10,891
475	3,455	7,617	4,040	8,907	4,595	10,130	4,890	10,781
450	3,420	7,540	3,995	8,807	4,545	10,020	4,835	10,659
425	3,380	7,452	3,950	8,708	4,500	9,921	4,785	10,549
400	3,345	7,374	3,910	8,620	4,450	9,811	4,735	10,439
375	3,305	7,286	3,865	8,521	4,400	9,700	4,680	10,318
350	3,270	7,209	3,820	8,422	4,355	9,601	4,630	10,207
325	3,230	7,121	3,780	8,333	4,310	9,502	4,580	10,097
300	3,195	7,044	3,735	8,234	4,260	9,392	4,530	9,987
275	3,155	6,956	3,695	8,146	4,215	9,292	4,480	9,877
250	3,120	6,878	3,655	8,058	4,170	9,193	4,430	9,766
225	3,085	6,801	3,610	7,959	4,125	9,094	4,385	9,667
200	3,045	6,713	3,570	7,870	4,080	8,995	4,335	9,557
175	3,010	6,636	3,530	7,782	4,035	8,896	4,285	9,447
150	2,975	6,559	3,490	7,694	3,990	8,796	4,240	9,348
125	2,940	6,482	3,450	7,606	3,945	8,697	4,190	9,237
100	2,905	6,404	3,410	7,518	3,900	8,598	4,145	9,138
75	2,865	6,316	3,370	7,430	3,860	8,510	4,100	9,039
50	2,830	6,239	3,335	7,352	3,815	8,411	4,050	8,929
25	2,795	6,162	3,295	7,264	3,770	8,311	4,005	8,830
0	2,760	6,085	3,255	7,176	3,730	8,223	3,960	8,730

Figure 2.6.3-3: Atlas V 5X1 3-Burn Delta-V to Geosynchronous Orbit

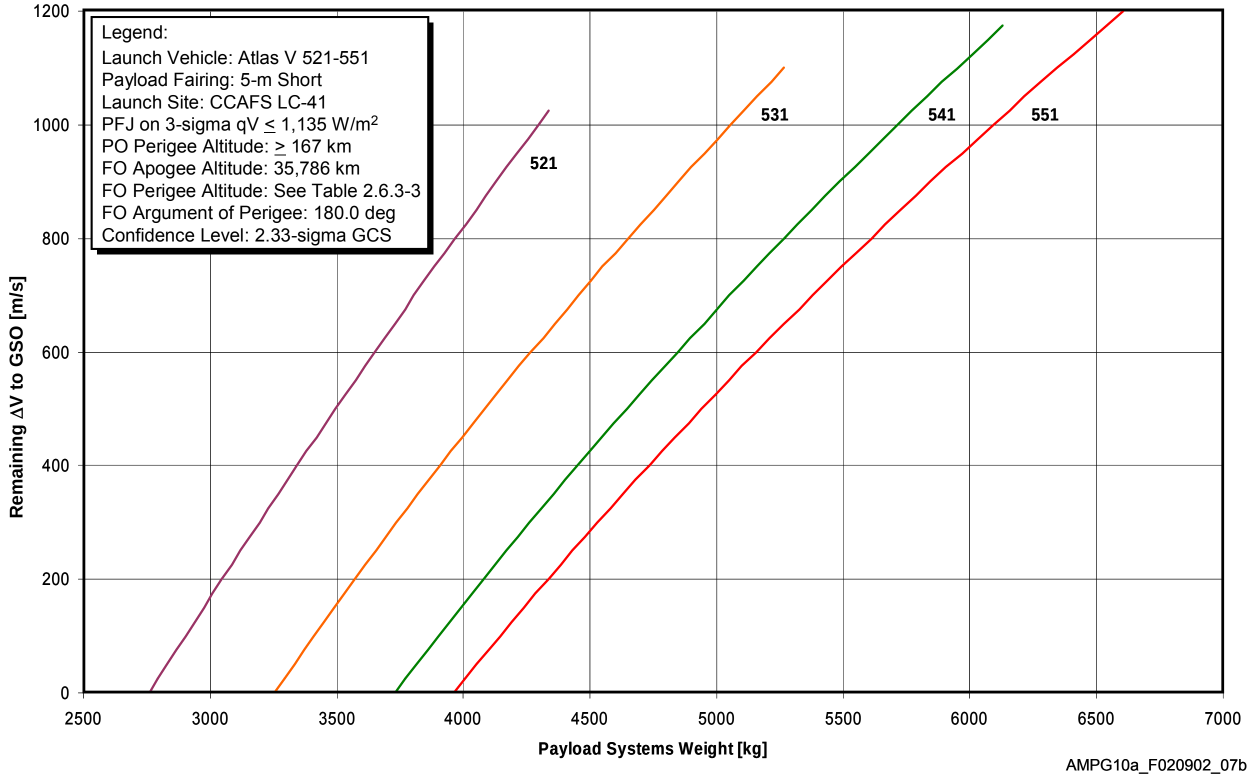


Table 2.6.3-3a: Atlas V 521 3-Burn Delta-V to Geosynchronous Orbit – Apogee Cap

Atlas V 521 3-Burn - ∆V to GSO at 35,786 km (19,323 nmi)						
∆V to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,025	7,608	4,108	10.7	179.7	4,335	9,557
1,000	7,971	4,304	10.3	179.7	4,295	9,469
975	8,345	4,506	10.0	179.7	4,255	9,381
950	8,731	4,714	9.7	179.5	4,210	9,281
925	9,118	4,924	9.3	179.7	4,170	9,193
900	9,512	5,136	9.0	179.7	4,130	9,105
875	9,931	5,362	8.6	179.8	4,090	9,017
850	10,365	5,597	8.3	179.9	4,050	8,929
825	10,787	5,824	8.0	179.5	4,010	8,841
800	11,236	6,067	7.7	179.5	3,965	8,741
775	11,703	6,319	7.4	179.6	3,925	8,653
750	12,171	6,572	7.1	179.6	3,885	8,565
725	12,653	6,832	6.8	179.7	3,845	8,477
700	13,159	7,105	6.5	179.7	3,805	8,389
675	13,663	7,377	6.2	179.9	3,770	8,311
650	14,202	7,668	5.9	-180.0	3,730	8,223
625	14,754	7,966	5.6	-179.9	3,690	8,135
600	15,299	8,261	5.3	-179.7	3,650	8,047
575	15,888	8,579	5.1	-179.9	3,610	7,959
550	16,493	8,906	4.8	-179.9	3,575	7,882
525	17,090	9,228	4.6	-179.9	3,535	7,793
500	17,724	9,570	4.3	-179.8	3,495	7,705
475	18,390	9,930	4.1	-179.6	3,455	7,617
450	19,047	10,285	3.8	-179.9	3,420	7,540
425	19,716	10,646	3.5	-179.9	3,380	7,452
400	20,451	11,043	3.3	-180.0	3,345	7,374
375	21,184	11,439	3.1	-179.7	3,305	7,286
350	21,962	11,858	2.9	-179.9	3,270	7,209
325	22,733	12,275	2.6	-179.7	3,230	7,121
300	23,543	12,712	2.4	-179.9	3,195	7,044
275	24,373	13,160	2.2	179.8	3,155	6,956
250	25,248	13,633	2.0	179.7	3,120	6,878
225	26,154	14,122	1.8	179.9	3,085	6,801
200	27,110	14,638	1.6	179.0	3,045	6,713
175	28,020	15,129	1.4	-179.8	3,010	6,636
150	29,018	15,668	1.2	179.4	2,975	6,559
125	30,039	16,220	0.9	179.7	2,940	6,482
100	31,084	16,784	0.7	179.6	2,905	6,404
75	32,214	17,394	0.6	179.6	2,865	6,316
50	33,354	18,010	0.4	179.1	2,830	6,239
25	34,543	18,652	0.2	178.6	2,795	6,162
0	35,786	19,323	0.0	--	2,760	6,085

Notes:
Launch Site: CCAFS LC-41
Confidence Level: 2.33 Sigma GCS
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
FO Apogee Altitude: 35,786 km
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
FO Argument Perigee = 180 deg
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-3b: Atlas V 531 3-Burn Delta-V to Geosynchronous Orbit – Apogee Cap

Atlas V 531 3-Burn - ΔV to GSO at 35,786 km (19,323 nmi) Cap						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,100	6,578	3,552	11.8	179.6	5,265	11,607
1,075	6,959	3,758	11.6	179.6	5,215	11,497
1,050	7,258	3,919	11.1	179.7	5,160	11,376
1,025	7,606	4,107	10.7	179.7	5,105	11,255
1,000	7,972	4,305	10.4	179.7	5,055	11,144
975	8,351	4,509	10.0	179.8	5,005	11,034
950	8,720	4,709	9.6	179.8	4,950	10,913
925	9,115	4,922	9.3	179.8	4,900	10,803
900	9,524	5,142	9.0	179.8	4,850	10,692
875	9,939	5,367	8.6	179.9	4,800	10,582
850	10,355	5,591	8.3	179.9	4,750	10,472
825	10,775	5,818	8.0	179.9	4,700	10,362
800	11,237	6,068	7.7	180.0	4,650	10,251
775	11,700	6,317	7.4	179.9	4,600	10,141
750	12,181	6,577	7.1	-180.0	4,550	10,031
725	12,658	6,835	6.8	-180.0	4,505	9,932
700	13,158	7,105	6.5	-180.0	4,455	9,822
675	13,711	7,403	6.2	179.9	4,410	9,722
650	14,207	7,671	5.9	-180.0	4,360	9,612
625	14,749	7,964	5.6	-180.0	4,315	9,513
600	15,315	8,269	5.4	-179.7	4,265	9,403
575	15,895	8,582	5.1	-179.8	4,220	9,303
550	16,486	8,902	4.8	-179.8	4,175	9,204
525	17,095	9,231	4.6	-179.8	4,130	9,105
500	17,731	9,574	4.3	-179.8	4,085	9,006
475	18,394	9,932	4.1	-179.8	4,040	8,907
450	19,026	10,273	3.8	-179.7	3,995	8,807
425	19,739	10,658	3.6	-179.9	3,950	8,708
400	20,453	11,044	3.3	-179.7	3,910	8,620
375	21,185	11,439	3.1	-179.8	3,865	8,521
350	21,956	11,855	2.9	-179.8	3,820	8,422
325	22,736	12,276	2.6	-179.8	3,780	8,333
300	23,523	12,701	2.4	-179.9	3,735	8,234
275	24,369	13,158	2.2	-179.8	3,695	8,146
250	25,231	13,623	2.0	179.3	3,655	8,058
225	26,152	14,121	1.8	179.4	3,610	7,959
200	27,054	14,608	1.5	179.4	3,570	7,870
175	28,022	15,131	1.4	179.3	3,530	7,782
150	29,029	15,674	1.2	179.4	3,490	7,694
125	30,042	16,221	1.0	179.2	3,450	7,606
100	31,099	16,792	0.7	179.0	3,410	7,518
75	32,213	17,394	0.6	178.8	3,370	7,430
50	33,357	18,011	0.4	178.6	3,335	7,352
25	34,546	18,653	0.2	179.0	3,295	7,264
0	35,786	19,323	0.0	--	3,255	7,176

Notes:
Launch Site: CCAFS LC-41
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 2.33 Sigma GCS
FO Apogee Altitude: 35,786 km
FO Argument Perigee = 180 deg
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-3c: Atlas V 541 3-Burn Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 2)

Atlas V 541 3-Burn - ∆V to GSO at 35,786 km (19,323 nmi)						
∆V to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,175	5,645	3,048	13.1	179.4	6,130	13,514
1,150	5,934	3,204	12.6	179.4	6,070	13,382
1,125	6,242	3,370	12.2	179.4	6,010	13,250
1,100	6,753	3,646	12.2	-178.6	5,950	13,117
1,075	6,979	3,769	11.6	-178.9	5,890	12,985
1,050	7,253	3,916	11.1	-179.0	5,830	12,853
1,025	7,620	4,115	10.7	-178.7	5,770	12,721
1,000	7,984	4,311	10.4	-178.9	5,715	12,599
975	8,358	4,513	10.0	-180.0	5,655	12,467
950	8,726	4,712	9.6	-179.6	5,600	12,346
925	9,118	4,923	9.3	179.9	5,545	12,225
900	9,523	5,142	9.0	179.9	5,485	12,092
875	9,937	5,365	8.6	179.9	5,430	11,971
850	10,361	5,594	8.3	179.9	5,375	11,850
825	10,798	5,831	8.0	180.0	5,320	11,729
800	11,235	6,066	7.7	180.0	5,265	11,607
775	11,700	6,317	7.4	-179.9	5,210	11,486
750	12,173	6,573	7.1	179.8	5,160	11,376
725	12,655	6,833	6.8	-179.9	5,105	11,255
700	13,164	7,108	6.5	-179.9	5,050	11,133
675	13,679	7,386	6.2	179.9	5,000	11,023
650	14,200	7,668	5.9	-179.9	4,950	10,913
625	14,747	7,963	5.6	-179.9	4,895	10,792
600	15,312	8,268	5.4	180.0	4,845	10,681
575	15,887	8,579	5.1	-179.8	4,795	10,571
550	16,479	8,898	4.8	179.7	4,745	10,461
525	17,143	9,256	4.6	178.7	4,695	10,351
500	17,845	9,635	4.5	179.9	4,645	10,240
475	18,373	9,921	4.1	179.7	4,595	10,130
450	19,041	10,282	3.8	179.9	4,545	10,020
425	19,742	10,660	3.6	179.8	4,500	9,921
400	20,447	11,041	3.3	179.9	4,450	9,811
375	21,195	11,444	3.1	179.7	4,400	9,700
350	21,933	11,843	2.8	179.6	4,355	9,601
326	22,662	12,236	2.6	-179.1	4,310	9,502
300	23,543	12,712	2.4	180.0	4,260	9,392
275	24,378	13,163	2.2	-179.9	4,215	9,292
250	25,270	13,645	2.0	-179.7	4,170	9,193

Notes:
Launch Site: CCAFS LC-41
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 2.33 Sigma GCS
FO Apogee Altitude: 35,786 km
FO Argument Perigee = 180 deg
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-3c: Atlas V 541 3-Burn Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 2)

Atlas V 541 3-Burn - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
225	26,141	14,115	1.8	179.0	4,125	9,094
200	27,068	14,615	1.6	179.5	4,080	8,995
175	28,027	15,134	1.4	179.4	4,035	8,896
150	29,012	15,665	1.1	179.3	3,990	8,796
125	30,001	16,199	0.9	179.6	3,945	8,697
100	31,103	16,794	0.8	179.3	3,900	8,598
75	32,216	17,395	0.6	-179.8	3,860	8,510
50	33,360	18,013	0.4	-179.7	3,815	8,411
25	34,549	18,655	0.2	-179.4	3,770	8,311
0	35,786	19,323	0.0	--	3,730	8,223

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr) — FO Apogee Altitude: 35,786 km
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — FO Argument Perigee = 180 deg
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-3d: Atlas V 551 3-Burn Delta-V to Geosynchronous Orbit – Apogee Cap (1 of 2)

Atlas V 551 3-Burn - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
1,200	5,352	2,890	13.5	179.5	6,605	14,562
1,175	5,651	3,051	13.1	179.5	6,540	14,418
1,150	5,960	3,218	12.7	179.6	6,475	14,275
1,125	6,273	3,387	12.3	179.6	6,410	14,132
1,100	6,601	3,564	11.9	179.7	6,345	13,988
1,075	6,935	3,745	11.5	179.7	6,280	13,845
1,050	7,272	3,926	11.1	179.7	6,215	13,702
1,025	7,623	4,116	10.7	179.8	6,155	13,569
1,000	7,984	4,311	10.4	179.7	6,090	13,426
975	8,356	4,512	10.0	179.8	6,030	13,294
950	8,735	4,716	9.7	179.9	5,970	13,162
925	9,125	4,927	9.3	179.7	5,905	13,018
900	9,530	5,146	9.0	179.8	5,845	12,886
875	9,938	5,366	8.6	179.9	5,785	12,754
850	10,363	5,596	8.3	179.9	5,725	12,621
825	10,795	5,829	8.0	-180.0	5,665	12,489
800	11,252	6,075	7.7	180.0	5,610	12,368

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr) — FO Apogee Altitude: 35,786 km
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — FO Argument Perigee = 180 deg
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Table 2.6.3-3d: Atlas V 551 3-Burn Delta-V to Geosynchronous Orbit – Apogee Cap (2 of 2)

Atlas V 551 3-Burn - ΔV to GSO at 35,786 km (19,323 nmi)						
ΔV to GSO	Perigee		Inclination	True Anomaly	Payload Systems Weight	
[m/s]	[km]	[nmi]	[deg]	[deg]	[kg]	[lb]
775	11,701	6,318	7.4	180.0	5,550	12,236
750	12,185	6,580	7.1	-180.0	5,495	12,114
725	12,674	6,843	6.8	-179.9	5,435	11,982
700	13,163	7,108	6.5	-179.9	5,380	11,861
675	13,675	7,384	6.2	-179.8	5,325	11,740
650	14,210	7,673	5.9	-179.9	5,265	11,607
625	14,754	7,967	5.6	-179.9	5,210	11,486
600	15,321	8,272	5.4	-179.8	5,155	11,365
575	15,907	8,589	5.1	-179.8	5,100	11,244
550	16,489	8,903	4.8	-179.8	5,050	11,133
525	17,097	9,231	4.6	-179.7	4,995	11,012
500	17,733	9,575	4.3	-179.9	4,940	10,891
475	18,384	9,927	4.1	-179.8	4,890	10,781
450	19,060	10,291	3.8	-179.7	4,835	10,659
425	19,741	10,659	3.6	-179.8	4,785	10,549
400	20,447	11,040	3.3	-179.7	4,735	10,439
375	21,197	11,445	3.1	-179.7	4,680	10,318
350	21,953	11,854	2.9	-179.6	4,630	10,207
325	22,735	12,276	2.6	-179.7	4,580	10,097
300	23,543	12,712	2.4	-179.8	4,530	9,987
275	24,385	13,167	2.2	-179.7	4,480	9,877
250	25,249	13,633	2.0	-179.5	4,430	9,766
225	26,146	14,118	1.8	-179.6	4,385	9,667
200	27,073	14,618	1.6	-179.6	4,335	9,557
175	28,022	15,130	1.4	-179.6	4,285	9,447
150	29,020	15,669	1.2	-179.5	4,240	9,348
125	30,042	16,221	1.0	-179.3	4,190	9,237
100	31,108	16,797	0.8	-179.5	4,145	9,138
75	32,214	17,394	0.6	-179.4	4,100	9,039
50	33,361	18,013	0.4	-179.6	4,050	8,929
25	34,548	18,654	0.2	-179.4	4,005	8,830
0	35,786	19,323	0.0	--	3,960	8,730

Notes:
Launch Site: CCAFS LC-41 — Confidence Level: 2.33 Sigma GCS
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr) — FO Apogee Altitude: 35,786 km
Park Orbit Perigee Altitude ≥ 167 km (90 nmi) — FO Argument Perigee = 180 deg
All parameters are at Spacecraft Separation except Apogee, which is at 1st Spacecraft Apogee
Only oblate Earth effects were taken into account when propagating to 1st Spacecraft Apogee.

Figure 2.6.4-1a: Atlas V 401 Earth Escape Performance (C3 Curves) - CCAFS

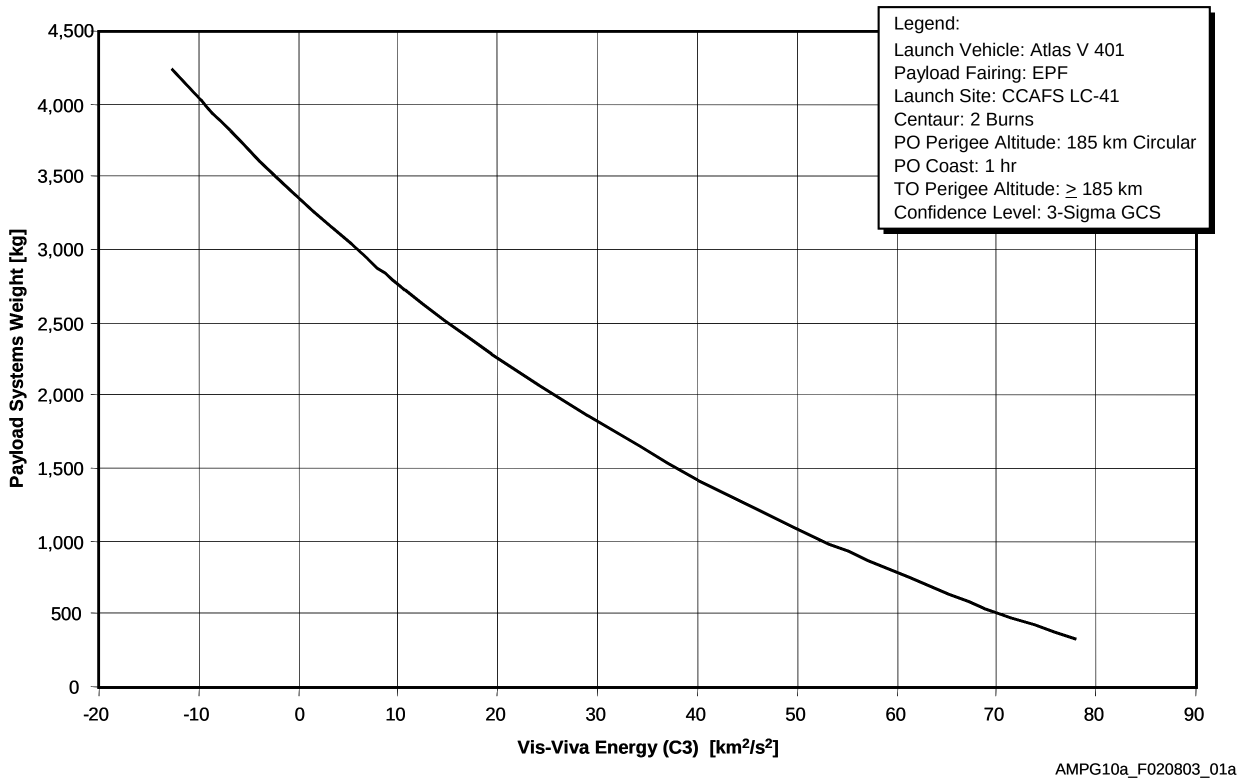


Figure 2.6.4-1b: Atlas V 411 Earth Escape Performance (C3 Curves) – CCAFS

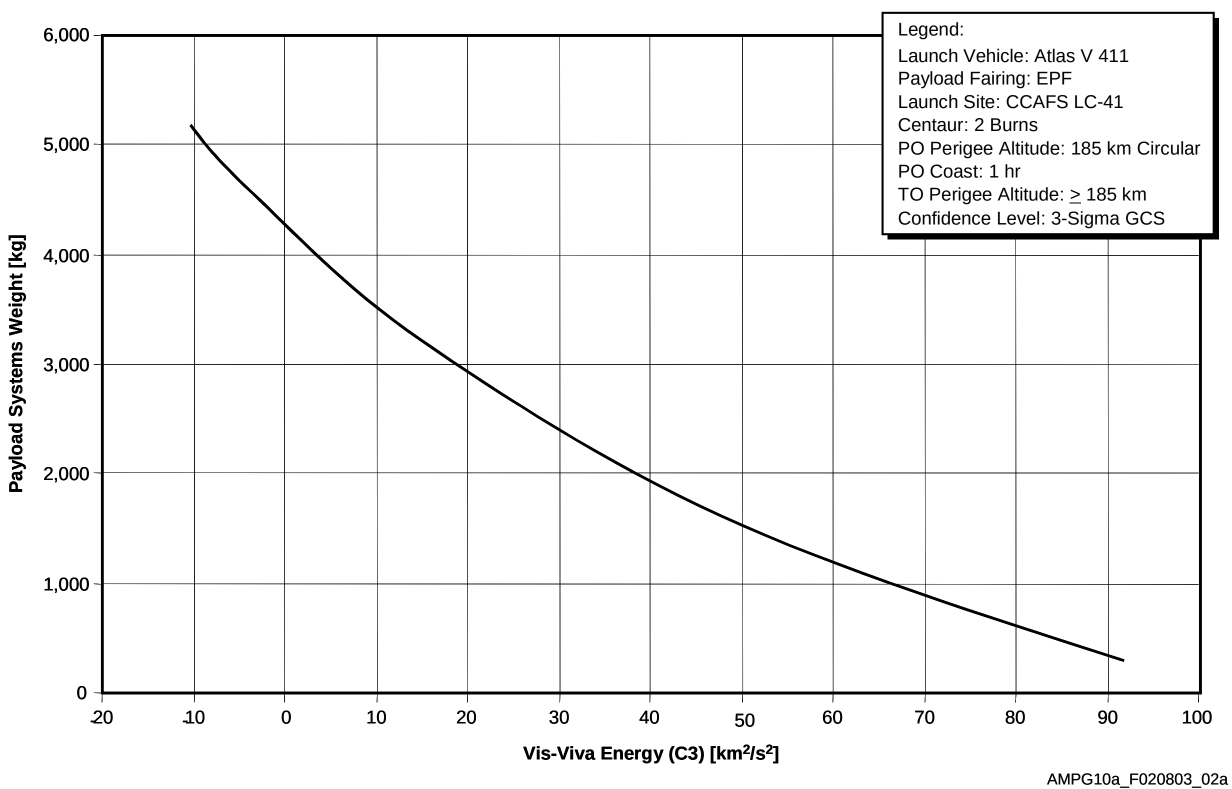
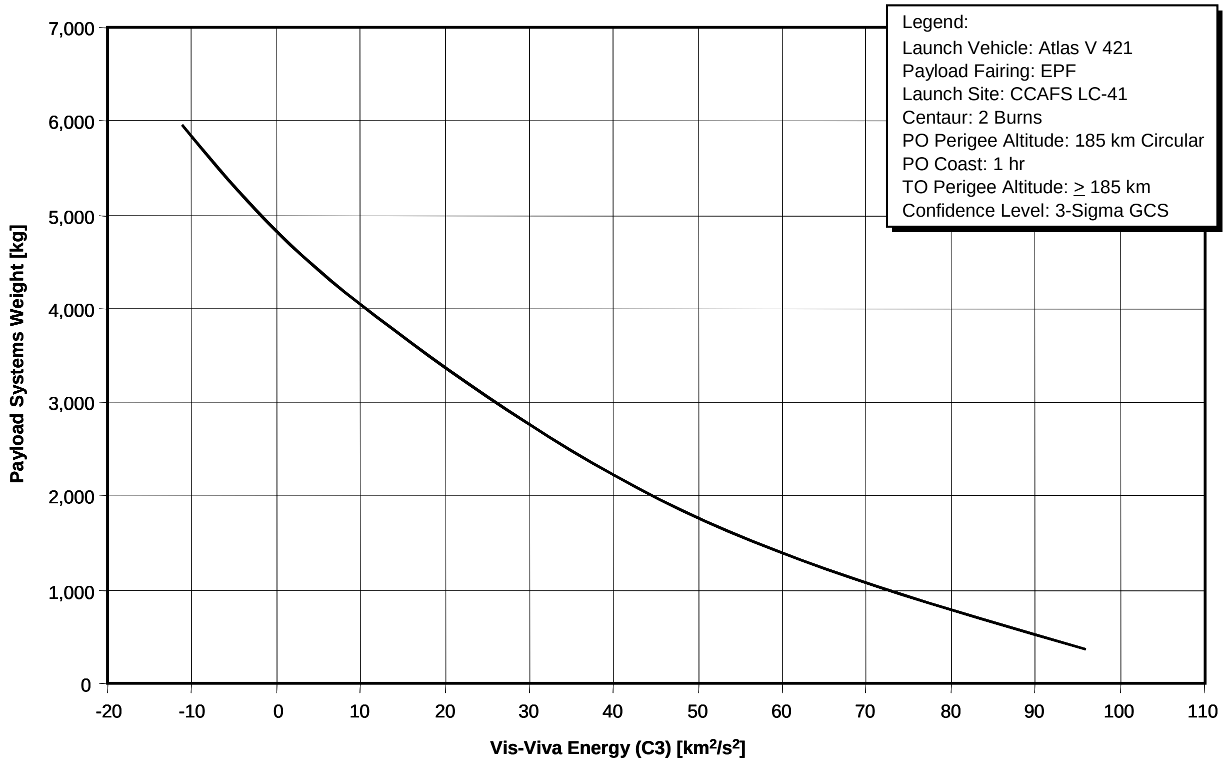
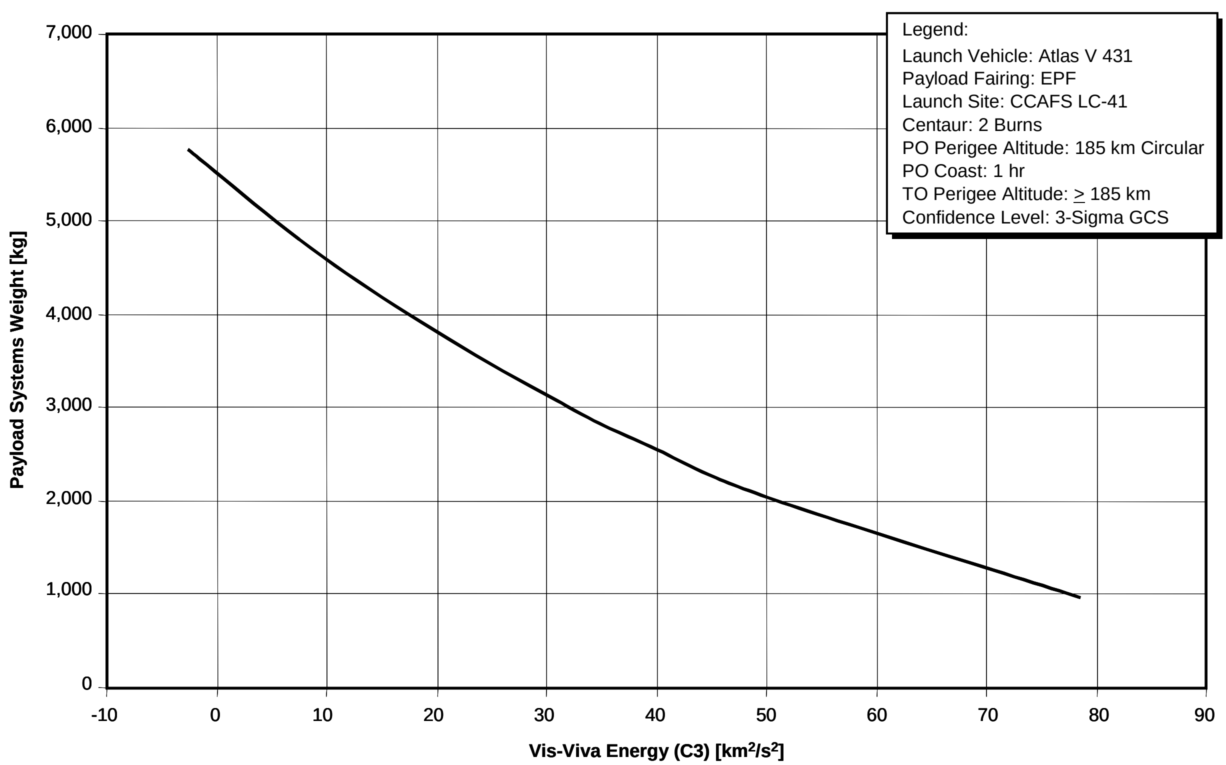


Figure 2.6.4-1c: Atlas V 421 Earth Escape Performance (C3 Curves) - CCAFS



AMPG10a_F020803_03a

Figure 2.6.4-1d: Atlas V 431 Earth Escape Performance (C3 Curves) - CCAFS



AMPG10a_F020803_04a

Figure 2.6.4-2a: Atlas V 501 Earth Escape Performance (C3 Curves) - CCAFS

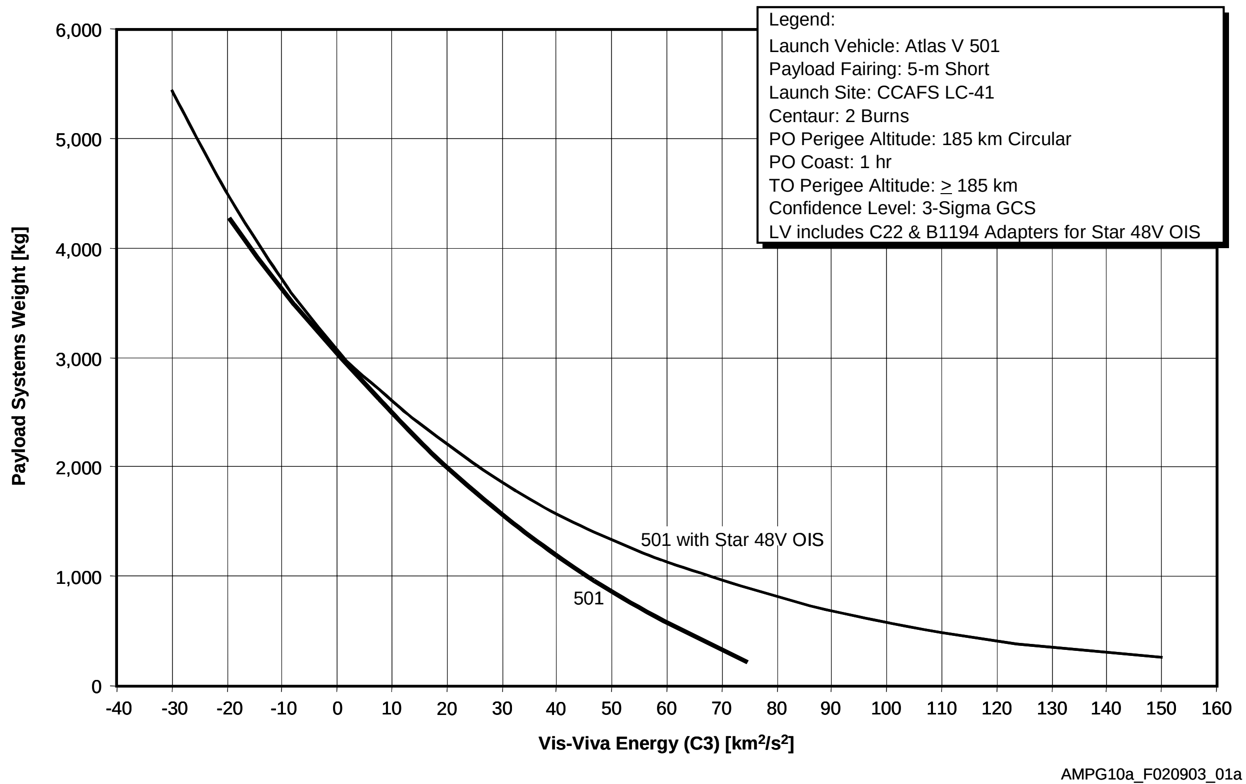


Figure 2.6.4-2b: Atlas V 511 Earth Escape Performance (C3 Curves) - CCAFS

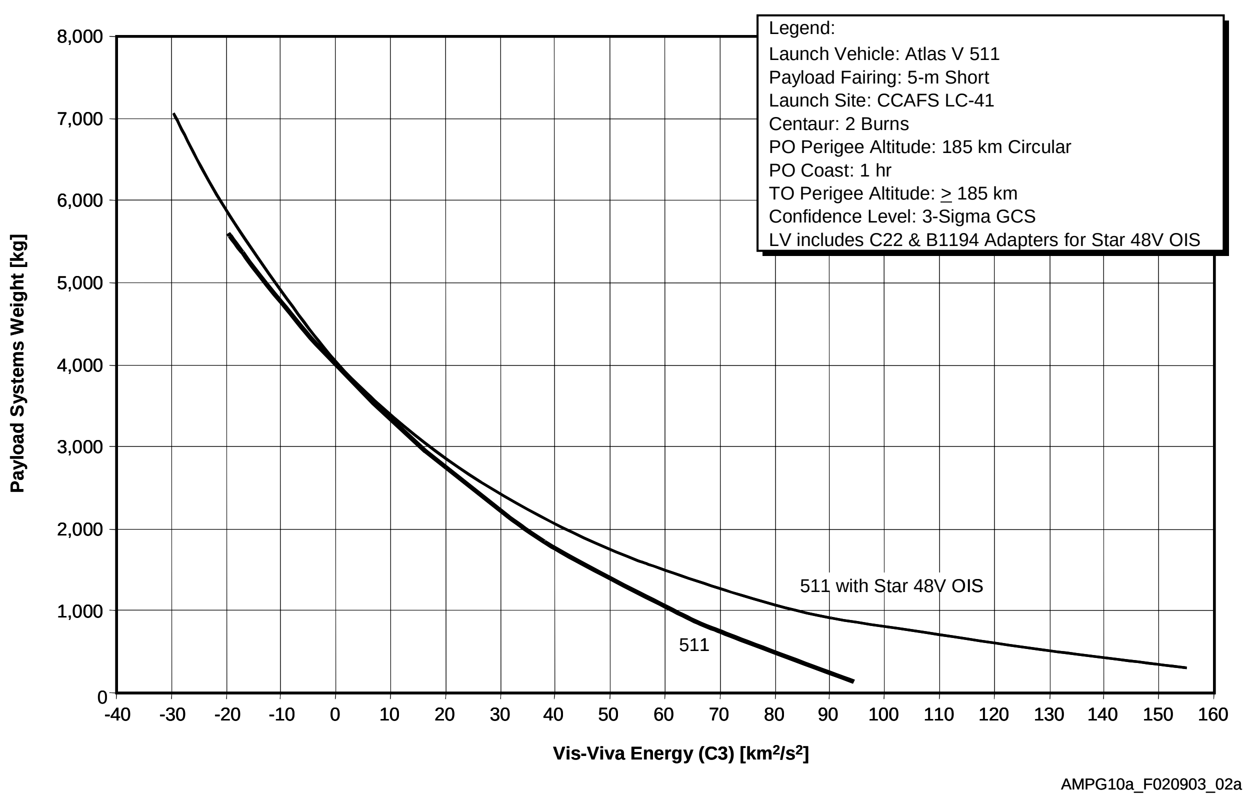
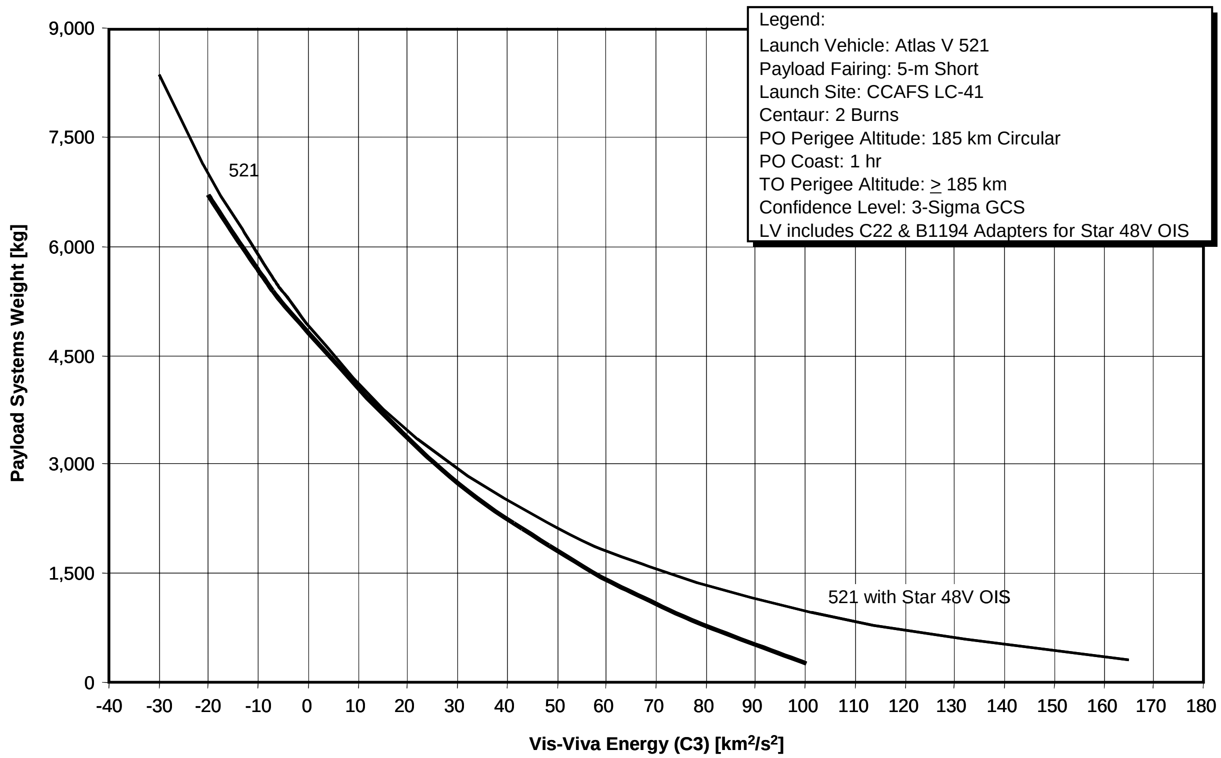
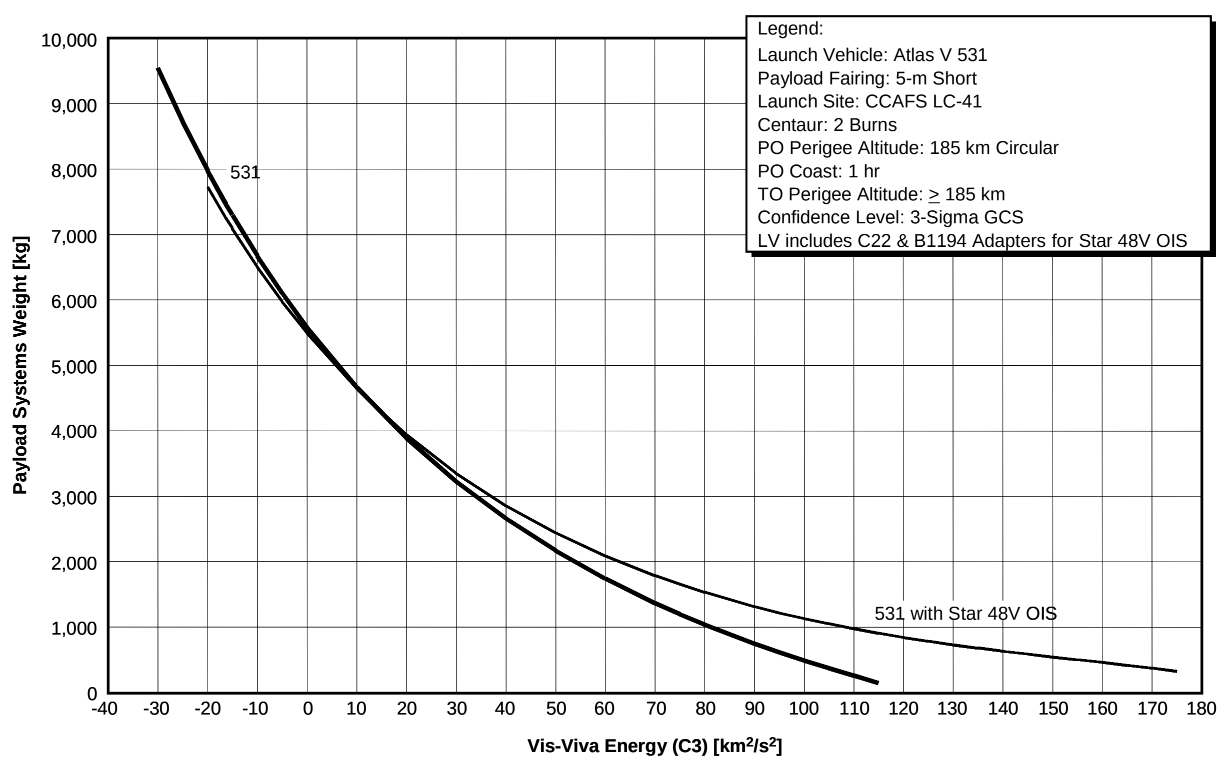


Figure 2.6.4-2c: Atlas V 521 Earth Escape Performance (C3 Curves) - CCAFS



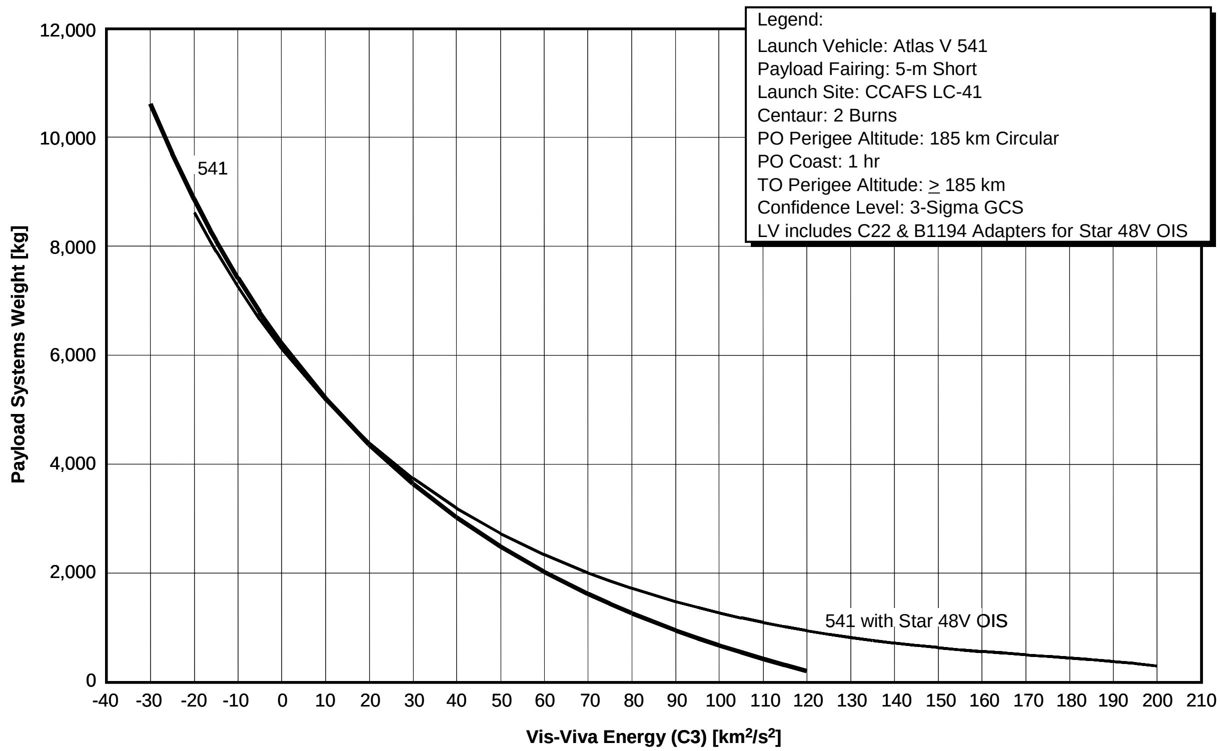
AMPG10a_F020903_03a

Figure 2.6.4-2d: Atlas V 531 Earth Escape Performance (C3 Curves) - CCAFS



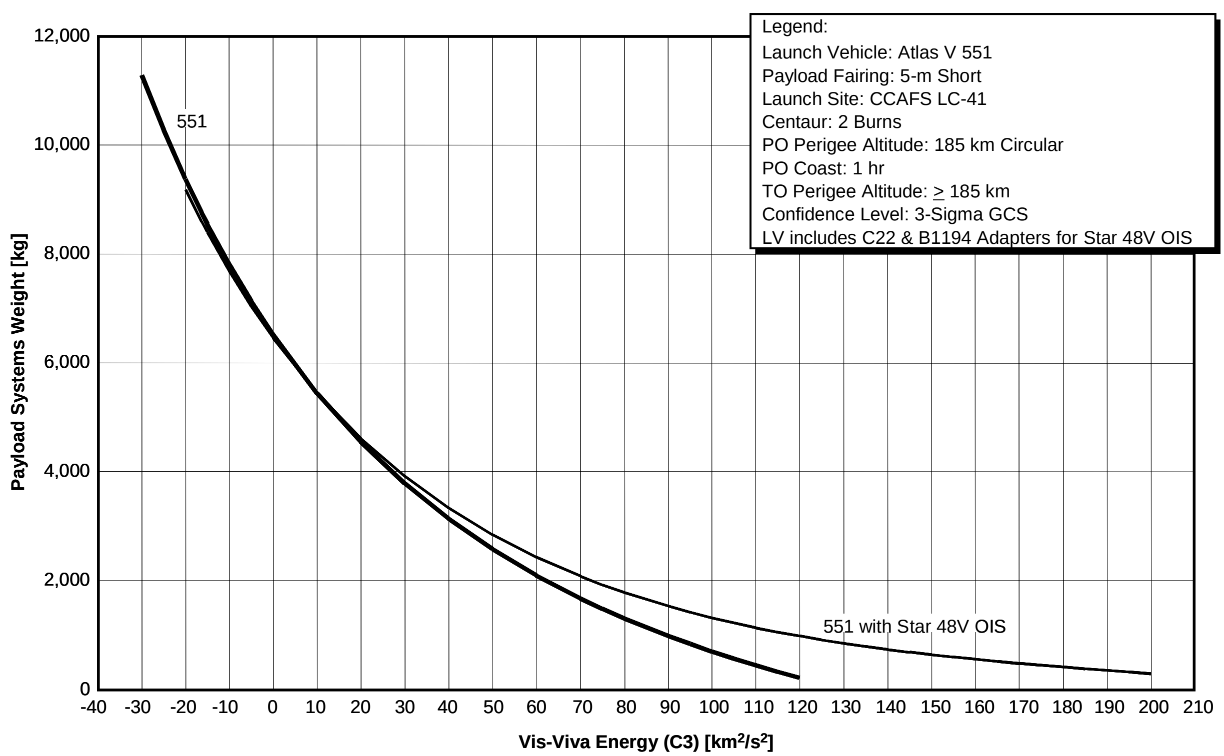
AMPG10a_F020903_04a

Figure 2.6.4-2e: Atlas V 541 Earth Escape Performance (C3 Curves) - CCAFS



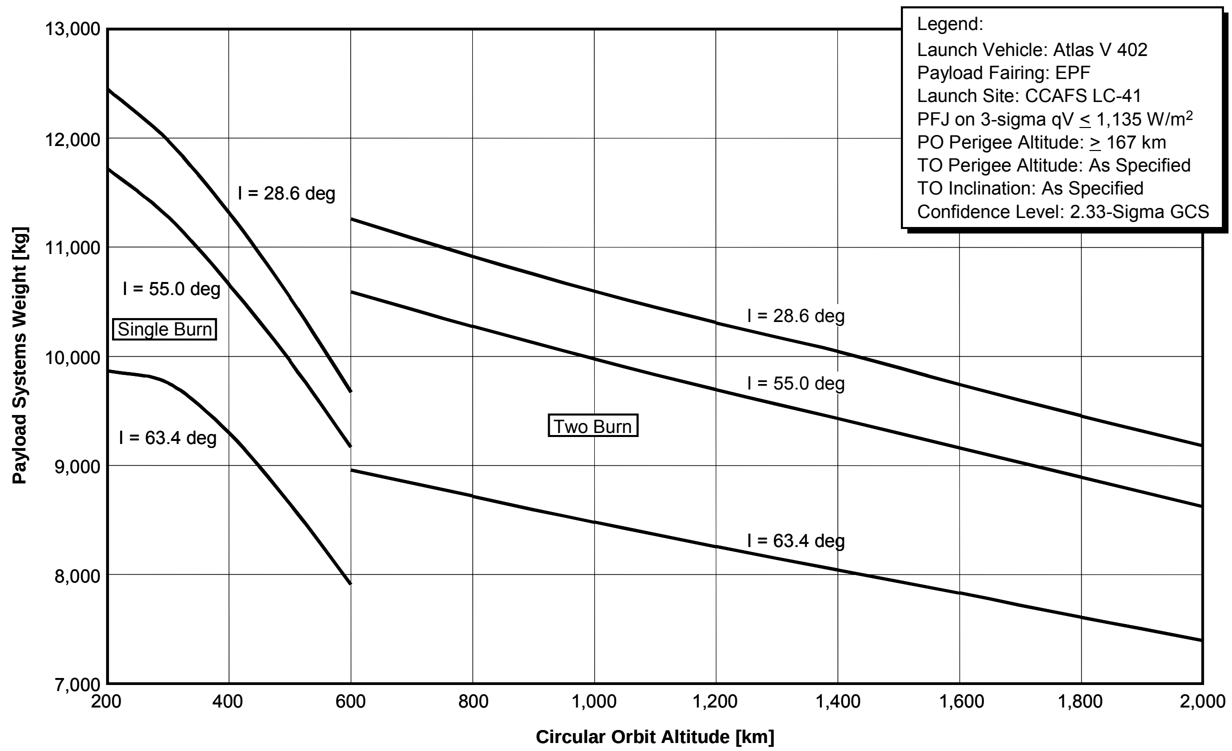
AMPG10a_F020903_05a

Figure 2.6.4-2f: Atlas V 551 Earth Escape Performance (C3 Curves) - CCAFS



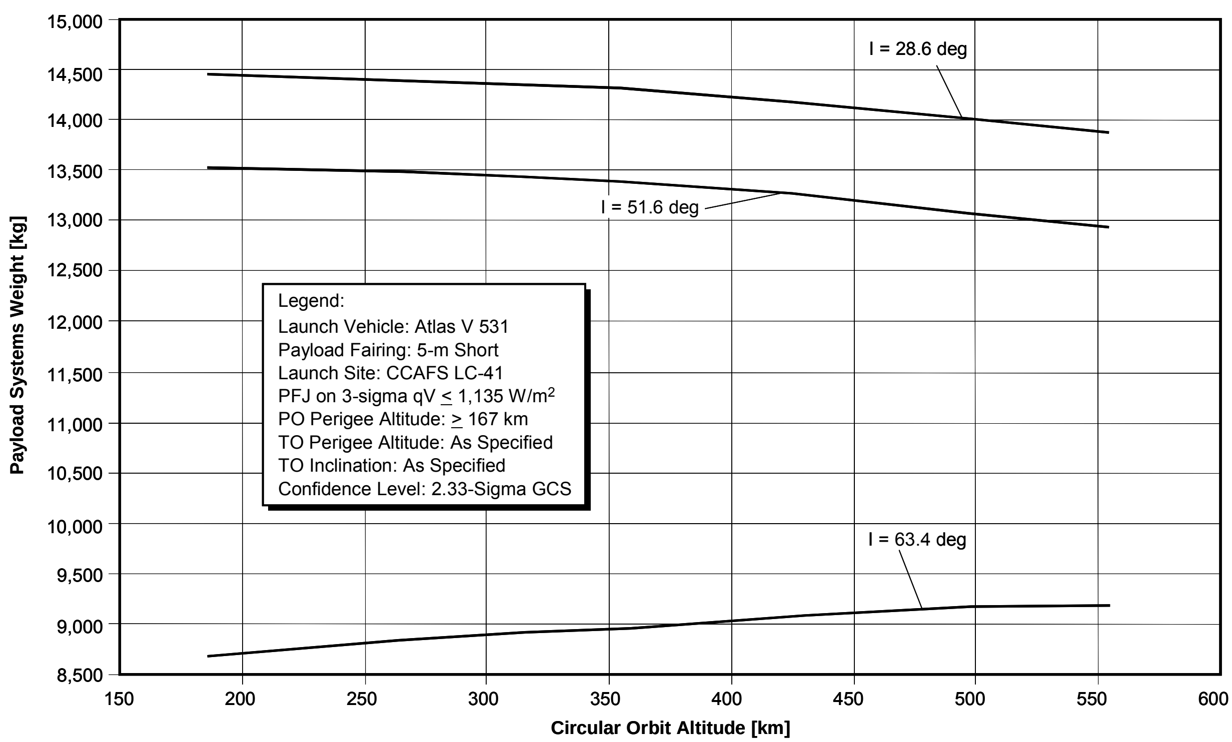
AMPG10a_F020903_06a

Figure 2.6.5-1: Atlas V 402 Low-Earth Orbit Performance - CCAFS



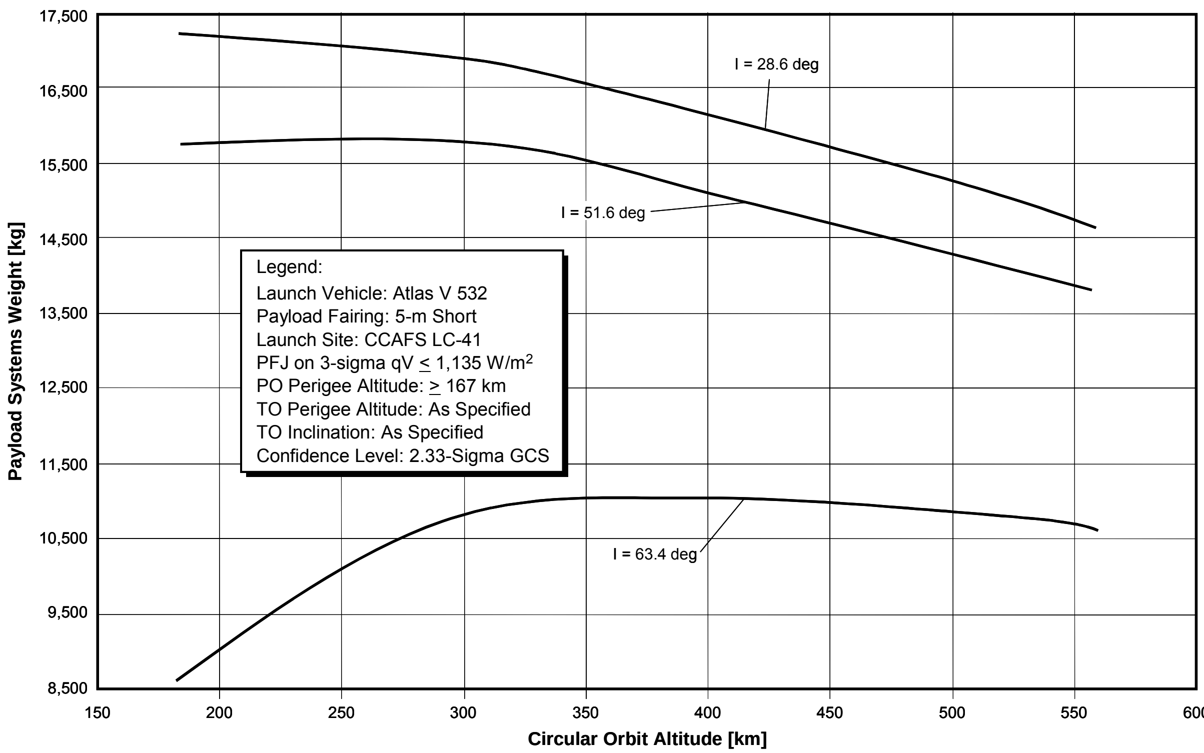
AMPG10a_F020804_01b

Figure 2.6.5-2a: Atlas V 531 Low-Earth Orbit Performance - CCAFS



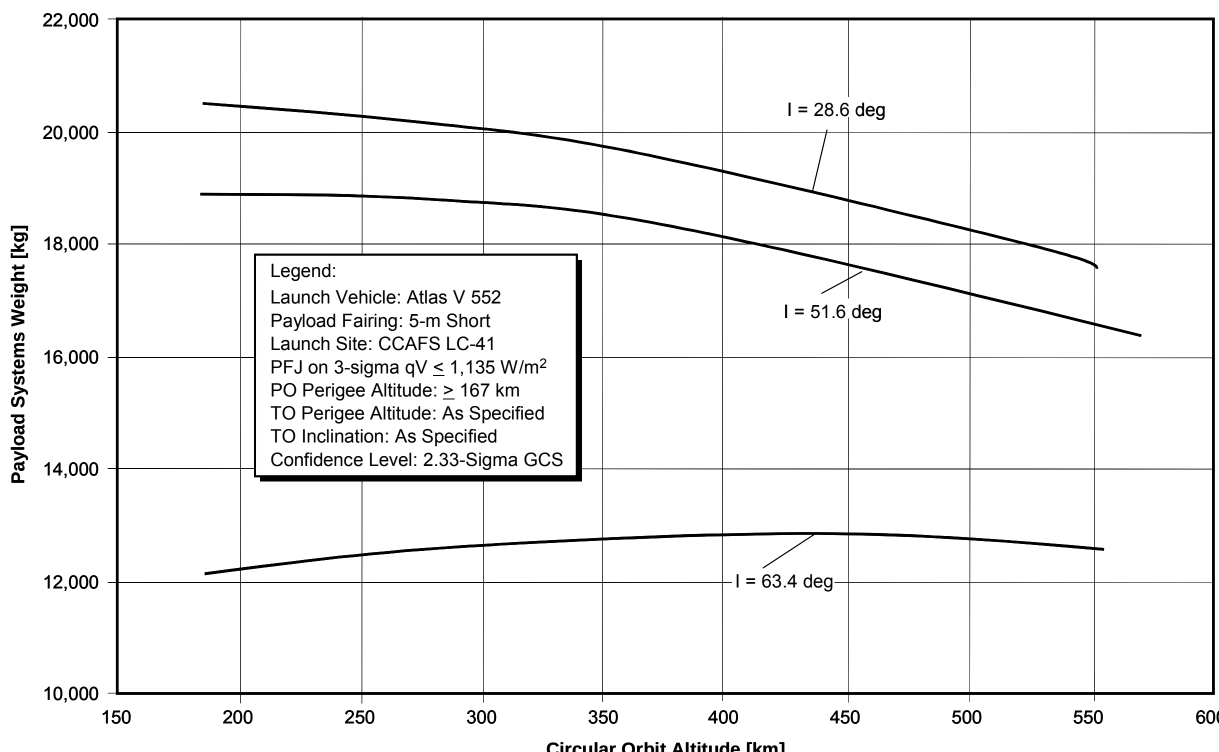
AMPG10a_F020904_01a

Figure 2.6.5-2b: Atlas V 532 Low-Earth Orbit Performance - CCAFS



AMPG10a_F020904_02a

Figure 2.6.5-2c: Atlas V 552 Low-Earth Orbit Performance - CCAFS



AMPG10a_F020904_03a

Figure 2.6.6-1a: Atlas V 401 Low-Earth Orbit Performance – VAFB

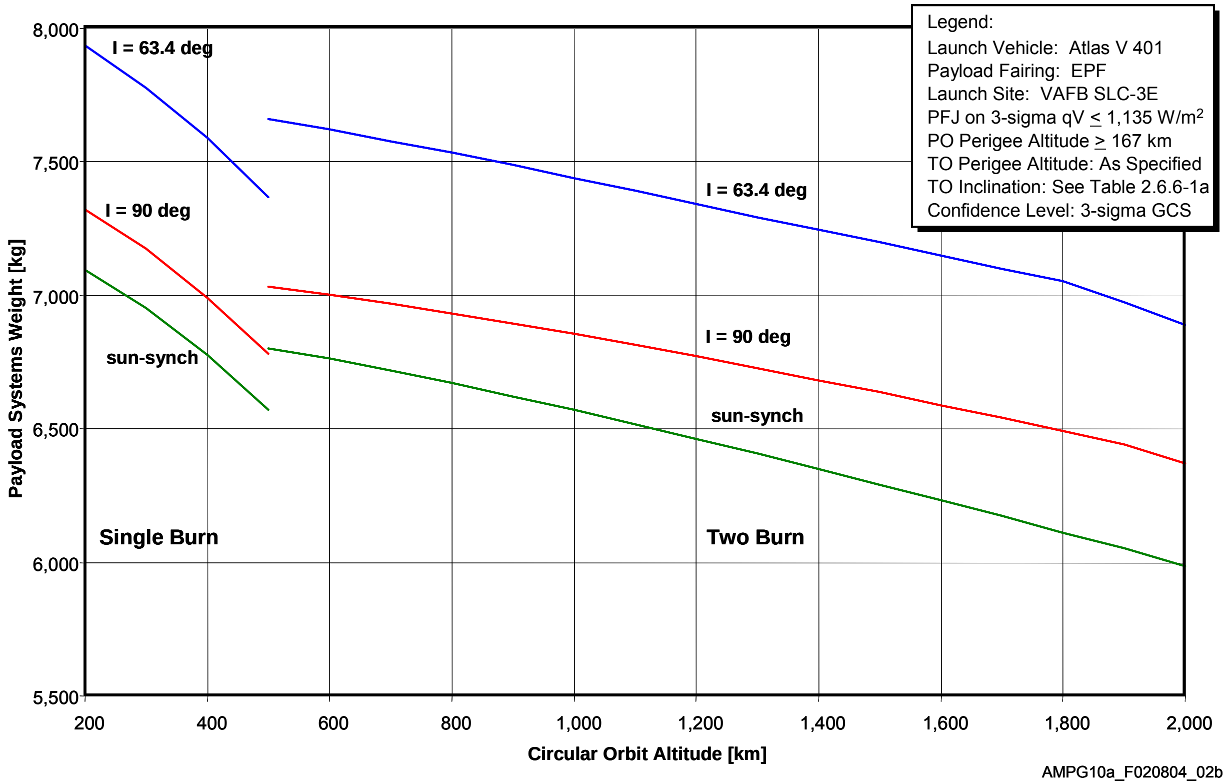


Table 2.6.6-1a: Atlas V 401 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 401 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	6,370	14,043	6,890	15,190	5,985	13,195
1,900	1,026	6,440	14,198	6,970	15,366	6,050	13,338
1,800	972	6,490	14,308	7,050	15,543	6,110	13,470
1,700	918	6,540	14,418	7,100	15,653	6,170	13,603
1,600	864	6,590	14,528	7,150	15,763	6,230	13,735
1,500	810	6,635	14,628	7,195	15,862	6,290	13,867
1,400	756	6,680	14,727	7,245	15,972	6,350	13,999
1,300	702	6,725	14,826	7,290	16,072	6,405	14,121
1,200	648	6,770	14,925	7,340	16,182	6,460	14,242
1,100	594	6,810	15,013	7,390	16,292	6,515	14,363
1,000	540	6,855	15,113	7,440	16,402	6,570	14,484
900	486	6,895	15,201	7,485	16,502	6,620	14,595
800	432	6,930	15,278	7,530	16,601	6,670	14,705
700	378	6,970	15,366	7,575	16,700	6,720	14,815
600	324	7,000	15,432	7,620	16,799	6,760	14,903
500	270	7,030	15,498	7,660	16,887	6,800	14,991
One Burn							
500	270	6,780	14,947	7,365	16,237	6,570	14,484
400	216	6,990	15,410	7,585	16,722	6,775	14,936
300	162	7,170	15,807	7,775	17,141	6,950	15,322
200	108	7,320	16,138	7,935	17,494	7,095	15,642

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: EPF
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-1b: Atlas V 411 Low-Earth Orbit Performance – VAFB

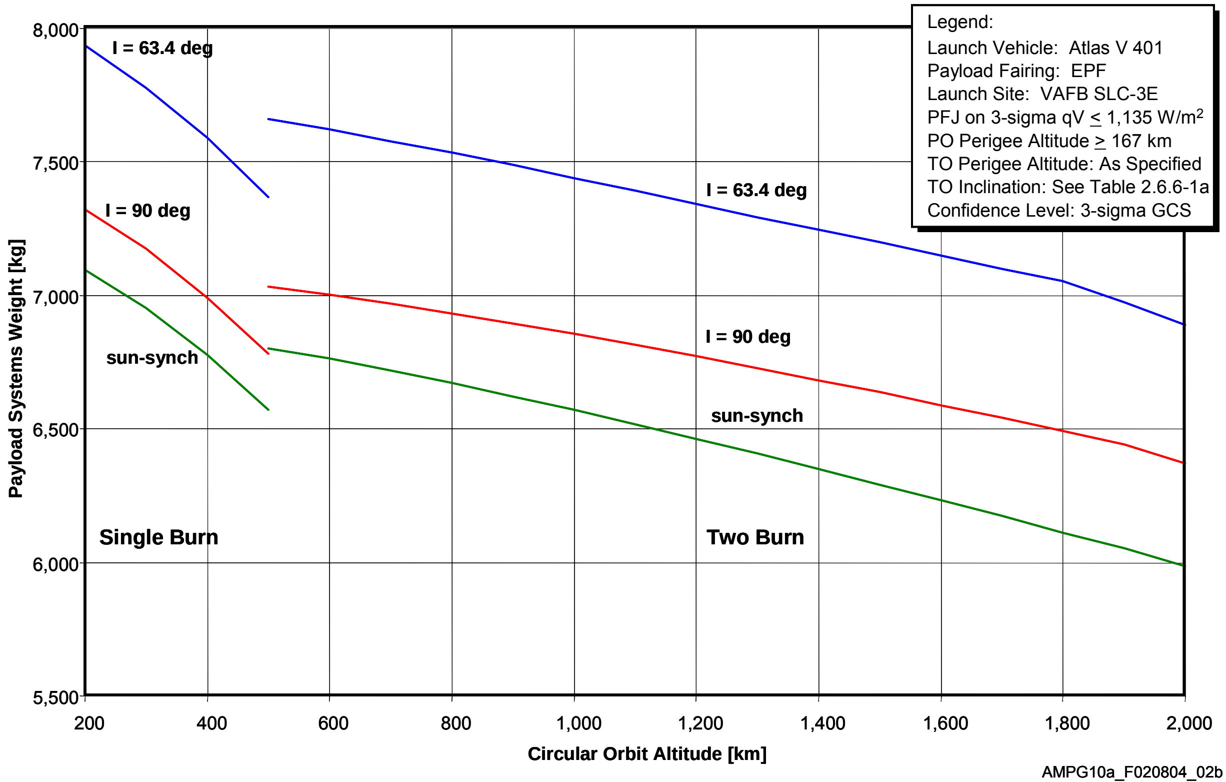


Table 2.6.6-1b: Atlas V 411 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 411 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	7,790	17,174	8,405	18,530	7,350	16,204
1,900	1,026	7,880	17,372	8,500	18,739	7,445	16,413
1,800	972	7,970	17,571	8,600	18,960	7,555	16,656
1,700	918	8,065	17,780	8,690	19,158	7,655	16,876
1,600	864	8,150	17,968	8,785	19,368	7,745	17,075
1,500	810	8,245	18,177	8,895	19,610	7,865	17,339
1,400	756	8,330	18,365	8,970	19,775	7,950	17,527
1,300	702	8,420	18,563	9,090	20,040	8,045	17,736
1,200	648	8,505	18,750	9,165	20,205	8,150	17,968
1,100	594	8,590	18,938	9,270	20,437	8,255	18,199
1,000	540	8,665	19,103	9,360	20,635	8,345	18,398
900	486	8,735	19,257	9,435	20,801	8,425	18,574
800	432	8,800	19,401	9,505	20,955	8,495	18,728
700	378	8,845	19,500	9,555	21,065	8,555	18,861
600	324	8,875	19,566	9,580	21,120	8,555	18,861
500	270	8,885	19,588	9,585	21,131	8,590	18,938
One Burn							
500	270	8,400	18,519	9,105	20,073	8,150	17,968
400	216	8,635	19,037	9,350	20,613	8,390	18,497
300	162	8,835	19,478	9,560	21,076	8,590	18,938
200	108	8,995	19,831	9,730	21,451	8,760	19,312

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: EPF
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-1c: Atlas V 421 Low-Earth Orbit Performance – VAFB

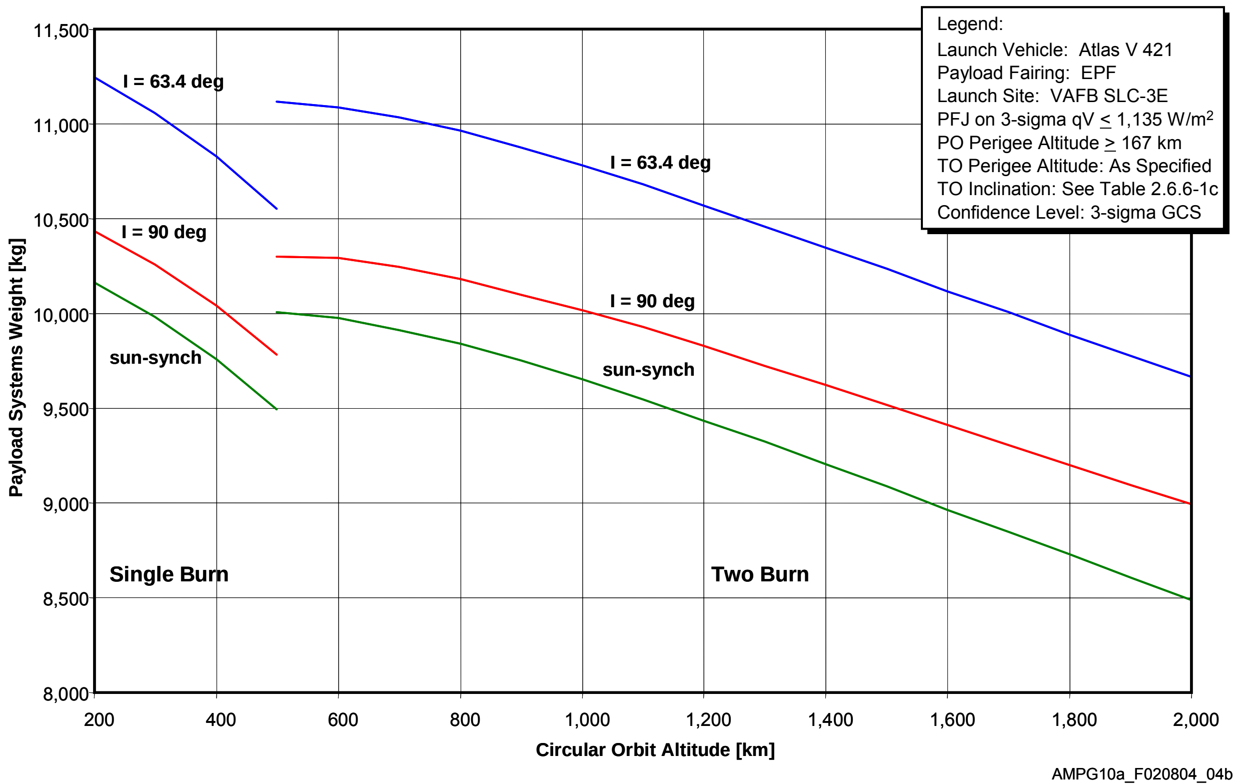


Table 2.6.6-1c: Atlas V 421 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 421 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	8,990	19,820	9,660	21,297	8,485	18,706
1,900	1,026	9,095	20,051	9,775	21,550	8,605	18,971
1,800	972	9,200	20,283	9,885	21,793	8,725	19,235
1,700	918	9,305	20,514	10,000	22,046	8,845	19,500
1,600	864	9,410	20,746	10,115	22,300	8,965	19,764
1,500	810	9,515	20,977	10,230	22,553	9,085	20,029
1,400	756	9,620	21,208	10,345	22,807	9,200	20,283
1,300	702	9,725	21,440	10,460	23,060	9,320	20,547
1,200	648	9,825	21,660	10,570	23,303	9,435	20,801
1,100	594	9,925	21,881	10,680	23,545	9,540	21,032
1,000	540	10,020	22,090	10,780	23,766	9,650	21,275
900	486	10,100	22,267	10,875	23,975	9,750	21,495
800	432	10,180	22,443	10,965	24,174	9,840	21,693
700	378	10,245	22,586	11,035	24,328	9,910	21,848
600	324	10,290	22,686	11,085	24,438	9,975	21,991
500	270	10,300	22,708	11,115	24,504	10,005	22,057
One Burn							
500	270	9,780	21,561	10,555	23,270	9,490	20,922
400	216	10,040	22,134	10,830	23,876	9,755	21,506
300	162	10,255	22,608	11,060	24,383	9,980	22,002
200	108	10,435	23,005	11,245	24,791	10,165	22,410

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: EPF
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-1d: Atlas V 431 Low-Earth Orbit Performance – VAFB

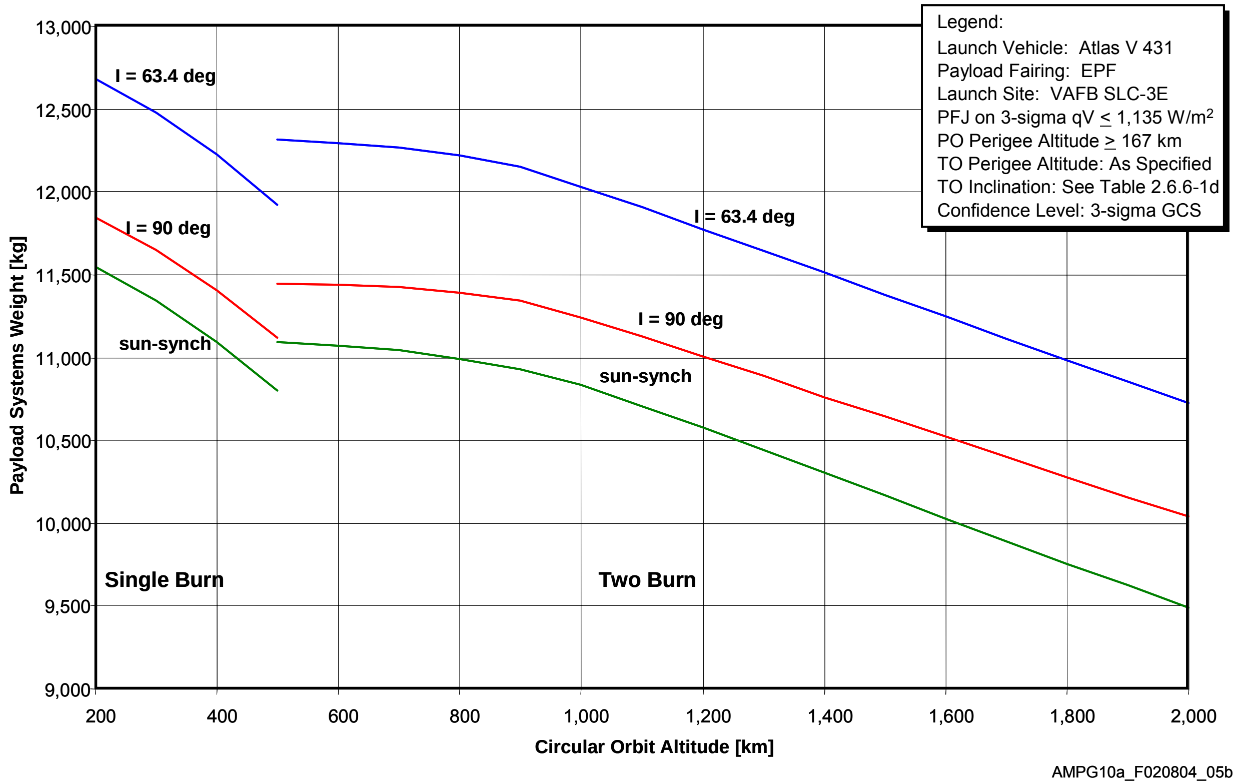


Table 2.6.6-1d: Atlas V 431 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 431 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	10,035	22,123	10,720	23,634	9,485	20,911
1,900	1,026	10,155	22,388	10,850	23,920	9,620	21,208
1,800	972	10,275	22,653	10,980	24,207	9,755	21,506
1,700	918	10,395	22,917	11,110	24,493	9,890	21,804
1,600	864	10,515	23,182	11,245	24,791	10,025	22,101
1,500	810	10,640	23,457	11,375	25,078	10,165	22,410
1,400	756	10,760	23,722	11,510	25,375	10,300	22,708
1,300	702	10,885	23,997	11,640	25,662	10,435	23,005
1,200	648	11,005	24,262	11,770	25,948	10,570	23,303
1,100	594	11,120	24,515	11,900	26,235	10,705	23,600
1,000	540	11,240	24,780	12,030	26,522	10,835	23,887
900	486	11,340	25,000	12,150	26,786	10,930	24,097
800	432	11,390	25,111	12,220	26,940	10,990	24,229
700	378	11,425	25,188	12,260	27,029	11,040	24,339
600	324	11,440	25,221	12,290	27,095	11,070	24,405
500	270	11,440	25,221	12,310	27,139	11,090	24,449
One Burn							
500	270	11,115	24,504	11,920	26,279	10,795	23,799
400	216	11,400	25,133	12,220	26,940	11,085	24,438
300	162	11,645	25,673	12,475	27,503	11,340	25,000
200	108	11,845	26,114	12,680	27,955	11,545	25,452

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: EPF
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-2a: Atlas V 501 Low-Earth Orbit Performance - VAFB

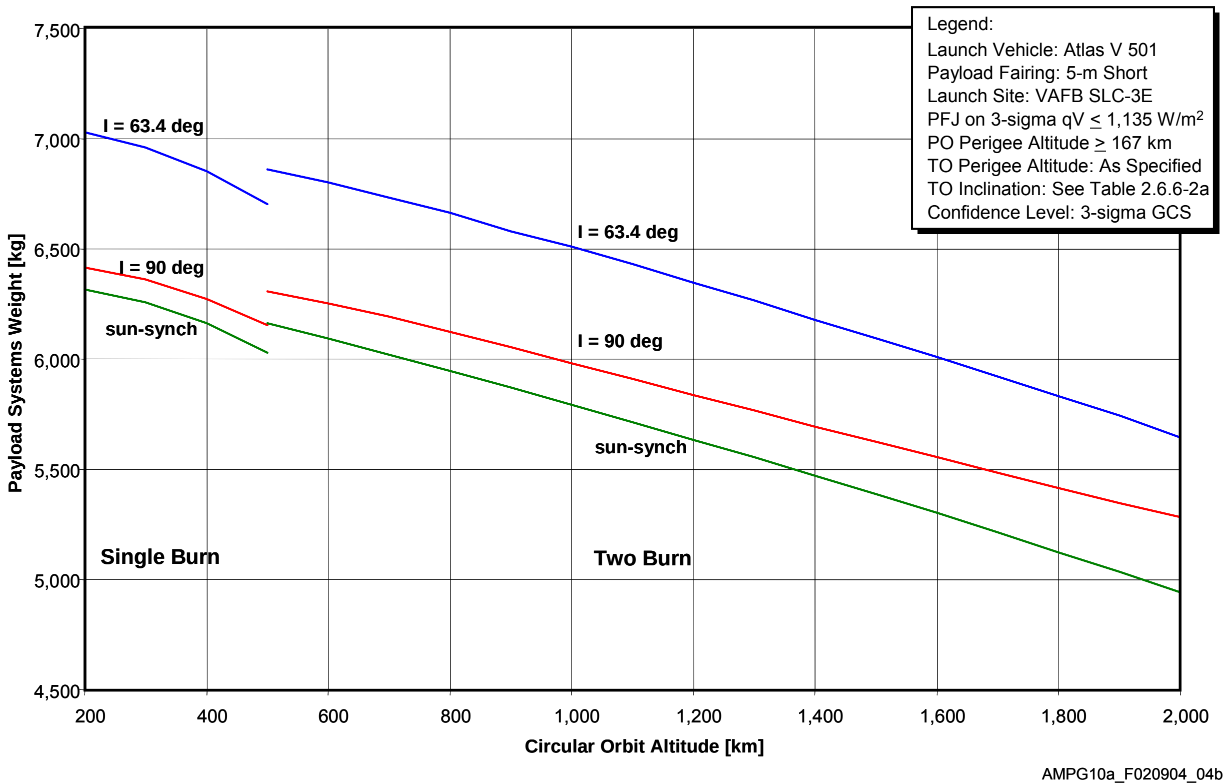


Table 2.6.6-2a: Atlas V 501 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 501 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	5,280	11,640	5,645	12,445	4,940	10,891
1,900	1,026	5,345	11,784	5,740	12,655	5,030	11,089
1,800	972	5,415	11,938	5,830	12,853	5,120	11,288
1,700	918	5,480	12,081	5,920	13,051	5,210	11,486
1,600	864	5,550	12,236	6,005	13,239	5,300	11,685
1,500	810	5,620	12,390	6,095	13,437	5,385	11,872
1,400	756	5,690	12,544	6,180	13,625	5,470	12,059
1,300	702	5,765	12,710	6,265	13,812	5,550	12,236
1,200	648	5,835	12,864	6,345	13,988	5,630	12,412
1,100	594	5,910	13,029	6,425	14,165	5,710	12,588
1,000	540	5,980	13,184	6,505	14,341	5,790	12,765
900	486	6,050	13,338	6,575	14,495	5,865	12,930
800	432	6,120	13,492	6,660	14,683	5,945	13,106
700	378	6,190	13,647	6,735	14,848	6,015	13,261
600	324	6,250	13,779	6,800	14,991	6,090	13,426
500	270	6,305	13,900	6,860	15,124	6,160	13,580
One Burn							
500	270	6,150	13,558	6,700	14,771	6,030	13,294
400	216	6,270	13,823	6,850	15,102	6,160	13,580
300	162	6,360	14,021	6,960	15,344	6,255	13,790
200	108	6,415	14,143	7,030	15,498	6,315	13,922

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: 5-m Short
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-2b: Atlas V 511 Low-Earth Orbit Performance - VAFB

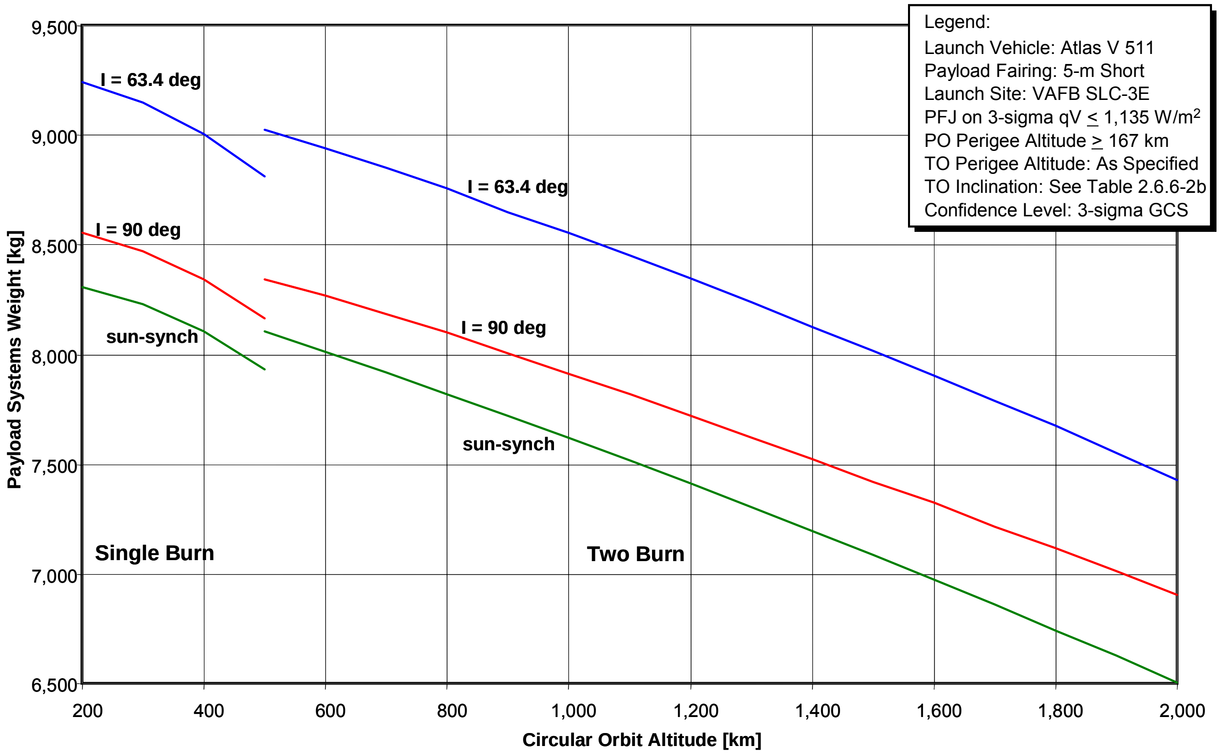


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Table 2.6.6-2b: Atlas V 511 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 511 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	6,905	15,223	7,425	16,369	6,505	14,341
1,900	1,026	7,010	15,454	7,555	16,656	6,625	14,606
1,800	972	7,115	15,686	7,675	16,920	6,740	14,859
1,700	918	7,215	15,906	7,790	17,174	6,860	15,124
1,600	864	7,325	16,149	7,900	17,417	6,975	15,377
1,500	810	7,420	16,358	8,015	17,670	7,085	15,620
1,400	756	7,520	16,579	8,125	17,913	7,195	15,862
1,300	702	7,620	16,799	8,240	18,166	7,305	16,105
1,200	648	7,720	17,020	8,345	18,398	7,410	16,336
1,100	594	7,815	17,229	8,450	18,629	7,515	16,568
1,000	540	7,915	17,450	8,555	18,861	7,620	16,799
900	486	8,005	17,648	8,650	19,070	7,720	17,020
800	432	8,100	17,857	8,755	19,301	7,820	17,240
700	378	8,185	18,045	8,850	19,511	7,915	17,450
600	324	8,265	18,221	8,940	19,709	8,010	17,659
500	270	8,345	18,398	9,020	19,886	8,105	17,868
One Burn							
500	270	8,165	18,001	8,815	19,434	7,930	17,483
400	216	8,340	18,387	9,005	19,853	8,100	17,857
300	162	8,470	18,673	9,145	20,161	8,225	18,133
200	108	8,555	18,861	9,240	20,371	8,310	18,320

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: 5-m Short
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-2c: Atlas V 521 Low-Earth Orbit Performance - VAFB

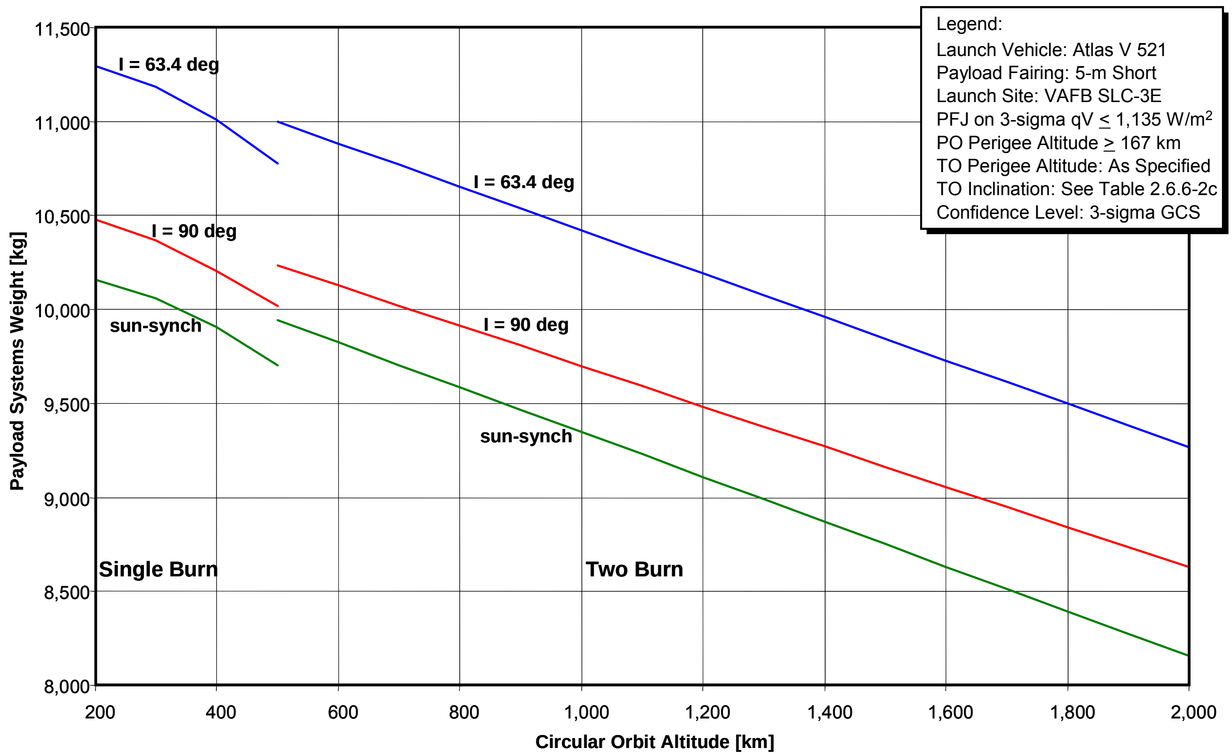

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Table 2.6.6-2c: Atlas V 521 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 521 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	8,625	19,015	9,265	20,426	8,155	17,979
1,900	1,026	8,735	19,257	9,380	20,679	8,275	18,243
1,800	972	8,840	19,489	9,495	20,933	8,390	18,497
1,700	918	8,945	19,720	9,610	21,186	8,510	18,761
1,600	864	9,055	19,963	9,725	21,440	8,630	19,026
1,500	810	9,160	20,194	9,845	21,705	8,750	19,290
1,400	756	9,270	20,437	9,960	21,958	8,870	19,555
1,300	702	9,375	20,668	10,075	22,212	8,990	19,820
1,200	648	9,480	20,900	10,190	22,465	9,105	20,073
1,100	594	9,590	21,142	10,305	22,719	9,225	20,338
1,000	540	9,695	21,374	10,420	22,972	9,345	20,602
900	486	9,805	21,616	10,535	23,226	9,465	20,867
800	432	9,910	21,848	10,650	23,479	9,585	21,131
700	378	10,020	22,090	10,765	23,733	9,705	21,396
600	324	10,125	22,322	10,885	23,997	9,820	21,649
500	270	10,230	22,553	11,000	24,251	9,940	21,914
One Burn							
500	270	10,020	22,090	10,775	23,755	9,700	21,385
400	216	10,205	22,498	11,010	24,273	9,910	21,848
300	162	10,365	22,851	11,185	24,659	10,060	22,179
200	108	10,475	23,093	11,295	24,901	10,160	22,399

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: 5-m Short
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-2d: Atlas V 531 Low-Earth Orbit Performance - VAFB

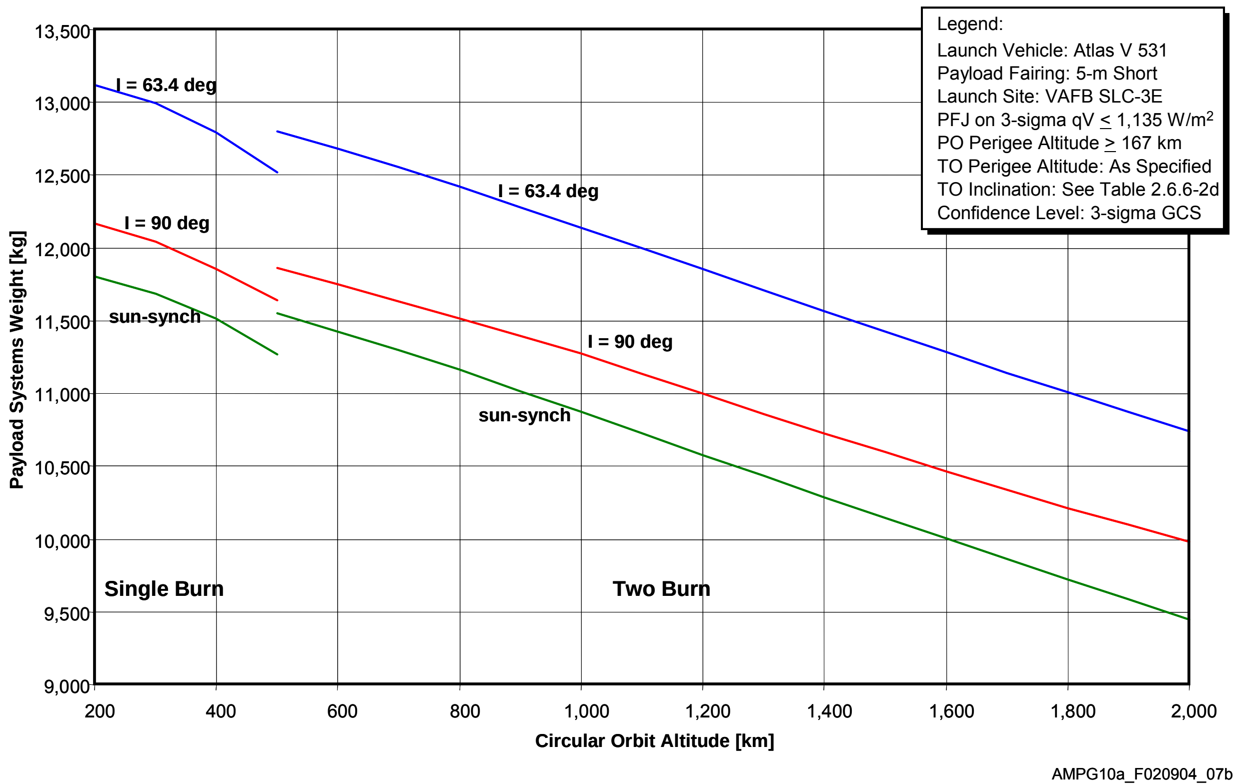

Table 2.6.6-2d: Atlas V 531 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 531 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	9,980	22,002	10,740	23,678	9,445	20,823
1,900	1,026	10,095	22,256	10,875	23,975	9,585	21,131
1,800	972	10,215	22,520	11,005	24,262	9,720	21,429
1,700	918	10,335	22,785	11,145	24,571	9,860	21,738
1,600	864	10,465	23,071	11,280	24,868	10,000	22,046
1,500	810	10,595	23,358	11,420	25,177	10,145	22,366
1,400	756	10,725	23,645	11,565	25,496	10,290	22,686
1,300	702	10,860	23,942	11,705	25,805	10,435	23,005
1,200	648	10,995	24,240	11,850	26,125	10,580	23,325
1,100	594	11,135	24,548	11,995	26,444	10,725	23,645
1,000	540	11,275	24,857	12,140	26,764	10,870	23,964
900	486	11,395	25,122	12,280	27,073	11,015	24,284
800	432	11,515	25,386	12,420	27,381	11,160	24,604
700	378	11,630	25,640	12,550	27,668	11,295	24,901
600	324	11,745	25,893	12,685	27,966	11,420	25,177
500	270	11,860	26,147	12,795	28,208	11,550	25,463
One Burn							
500	270	11,635	25,651	12,515	27,591	11,270	24,846
400	216	11,855	26,136	12,790	28,197	11,510	25,375
300	162	12,040	26,544	12,990	28,638	11,685	25,761
200	108	12,170	26,830	13,120	28,925	11,800	26,015

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: 5-m Short
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-2e: Atlas V 541 Low-Earth Orbit Performance - VAFB

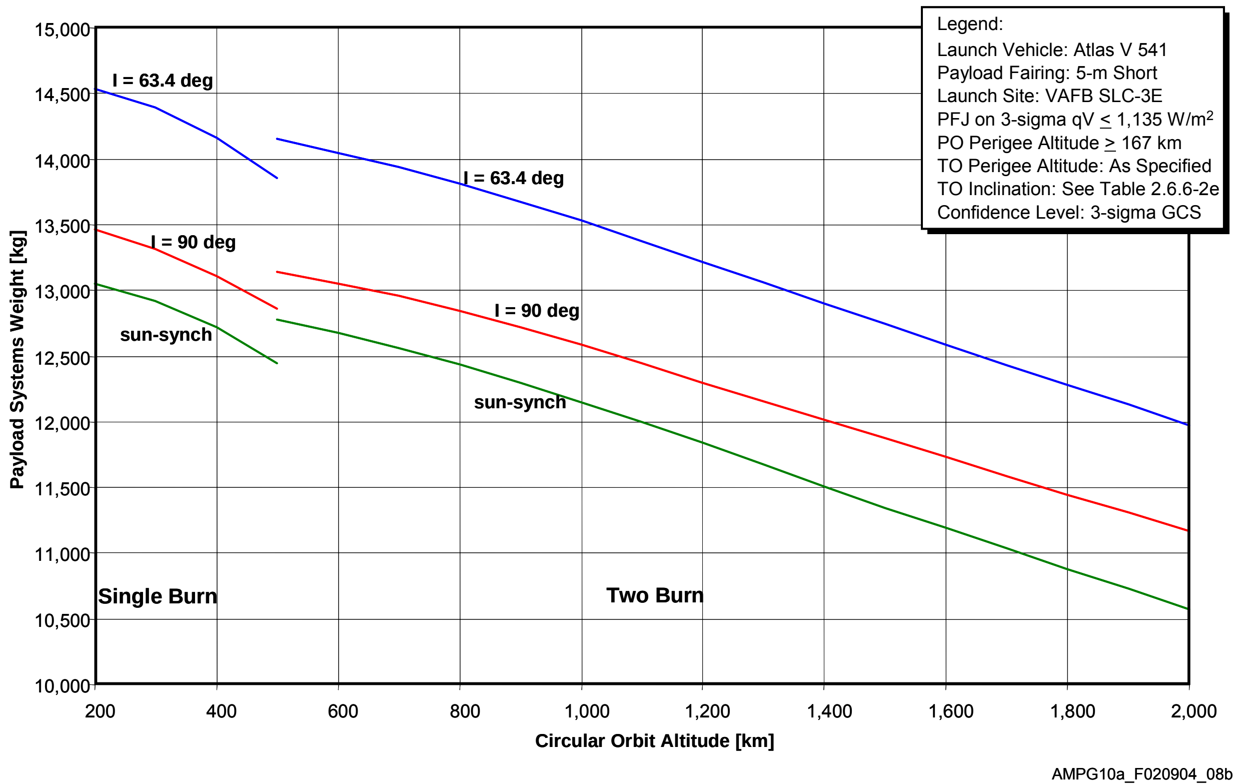

Table 2.6.6-2e: Atlas V 541 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 541 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	11,165	24,615	11,975	26,400	10,570	23,303
1,900	1,026	11,305	24,923	12,125	26,731	10,725	23,645
1,800	972	11,445	25,232	12,275	27,062	10,880	23,986
1,700	918	11,585	25,541	12,430	27,403	11,035	24,328
1,600	864	11,730	25,860	12,585	27,745	11,190	24,670
1,500	810	11,870	26,169	12,745	28,098	11,345	25,011
1,400	756	12,015	26,489	12,905	28,451	11,510	25,375
1,300	702	12,155	26,797	13,060	28,792	11,675	25,739
1,200	648	12,300	27,117	13,220	29,145	11,835	26,092
1,100	594	12,440	27,426	13,375	29,487	11,995	26,444
1,000	540	12,585	27,745	13,525	29,818	12,145	26,775
900	486	12,715	28,032	13,670	30,137	12,290	27,095
800	432	12,845	28,318	13,815	30,457	12,435	27,414
700	378	12,955	28,561	13,940	30,732	12,560	27,690
600	324	13,050	28,770	14,050	30,975	12,675	27,944
500	270	13,140	28,969	14,150	31,195	12,780	28,175
One Burn							
500	270	12,860	28,351	13,855	30,545	12,445	27,437
400	216	13,110	28,903	14,165	31,228	12,720	28,043
300	162	13,320	29,366	14,390	31,725	12,915	28,473
200	108	13,465	29,685	14,535	32,044	13,045	28,759

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: 5-m Short
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-2f: Atlas V 551 Low-Earth Orbit Performance - VAFB

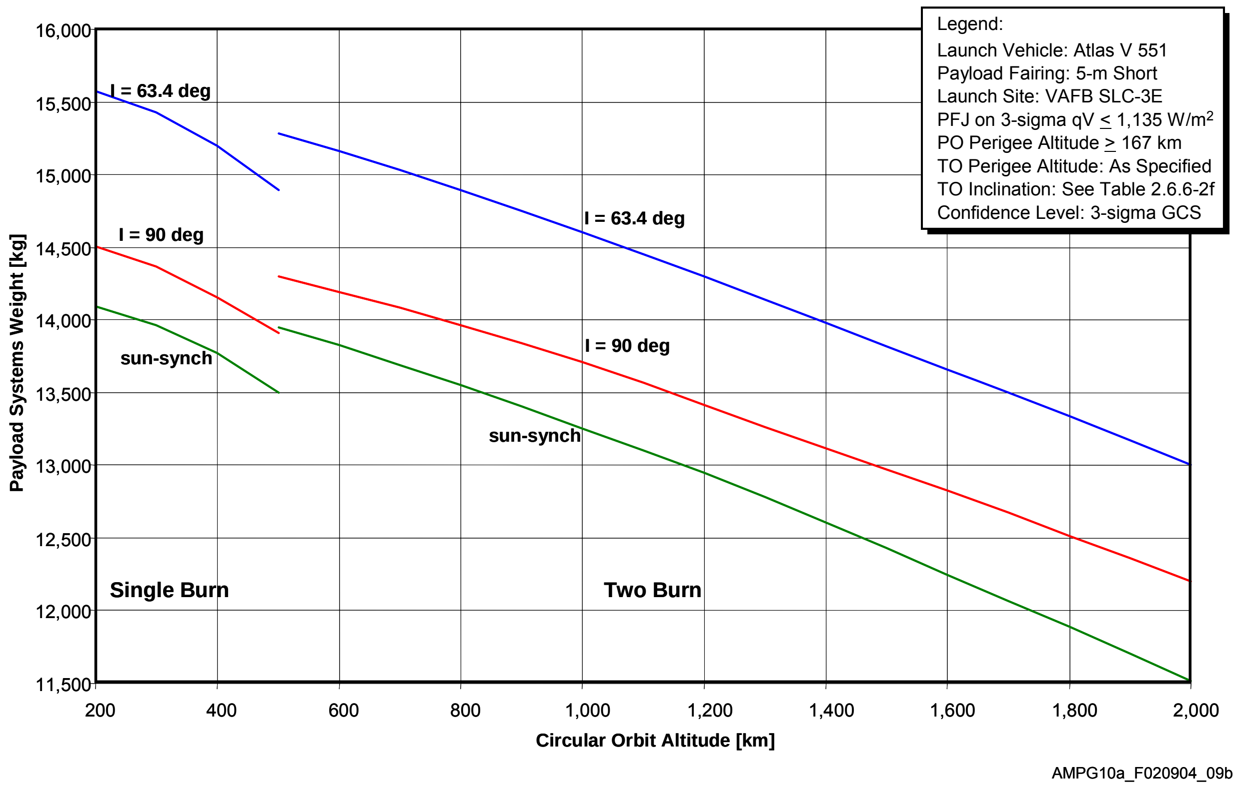

Table 2.6.6-2f: Atlas V 551 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 551 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	12,200	26,896	13,005	28,671	11,515	25,386
1,900	1,026	12,355	27,238	13,170	29,035	11,700	25,794
1,800	972	12,515	27,591	13,335	29,399	11,885	26,202
1,700	918	12,670	27,933	13,495	29,751	12,065	26,599
1,600	864	12,825	28,274	13,660	30,115	12,245	26,996
1,500	810	12,970	28,594	13,820	30,468	12,425	27,392
1,400	756	13,120	28,925	13,980	30,821	12,605	27,789
1,300	702	13,265	29,244	14,140	31,173	12,780	28,175
1,200	648	13,410	29,564	14,295	31,515	12,945	28,539
1,100	594	13,560	29,895	14,450	31,857	13,100	28,881
1,000	540	13,710	30,225	14,600	32,187	13,255	29,222
900	486	13,840	30,512	14,750	32,518	13,405	29,553
800	432	13,965	30,788	14,895	32,838	13,550	29,873
700	378	14,080	31,041	15,030	33,135	13,690	30,181
600	324	14,190	31,284	15,160	33,422	13,825	30,479
500	270	14,295	31,515	15,285	33,698	13,950	30,754
One Burn							
500	270	13,910	30,666	14,895	32,838	13,495	29,751
400	216	14,155	31,206	15,200	33,510	13,770	30,358
300	162	14,365	31,669	15,425	34,006	13,965	30,788
200	108	14,505	31,978	15,570	34,326	14,095	31,074

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: 5-m Short
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.6-2g: Atlas V 552 Low-Earth Orbit Performance - VAFB

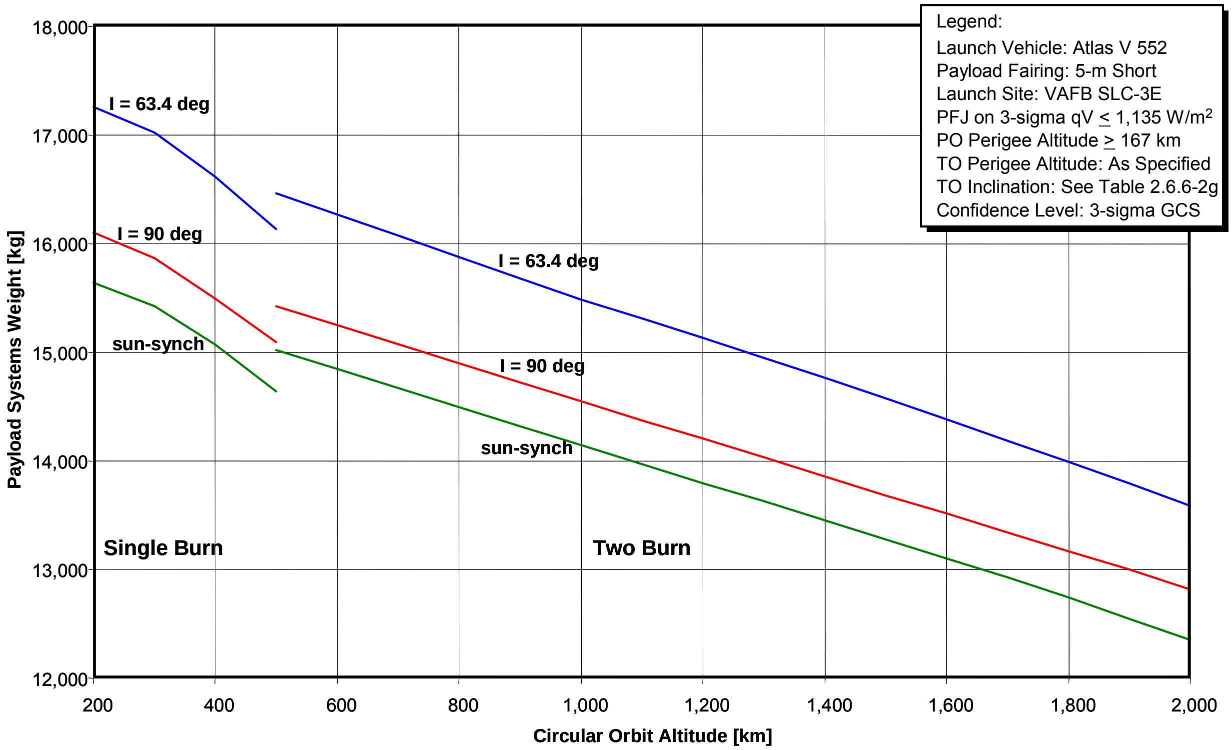

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Table 2.6.6-2g: Atlas V 552 Low Earth Orbit Performance – PSW vs Altitude

Atlas V 552 LEO - VAFB							
Circular Orbit Altitude		Payload Systems Weight					
		I = 90 deg		I = 63.4 deg		Sun Synch	
[km]	[nmi]	[kg]	[lb]	[kg]	[lb]	[kg]	[lb]
Two Burn							
2,000	1,080	12,810	28,241	13,590	29,961	12,350	27,227
1,900	1,026	12,995	28,649	13,790	30,402	12,545	27,657
1,800	972	13,165	29,024	13,985	30,832	12,735	28,076
1,700	918	13,340	29,410	14,180	31,262	12,930	28,506
1,600	864	13,510	29,784	14,375	31,691	13,105	28,892
1,500	810	13,685	30,170	14,570	32,121	13,275	29,266
1,400	756	13,855	30,545	14,760	32,540	13,450	29,652
1,300	702	14,030	30,931	14,945	32,948	13,620	30,027
1,200	648	14,200	31,306	15,130	33,356	13,795	30,413
1,100	594	14,375	31,691	15,305	33,742	13,970	30,799
1,000	540	14,545	32,066	15,480	34,128	14,140	31,173
900	486	14,720	32,452	15,675	34,557	14,315	31,559
800	432	14,895	32,838	15,870	34,987	14,490	31,945
700	378	15,070	33,224	16,070	35,428	14,665	32,331
600	324	15,245	33,609	16,265	35,858	14,840	32,717
500	270	15,420	33,995	16,465	36,299	15,020	33,113
One Burn							
500	270	15,085	33,257	16,130	35,561	14,635	32,265
400	216	15,495	34,161	16,610	36,619	15,065	33,213
300	162	15,865	34,976	17,015	37,512	15,420	33,995
200	108	16,095	35,483	17,255	38,041	15,635	34,469

Notes:
Launch Site: VAFB SLC-3E
Payload Fairing: 5-m Short
PFJ at 3-sigma qV ≤ 1,135 W/m^2 (360 BTU/ft^2-hr)
Park Orbit Perigee Altitude ≥ 167 km (90 nmi)
Confidence Level: 3 Sigma GCS

Figure 2.6.7-1a: Atlas V 531 Intermediate Orbit Performance - CCAFS

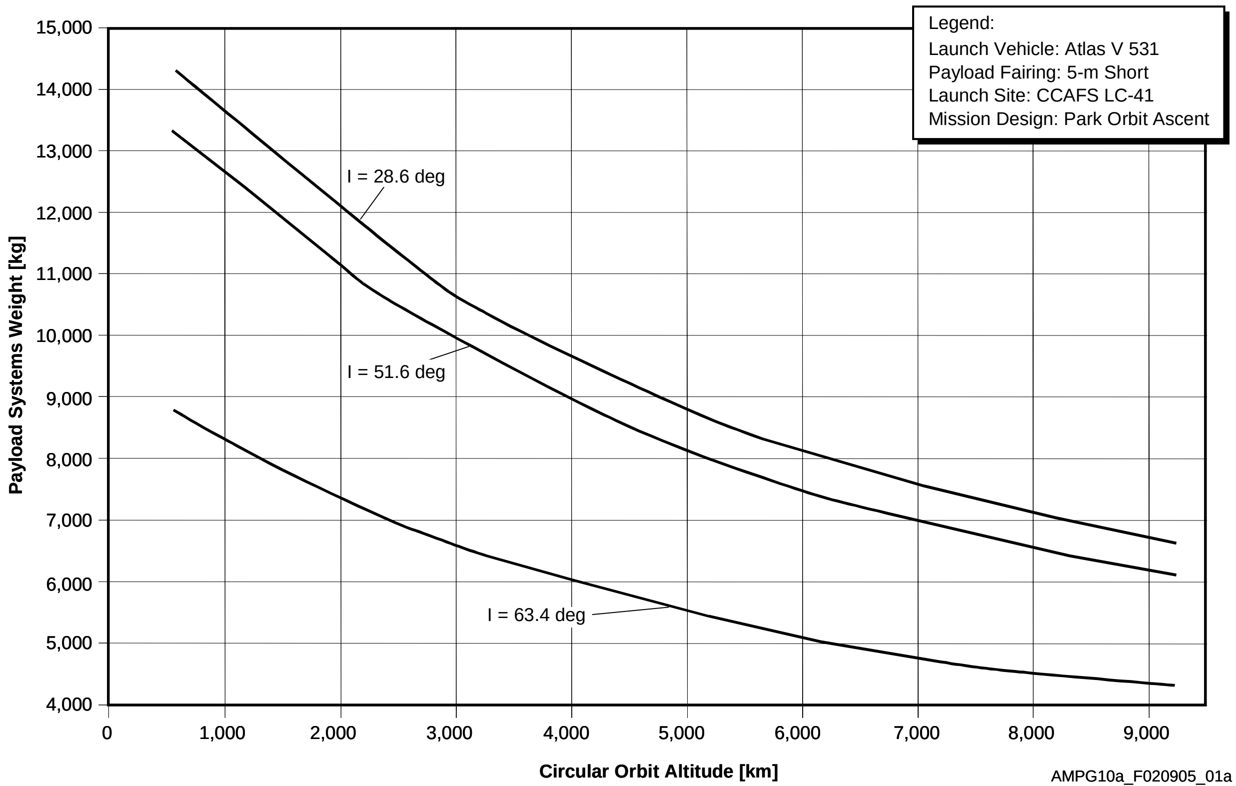


Figure 2.6.7-1b: Atlas V 532 Intermediate Orbit Performance - CCAFS

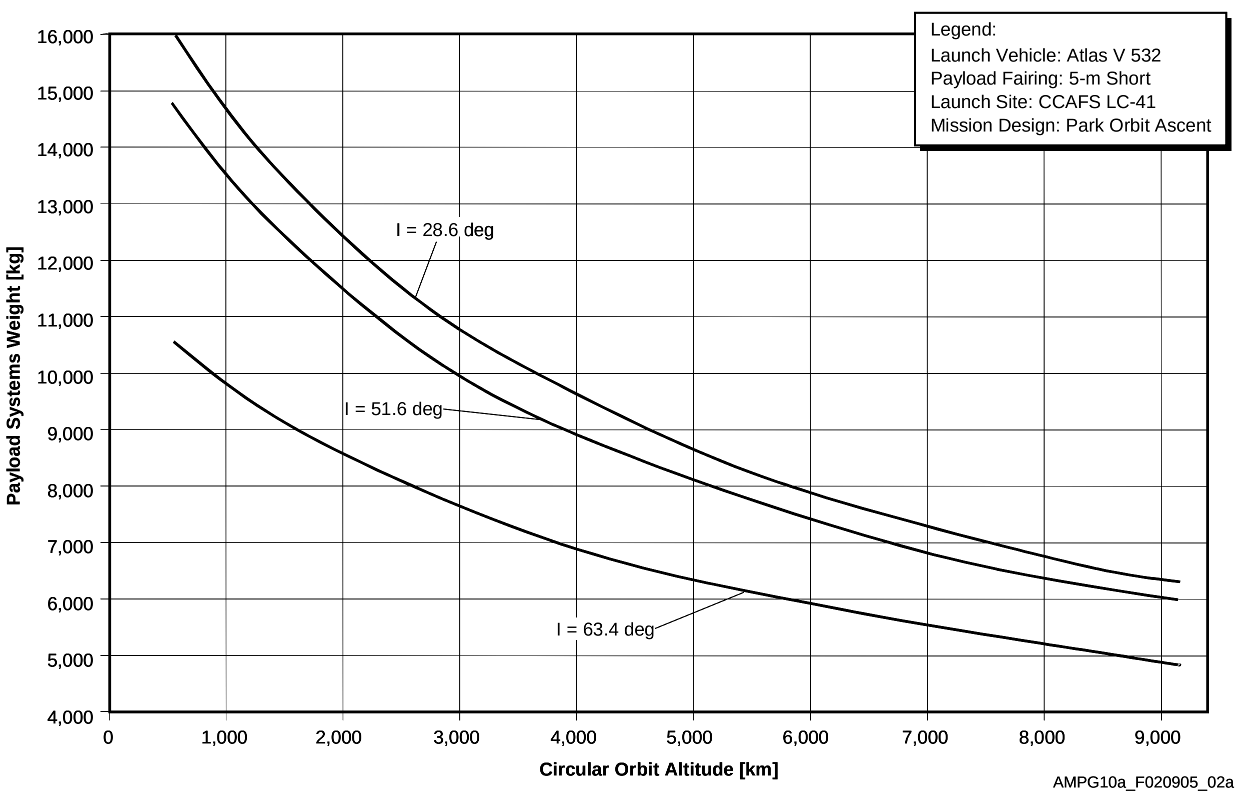
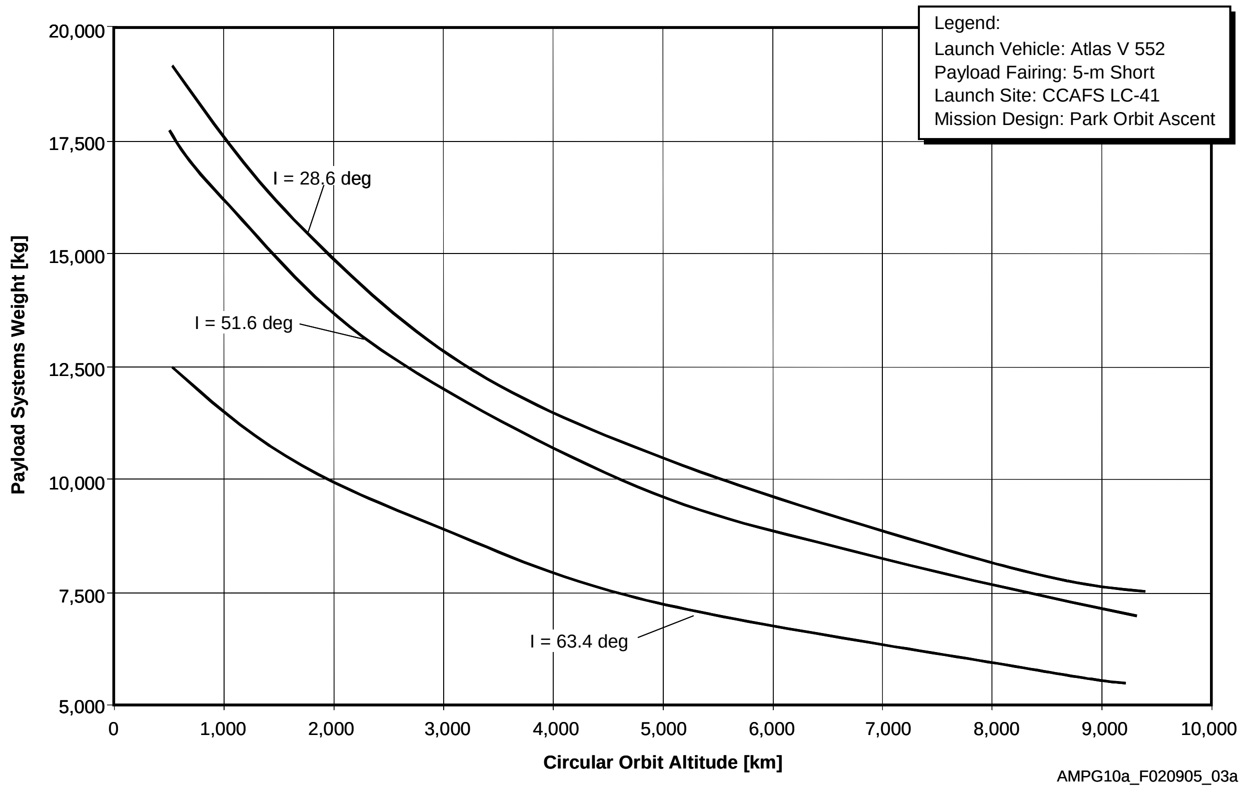


Figure 2.6.7-1c: Atlas V 552 Intermediate Orbit Performance - CCAFS



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3. ENVIRONMENTS

This section describes the Atlas V prelaunch and flight environments to which the spacecraft is exposed. Detailed spacecraft environmental data for Atlas V launch vehicles are provided for Launch Complex 41 (LC-41) at Cape Canaveral Air Force Station (CCAFS) and for Space Launch Complex (SLC)-3E 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 test requirements are described in Section 3.3.

3.1 PRELAUNCH ENVIRONMENTS

3.1.1 Thermal

The spacecraft thermal environment is maintained during ground transport and hoist. The encapsulated spacecraft thermal environment is controlled after mate to the launch vehicle and during prelaunch activity, as described in the following paragraphs.

Payload Processing Facility — Payload Processing Facility (PPF) environments in the spacecraft processing areas at Astrotech/CCAFS 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.

Ground Transport From PPF to Launch Site — During ground transport from the PPF to the Vertical Integration Facility (VIF), the temperature within the Atlas V 4-m Payload Fairing (PLF) remains between 4-30 °C (40-86 °F), with conditioning provided by an air Environmental Control System (ECS) or a Gaseous Nitrogen (GN_2) purge. If required, the ECS can maintain air temperatures between 10-25 °C (50-77 °F). The maximum dew point temperature is -37 °C (-35 °F) for GN_2 and 4.4 °C (40 °F) for air. At VAFB SLC-3E, the transporter environmental control unit maintains temperature between 10-27 °C (50-80 °F) at a maximum dew point temperature of 4.4 °C (40 °F). A GN_2 backup system is available at VAFB.

The air temperature within the Atlas V 5-m payload fairing is maintained between 4-30 °C (40-86 °F) during transport from the PPF to the VIF. This is accomplished with a mobile environmental control unit that guarantees a maximum dew point temperature of 4.4 °C (40 °F).

Hoisting Operations — During hoisting operations at the VIF and SLC-3E, the encapsulated spacecraft is purged with dry GN_2 with a maximum dew point of -37 °C (-35 °F).

Post-Spacecraft Mate to Centaur — After spacecraft mate to launch vehicle, gas conditioning is provided to the PLF at the required temperature, humidity, and flow rate. Air with a maximum dew point of 4.4 °C (40 °F) is used until approximately 4 hours before launch at LC-41 and approximately 3 hours before launch at SLC-3E, 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 Atlas V 400 and 500 series configurations at LC-41 and SLC-3E. Mission-peculiar arrangements for dedicated purges of specific components can be provided.

The ECS flow to the payload compartment is supplied through a ground/airborne disconnect on the PLF and is controlled by primary and backup environmental control units. These units provide conditioned air or GN_2 to the specifications in Table 3.1.1-2. Figure 3.1.1-1 shows the general gas-conditioning layout for the 4-m PLF configuration and Figure 3.1.1-2 shows the general gas-conditioning layout for 5-m PLF configuration.

Internal ducting in the 4-m PLF 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 PLF. The PLF air distribution system will provide a maximum airflow velocity in all directions of no more than 9.75 mps

Table 3.1.1-1: Atlas V Gas Conditioning Capabilities

			Temperature Range Inside Payload Fairing**	
			Atlas V 400 Series	**Atlas V 500 Series**
Location	**Inlet Temperature Capability***	**Inlet Flow Rate Capability, kg/min (lb/min)**	**LPF, EPF & XEPF**	**5-m Short, Medium & Long**
Post-LV Mate Through Move to Launch Configuration	10-29 °C (50-85 °F)	Atlas V 400 Series 22.7-72.6 (50-160) Atlas V 500 Series 22.7-136.2 (50-300)	6-20 °C (43-68 °F)	6-20 °C (43-68 °F)
Post-Move to Launch Configuration	10-29 °C (50-85 °F)	Atlas V 400 Series 22.7-72.6 (50-160) Atlas V 500 Series 22.7-136.2 (50-300)	6-21 °C (43-70 °F)	6-21 °C (43-70 °F)

Notes: * Inlet Temperature Is Adjustable (Within System Capability) According to Spacecraft Requirements
** Temperature Ranges Are Based on Worst-Case Minimum & Maximum External Heating Environment.
Ranges Shown Assume a 72.6 kg/min (160 lb/min) for 4-m ECS Flow Rate
136.2 kg/min (300 lb/min) for 5-m ECS Flow Rate

Table 3.1.1-2: Conditioned Air/GN_2 Characteristics

Parameter	Description Atlas V (400/500 Series)
Cleanliness	• Class 5,000 per FED-STD-209D
Inlet Temperature	• Setpoint from 10-29 °C (50-85°F) • 10-21 °C (50-70 °F) for Sensitive Operations
Inlet Temperature Control	• ±3 °C (±5 °F)
Filtration	• 99.97% HEPA Not Dioctyl Phthalate (DOP) Tested
Flow Rate	• **Atlas V 400 Series**: 22.7-72.6 kg/min ± 2.3 kg/min (50–160 lb/min ±5 lb/min) • **Atlas V 500 Series**: 22.7-136.2 kg/min ± 5.7 kg/min (50–300 lb/min ±12.5 lb/min)
Dew point (Maximum)	• 20-50% RH Air • 35-50% RH (Sensitive Operation) Air • -37.2 °C (-35 °F) GN_2

(32 fps) for the Atlas V 400 series and 10.67 mps (35 fps) for the Atlas V 500 series. There will be localized areas of higher flow velocity at, near, or associated with the air conditioning outlet. Maximum airflow velocities correspond to maximum inlet mass flow rates. Reduced flow velocities are achievable using lower inlet mass flow rates. If required, a computational fluid dynamics analysis can be performed to verify mission-unique gas impingement velocity limits.

The conditioned air is typically delivered near the top of the PLF and the flow can be divided so up to 40% or more of the gas flow is directed to the base of the payload compartment when required for spacecraft battery cooling or other ground operations.

Mission-specific arrangements for dedicated grade B or C GN_2 purges with a maximum dew point of -37.2 °C (-35 °F) of specific spacecraft components can be provided at a rate of up to 14.2 standard m^3/hr (500 standard ft^3/hr).

Figure 3.1.1-1: 4-m PLF Environmental Conditioning System

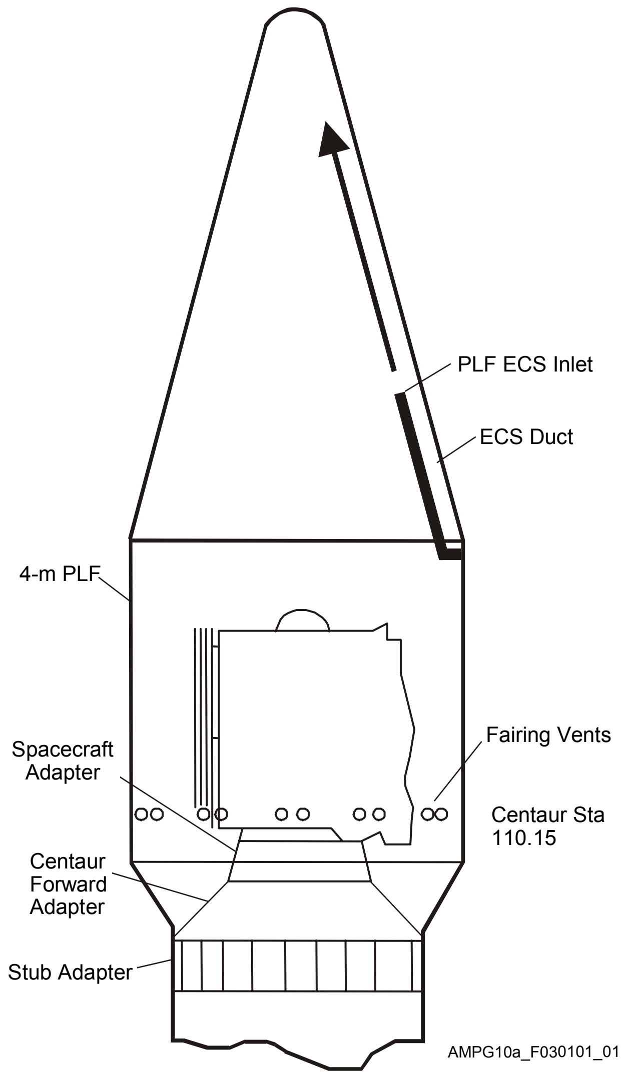
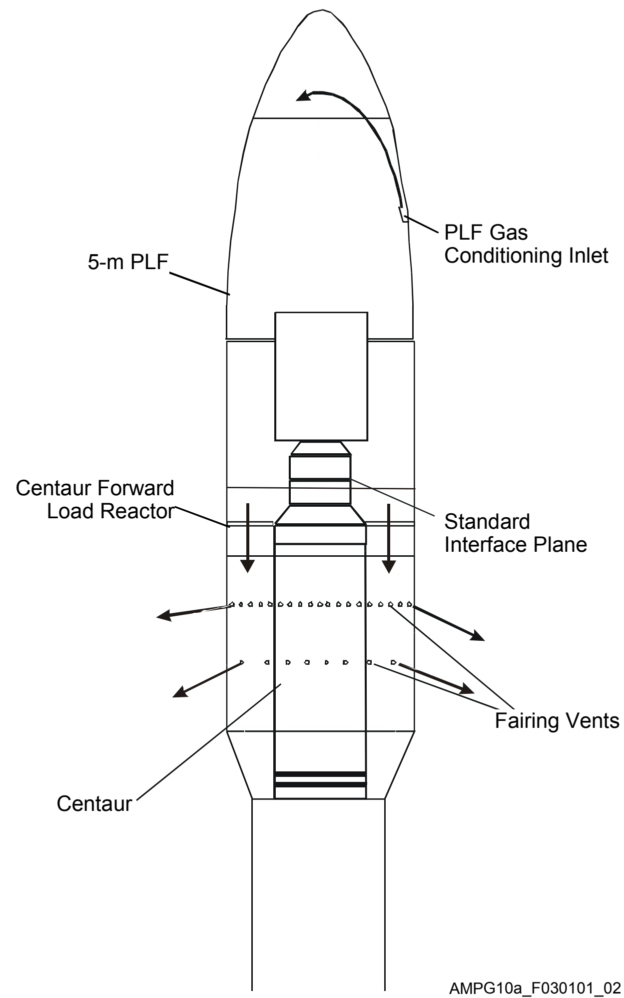


Figure 3.1.1-2: 5-m PLF Environmental Conditioning System



3.1.2 Radiation and Electromagnetics

The Electromagnetic (EM) environment is thoroughly evaluated to ensure that Electromagnetic Compatibility (EMC) is achieved for each launch. The launch services customer will be required to provide all spacecraft data necessary to support EMC analyses (See Appendix C) used for this purpose.

3.1.2.1 Launch Vehicle Intentional RD Emissions

Launch vehicle intentional transmissions are limited to the S-band telemetry transmitters (for operation with the Tracking and Data Relay Satellite System) and the C-band beacon transponder.

Launch vehicle transmitter characteristics for the Atlas V 400 and 500 series launch vehicles are shown in Tables 3.1.2.1-1 through 3.1.2.1-4.

Figure 3.1.2.1-1 shows the theoretical worst-case intentional radiated E-field emissions generated by the launch vehicle. The curve is based on transmitter fundamental 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 spacecraft (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. Intentional E-field levels at Mission Peculiar Standard Interface Plane (SIP) locations shall be addressed on an individual basis.

Table 3.1.2.1-1: Atlas V 400 Series C-Band Transponder & RF Characteristics

Atlas V 400 Series C-Band Transponder & RF Characteristics

Transmitter	
Part Number:	57-03101-3
Frequency:	5765 MHz
Frequency Stability:	± 3 MHz
Bandwith:	6 MHz
Modulation:	Pulse
Pulse Frequency:	2600 pps (max ops) 1000 pps (nominal ops)
Pulsewidth:	0.5 ± 0.1 µsec
Fixed Delay Setting:	2.5 ± 0.1 µsec
Output Power (peak):	700 watts (max)
Antenna System	
Part Number:	57-03102-4 or –5
Type:	Microstrip RHC Polarized
Locations:	
Centaur Station:	106.7 in.
Centaur Clock Angle:	162 & 342 deg
PLF Station:	N/A (No PLF Antennas)
Vehicle Minimum RF Loss:	2.6 dB

Clock Angle Pattern Cut (0 to 360 deg) in 10 deg (Theta) bands		
	Includes 3 dB Power Split	
Theta (deg.)	Omni, PLF, Gain (dBi Max)	Omni, no PLF, Gain (dBi Max)
+90	-14.64	-8.54
+80	-11.81	-4.76
+70	-5.90	-0.66
+60	+0.09	-0.30
+50	+1.14	-0.11
+40	+3.48	+1.92
+30	+3.93	+1.92
+20	+3.08	+1.28
+10	+2.25	+1.95
-10	+1.82	+1.69
-20	+1.39	+1.57
-30	+1.39	+1.05
-40	+1.22	+1.49
-50	+0.96	+1.49
-60	-0.42	-1.31
-70	-1.03	-2.67
-80	-2.47	-3.01
-90	-5.74	-4.43

Table 3.1.2.1-2: Atlas V 400 Series S-Band Transmitter & RF Characteristics

Atlas V 400 Series S-Band Transmitter & RF Characteristics

Transmitter	
Part Number:	57-03053-3
Frequency:	2211 MHz
Frequency Stability:	4.4 kHz
Bandwith:	±4 MHz
Modulation:	BPSK/QPSK
Telemetry Rate (NRZ-M):	512 or 256 kbps I-Channel
Forward Error Correction:	Convolutional Rate ½, k=7 3200 or 200 kbps Q-Channel
Output Power:	
Single Port:	26.0 watt min. 45.0 watt max.
Dual Port:	11.6 watts min. 25.4 watt max.
Antenna System	
Part Number:	57-03000-4 or –6
Type:	Raised Patch RHC Polarized
Locations:	
Centaur Station:	97.8 in.
Centaur Clock Angle:	177 & 357 deg
PLF Station:	N/A (No PLF Antennas)
Vehicle Minimum RF Loss:	1.3 dB

Clock Angle Pattern Cut (0 to 360 deg) in 10 deg (Theta) bands

	Includes 3 dB Power Split		No 3 dB Power Split
Theta (deg.)	**Omni, PLF, Gain (dBi Max)**	**Omni, no PLF, Gain (dBi Max)**	**Single Ant., No PLF, Gain (dBi Max)**
+90	-12.35	-2.72	-3.43
+80	-8.69	-1.47	-0.78
+70	-5.75	+0.80	+1.93
+60	-1.08	+1.33	+3.77
+50	+3.44	+1.61	+4.77
+40	+4.39	+2.57	+5.27
+30	+4.37	+2.97	+5.83
+20	+3.81	+3.29	+6.37
+10	+3.40	+3.86	+6.82
-10	+2.64	+3.86	+6.90
-20	+1.61	+3.49	+6.45
-30	+1.55	+2.29	+5.27
-40	+0.80	+0.85	+3.95
-50	+0.33	-0.05	+2.89
-60	-0.74	-0.41	+2.51
-70	-1.43	-1.15	+1.74
-80	-1.83	-1.62	+0.67
-90	-2.30	-1.62	+0.49

Table 3.1.2.1-3: Atlas V 500 Series C-Band Transponder & RF Characteristics

Atlas V 500 Series C-Band Transponder & RF Characteristics

Transmitter	
Part Number:	57-03101-3
Frequency:	5765 MHz
Frequency Stability:	± 3 MHz
Bandwith:	6 MHz
Modulation:	Pulse
Pulse Frequency:	2600 pps (max ops) 1000 pps (nominal ops)
Pulsewidth:	0.5 ± 0.1 µsec
Fixed Delay Setting:	2.5 ± 0.1 µsec
Output Power (peak):	700 watts (max)
Antenna System	
Part Number:	57-03102–5
Type:	Microstrip RHC Polarized
Locations:	
Centaur Station:	106.7 in.
Centaur Clock Angle:	162 & 342 deg
PLF Station:	2530.7 in.
PLF Clock Angle:	120 and 300 deg
Vehicle Minimum RF Loss:	7.0 dB, with PLF 3.6 dB, no PLF

Clock Angle Pattern Cut (0 to 360 deg) in 10 deg (Theta) bands		
	Includes 3 dB Power Split	
Theta (deg.)	Omni, PLF, Gain (dBi Max)	Omni, no PLF, Gain (dBi Max)
+90	-5.03	-5.89
+80	-4.06	-3.02
+70	-2.44	-1.20
+60	-1.78	-0.07
+50	-0.44	+0.51
+40	+0.43	+1.30
+30	+1.32	+1.63
+20	+1.51	+1.66
+10	+2.18	+1.64
-10	+2.18	+1.85
-20	+1.88	+1.30
-30	+1.86	+1.28
-40	+1.38	+0.69
-50	+0.98	+0.58
-60	-0.74	-0.39
-70	-2.21	-2.33
-80	-4.30	-3.22
-90	-4.95	-6.07

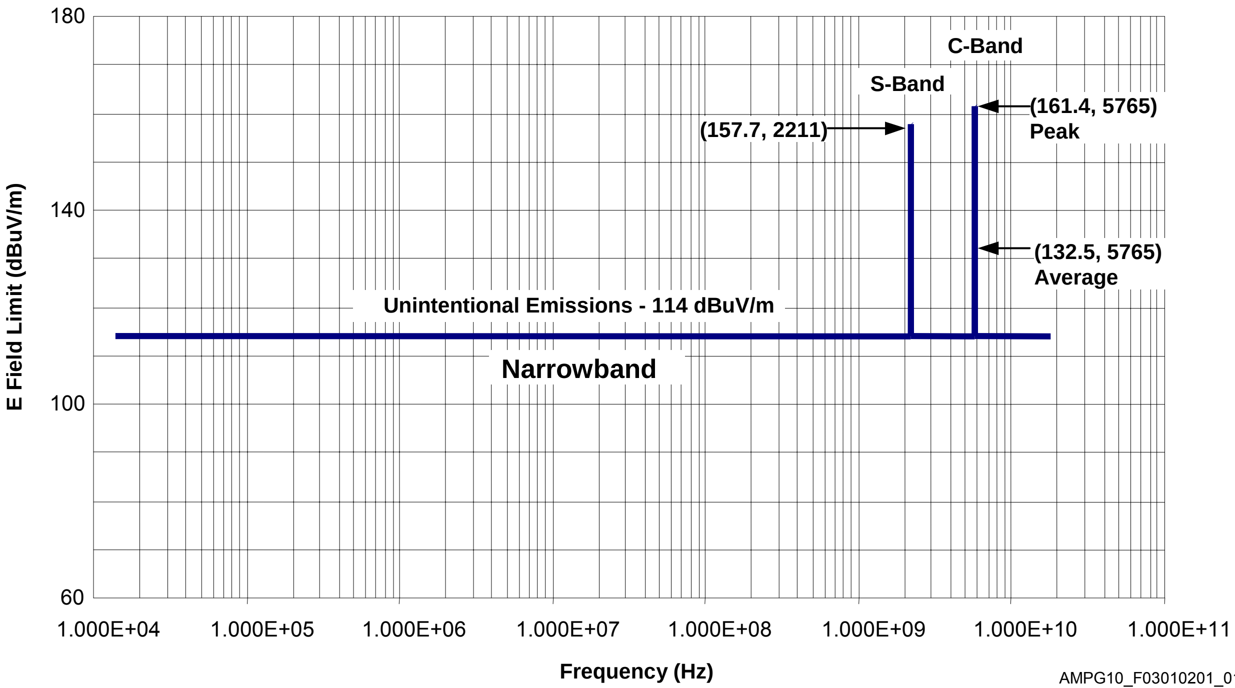
Table 3.1.2.1-4: Atlas V 500 Series S-Band Transmitter & RF Characteristics

Atlas V 500 Series S-Band Transmitter & RF Characteristics

Transmitter	
Part Number:	57-03053-3
Frequency:	2211 MHz
Frequency Stability:	4.4 MHz
Bandwith:	±4 kHz
Modulation:	BPSK/QPSK
Telemetry Rate (NRZ-M):	512 or 256 kbps I-Channel
Forward Error Correction:	Convolutional Rate ½, k=7 3200 or 200 kbps Q-Channel
Output Power:	
Single Port:	26.0 watt min. 45.0 watt max.
Dual Port:	11.6 watts min. 25.4 watt max.
Antenna System	
Part Number:	57-03000–6
Type:	Raised Patch RHC
Polarized	
Locations:	
Centaur Station:	97.8 in.
Centaur Clock Angle:	177 & 357 deg
Booster Station:	2041.6 in.
Booster Clock Angle:	180 & 357.6 deg
Vehicle Minimum RF Loss:	8.8 dB with PLF 1.3 dB, No PLF

Clock Angle Pattern Cut (0 to 360 deg) in 10 deg (Theta) bands			
	Includes 3 dB Power Split		**No 3 dB Power Split**
Theta (deg.)	**Omni, PLF, Gain (dBi Max)**	**Omni, no PLF, Gain (dBi Max)**	**Single Ant., No PLF, Gain (dBi Max)**
+90	-2.80	-2.91	-3.84
+80	-1.34	-1.99	-1.52
+70	-1.31	+0.09	+1.28
+60	-0.16	+0.95	+3.07
+50	+0.77	+2.11	+4.76
+40	+1.63	+2.51	+5.27
+30	+2.20	+2.64	+5.62
+20	+2.66	+3.05	+6.08
+10	+2.80	+3.47	+6.46
-10	+2.95	+3.56	+6.45
-20	+2.66	+2.98	+5.99
-30	+2.33	+1.86	+4.75
-40	+1.62	+0.47	+3.49
-50	+0.71	-0.56	+2.33
-60	-0.05	-0.70	+2.12
-70	-0.99	-1.45	+1.18
-80	-1.86	-1.39	+0.41
-90	-2.92	-1.39	+0.35

Figure 3.1.2.1-1: Launch Vehicle Field Radiation from Antennas



3.1.2.2 Launch Vehicle Unintentional RF Emissions

The unintentional RF emissions generated by the launch vehicle are depicted in Figures 3.1.2.2-1. Launch vehicle unintentional emissions shall not exceed an E-field level of 114 dBμV/m in the frequency range from 14 kHz to 18 GHz, as defined in Figure 3.1.2.2-1.

3.1.2.3 Launch Range Electromagnetic Environment

An EMC analysis will be performed to ensure EMC of the spacecraft/launch vehicle with the Eastern Test Range environment, including updates as available. On customer request, the launch vehicle will coordinate with ETR to address spacecraft issues with range controlled EM emitters during transport from Astrotech to LC-41 and the at launch complex itself.

Table 3.1.2.3-1 documents EM emitters in the vicinity of CCAFS. The EM environment of the launch range is primarily based on information in TOR-2001 (1663)-1 “Cape Canaveral Spaceport Radio Frequency Environment.” E-field intensities during transport from Astrotech and at LC-41 are provided. Data can be provided for other locations upon request.

Table 3.1.2.3-1 Worst-Case RF Environment for CCAFS

Transport from Astrotech to LC-41				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, V/m**	**Duty Cycle**	**Mitigation**
Radar 0.14	5,690	83.8	0.0016	Procedure Mask
Radar 1.16	5,690	61.6	0.00064	None
Radar 19.39	5,690 & 5,800	84.9	0.004	None
Radar 19.14	5,690	183.8	0.0016	None
Radar 19.17	5,690	106.5	0.0008	None
Radar 28.14	5,690	15.8	0.0016	Topography
Radar 1.8	9,410	2.1	0.0012	None
Radar ARSR-4	1,244.06 & 1,326.92	1.7	0.0006	None
Radar GPN-20	2,750 & 2,840	6.1	0.0008	None
WSR-74C	5,625	12.7	0.0064	None
WSR-88D	2,879	14.7	0.006	None
GPS Gnd Station	1,784	2.4	CW	Ops Min 3 deg
NASA STDN	2,025 & 2,120	16.9	CW	None
CSAS Orbit	416.5 & 421	0.5	CW	None
SRB Retrieval Ship	3,049.4	13.0	0.00065	None
SRB Retrieval Ship	9,413.6	3.7	0.0015	None
ET Barge	9,410	200.0	0.0012	None
LC-41				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, V/m**	**Duty Cycle**	**Mitigation**
Radar 0.14	5690	71.7	0.0016	Procedure Mask
Radar 1.16	5690	52.6	0.00064	Procedure Mask
Radar 19.39	5690 & 5800	29.5	0.004	Procedure Mask
Radar 19.14	5690	106.5	0.0016	Procedure Mask
Radar 19.17	5690	55.1	0.0008	Procedure Mask
Radar 28.14	5690	15.5	0.0016	Topography
Radar 1.8	9410	2.0	0.0012	Procedure Mask
Radar ARSR-4	1,244.06 & 1,326.92	1.4	0.0006	None
Radar GPN-20	2750 & 2840	5.2	0.0008	None
WSR-74C	5625	10.6	0.0064	None
WSR-88D	2879	12.7	0.006	None
GPS Gnd Station	1784	2.3	CW	Ops Min 3°
NASA STDN	2025 & 2120	1.0	CW	None
CSAS Orbit	416.5 & 421	0.4	CW	None
SRB Retrieval Ship	3,049.4	11.0	0.00065	None
SRB Retrieval Ship	9,413.6	3.2	0.0015	None
ET Barge	9,410	10.0	0.0012	None

Note: Sources Taken from Aerospace Report TOR-2001 (1663)-1, “Cape Canaveral Spaceport Radio Frequency Environment,” October 2000

Avg V/m = Pk, V/m*sqrt (Duty Cycle); CW = Continuous Wave

Non-shaded Sources 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 lists EM emitters in the vicinity of VAFB. This data is based on EM site survey data. Measured and theoretical, presented in CDRL B534, “Field Intensity Database,” provided by Western Test Range (WTR).

Launch trajectory and uncontrolled emitters in the area, such as nearby cruise ships or Navy vessels, may cause Radio Frequency (RF) environments to exceed the levels shown in these tables. Emitters less than 1 V/m are not recorded. On customer request, the launch vehicle will coordinate with WTR to address spacecraft issues with range controlled EM emitters during transport from the Integrated Processing Facility (IPF) and at the SLC-3E launch complex.

Table 3.1.2.3-2: Worst-Case RF Environment for VAFB

SLC-3E				
Emitter Name	**Frequency, MHz**	**Measured Intensity, V/m**	**Duty Cycle**	**Mitigation**
CT-2	416.5	2.54	CW	None
ARSR-4	1,262/1,345	18.2	0.0006	None
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	11.4 (Est)	0.0064	None
HAIR	5,400-5,900	279.25	0.002	Procedure Mask
MOTR	5,400-5,900	93.33	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	8.12	0.002	None
DRWP	404.37	NSO	0.067	Vertical Emitter
Integrated Processing Facility				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, V/m**	**Duty Cycle**	**Mitigation**
CT-2	416.5	1.14	CW	None
ARSR-4	4,262/1,345	36.4	0.0006	Procedure Mask
AN/GPN-12	2,800	100+	0.000835	None
FPS-16-1	5,725	146.5	0.001	Procedure Mask
FPS-77	5,450 & 5,650	27	0.0064	None
HAIR	5,400-5,900	286.4	0.002	Procedure Mask
MOTR	5,400-5,900	14	0.005	Procedure Mask
TPQ-18	5,840	298.3	0.0016	Procedure Mask
TPQ-39	9,180	19.0	0.00024	Procedure Mask
NEXRAD	2,890	28.21	0.002	None
DRWP	404.37	NSO	0.0067	None

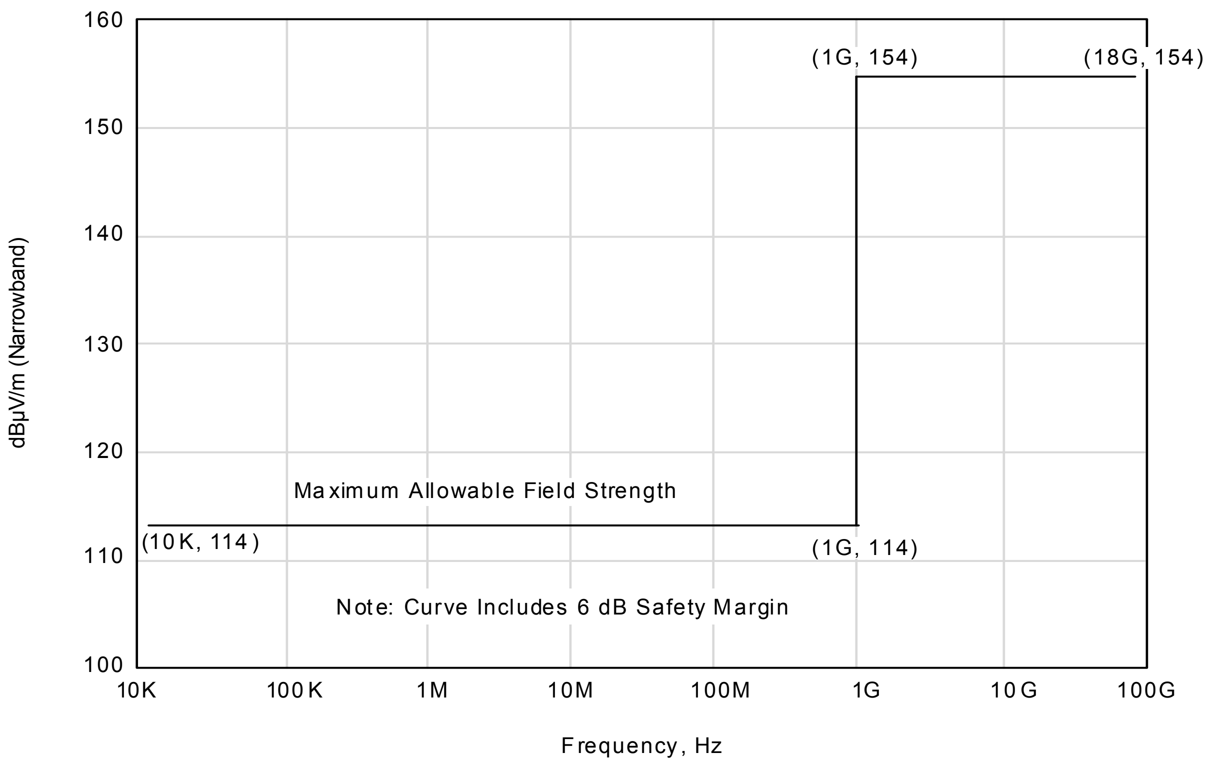
Note: Theoretical intensity data taken from the “Field Strength Measurement Data Summary at the Western Range” March 7, 1994, prepared for the 30th Space Wing.
CT: Command Transmitter; CW: Continuous Wave
Estimated (Est) values are theoretical and were not operating when measurement were taken.

3.1.2.4 Spacecraft Generated EMC Environment Limitation

During ground and launch operation time frames 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. Spacecraft shall provide available unintentional radiated emissions data to the launch vehicle in the frequency ranges form 410 to 430 MHz and from 5660 to 5720 Mhz.

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 spacecraft contractor.

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



3.1.2.5 Electrostatic Discharge Events

3.1.2.5.1 Centaur Thermal Blankets

The Centaur vehicle, depending on the trajectory employed to deliver the spacecraft to orbit, can transition through parts of the Van Allen Radiation belts. If the trajectory up through release altitude and location indicates no passage through the Van Allen belts, charging of the Centaur thermal blankets will not occur. However, if the trajectory indicates passage through the Van Allen belts, charging of the Centaur thermal blankets can occur resulting in an Electostatic Discharge (ESD) event. Figure 3.1.2.5.1-1 displays the expected ESD broadband E-field emissions that would be present at the Centaur Forward Adapter (Centaur Station 61.9).

3.1.2.5.2 Payload Fairing ESD

The 4-m PLF, when transitioning through the atmosphere, can be exposed to tribo-electric charging (P-Static) which, in turn, can result in ESD on the exterior surface. The particular area of concern is in the cone section of the PLF that has its closest approach to the spacecraft structure. If an ESD event occurred on the exterior of the PLF, the resulting E-field generated at various distances from the interior surface of the PLF are displayed in Figure 3.1.2.5.2-1. Figure 3.1.2.5.2-1 shows a significant reduction in ESD field strength when moving away from the PLF wall toward the spacecraft. Note that the ESD curves of Figure 3.1.2.5.2-1 are enveloped by the Electrostatic Arc Discharge Profile curve of MIL-STD-1514A, as shown in Figure 3.1.2.5.2-2. The 5-m PLF is not subject to this environment due to conductive paint that dissipates the charge.

Figure 3.1.2.5.1-1: Centaur Thermal Blanket ESD Driven Broadband E-field Emissions

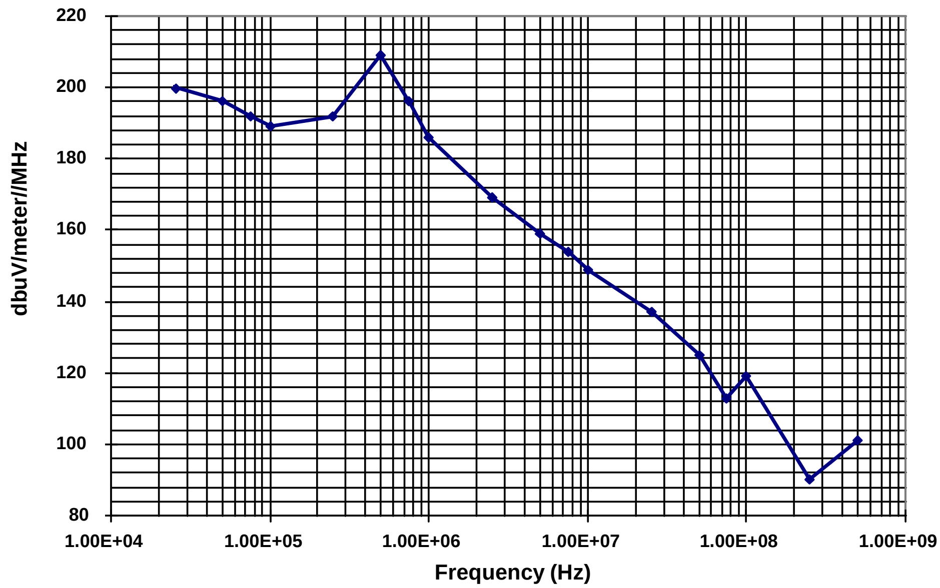


Figure 3.1.2.5.2-1: 4-m PLF ESD Driven Broadband E-field Emissions at Varying Distances from PLF Interior Wall Surface

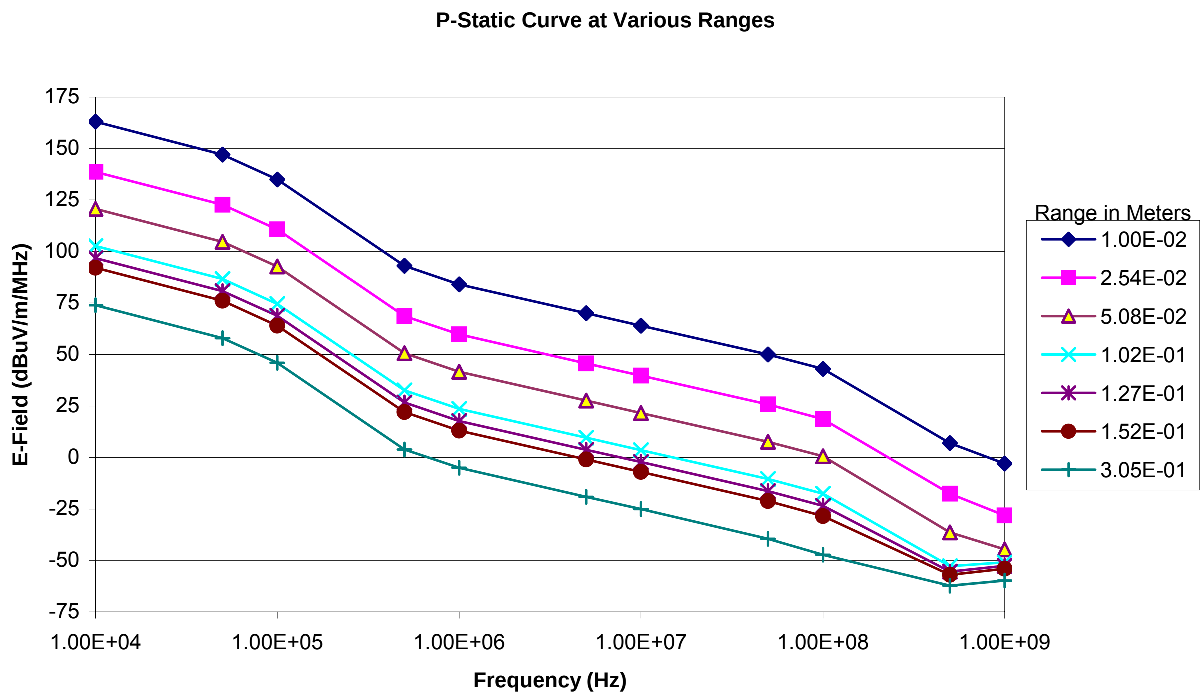
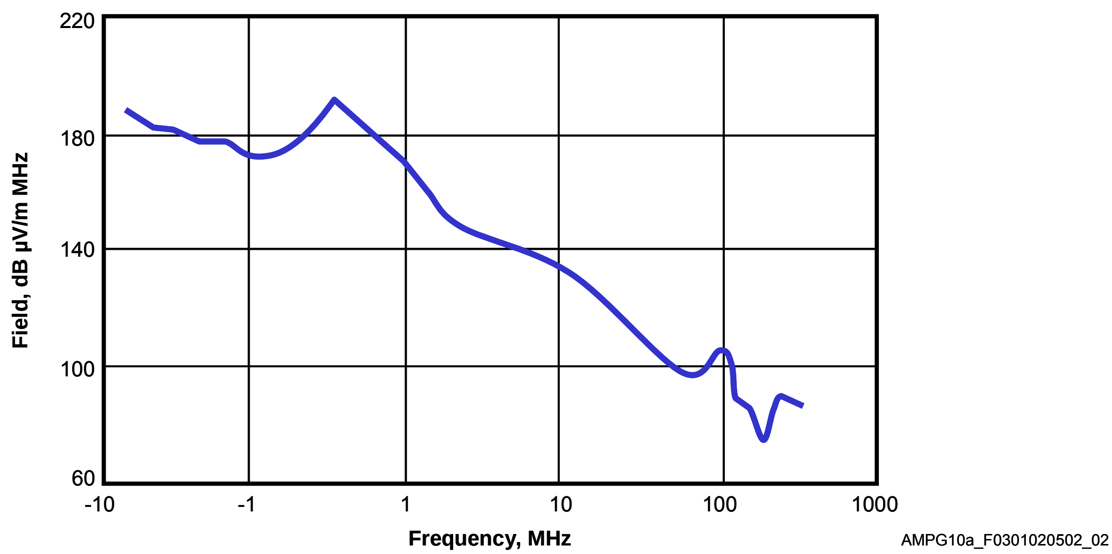


Figure 3.1.2.5.2-2: MIL-STD-1541A Arc Discharge Broadband E-field Emissions



3.1.2.6 Lightning Protection

At CCAFS, a Catenary system at the LC-41 pad and a down conduct system at the Vertical Integration Facility (VIF) provide protection against direct lightning attachment to the launch vehicle. Lightning protection at VAFB is achieved via the Mobile Service Tower (MST) when in the service position and by an air terminal located on the Umbilical Tower (UT) when the MST is in the park position. In addition, shielded payload umbilical cabling is provided between the launch vehicle and the ground support equipment at each site. Additional measures beyond these standard capabilities will be addressed on a mission unique basis.

3.1.3 Prelaunch Contamination Control and Cleanliness

Launch vehicle hardware that comes into contact with the spacecraft environment has been designed and manufactured according to strict contamination control guidelines. This hardware is defined as “contamination critical.” Contamination critical surfaces include the Centaur forward adapter, PLF interior, boattail interior, payload adapter, and for the 5-m PLFs, the top of the Centaur forward load reactor (See Section A.2.1).

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. A mission-unique appendix will be written to supplement the contamination control plan if the mission-specific requirements identified in the Interface Control Document (ICD) are more stringent than those baselined in the control plan. Some guidelines and practices used in the plan are found in the following paragraphs. Analysis of launch vehicle contamination of the spacecraft is discussed in Section 5.2.10.

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. Contamination critical hardware is entered into a controlled production phase in which the hardware is cleaned and maintained clean to prevent contaminants in difficult-to-clean places at the end of production. To support this effort, final assembly of payload adapters, the PLF, and the Centaur vehicle is performed 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. Plastic wrapping is used to protect critical surfaces during contaminant generating activities.

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. Materials that are cadmium-plated, zinc-plated, or made of unfused electrodeposited tin are avoided 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 criteria that were developed using National Aeronautics and Space Administration (NASA) SP-R-0022 as a starting point.

3.1.3.2 Contamination Control Before Spacecraft Encapsulation

Cleanliness Levels — Contamination critical hardware surfaces are cleaned and inspected to Visibly Clean Level 2. 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 foot-candles). 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 restore this level of cleanliness if the hardware becomes contaminated. Contingency cleaning procedures outside of the encapsulation facility before encapsulation are subject to Atlas V program engineering approval. The cognizant spacecraft engineer must approve any required cleaning of launch vehicle hardware in the vicinity of the spacecraft.

Certain spacecraft 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 — The Atlas V program 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, reviewed and approved by Material and Processes Engineering. Final PLF cleaning and encapsulation is performed in a Class 100,000 facility.

Cleanliness Verification — All contamination critical hardware surfaces are visually inspected to verify Visibly Clean Level 2 criteria as described above. For Atlas V, contamination critical surfaces are also verified to have less than 1 mg/ft^2 of Nonvolatile Residue (NVR). The additional verification techniques shown below can be provided on a mission-unique basis:

1. Particulate Obscuration — Tape lift sampling

2. Particulate Obscuration — Ultraviolet light inspection

3. NVR — Solvent wipe sampling

4. Particulate and Molecular Fallout — Witness plates

3.1.3.3 Contamination Control After Encapsulation

Contamination Diaphragm — Contamination barriers for 4-m and 5-m PLFs are provided to protect the spacecraft from possible contamination during transport and hoist. After the two halves of the PLF are joined, the encapsulation is completed by closing the aft opening with a ground support equipment Kevlar-reinforced, Teflon-coated diaphragm. The toriodal diaphragm stretches from the payload adapter to the aft end of the PLF cylinder and creates a protected environment for the spacecraft from mating to the Centaur through final launch vehicle closeouts prior to launch, typically one day before launch.

For the 5-m payload fairings, the entire Centaur is contained within the PLF. In this case, the launch vehicle is designed so the direction of the conditioned airflow is toward the aft end of the PLF, through the Centaur Forward Load Reactor (CFLR) deck, to minimize spacecraft exposure from sources aft of the CFLR.

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.

Personnel Controls — 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 at the launch complex to ensure that they are familiar with the procedures.

SLC-3E Controlled Work Area — The encapsulated spacecraft is contained within an environmentally controlled area on the Mobile Service Tower. The conditioned air supply to this facility is 90% filtered (removing 90% of particles 0.7 microns and larger).

The Vertical Integration Facility at LC-41 — Access to the encapsulated spacecraft is performed from work stands situated on Levels 5, 6, and 7. 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 making PLF entry at the Centaur forward adapter station to provide optimum cleanliness control as dictated by spacecraft requirements. A portable clean room tent is available for entry through mission-peculiar access doors as required (portable between VIF levels 5, 6, 6.5, and 7).

3.1.3.4 Payload Fairing Helium Environment in Prelaunch Operations

The volume between the Centaur Liquid Hydrogen (LH_2) tank and the Centaur forward adapter is purged with helium while the vehicle is on the pad with the LH_2 tank loaded, to prevent condensation on the hydrogen tank forward bulkhead. Although the Centaur forward adapter is sealed, some helium may leak into the payload compartment. The helium mixes with the GN_2 being provided by the PLF and Centaur forward adapter environmental control systems. At T-8 seconds, a pyroactivated helium vent door opens, venting the Centaur forward adapter helium into the payload compartment. Environmental control 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 forward adapter are variable and no specific exposure level can be guaranteed until further testing is performed. Some solutions, such as GN_2 purge of sensitive components, are available to helium-sensitive spacecraft on a mission-unique basis.

The Interstage Adapter (ISA) is filled with helium for launch of the Atlas V 500 series and Heavy Lift Vehicle configurations to attenuate the acoustic environment within the ISA. To accomplish this, the ISA is purged with helium for the last 5 minutes before launch. The ISA is vented through the 5-m PLF base module compartment to the PLF vents just below the CFLR deck. Flow of this helium through the CFLR into the payload compartment is minimized by the flow of GN_2 from the payload module ECS, which enters into the payload compartment at a high flow rate and is vented down through the CFLR and out the vents in the PLF base module compartment. However, no specific helium exposure level can be guaranteed at this time.

3.2 LAUNCH AND FLIGHT ENVIRONMENTS

This section describes general environmental conditions that may be encountered by a spacecraft during launch and flight of the Atlas V launch vehicle. A thorough description of each environment is presented in Section 3.2, as well as an outline of necessary spacecraft compatibility testing in Section 3.3. All flight environments defined in this section are maximum expected levels and do not include margins typically associated with qualification tests. Verification analysis necessary to assure spacecraft compatibility with Atlas V environments is performed during the Mission Integration process as described in Section 5.2.

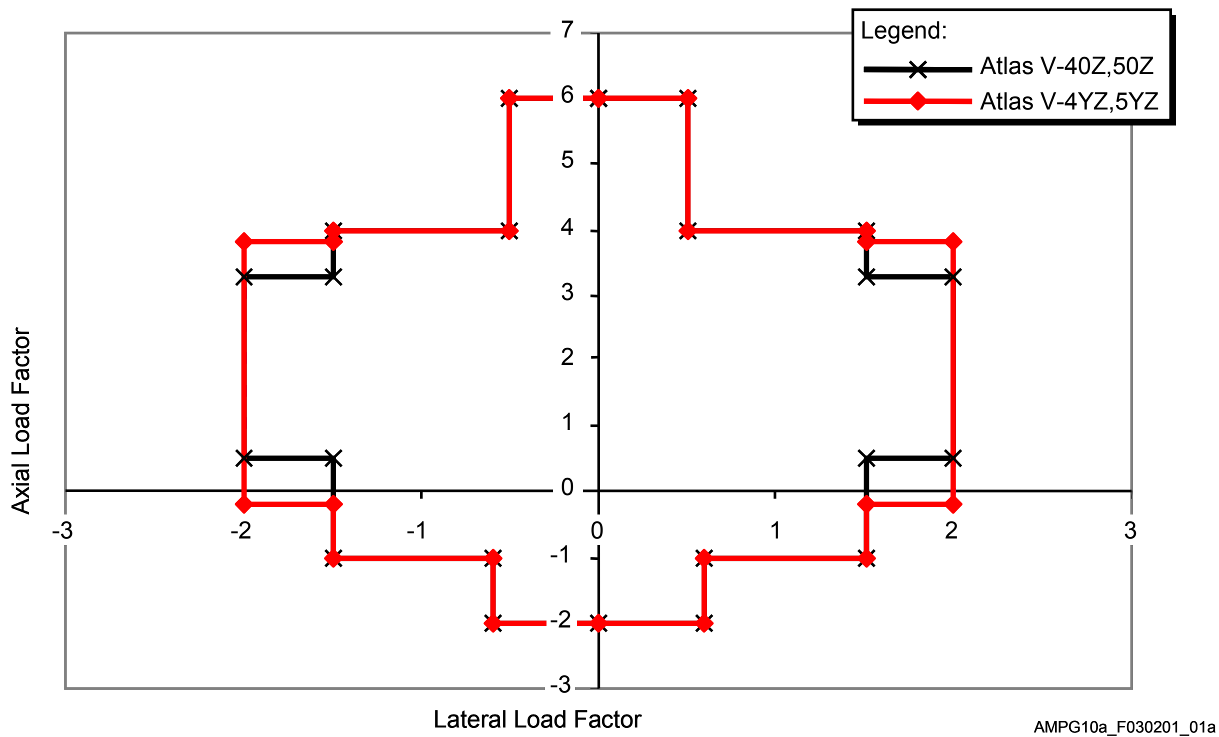
3.2.1 Spacecraft Design Load Factors

Design Load Factors (DLF) are provided in Table 3.2.1-1 and Figure 3.2.1-1 for use in preliminary design of primary structure and/or evaluation of compatibility of existing spacecraft with Atlas V launch vehicles. Factors are provided for each transient event that produces significant spacecraft loading, including any engine-induced thrust oscillation effects such as Centaur Longitudinal Event (CLE) and Booster Engine Thrust Oscillation (BETO). The total DLF for any direction can be determined by addition of the quasi-steady-state and oscillatory dynamic components provided. No uncertainty factors related to spacecraft design maturity are included in the DLF definitions.

Table 3.2.1-1: Spacecraft Design Limit Load Factors

Load Condition	Direction	Atlas V 40Z, 50Z		Atlas V 4YZ, 5YZ		Atlas V HLV	
		Steady State, g	Dynamic, g	Steady State, g	Dynamic, g	Steady State, g	Dynamic, g
Launch	Axial	1.2	±0.5	1.5	±1.5	1.2	±0.5
	Lateral	0.0	±1.0	0.0	±2.0	0.0	±1.3
Flight Winds	Axial	1.0 – 2.8	±0.5	1.0 – 2.8	±0.5	1.0 – 2.8	±0.5
	Lateral	±0.4	±1.6	±0.4	±1.6	±0.4	±1.6
Strap–On Separation	Axial	—	—	3.3	±0.5	4.1	±0.5
	Lateral	—	—	0.0	±0.5	0.0	±0.5
BECO/BETO							
(Max Axial)	Axial	5.5	+0.5	5.5	+0.5	5.5	+0.5
	Lateral	0.0	±0.5	0.0	±0.5	0.0	±0.5
(Max Lateral)	Axial	3.0-0.0	±1.0	3.0-0.0	±1.0	3.0-0.0	±1.0
	Lateral	0.0	± 1.5	0.0	± 1.5	0.0	± 1.5
SECO	Axial	—	—	—	—	—	—
	Lateral	—	—	—	—	—	—
MECO/CLE							
(Max Axial)	Axial	4.5-0.0*	±1.0	4.5-0.0*	±1.0	4.5-0.0*	±1.0
	Lateral	0.0	±0.3	0.0	±0.3	0.0	±0.3
(Max Lateral)	Axial	0.0	±2.0	0.0	±2.0	0.0	±2.0
	Lateral	0.0	±0.6	0.0	±0.6	0.0	±0.6
Sign Convention Longitudinal Axis: + (Positive) = Compression – (Negative) = Tension ± May Act in Either Direction							
Lateral & Longitudinal Loading May Act Simultaneously During Any Flight Event Loading Is Applied to the Spacecraft cg							
"Y" in vehicle designator is number of SRBs and ranges from 1 to 3 (400 Series) or 1 to 5 (500 Series) "Z" in vehicle designator is number of Centaur engines and is 1 or 2							
* Decaying to Zero							

Figure 3.2.1-1: Spacecraft Design Limit Load Factors



The load factors were derived for application to the Center of Gravity (cg) of a rigid spacecraft to generate a conservative estimate of interface loading. The actual responses of a spacecraft due to launch vehicle transients will depend on its specific static and structural dynamic characteristics; however, the values provided have generally proven conservative for spacecraft in the weight range of 1,800 to 9,100 kg (3,968 to 20,062 lb) for Atlas V 400 series vehicles, and 4,500 to 19,000 kg (9,921 to 41,888 lb) for the Atlas V 500 series vehicles and HLV. The spacecraft cantilevered fundamental mode frequencies are assumed to be a minimum of 8 Hz lateral and 15 Hz axial to ensure applicability of the design load factors. Spacecraft that do not meet these criteria will require configuration specific analyses for assessing compatibility with Atlas V launch vehicles.

Coupled Loads Analyses (CLA) are dynamic analyses conducted as part of the mission integration activity to provide spacecraft primary and secondary structure loads, accelerations, and deflections for use in design, test planning, and verification of minimum margins of safety. CLA results supercede loads derived from the Table 3.2.1-1 load factors.

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 any other. The highest acoustic level occurs for approximately 10 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 aerodynamic shock waves and a highly turbulent boundary layer. The acoustic level inside the PLF will vary slightly with different spacecraft due to acoustic absorption that varies with spacecraft size, shape, and surface material properties. Acoustic sound pressure levels for Atlas V 4-m and Atlas V 5-m PLFs are provided in Figures 3.2.2-1 and 3.2.2-2, respectively. These figures represent the maximum expected environment based on a 95% probability and 50% confidence (limit level). The levels presented are for spacecraft of square cross sectional shape with typical cross sectional fill ratios of 50-75% for the 4-m PLF and 40-50% for the 5-m PLF. A mission-peculiar acoustic analysis is required for spacecraft with other fill factors. Spacecraft should be capable of functioning properly after 1-minute exposure to these levels.

For 4-m payload fairings, special consideration should be given to components within 76 cm (30 in.) of PLF vents. Sound pressure levels for components near the PLF vents are provided in Figure 3.2.2-3.

Figure 3.2.2-1: Acoustic Levels for Atlas V 400 Series with 4-m Payload Fairings (with Blankets)

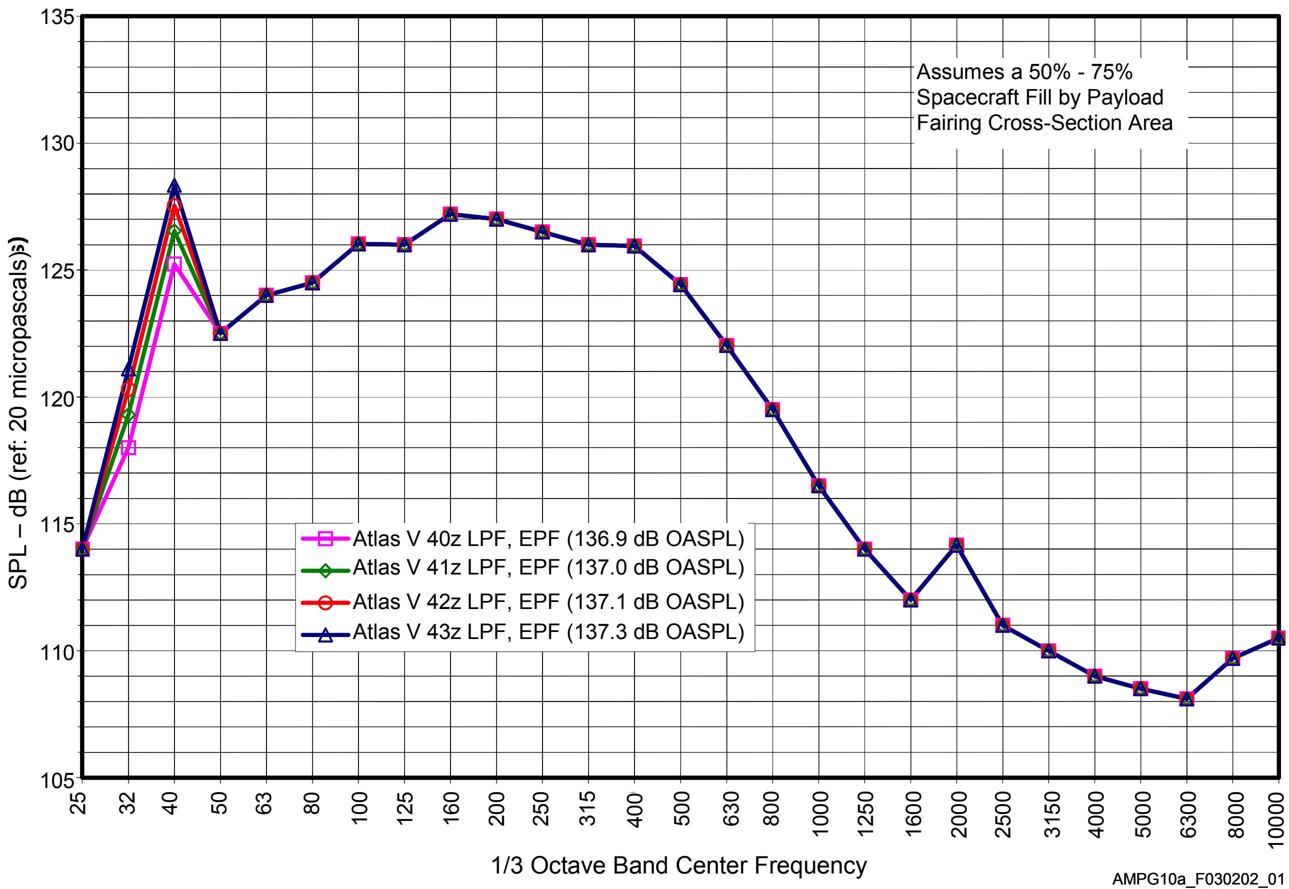


Figure 3.2.2-2: Acoustic Levels for Atlas V 500 Series with 5-m Payload Fairings

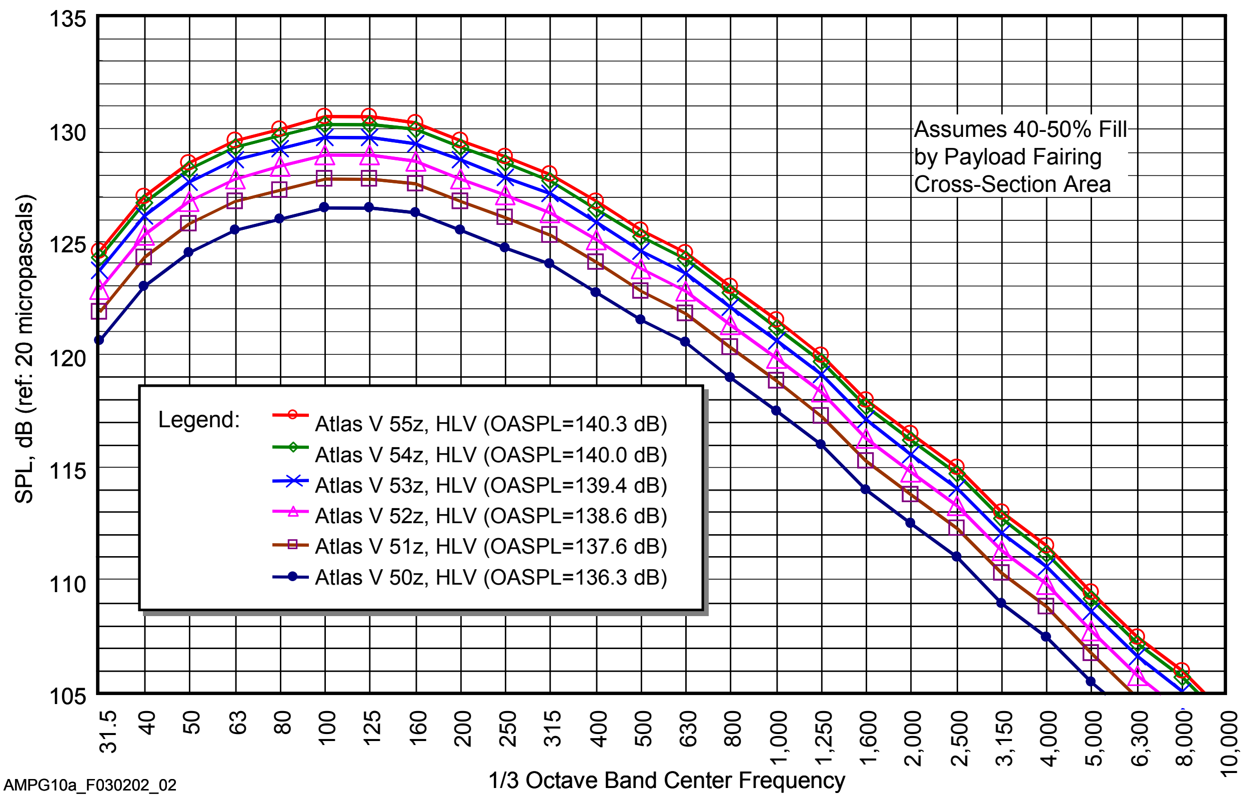
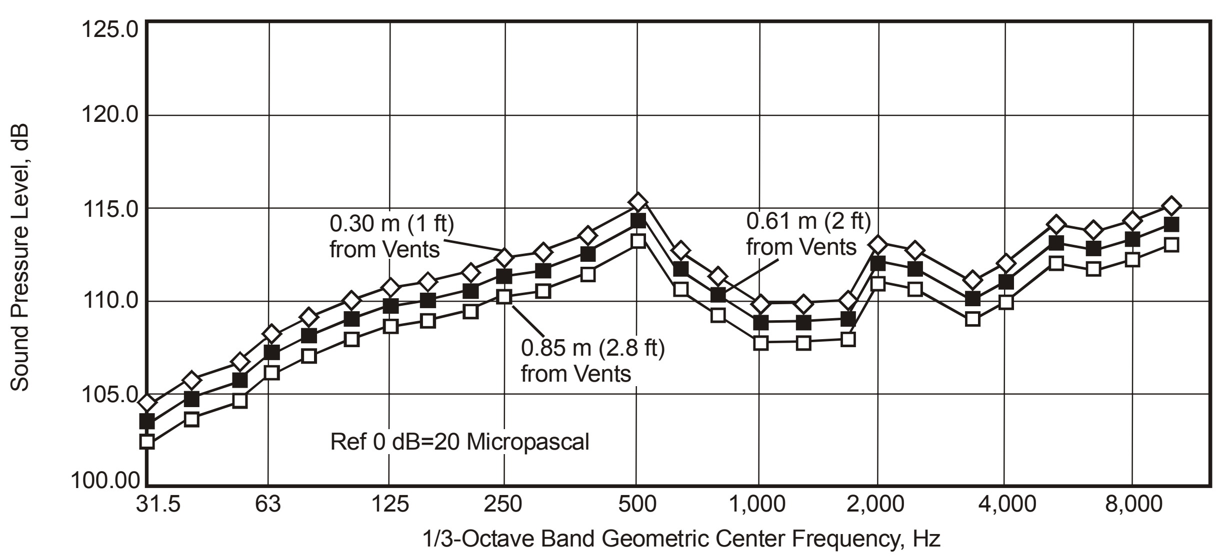


Figure 3.2.2-3: Acoustic Levels Near Vents with 4-m Payload Fairings



3.2.3 Vibration

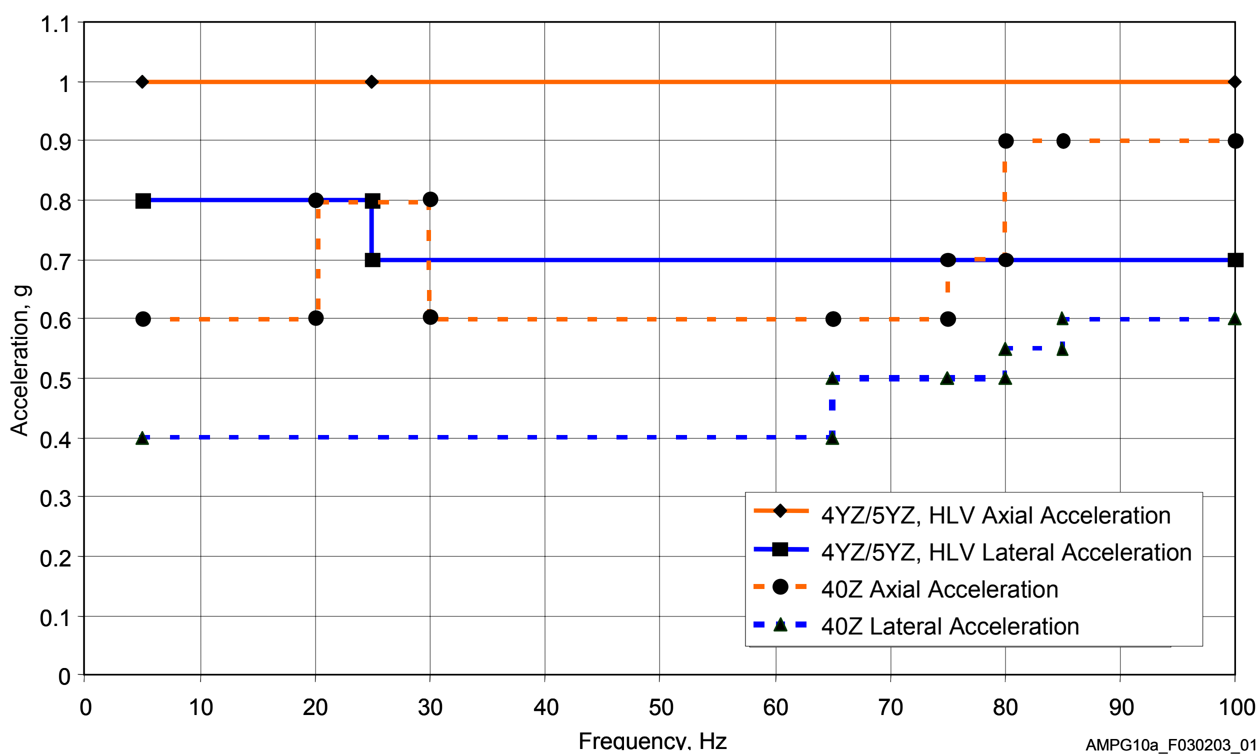
The spacecraft is exposed to a vibration environment that may be divided into two general frequency ranges: low frequency quasi-sinusoidal vibration, and high frequency broadband random vibration.

The low frequency vibration tends to be the design driver for spacecraft structure. The flight measured equivalent sine vibration near the spacecraft interface for the Atlas V 400 and 500 series vehicles are shown in Figure 3.2.3-1. The envelope shown is based on a 99% probability and 90% confidence statistical envelope. Most peak responses occur for a few cycles during transient events, such as launch, wind gusts, Booster Engine Cutoff, jettison events, and Centaur Main Engine Cutoff. Other flight events produce multicycle responses such as buffet, Centaur Longitudinal Event, and Booster Engine Thrust Oscillation.

The low frequency transient response environment during Atlas/Centaur flight is characterized by a combination of the equivalent sinusoidal vibration specified at the spacecraft interface and Coupled Dynamic Response Analysis. Verification of minimum factors of safety is required by performance of a system level vibration test (See Section 3.3). CLA is used to notch the test input in the 0 to 50 Hz range to maintain component responses below design levels. Notching in the 50 to 100 Hz frequency range is not recommended without technical discussion concerning flight equivalent levels as related to actual spacecraft damping.

High frequency random vibration that the spacecraft experiences is primarily due to the acoustic noise field, with a very small portion being mechanically transmitted through the spacecraft interface. The acoustically excited random vibration environment tends to be the design driver for lightweight components and small structural 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 adapter interface depends on the adjacent structure above and below the interface, the exact interface levels depend on the structural characteristics of the lower portion of the spacecraft, the particular payload adapter, and the

Figure 3.2.3-1: Quasi-Sinusoidal Vibration Levels for Atlas V 400 Series and Atlas V 500 Series



influence of the acoustic field for the particular spacecraft. Acoustic test of the spacecraft will therefore be the most accurate simulation of the high frequency environment experienced in flight and is preferable to base input random vibration test. If the spacecraft is mounted to a test fixture that has structural characteristics similar to the payload adapter, then vibration levels at the interface will be similar to flight levels. To accurately reflect the flight environment, it is not recommended to attach the spacecraft to a rigid fixture during acoustic testing.

3.2.4 Shock

Four pyrotechnic shock events occur during flight on the Atlas V vehicles. These events are PLF jettison, Centaur separation from the Atlas V booster, spacecraft separation, and separation of the CFLR for the Atlas V 500 series vehicles. The CFLR provides a structural connection between the top of the Centaur and the 5-m PLF for reduced loss of clearance within the payload compartment. In general, the spacecraft separation device generates the maximum shock environment for the spacecraft. While the other events do produce noticeable shock, the levels are significantly lower than separation system shock due to distance attenuation.

Figure 3.2.4-1 shows the maximum expected spacecraft separation shock levels for a typical spacecraft at the separation plane for the Atlas V standard payload adapters. The most significant shock generated by the Atlas V launch vehicle prior to spacecraft separation is PLF jettison. The shock level for the PLF jettison event is shown on the figure for comparison. All shock environments are defined on the Atlas V launch vehicle side of the interface and represent a 95% probability and 50% confidence envelope with a resonant amplification factor (Q) of 10. Response on the spacecraft side will be dependent on the unique characteristics of the spacecraft's interface structure.

Figure 3.2.4-1: Typical Maximum Atlas Shock Levels – Atlas V Standard Payload Adapters

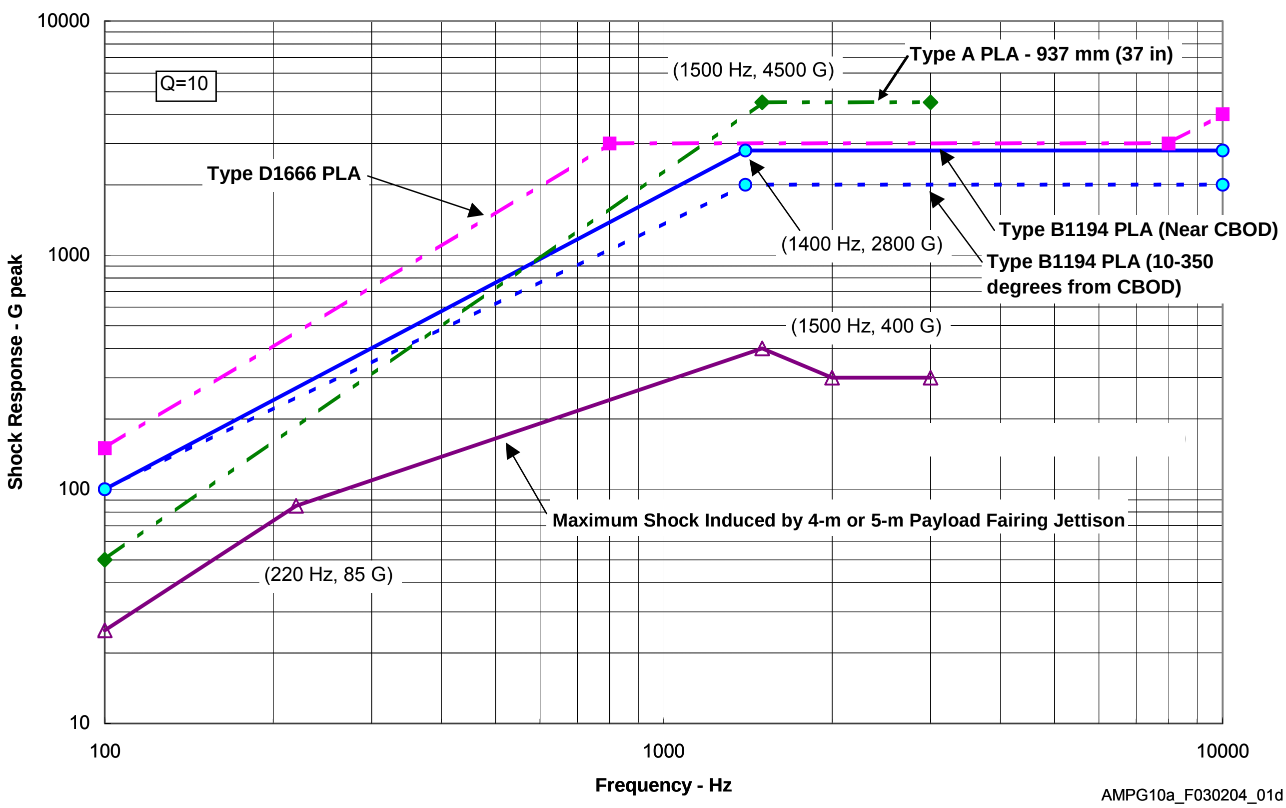
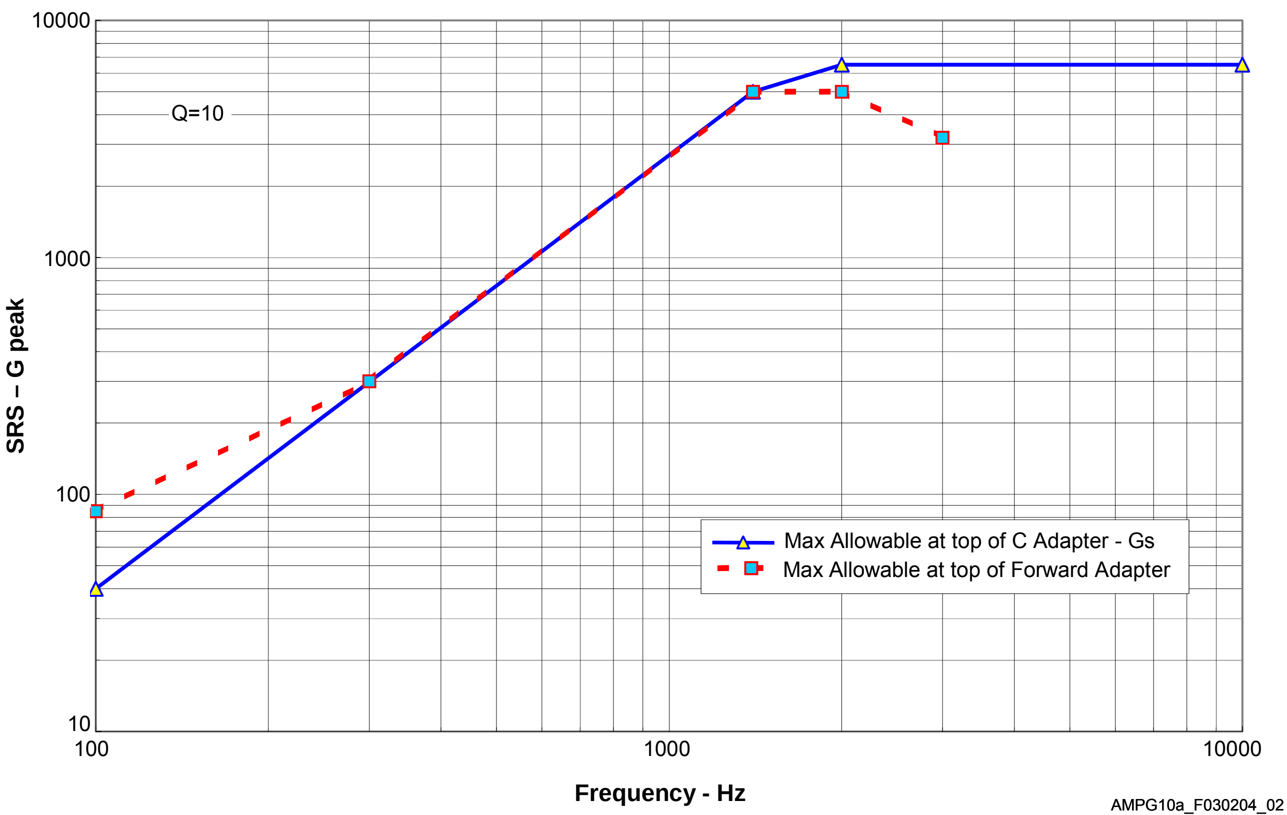


Figure 3.2.4-2 shows the maximum acceptable shock level at the Centaur forward adapter and C22 launch vehicle adapter interface for a customer provided payload adaptor and/or separation system. This requirement is necessary to assure Atlas V launch vehicle component compatibility with spacecraft-induced shock.

Figure 3.2.4-2: Maximum Allowable Spacecraft-Produced Shock at Centaur Forward Adapter



3.2.5 Thermal

Within Payload Fairing — The PLF protects the spacecraft during ascent to an altitude of approximately 113,000 m (370,000 ft). Aerodynamic heating on the PLF results in a time-dependent radiant heating environment around the spacecraft before PLF jettison. The PLF uses cork on the external surface to minimize PLF skin temperatures.

For the Atlas V 4-m PLF, the inner PLF bare skin surfaces of the cone and cylinder have a low emittance finish (ε <0.1) that minimizes heat transfer to the spacecraft. The PLF boattail and inner acoustic blanket surfaces facing the spacecraft for the 4-m PLF have an emittance of ≤0.9 (ε ≤0.9). The peak heat flux radiated by the 4-m PLF bare cone and cylinder inner surfaces is less than 400 W/m^2 (125 Btu/hr-ft^2) and peak PLF skin temperatures remain below 204 °C (400°F) at the warmest location. Inner acoustic blanket temperatures remain below 49°C (120°F).

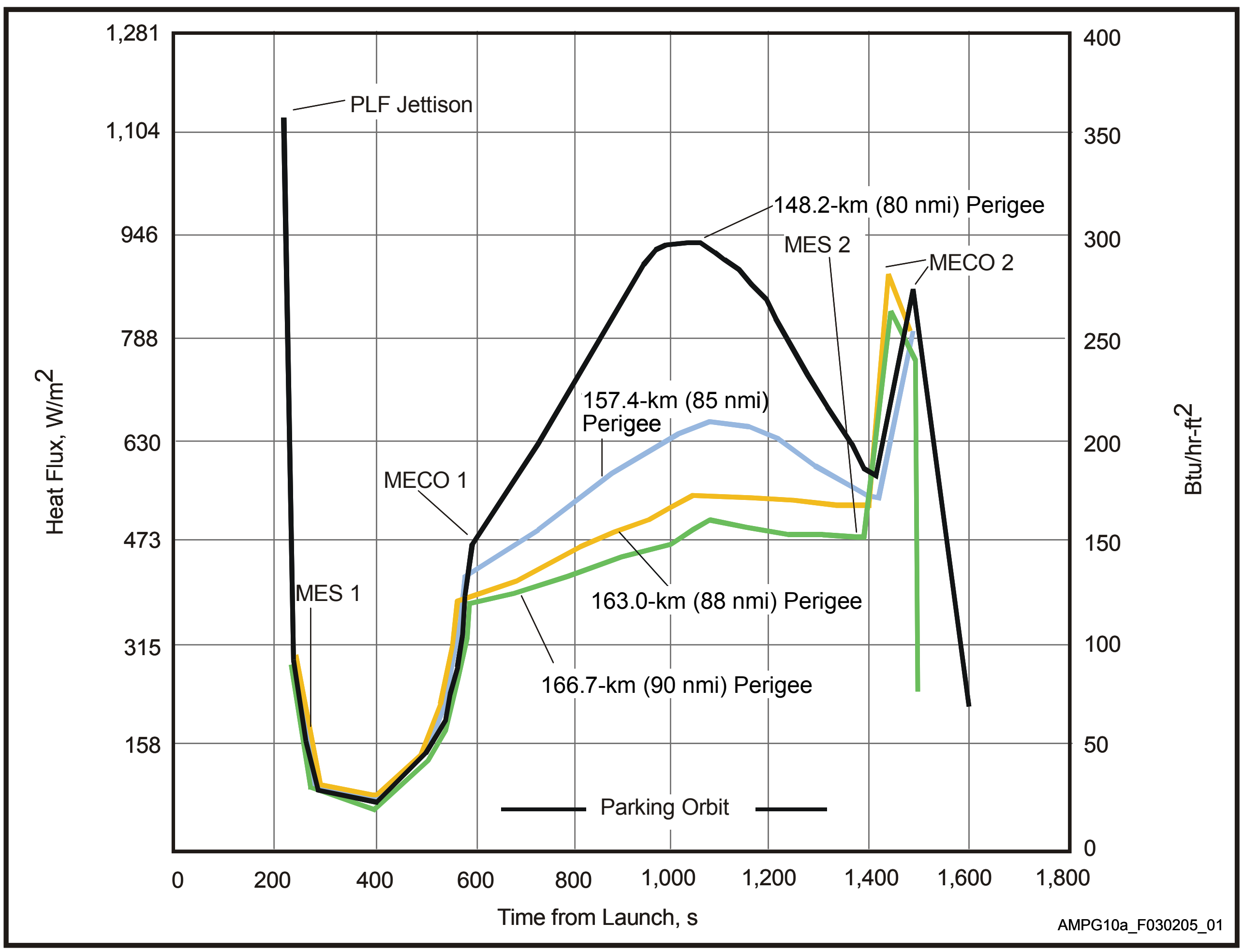
For the Atlas V 5-m PLF, the spacecraft thermal environment is attenuated by the acoustic suppression system that is baselined for the PLF cylinder and the lower portion of the ogive nose section. The inner surfaces of the bare composite 5-m PLF nose and cylinder have an emittance of 0.9. The inner surfaces of the acoustic suppression system have an emittance of ≤ 0.8. The peak heat flux radiated by the inner surfaces of the bare cone and cylinder of the 5-m PLF is less than 536 W/m^2 (170 Btu/hr-ft^2), and peak temperatures remain below 93°C (200°F), at the warmest location. The inner acoustic suppression system surface temperatures for the 5-m PLF remain below 93°C (200°F).

After Payload Fairing Jettison — PLF jettison typically occurs when the 3-sigma maximum free molecular heat flux decreases to 1,135 W/m^2 (360 Btu/hr-ft^2). PLF jettison timing can be adjusted to meet specific mission requirements. Free Molecular Heating (FMH) profiles are highly dependent on the trajectory flown. Typical FMH profiles for standard parking orbit coast length missions (≤ 15 minutes) 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. Raising the parking orbit perigee altitude can reduce peak FMH levels; however, it will have a minor negative effect on delivered launch vehicle performance.

The spacecraft thermal environment following PLF jettison includes free molecular heating, solar heating, Earth albedo heating, Earth thermal heating, and radiation to the upper stage and to deep space. The spacecraft also is conductively coupled to the forward end of the Centaur through the payload 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.

For a typical 30-minute geosynchronous transfer orbit mission, 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 0 to 49°C (32 to 120°F) at the forward end of the payload adapter. Neither the Centaur main engine plumes nor reaction control system engine plumes provide any significant heating to the spacecraft. The Centaur main engine plumes are nonluminous due to the high purity of LH_2 and Liquid Oxygen (LO_2) reactants.

Figure 3.2.5-1: Typical FMH Flux Profiles (Information Use Only)



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 PLF design and trajectory.

The 4-m and the 5-m PLFs were designed to have a depressurization rate of no more than 6.20 kPa/s (0.9 psi/s). The pressure decay rate will generally be less than 2.5 kPa/s (0.36 psi/s), except for a short period around transonic flight when the decay rate will generally not exceed 5.0 kPa/s (0.73 psi/s). Typical depressurization rates are less than these values. Figures 3.2.6-1 through 3.2.6-2 illustrate typical bounding pressure profiles and depressurization rates for the 4-m PLF. Figures 3.2.6-3 and 3.2.6-4 illustrate typical pressure profiles and depressurization rates for the 5-m PLF.

3.2.7 In-flight Contamination Control

Launch system ground contamination sources were addressed in Paragraphs 3.1.3.2 and 3.1.3.3. Launch system ascent contamination sources are discussed below.

3.2.7.1 Molecular Outgassing

Nonmetallic materials outgas molecules that can deposit on spacecraft surfaces. This source is limited by choosing low outgassing materials where possible and by limiting, encapsulating, or vacuum baking higher outgassing materials.

3.2.7.2 Nonvolatile Residue Redistribution

PLF and other surfaces in the PLF volume will have small amounts of adsorbed molecules that desorb when these surfaces are warmed. They can deposit on spacecraft surfaces that are cooler than the condensation temperature of these molecules. Atlas V hardware is cleaned and tested to less than 1 mg/ft^2 of NVR.

3.2.7.3 Particle Redistribution

Particles on surfaces within the PLF volume can shake loose and redistribute to spacecraft surfaces during launch and flight. This source is limited by cleaning hardware to Visibly Clean Level 2 criteria before encapsulation. The PLF is also vibrated as part of the cleaning process. Additional launch vehicle hardware cleanliness levels may be specified to meet mission unique requirements.

3.2.7.4 Payload Fairing Separation

Separation of the 4-m PLF is accomplished using pyrotechnic separation bolts and jettison springs. The pyrotechnic bolts on the PLF are located in individual cavities that isolate them from the spacecraft. Particle production from PLF jettison springs has been tested and is negligible. The 5-m PLF halves are separated by a linear pyrotechnic separation system that is fully contained in an expandable bellows. The bellows expands forcing the shearing of a rivet line. The sheared rivets are also retained by tape. Spacecraft contamination from this system is negligible based on the results of analyses and tests.

Figure 3.2.6-1: Typical Static Pressure Profiles Inside the EPF for Atlas V 431

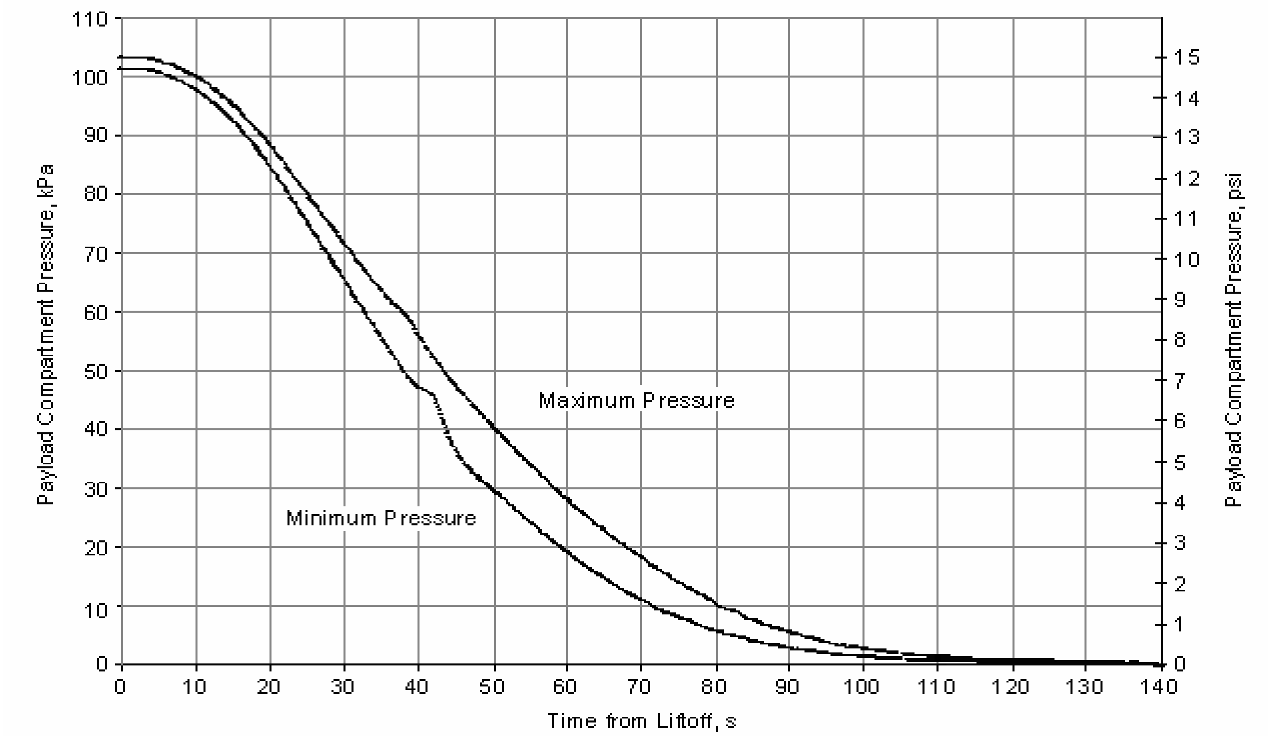


Figure 3.2.6-2: Typical Payload Compartment Pressure Decay Rate Inside the EPF for Atlas V 431

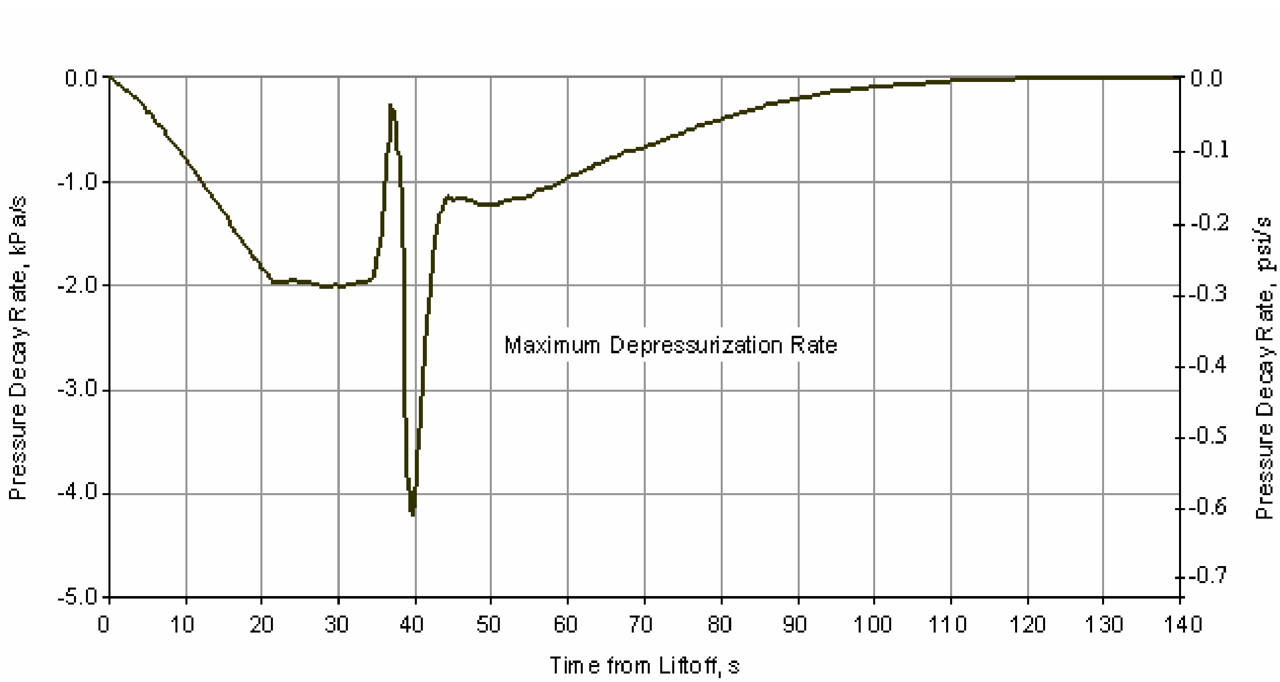


Figure 3.2.6-3: Typical Static Pressure Profiles Inside the 5-m Fairing

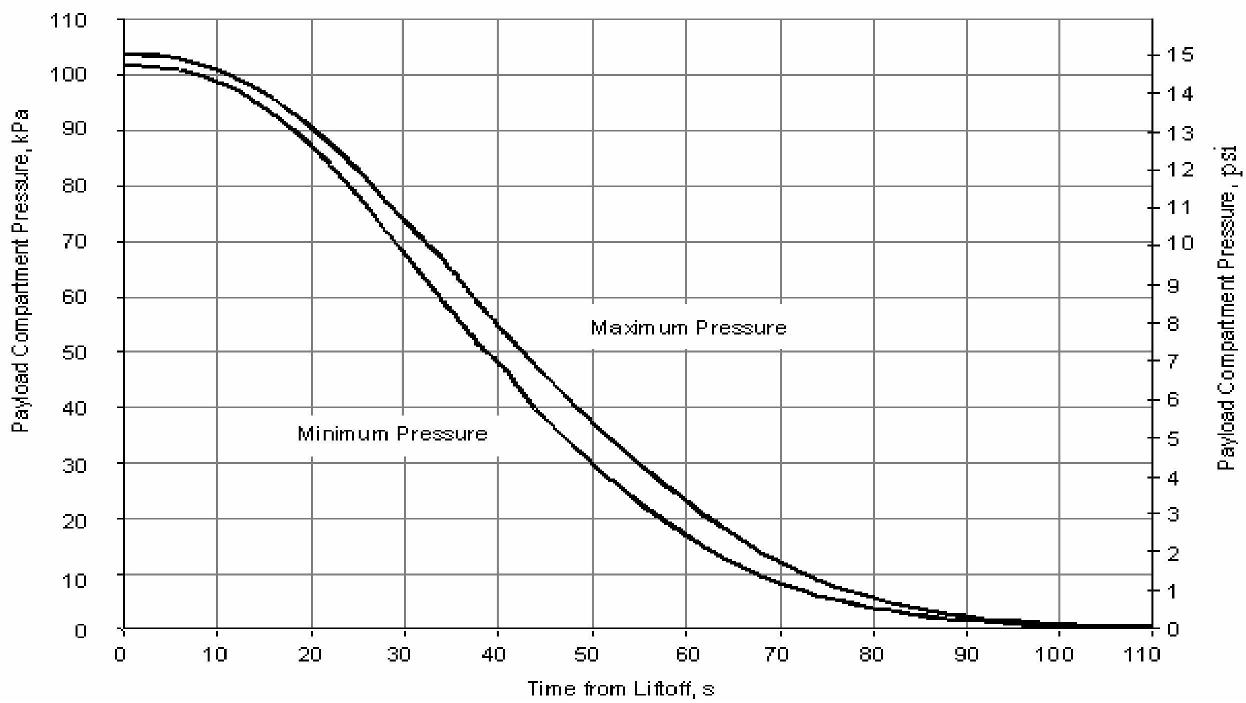
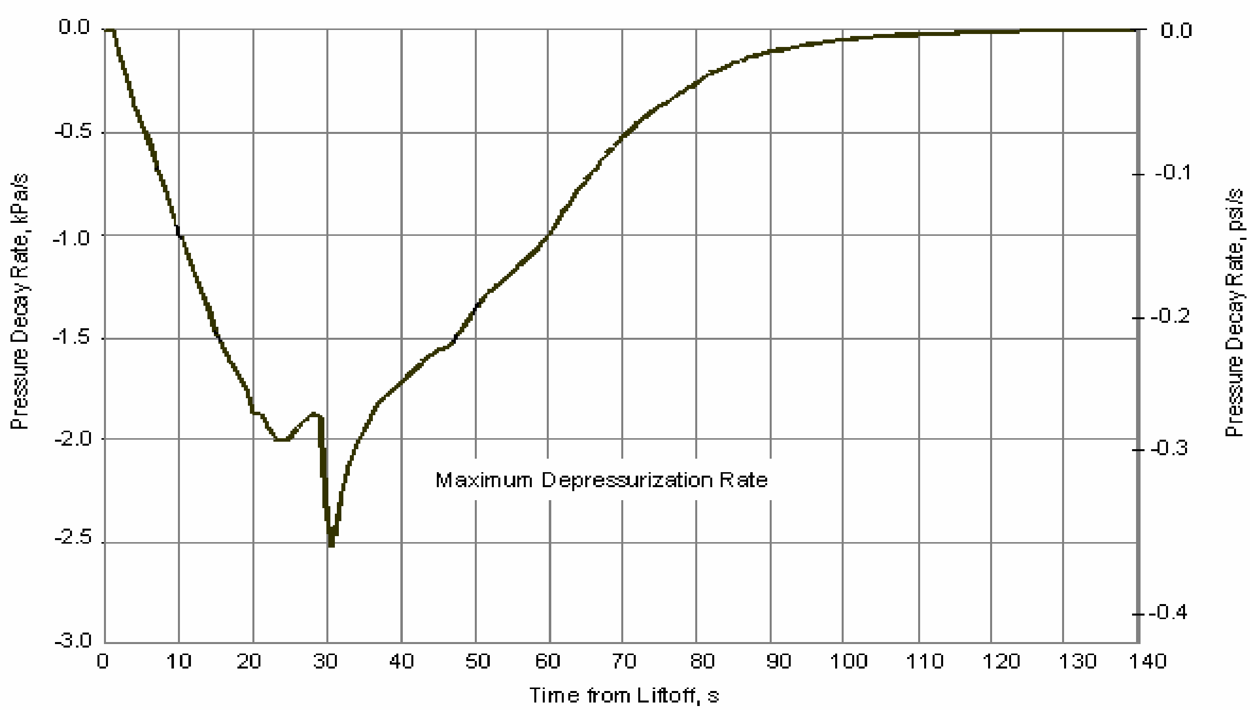


Figure 3.2.6-4: Typical Payload Compartment Pressure Decay Rate for the 5-m Fairing



3.2.7.5 Booster Separation

The Atlas V booster is separated from the Centaur by a linear charge and propelled away from it by retrorockets. These two systems are discussed in the following paragraphs.

3.2.7.5.1 Atlas V 400 Series

The Atlas V 400 series mission designs have the PLF in place on the vehicle during detonation of the linear charge and firing of the retrorockets. PLF leak areas are small and entrance of gases through vent areas is controlled by one-way flapper doors on all PLF designs. This virtually eliminates the separation system as a contamination source since there is no credible transport path from the retrorockets to the spacecraft.

3.2.7.5.2 Atlas V 500 Series and HLV

The booster is separated from Centaur with a frangible joint, similar to the PLF Horizontal Separation System (HSS), located just aft of the Centaur aft tank ring. This event occurs inside the PLF boattail that remains even though the PLF has been jettisoned before the booster separation event.

3.2.7.5.3 Booster Retrorockets

After separation, eight retrorockets located in the booster intertank compartment are fired to ensure the expended booster moves away from Centaur. These eight retrorockets use solid propellants and are canted outboard. The resulting plume products consist of small solid particles and very low density gases. The boattail shields the spacecraft from particles in the retrorocket plume. Retrorocket plume gases that expand around the boattail and impinge on the spacecraft are rarefied and only a small fraction are condensable because spacecraft surfaces are still relatively warm from prelaunch payload compartment gas conditioning. Moderate molecular depositions can be expected on aft facing surfaces.

3.2.7.6 Spacecraft Separation Event

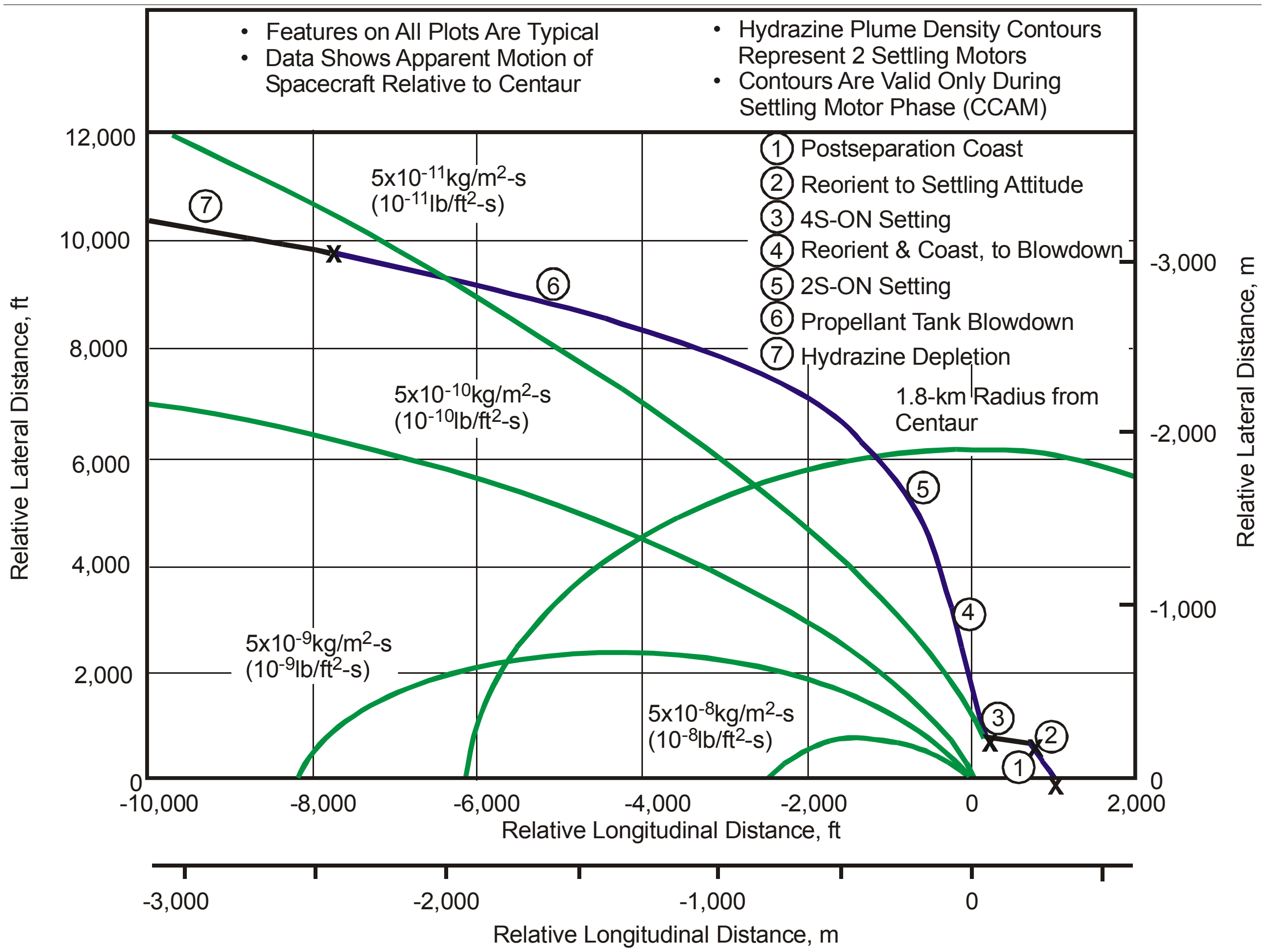
Atlas V uses three types of spacecraft separation systems. The first system used on Type A payload adapters, uses two pyrotechnically initiated bolt cutters to release two clamp halves that hold the spacecraft to the launch vehicle payload adapter. Upon actuation, small particles can be generated by this system. The kinetic energy of particles observed during separation system testing was approximately two orders of magnitude less than the micrometeoroid 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.). This system is similar to those used throughout the launch vehicle industry for many years. The second system, termed the Low Shock Payload Separation System (LSPSS), employs a sealed pyrotechnic pin-puller to release potential energy stored as component tension to release the clamp band device. This system does not have the bolt cutter and produces a few small particles, but no debris. Sealed pyrotechnic bolts are sometimes used. These also produce only minor quantities of small particles.

3.2.7.7 Collision and Contamination Avoidance Maneuver

The Centaur Reaction Control System (RCS) consists of 12 hydrazine (N_2H_4) thrusters for Centaur propellant settling and attitude-control requirements. Four thrusters provide axial thrust and eight provide roll, pitch, and yaw control. Thrusters are located slightly inboard on the Centaur aft bulkhead. Thrusters produce a plume that has an extremely low contaminant content. Before Centaur/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 and Contamination Avoidance Maneuver (CCAM). CCAM is designed to prevent recontact of the Centaur with the separated spacecraft while minimizing contamination of the spacecraft.

A typical CCAM sequence is shown in Figure 3.2.7.7-1. 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 (two settling motors 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. Analysis of spacecraft contamination from this event predicts a maximum deposition of 3.0 x 10^{-9} g/cm^2 (0.3Å) on spacecraft surfaces from the Centaur vehicle.

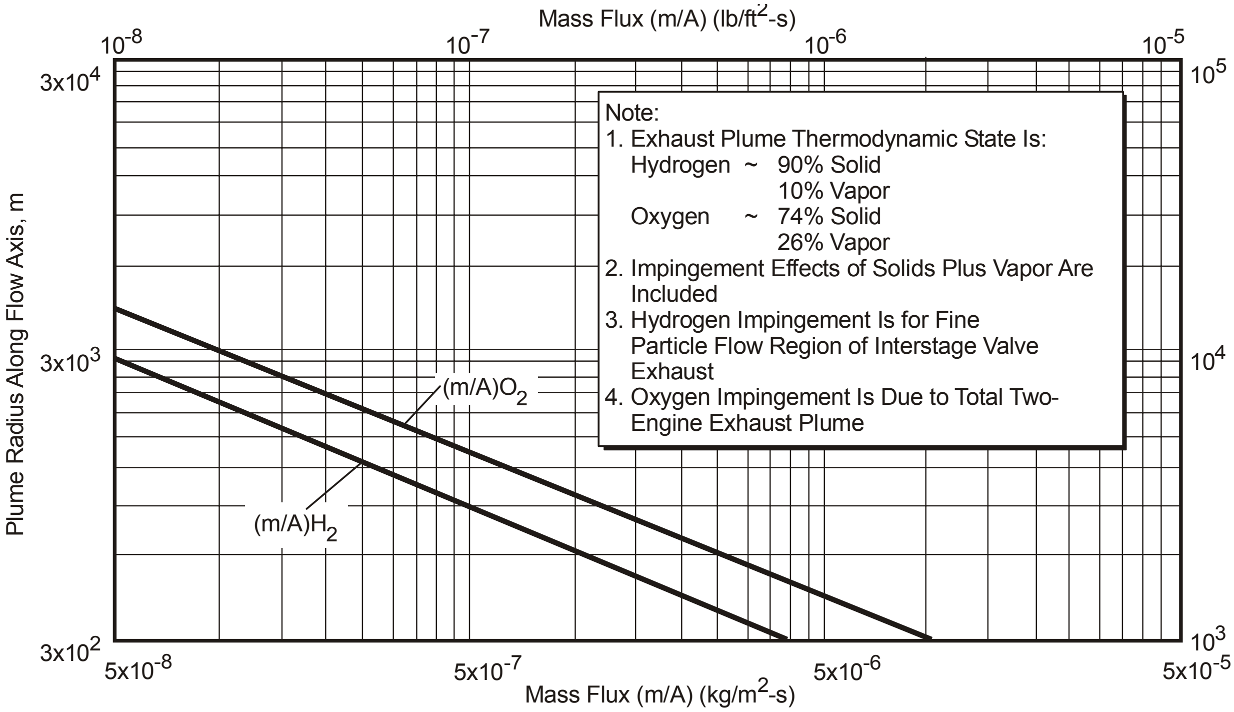
Figure 3.2.7.7-1: Typical Spacecraft Motion Relative to Centaur



3.2.7.8 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 Centaur/spacecraft separation distance. Hydrogen is expelled out the engine cool down 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. Main engine blowdown does not begin until a relative separation distance ≥ 1.9 km (1.0 nmi) has been attained. Figure 3.2.7.8-1 identifies typical main engine blowdown exhaust product impingement rates on the spacecraft.

Figure 3.2.7.8-1: Typical Spacecraft Impingement Fluxes During Propellant Tank Blowdown



3.2.8 Radiation and EMC

The description of environments in Section 3.1.2 encompasses worst case flight environments with some individual exceptions, which are dependent upon launch trajectory. Any individual exceedances will be addressed for specific launches.

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 compatibility of the spacecraft with flight environments.

Table 3.3-1: Spacecraft Qualification and Acceptance Test Requirements

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

The spacecraft testing required for demonstration of compatibility is listed in Table 3.3-1. Table 3.3-2 describes structural environment test 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 system level vibration test shall include wet propellant tanks for new designs if acceptance testing will not include this important effect.

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.

Flight hardware fit checks are performed to verify mating interfaces and envelopes. Verification of spacecraft compatibility with shock produced by separation systems provided by the Lockheed Martin is typically demonstrated by firing of a flight configured system following the FM fit check. This test may also be performed during STM or PFM testing to establish a mapping of shock levels for component locations near the interface. Component unit qualification testing must envelop the mapped environment. For user-supplied adapters and separation systems, firing of the actual separation device on a representative payload adapter and spacecraft to measure the actual level and/or qualify the spacecraft is recommended.

Table 3.3-2: 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.

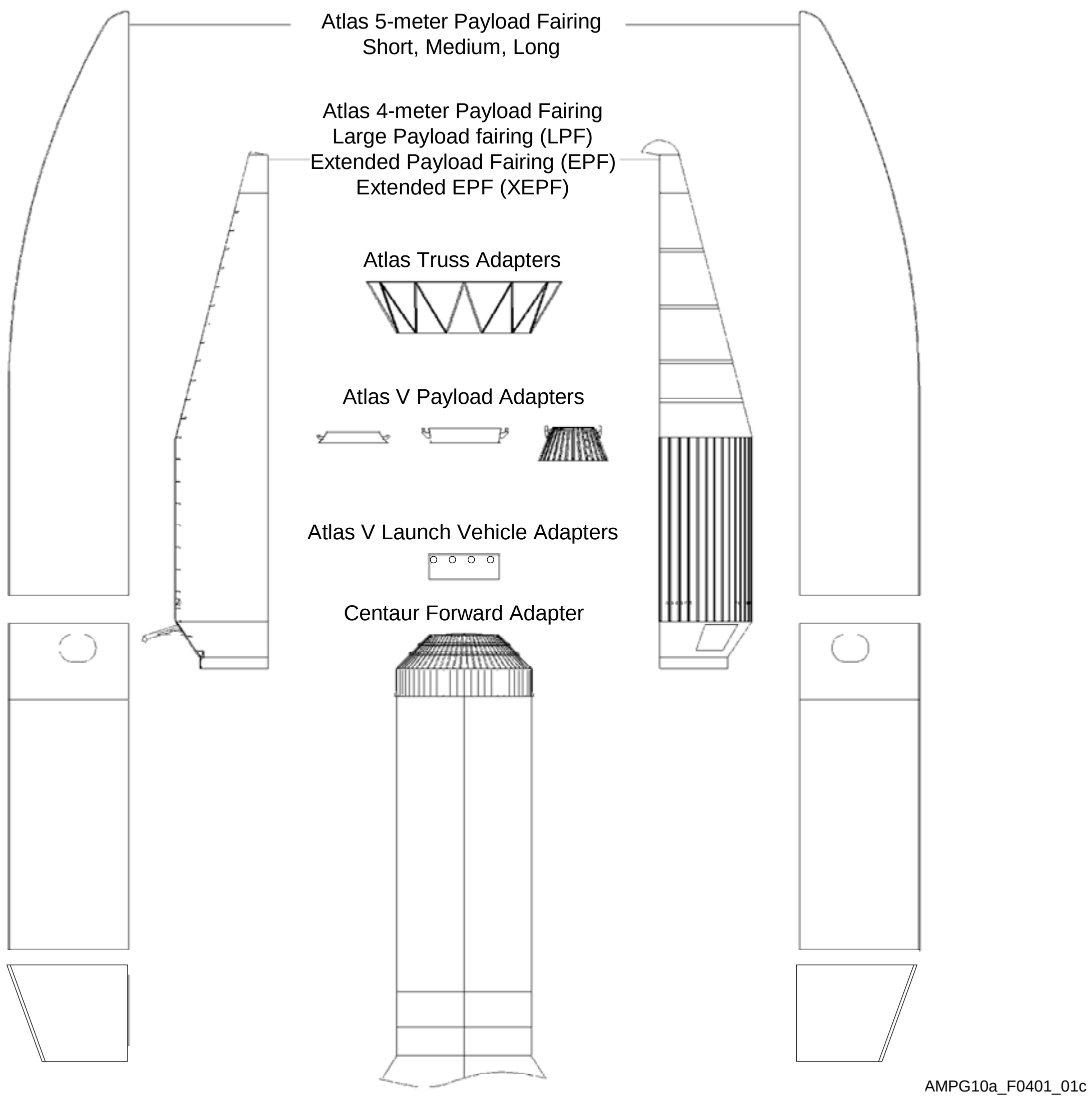
Lockheed Martin also suggests that the spacecraft contractor demonstrate the spacecraft capability to withstand thermal and EMI/EMC environments.

4. SPACECRAFT INTERFACES

4.1 SPACECRAFT-TO-LAUNCH VEHICLE INTERFACES

The Atlas V launch system is designed to meet requirements of currently defined spacecraft and offers the flexibility to adapt to mission-specific needs. Primary interfaces between the Atlas V launch vehicle and the spacecraft consist of a Payload Fairing (PLF) that encloses and protects the spacecraft and a Payload Adapter (PLA) that supports the spacecraft on top of the launch vehicle. These components are designed to provide mechanical and electrical interfaces required by the spacecraft and to provide a suitable environment during integration and launch activities. Payload adapters and payload fairings that are offered by the Atlas V program are shown in Figure 4.1-1.

Figure 4.1-1: Atlas V Payload Interface Options



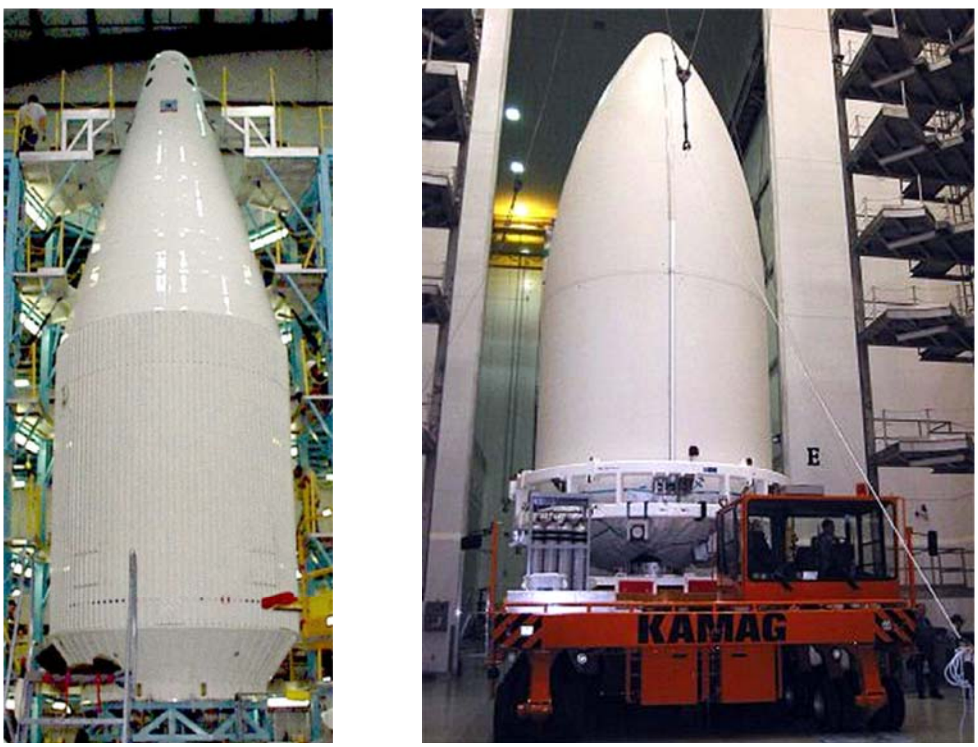
The following sections summarize these interface options. Detailed descriptions of payload fairings are included in Appendix D, and of payload adapters are included in Appendix E. The interface information in the following sections should be used only as a guideline. Modifications to these systems may be accommodated on a mission-specific basis.

Ultimate control of interface information for a given mission is governed through the Interface Control Document (ICD) developed and maintained during the mission integration process, as discussed in Section 5.

4.1.1 Atlas V Payload Fairings

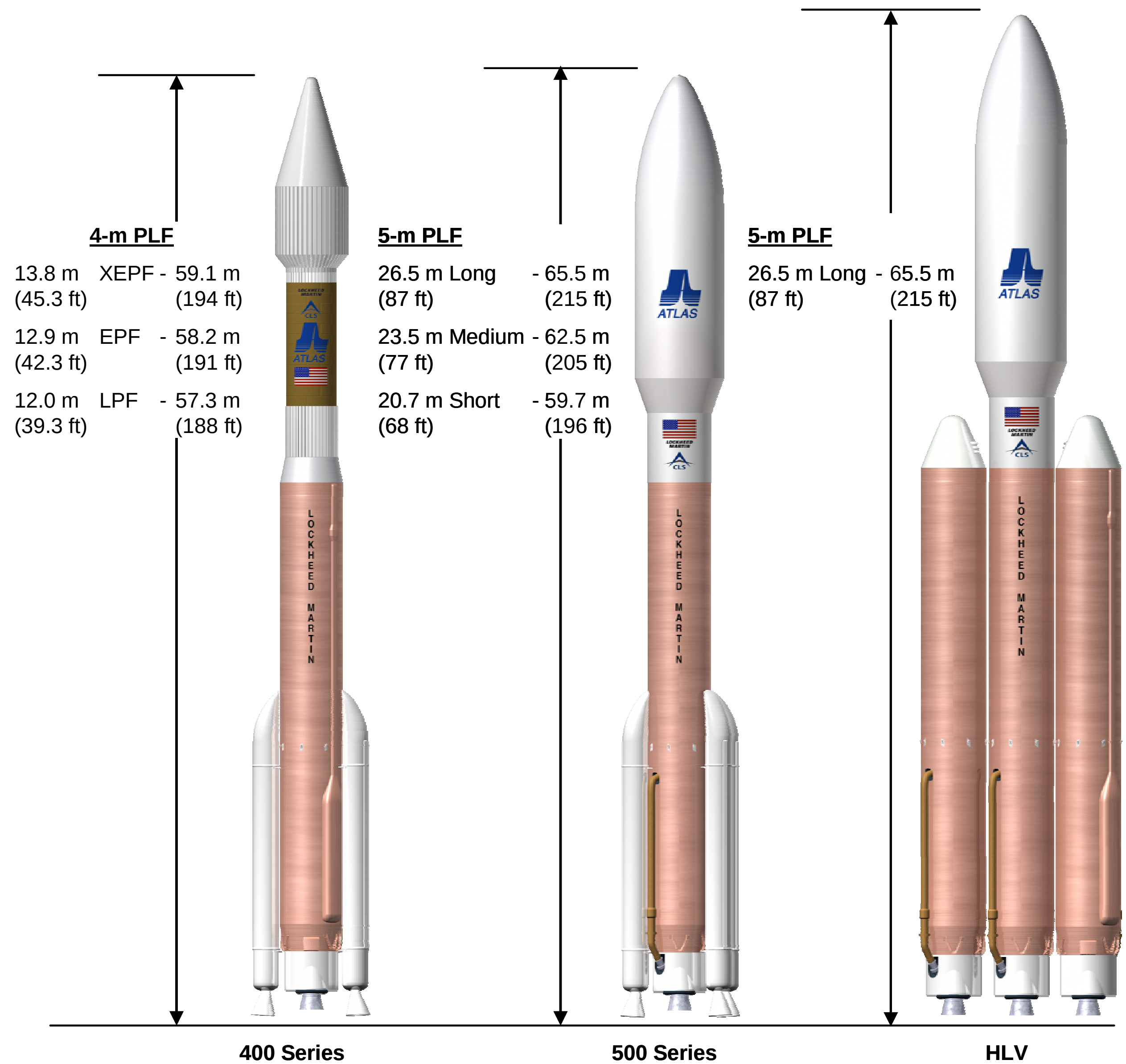
The PLF encloses and protects the spacecraft during ground operations and launch vehicle ascent. The PLF incorporates hardware to control thermal, acoustic, electromagnetic, and cleanliness environments for the spacecraft. The PLF has standard access door locations and may also be tailored to provide additional spacecraft access and Radio Frequency (RF) communications to the encapsulated spacecraft. The Atlas V user has a choice between the 4-m diameter (400 series) and 5-m diameter (500 series) PLF configurations as shown in Figure 4.1.1-1. Both the 4-m and 5-m payload fairings are available in different lengths to meet spacecraft mission requirements as shown in Figure 4.1.1-2.

Figure 4.1.1-1: Atlas V Payload Fairings



Atlas V 4-m EPF **Atlas V 5-m PLF** AMPG10a_F040101_01a

Figure 4.1.1-2: Atlas V Vehicle Lengths with Payload Fairing Options



AMPG10a_F040101_02d

The static payload envelope defines the usable volume for a spacecraft. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft and payload adapter interface. For clearances between the spacecraft and PLF, primary clearance concerns are for dynamic deflections of the spacecraft and PLF and the resulting relative loss of clearance between these components. For clearances between the spacecraft and PLA, primary envelope concerns are for access to the mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft. For all areas of the static payload envelope, clearance layouts and analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft and launch vehicle integration operations to ensure a positive clearance is maintained. Overall views of static payload envelopes for Atlas V 400 series launch vehicle configurations are shown in Figure 4.1.1-3 and for Atlas V 500 series launch vehicle configurations are shown in Figure 4.1.1-4. Detailed descriptions are included in Appendix D.

Figure 4.1.1-3: Atlas V 4-m Static Payload Envelope

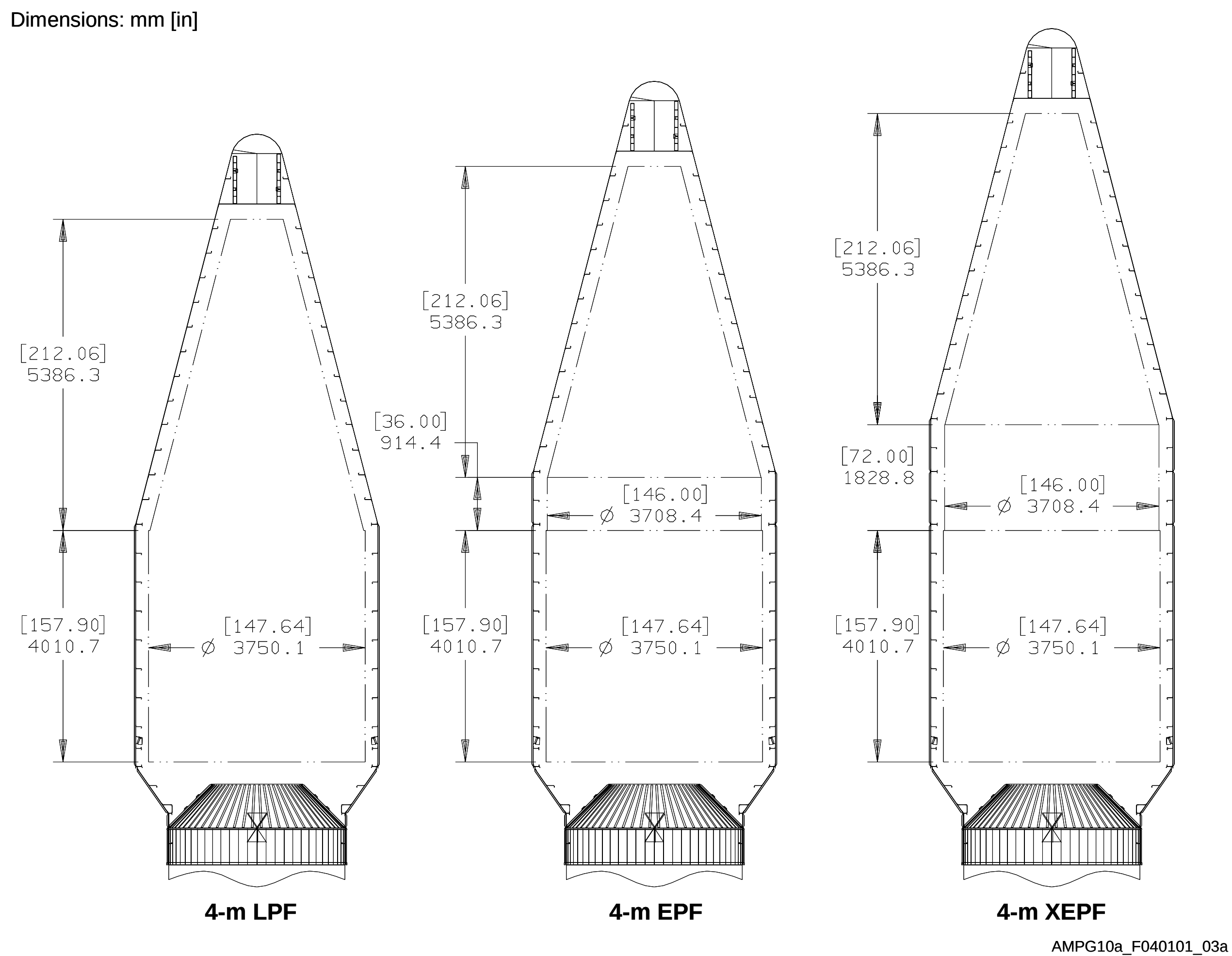
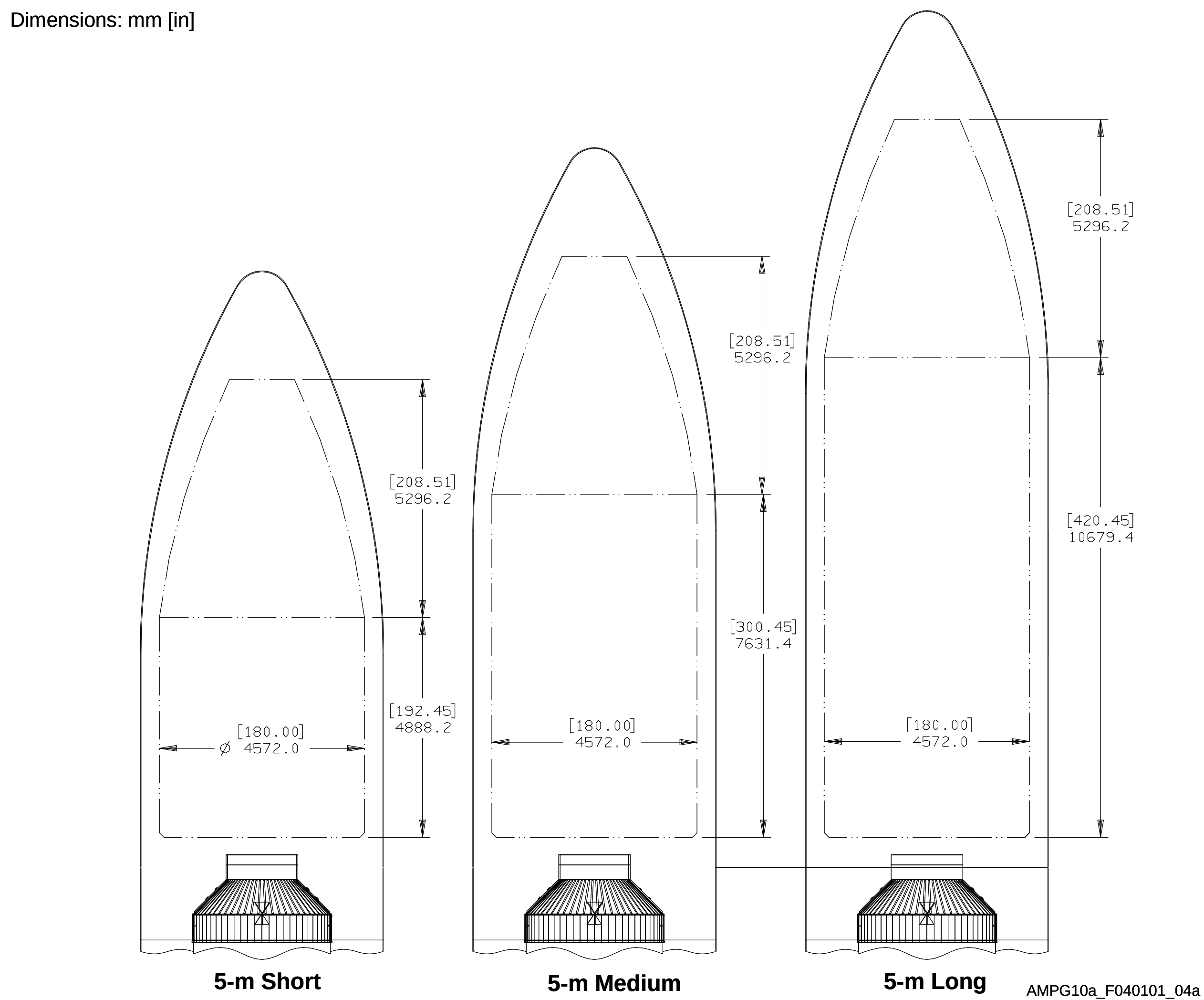


Figure 4.1.1-4: Atlas V 5-m Static Payload Envelope



4.1.1.1 Atlas V 4-m Payload Fairings

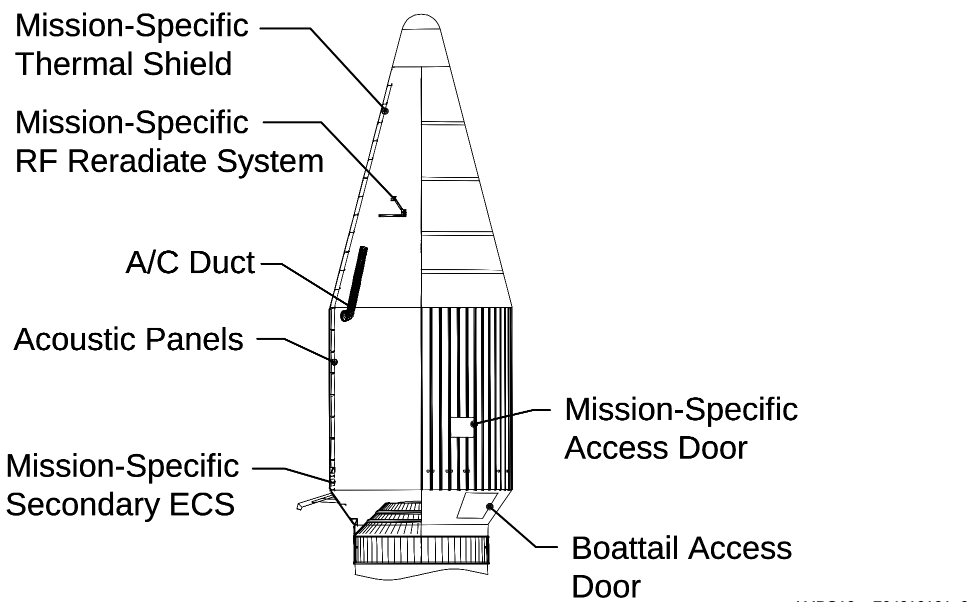
The Atlas V Large Payload Fairing (LPF), Extended Payload Fairing (EPF), and Extended EPF (XEPF) have a common 4-m diameter cylindrical section topped by a conical section. Major sections of these payload fairings are the boattail, the cylindrical section, and the nose cone that is topped by a spherical. The EPF was developed to support launches of larger volume spacecraft by adding a 0.9-m (36-in.) high cylindrical plug to the top of the cylindrical section of the LPF. The XEPF is a modified version of the EPF that incorporates an additional 0.9-m (36-in.) high cylindrical plug to further increase the available payload volume.

The PLF is designed to provide a controlled environment for spacecraft. For the Atlas V 400 series launch vehicles, acoustic panels are provided in the PLF to attenuate the sound pressure levels to acceptable limits. For thermal control, the external surface of the conical section fairing is insulated with cork to limit temperatures to acceptable values. Thermal shields may be added in the conical section of the PLF to provide additional thermal control. During prelaunch activities, conditioned air is provided through the air conditioning duct that is located in the cylindrical and nose cone portion of the PLF. Vent holes and housings are mounted on the lower part of the cylindrical section to allow air from the air conditioning system to exit the PLF and to allow depressurization during ascent. A secondary environmental control system may be added as a mission specific option to provide additional cooling or to direct cooling air to specific points on the spacecraft. The four large access doors in the boattail section of the 4-m PLF provide primary access to the Centaur forward adapter packages and the encapsulated spacecraft. Work platforms can be inserted through these access doors into the payload compartment to allow access to spacecraft hardware near the aft end of the payload compartment. If access to other portions of the spacecraft is required, additional doors can be provided on the cylindrical section of the PLF on a mission-specific basis.

A customer specified logo may be placed on the cylindrical section of the PLF. The Atlas V program will work with the customer and provide layouts of the logo on the launch vehicle to assist in determining the proper size and location for these logos.

Additional mission-specific items that can be mounted on the PLF to provide support services for the spacecraft include an RF reradiating antenna. The reradiating antenna allows spacecraft RF communications after the spacecraft is encapsulated inside the PLF. Standard and mission specific features of the PLF are shown in Figure 4.1.1.1-1. Detailed information on the Atlas V 4-m payload fairings may be found in Appendix D.

Figure 4.1.1.1-1: Atlas V 4-m Payload Fairing Interface Features



4.1.1.2 Atlas V 5-m Payload Fairings

The Atlas V 5-m short, medium and long payload fairings were developed along with the increased launch vehicle performance to accommodate evolving spacecraft requirements. The 5-m PLF is a bisector PLF with a composite structure made from sandwich panels with carbon fiber face sheets and a vented aluminum honeycomb core. There are two major components of the 5-m PLF. The lower section is the base module that encapsulates the Centaur. The upper section is the Common Payload Module (CPM) that encapsulates the spacecraft. The CPM consists of a cylindrical section which transitions into a constant radius ogive nose section topped by a spherical nose cap. For the 5-m medium PLF, a 2,743-mm (108-in.) lower payload module section is added to the base of the common payload module to increase the available payload volume. The PLF interfaces with the launch vehicle at the fixed conical boattail that is attached to the launch vehicle booster. Clearance losses for spacecraft are minimized by the Centaur Forward Load Reactor (CFLR) system that stabilizes the top of the Centaur and thereby reduces the relative motion between the PLF, Centaur, and spacecraft. PLF sections provide mounting provisions for various secondary systems.

Payload compartment cooling system provisions are in the ogive-shaped portion of the PLF. Electrical packages required for the PLF separation system are mounted on the internal surface of the PLF.

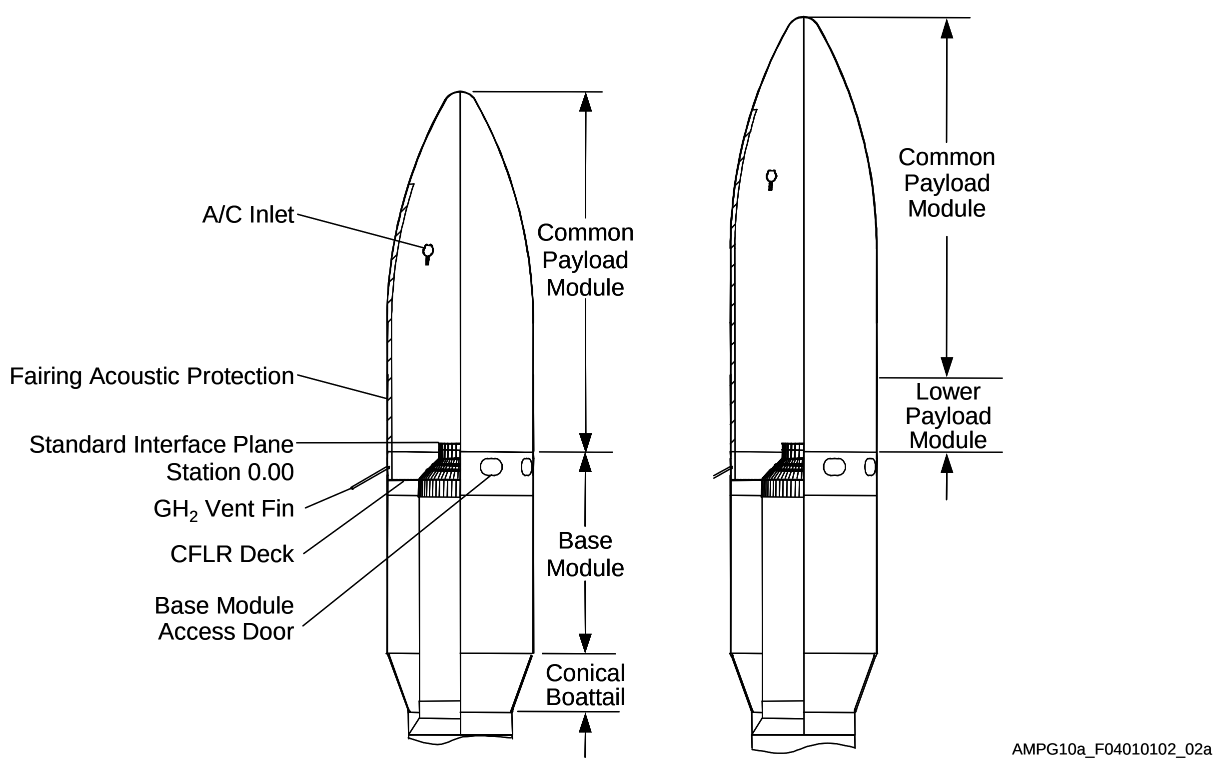
The PLF and boattail provide a protective enclosure for the spacecraft and Centaur during prelaunch operations and launch vehicle ascent. Fairing Acoustic Protection (FAP) is provided as a standard service to attenuate acoustic sound pressure levels to acceptable limits. For thermal control, external PLF surfaces are insulated with cork and painted white to limit temperatures to acceptable values. During prelaunch activities, conditioned air is provided through the air conditioning inlet that is located in the ogive section of the PLF. This inlet directs conditioned air to provide thermal and humidity control for the payload compartments and prevents direct impingement of this flow on the spacecraft. Vent ports and vent port assemblies are mounted in the mid-section of the base module for air from the air conditioning system to exit the PLF and to allow depressurization during launch vehicle ascent. A secondary environmental control system may be added as a mission specific option to provide additional cooling or to direct cooling air to specific points on the spacecraft.

The 5-m payload PLF has four large access doors in the base module portion of the PLF to provide primary access to the Centaur forward adapter. These doors also allow access to the spacecraft hardware near the aft end of the payload module. If access to other portions of the spacecraft is required, additional access doors can be provided on a mission-specific basis. The 5-m PLF interface features are shown in Figure 4.1.1.2-1.

A customer-specified logo may be placed on the cylindrical section of the PLF. Lockheed Martin will work with the customer and provide layouts of the logo on the launch vehicle to assist in determining the proper size and location for these logos.

Additional mission-specific items that can be mounted on the PLF to provide support services for the spacecraft include an RF reradiating antenna. The reradiating antenna allows spacecraft RF communications after the spacecraft is encapsulated inside the PLF. Detailed information on the Atlas V 5-m

Figure 4.1.1.2-1: Atlas V 5-m Payload Fairing Interface Features

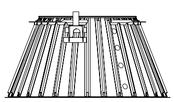
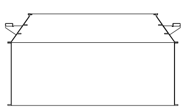
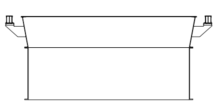


payload fairings may be found in Appendix D.

4.1.2 Mechanical Interface—Payload Adapters

The Atlas V offers a range of mechanical interface options including standardized bolted interfaces, payload adapters, and trusses that mate the spacecraft to the launch vehicle. These options include provisions for mechanical and electrical interfaces between the launch vehicle and spacecraft and for mission-specific services. Payload interface options currently offered by the Atlas V program include a bolted interface at the standard interface plane, Atlas V standard payload adapters that are designed to handle heavier spacecraft taking advantage of the higher performance Atlas V vehicle, and truss interfaces that allow the user to take advantage of the full volume and performance capabilities of the Atlas V 500 series. These spacecraft interface options are summarized in Figure 4.1.2-1.

Figure 4.1.2-1: Atlas V Payload Support Interface Options

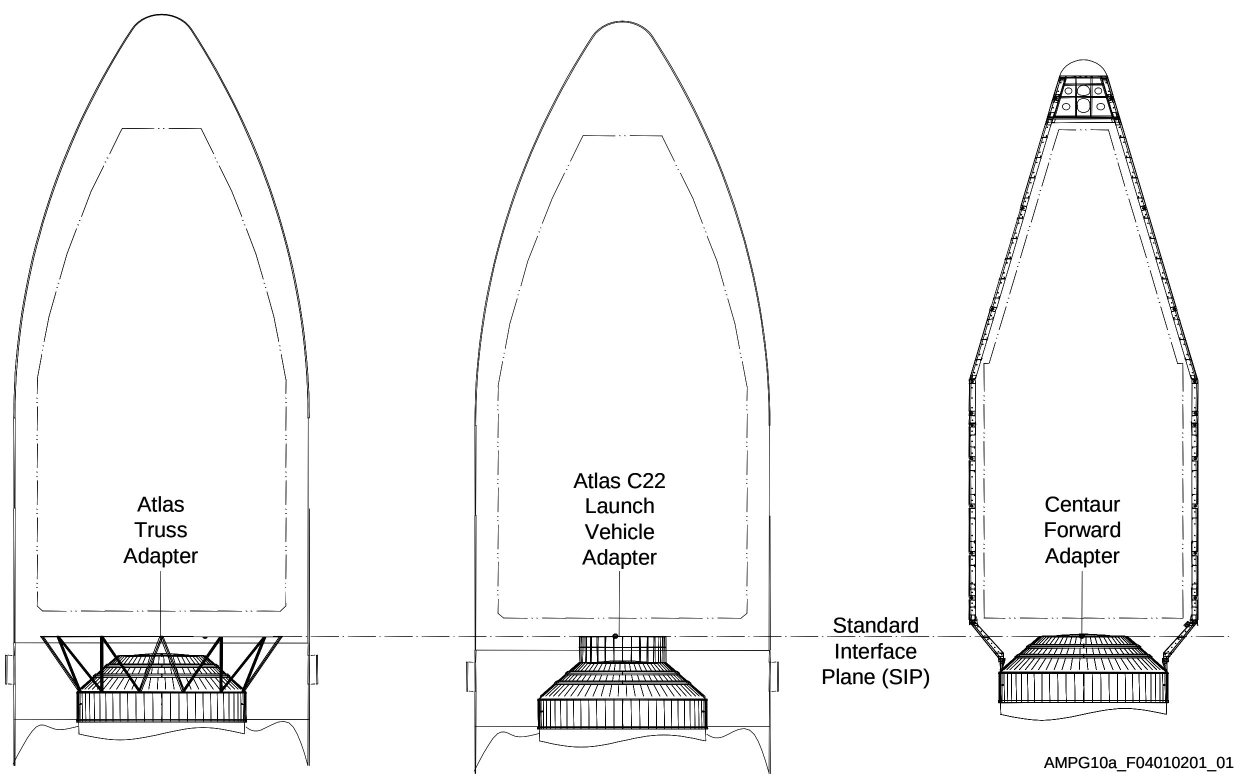
Standard Interface Plane		
SIP	Bolted Interface: 121 Fasteners Bolt Circle Diameter: 1,575.0 mm (62.010 in.)	Appendix E.1
Launch Vehicle Adapters		
C13 C15 C22 C29 Cxx - (Mission Specific)	Integrally Machined Aluminum Forging Bolted Interface: 120 or 121 Fasteners Bolt Circle Diameter: 1,575.0 mm (62.010 in.) Height: 330.2 mm to 736.6 mm (13 in. to 29 in.) Mass: See PSW, Section 2.5.1	Appendix E.2
Atlas V Standard Payload Adapters		
A 	Aluminum Skin/Stringer/Frame Construction Forward Ring Diameter: 945.3 mm (37.215 in.) Separation System: PSS37 Marmon Type Clampband Height: 762.0 mm (30.00 in.) Mass: See PSW, Section 2.5.1	Appendix E.3
B1194 	Integrally Machined Aluminum Forging Forward Ring Diameter: 1,215.0 mm (47.835 in.) Separation System: LSPSS1194 Low-Shock Marmon Type Clampband Height: 812.8 mm (32.00 in.) Standard 584.2 – 990.6 mm (23 – 39 in.) Available Mass: See PSW, Section 2.5.1	Appendix E.4
D1666 	Integrally Machined Aluminum Forging Separation System: LSPSS1666 Low-Shock Marmon-Type Clampband Forward Ring Diameter: 1,666.1 mm (65.594 in.) Height: 889.0 mm (35.00 in.) Standard 660.4 – 1066.8 mm (26 – 42 in.) Available Mass: See PSW, Section 2.5.1	Appendix E.5
Atlas V Truss Adapters		
T4394 (173 in.)	Bolted Interface: 18 Places Forward Ring Bolt Circle Diameter: 4,394.2 mm (173.0 in.) Height: 1,168.4 mm (46.00 in.)	Section 4.1.2.4
T3302 (130 in.)	Bolted Interface: 14 Places Forward Ring Bolt Circle Diameter: 3,302.0 mm (130.0 in.) Height: 914.4 mm (36.00 in.)	Section 4.1.2.4

4.1.2.1 Atlas Standard Interface Plane

The reference plane for the spacecraft to launch vehicle interface is the Atlas V Standard Interface Plane (SIP) as shown in Figure 4.1.2.1-1. For vehicles using a 4-m PLF, this plane is at the top of the Atlas V Centaur forward adapter. For vehicles using a 5-m PLF, this plane occurs at the top of an Atlas V program provided C22 launch vehicle adapter that is mounted on top of the Centaur forward adapter. The C22 adapter is standard with the 5-m PLF to allow launch vehicle ground support equipment interfaces and is designed to provide an interface that is identical to that provided by the Centaur forward adapter. The C22 adapter is considered to be part of the basic launch vehicle and does not count against the Payload Systems Weight (PSW). For vehicles using a 5-m PLF and a launch vehicle provided truss adapter, the C22 adapter is not used and the SIP plane occurs at the top of the truss.

For customers that provide their own adapter, the SIP is the interface point between the spacecraft provided hardware and the Atlas V launch vehicle. 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 torus arm fittings and an encapsulation diaphragm unless an Atlas V launch vehicle supplied intermediate adapter is used. Detailed information on the SIP and on interfaces to launch vehicle flight and Ground Support Equipment (GSE) hardware may be found in Appendix E.

Figure 4.1.2.1-1: Atlas V Standard Interface Plane



4.1.2.2 Atlas V Launch Vehicle Adapters

The Atlas V launch vehicle adapters were developed to provide a common interface for launch vehicle required ground support equipment that interfaces with payload adapter systems. The launch vehicle adapter is a machined aluminum component in the form of a monocoque cylinder and is available in heights from 330.2 mm to 736.6 mm (13.00 in. to 29.00 in.). Table 4.1.2.2-1 lists the standard configuration available for the Atlas V C13, C15, C22 (shown in Figure 4.1.2.2-1) and C29 launch vehicle adapter.

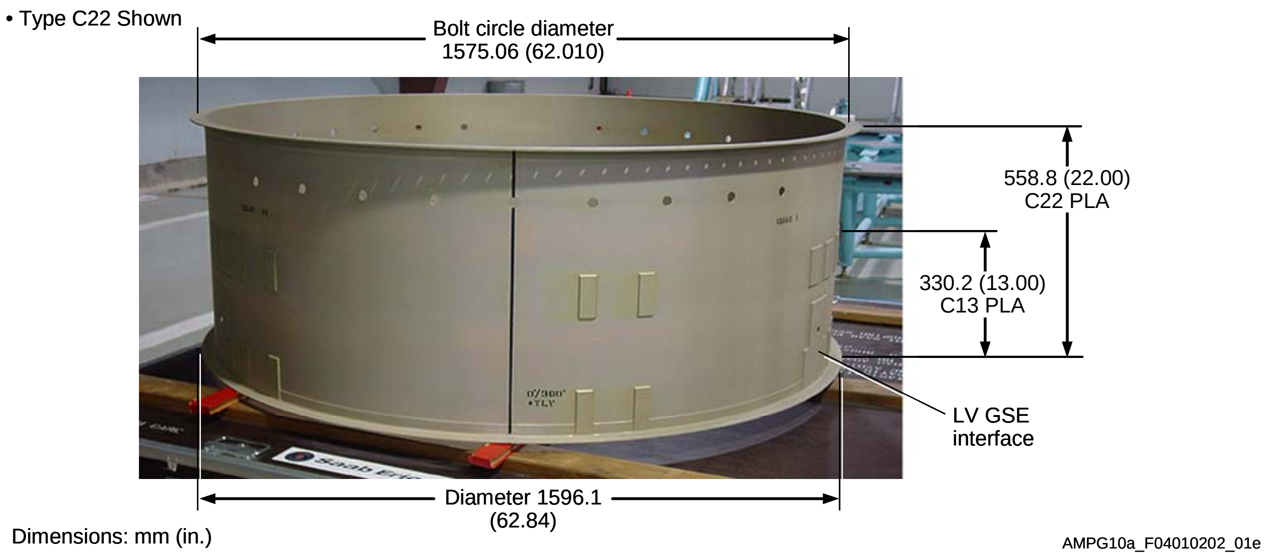
Table 4.1.2.2-1 Atlas V Launch Vehicle Adapters

Launch Vehicle Adapter	Standard Height
C13	330.2-mm (13.00-in.)
C15	384.8-mm (15.15-in.)
C22	558.8-mm (22.00-in.)
C29	736.6-mm (29.00-in.)

On the Atlas V 500 series launch vehicle, the C22 launch vehicle adapter is mounted to the top of the Atlas V Centaur forward adapter and provides an interface surface and bolt hole pattern at its forward end that is compatible with SIP requirements. The C22 adapter is standard with the 5-m PLF to allow clearance for launch vehicle ground support equipment. In this configuration, cost and performance impacts of the C22 launch vehicle adapter are considered to be a part of the basic launch vehicle service and are not counted against PSW.

For customers that provide their own payload adapter and payload separation system, launch vehicle adapters are available as a mission specific option. This allows the customer to raise the position of the SIP relative to the launch vehicle for additional clearance or to take advantage of standard GSE interfaces that are built into the launch vehicle adapter.

Figure 4.1.2.2-1: Atlas V Launch Vehicle Adapter Configuration



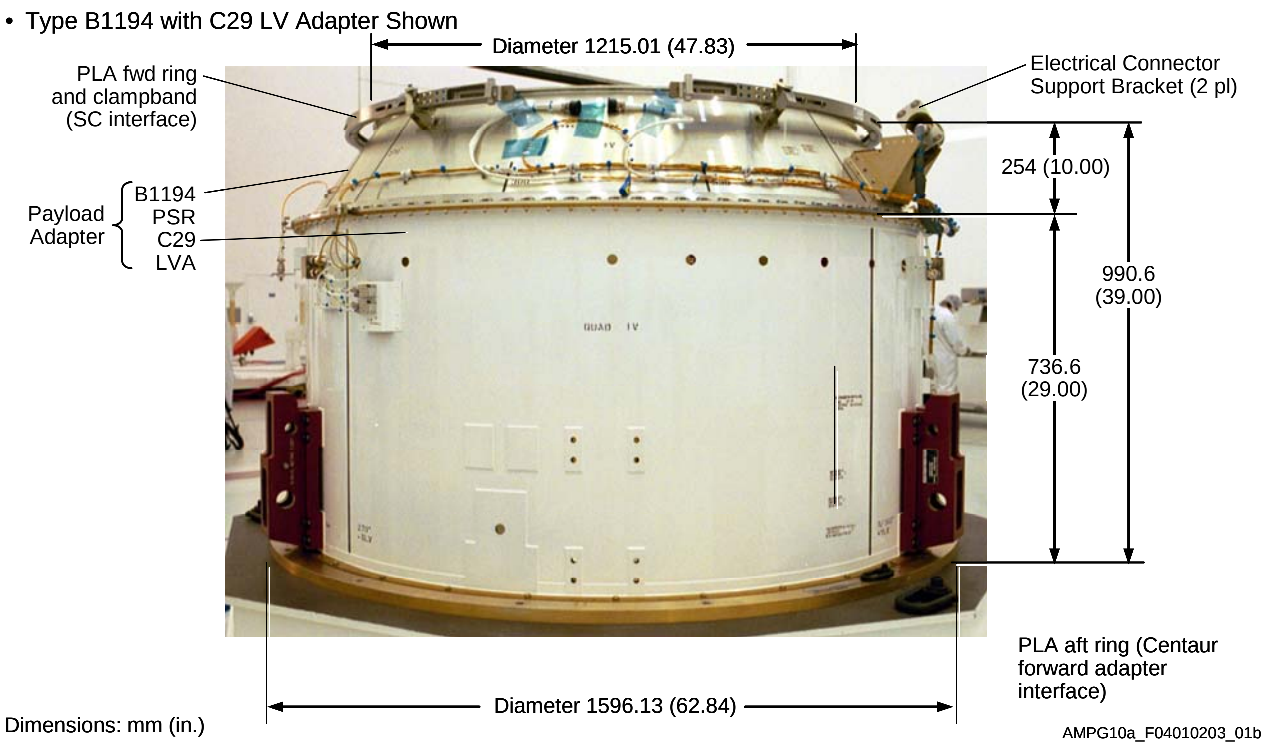
4.1.2.3 Atlas V Standard Payload Adapters

The Atlas V standard payload adapters have been designed to handle heavier spacecraft that can take advantage of the higher performance of the Atlas V vehicle. The available systems include the Atlas V Type A, B1194, and D1666 payload adapters.

These payload adapters consist of two major sections, the Payload Separation Ring (PSR) and the launch vehicle adapter shown in Figure 4.1.2.3-1. The PSR is a machined aluminum component in the form of a truncated cone. The forward ring forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 evenly spaced bolt holes that allow it to be joined to the launch vehicle adapter. This symmetrical bolt hole pattern allows the payload separation ring and attached spacecraft to be rotated relative to the launch vehicle in 3-degree increments to meet mission specific requirements. The PSR supports all hardware that directly interfaces with spacecraft including the payload separation system, electrical connectors, and mission-specific options.

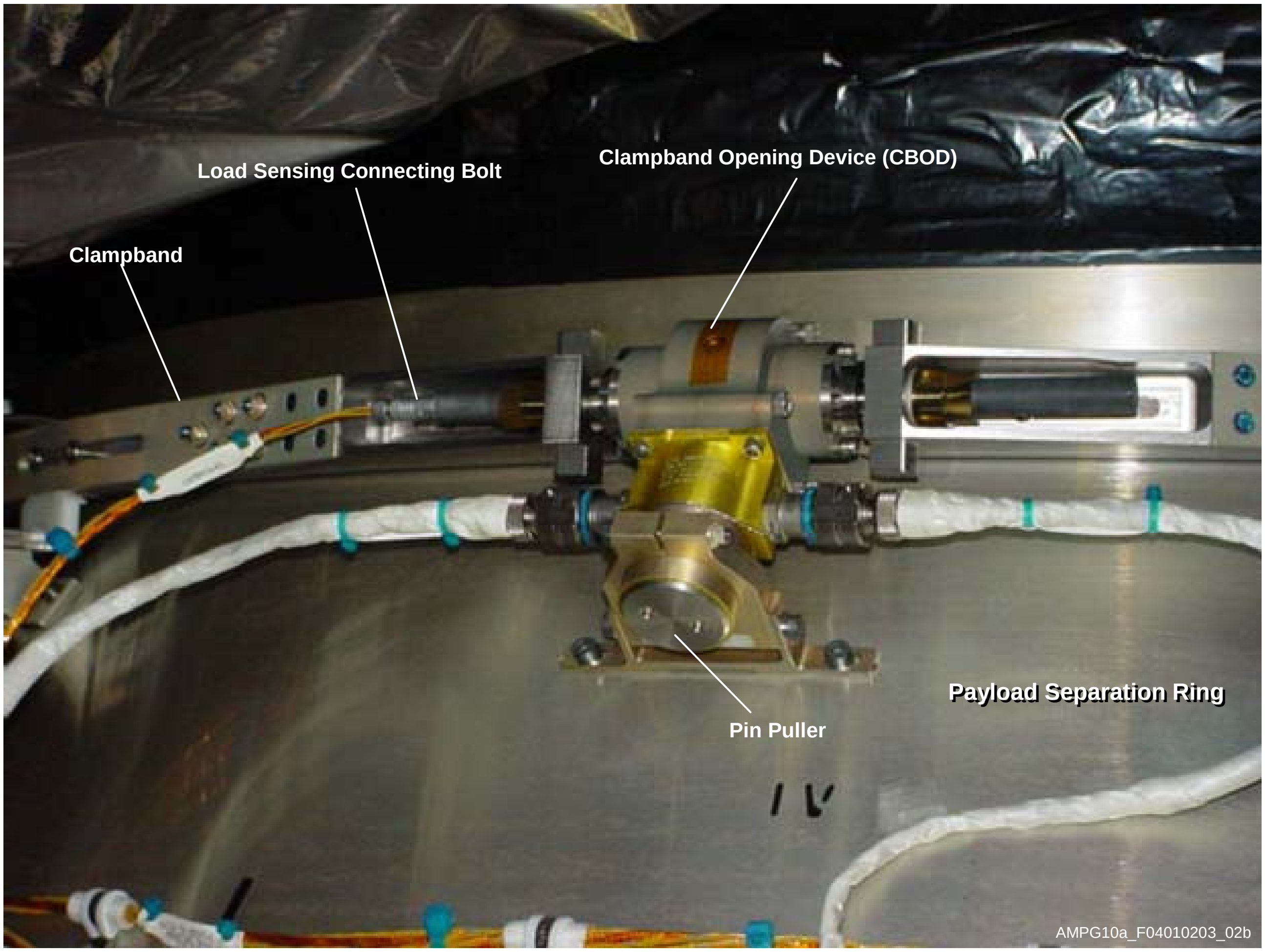
The launch vehicle adapter is a machined aluminum component in the form of a monocoque cylinder. The forward ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 bolt holes spaced evenly every 3 degrees that allow it to be joined to the payload separation ring. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 bolt holes that match up with Atlas V SIP requirements. The nominal height of the launch vehicle adapter is 558.8 mm (22.00 in.) but this height may be varied from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.) to meet mission-specific requirements. The launch vehicle adapter includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Figure 4.1.2.3-1: Atlas V Standard Payload Adapter Configuration – PLA with LV Adapter



The Atlas Type B1194 and D1666 payload adapters use a launch vehicle provided low-shock Marmon-type clampband payload separation system. This separation system shown in Figure 4.1.2.3-2 consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to catch and retain the clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a single-piece aluminum retaining band that holds the clamp segments in place. The low-shock Clampband Opening Device (CBOD) holds the ends of the retaining band together. The CBOD includes release bolts that engage the ends of the clampband. These release bolts are threaded into a flywheel mechanism. During installation and flight, the flywheel is restrained against rotation by a restraining pin. For separation, a pyrotechnically activated pin-puller retracts this pin from the flywheel, allowing it to rotate and eject the release bolts. This separation system reduces shock compared to a conventional bolt-cutter system and is resettable, allowing the actual flight hardware to be tested during component qualification and acceptance testing. Detailed information on each Atlas V standard payload adapter is in Appendix E.

Figure 4.1.2.3-2: Atlas V Low-Shock Payload Separation System Configuration



4.1.2.4 Atlas V Truss Adapters

T4394 Truss Adapter — A payload truss with a forward spacecraft interface of 4,394-mm (173-in.) diameter has been developed to meet future spacecraft interface requirements. The 1,168-mm (46-in.) high truss attaches at 12 locations to the Centaur forward adapter. The forward ring of the payload truss has 18 equally spaced mounting points for the spacecraft, but could be easily modified to accommodate alternate mounting configurations. The truss has 24 graphite-epoxy struts with titanium alloy end fittings, aluminum alloy forward and aft brackets, and an aluminum alloy forward ring. When this truss is used, the static payload envelope extends to the top surface of the payload support truss forward ring.

T3302 Truss Adapter — A payload truss with a forward spacecraft interface of 3302-mm (130-in) diameter is under development to meet future spacecraft interface requirements. The 914.4-mm (36-in) high truss attaches at 12 locations to the Centaur forward adapter. The forward ring of the payload truss has 14 equally spaced mounting points for the spacecraft, but could be easily modified to accommodate alternate mounting configurations. The truss has 24 graphite-epoxy struts with titanium alloy end fittings, aluminum alloy forward and aft brackets, and an aluminum alloy forward ring. When this truss is used, the static payload envelope extends to the top surface of the payload support truss forward ring.

4.1.3 Electrical Interfaces

Spacecraft and launch vehicle electrical interfaces are shown in Figures 4.1.3-1 and 4.1.3-2. Standard interfaces include:

1. A spacecraft dedicated umbilical interface between the umbilical disconnect located on the Centaur forward adapter and electrical In-Flight Disconnects (IFD) at the spacecraft and launch vehicle interface;

2. Spacecraft and launch vehicle separation indicators (continuity loop wiring) located in the spacecraft and launch vehicle IFDs to verify separation;

3. Standard IFDs or other customer supplied connectors that may be required by mission-peculiar requirements. For standard connectors, the following part numbers (or equivalent substitutes) apply:

   — 37 Contact MS3446E37-50P (Launch Vehicle) MS3464E37-50S (Spacecraft)

   — 61 Contact MS3446E61-50P (Launch Vehicle) MS3464E61-50S (Spacecraft)

The Atlas V launch vehicle can also be configured to provide electrical interfaces for various mission-peculiar requirements. The following paragraphs describe the Atlas V electrical interfaces in more detail.

Figure 4.1.3-1: Typical Spacecraft/Launch Vehicle Electrical Interface

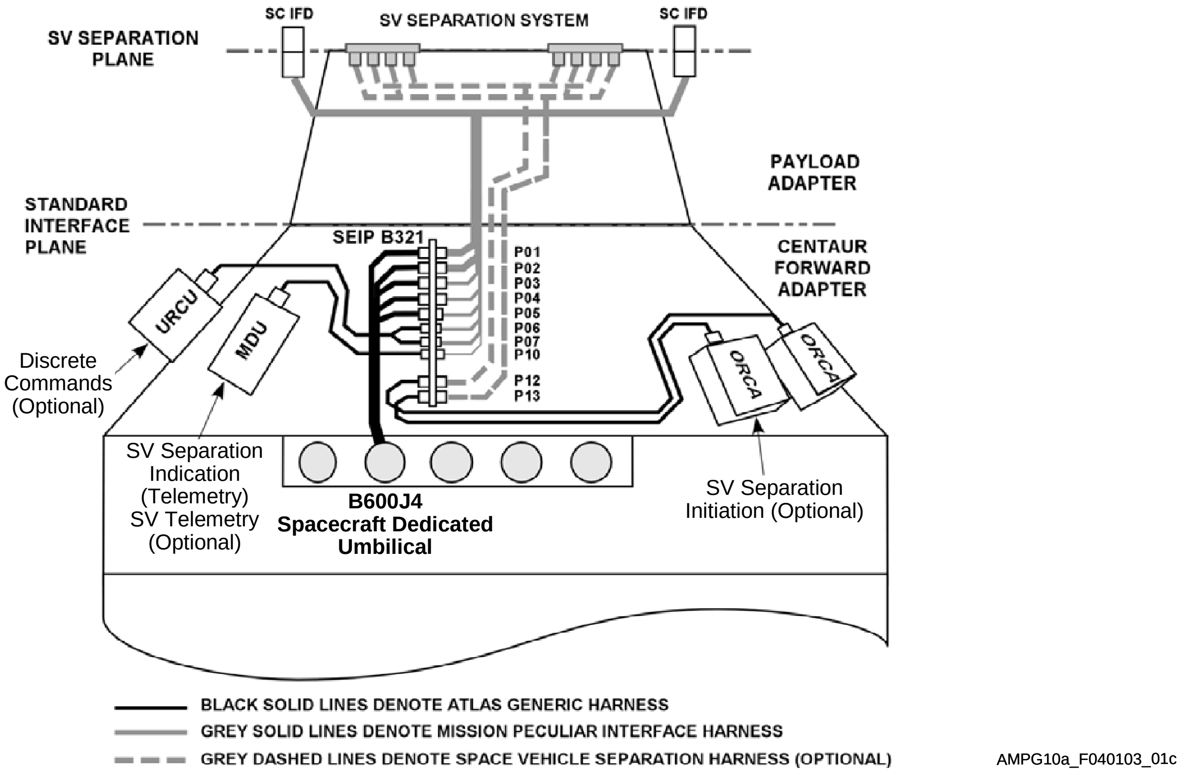
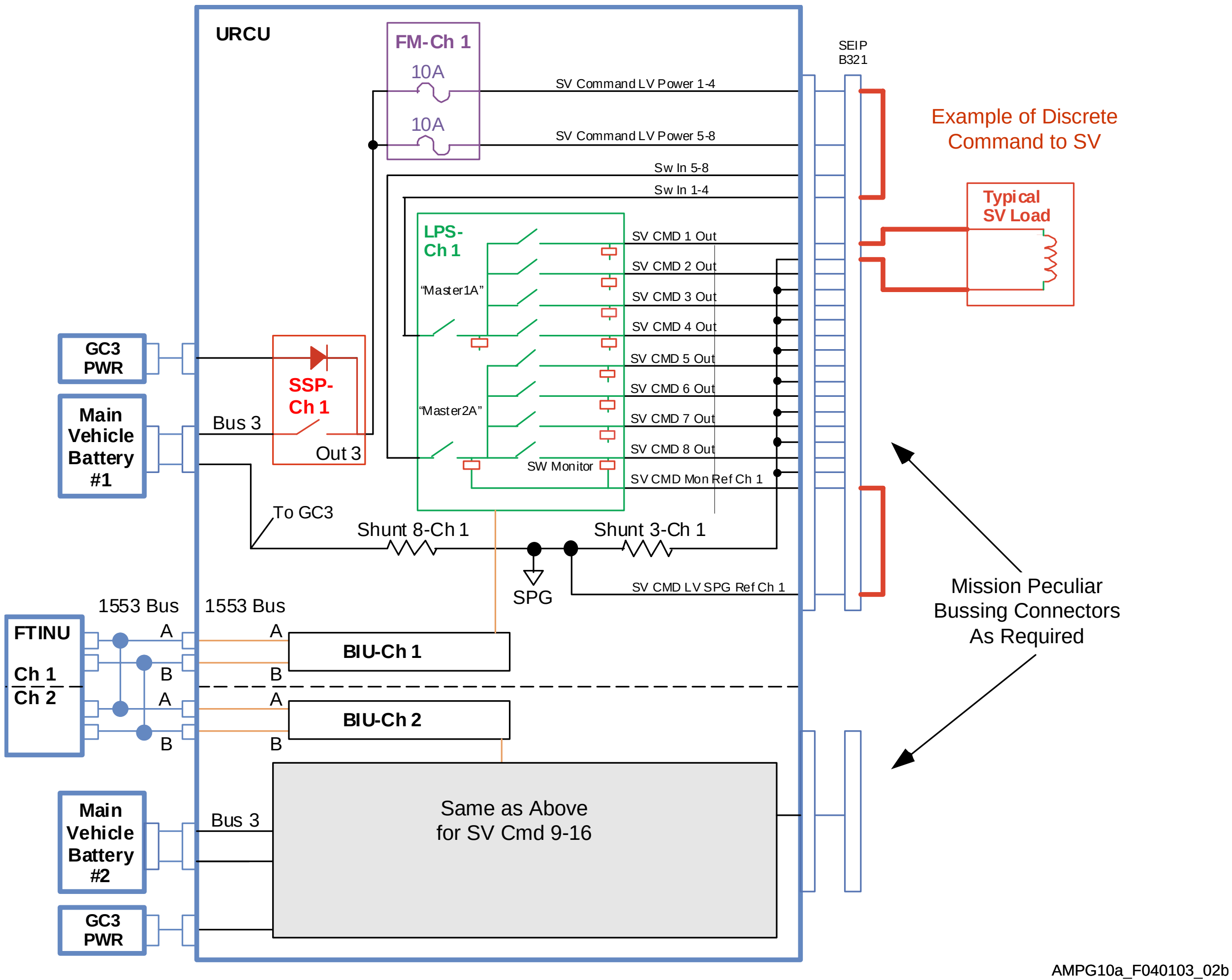


Figure 4.1.3-2: Spacecraft Switch Interface



4.1.3.1 Umbilical Spacecraft-to-GSE Interface

One spacecraft dedicated umbilical disconnect for on-pad operations is located on the Centaur forward umbilical panel. This umbilical interface provides signal paths between the spacecraft and GSE for spacecraft system monitoring during prelaunch and launch countdown operations. The umbilical disconnect separates at liftoff from the Centaur receptacle. The spacecraft in-flight disconnect connectors disengage during separation of the spacecraft from the Centaur. The following are the complement of spacecraft umbilical wires for the Atlas V vehicles.

1. Twelve — 12 AWG twisted wire pairs

2. Sixty — 20 AWG twisted shielded wire pairs

3. Four — 20 AWG twisted shielded wire triples

4. Six — 78 ohms twisted shielded controlled impedance wire pairs

4.1.3.2 Electrical In-Flight Disconnects

Standard payload adapters provide two in-flight disconnect options of either 37 pins or 61 pins for the spacecraft interface. These IFDs typically provide a spacecraft dedicated umbilical interface between the spacecraft and GSE, and any launch vehicle command and monitoring required to support the spacecraft during ascent. The Atlas V program can provide more IFDs on a mission-peculiar basis if the spacecraft requires them. Mechanical interface requirements for the IFDs are shown in Appendix E.

4.1.3.3 Spacecraft Separation System

The baseline separation system for spacecraft and Centaur separation is a pyrotechnic V-type clampband system. The separation sequence is initiated by redundant commands from the Centaur guidance system. Positive spacecraft separation is detected by continuity loops installed in the spacecraft IFDs that are wired to the Centaur instrumentation system. The separation event is telemetered to the ground.

4.1.3.4 Control Command Interface

For the Atlas V launch vehicle configured with the current avionics, the Centaur Remote Control Unit (CRCU) provides as many as 16 control commands to the spacecraft. These circuits are solid-state switches configured as either launch vehicle 28 Vdc discrete commands or switch closure (dry-loop) commands depending on spacecraft requirements. The Inertial Navigation Unit (INU) controls the switches. Digital data from the INU are decoded in the CRCU, and addressed switches are energized or de-energized under INU software control. Command feedback provisions are also incorporated to ensure that control commands issued to the spacecraft are received through spacecraft and launch vehicle IFDs. The spacecraft is responsible for providing a feedback loop on the spacecraft side of the interface, including fault isolation circuits.

For the Atlas V launch vehicle configured with the fault tolerant avionics, the Upperstage Remote Control Unit (URCU) provides as many as 16 control commands (8 primary and 8 secondary) to the spacecraft as shown in Figure 4.1.3-2. The URCU channelization architecture allows each group of 8 commands to be powered by an independent battery. In addition, there is a series inhibit (“master switch”) per each group of 4 command switches that provide protection against a single “stuck ON” switch failure issuing a premature command. Each group of 8 commands, along with 2 master switches, can be independently configured as a 28 Vdc (nominal) discrete from the launch vehicle, or as a switch closure (dry-loop) function (using SV provided power) depending on spacecraft requirements. One of the two command types, discrete or closure, must be selected within a channel, however, each channel can be configured independently. Only one voltage can be applied to each group/channel (e.g. 28 Vdc from the launch vehicle to the spacecraft is fed to all 8 switches, independent voltage to each switch cannot be accommodated).

The following are the standard electrical switching capabilities:

1. The maximum allowable current through each master switch is 6 amps;

2. Each individual command switch is rated at 4 amps maximum; however, the total current from four simultaneous ON commands cannot exceed the rating of the corresponding master switch;

3. The allowable voltage range which can be switched is 22-33 Vdc;

4. The system has been optimized for launch vehicle discrete commands at 0.5 amps each or for switch closure commands at 1.0 amps each. Other mission unique requirements can be accommodated;

5. Commands can be controlled in 20 msec timing increments, individually or simultaneously per mission requirements.

Mission-peculiar compatibility analyses are performed for any interfaces that use these commands in order to verify proper circuit interaction and appropriate circuit electrical deratings.

4.1.3.5 Spacecraft Environment Instrumentation

The Atlas V program offers a suite of instrumentation options to capture spacecraft environments.

4.1.3.5.1 Spacecraft Telemetry Interface

A mission satisfaction telemetry kit is provided as an option by the launch vehicle to ascertain vehicle environments and ensure ICD requirements are met. Low frequency data (less than 200 Hz) can be captured through the launch vehicle Data Acquisition System (DAS) located on the Centaur. The Digital Telepak and Telemetry and Data Relay Satellite System (TDRSS) transmitter combination can provide up to 32 channels of wideband data (up to 8-kHz data) with a Pulse-Code Modulation (PCM) (Quadrature Phase-Shift Keying [QPSK]) telemetry format. Additional instrumentation may be added optionally up to the bandwidth and interface capabilities of these data systems. Data recorded from these measurements are provided, postlaunch, to evaluate compliance with ICD environmental requirements. The available mission satisfaction instrumentation consists of: (1) two acoustic measurements located inside the PLF, (2) three acceleration measurements at the spacecraft and launch vehicle separation plane spaced 120 degrees apart, (3) three orthogonal vibration measurements at the spacecraft and vehicle separation plane, (4) three orthogonal shock measurements at the spacecraft and launch vehicle separation plane, (5) one absolute pressure measurement located in the spacecraft compartment, and (6) one temperature measurement located in the spacecraft compartment.

4.1.3.5.2 Spacecraft Telemetry Options

The Atlas V program offers two options for transmission of spacecraft data.

4.1.3.5.2.1 RF Reradiation GSE Interface

A modification of the Atlas V PLF is made to accommodate an RF reradiating antenna system to reradiate spacecraft RF telemetry and command signals from inside the PLF to a GSE site before launch.

4.1.3.5.2.2 Spacecraft Serial Data Interface

The Atlas V launch vehicle can provide transmission of two spacecraft serial data interfaces. Spacecraft data are interleaved with launch vehicle DAS data and serially transmitted in the PCM bit stream. For each data interface, the spacecraft provides Nonreturn-to-Zero Level (NRZ-L) coded data and a clock from dedicated drivers (as an input to the launch vehicle). Spacecraft data and clock signals must be compliant with Electronics Industry Association RS-422, Electrical Characteristics of Balanced Voltage Digital Interface Circuits, with a maximum data bit rate of 2 kbps. Spacecraft data are sampled by the launch vehicle on the falling edge of the spacecraft clock signal. The clock-to-data skew must be less than 50 microseconds and the signal and clock duty cycles must be 50% ±5%. Cabling from the signal driver to the launch vehicle has a nominal characteristic impedance of 78 ohms. Data are presented as the original NRZ-L data stream in real time for those portions of prelaunch operations and flight for which Atlas V data are received. For postflight analysis, spacecraft data can be recorded to digital media.

4.1.3.6 Spacecraft Destruct Option

If required for Range Safety considerations, Atlas V can provide a spacecraft destruct capability. A safe and arm initiator receives the destruct command from the Centaur Flight Termination System (FTS). The initiator ignites electrically initiated detonators, which set off a charge on the booster. The charge ignites a mild detonating fuse that 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 Launch Service Building at SLC-3E, and the payload van at LC-41) for installation of a spacecraft ground control console(s). This console(s) is provided by the user, and interfaces with Lockheed Martin provided control circuits through the dedicated umbilical to the spacecraft. Control circuits provided for spacecraft use are isolated physically and electrically from those of the launch vehicle to minimize Eectromagnetic Interference (EMI) effects. Spacecraft that require a safe and arm function for apogee motors will also interface with the range operated pad safety console. The Atlas V program will provide cabling between the spacecraft console and the pad safety console. The safe and 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 it is safe to arm the system.

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 Uninterruptible Power System (UPS). The dual UPS consists of battery chargers, batteries, and a static inverter. UPS power is available for spacecraft use in the launch service building, umbilical tower, and payload van. Twenty-eight Vdc power can be provided for spacecraft use in the launch service building. 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 requirements restricting ozone-depleting chemicals.

Gaseous Nitrogen (GN_2)—Three pressure levels of GN_2 are available on the service tower or Vertical Integration Facility (VIF) for spacecraft use. Nominal pressure settings are 13,790 kN/m^2 (2,000 psi), 689.5 kN/m^2 (100 psi), and approximately 68.95 kN/m^2 (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 service towers at both Cape Canaveral Air Force Station (CCAFS) and Vandenberg Air Force Base (VAFB). At CCAFS, GHe at 15,169 kN/m^2 (2,200 psi) is available. At VAFB SLC-3E, GHe at 589.5 kN/m^2 (100 psig) and 34,475 kN/m^2 (5,000 psig) is available.

Liquid Nitrogen (LN_2)—LN_2 is available at the CCAFS launch complex storage facility. LN_2 is used primarily by the Atlas V pneumatic and LN_2 cooling systems. Small Dewars can be filled at Range facilities and brought to the Atlas V 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/m^2 (100 psig); 2,758 kN/m^2 (400 psig); 24,822 kN/m^2 (3,600 psig); and 34,475 kN/m^2 (5,000 psig).

Gaseous Xenon (Xe)—Gaseous xenon is available for use at CCAFS and VAFB. At both locations, xenon can be provided at various pressures and quantities subject to the user’s specific requirements.

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 (e.g., helium, nitrogen, xenon, and breathing air) and liquids (e.g., 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 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 fixed work decks do not suffice. Access can be provided inside the encapsulated nose fairing. Access requirements will be developed during the planning stage of each mission.

4.3 RANGE AND SYSTEM SAFETY INTERFACES

4.3.1 Requirements and Applicability

To launch from either CCAFS or VAFB, launch vehicle and spacecraft design and ground operations must comply with applicable launch site Range Safety regulations, U.S. Air Force (USAF) requirements concerning explosives safety, and U.S. consensus safety standards. In addition, when using spacecraft processing facilities operated by Spaceport Systems International, Astrotech International Corporation, the National Aeronautics and Space Administration (NASA), or the USAF, compliance with applicable facility safety policies is also required.

CCAFS and VAFB Range Safety organizations have regularly updated their safety requirements documents. Effective 1 July 2004, the single safety document for both CCAFS and VAFB (Eastern/Western Range Regulation [EWR] 127-1) was replaced by AFSPC 91-710. Existing launch vehicle and spacecraft programs are not affected by the new 91-710 regulations. New programs introduced after 1 July 2004 will negotiate applicable regulations with the Range Safety Office (45th Space Wing for Eastern Range and 30th Space Wing for the Western Range). Earlier versions of Range regulations may still apply to a given spacecraft or mission, depending on when the spacecraft bus was originally designed and constructed and approved by the Eastern and/or Western Range Safety organizations. Earlier versions of Range Safety regulations 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, 31 October 1997, and 31 December 1999.

Applicable safety compliance documents are determined during negotiations with the Range, Lockheed Martin, and the spacecraft contractor at the outset of the mission integration process. Other safety documents that may also apply to the launch site safety interface are:

1. Radiation Protection Program, 45 Space Wing Instruction 40-201
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. NASA KHB 1710.2, KSC Safety Practices Handbook (for spacecraft processed at KSC facilities)

7. Atlas Launch Site Safety Manual

At the start of the safety integration process (Section 4.3.2), Range Safety documents applicable to spacecraft design and ground processing operations will be determined. Safety requirements applicable to new designs and ground processing operations will be reviewed and tailored for specific spacecraft and mission applications.

Lockheed Martin System Safety engineers will evaluate mission-specific spacecraft designs and ground processing operations and provide guidance for successful completion of the Range review and approval process. Should areas of noncompliance be identified, Lockheed Martin will evaluate each area and provide guidance for resolution of specific noncompliance items while still meeting the intent of the applicable safety requirements. For commercial programs, Lockheed Martin will act as the spacecraft contractor’s liaison for interface activities with the launch site Range Safety Office.

EWR 127-1 Range Safety regulations require three inhibits (dual-fault tolerance) if system failures could result in catastrophic events and two inhibits (single-fault tolerance) if failures could result in critical events. Critical and catastrophic events are defined in EWR 127-1. The Range typically applies the three-inhibit requirement to safety critical electrical systems, including the spacecraft’s Category A ordnance circuits, during ground processing operations at the launch site (e.g., encapsulation of spacecraft at the processing facility, transport of encapsulated spacecraft to LC-41, mate of encapsulated assembly to the launch vehicle). During final ground processing of Atlas V payloads, the integrated spacecraft and launch vehicle stack (in launch configuration) will be transported from the VIF to the LC-41 pad. When Category A spacecraft circuits use the Atlas V launch vehicle’s ordnance controller during transport from VIF to pad, the three-inhibit requirement is satisfied. If the spacecraft’s Category A ordnance circuits will be independent from the launch vehicle, spacecraft customers should review their bus designs and ground operations plans and notify the Atlas V program if spacecraft systems do not provide the required fault tolerance. As stated above, Lockheed Martin will then assess mission-specific designs, evaluate hazard controls, and work with the Range to develop and implement “meets-intent” resolutions.

For each spacecraft and mission, compliance with applicable Range regulations (as tailored) will be addressed in the mission specific safety submittals defined in Section 4.3.2.

4.3.2 Safety Integration Process

The process used by Lockheed Martin to facilitate Range and System Safety coordination and receive Range approval and/or permission to launch is shown in Figure 4.3.2-1. This figure identifies responsibilities of the spacecraft contractor, Lockheed Martin, and the Range. Timelines identified in this process are typical and may vary to accommodate mission specific requirements.

For each mission integration effort, Lockheed Martin will provide qualified engineers to assist the spacecraft contractor during the Range review and approval process. Lockheed Martin obtains all Range safety and system safety approvals. The following paragraphs summarize our safety integration process and define safety data to be developed by the spacecraft customer during implementation of this process. Refer to Appendix C, Section C.3.6, for additional information on spacecraft data requirements.

Mission Orientation — Soon after contract award, Lockheed Martin and the spacecraft contractor will introduce a new system or mission to the Range during a mission orientation meeting at the Range Safety Office. Figure 4.3.2-1. Block A, shows basic elements of this orientation. The orientation provides a general overview of the mission and provides a forum for coordination of mission specific requirements, schedules, and data submittals. Mission peculiar designs and operational issues are reviewed so agreements can be established during the early phase of mission integration. Range Safety requirements that will be imposed on spacecraft designs and ground processing operations are identified.

For follow-on missions, a formal meeting is generally not necessary. Lockheed Martin will develop and submit a mission orientation letter to coordinate mission specific requirements, schedules, and data submittals. The spacecraft contractor will provide inputs to the mission orientation letter.

Spacecraft and Launch Vehicle Safety Assessments — Mission-specific spacecraft designs and ground processing operations are documented in the Missile System Prelaunch Safety Package (MSPSP) (Figure 4.3.2-1, Block B). The spacecraft contractor develops the spacecraft MSPSP to describe the spacecraft, document potential hazards associated with ground processing operations at the Range (e.g., pressure systems, ordnance control systems, toxic materials, spacecraft access requirements, RF testing, etc.), and define the means by which each hazard is eliminated or controlled to an acceptable level. Range Safety regulations provide details on the format and contents of the MSPSP.

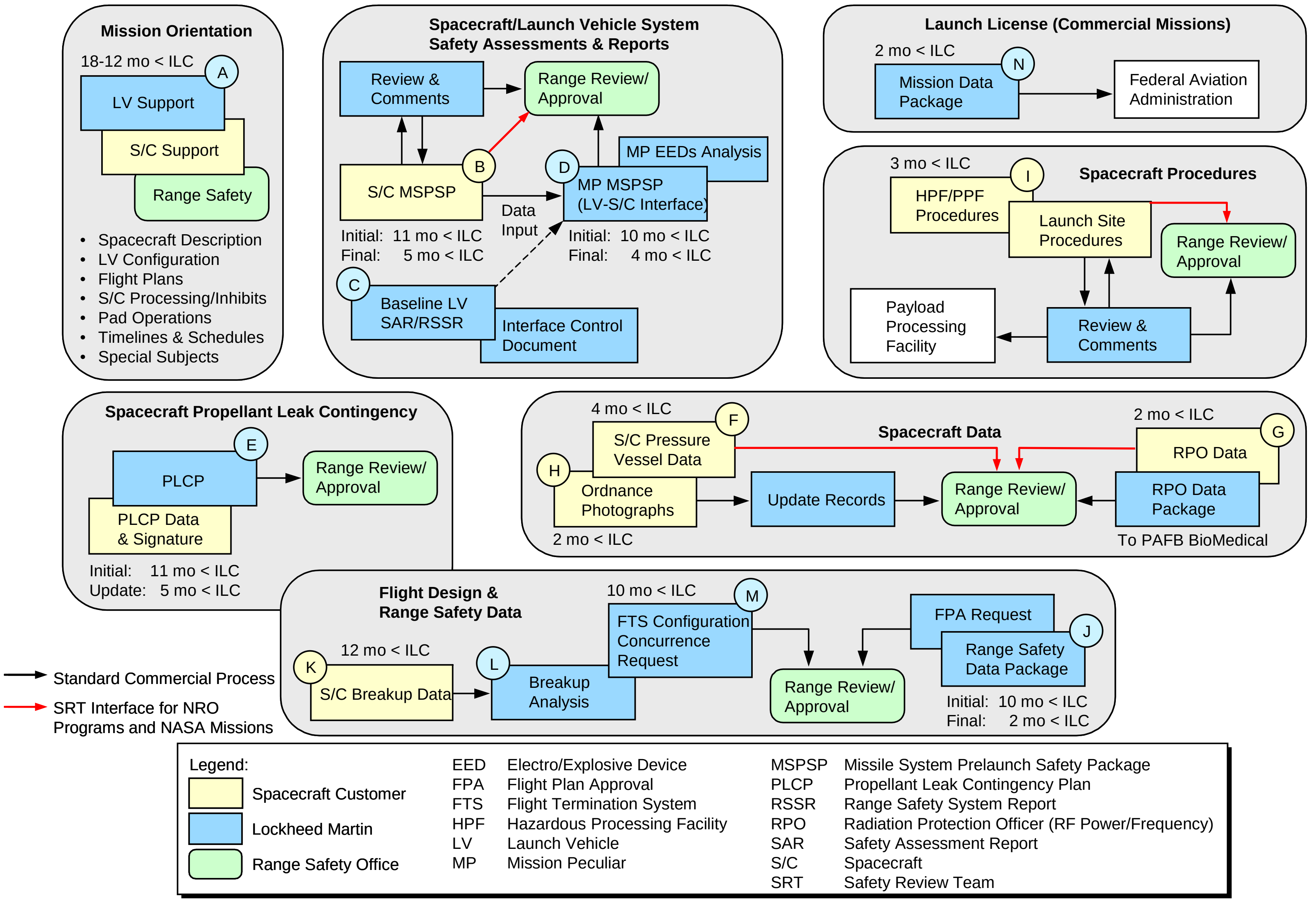
The initial spacecraft MSPSP is typically submitted to Lockheed Martin approximately 11 months before Initial Launch Capability (ILC). Lockheed Martin will review the document, provide comments if necessary, and forward the document with comments to the Range for formal review and comment. Lockheed Martin will then forward the spacecraft MSPSP review comments to the spacecraft contractor for incorporation into the final submittal of the MSPSP. The final spacecraft MSPSP is typically submitted to Lockheed Martin about 5 months before scheduled ILC.

Lockheed Martin will combine data from the spacecraft MSPSP with data from existing baseline Atlas V launch vehicle safety reports (Figure 4.3.2-1, Block C) and the mission-specific ICD to perform and document a safety assessment of the launch vehicle to spacecraft interface. Results of this assessment will be delivered to the Range as the mission unique launch vehicle MSPSP (Figure 4.3.2-1, Block D).

For Western Range programs, Lockheed Martin will develop and submit a Missile System Ground Safety Approval (MSGSA) request. Formal ground safety approval from the 30th Space Wing is typically received approximately two weeks prior to ILC.

Spacecraft Propellant Leak Contingency — Based on data supplied by the spacecraft contractor (e.g., hardware locations, access requirements, ground support equipment), Lockheed Martin will develop the spacecraft Propellant Leak Contingency Plan (PLCP) (Figure 4.3.2-1, Block E). The PLCP provides a top-level plan for offload of spacecraft propellants should leakage occur during ground processing operations at the launch pad. The PLCP requires development of detailed procedures by the spacecraft contractor to implement offload operations.

Figure 4.3.2-1: Atlas V Safety Integration Process*



*Timelines Typical - Mission Specific Adjustments as Required

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Spacecraft Data — The spacecraft contractor will provide pressure vessel qualification and acceptance test data to the Range (through Lockheed Martin) for review and acceptance. These data are shown in Figure 4.3.2-1, Block F. For follow-on missions, if the pressure vessel data remains unchanged, only acceptance data is required.

The spacecraft contractor will also submit data specifying the type and intensity of RF radiation that the spacecraft will transmit during ground testing, processing, and launch at the Range. Lockheed Martin will forward these data to the Radiation Protection Officer (RPO) for review and approval of RF related operations to be performed at the launch site. RPO data are shown in Figure 4.3.2-1, Block G. Refer to Appendix C, Section C.3.6.3, for a description of data required by the Range RPO. The spacecraft contractor will be required to complete the appropriate RPO forms.

The Range requires photographs showing locations of ordnance items installed on the spacecraft. These data are shown in Figure 4.3.2-1, Block H. spacecraft ordnance photographs may be submitted to the Range through Lockheed Martin, or the spacecraft contractor may submit ordnance photographs directly to the Range Safety Office. If the spacecraft contractor selects the direct submittal option, Lockheed Martin requires notification that photographs have been delivered. A follow-up meeting between the Range and the spacecraft contractor is typically required to review ordnance data.

Spacecraft Procedures — Through Lockheed Martin, the spacecraft contractor will submit onsite processing procedures (Payload Processing Facility [PPF] procedures and launch pad procedures) (Figure 4.3.2-1, Block I) to the operator of the PPF (e.g., Astrotech SSI, NASA, or Air Force) and the Range for review and approval. As indicated in Section 4.3.1, PPF procedures must comply with the applicable processing facility’s safety policy. Procedures to be implemented at the launch pad will comply with applicable Lockheed Martin and Range Safety regulations. For first time missions, the Range requires submittal of all spacecraft procedures. For follow-on missions, only hazardous procedures must be submitted.

Flight Design and Range Safety Data — Lockheed Martin’s flight design group will develop a Range Safety data package (Figure 4.3.2-1, Block J) that describes the basic spacecraft configuration, the preliminary flight profile, and the time of launch. Lockheed Martin will submit the preliminary (initial) package to the Range approximately 10 months before ILC. The initial data package will include a Flight Plan Approval (FPA) request to receive preliminary approval to fly the mission on the Range, as designed. Approximately 2 months before ILC, Lockheed Martin will submit the final Range Safety data package with a request for final FPA. Final FPA is usually received from the Range approximately 7 days before ILC.

To support development of the Range Safety data package, the spacecraft contractor will provide spacecraft breakup data (Figure 4.3.2-1, Block K) to Lockheed Martin. Lockheed Martin will use the breakup data to perform a breakup analysis on the spacecraft under expected mission conditions (Figure 4.3.2-1, Block L). Refer to Appendix C, Section C.3.6.4, for additional information on spacecraft breakup data and analysis.

Based on results of the breakup analysis, Lockheed Martin will submit an FTS configuration concurrence request to the Range (Figure 4.3.2-1, Block M). The purpose of this concurrence request is to obtain an agreement with the Range regarding requirements for a designated spacecraft destruct capability. Because there are no appreciable and/or additional public safety hazards with typical missions, Lockheed Martin typically pursues FTS concurrence without a separate spacecraft destruct system.

Launch License (Commercial Missions) — For commercial missions, Lockheed Martin maintains a launch license from the Federal Aviation Authority (FAA). The Atlas launch license requires periodic updates to address each commercial mission. Lockheed Martin will develop a mission specific addendum to the baseline license for each commercial flight and submit this data package (Block N, Figure 4.3.2-1) to the FAA. Spacecraft information included in the FAA data package will include MSPSP approval status and overviews of hazardous spacecraft commodities (propellants, pressure systems, batteries, etc).

4.4 ATLAS V IN-FLIGHT VIDEO OPTION

Lockheed Martin has the capability to provide a non-standard launch service option of flight proven in-flight video during the Atlas V launch vehicle mission. The Atlas V launch vehicle can be custom outfitted with up to two real-time NTSC video streams for pre-flight and in-flight video. The two video streams can be configured from video inputs from up to four standard camera locations with a user-defined camera switching sequence. Video cameras have been installed on previously flown missions for viewing launch vehicle ascent, launch vehicle staging events, payload fairing separation and spacecraft separation events. The in-flight video is provided to the customer in real-time viewing mode and high quality recordings are generated from supporting ground stations for post-flight use.

Table 4.4-1 lists the Atlas launch vehicles custom outfitted with in-flight video. To date, the Atlas family of launch vehicles has a 100% successful history of in-flight video performance. Figure 4.4-1 shows examples of in-flight video images taken during ascent.

Table 4.4-1 Atlas Launch Vehicle In-Flight Video Missions

Mission	Atlas Vehicle	Spacecraft	Launch Site	Date
AC-201	Atlas IIIA – First Flight from CCAFS	Eutelsat-W4	CCAFS	May 24, 2001
AC-160	Atlas IIAS	MLV-10	VAFB	September 8, 2001
AV-001	Atlas V 401 – First Flight from CCAFS	Hotbird-6	CCAFS	August 21, 2002
AV-003	Atlas V 521 – First Flight from CCAFS	Rainbow 1	CCAFS	July 17, 2003
AC-164	Atlas IIAS	MLV-14	VAFB	December 2, 2003
AV-004	Atlas V 431 – First Flight with 3 SRBs	MRSAT 4F1	CCAFS	ILC 2005
AC-206	Atlas III	MLV-15	CCAFS	ILC 2005

Figure 4.4-1 Atlas Launch Vehicle In-Flight Video Image – Solid Rocket Booster Separation

Figure 4.4-2 Atlas Launch Vehicle In-Flight Video Image – Payload Fairing Separation

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5. MISSION INTEGRATION AND MANAGEMENT

5.1 INTEGRATION MANAGEMENT

Clear communication between spacecraft and launch vehicle contractors is vital to accomplishing mission success. 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 commercial and government Atlas program missions. These proven processes and interfaces have enabled mission integration in as little as 4 months, with 12 months being a more typical schedule for commercial missions.

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

5.1.1 Launch Vehicle Responsibilities

Lockheed Martin Atlas Programs is responsible for Atlas V design, integration, checkout, and launch. This work is performed at facilities in Denver, Colorado; San Diego, California; and Harlingen, Texas. Major subcontractors include Pratt & Whitney (Centaur main engines and Atlas V booster engines), Honeywell (inertial navigation unit), Contraves (5-m payload fairings), and Aerojet (solid rocket boosters). As the spacecraft-to-launch vehicle integrating contractor, Lockheed Martin Atlas Programs is responsible for spacecraft integration, including electrical, mechanical, environmental, and electromagnetic compatibilities; guidance system integration; mission analysis; software design; Range Safety documentation and support; launch site processing and coordination.

5.1.2 Spacecraft Responsibilities

Since each spacecraft mission has unique requirements, interested Atlas V customers are encouraged to discuss their particular needs with the appropriate representatives of the Atlas Team. 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 meeting between the Atlas Team and the customer to assist in determining spacecraft and launch vehicle compatibility.

Customers are encouraged to contact the Atlas Team 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

The Atlas Mission Integration organization assigns an integration manager for each Atlas V mission. An Integration Manager will focus on engineering integration of the mission/spacecraft with the Atlas V launch vehicle, development of the Interface Control Document (ICD), tracking of action items, and coordination of requirements across engineering disciplines. This team will ensure on-time delivery of hardware, manage mission unique and launch readiness reviews, and coordinate with other areas of the broader Atlas V program. This team works to satisfy all customer requirements and keep the customer up-to-date on the status of the Atlas V program.

For commercial or international Atlas V missions, Lockheed Martin Commercial Launch Services (LMCLS) management assigns a program director. The program director serves as the primary interface with the customer, coordinates activities at Atlas facilities required to support the mission, and ensures that technical and contractual issues are addressed in a timely manner. LMCLS responds to a customer order by arranging services from several organizations such as Astrotech, U.S. Air Force (USAF), and Lockheed Martin Atlas Programs. Astrotech spacecraft integration facilities are contracted for most Atlas V launches, with additional support from other government facilities when required. Lockheed Martin Atlas Programs is responsible for Atlas V production, launch, and mission-peculiar integration processing. LMCLS has contracts with the U.S. government for use of Atlas V launch complexes and spacecraft integration facilities and with the USAF for range and launch site services.

To provide maximum efficiency in managing the many launch site operations, a launch site payload integrator is assigned to each mission. The launch site payload integrator is responsible for the development, integration, and installation of all spacecraft mission peculiar items at the launch site and at arrival of the spacecraft. This program organization concept has been used successfully for the Atlas program.

5.1.4 Integration Program Reviews

Integration program reviews focus management attention on significant milestones during the launch system design and launch preparation process to support major program milestones. As with working group meetings, these reviews can be tailored to customer requirements; however, for a first-of-a-kind launch, they may include a leading-edge design review, a Mission Peculiar Design Review, and a Launch Readiness Review. For typical communications spacecraft launches, only one design review is required. Additionally, Program Management Reviews are convened periodically to provide status.

Launch system meetings and reviews are scheduled according to the mission integration schedule. Spacecraft customer representatives may have access to Atlas facilities to attend reviews, meetings, and related activities. The incremental launch preparation review process provides management with an assessment of the readiness of the launch vehicle systems to proceed with launch preparation, and assurance that all mission functional and support elements are ready to support Range Safety, countdown, and launch activities. The program level review process is the primary mechanism for providing management with the visibility required to establish maximum confidence in mission success achievement for Atlas V launch systems.

Technical working group meetings are convened as necessary during the mission integration effort to define technical interfaces and resolve technical issues. At a minimum, technical meetings include a kick-off meeting, a mission peculiar design review, a Ground Operations Working Group, a systems review, and Ground Operations Readiness Review.

5.1.4.1 Mission Peculiar Design Reviews

Through significant experience with most communications spacecraft contractors, we have found that in the majority of cases, one Mission Peculiar Design Review (MPDR) can successfully meet goals and requirements that were, in the past, met by both a Preliminary Design Review (PDR) and a Critical Design Review (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. The Atlas Team prepares and presents the review with participation from the spacecraft contractor, launch services customer, and launch vehicle management.

Missions that require unique spacecraft interfaces with the launch vehicle or the mission design, two design reviews may be deemed necessary after discussions with the launch services customer and spacecraft contractor.

The MPDR includes the following subjects:

1. Requirements updated since the mission integration kickoff meeting;
2. Mission design, performance capability, and margin results;
3. Coupled loads analysis results summary;
4. Integrated thermal analysis results;
5. Spacecraft separation analysis preliminary results;
6. Guidance system accuracy analysis preliminary results;
7. Mission-peculiar flight software and parameter implementation and design updates (if required);
8. Radio frequency compatibility analysis, electromagnetic interference/electromagnetic compatibility analysis, and link margin analysis results;
9. Mission-peculiar electrical and mechanical interface hardware design;
10. Launch site implementation of unique requirements;
11. Mission-peculiar range asset requirement summary;
12. Range Safety documentation submittal status;
13. Venting analysis;
14. Contamination assessment;
15. Environmental Control System velocity impingement on spacecraft.

5.1.4.2 Launch Readiness Review

The Launch Readiness Review (LRR), conducted approximately two 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 contractors' individual certifications of launch readiness. The purpose of the LRR is to ensure that spacecraft systems, launch vehicle systems, facilities and Aerospace Ground Equipment (AGE), and all supporting organizations are ready and committed to support the final launch preparations, countdown, and launch. Atlas Team management representatives participate in the LRR, along with representatives from spacecraft customer organizations. Representatives from each key organization summarize their preparations and rationale for their readiness to proceed with the final launch preparations and countdown. The meeting concludes with a poll of each organization to express their readiness and commitment to launch.

5.1.4.3 Ground Operations Working Group

The Ground Operations Working Group (GOWG) meeting is convened at the launch site during the early part of the standard integration cycle. The GOWG includes representatives of the spacecraft customer, contractors, and all launch site organizations involved in operations. The GOWG provides a forum for coordinating launch site activities and resolving operational issues and concerns. It is usually co-chaired by the launch site spacecraft integrator and the spacecraft customer launch operations lead. At the GOWG, the following items are coordinated: the ground operations activities flow, operational timeline modifications for mission-peculiar spacecraft operational considerations, ICD interface requirements definition for launch site facilities and ground support equipment, hazardous operations with the Range, and ground test requirements. The GOWG reviews and approves the system documentation, operational timelines, and operational procedures required to process, test, and launch the integrated Atlas V/spacecraft vehicle. For spacecraft customers with nearly identical spacecraft (follow-on) and who are very familiar with Atlas V site processing, the GOWG may be waived.

5.1.4.4 Ground Operations Readiness Review

The Ground Operations Readiness Review (GORR) is conducted just before the arrival of the spacecraft at the launch site spacecraft processing facility. The meeting objectives are to formally kickoff the launch campaign; review the readiness of the facility to receive the spacecraft; and ensure that processing plans, schedules, procedures, and support requirements are coordinated. The technical chairperson is the Atlas V Launch Operations chief for spacecraft operations and the launch site spacecraft integrator is responsible for documenting meeting minutes and action items. A poll at the conclusion of the meeting ensures that all participating agencies, including spacecraft customer representatives, concur with the plan to be implemented for the launch campaign.

5.1.5 Integration Control Documentation

5.1.5.1 Program Master Schedule

This top-level schedule is prepared and monitored by the Atlas V 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 tasks and schedule requirements identified during initial integration meetings and is used by all participating organizations and working groups to develop and update sub-tier schedules. The mission integration schedule facilitates a systematic process to manage program activities. The mission integration schedule is used to track and monitor the mission progress to avoid significant schedule issues and possible cost impacts. The mission integration schedule contains sufficient mission details and contract Statement of Work (SOW) milestones to assist the program director and integration manager in managing the launch service.

5.1.5.2 Interface Requirements Documents

The customer creates the Interface Requirements Document (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 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

The Interface Control 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 Atlas Programs and is under configuration control after formal signoff. The document contains 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 Atlas V program management and the launch service customer. Subsequent changes to the mission ICD require agreement of the signing parties. If any conflict or inconsistency exists between the signed mission ICD and the IRD or the contract SOW, the signed mission ICD is given precedence.

ICD technical development is led by the integration manager with management decisions coordinated by the program director. The ICD is the top-level interface requirements document between the launch vehicle and the spacecraft. It contains physical, functional, environmental, operational, and performance requirements for the interface and is a contractually binding document. The document establishes how each interface requirement is to be verified to ensure that all interface details have been accomplished in compliance with ICD requirements. It identifies interface verification activities that link the designed, built, and tested interface back to the functional and performance requirement the interface was meant to satisfy.

5.2 MISSION INTEGRATION ANALYSIS

Lockheed Martin Atlas Programs will perform analyses, summarized in Table 5.2-1, to support a given mission. This table indicates the specific output of analyses to be performed, required spacecraft data, the timing during the integration cycle that the analysis is to be completed, and the application of 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. For a follow-on mission, all analyses are reassessed to ensure the original analysis is still applicable. Table 5.2-1 represents standard integration analyses. For many missions, Lockheed Martin Atlas Programs uses generic versions of these analyses and may not be required to perform a mission-peculiar version.

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

The Atlas program utilizes “Class Analysis” within Mission Integration. Class Analysis mission integration is based on broad envelopes of parameters rather than parameters for a specific mission. Analyses are performed up-front in the development phase of the program to encompass the known variation in individual vehicles and missions. The Atlas V program with its robust margins is able to extend the use of Class Analysis to all applicable mission design processes and analyses. Vehicle-to-vehicle variations and mission-to-mission variations are accounted for when defining boundaries of a class. Class Analysis reduces recurring mission integration span time while maintaining mission success by eliminating recurring characterization of the Atlas V launch vehicle and minimizing the need for recurring analyses. The Class Analysis method is enabled by the robustness and flexibility of the hardware design, software design, and standard spacecraft interface. Standardized designs can use standardized (class) analyses. Recurring analyses are replaced by assessments of the applicability of the previously performed Class Analysis and replace recurring flight software development with recurring parameter generation. The mission unique designs that require mission unique analyses are only those needed to verify unique requirements in the ICD. The following paragraphs describe mission integration Class Analyses.

Table 5.2-1: Summary of Typical Atlas V Mission Integration Analyses

Analysis	S/C Data	Analysis Products	No. of Cycles	Schedule	First-of-a-Kind	Follow-On
1. Coupled Loads	S/C Dynamic Math Model	• Spacecraft Loads • Dynamic Loss of Clearance • Launch Availability • PLF Jettison Evaluation	2	Model Delivery + 4 month	X	
2. Integrated Thermal	S/C Geometric & Thermal Math Models & Power Dissipation Profile	• Spacecraft Component Temperatures • Prelaunch Gas Conditioning & Set Points	1	Model Delivery + 4 month	X	
3. PLF Venting	S/C Venting Volume	• Pressure Profiles • Depressurization Rates	1	S/C Data + 2 month	X	
4. Critical Clearance	S/C Geometric Model; S/C Dynamic Model	• S/C-to-PLF Loss of Clearance (Dynamic + Static)	2	S/C Model Delivery + 4 month	X	
6. Spacecraft Separation & Clearance	S/C Mass Properties	• S/C Sep Clearance • S/C Sep Attitude, Rate & Spin-Up Verification	1	Design Review	X	
7. Post Separation Clearance		• LV-S/C Separation History	1	Design Review	X	
8. Pyro Shock	S/C Interface Definition	• S/C Shock Environment	1	Design Review	X	
9. Acoustics	S/C Geometry Fill Factors	• S/C Acoustics Environment	1	Design Review	X	
10. EMI/EMC	• S/C Radiated Emissions Curve • S/C Radiated Susceptibility Curve • S/C Rec Op & Demise Thresholds • S/C Diplexer Rejection	• Confirmation of Margins • Integrated EMI/EMC Analysis	1	Design Review	X	
11. Contamination	S/C Contamination Limits for Sensitive, Critical, Vertical & Horizontal Surfaces	• Contamination Analysis or Contamination Assessment	1	Design Review	X	
12. RF Link Compatibility & Telemetry Coverage (Airborne)	S/C Transmitter & Receiver Characteristics	• Link Margins • Frequency Compatibility • EEDs Susceptibility	1	Design Review	X	
13. RF Link Compatibility & Telemetry Coverage (Ground)	S/C Transmitter & Receiver Characteristics	• Link Margins • Identify Required Hardware	1	Design Review	X	
14. Performance	S/C Mass & Mission Requirements Definition	• Config, Performance & Weight Status Report • Performance Margin	3	ATP + 3-month, Design Review, Final Targeting	X	X

Analysis	S/C Data	Analysis Products	No. of Cycles	Schedule	First-of-a-Kind	Follow-On
15. Stability		• Control System Margins • RCS Use	1	L-2 month	X	
16. Mass Properties		• Mass Properties of Launch Vehicle		Coincident with Perf Reports	X	X
17. Trajectory Analysis	S/C 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	S/C Mass Properties	• LV Ground & Flight Winds Restrictions	1	Design Review	X	
22. Range Safety	S/C Breakup Data & Propulsion Characteristics	• Trajectory Data & Tapes for Range Approval	2	L-1 year, Prelimin-ary L-7 week, Final	X	
23. Electrical Compatibility	Electrical Interface Requirements	• End-to-End Circuit Analysis	1	Design Review	X	
24. Postflight		• Eval of Mission Data, LV Performance & Environment	1	L + 60/days	X	X
25. Destruct System		• Confirmation of Meeting Range Safety Requirements	1	Design Review	X	
26. Mission Targeting	Orbit Requirements	• Flight Constants Tapes • Firing Tables	1	L-1 month	X	X
27. Flight Software	Mission Requirements	• FCS Software	1	L-3 weeks	X	X

5.2.1 Coupled Loads Analysis

During the Atlas V program, a set of test-correlated three-dimensional (3-D) analytical launch vehicle models is generated for the mission-peculiar dynamic Coupled Loads Analysis (CLA). Mission-peculiar analyses will be performed where launch vehicle and spacecraft parameters may be affected (Sect. 3.2.1). While not all flight events are analyzed in each load cycle for each vehicle configuration, analyses typically performed as part of the CLA are identified below.

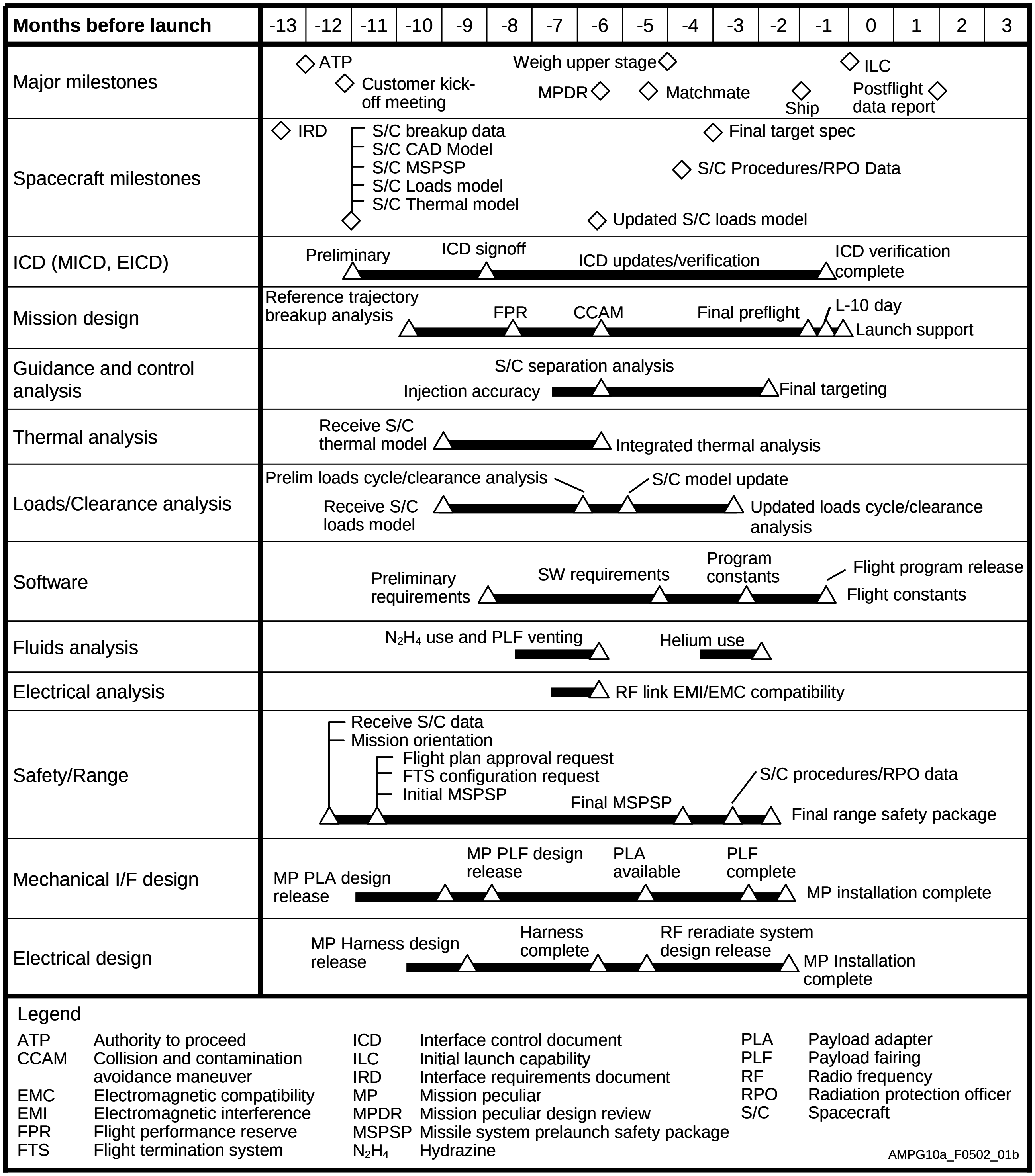
1. Spacecraft loading for critical flight events: liftoff, transonic buffet and gust, maximum dynamic pressure buffet and gust, Solid Rocket Booster (SRB) ignition, SRB burnout and jettison, Booster Engine Cutoff (BECO), maximum g-loading, Booster Package Jettison (BPJ), Booster Engine Thrust Oscillation (BETO), Centaur Main Engine Cutoff (MECO), and Centaur Longitudinal Event (CLE);

2. Flight wind launch availability assessment;

3. Payload Fairing (PLF) jettison evaluation applicable to a mission spacecraft;

4. Spacecraft loss of clearance evaluation for all critical flight events.

Figure 5.2-1: Twelve-month Generic Mission Integration Schedule



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 describes the type and format of the spacecraft dynamic model.

5.2.2 Integrated Thermal Analysis — Preflight and Flight

An integrated launch vehicle-spacecraft analysis of thermal environments imposed on the spacecraft under prelaunch conditions and for flight mission phases up to spacecraft separation will be performed. The Integrated Thermal Analysis (ITA) will be performed with customer-supplied spacecraft geometric and thermal math models and a detailed spacecraft power dissipation timeline. Results will be 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, PLF aeroheating analyses, PLF gas conditioning analyses, and free molecular heating analyses will be performed 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 spacecraft gas velocity analyses verify that impingement velocities are compatible with the defined spacecraft. A worst-case analysis is performed (using the maximum air conditioning supply rate) to determine flow conditions inside the PLF.

The gas conditioning 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 inlet temperature and flow rate conditions with spacecraft power dissipation levels.

PLF internal surface temperature ranges are predicted by analyzing flight aerodynamic heating.

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

5.2.3 PLF Venting Analysis (Ascent Phase)

A PLF venting analysis is performed 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)

The static payload envelope defines the usable volume for a spacecraft. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft and payload adapter interface. For clearances between the spacecraft and PLF, the primary clearance concerns are for dynamic deflections of the spacecraft and PLF and the resulting relative loss of clearance between these components. A critical clearance analysis is performed to verify that these deflections do not result in contact between the spacecraft and launch vehicle hardware. This analysis considers spacecraft and PLF static tolerances and misalignments, dynamic deflections, and out-of-tolerance conditions, and ensures that a minimum 25-mm (1-in.) clearance between the spacecraft and the PLF is maintained. During this analysis, dynamic deflections are calculated for ground handling, flight (from the coupled dynamic loads analysis, Sect. 5.2.1), and PLF jettison conditions. Clearance layouts and analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured after the spacecraft is encapsulated inside the PLF to ensure positive clearance during flight.

5.2.5 Spacecraft Separation Analysis

Extensive Monte Carlo analysis of 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 and spacecraft separation event is performed using finalized spacecraft mass properties to verify the Centaur will not recontact the spacecraft after separation system release. This analysis demonstrates minimum relative separation velocity, ensuring that adequate separation distance is achieved before initiating post separation Centaur maneuvers.

5.2.6 Spacecraft Post Separation 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 (RCS) uses 12 hydrazine thrusters. Four thrusters are dedicated to propellant settling (axial) control and eight are allocated to roll, pitch, and yaw control. 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, thrust directions are either 90 degrees or 180 degrees 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 two or three attitude maneuvers, combined with axial thrust from the RCS settling motors and blowdown of the Centaur tanks. For Atlas V vehicles, shortly after spacecraft separation, the Centaur typically turns 50 degrees from the separation attitude 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 and 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 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.

5.2.7 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 Payload Separation System (PSS). Verification of this environment has been accomplished by ground PSS testing of our existing separation systems.

5.2.8 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/spacecraft configurations.

5.2.9 Electromagnetic Interference/Electromagnetic Compatibility Analysis

An Electromagnetic Interference (EMI)/Electromagnetic Compatibility (EMC) plan is maintained to ensure compatibility between all avionics equipment. This plan covers requirements for bonding, lightning protection, wire routing and shielding, and procedures. Intentional and unintentional RF sources will be analyzed to confirm 6-decibel (dB) margins with respect to all general EMI/EMC requirements. In addition, Electroexplosive Device (EED) RF susceptibility analyses are performed to Range requirements for both the launch vehicle and spacecraft. The spacecraft analysis is performed by the spacecraft manufacturer and reviewed by Lockheed Martin Atlas Programs. The presence of an RF environment will affect safety margins of EEDs. This analysis is intended to confirm a minimum 20-dB margin with respect to the direct current (dc)

EED no-fire power level. The purpose of the EED susceptibility analysis is to demonstrate that safety margins of each EED are maintained when exposed to the flight vehicle and site sources RF environment. Comprehensive reports are published describing requirements and results of these analyses.

5.2.10 Contamination Analysis

Control of contamination (to meet analysis assumptions) is discussed in Section 3.1.3 for ground operations and in Section 3.2.7 for flight operations.

The spacecraft customer will be provided with an assessment of contamination contributions from Atlas V launch vehicle sources, as required. Starting from PLF encapsulation of the spacecraft through CCAM, contamination sources are identified and analyzed. This provides a qualitative assessment of the factors affecting spacecraft contamination to allow the spacecraft customer to approximate final on-orbit contamination budgets. A more detailed mission-peculiar analysis can be provided to the spacecraft customer if mission unique deposition requirements are specified in the ICD.

5.2.11 RF Link Compatibility and Telemetry Coverage Analysis (Airborne)

An airborne link analysis will be performed on all RF links between ground stations and the Atlas/Centaur vehicle to determine whether the signal strength between the RF system on the launch vehicle and the RF system at the receiving station meets mission requirements. The S-band telemetry system, the active C-band vehicle tracking system, and the flight termination system are analyzed. A program that considers airborne and ground station equipment characteristics, vehicle position, and attitude will be used. This analysis includes maximizing link margins with the receiving ground stations and the Tracking and Data Relay Satellite System (TDRSS) when the TDRSS-compatible transmitter is used on Centaur. A comprehensive report is published describing link requirements and results.

An RF compatibility analysis between all active airborne RF transmitters and receivers to ensure proper function of the integrated system will be performed. Transmit frequencies and their harmonics are analyzed for potential interference to each active receiver. If interference exists, a worst-case power-level analysis is performed to determine what effect the interference frequency has on the receiver’s performance. In addition, strong site sources, such as C-band site radar, are also analyzed. The spacecraft contractor provides details of active transmitters and receivers for this analysis. A comprehensive report is published describing analysis requirements and results.

5.2.12 RF Link Compatibility and Telemetry Coverage Analysis (Ground)

For customers who require communication with their spacecraft during prelaunch activities, a ground link analysis on spacecraft RF systems will be conducted to ensure that a positive link exists between the spacecraft and the spacecraft checkout equipment to checkout the spacecraft telemetry and command system. The RF reradiate system will provide sufficient margin to minimize effects of deviations or fluctuations in RF power and will provide consistent system performance to ensure positive link margin during the required time periods. Information about spacecraft requirements is in the ICD. A technical report is published describing link requirements and implementation of the link system.

5.2.13 Performance Analysis

The capability of Atlas V to place the spacecraft into the required orbit(s) is evaluated through our trajectory simulation tools. Vehicle performance capability is provided through our configuration, performance, and weight status report. This report is tailored to accommodate 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 list of the vehicle configuration status, mission-peculiar ground rules and inputs, and vehicle masses for performance analysis is included. The report also provides the more commonly used spacecraft partial derivatives (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.

5.2.14 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 Atlas V and Centaur autopilot configurations; establish gain and filter requirements for satisfactory rigid body, slosh, and elastic mode stability margins; verify vehicle and 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 and nonlinear simulation methods, ensuring that the Atlas V autopilot maintains robust stability throughout the defined mission. Correlation of simulation results with previous postflight data has confirmed the adequacy of these techniques.

5.2.15 Mass Properties Analysis

Mass properties will be analyzed, reported, and verified to support performance evaluation, structural loads analysis, control system software configuration development, ground operations planning, airborne shipping requirements, and customer reporting requirements.

5.2.16 Trajectory Analysis and Design

The 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 required performance reserves.

The trajectory design and simulation process provides the vehicle performance capability for the mission. It provides the basis, by simulation of dispersed 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. 5.2.13). Detailed insight into the tradeoffs used for the trajectory design is provided in the trajectory design report.

Our trajectory analysis tools incorporate detailed propulsion, mass properties, aerodynamic, and steering control modeling, as well as 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.

These simulation tools interface directly with actual flight computer software. This feature bypasses the need to have engineering equivalents of flight software. Another powerful feature 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.

5.2.17 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.

5.2.18 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 off-nominal effects of guidance system gyros and accelerometers. Software errors include Inertial Navigation Unit (INU) computation errors and vehicle dispersion effects. Positive and negative dispersions of more than 30 independent vehicle and atmospheric parameters that perturb Atlas V and Centaur performance are simulated. The accuracy analysis includes sensor noise, effects of vehicle prelaunch twist and sway on guidance system alignment during gyro compassing, and the covariance error analysis of the guidance hardware.

5.2.19 Launch Window Analysis

Launch window analyses are performed to define the open and close of mission-specific launch windows that satisfy mission-specific requirements on each launch day within the launch period. The Atlas/Centaur launch vehicle can accommodate launch windows, any time of day, any day of the year within performance capability constraints for a given mission design. Customers are requested to provide opening and closing times for the maximum launch window the spacecraft is capable of supporting. If the launch windows are several hours long or multiple windows in a single day, then a span within the total launch opportunity will be jointly chosen by the Atlas Team 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 further analysis. For example, launch system performance capability constrains 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.

Any launch window duration can be accommodated. However, a window of 30 minutes or more is recommended. Shorter windows increase the risk of a launch delay if exceeded due to weather or technical problem resolution. Windows longer than 3 hours for Atlas V may be limited by liquid oxygen supplies or crew rest.

5.2.20 Wind Placard Analysis (Prelaunch, Flight)

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

5.2.21 Range Safety Analyses

Flight analyses will be conducted as 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 one year before launch for the initial request and approximately 60 days before launch for the second. Reports and digital media of required information are provided to the Range agency in required formats and include nominal and dispersed trajectories and impact locations of jettisoned hardware. During spacecraft integration, a Range support plan is prepared documenting our planned coverage.

5.2.22 End-to-End Electrical Compatibility Analysis

An end-to-end electrical circuit analysis will be conducted to verify proper voltage and current parameters and any required timing and sequencing interfaces between all spacecraft and launch vehicle airborne interfaces (through to the end function). This analysis requires spacecraft data, such as contact assignments, wiring interfaces, and circuit detail of avionics (first level) to verify end-to-end (spacecraft-to-launch vehicle) compatibility. All “in-between” wiring and 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.

5.2.23 Postflight Data Analysis

Individual stage and spacecraft performance information are derived from available launch vehicle telemetry data using proven techniques. Main outputs of the analysis are: (1) Atlas V 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 parameters of interest. The report provides a trajectory listing of simulated Centaur flight that effectively matches observed data from the actual flight.

A primary input into the postflight vehicle performance analysis task is flight telemetry data. Telemetered outputs from the Propellant Utilization (PU) system are used to obtain 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 V stage shutdown, compared to the predicted nominal values, provide sufficient knowledge to obtain the Atlas V 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 injection conditions in terms of orbital parameters and deviations from target values and spacecraft separation attitude and rates. The report also documents spacecraft environments to the extent that the launch vehicle instrumentation permits. These environments could include interface loads, acoustics, vibration, and shock.

Finally, the report presents analyses of individual launch vehicle system performance and documents any anomalies noted during the mission. Launch vehicle and landline telemetry data provide the primary source of information for these analyses. Additionally, results of the review of optical data (from both fixed cameras at the launch site and tracking cameras) and radar data are also presented in the report.

5.2.24 Destruct System Analysis

Launch vehicle destruct system analysis is provided in our Range Safety System Reports (RSSR). Atlas V vehicle configurations are addressed in RSSR 14000-00-021. The reports comply with requirements specified in Appendix 4A of EWR 127-1, October 1997.

RSSR documents provide an overview of each vehicle configuration and detailed descriptions of the Flight Termination System (FTS), C-band tracking system, S-band telemetry system, and ground support equipment for Eastern and Western Range Safety systems. Component and system-level testing is also described. Antenna patterns, link margins, and FTS battery load capacity analyses are included.

As indicated in Section 4.3, the Atlas V program develops a spacecraft FTS configuration concurrence request for each mission (dedicated spacecraft destruct capabilities are generally not required for communications spacecraft).

5.2.25 Mission Targeting

Mission targeting is conducted to define 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 publication of flight constants parameter loads used for the flight computer and mission-peculiar firing tables documentation.

5.2.26 Mission Peculiar Flight Software

Our mission-peculiar software activity for mission integration is a controlled process that ensures the generation and release of validated Flight Control Subsystem (FCS) software to support the launch schedule. Our modular software design minimizes the impact of changes due to mission-peculiar requirements. This is achieved through the generic software design philosophy, which has been applied during development and evolution of the Atlas V FCS architecture. A parameterized software design has been implemented so that baseline FCS software is able to support all functionality necessary to fly most Atlas V missions. Parameters are then selected to properly implement the required mission-peculiar functionality.

Periodic updates to FCS software baselines are scheduled to support updates in the vehicle hardware configuration or to implement capability enhancements as required by the Atlas V program.

A rigorous software validation test program is run using the specific-mission trajectory and targeting parameters to validate the flight software and parameter data load under nominal, 3-sigma dispersed, severe stress, and failure mode environments, before release for flight. Testing and validation are completed in our Systems Integration Laboratory (SIL), which includes flight-like avionics components operating within a real-time simulation environment.

5.3 POLICIES

This section provides 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 spacecraft.

5.3.1 Launch Vehicle Logos

As part of our standard launch service, the Atlas V 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 V program typically needs to have final artwork for the logo by 6 months before launch. This timeframe allows the engineering organization to transform the artwork into a template to be used 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. Changes to the logo shall be supplied at a time that supports scheduled PLF completion date.

5.3.2 Launch Scheduling

Atlas V has a launch rate capability of 12 to 15 launches per year from LC-41 at Cape Canaveral Air Force Station (CCAFS) and 4 to 6 launches per year from SLC-3E at Vandenberg Air Force Base (VAFB).

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 months or less. The earlier a desired schedule position is contracted for the more likely it will be available.

Scheduling and rescheduling launches in the manifest require the equitable treatment of all customers (Table 5.3.2-1). 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). The Atlas Team 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.

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

General Policy
• Scheduling missions in the manifest and adjusting schedules due to Atlas V delays or customer delays; • Require consistent and even-handed treatment while minimizing cost and revenue impacts to both the customer and the Atlas V program • Positions are assigned to satisfy contract provisions and maintain customer satisfaction, while avoiding or minimizing aggregate delays to the manifest
Ground rules
• 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, spacecraft may be interchanged subject to mutual agreement
Delays
• Once in queue, customers will stay in queue if Atlas 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 spacecraft 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 and subject to change

5.3.3 Spacecraft Launch Window Options

Atlas V 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, CCAFS missions that will be scheduled during 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 schedule delay potential.

6. SPACECRAFT AND ATLAS V LAUNCH FACILITIES

Lockheed Martin has the capability to launch the Atlas V from launch facilities on the East and West coasts of the United States. Long-term use agreements with the United States Air Force (USAF) are in place for Launch Complex 41 (LC-41) at CCAFS, Figure 6-1, and for Space Launch Complex 3E (SLC-3E) at VAFB that encompass facilities, range services and equipment.

This section summarizes launch facility capabilities available to Atlas V customers at both CCAFS and VAFB.

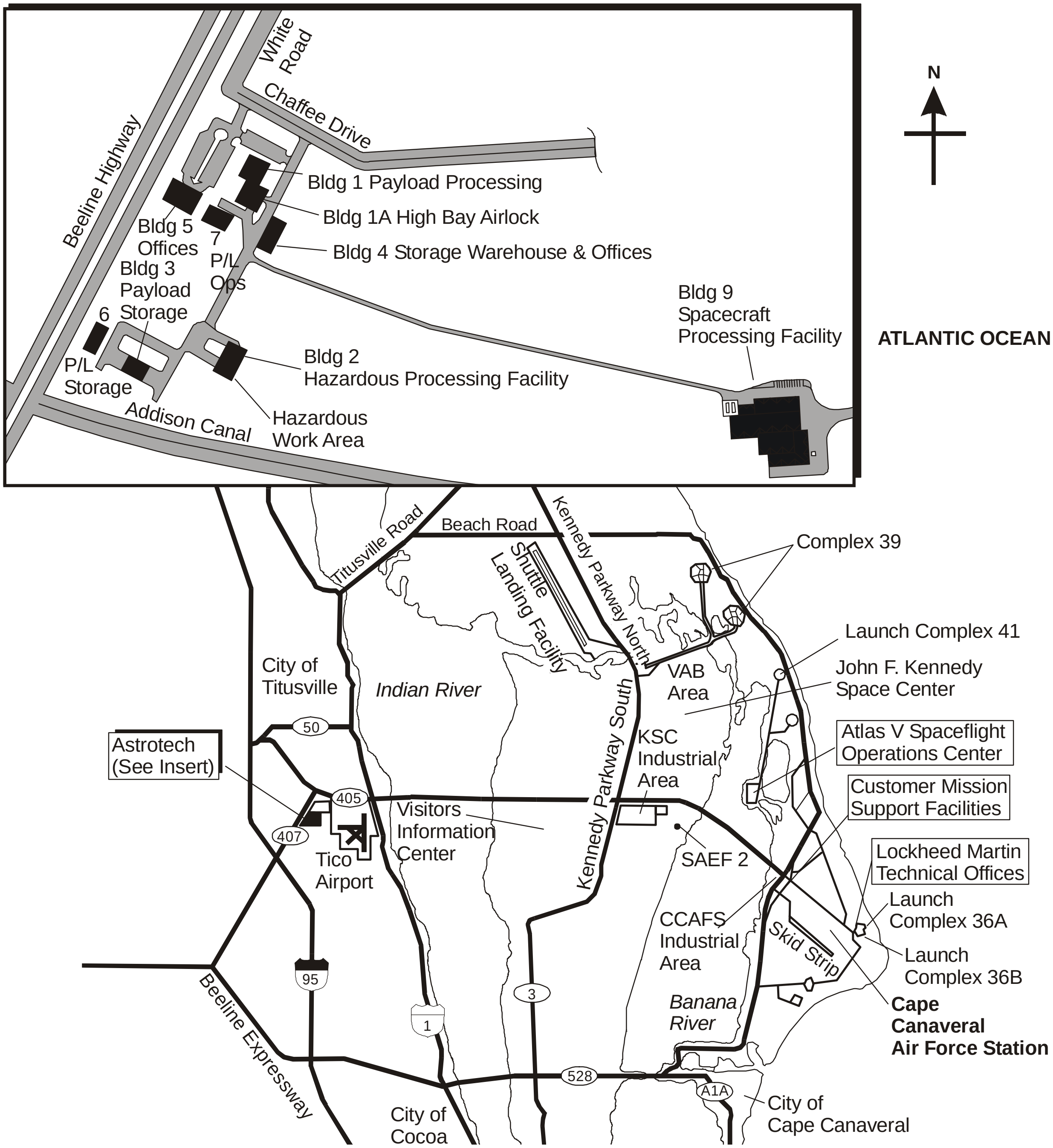
6.1 CCAFS ATLAS V LAUNCH FACILITIES

CCAFS facilities include spacecraft processing facilities available to commercial and users. Figure 6.1-1 identifies the location of facilities at CCAFS.

Figure 6-1 Atlas V CCAFS Launches

Atlas V 400 Series **Atlas V 500 Series**

Figure 6.1-1 CCAFS Facility Locations



6.1.1 CCAFS Spacecraft Processing Facilities

CCAFS facilities include spacecraft processing facilities available to commercial and U.S. government users. The Astrotech Payload Processing Facility (PPF), owned and operated by Spacehab Inc., is the primary facility for processing Atlas V class civil, government, and commercial spacecraft. Government facilities are available should the Astrotech facility not adequately satisfy spacecraft processing requirements, as outlined in Lockheed Martin Corporation/NASA and Lockheed Martin Corporation/USAF agreements. The Astrotech spacecraft processing and encapsulation facilities are described in the following section.

6.1.1.1 Astrotech Commercial Payload Processing Facility

The Astrotech facility, shown in Figure 6.1.1.1-1, 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.

Astrotech PPF — Astrotech Building 1 is considered the primary PPF and contains a total of four (4) high bay complexes with a supporting airlock. The PPF is used for nonhazardous spacecraft operations including final spacecraft assembly and checkout. Its floor plan is depicted in Figures 6.1.1.1-2 and 6.1.1.1-3. Table 6.1.1.1-1 lists details of Building 1 facility room dimensions, cleanliness, and crane capabilities. 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. Four 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.

Astrotech Hazardous Processing Facility — Astrotech's Building 2 is a Hazardous Processing Facility (HPF) located within the controlled hazardous work area of the complex. This facility supports liquid propellant transfer operations, solid propellant rocket motor and ordnance preparations, dynamic balancing, and spacecraft encapsulation operations. Table 6.1.1.1-2 lists the facility's room dimensions, cleanliness levels, and crane capabilities. 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 spacecraft encapsulation high bays,
4. One spin balance bay.

The Building 2 floor plan is depicted in Figure 6.1.1.1-4. 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.

Astrotech Spacecraft Processing Facility — Astrotech's Building 9 Spacecraft Processing Facility (SPF) provides an additional HPF. This facility is capable of processing the larger Atlas V 5-m Payload Fairing (PLF). Final preparations of the spacecraft and the PLF are carried out in the SPF in a similar manner to those in the HPF. Table 6.1.1.1-3 lists the facility's room dimensions, cleanliness levels, and crane capabilities. Major areas of the SPF are:

1. One airlock,

2. One encapsulation bay,

3. Two high bays,

4. Two propellant cart storage rooms,

5. Two garment change rooms,

6. Three control rooms.

The Building 9 floor plan is depicted in Figure 6.1.1.1-5. Each processing cell is built to explosion-proof or equivalent standards to support operations involving liquid propellant transfer, solid propellant motor preparations, and ordnance installation.

Astrotech Spacecraft Support Facilities — Building 3 at Astrotech is a thermally controlled, short-term spacecraft storage area with spacecraft processing activities or long-term spacecraft storage. Storage bay and door dimensions and thermal control ranges are identified in Table 6.1.1.1-4.

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 6.1.1.1-5 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 offices with a reception area sufficient to accommodate up to three secretaries.

Astrotech's Building 6 provides storage primarily for Lockheed Martin Payload Fairing (PLF) support equipment. However, if customers require additional nonenvironmentally controlled storage, this additional space can be made available.

Spacecraft Services — Full services for spacecraft 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 fc of illumination. Control rooms, offices, and conference areas have 35-W fluorescent lamps to maintain 70 fc 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 (OIS) and Transistorized Operational Phone System (TOPS) to provide multiple-channel voice communications between the Astrotech facility and selected locations at CCAFS.

Closed-Circuit Television — Closed-Circuit Television (CCTV) cameras are located in the high bays of Building 2 and can be placed in the 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)/CCAFS via a dedicated microwave link.

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

Astrotech provides antennas for direct S-band, C-band, and Ku-band airlinks from the Astrotech facility to LC-41 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.

Customer Local Area Network — Astrotech provides customer assistance for connectivity into an existing Local Area Network (LAN) that has drops available in the PPF and HPF. System interface is via standard RJ 45 connectors. End-item instruments, such as hubs, are customer provided. A limited number of drops on this system are available at the Launch Services Buildings (LSB). If the customer has arranged for a T1 line from an outside provider for LAN connectivity to their home facilities, Astrotech will ensure it is properly connected to the customer's PPF control room. Customers are responsible for verifying connectivity of the T1 line from the Astrotech facility back to their own facility.

Remote Spacecraft Control Center — Astrotech has the capability to link remote ground stations (voice and data) between Astrotech and CCAFS 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. Cypher locks on all doors leading into spacecraft processing areas provide internal security. 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.

Figure 6.1.1.1-1 Astrotech Facility

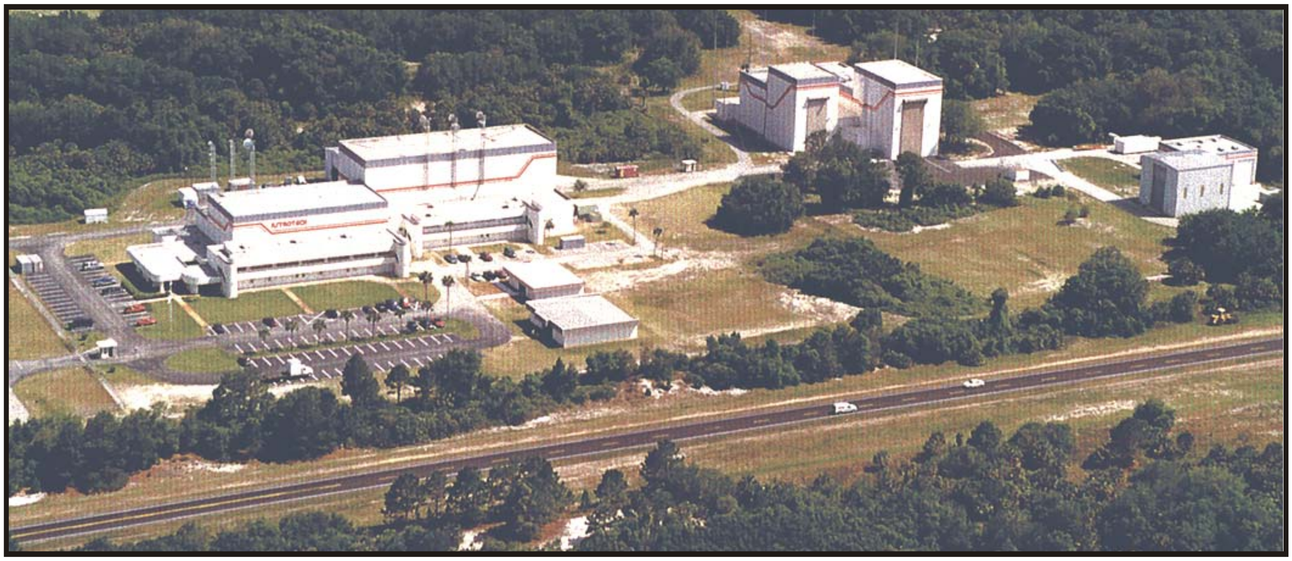
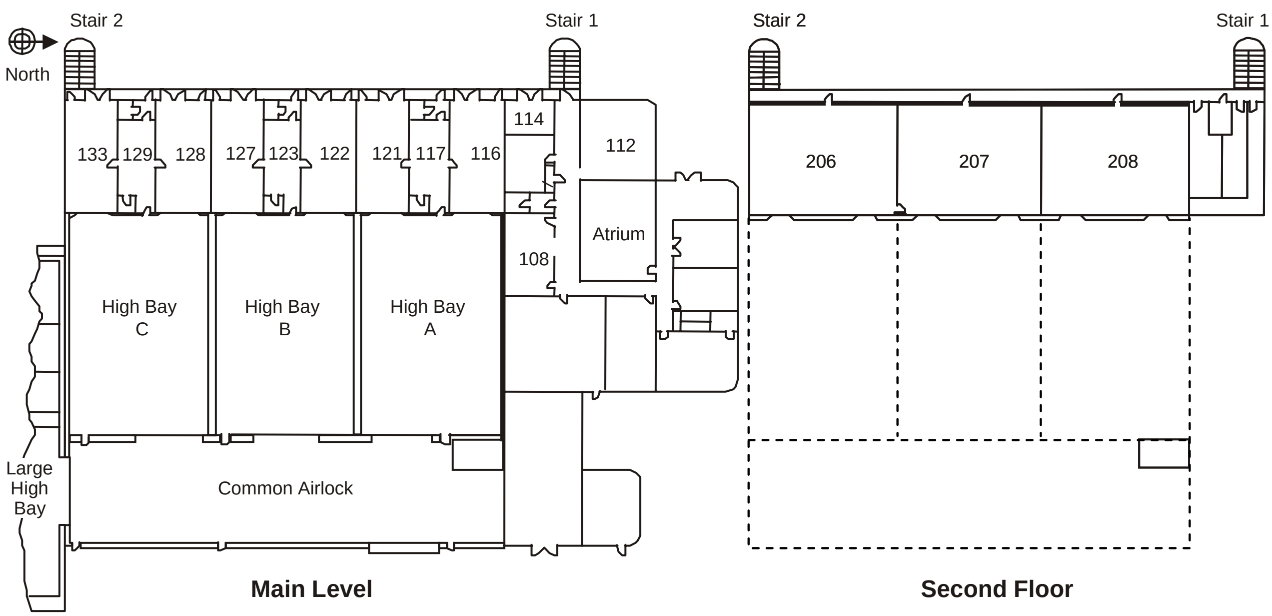


Figure 6.1.1.1-2 Astrotech Building 1 Detailed Floor Plan



Room	Function	Room	Function	Room	Function
108	Conference Room	121	Control Room A2	129	Change Room C
112	Break/Lunch Room	122	Control Room B1	133	Control Room C2
114	Machine Shop	123	Change Room B	206	Office Area C
116	Control Room A1	127	Control Room B2	207	Office Area B
117	Control Room A	128	Control Room C1	208	Office Area A

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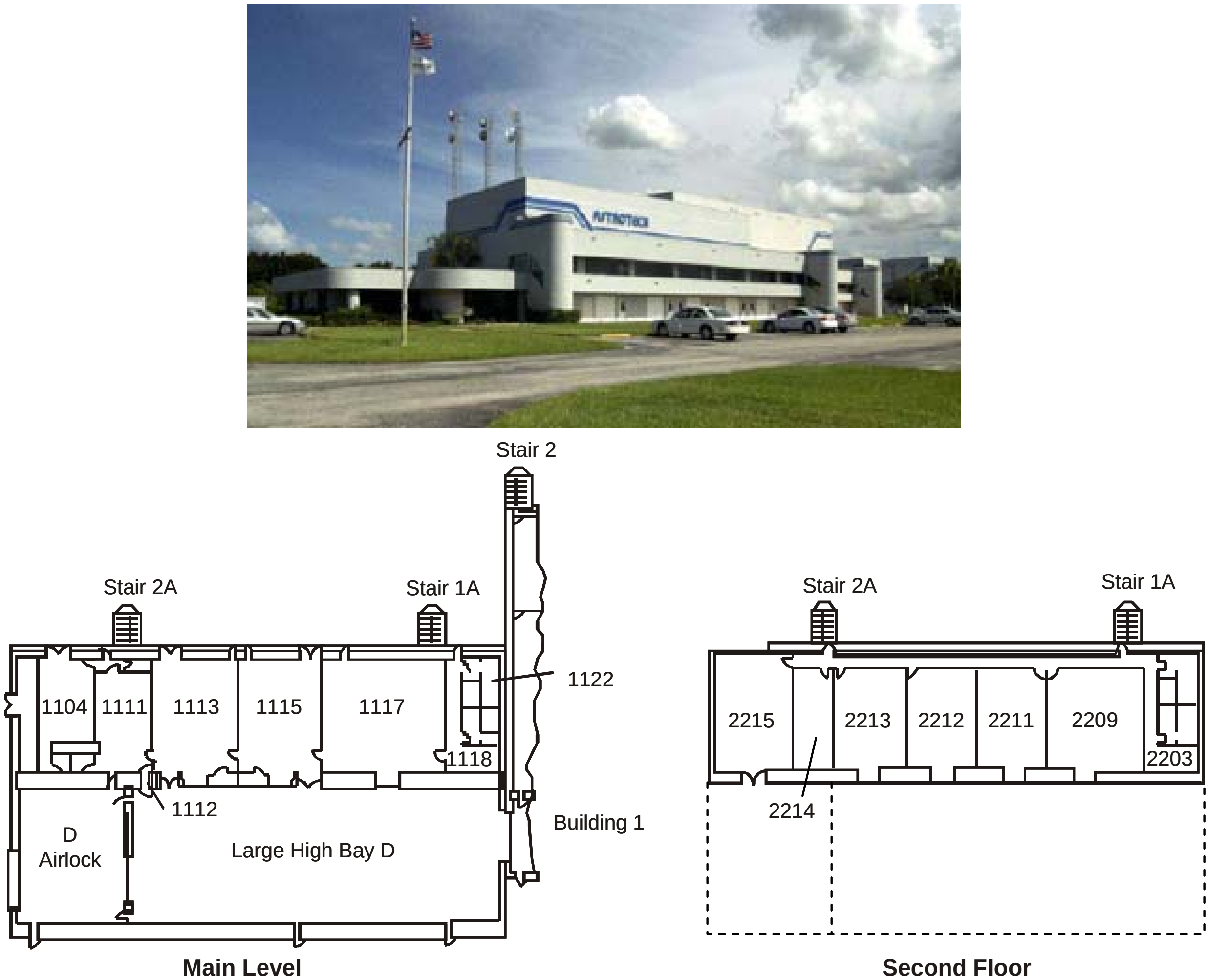
Figure 6.1.1.1-3 Astrotech Building 1 Large Bay Detailed Floor Plan – First Floor



Room	Function	Room	Function	Room	Function
1104	Conference D1	1117	Office Area	2212	Office Area
1111	Change Room D	1118	Break Room	2213	Office Area
1112	Air Shower	2203	Break Room	2214	Office Area
1113	Control Room D2	2209	Office Area	2215	Office Area
1115	Control Room D1	2211	Office Area		

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Table 6.1.1.1-1: Astrotech Building 1 Payload Processing Facility

High Bays A, B, C (3) - Class 100,000 Clean Room			**Large High Bay D - 100,000 Clean Room**		
Temperature	23.8 ±2.8 °C*	75 ±5 °F*	Temperature	23.8 ±2.8 °C	75 ±5 °F
Relative Humidity	50 ±5%*		Relative Humidity	50 ±5%	
Usable Floor Space	18 x 12.19 m	60 x 40 ft	Usable Floor Space	38.1 x 15.5 m	125 x 51 ft
Ceiling Height	13.2 m	43.5 ft	Ceiling Height	18.3 m	60 ft
Crane Type (Each Bay)	Bridge		Crane Type (Each Bay)	Bridge	
Crane Capacity	9.07 tonne	10 ton	Crane Capacity[1]	27.21 tonne	30 ton
Crane Hook Height	11.3 m	37.1 ft	Crane Hook Height	15.2 m	50 ft
Roll-Up Door Size (w x h)	6.1 x 7 m	20 x 23 ft	High Bay Airlock Door Size (w x h)	6.1 x 15.2 m	20 x 50 ft
			Main Airlock Door Size (w x h)	6.1 x 7 m	20 x 23 ft
High Bay Control Rooms A, B, C (3)			**Large Bay Control Rooms (2)**		
Size	9.1 x 12.8 m	30 x 42 ft	Size	9.1 x 10.7 m	30 x 35 ft
Ceiling Height	2.67 m	8.75 ft	Ceiling Height	2.8 m	9.33 ft
Door Size (w x h)	2.44 x 2.44 m	8 x 8 ft	Door Size (w x h)	2.44 x 2.44 m	8 x 8 ft
Temperature	23.8 ±2.8 °C	75 ±5 °F	Temperature	23.8 ±2.8 °C	75 ±5 °F
Bay Window Size	1.22 x 2.44 m	4 x 8 ft			
Common Airlock: Class 100,000 Clean Room			**Large Bay Airlock**		
Temperature	23.8 ±2.8 °C*	75 ±5 °F*	Temperature	23.8 ±2.8 °C	75 ±5 °F
Relative Humidity	50 ±5%*		Relative Humidity	50 ±5%	
Usable Floor Space	36.6 x 9.14 m	120 x 30 ft	Usable Floor Space	12.2 x 15.5 m	40 x 51 ft
Ceiling Height	7.0 m	23 ft	Ceiling Height	18.3 m	60 ft
Door Size (w x h)	6.1 x 7 m	20 x 23 ft	Door Size (w x h)	6.1 x 15.2 m	20 x 50 ft
			Crane Hook Height	15.2 m	50 ft
			Crane Capacity[1]	27.21 tonne	30 ton
Offices: Second Floor			**Offices: Second Floor**		
Floor Size	12.5 x 7.3 m	41 x 24 ft	Floor Size (D1)	10.7 x 12.5 m	35 x 41 ft
Ceiling Height	2.43 m	8 ft	Floor Size (D2)	9.1 x 10.7 m	30 x 35 ft
Door Size (w x h)	0.91 x 2.04 m	3 x 6.7 ft	Floor Size (D3)	7.6 x 9.1 m	25 x 30 ft
Bay Window Size	1.22 x 1.22 m	4 x 4 ft	Ceiling Height	2.8 m	9.33 ft

[1]Building 1 D high bay and airlock crane capacity is 27.21 tonnes (30 tons). A 13.6 tonne (15 ton) hook is installed as standard; however, the 27.2 tonne (30 ton) hook can be installed at the request of the customer. The crane is normally proof-tested for the 13.6 tonne (15 ton) hook only.

Figure 6.1.1.1-4 Astrotech Building 2 Hazardous Operations Facility

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Table 6.1.1.1-2: Astrotech Building 2- Three Blast-Proof Processing Rooms

North and South High Bay - Class 100,000 Clean Room			Airlock - Class 100,000 Clean Room		
Temperature	23.9 ±2.8 °C*	75 ±5 °F*	Temperature	23.9±2.8 °C*	75±5 °F*
Relative Humidity	50 ±5%*		Relative Humidity	50 ±5%*	
Floor Space	18.3 x 11.3 m	60 x 37 ft	Floor Space	11.6 x 8.8 m	38 x 29 ft
Ceiling Height	13.1 m	43 ft	Ceiling Height	13.1 m	43 ft
Crane Type (Each Bay)	Bridge		Crane Type	Monorail	
Crane Capacity	9.1 tonne	10 ton	Crane Capacity	1.8 tonne	2 ton
Crane Hook Height	11.0 m	36.2 ft	Crane Hook Height	11.3 m	37 ft
Fuel Island	7.6 x 7.6 m	25 x 25 ft	South Encapsulation Bay Door (w x h)	6.1 x 12.2 m	20 x 40 ft
Spin Bay Door (w x h)	6.1 x 13.1 m	20 x 43 ft			
North Encapsulation Bay & South Air Lock Doors (w x h)	6.1 x 12.2 m	20 x 40 ft	South High Bay Door (w x h)	6.1 x 12.2 m	20 x 40 ft
North High Bay Control Room			**North Encapsulation High Bay Class 100,000 Clean Room**		
Temperature	21.1 to 23.9 °C	70 to 75 °F			
Floor Space	7.6 x 9.1 m	25 x 30 ft	Temperature	23.9±2.8 °C*	75±5 °F*
Ceiling Height	2.84 m	9.33 ft	Relative Humidity	50±5%*	
Outside Door (w x h)	2.4 x 2.4 m	8 x 8 ft	Floor Space	15.2 x 12.2 m	50 x 40 ft
Bay Window (w x h)	0.81 x 0.76 m	2.67 x 2.50 ft	Fuel Island	7.6 x 7.6 m	25 x 25 ft
South High Bay Control Room			Ceiling Height	19.8 m	65 ft
Temperature	21.1 to 23.9 °C	70 to 75 °F	Crane Type	Bridge	
Floor Space	7.6 x 7.6 m	25 x 25 ft	Crane Capacity	27.2 tonne	30 ton
Ceiling Height	2.84 m	9.33 ft	Crane Hook Height	16.99 m	55.75 ft
Outside Door (w x h)	2.4 x 2.4 m	8x8 ft	Outside Door (w x h)	6.1x15.2 m	20 x 50 ft
Bay Window (w x h)	0.81 x 0.76 m	2.67 x 2.50 ft	North High Bay Door (w x h)	6.1x12.2 m	20 x 40 ft
Spin Balance High Bay Class 100,000 Clean Room			**South Encapsulation High Bay Class 100,000 Clean Room**		
Temperature	23.9±2.8 °C*	75±5 °F*	Temperature	23.9 ±2.8 °C*	75 ±5 °F*
Relative Humidity	50±5%*		Relative Humidity	50 ±5%*	
Floor Space	14.6x8.2 m	48x27 ft	Floor Space	13.7 x 21.4 m	45 x 70 ft
Ceiling Height	13.1 m	43 ft	Ceiling Height	19.8 m	65 ft
Crane Type	Bridge		Crane Type	Bridge	
Crane Capacity	9.1 tonne	10 ton	Crane Capacity	27.2 tonne	30 ton
Crane Hook Height	11.0 m	36.2 ft	Crane Hook Height	17.1 m	56.7 ft
North & South High Bay Door (w x h)	6.1 x 13.1 m	20 x 43 ft	Outside Door (w x h)	7.0 x 16.8 m	23 x 55 ft
			South Airlock Door (w x h)	6.1 x 12.2 m	20 x 40 ft
Spin Balance High Bay Control Room			**Propellant (Oxidizer & Fuel) Cart Rooms (2)**		
Temperature	23.9 ±2.8 °C	75 ±5 °F	Floor Space	6.1 x 6.1 m	20 x 20 ft
Floor Space	3 x 4.6 m	10 x 15 ft	Ceiling Height	2.84m	9.33 ft
Ceiling Height	2.84 m	9.33 ft	South High Bay Door	3.1 x 2.4 m	10 x 8 ft
South Control Room Door (w x h)	1.8 x 2.03 m	6 x 6.67 ft	Outside Door (w x h)	1.8 x 2.03 m	6 x 6.67 ft
* Can be Adjusted as Needed					

Figure 6.1.1.1-5: Astrotech Building 9 Layout

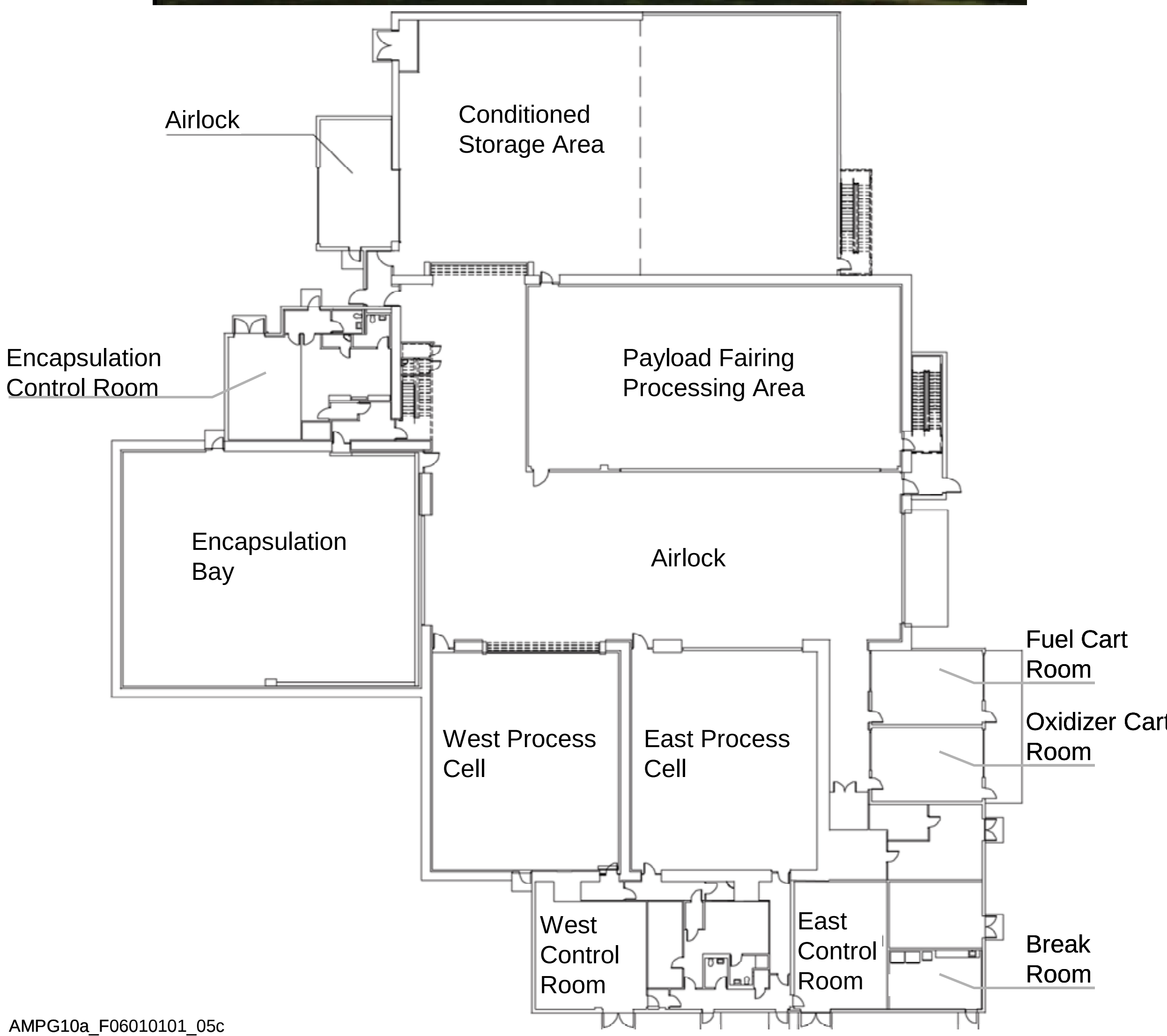

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Table 6.1.1.1-3: Astrotech Building 9 - Two Processing Rooms

Airlock - Class 100,000 Clean Room		
Temperature	23.9±2.8 °C*	75 ±5 °F*
Relative Humidity	50 ±5%*	
Floor Space	38.7x30.2 m	127 x 99 ft
Ceiling Height	30.8 m	101 ft
Crane	Bridge/Trolley	
Crane Capacity	27.15 tonne	30 ton
Crane Hook Height	27.7 m	91 ft
Encapsulation Bay Door (w x h)	9.1 x 27.7 m	30 x 91 ft
East/West High Bays Doors (w x h)	9.1 x 19.8 m	30 x 65 ft
Outside Door (w x h)	9.1 x 27.7 m	30 x 91 ft
Fairing Process Room Door (w x h)	21.3 x 7.6 m	70 x 25 ft
Fairing Cnd Storage Door (w x h)	7.6 x 19.8 m	25 x 65 ft
East High Bay - Class 100,000 Clean Room		
Temperature	23.9±2.8 °C*	75 ±5 °F*
Relative Humidity	50 ±5%*	
Floor Space	18.3x15.2 m	60 x 50 ft
Ceiling Height	24.4 m	80 ft
Access Door (w x h)	0.9 x 2.1 m	3 x 7 ft
Crane Type (Each Bay)	Bridge/Trolley	
Crane Capacity	22.67 tonne	25 ton
Crane Hook Height	22.3 m	73 ft
East High Bay Control Room		
Temperature	21.1-23.9 °C	70-75 °F
Floor Space	10.7 x 7.6 m	35 x 25 ft
Ceiling Height	3.2 m	10.5 ft
Door for EGSE Installation (wxh)	2.4 x 3.0 m	8 x 10 ft
Access Door (w x h)	0.9 x 2.1 m	3 x 7 ft
West High Bay - Class 100,000 Clean Room		
Temperature	23.9±2.8 °C*	75 ±5 °F*
Relative Humidity	50 ±5%*	
Floor Space	18.3x15.2 m	60 x 50 ft
Ceiling Height	24.4 m	80 ft
Access Door (w x h)	0.9 x 2.1 m	3 x 7 ft
Crane Type (Each Bay)	Bridge/Trolley	
Crane Capacity	27.21 tonne	30 ton
Crane Hook Height	22.3 m	73 ft
West High Bay Control Room		
Temperature	21.1-23.9 °C	70-75 °F
Floor Space	9.3 x 9.3 m	30.5 x 30.5 ft
Ceiling Height	3.2 m	10.5 ft
Door for EGSE Installation (wxh)	2.4 x 3.0 m	8 x 10 ft
Access Door (w x h)	0.9 x 2.1 m	3 x 7 ft

Encapsulation Bay - Class 100,000 Clean Room		
Temperature	23.9±2.8°C*	75 ±5 °F*
Relative Humidity	50±5%*	
Floor Space	24.4x19.8 m	80 x 65 ft
Ceiling Height	33.5 m	110 ft
Crane Type	Bridge/Trolley	
Crane Capacity	45.25 tonne	50 ton
Crane Hook Height	30.5 m	100 ft
Encapsulation Control Room		
Temperature	21.1-23.9 °C	70-75 °F
Floor Space	8.5 x 5.8 m	28 x 19 ft
Ceiling Height	3.2 m	10.5 ft
Door for EGSE Installation (w x h)	2.4 x 3.0 m	8 x 10 ft
Access Door (w x h)	0.9 x 2.1 m	3 x 7 ft
Encapsulation Bay Garment Change Room		
Floor Space	5.2 x 7.3 m	17 x 24 ft
Ceiling Height	2.7 m	9 ft
Fairing Processing Area		
Floor Space	30.8x15.7 m	101x51.5 ft
Ceiling Height	9.1 m	30 ft
Conditioned Storage Area		
Floor Space	30.8x15.7 m	101x51.5 ft
Ceiling Height	19.8&6.1 m	65 & 20 ft
Conditioned Storage Airlock		
Floor Space	11.6 x 9.1 m	38 x 30 ft
Ceiling Height	11.6 m	11.6 m
Outside Door (w x h)	6.1 x 9.1 m	20 x 30 ft
Airlock Door (w x h)	6.1 x 9.1 m	20 x 30 ft
Crane Capacity	13.6 tonne	15 ton
Crane Hook Height	9.1 m	30 ft
Propellant (Oxidizer & Fuel) Cart Rooms (2)		
Floor Space	6.7 x 4.4 m	22 x 14.5 ft
Height	2.7 m	9 ft
Airlock Door (w x h)	3.05 x 3.0m	10 x 10 ft
Outside Door (w x h)	3.7 x 3.0 m	12 x 10 ft
Floor Space	6.7 x 4.4 m	22 x 14.5 ft
Height	2.7 m	9 ft
Break Room		
Floor Space	7.6 x 5.2 m	25 x 17 ft
Ceiling Height	2.7 m	9 ft

*Can be Adjusted as Needed

Table 6.1.1.1-4: Astrotech Building 3 – Short Term Spacecraft Storage

Building 3: Thermally Controlled Storage Facility		
Temperature Control	25.6 °C	70 to 78 °F
Relative Humidity	50±10%	
Height (All Bays)	8.5 m	28 ft
Floor Space		
• Bays A, C, D & F	7.6 x 6.7 m	25 x 22 ft
• Bays B & E	7.6 x 7.3 m	25 x 24 ft
Door Size (w x h)		
• Bays A, C, D & F	6.1x7.6 m	20x25 ft
• Bays B & E	5.5x7.6 m	18x25 ft
Crane	None	

Table 6.1.1.1-5: Astrotech Building 4 - Warehouse Storage Area

Building 4: Storage Without Environmental Control		
Floor Space	15.2x38.1 m	50x125 ft
Height	8.5 m	28 ft
Door Size	6.1x7.9 m	20x26 ft
Crane	None	

6.1.2 Encapsulated Spacecraft Transport to LC-41

The primary encapsulated spacecraft transporter is a nine-axle KMAG 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 within the PLF during the transport period. The transporter includes an environmental monitoring instrumentation system that provides continuous digital display of the PLF and ECS environment during transport.

6.1.3 CCAFS Atlas V Launch Site Facilities

Atlas V uses three primary facilities and integrated GSE that support spacecraft processing in addition to the encapsulation facilities identified in Section 6.1.1. The following section describes LC-41 consisting of the Vertical Integration Facility, Payload Van, the Mobile Launch Platform, the launch pad and the Atlas Spaceflight Operations Center.

6.1.3.1 Vertical Integration Facility

The Vertical Integration Facility (VIF) is a weather-enclosed steel structure, approximately 22.9 m (75 ft) square and 87.2 m (286 ft) tall, with a fabric roll-up door, a hammerhead bridge crane, platforms, and servicing provisions required for launch vehicle integration and checkout (Figure 6.1.3.1-1). Launch vehicle processing in the VIF includes stacking booster(s) and Centaur, performing launch vehicle subsystem checks and system verification, installing the encapsulated spacecraft, performing integrated system verification, final installations, and vehicle closeouts.

No provisions for spacecraft propellant loading are available at the launch complex, either on-pad or in the VIF. The spacecraft is fueled at the PPF before encapsulation. If required, emergency spacecraft detanking on the launch complex will occur in the VIF.

The facility includes two stairways and one freight elevator for access to the platforms. Elevator capacity is 1,820 kg (4,000 lb), with maximum cargo dimensions of 1.7 m (5 ft, 7 in.) width, 2.4 m (7 ft, 9 in.) length, and 2.1 m (7 ft) height.

Access to the spacecraft through the PLF access doors occurs on VIF Levels 5 to 7, depending on the launch vehicle and spacecraft configuration. Access to the spacecraft, if required, is provided using portable access stands. Class 5,000 conditioned air and localized GN_2 purges (if required) are provided to the PLF to maintain the spacecraft environment. Spacecraft conditioned air cleanliness, temperature, humidity and flow rate are as follows:

1. Cleanliness Class 5,000;

2. Temperature 10-29 °C (50-85 °F), selectable with tolerance ±2.8 °C (±5 °F);

3. Relative humidity 20-50%, maximum dewpoint 4.4 °C (40 °F) and 35-50% when required for sensitive operations;

4. Flow rate: 0.38-1.21 kg/s (50-160 lb/min) selectable with tolerance ±0.038 kg/s (±5 lb/min) (400 series), 0.38-2.27 kg/s (50-300 lb/min) selectable with tolerance 0.095 kg/s (±12.5 lb/min) (500 series).

Accommodations for spacecraft personnel and test equipment are provided on VIF Levels 5, 6, 6.5, and 7. Table 6.1.3.1-1 identifies power, commodities, and communications interfaces on these levels. Spacecraft customers are encouraged to contact Lockheed Martin for further details and capabilities.

Figure 6.1.3.1-1: VIF

The 54,000-kg (60-ton) facility crane used for lifting and mating the encapsulated spacecraft has creep speed controls down to 30 mm/min (1.2 in/min) on the bridge and trolley, and 12.5 mm/min (0.5 in/min) on hoist, with maximum acceleration/deceleration of 75 mm/s^2 (0.25 ft/s^2).

Critical power (redundant facility power) and facility power (120 V) are provided to support spacecraft operations in the VIF. Communications on spacecraft access levels in the VIF include unsecure operational voice system, telephone, public address and user interfaces. Lighting to 50 fc is provided on spacecraft access levels in the VIF. If it should be necessary to perform an emergency spacecraft propellant detank, the VIF includes fuel and oxidizer drains and vent lines, emergency propellant catch tanks, hazardous vapor exhaust, and breathing air support.

6.1.3.2 Payload Support Van

The Payload Support Van (PVan) provides electrical, gas, and communication interfaces between the spacecraft ground support equipment and the spacecraft, initially at the VIF for prelaunch testing, and subsequently at the pad during launch. The Pvan, shown in Figure 6.1.3.2-1, consists of a rail car undercarriage and support container that houses the spacecraft ground support equipment. The PVan provides 23.2 m^2 (250 ft^2) of floor space for spacecraft mechanical, electrical, and support equipment. The PVan also provides power, air conditioning, lighting, and environmental protection.

The PVan provides the electrical interface between the spacecraft ground support equipment and the Atlas V T-0 umbilical that supplies the ground electrical services to the spacecraft. The PVan provides 20-kVA Uninterruptible Power Supply (UPS) power for spacecraft GSE racks. Power receptacles provided include eight 120-Vac, 15-A receptacles; two 120-Vac, 30-A receptacles; and four 120/208-Vac, three-phase, four-pole, five-wire receptacles. The PVan provides the supply and interface through the T-0 umbilical for Grade B or Grade C GN_2 at a flow rate up to 14.2 scmh (500 scfm) for spacecraft instrument purge.

The Atlas V communication system provides spacecraft communication connectivity from spacecraft ground support equipment in the PVan to the Atlas V fiber-optic network. The communication network provides interfaces to the spacecraft remote command and control station located at the Atlas Spaceflight Operations Center or Astrotech as illustrated in Figure 6.1.3.2-2. Additional remote spacecraft processing sites may also access spacecraft data by precoordinating with the Range for connectivity to the Atlas Spaceflight Operations Center. Spacecraft RF communications are routed from the PLF RF window or reradiating antenna to the PVan and then through the fiber-optic network. Options are also available for spacecraft to radiate directly from the RF window to the PPF. This connection is available while in the VIF or at the pad. During the move from the VIF to the pad, the spacecraft RF signal is available at the PVan for local spacecraft monitoring and recording only. The communication system provides spacecraft RF uplink and downlink capability at the VIF and at the pad. During flight, spacecraft data can be interleaved with the launch vehicle telemetry stream.

The PVan is air-conditioned to maintain 20.6-25 °C (69-77 °F) and 35-70% relative humidity, assuming a spacecraft support equipment heat load of 21,000 Btu per hour. The PVan provides interior lighting of 50 fc in the personnel work area. The PVan limits dynamic loads to 1.5 g during transportation. When the PVan is housed within the Pad Equipment Building (PEB), spacecraft support equipment is protected from the launch induced environment, including overpressure, acoustics (<110 db), and thermal.

The PVan can be staffed during operations at the VIF, during transit to the pad, and on pad until final pad clear operations. PVan reconfiguration for each launch consists of transporting the PVan from the pad to the VIF, removing the previous user’s equipment, performing baseline electrical checkout, and installing the spacecraft support equipment for the current user.

Table 6.1.3.1-1: Vertical Integration Facility Spacecraft Accommodations

Accommodation	Description	Qty
VIF Level 5 (Spacecraft Mate)		
Critical Power	120V, 20A, 60 Hz, 1Ø, 3W, 2P, Explosion Proof	3
	120/208V, 30A, 60 Hz, 3Ø, 5W, 4P, Explosion Proof	1
	120/208V, 60A, 60 Hz, 3Ø, 5W, 4P, Explosion Proof	1
Shop Air Outlets	GN_2 Service	2
Fuel & Oxidizer Lines	Emergency Propellant Detanking	1 Set
LMAO/Customer Work Area	Desk/Phone/User Interface	2
User Interface	Hazardous, Fiber Optic	2
Telephone		7
Facility Ground Plate		2
Technical Ground Plate		3
VIF Level 6 (Spacecraft Access Through PLF Doors)		
Critical Power	120V, 20A, 60 Hz, 1Ø, 3W, 2P, Explosion Proof	2
Shop Air Outlets	GN_2 Service	2
Fuel & Oxidizer Lines	Emergency Propellant Detanking	1 Set
Customer Work Area	Desk/Phone/User Interface	2
User Interface	1 Each Nonhazardous, 2 Each Hazardous, Fiber Optic	3
Telephone		7
Facility Ground Plate		2
Technical Ground Plate		2
VIF Level 7 (Access to Top of PLF)		
Critical Power	120V, 20A, 60 Hz, 1Ø, 3W, 2P, Explosion Proof	4
Shop Air Outlets	GN_2 Service	2
Fuel & Oxidizer Lines	Emergency Propellant Detanking	1 Set
Customer Work Area	Desk/Phone/User Interface	2
User Interface	2 Each Nonhazardous, 2 Each Hazardous, Fiber Optic	4
Telephone		6
Facility Ground Plate		3
Technical Ground Plate		3
Freight Elevator		
Size	Length x Width: 2,438.4 mm x 1,828.8 mm (8 ft x 6 ft)	
Door Clearance	Width x Height: 1,828.8 mm x 2,133.6 mm (6 ft x 6 ft 11 in.)	
Capacity	2,267.9 kg (5,000 lb)	

Figure 6.1.3.2-1: Payload Van

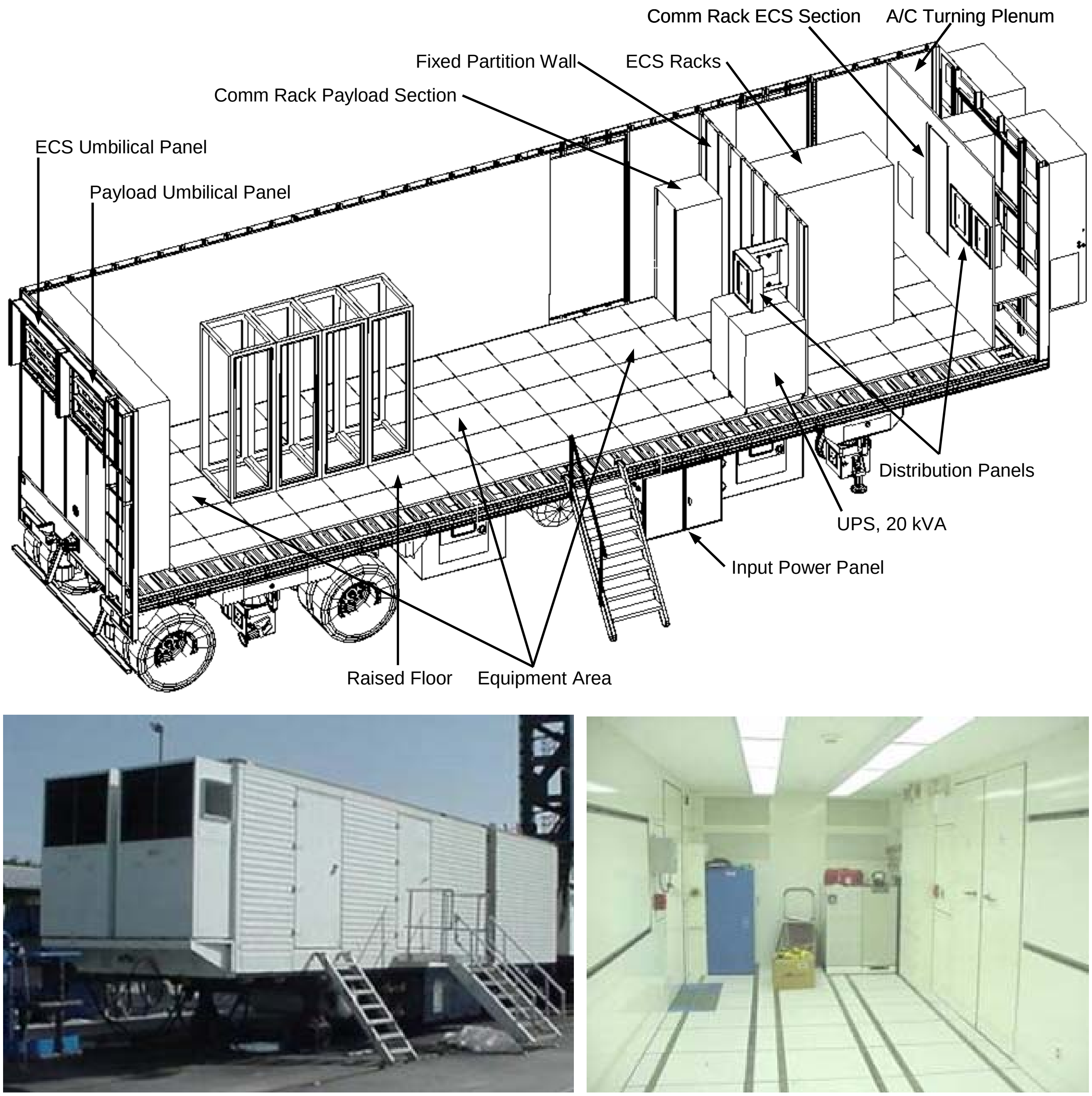

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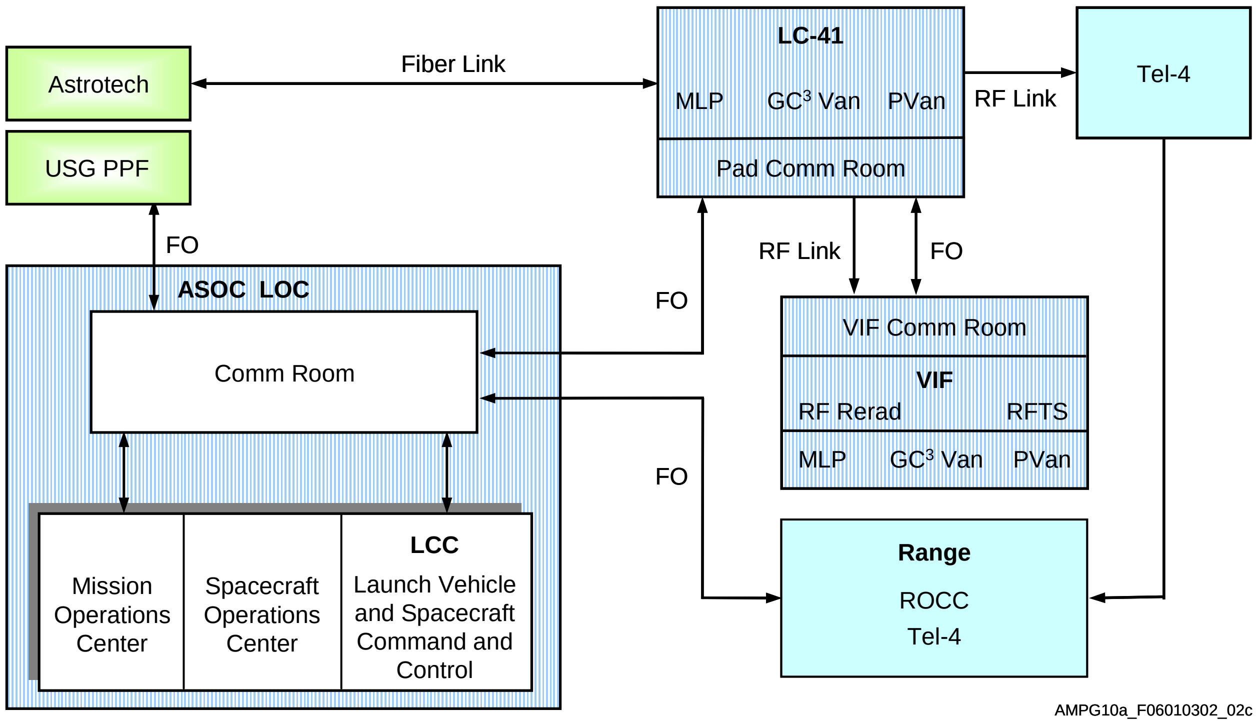
Figure 6.1.3.2-2: Communications Network

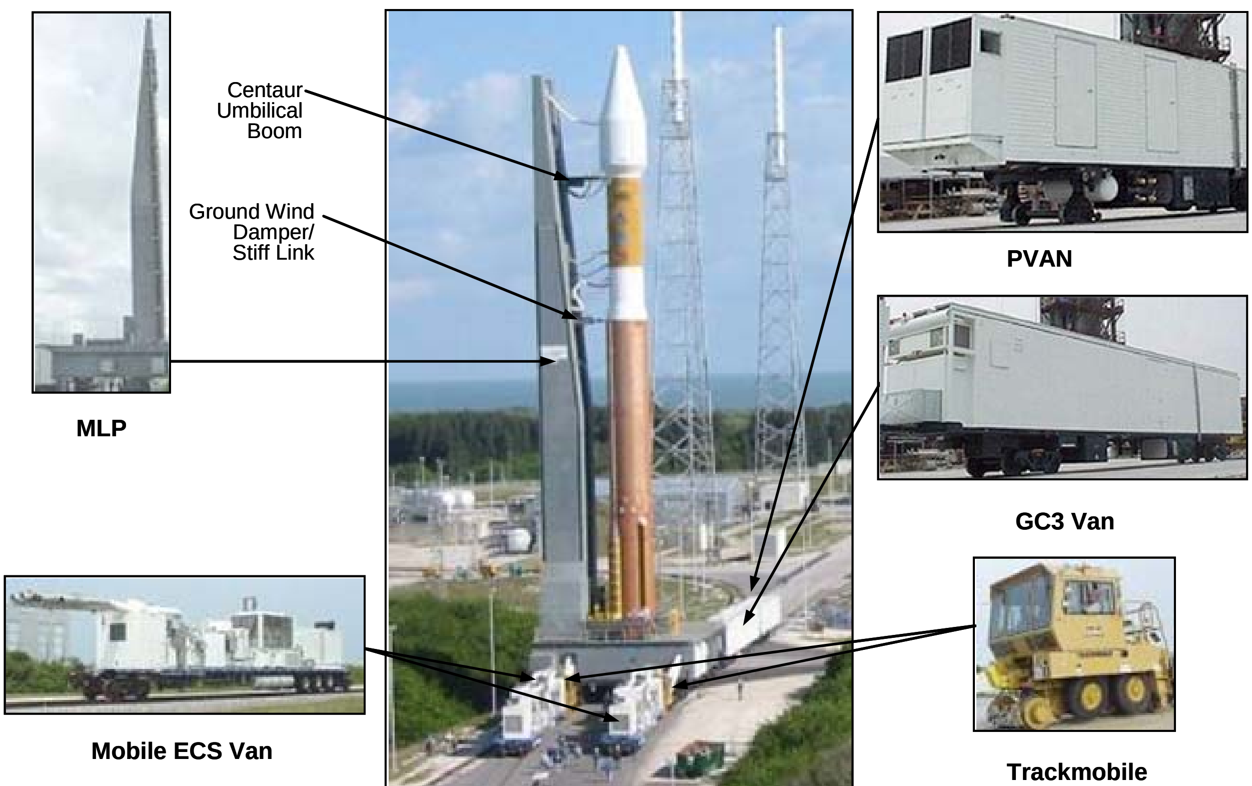


6.1.3.3 Mobile Launch Platform

The Mobile Launch Platform (MLP), shown in Figure 6.1.3.3-1, consists of a structural steel frame capable of supporting the various Atlas V 400 and 500 series launch vehicle configurations. Supported operations include integration of the booster(s), mating of the Centaur and spacecraft in the VIF, transport to the launch pad, launch vehicle fueling, final preparation for launch, thrust hold-down, and release of the launch vehicle at launch. This frame is supported underneath by piers at the VIF and at the launch pad. The frame is rolled to these locations using four 227,000-kg (250-ton) rail cars equipped with a hydraulic jacking system for raising the MLP for movement and lowering onto the piers for stability. The MLP is moved between the VIF and launch pad by two tugs that ride on a rail system. The MLP frame also supports the umbilical mast.

Mobile Launch Platform Umbilical Mast — The Atlas V MLP includes an umbilical mast for electrical, fluids, and gas servicing during final countdown, eliminating the need for an on-pad umbilical tower. All umbilical interfaces are connected and checked out in the VIF before rollout to the pad. The T-0 umbilicals remain connected up to launch. During transit from the VIF to the launch pad, launch vehicle and spacecraft conditioned air is provided using an air conditioning trailer connected to the MLP. The MLP includes common ductwork allowing switching between facility-provided and trailer-provided conditioned air sources without interruption of conditioned air services.

Figure 6.1.3.3-1: Mobile Launch Platform



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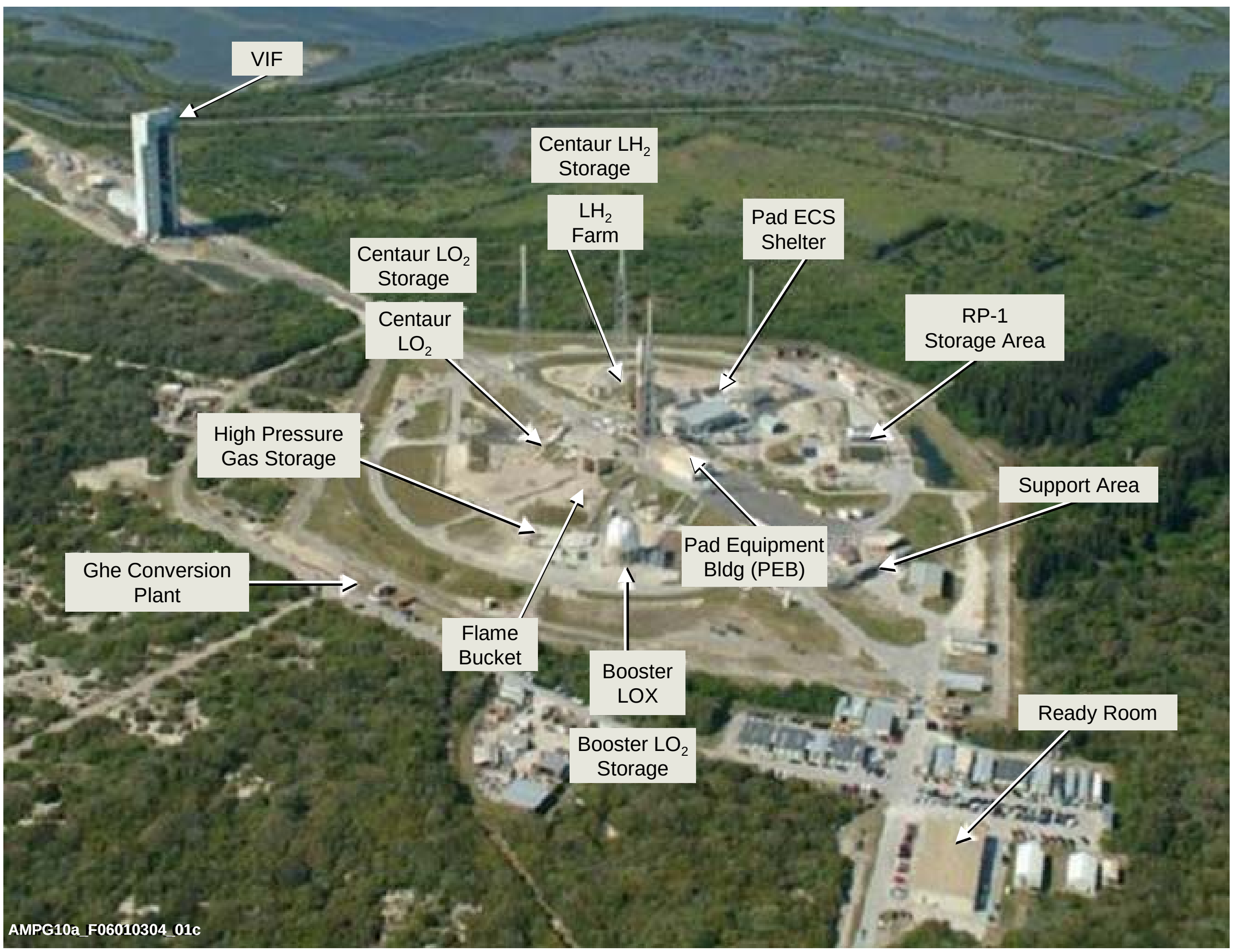
6.1.3.4 Launch Pad

The Atlas V launch pad, shown in Figure 6.1.3.4-1, is located within the LC-41 complex at CCAFS. The launch complex uses a "clean-pad" launch processing approach whereby the launch vehicle is fully integrated off-pad on the MLP in the VIF, and the launch pad is used only for launch day launch vehicle propellant loads and launch countdown. There are no provisions for spacecraft access while the MLP is on the launch pad. (All final spacecraft access activities, including removal of ordnance safe and arm devices, are made in the VIF.) All spacecraft umbilicals needed at the launch pad are flyaway disconnects. Rollback from the launch pad to the VIF can be accomplished in 6 hours if launch vehicle propellants have not been loaded, or within 18 hours if launch vehicle propellants must be detanked.

The launch pad operations continue to provide MLP and PVan interfaces. Payload compartment conditioned air is provided to the same parameters provided in the VIF. Spacecraft conditioned air is switched to GN_2 approximately 1 hour before Centaur cryogenic propellant operations. GN_2 cleanliness, temperature, and flow rate are the same as conditioned air, with maximum dew point of -37 °C (-35 °F).

Figure 6.1.3.4-1: Launch Complex 41



6.1.4 Atlas Spaceflight Operations Center

The Atlas Spaceflight Operations Center (ASOC) is a multifunctional facility supporting launch vehicle hardware receipt and inspection, horizontal testing, and launch control. The ASOC is located approximately 4 miles from the LC-41 complex. The Launch Operations Center (LOC) in the ASOC (Figures 6.1.4-1 and 6.1.4-2) provides spacecraft interfaces for command, control, monitoring, readiness reviews, anomaly resolution, office areas, and day of launch viewing. The LOC design provides maximum flexibility to support varying customer requirements. The main areas of the LOC include the Launch Control Center, the Mission Operations Center, the Spacecraft Operations Center, the Engineering Operations Center and the Customer Support Facility. Entry into the LCC is controlled by a personnel entry door equipped with electronic access system controls.

Launch Control Center — The Launch Control Center (LCC) is the Atlas V launch vehicle command and control center for Lockheed Martin engineering operating the real-time system, monitoring critical data and for launch conductor coordination and control of the launch process.

Spacecraft Operations Center — The Spacecraft Operations Center (SOC) provides four spacecraft customer positions and one Lockheed Martin position. Each position includes access to launch vehicle and spacecraft data, voice, and video. The stadium seating provides excellent viewing of the Launch Control Center (LCC) video wall.

Mission Operations Center — The Mission Operations Center (MOC) provides four spacecraft customer positions combined with the Lockheed Martin launch management team. Each position includes access to launch vehicle data, voice, and video. The stadium seating provides excellent viewing of the LCC video wall.

Engineering Operations Center — The Engineering Operations Center (EOC) provides twenty operator positions for Lockheed Martin personnel. . Each position includes access to launch vehicle data, voice, and video. The stadium seating provides excellent viewing of the LCC video wall.

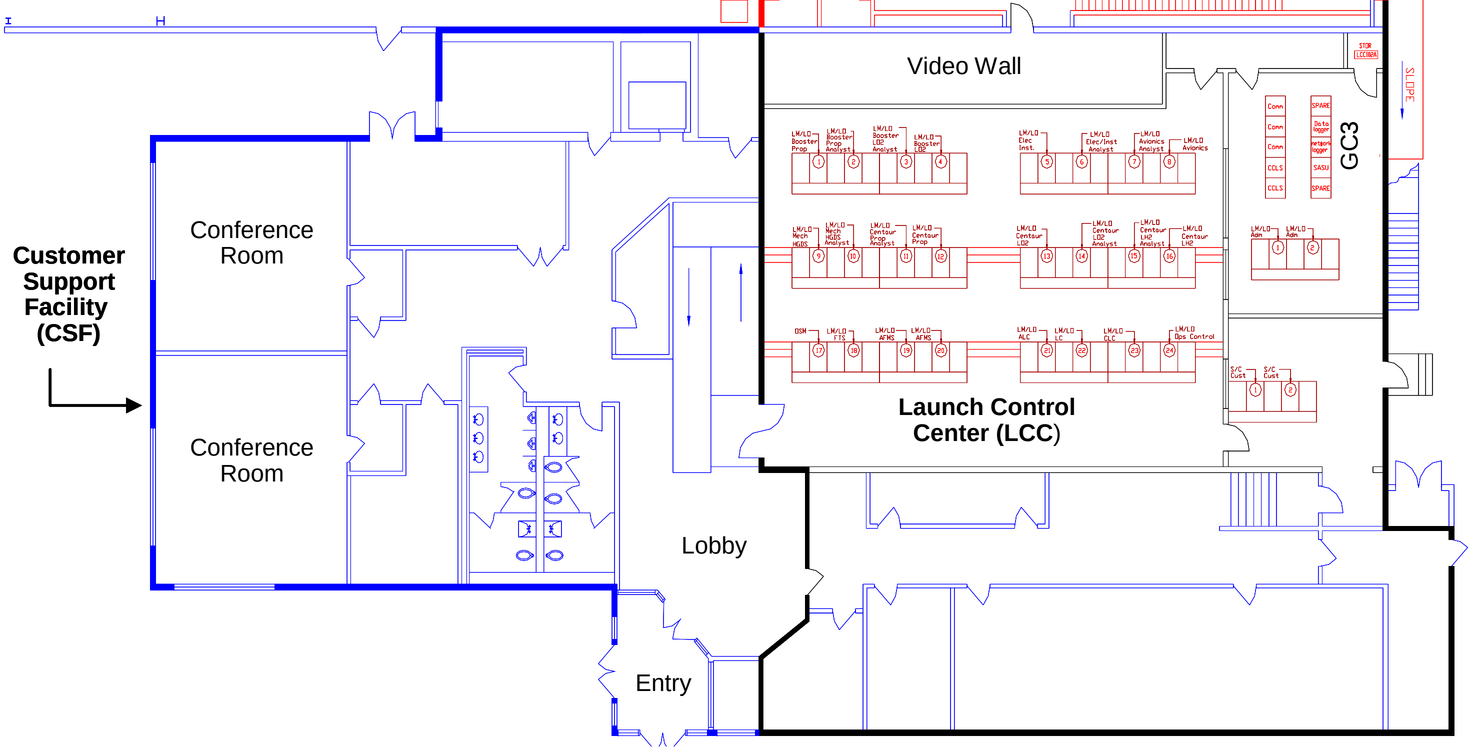
Customer Support Facility — The Customer Support Facility (CSF) provides office space, briefing and anomaly resolution areas, data, voice, video, hospitality area, and day of launch viewing. The CSF includes easy access to other areas within the LOC.

Operations Communication Center — The Operations Communication Center (OCC) is the communications center for all Atlas V operations including networks, data circuits, voice, video and telephone systems.

Mission Support Rooms — The Mission Support Rooms (MSR) provide space for customers and subcontractors use. Accommodations include data, communications, and video display with easy access to the conference rooms and the CSF.

GC3 Support Area – This area contains the ground processing computers and equipment, including system administration consoles and provisions for data recording and archiving.

Figure 6.1.4-1: ASOC Launch Operations Center First Floor



First Floor–LOC

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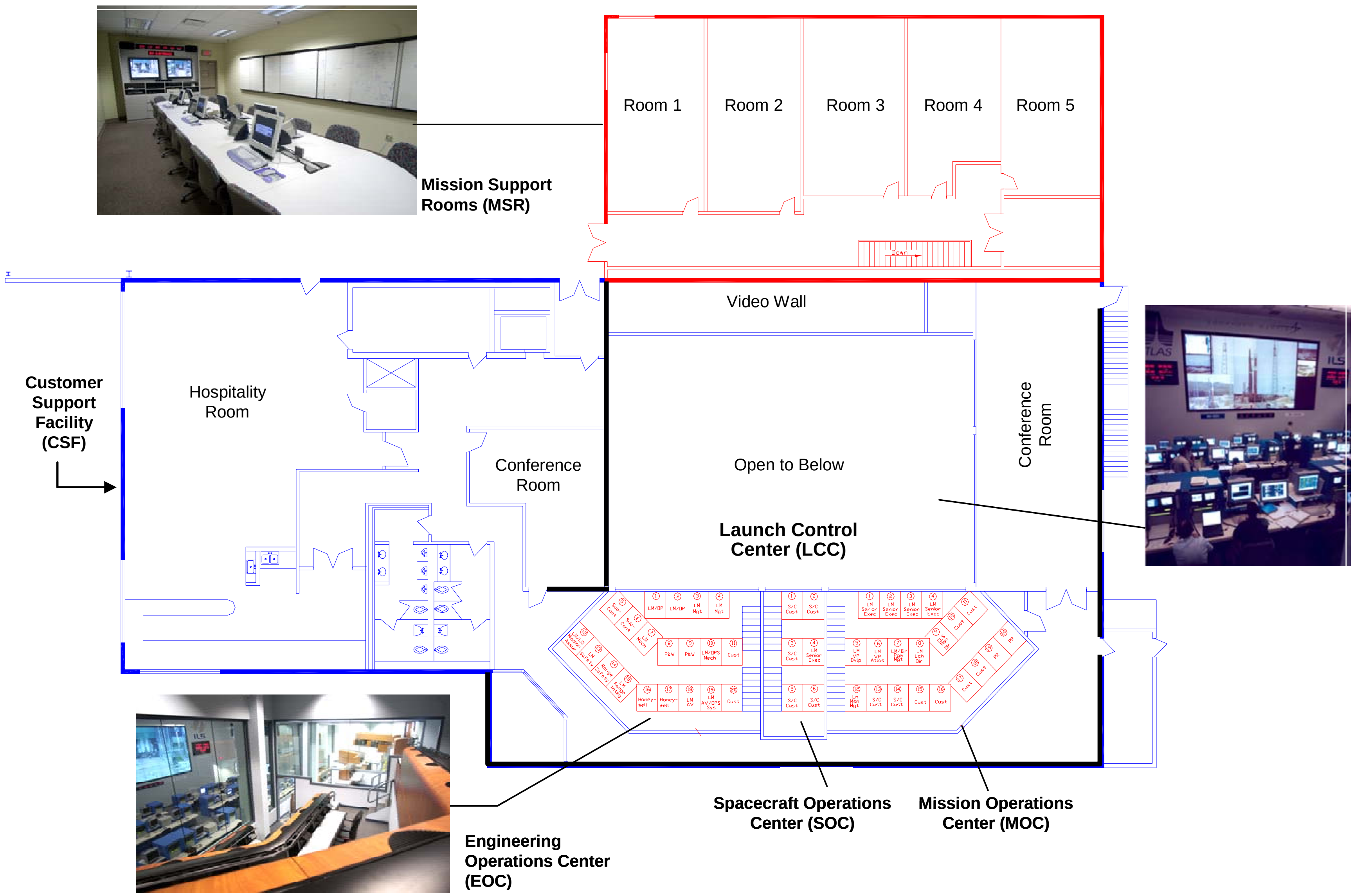
Figure 6.1.4-2: ASOC Launch Operations Center Second Floor



Second-Floor LOC

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6.1.5 Spacecraft Instrumentation Support Facilities

CCAFS 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 known as TEL4. 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 electronic media 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/Merritt Island Launch Area Station — The Goddard Space Flight Center (GSFC)/Merritt Island Launch Area (MILA) station is located on Merritt Island. MILA is the Eastern Range launch area station for NASA’s Ground Satellite Tracking and Data Network (GSTDN) Tracking and Data Relay Satellite System (TDRSS). This system includes satellite ground terminals providing access to worldwide communications. Circuits from MILA to the HPF, PPF, and LC-41 complex are available to support checkout and network testing during prelaunch operations and spacecraft telemetry downlinking during day of 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.

Jet Propulsion Laboratory MIL-7.1 Station — The Jet Propulsion Laboratory (JPL) MIL-7.1 station is co-located 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/Transportable Vehicle Checkout Facility — The Eastern Vehicle Checkout Facility (EVCF)/Transportable Vehicle Checkout Facility (TVCF) is an Air Force Space Control Network (AFSCN) ground station. It provides an S-band interface to AFSCN resources.

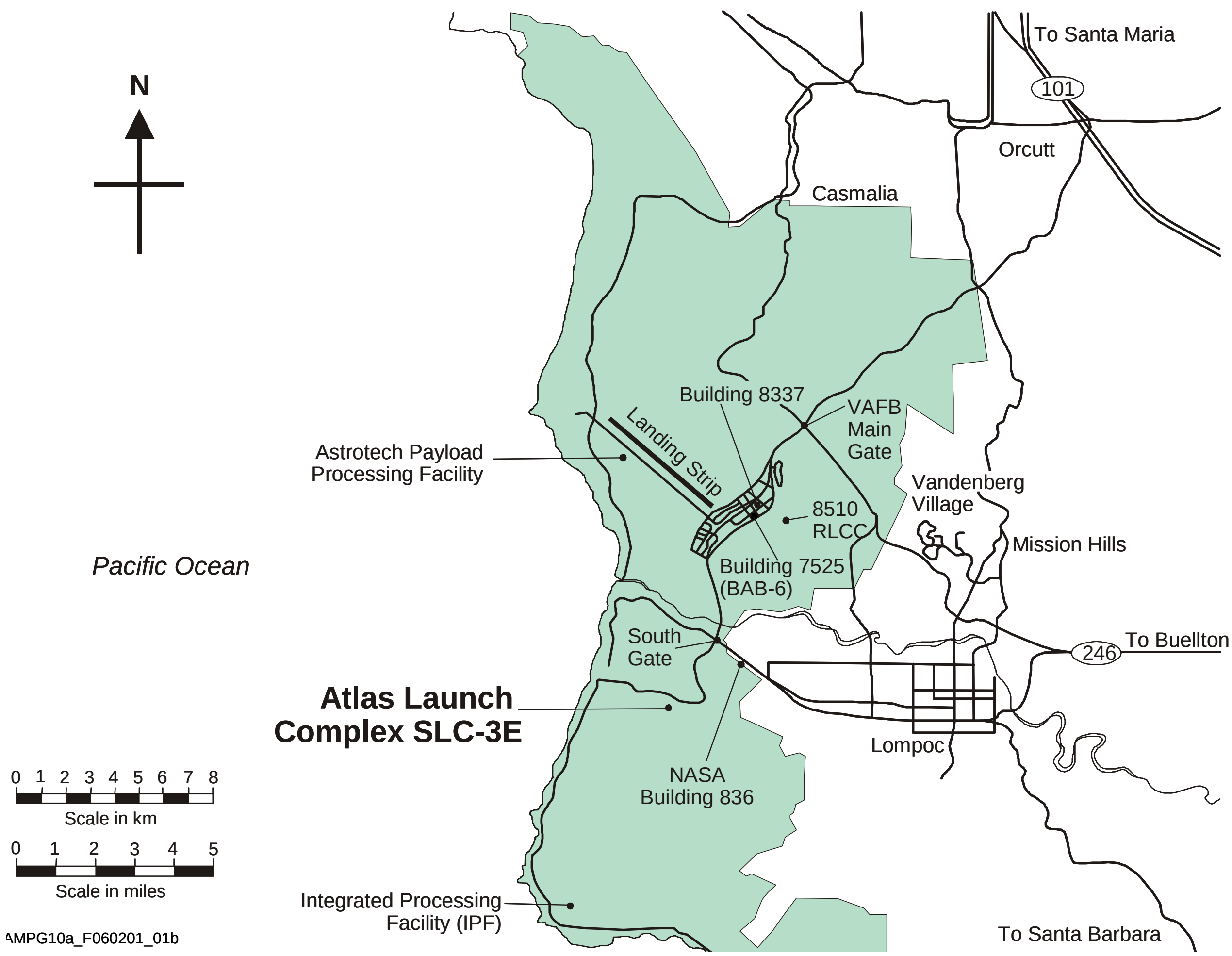
6.2 VAFB ATLAS V LAUNCH FACILITIES

6.2.1 VAFB SPACECRAFT PROCESSING FACILITIES

Spacecraft facilities available to commercial and U.S. government facilities at VAFB include the VAFB Astrotech Payload Processing Facility and the Space Systems International Integrated Processing Facility. Figure 6.2.1-1 illustrates the location of these facilities and the Atlas V Launch Complex SLC-3E.

Government facilities, as outlined in Lockheed Martin/NASA and Lockheed Martin/USAF agreements, are available for use should the commercial facilities not adequately satisfy spacecraft processing requirements. These facilities are described in the following sections.

Figure 6.2.1-1: VAFB Facility Locations



6.2.1.1 Astrotech Facilities

The Astrotech/VAFB commercial Payload Processing Facility, owned and operated by Spacehab Inc. is available for processing Atlas V class civil, government, and commercial spacecraft. The VAFB facility is capable of processing Atlas V 4-meter payload fairings. This facility supports final checkout and encapsulation of spacecraft. Depending on VAFB launch interest and U.S. government spacecraft processing requirements for Atlas/Centaur class spacecraft, the Astrotech/VAFB facility offers compatibility with spacecraft contractor and Lockheed Martin use requirements.

This facility 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.

Astrotech PPF

The Astrotech/VAFB PPF can be used for all spacecraft preparation operations, including liquid propellant transfer, Solid Rocket Booster (SRB) and ordnance installations, and spacecraft encapsulation. The facility is near the VAFB airfield, approximately 12 km (7.5 miles) from SLC-3E. The PPF, Figure 6.2.1.1-1, 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 and customer support office approximately 21.3 m wide x 24.4 m long (70 ft x by 80 ft) is available (Figure 6.2.1.1-2), 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.

Spacecraft Services — A full complement of services can be provided at Astrotech to support spacecraft 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. A diesel generator backs up standard power 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. Transistorized Operational Phone System (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.

Figure 6.2.1.1-1: Astrotech/VAFB Payload Processing Facility Layout

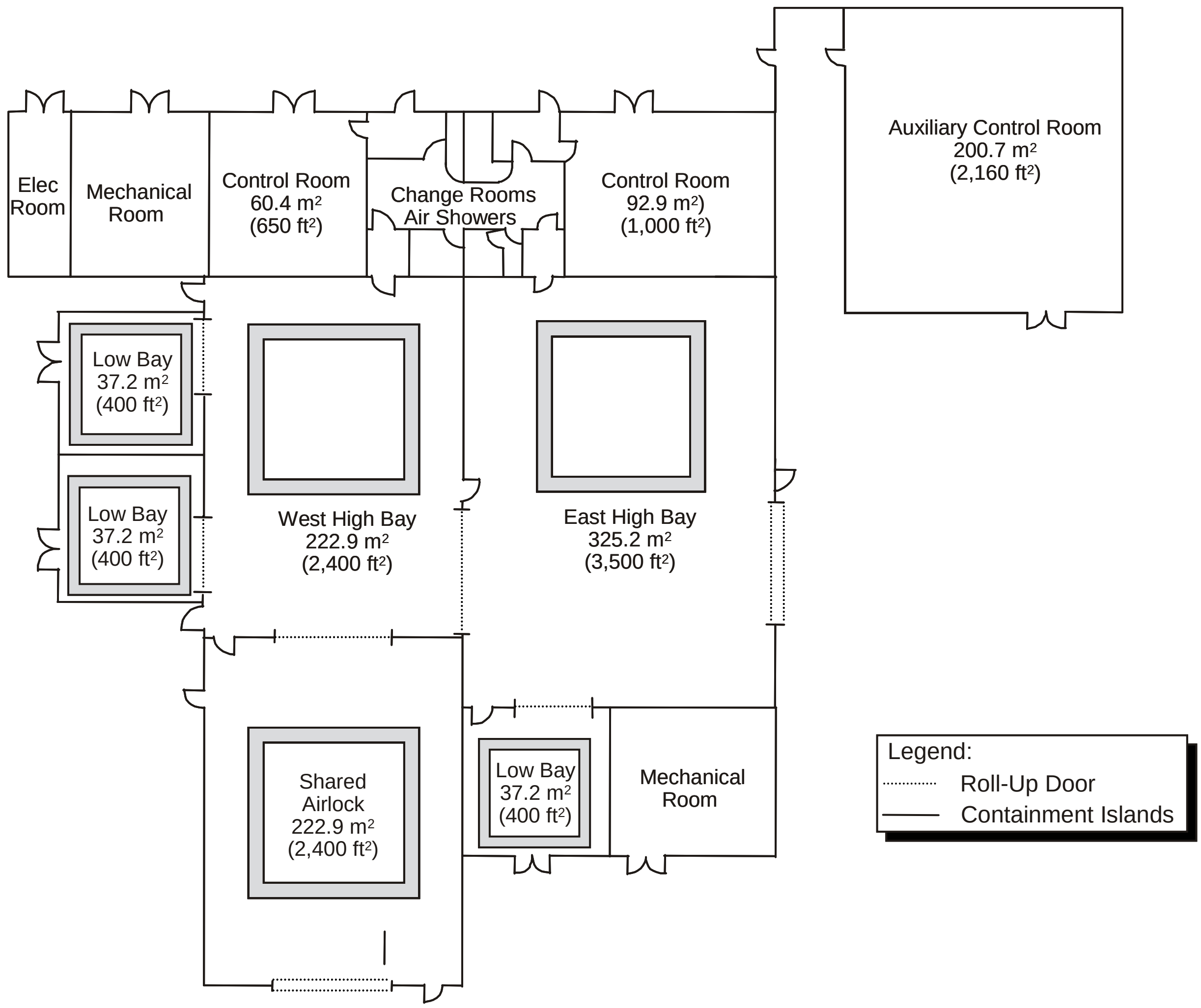


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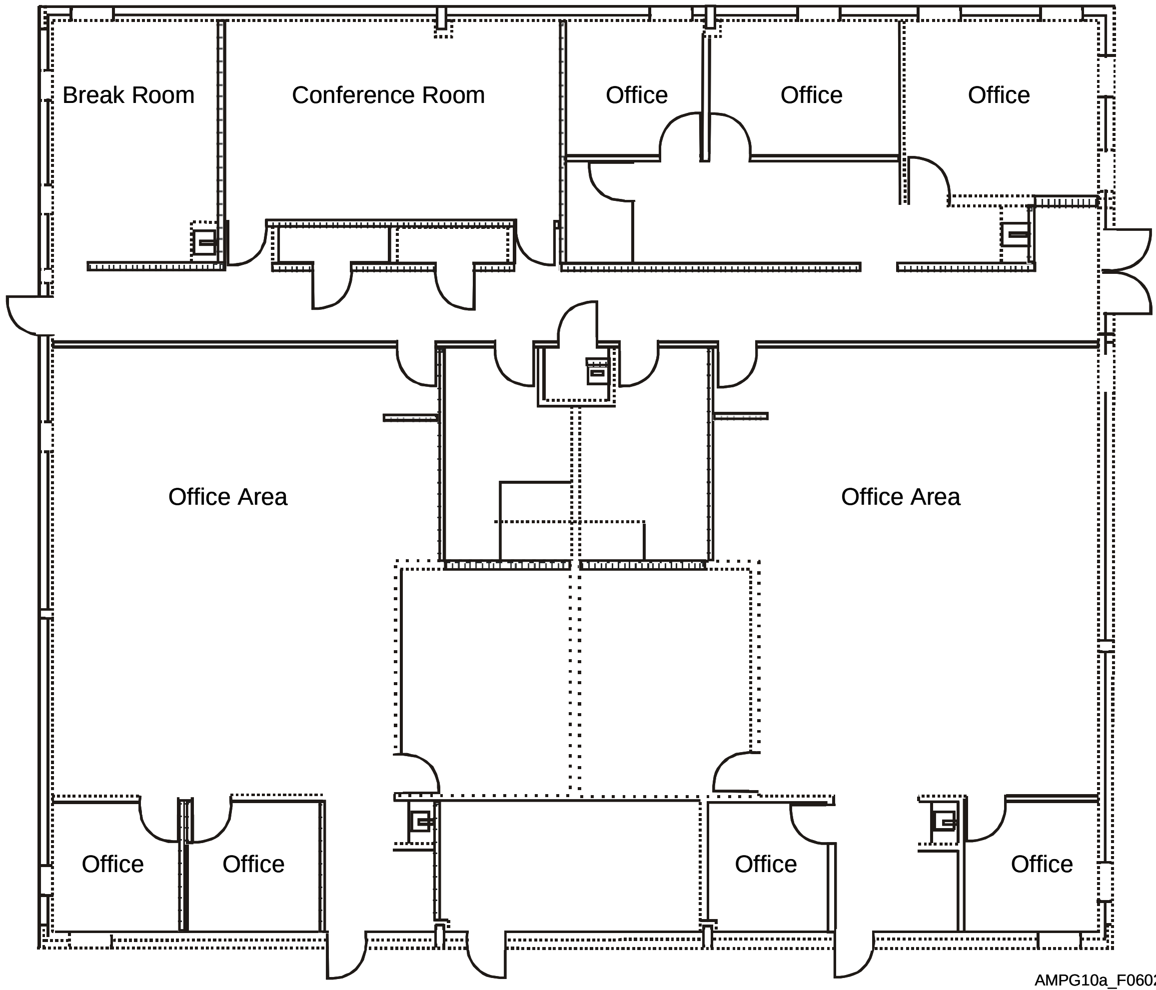
Figure 6.2.1.1-2: Technical Support Building



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 clean room 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 clean room 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 clean rooms to prevent toxic vapor leaks into adjacent areas.

Compressed Air — A stationary, two-stage, rotary-tooth compressor supplies oil-free compressed air for breathing, shop air, and pallet air applications. The 30-horse power 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. The alarm system is designed to allow completely segregated operations in the two processing high bays.

6.2.1.2 Spaceport Systems International Facilities

The Integrated Processing Facility (IPF) owned and operated by Spaceport Systems International (SSI) is available for processing Atlas V class civil and government spacecraft. The IPF is capable of processing both 4-m and 5-m payload fairings. This facility supports final checkout and encapsulation of spacecraft. Depending on VAFB launch interest and U.S. government spacecraft processing requirements for Atlas/Centaur class spacecraft, the IPF facility offers compatibility with spacecraft contractor and Lockheed Martin use requirements.

The IPF (Figure 6.2.1.2-1) contains hazardous processing cells, storage capabilities, and technical support areas. The facility floor plan is described in the following sections. The IPF complies fully with all applicable federal, state, regional, and local statutes, ordinances, rules, and regulations relating to safety and environmental requirements.

The IPF can be used for all spacecraft preparation operations, including liquid propellant transfer, SRB and ordnance installations, and spacecraft encapsulation. The facility is located on south VAFB approximately 11 km (6.8 miles) from SLC-3E. The PPF contains the following as shown in Figures 6.2.1.2-2 and 6.2.1.2-3:

1. One Air Lock
2. One Highbay
3. Three Checkout and Processing Cells

Figure 6.2.1.2-1: Integrated Processing Facility

Figure 6.2.1.2-2: IPF Floor Plan

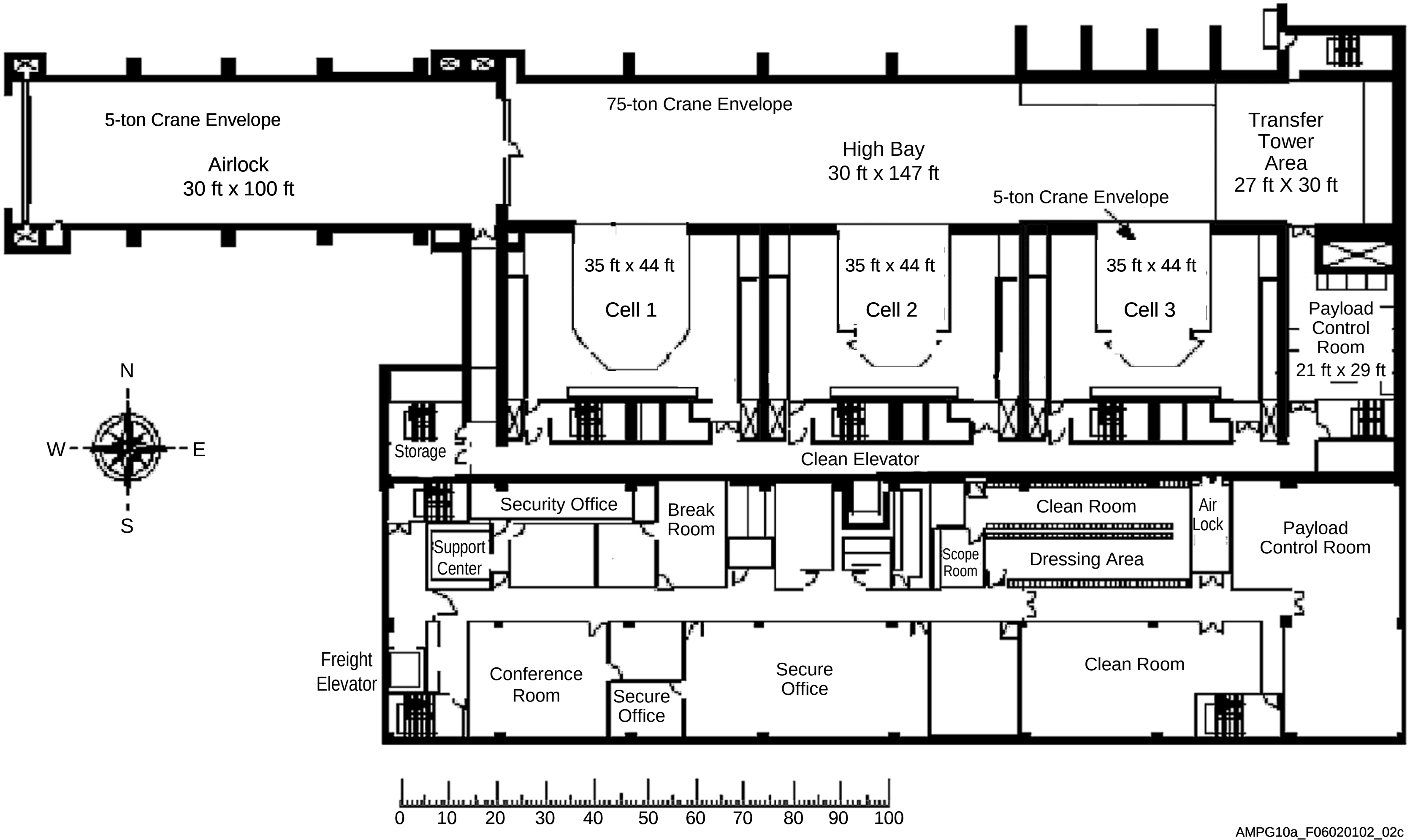
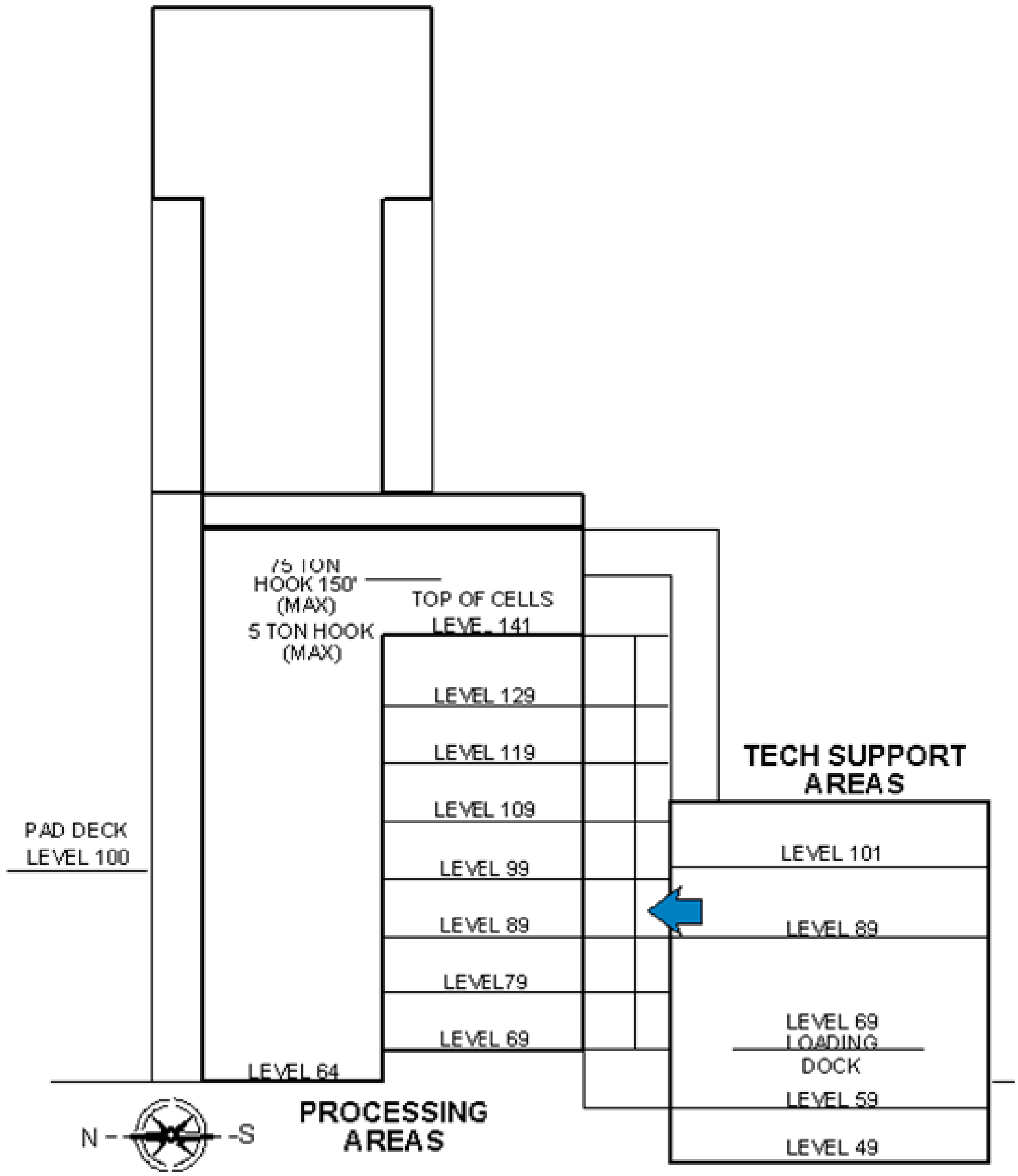


Figure 6.2.1.2-3: IPF Side View



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4. Control Room

5. Common and Technical Support Areas

Airlock — As a buffer area for operations between the external environment and the main processing areas, the IPF airlock has significant processing capability of its own, including:

- A 7.3 m wide x 8.5 m high (24 ft x 28 ft, 1in) door between the Airlock and the Highbay
- 11.8 m x 39.4 m (30 ft x 100 ft) working area
- Two 4.5 tonne (5 ton) overhead cranes with a maximum hook height of 14 m (35 ft, 5 in)
- 125 VAC utility power
- Facility ground
- Explosion proof fixtures
- Shop air
- Class 100,000 clean (10,000 functional)
- Central vacuum system
- CCTV capability
- OVS capability
- Telephones
- Single and multi-mode fiber optic interface (future)

Highbay — In addition to providing an access area to the Payload Checkout Cells, highbay capacity allows use for full processing of spacecraft and/or booster components.

- Class 100,000 clean (10,000 functional)
- Access to the Highbay is through the 9.5 m (24 ft) wide x 11 m (28 ft) door from the Airlock.
- Work area is 11.8 m x 57.9 m (30 ft x 124 ft, 7 in)
- 68 tonne (75 ton) bridge crane. The hook height in the highbay is 34 m (8 6ft, 4 in).
- Shop Air
- Central Vacuum System
- Explosion proof fixtures
- 125 VAC utility power, 120/208 VAC tech power
- Facility Ground
- CCTV capability

- Telephones
- The Airlock and the Highbay are on Level 64

Payload Processing Cells — Three 10,000 clean 13.8 m x 17.3 m (35 ft x 44 ft) Payload Checkout Cells serviced by a 68 tonne (75 ton) bridge crane with 32 m (81 ft, 4 in) hook height. Each cell also has 4.5 tonne (5 ton) crane support with hook height of 28.3 m (71 ft, 11 in). Access to each cell is through doors from the Highbay with a total horizontal opening of 8.4 m (21 ft, 2 in).

- GN_2 and GHe are supplied by K-bottles or tube banks,
- 60 tonne (75 ton) crane with micro drive services all cells and the Highbay,
- 4.5 tonne (5 ton) crane with micro drive in each cell,
- Class 100,000 clean (10,000 functional),
- In addition to the cell floor at Level 69, there are six platform levels in each of the three processing cells – 24.1 m (79 ft), 27.1 m (89 ft), 30.2 m (99 ft), 33.2 m (109 ft), 36.3 m (119 ft) and 39.3 m (129 ft).
- Finger platforms for adjustable diameter segments (not in Cell 1)
- 125, 120/208 and 480 VAC power
- Facility and technical grounds
- Cell separators for 2 levels
- Elevator/corridor access for GSE; 1.2m x 1.8m x 2.1m (4ft x 6ft x 7ft) envelope at 1,493 kg (4,000 lbs.) max
- Shop Air
- Central Vacuum System
- Explosion-proof fixtures
- CCTV capability (Cells 2 & 3)
- OVS capability
- Telephones
- Single & multi-mode fiber optic interface (Cells 2 & 3)
- Fuel vent

Control Room — When configured as a Payload Control Room, SSI Spaceport control functions are located in one of several other IPF locations, and the Facility Monitoring and Control Workstation is relocated accordingly. There are several similar rooms located above Room 7903.

Technical Support Rooms — The facility provides the following common and technical support rooms.

Break Room — Room 8925 is available for use by all personnel in the IPF. It contains a microwave oven, refrigerator, and a selection of snacks for sale.

Conference Room — The Conference Room is available for use by all IPF users on an "as available" basis.

Technical Support Room — Room 8914 at 104.4 m^2 (1124 ft^2) is one of several technical support office areas, which can be outfitted to customer requirements. Room 10102 at 107.4 m^2 (1156 ft^2), room 10104 at 150.6m^2 (1621 ft^2) and room 10106 at 148.6 m^2 (1600 ft^2) are similar.

6.2.2 Spacecraft Encapsulation and Transportation to SLC-3E

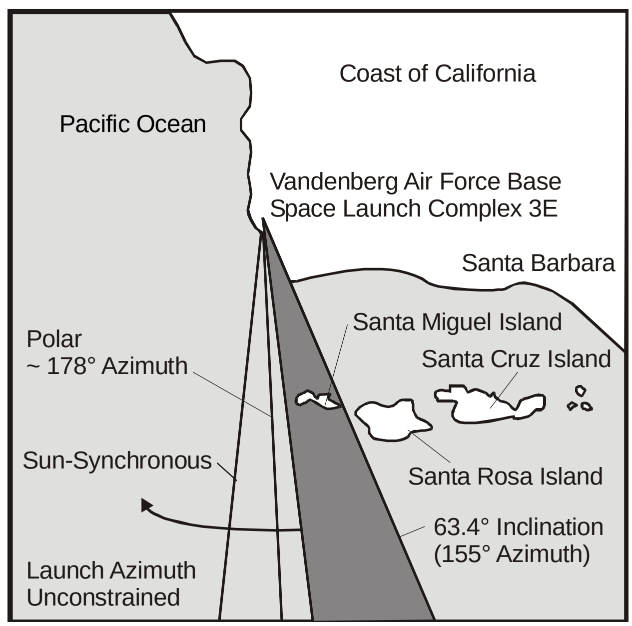
During the spacecraft processing campaign, Lockheed Martin will require use of the PPF/IPF for approximately 30 days to receive and verify cleanliness of the PLF and to encapsulate the spacecraft.

After spacecraft encapsulation, Lockheed Martin will transport the encapsulated spacecraft to SLC-3E and mate the encapsulated spacecraft assembly to the launch vehicle. Transportation hardware and procedures will be similar to those used at the East Coast LC-41 launch complex. Post-mate spacecraft testing can be performed from GSE located on the Mobile Service Tower, in the LSB payload user’s room, or through connectivity to the VAFB Fiber-Optics Transmission System (FOTS) from offsite locations.

6.2.3 ATLAS V Space Launch Complex-3E

Space Launch Complex-3E (SLC-3E) at VAFB in California has supported the launch of Atlas vehicles since the 1970s. In 1992, the USAF contracted with Lockheed Martin to convert the inactive SLC-3E site to support the launch of Atlas/Centaur to orbits not attainable from the East Coast CCAFS launch site (Figure 6.2.3-1). The Initial Operational Capability (IOC) of SLC-3E for Atlas IIAS was late 1997. The launch complex is being modified to accommodate Atlas V launch vehicles with a scheduled IOC of May 2005. This section discusses facilities that are or will be available to support spacecraft and Atlas V launch vehicle integration and launch.

Figure 6.2.3-1: SLC-3E Available Launch Azimuths



SLC-3E 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, SLC-3E, Building 8510 and Building 7525 are shown in Figure 6.2.1-1. Major facilities at SLC-3E include the Mobile Service Tower, Umbilical Tower, Launch Support Building and a Launch Operations Building.

The Remote Launch Control Center is 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 from the South Vandenberg nitrogen generation plant operated by the USAF.

6.2.3.1 Mobile Service Tower

The Mobile Service Tower (MST) is a multilevel, movable, totally enclosed steel-braced frame structure for servicing of launch vehicles and spacecraft (Figure 6.2.3.1-1). 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 54.4 tonne (60 ton) overhead bridge crane on Level 21 to move outside the MST for erection and mating of the Atlas V, Centaur and spacecraft. The various MST levels provide access to the Atlas V booster, the Centaur, and the spacecraft. It also provides a lighted, weather-protected work area for erection, mating, and checkout of the flight vehicle.

In addition to external siding, the MST incorporates a Payload Controlled Area (PCA) around the vehicle on Levels 14 through 17 to protect the spacecraft and spacecraft dedicated GSE located in the MST.

6.2.3.2 Umbilical Tower

The Umbilical Tower (UT) is a steel structure with 16 levels. The UT supports a draped umbilical and a ground wind damper. The draped umbilical is used to supply conditioned air to the spacecraft 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 Launch Services Building pad deck to appropriate distribution points.

6.2.3.3 Launch Services Building

The Launch Services Building is a reinforced concrete and steel structure that is the platform on which the Atlas V family of vehicles is assembled, tested, and launched. The top of the LSB, or LSB pad deck, provides support for the Atlas V launcher and the MST while in the service position. The LSB pad deck is also 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.

LSB equipment is 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 (Room 219) for locating spacecraft dedicated GSE.

6.2.3.4 Launch Operations Building

The Launch Operations Building (LOB) is a 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 Remote Launch Control Center. 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.

6.2.3.5 SLC-3E Spacecraft Support Services

Electrical interfaces exist in the LSB payload user’s room and the MST on Levels 14 and 17. 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 services are supplied as part of the facility on MST Levels 14, 15, and 16. 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 14, and 15. Contingency offload of spacecraft propellants is supported via the facility spacecraft propellant deservicing system. Propellant deservicing interfaces exist on MST Levels 14, and 15 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 on MST Levels

14, and 15, 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 (Room 219) is provided in the LSB, Figure 6.2.3.5-1, to support spacecraft testing. The user's room is electrically interconnected to the T-0 umbilical. Support cables are terminated at a connector interface panel for connection to the user's GSE located in Room 219. Capability also exists to connect the user's room to the FOTS for connectivity to offsite locations. Room 219 must be evacuated at approximately 3 hours prior to launch.

6.2.4 Remote Launch Control Center

The Remote Launch Control Center (RLCC) located in Building 8510, as shown in Figure 6.2.4-1, 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 Computer Controlled Launch Set (CCLS), launch vehicle and AGE, and, through the Safe/Arm and Securing Unit, control of safety-critical functions independent of the CCLS and RLCC-to-SLC-3E AGE interface links.

Similar to the LLC at the ASOC on the East Coast, the RLCC houses Atlas LCC, MOC, MOC annex (similar to SOC), EOC, MSRs, and GC3 functions in support of Atlas V processing and test operations.

Figure 6.2.3.1-1: SLC-3E Atlas V to MST Relationship

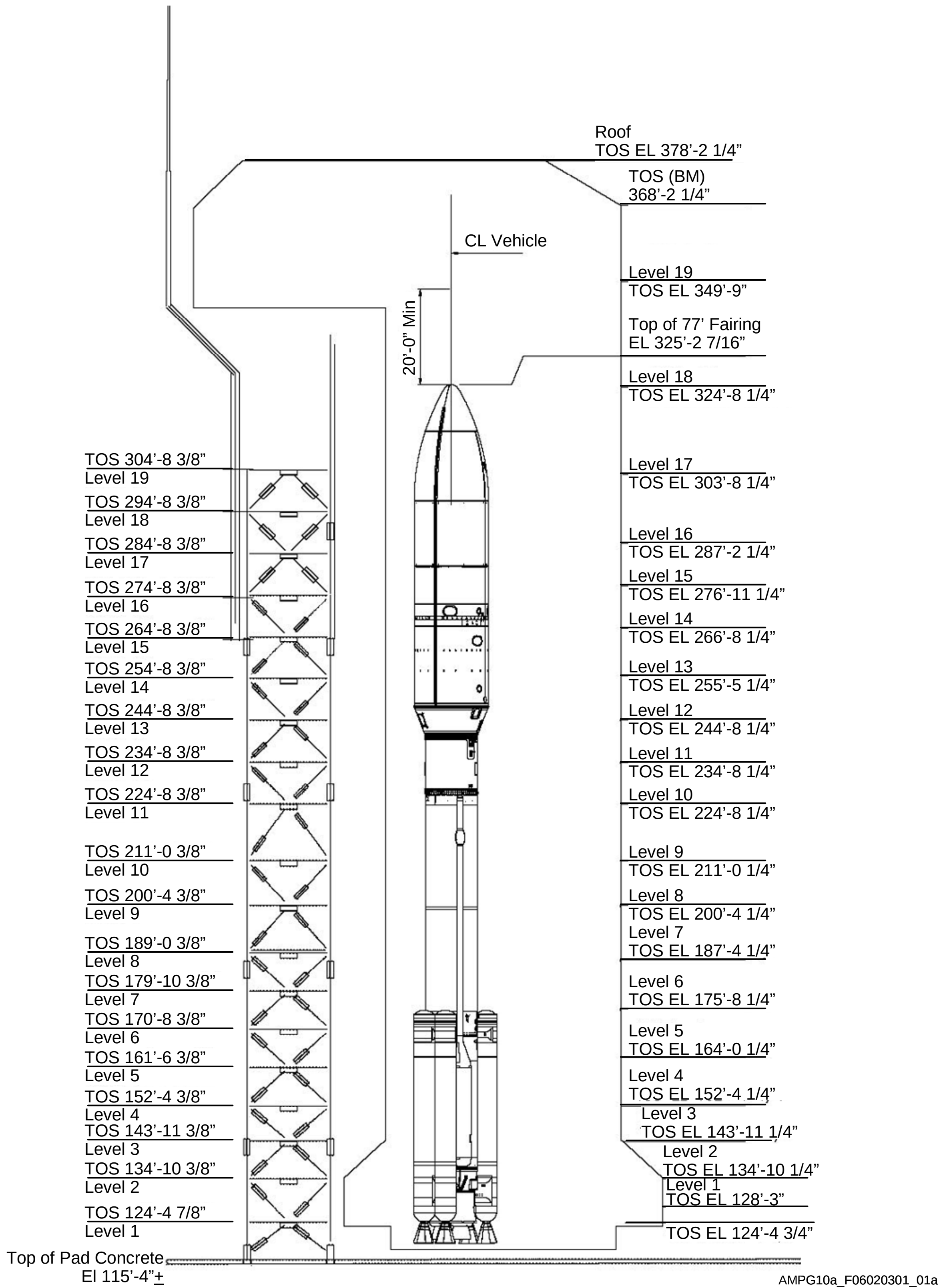


Figure 6.2.3.5-1: Launch Services Building Spacecraft User's Room 219

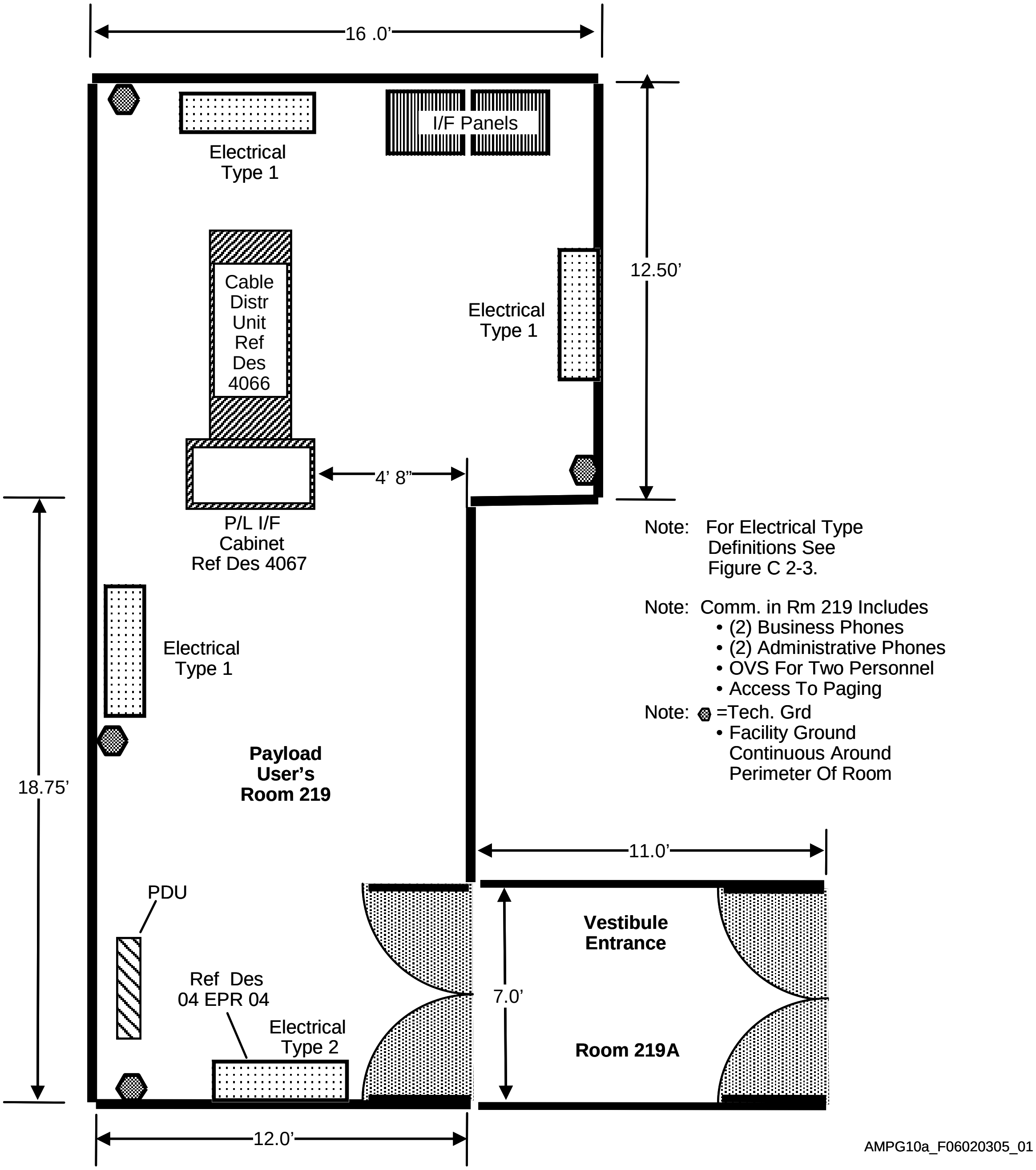
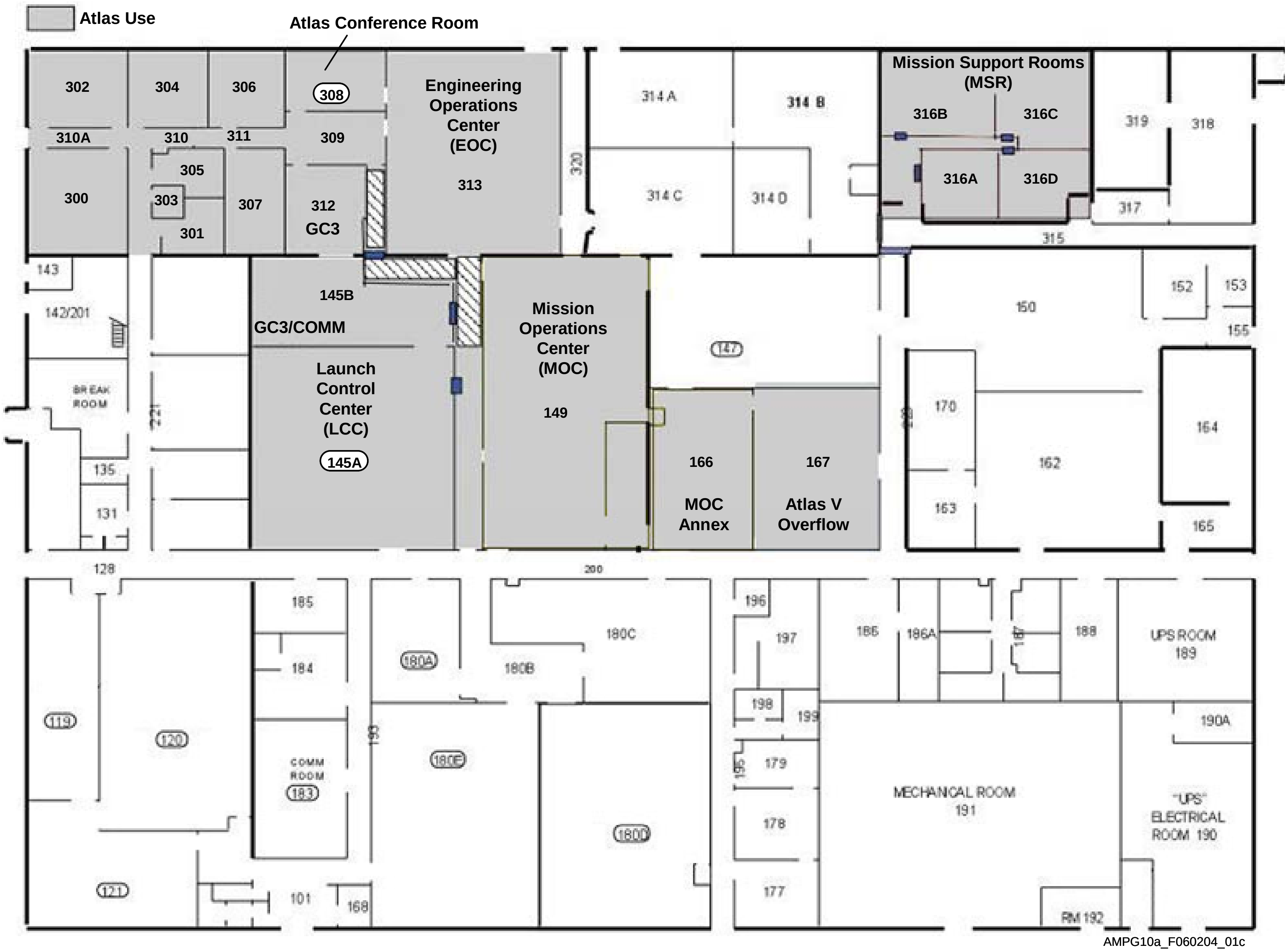


Figure 6.2.4-1: Atlas RLCC Area Within Building 8510

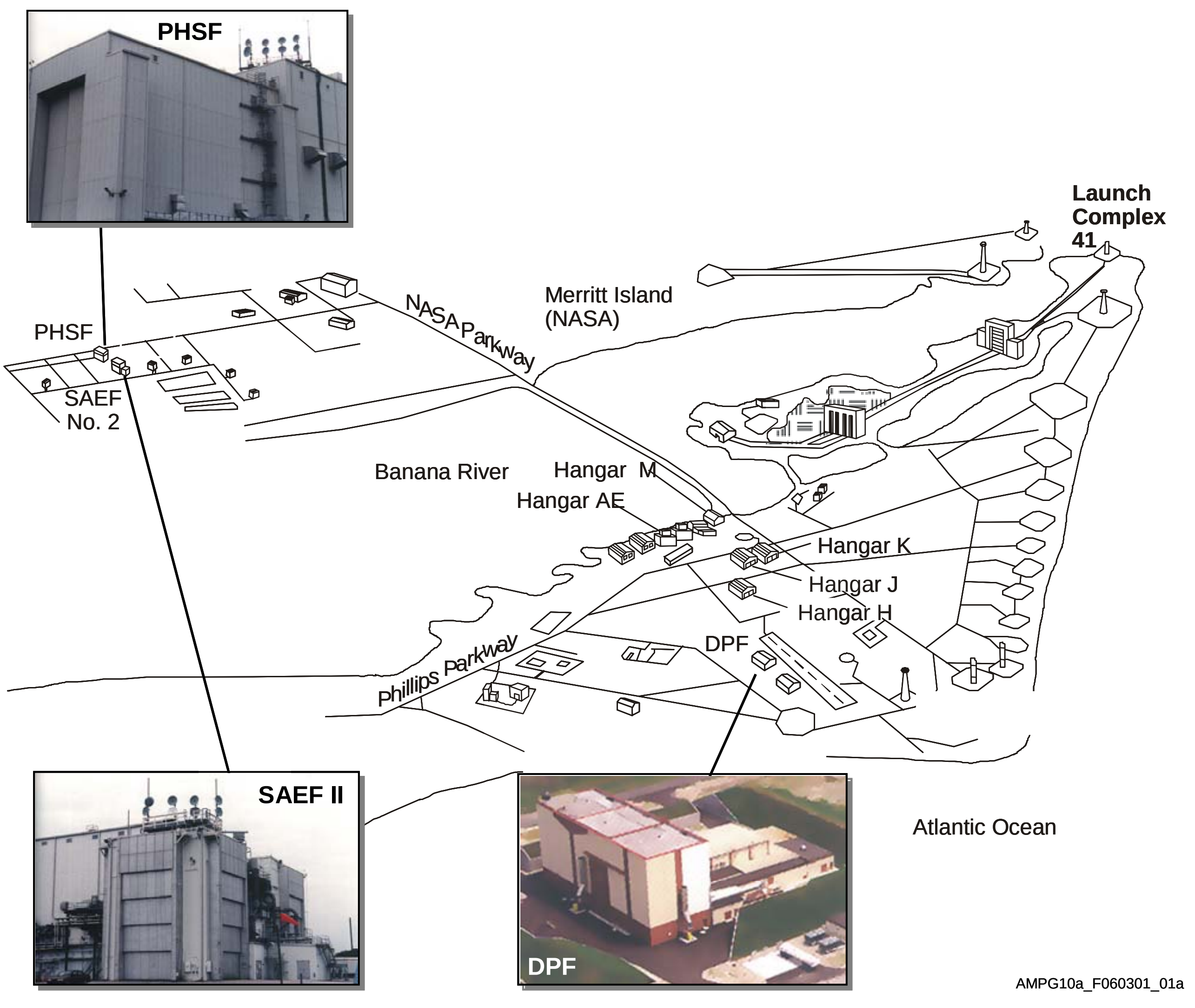


6.3 ADDITIONAL EAST COAST AND WEST COAST FACILITIES

Lockheed Martin has formal agreements with the United States Air Force, the National Aeronautics and Space Administration, and Spacehab Inc. for use of spacecraft and launch vehicle processing facilities at and near Atlas V launch sites at Cape Canaveral Air Force Station, Florida. Similar agreements for sites at Vandenberg Air Force Base in California have been implemented.

6.3.1 Additional CCAFS Processing Facilities

Figure 6.3.1-1 CCAFS Spacecraft Processing Facilities



6.3.1.1 NASA Facilities

6.3.1.1.1 Spacecraft Assembly and Encapsulation Facility No. 2 PPF

Spacecraft Assembly and Encapsulation Facility (SAEF) 2 (Figure 6.3.1.1.1-1) is a NASA facility located southeast of the KSC industrial area. Details of the SAEF 2 facility are listed in Table 6.3.1.1.1-1. It is the larger of the alternate NASA hazardous processing facilities. It features:

1. One high bay,
2. Two low bays,
3. One test cell,
4. Two control rooms.

6.3.1.1.2 Payload Hazardous Servicing Facility

The Payload Hazardous Servicing Facility (PHSF) is a NASA facility located southeast of the KSC industrial area (next to SAEF 2). Additional features of the PHSF Service Building are described in Table 6.3.1.1.2-1.

6.3.1.2 USAF Facilities

6.3.1.2.1 Defense System Communication Satellite Processing Facility

The Defense System Communication Satellite (DSCS) Processing Facility (DPF) is a USAF facility accommodating hazardous and nonhazardous spacecraft processing operations. It provides an area in which to process and encapsulate spacecraft off pad. Figure 6.3.1.2.1-1 is an overview of the DPF site. The facility was designed to accommodate a DSCS III class spacecraft consisting of a DSCS III spacecraft and integrated apogee boost subsystem.

Table 6.3.1.1.2-1: NASA PHSF Building - Hazardous Processing Facility

Service Bay		
Floor Space	18.4 x 32.6 m	60 x 107 ft
Ceiling Height	28.9 m	94 ft 10 in.
Door Dimensions	10.8 x 22.9 m	35 x 75 ft
Crane Capacity	45.40 tonne	50 ton
Hook Height	25.5 m	83 ft 6 in.
Airlock		
Floor Space	15.3 x 25.9 m	50 x 80 ft
Ceiling Height	27.4 m	89 ft 10 in.
Door Dimensions	10.8 x 22.9 m	35 x 75 ft
Crane Capacity	13.60 tonne	15 ton
Hook Height	22.9 m	75 ft
Equipment Airlock		
Usable Space	4.1 x 8.0 m	14 x 26 ft
Ceiling Height	3.2 m	10 ft 4 in.
Door Dimensions	3.0 x 3.0 m	10 x 10 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	

The facility can accommodate 9,000 kg (20,000 lb) of bipropellant and/or 9,000 kg (20,000 lb) of SRBs. 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 15.2 x 15.2 x 16.8-m high (50 x 50 x 55-ft high). The two cells in the HPF have been assigned the following functions:

1. East Bay — Prelaunch processing of the PLF and encapsulation of the spacecraft within the PLF. It can also be used as a fueling cell to assemble SRBs.

2. West Bay — Bipropellant loading cell that can also be used to assemble SRBs. There is no overhead crane in this room.

3. Main Bay — Intended for nonhazardous electrical and mechanical operations and integration of spacecraft elements before fueling. Leak testing and ordnance installation may be accomplished with approval from Range Safety. The main bay is a Class 100,000 clean room. Room environment is typically maintainable at 21.1±2.8°C (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 tonne (5 ton) crane with a hook height of 6.1 m (20 ft).

6.3.2 VAFB Facilities

6.3.2.1 Other Spacecraft Processing Facilities

Building 2520 PPF which is owned and operated by the USAF is capable of supporting all spacecraft preparation operations, including liquid propellant transfer, SRB and ordnance installations, and 4-m PLF encapsulation operations. This facilities availability is subject to conditions as outlined in Lockheed Martin/NASA and Lockheed Martin/USAF agreements

Table 6.3.1.1.1-1 NASA SAEF 2 Hazardous Processing Facility

Control Rooms (2)		
Floor Space	9.14 x 10.97 m	30 x 36 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	49 x 99 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 (Each Bay)	Bridge	
Crane Capacity	9.072 tonne	10 ton
Low Bays (2)		
Floor Size (No. 1)	5.79 x 21.95 m	19 x 72 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)	5.79 x 8.23 m	19 x 27 ft
Floor Area (No. 2)	47.7 m^2	513 ft^2
Test Cell		
Floor Size	11.28x11.28 m	37 x 37 ft
Floor Area	127.2 m^2	1,369 ft^2
Clear Ceiling Height	15.85 m	52 ft
Door Size	6.7 x 12.2 m	22 x 40(h) ft

Figure 6.3.1.1.1-1: SAEF 2 Building Floor Plan

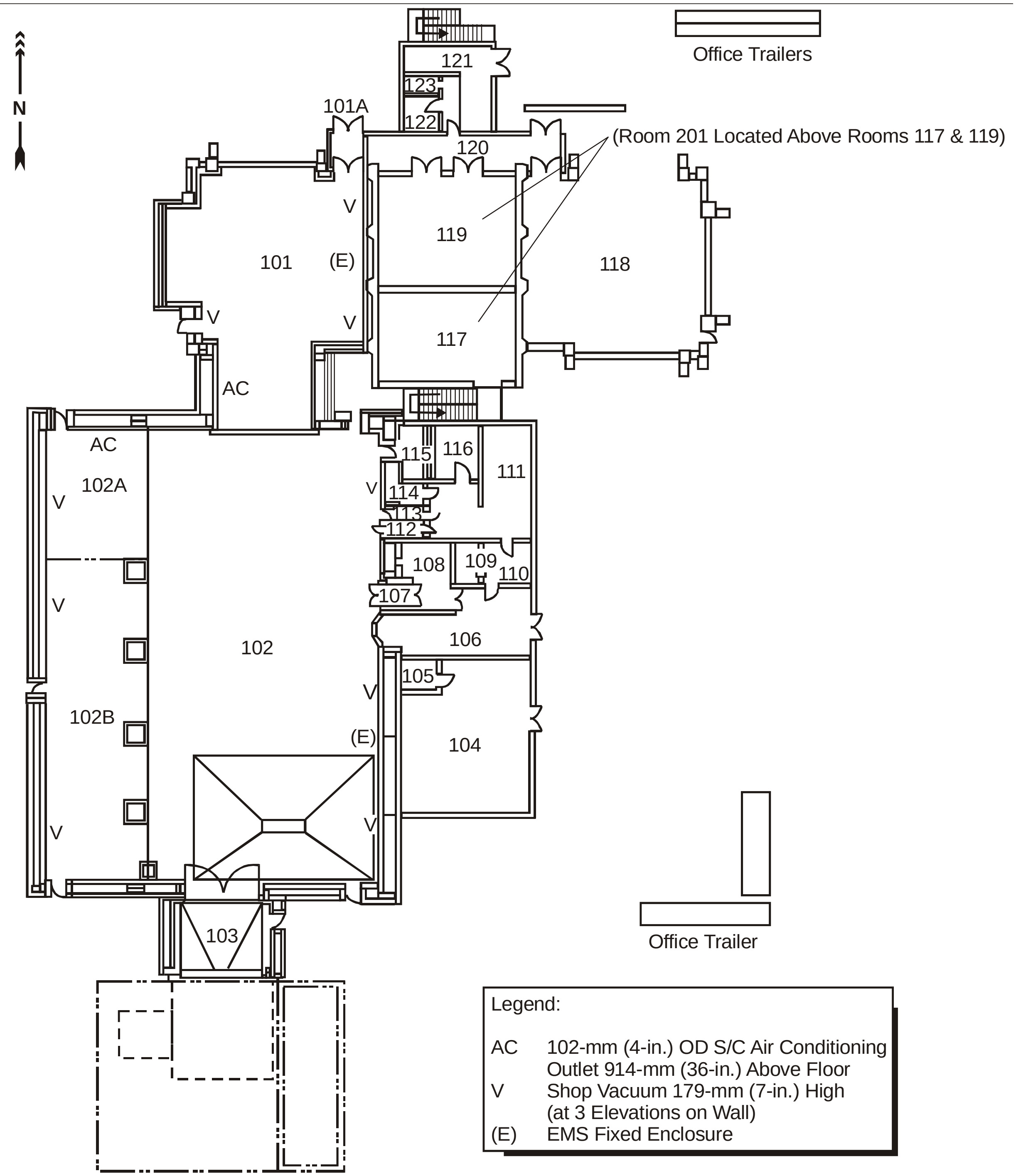
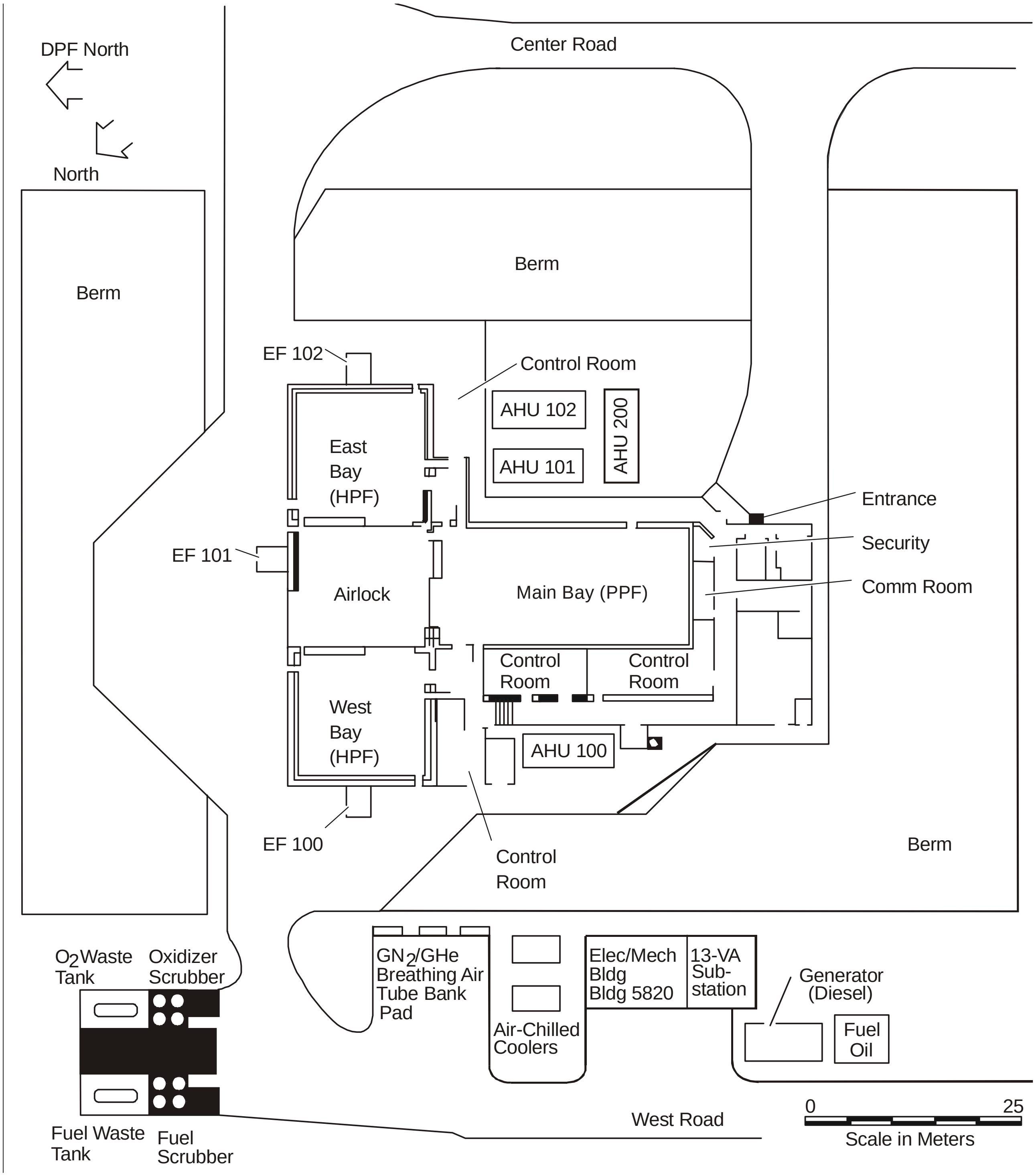


Figure 6.3.1.2.1-1: DPF Area Detail Site Plan

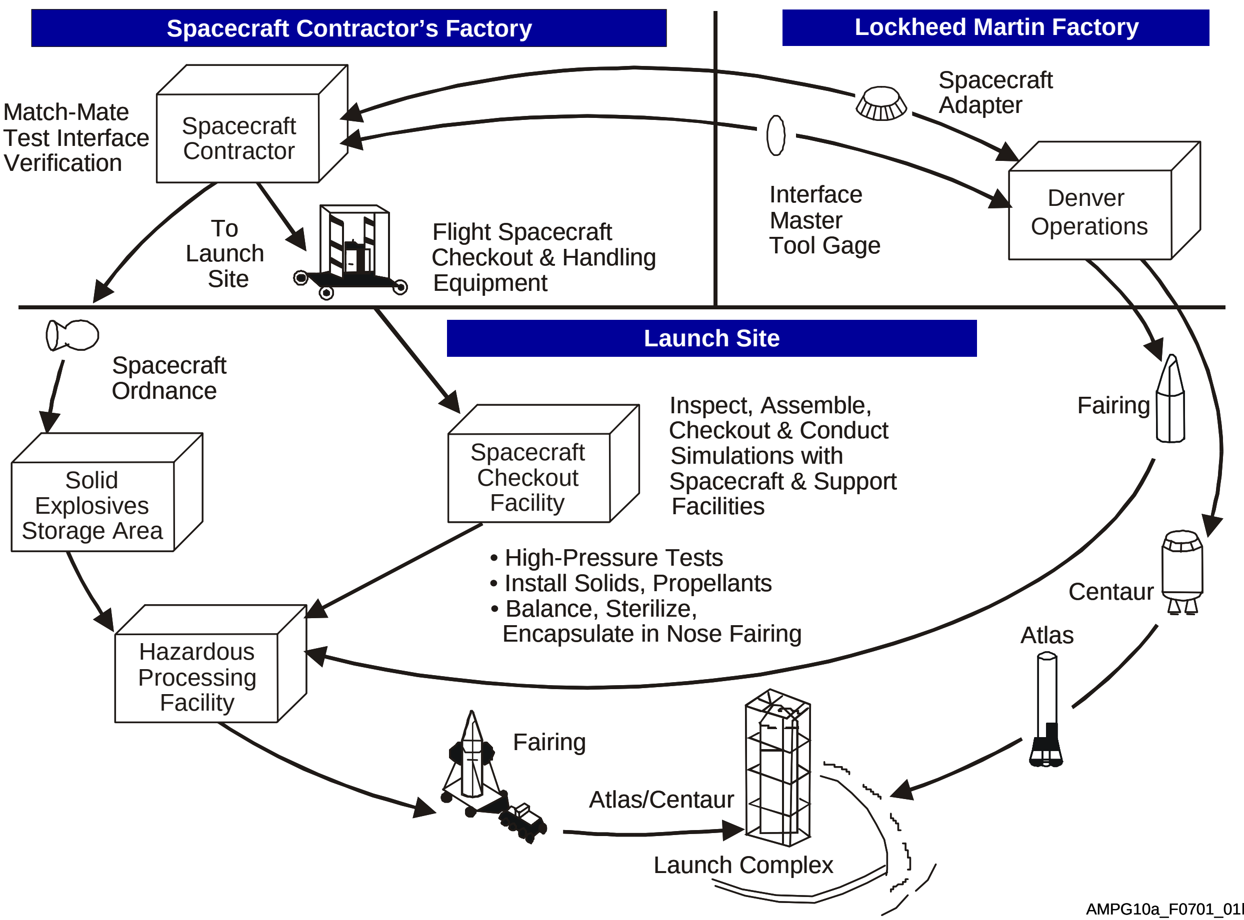


7. LAUNCH CAMPAIGN

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 Launch Vehicle and 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, 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 V 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 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 or Payload Van (PVan) and Launch Operations Center (LOC);
   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 spacecraft air conditioning;
   d. Provision of a spacecraft RF reradiate system in the Umbilical Tower (UT) /Mobile Launch Platform (MLP) mast (permitting on-stand spacecraft RF testing).
3. Coordination, preparation, and maintenance of required range support documents:
   a. Air Force System Command documents as required whenever support by any element of the Air Force Satellite Control Facility (AFSCF) is requested including the Operations Requirements Document (ORD) which details all requirements for support from the AFSCF Remote Tracking Stations (RTS) and/or Satellite Test Center (STC) during on-orbit flight operations;
   b. Range ground safety and flight safety documentation as required by the launch site Range Safety regulation and the Federal Aviation Authority (FAA);
      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.

iii) FAA launch license (for commercial missions), which includes items 3.b.i and 3.b.ii above as well as overviews of hazardous spacecraft commodities (propellants, pressure systems, etc.) The baseline FAA license is updated to address each commercial mission.

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 Centaur 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 (Reference Section 4.1.3.5.2 for requirements).

6. Post-flight 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 Force Station (CCAFS) 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 available 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 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.

Table 7.1.3-1: Hypergolic Propellants - CCAFS Fuel Storage Depot

Propellant, Hydrazine, Standard Grade, MIL-P-26536
Propellant, Hydrazine, Monopropellant Grade, MIL-P-26536
Propellant, Hydrazine/Uns-Dimethylhydrazine, MIL-P-27402
Monopropellant, High Purity Hydrazine, MIL-P-26536
Propellant, Monomethylhydrazine, MIL-P-27404
Propellant, Uns-Dimethylhydrazine, MIL-P-25604
Propellant, Nitrogen Tetroxide (NTO), NAS3620
Propellant, Nitrogen Tetroxide (MON-1), NAS3620
Propellant, Nitrogen Tetroxide (MON-3), NAS3620
Propellant, Mixed Oxides of Nitrogen (MON-10), MIL-P-27408
Propellant, Nitrogen Tetroxide (MON-3, Low Iron), NAS3620

7.2 INTEGRATED TEST PLAN

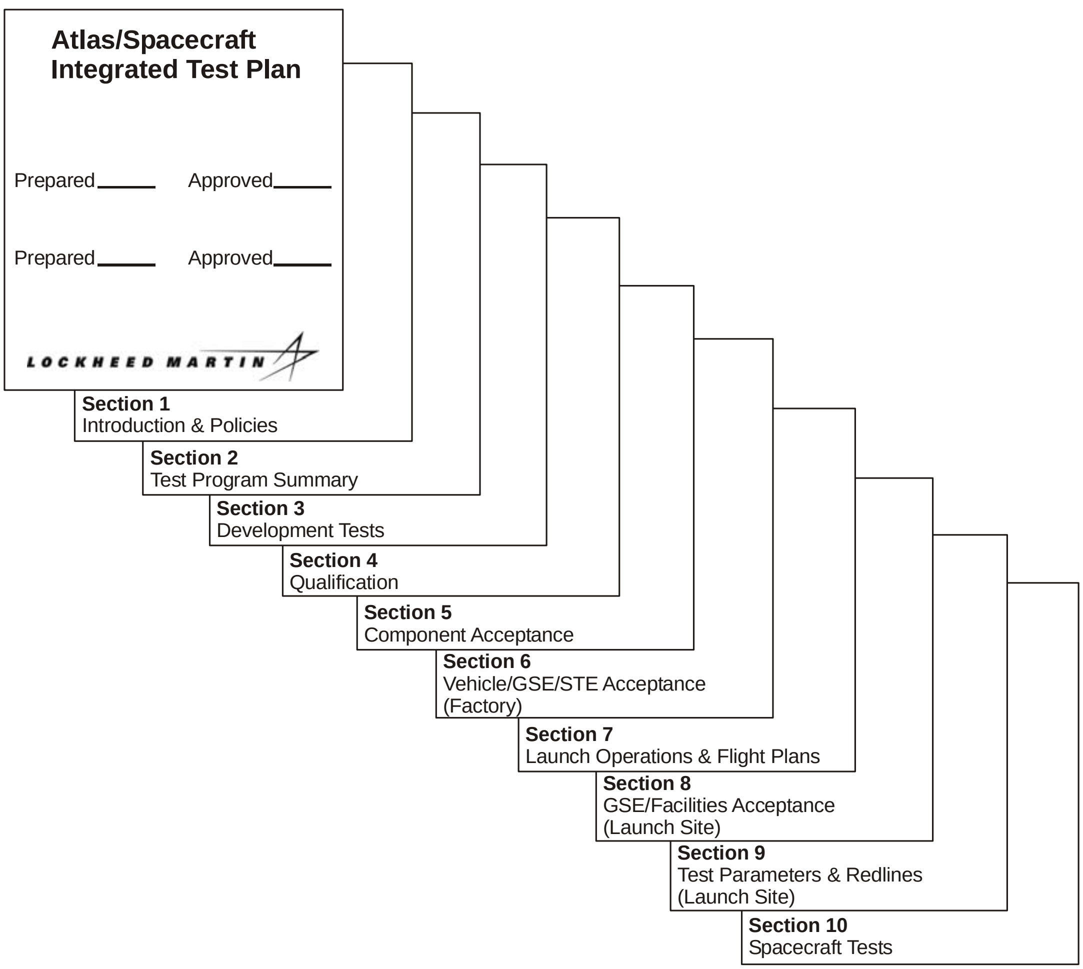
All testing performed during Atlas V design, development, manufacture, launch site checkout, and launch operations is planned and controlled through the Atlas V Integrated Test Plan (ITP). This encompasses all launch vehicle testing, including spacecraft mission-peculiar equipment and launch vehicle/spacecraft integrated tests.

The ITP 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 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) (Figure 7.2-1).

Subsections within these headings consist of the individual test plans for each Atlas V 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.

Figure 7.2-1: Integrated Test Plan Organization



7.3 TEST PROCEDURES

All test operations are performed according to documented test procedures prepared by test operations personnel using either the approved ITP subsections together with engineering drawings and specifications. The procedures for testing of Atlas V 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 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 requirements 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 ATLAS V 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 V launch vehicle.

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 Atlas V Launch Site Prelaunch Operations

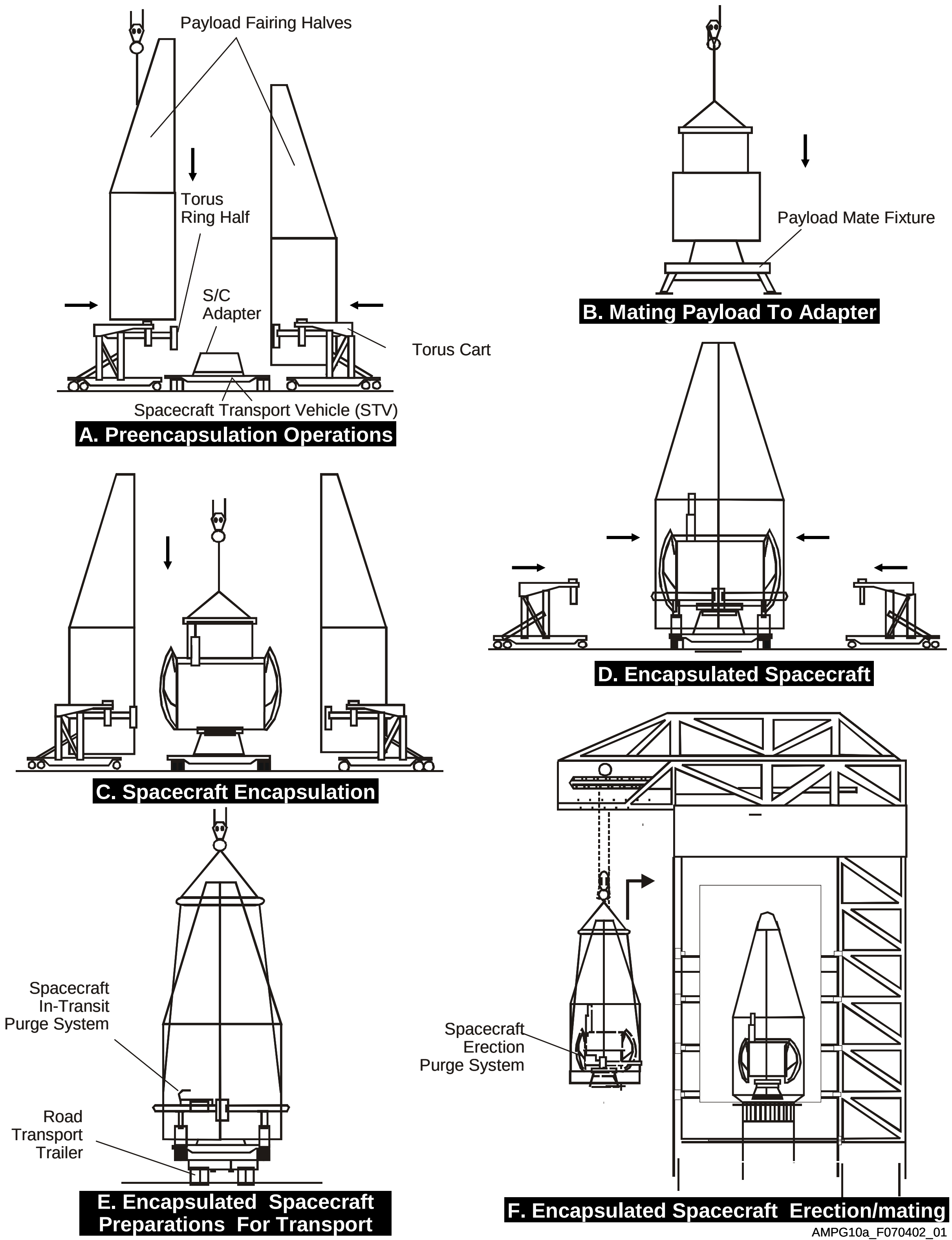
After erection of the Atlas V and connection of ground umbilical lines, subsystem and system level tests are performed to verify compatibility between airborne systems and associated ground support equipment in preparation for subsequent integrated system tests.

The Payload Fairing (PLF) halves and payload adapter are prepared for spacecraft encapsulation in the HPF at CCAFS or in the PPF at VAFB (Figure 7.4.2-1). Major tests are performed before the launch vehicle and launch pad are prepared to accept the spacecraft and start integrated operations.

Launch Vehicle Readiness Test — The Launch Vehicle Readiness Test (LVRT), the first major launch vehicle test within the Vertical Integration Facility (VIF) at CCAFS or the Mobile Service Tower (MST) at VAFB, verifies that the launch vehicle (complete, less the spacecraft and PLF) ground and airborne systems are compatible and capable of proper integrated system operation throughout a simulated launch countdown and plus-count flight sequence. Systems verified include Flight Termination System (FTS), ordnance staging, flight controls, RF, and engine hydraulics/alignment.

Wet Dress Rehearsal — The Wet Dress Rehearsal (WDR) consists of performing countdown, cryogenic tanking, abort, recycle, cryogenic detanking, and pad reentry procedures.

Figure 7.4.2-1: Payload Fairing Mate and Spacecraft Encapsulation



7.4.3 Atlas V Integrated Operations

7.4.3.1 Encapsulated Spacecraft Hoist/Mate

After a successful LVRT, the launch vehicle and the complex are prepared to accept the encapsulated spacecraft and start integrated operations. After arrival at the launch complex, the encapsulated spacecraft assembly is positioned for hoisting onto the Atlas V launch vehicle. Gas conditioning is transferred from the portable unit used during encapsulated spacecraft transport, to a facility source or GN_2 during hoisting operations. A hoisting sling is fastened to the encapsulated assembly lifting fixture and tie-downs to the ground transport vehicle are released. The encapsulated spacecraft assembly is then lifted into the VIF or MST and lowered onto the Atlas V launch vehicle.

After mechanical attachment of the spacecraft and PLF to the launch vehicle, the lifting fixture is removed, possibly necessitating a temporary detachment of the gas-conditioning duct. Electrical connections between the payload adapter and the launch vehicle are mated to complete the operation.

7.4.3.2 Postmate Functional Tests

The spacecraft customer may perform limited spacecraft functional tests shortly after spacecraft mating. These tests verify spacecraft/launch vehicle/launch complex/RF interfaces before initiation of more extensive spacecraft testing. Included as an integral part of these checks is a verification of the spacecraft launch umbilical and the spacecraft flight harness routed to the Atlas V standard electrical interface panel. The main operations performed during spacecraft testing at the VIF or MST are:

1. Umbilical and RF S-band, C-band, and/or K-band link checks (without spacecraft);
2. Spacecraft batteries trickle charge;
3. Telemetry/Telecommand operations in hardline telemetry configuration;
4. Telemetry/Telecommand operations in RF configuration (using a reradiation system);
5. Spacecraft flight configuration verifications;
6. Spacecraft/ground stations end-to-end test;
7. Stray voltage test with the launch vehicle.

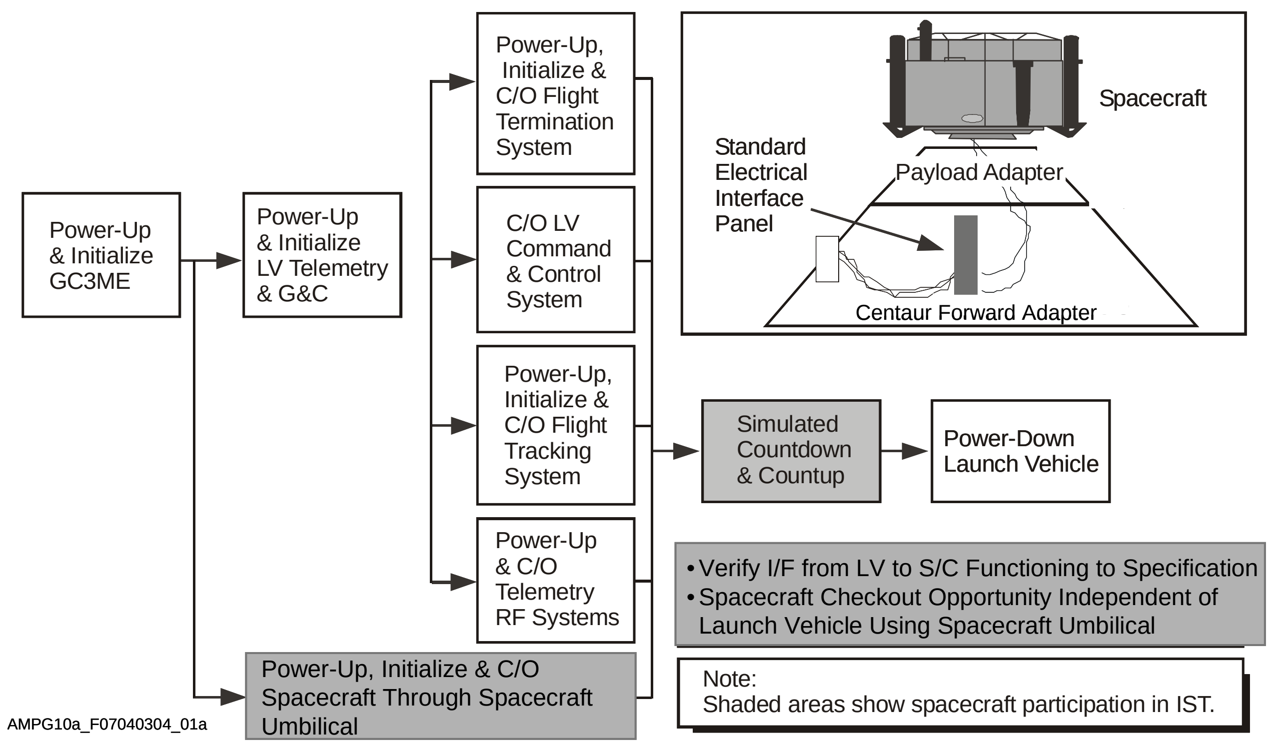
7.4.3.3 Postmate Special Tests

Special spacecraft tests may be necessary to investigate anomalies occurring in planned tests or operations, to reverify equipment operation or performance after changes were made to correct anomalies, or to accommodate spacecraft launch site schedules or operations planning. All special tests are conducted according to written and approved test procedures. For example, the spacecraft contractor may perform an operational spacecraft practice countdown to verify spacecraft timelines and coordinate Lockheed Martin support during final launch preparations. In some instances, a customer-provided spacecraft simulator may be required to validate electrical interfaces, spacecraft communications, and spacecraft Ground Support Equipment (GSE) before mating the encapsulated spacecraft with the launch vehicle.

7.4.3.4 Integrated System Test

Lockheed Martin performs the Integrated System Test (IST) with the spacecraft before moving to launch configuration and launch countdown. The spacecraft contractor provides input to the test procedure and participates in the test. An IST flow is depicted in Figure 7.4.3.4-1. This test exercises key elements of the countdown sequence arranged on an abbreviated timeline.

Figure 7.4.3.4-1: Integrated System Test Flow



Purpose — To provide a launch readiness verification of launch vehicle ground and airborne electrical systems (with a minimum of systems violations) after reconnection of ground umbilicals, after mating of the flight spacecraft assembly, and before pyrotechnic installation for launch. This is the final Electromagnetic Interference (EMI) compatibility test between the spacecraft and launch vehicle. A secondary objective of this integration test is to acquaint launch team operations personnel with communications systems, reporting, and status procedures 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.

Test Configuration — The Atlas V launch vehicle is in flight configuration within the VIF or MST, except for pyrotechnics, main propellants, and flight batteries. All batteries are simulated by support equipment prior to IST, which will use flight batteries. The spacecraft is mated in the launch configuration, including pyrotechnics, propellants, and batteries. Spacecraft test and control support equipment is installed in the PVan at CCAFS or in the LSB at VAFB.

Test Conditions — All launch vehicle umbilicals remain connected throughout the test. Vehicle power to the booster and Centaur is provided by airborne power. All launch vehicle electroexplosive devices are simulated by squib simulators. Atlas/Centaur telemetry, FTS, and C-band RF systems are open-loop tested. The spacecraft is afforded the opportunity to radiate open loop during this test, if required. This test is conducted from the Launch Control Center (LCC), and uses the full set of Ground, Command, Control, and Communications (GC^3) systems.

Test Description — Atlas V electronic systems operate through an abbreviated launch countdown that 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 for proper response. Centaur tank pressurization, engine valve actuation, prestart, and start phases are monitored for proper vehicle responses. Spacecraft participation is not required during the initial portions of the test, and no restrictions are imposed on spacecraft activity. Spacecraft and spacecraft launch team participation begins approximately 4 hours into the test. The spacecraft should be configured, as it will be on launch day, at the T-5 hour point. The test continues through an abbreviated countdown, and a nominal “plus” count, terminating at the separation command, and Contamination and Collision Avoidance Maneuvers (CCAM). Following this sequence, spacecraft participation in the IST is no longer required, and spacecraft testing and launch preparations may continue. A post-test critique is performed at the completion of the test.

7.4.3.5 Final Closeouts

Activities to be performed during final closeouts consist of final preparations necessary to ready the launch vehicle, spacecraft, and launch complex for start of launch day activities. 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.

Installation of Pyrotechnics — An RF-silence period will be imposed during which mechanical installation/connection of launch vehicle pyrotechnics will be performed.

Vehicle Compartment Closeout — After connection of pyrotechnics, vehicle compartments are readied for launch. Activities include final visual checks, final hardware configurations/remove-before-flight items, internal platform removal, closeout photographs, and airborne door installations.

Vehicle and Facility Preparations for Roll to Pad (CCAFS Only) — As vehicle compartments are readied for launch, the launch vehicle-to-MLP mast ground umbilicals/lanyards are configured for roll/flight. The VIF platforms, pad, and transport GSE are configured for transport

7.4.3.6 Launch Day Operations

Day-of-launch activities include transport of the launch vehicle/MLP from the VIF to the pad and securing the PVan and GC^3 vans in the Payload Equipment Building (PEB) at CCAFS or by MST roll-back at VAFB. After these operations are complete, propellant loading, systems verification test, and launch/plus count can commence. The launch countdown timeline for a launch from the LC-41 complex at CCAFS is shown in Figure 7.4.3.6-1 and from SLC-3E complex at VAFB is shown in Figure 7.4.3.6-2. The launch countdown timeline for a launch from the SLC-3E facility is similar with the exception of the MLP/MST differences.

7.4.4 Atlas V Launch Countdown Operations

Countdown Operations—The launch countdown consists of an approximate 10-hour count, which includes a built-in hold at T-180 minutes (30 minute hold) and at T-5 minutes (10 minute hold) to enhance the launch-on-time capability.

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 (Figure 7.4.4-1).

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.

Lockheed Martin prepares the overall countdown procedure for launch of the vehicle. However, the spacecraft customer prepares its own launch countdown procedure for controlling spacecraft operations. The two procedures are then integrated in a manner that satisfies the operations and safety requirements of both. This integration permits task synchronization through status checks at predetermined times early in the count and a complete mesh of operations during the final steps to launch.

Figure 7.4.3.6-1: Atlas V Launch Countdown - CCAFS

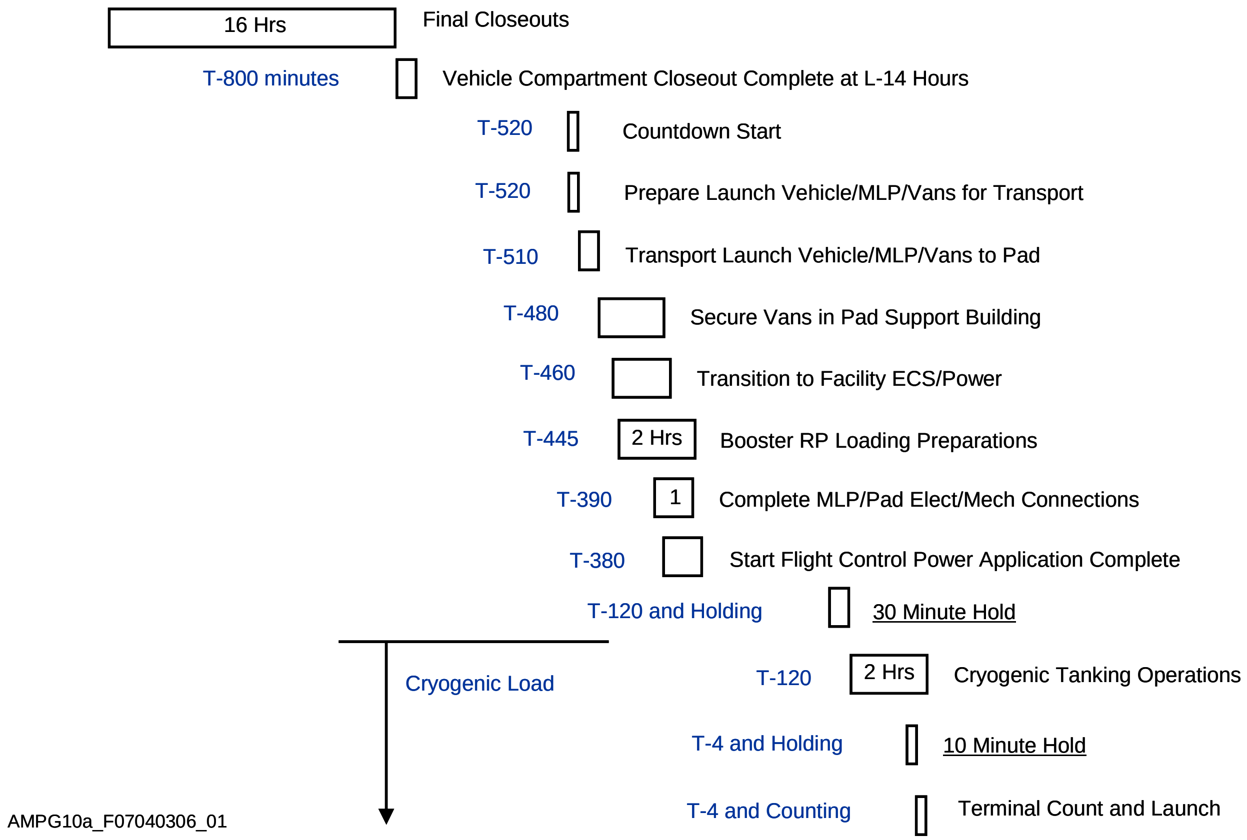


Figure 7.4.3.6-2: Atlas V Launch Countdown – VAFB

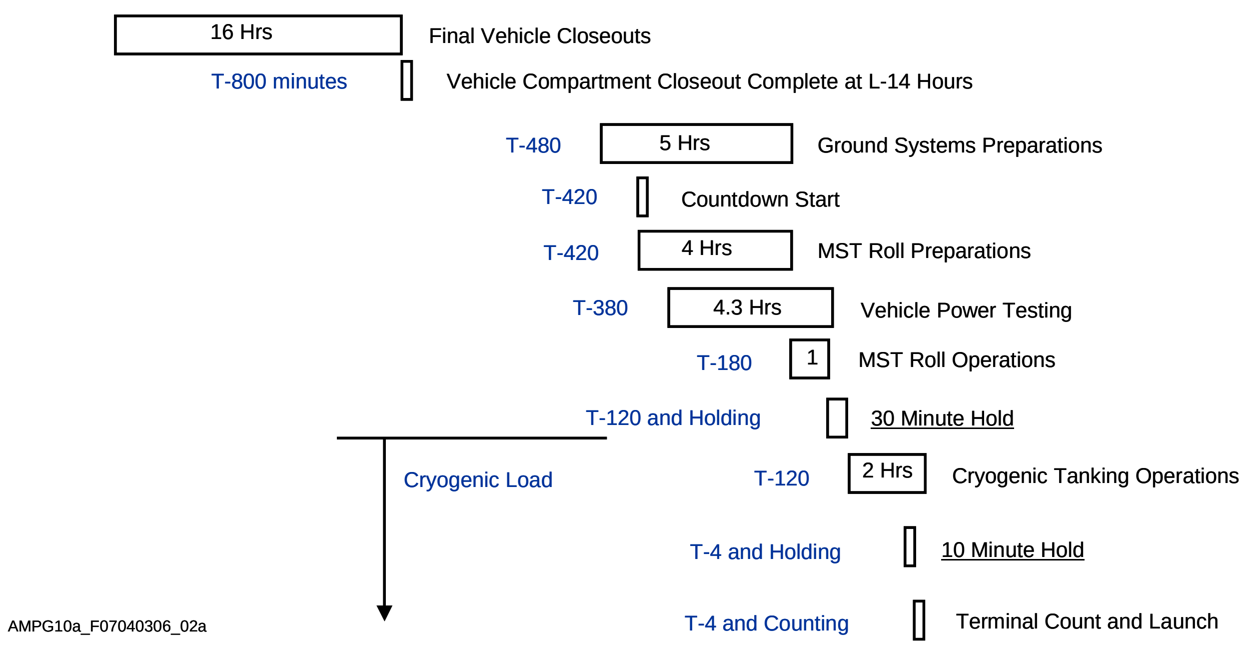
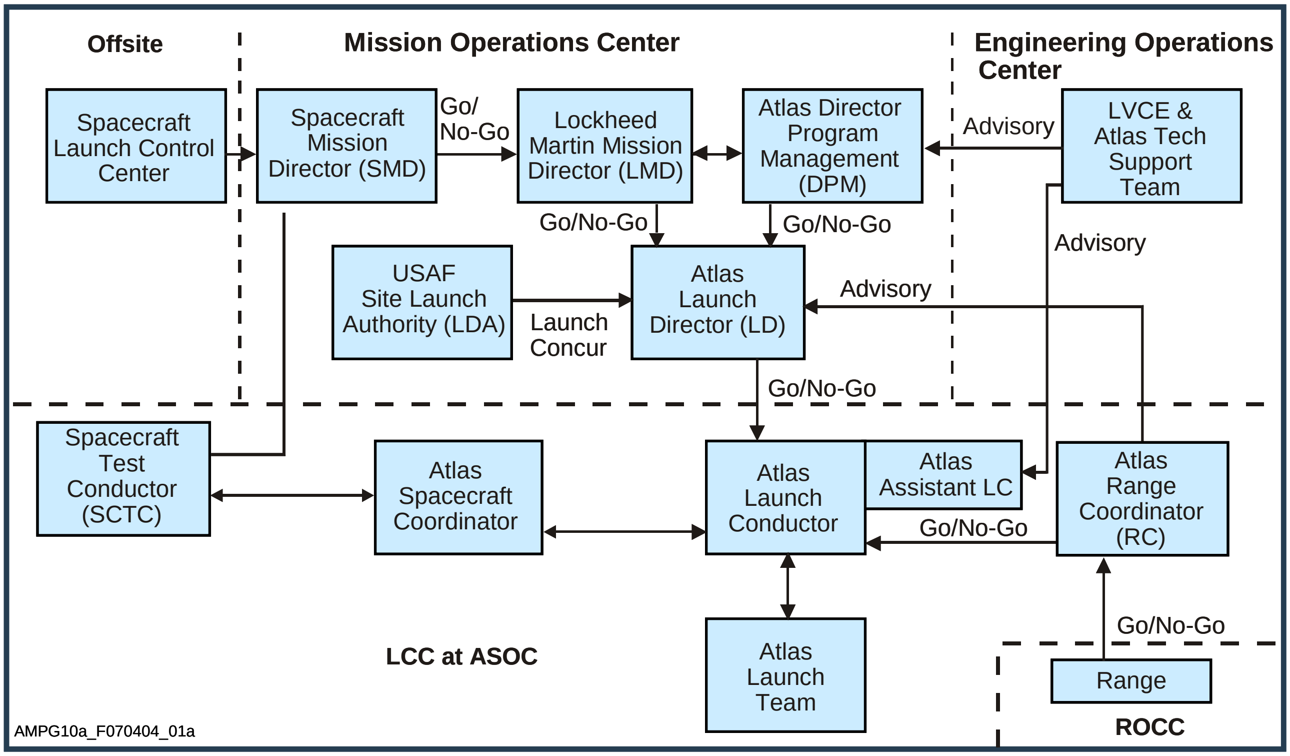


Figure 7.4.4-1: Typical Atlas V Launch Day Management Flow Diagram



7.5 LAUNCH CAPABILITY

In addition to the scheduled holds, additional hold time can be scheduled for up to 2 hours under normal environmental conditions or until the end of the scheduled launch window, whichever comes first. Launch window restrictions have typically been determined by the spacecraft mission requirements. The Atlas V 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

Lightning or equivalent weather conditions within a certain distance from the launch complex require a halt to all pad operations. A complete list of launch vehicle lightning constraints is in the appropriate generic launch vehicle Program Requirements Document (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 (-4 °F) level;

   b. Within 9 km (5 nmi) of cumulus clouds with tops higher than the -10 °C (14 °F) level;

   c. Through cumulus clouds with tops higher than the -5 °C (23 °F) 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 to -20 °C (32 to -4 °F) 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 (32 °F) 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 10 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 V launch vehicle is subject to ground wind restrictions during vehicle hoist, vehicle assembly, VAFB MST rollback or CCAFS MLP rollout, up through the time of launch. The Atlas V program has an established ground winds restriction procedure that provides 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 encapsulated spacecraft hoisting.

The ground winds monitoring system is designed to monitor vehicle loads when the launch vehicle is exposed on the launch pad prior to 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 V vehicle in flight can be calculated well before 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 V 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, Colorado 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 Centaur aft panel is ejected), the launch vehicle has a 24-hour turnaround capability after a launch abort due to a nonlaunch vehicle/GSE problem.

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 Centaur aft panel (ejected at T-4 seconds) and removal and replacement of the propellant pressurization line pyrovalves (fired upon aft panel ejection).

7.7.3 Safety COLA Constraints

Portions of the launch window may not be available due to a safety Collision Avoidance (COLA) closure. The safety COLA analysis is performed by the USAF Range Safety Organization. The analysis examines the trajectory (from liftoff to end of mission) and launch window to determine if there is a potential for the launch vehicle to collide (a conjunction) with a manned object or object capable of being manned, such as the International Space Station or Space Shuttle. Launch periods that would result in a high probability of conjunction at any time in the trajectory are identified and excluded from the daily launch window. Nominal and dispersed trajectories are included in the analysis. Safety COLA analyses are completed at L-48, L-24 and L-4 hours with the L-4 hour analysis used to establish the final safety COLA closures for launch day.

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8. ATLAS V SYSTEM ENHANCEMENTS

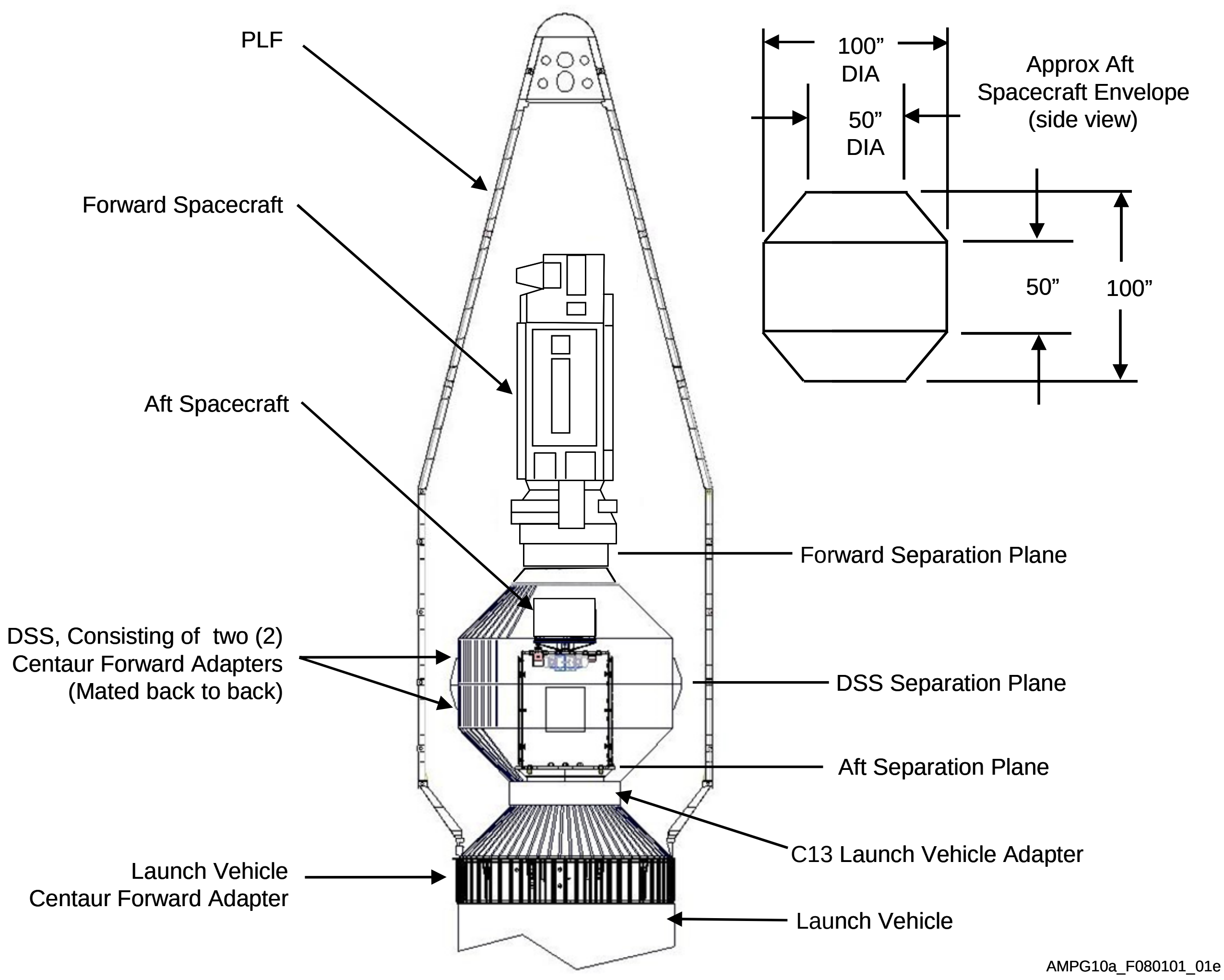
Maximizing on the combined strengths of the engineering, development, and production functions of Lockheed Martin, several Atlas V launch system enhancements are being implemented or planned to maintain the competitiveness of Atlas V in the launch services market. These initiatives are being supported to expand the operational capability of the Atlas/Centaur launch vehicle. Looking toward 21st century requirements, several major Atlas V modifications are planned and initial engineering is beginning. This section describes the initiatives being pursued by the Atlas V program.

8.1 MISSION-UNIQUE ENHANCEMENTS

8.1.1 Dual Spacecraft System

The Dual Spacecraft System (DSS) allows a Dual Manifest of two small- to medium-class spacecraft to be carried and launched at the same time on an Atlas V 400 series launch vehicle shown in Figure 8.1.1-1. The DSS consists of two modified flight-proven Centaur Forward Adapters mated back-to-back in a clamshell-type arrangement. This arrangement utilizes existing payload fairing explosive bolts to hold together and

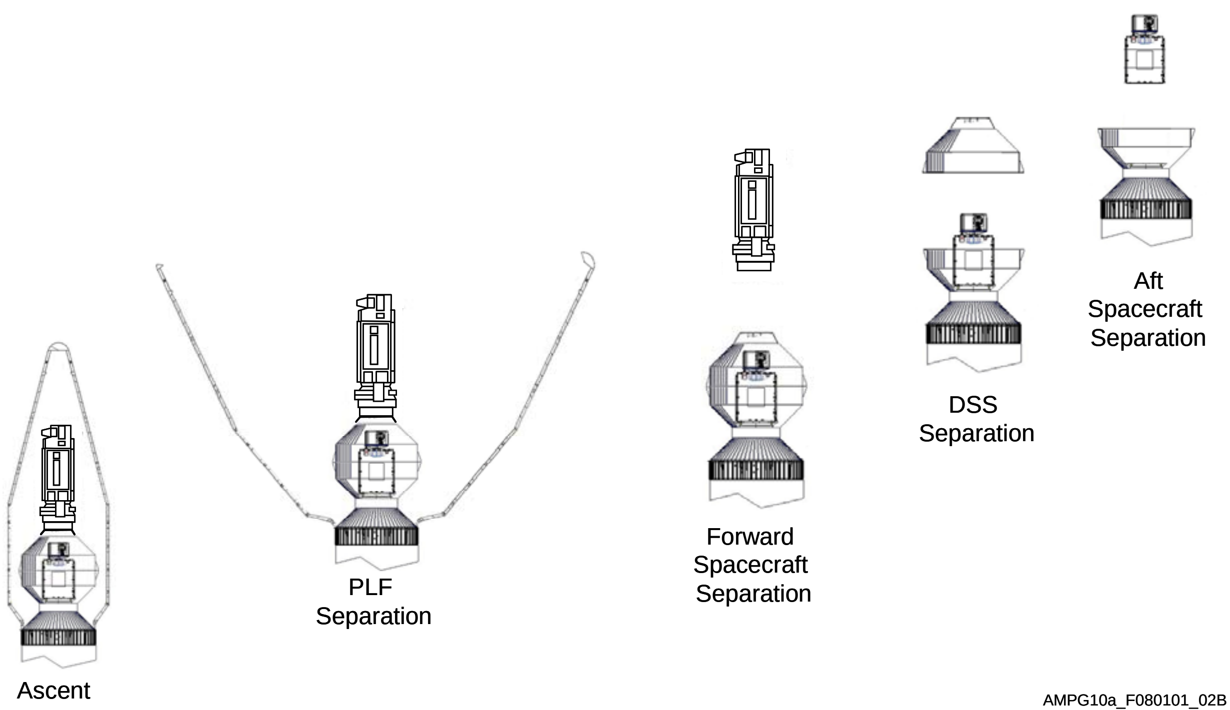
Figure 8.1.1-1: Dual Spacecraft System



separate the DSS halves. The Centaur forward adapter has successfully flown more than 100 missions atop Centaur, and the PLF/Boattail explosive bolts have a remarkable flight record on both Atlas and Titan/Centaur Programs.

The DSS fits entirely within the 4-m Atlas V PLF and is attached to an existing Atlas V C13 payload adapter. The forward spacecraft rides on top of the DSS, attached via an Atlas V or customer-provided payload adapter. The aft spacecraft is attached by either an Atlas V or customer-provided payload adapter and stays protected inside the confines of the DSS. The DSS can be flown with an LPF, EPF, or XEPF as needed based on the size of the forward spacecraft. Venting provisions are provided as well as Environmental Control System (ECS) cooling or heating into the DSS interior as required. The PLF is jettisoned prior to forward spacecraft separation as shown in Figure 8.1.1-2. When the correct orbit is reached, the forward spacecraft will be separated via its separation system. The Centaur can then maneuver and continue on to another orbit for the aft spacecraft as required. The top half of the DSS will then be jettisoned by firing the explosive bolts, and the upper half will separate due to the force applied by the separation springs. Upon completion of jettisoning the forward half of the DSS, the aft payload will separate via its separation system. The DSS, consisting entirely of existing flight-proven hardware, is conceptually straightforward, but has yet to undergo detailed engineering analysis.

Figure 8.1.1-2: Dual Spacecraft System Separation Sequence



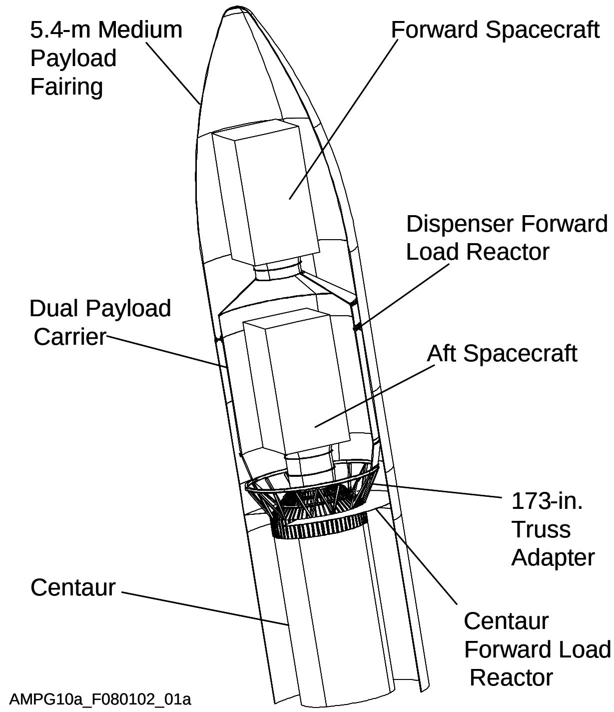
8.1.2 Dual Payload Carrier

Lockheed Martin has completed the preliminary design for a Dual Payload Carrier (DPC) for use on the Atlas V with the 5-m Medium PLF. The DPC will give the Atlas V the capability to simultaneously carry two medium- or intermediate-class spacecraft. The DPC will be adjustable in height to accommodate payloads of different heights.

The DPC fits entirely within the 5-m medium PLF and divides the PLF volume into two payload compartments (Figure 8.1.2-1). The forward compartment payload static envelope has a diameter of 4,572 mm (180.0 in) at the base, and conforms to the ogive shape of the PLF as it extends forward. The forward spacecraft mates to an Atlas V or customer-provided adapter that, in turn, mates to the 1,575-mm (62.01-in) diameter forward interface ring of the DPC.

The aft spacecraft is encapsulated by the DPC and mates to an Atlas V or customer-provided payload adapter that, in turn, mates to the 1,575-mm (62.01-in.) diameter forward interface ring of a C-type adapter. This C-type adapter mates to the top of the Centaur forward adapter. This aft compartment will have a static payload envelope with a minimum 4,000-mm (157.5-in.) diameter. Preliminary forward and aft payload static envelopes are shown in Figure 8.1.2-2.

Figure 8.1.2-1: Atlas V 500 Series Dual Payload Carrier

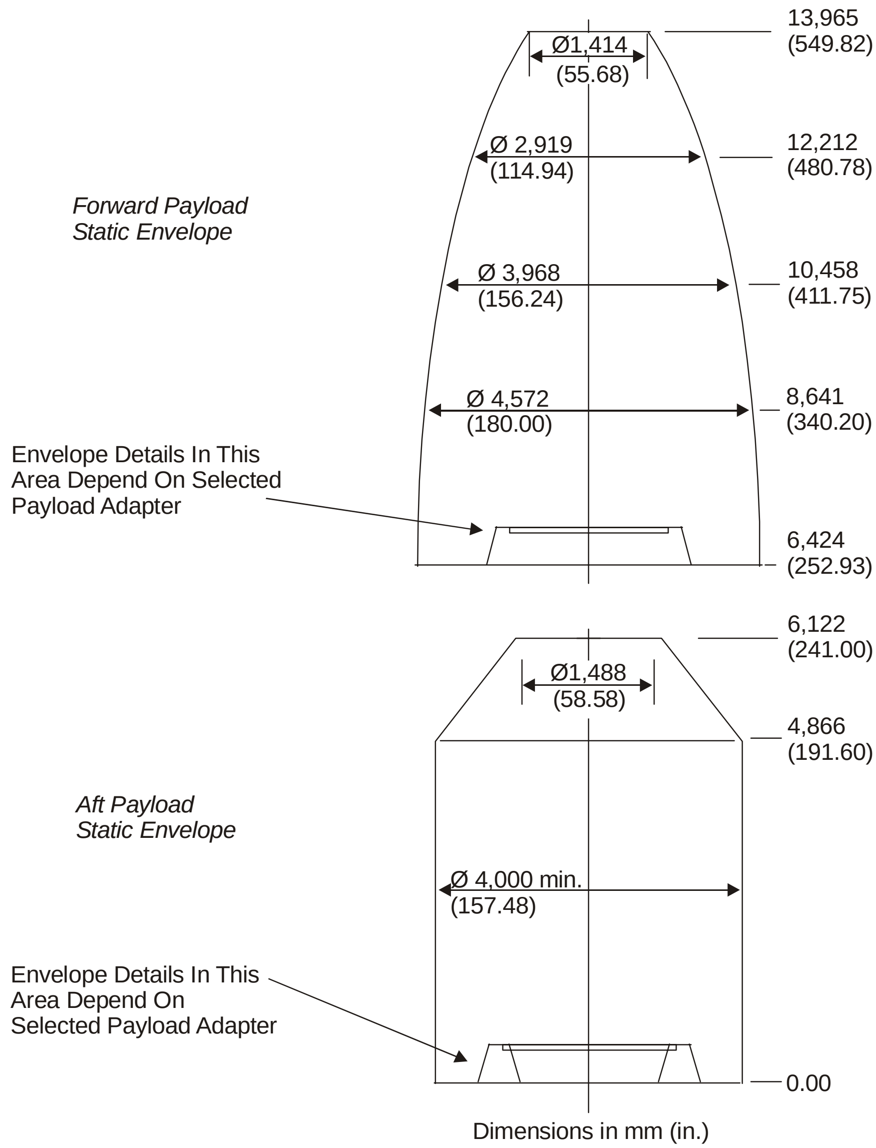


The DPC provides access to the aft spacecraft through standard 600-mm (23.6-in.) diameter doors, and it can provide accommodations for a reradiating system when required. Ports in the DPC structure will ensure adequate conditioned air passes through the aft compartment to maintain the required thermal environment for the aft spacecraft. The dispenser forward load reactor, near the top of the DPC cylinder section, controls relative motion between the DPC and PLF before PLF jettison.

The DPC structure, a lightweight, carbon fiber-reinforced composite sandwich structure, attaches to the forward interface of the 4,394-mm (173-in.) truss adapter.

To facilitate DPC jettison after forward spacecraft deployment, the DPC cylinder section contains a separation ring near its aft end. This pyrotechnic, frangible joint-type separation system contains all combustion byproducts and debris. After separation system actuation, a set of force-balanced springs pushes the DPC forward portion away from the aft spacecraft, ensuring adequate clearance is maintained between the DPC and aft spacecraft. The Centaur then turns to the required separation attitude and commands aft spacecraft separation.

Figure 8.1.2-2: Dual Payload Carrier - Preliminary Payload Envelopes



8.1.3 Dual Payload Carrier - Short

Lockheed Martin has completed an initial study for a Dual Payload Carrier – Short (DPC-S) for use on Atlas V with the 5-m short PLF. It is derived from the DPC but is approximately 11.5 feet shorter, allowing it to fit in a Short 5-m PLF without a load reacting device (like the DFLR for DPC). This will allow the 5-m short PLF to carry one intermediate class spacecraft and one small spacecraft. See Figure 8.1.3-1 for preliminary static payload envelopes.

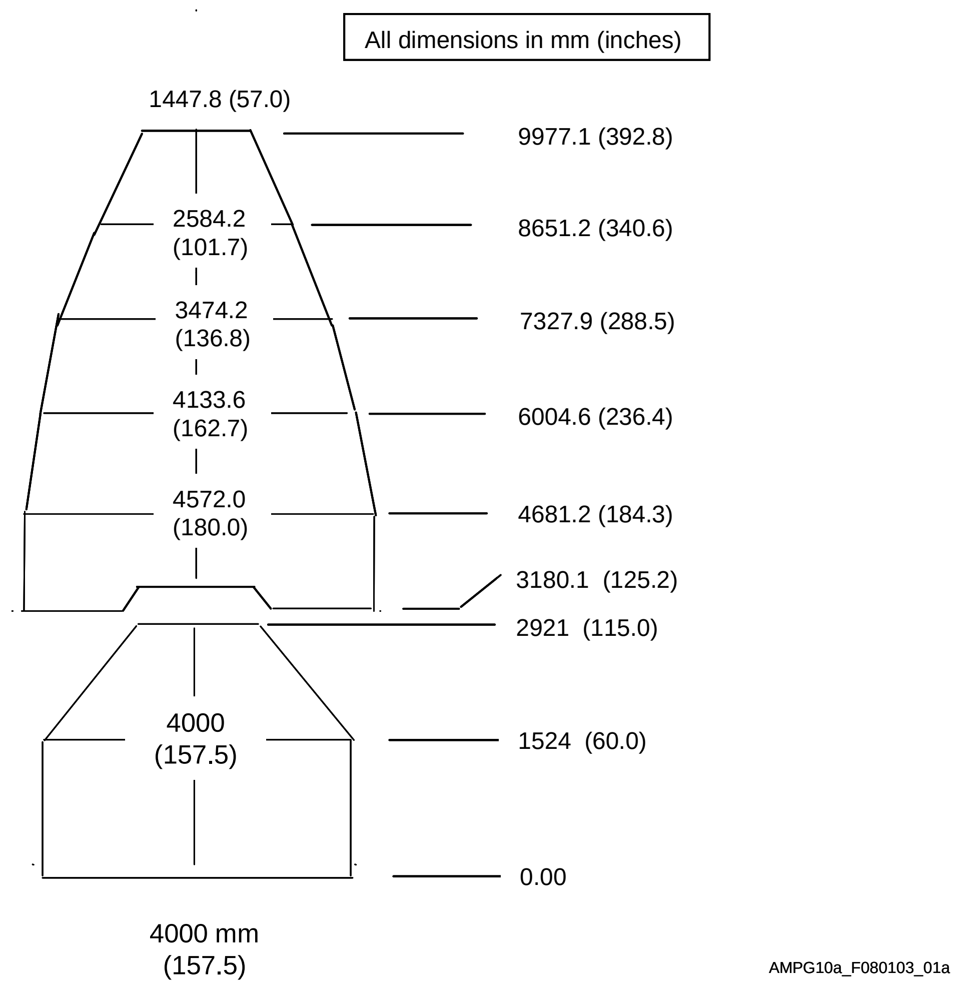
The forward payload compartment has a diameter of 4,572 mm (180.0 in) at the base and conforms to the ogive shape of the PLF as it extends forward. The forward spacecraft mates to an Atlas V or customer-provided payload adapter that, in turn, mates to a 1,575 mm (62.01 in) diameter C13-type PLA. This C-type adapter mates to the 1,575 mm (62.01 in) diameter forward interface ring of the DPC-S (duplicates the Centaur forward adapter payload adapter interface). The aft spacecraft is encapsulated by the DPC-S and mates to an Atlas V or customer-provided payload adapter that, in turn, mates to a 1,575 mm (62.01 in) diameter C-type adapter. This C-type adapter mates to the top of the Centaur forward adapter.

The DPC-S provides access to the aft spacecraft through standard 600 mm (23.6 inch) diameter doors that also ensure adequate conditioned air passes through the aft compartment to maintain the required thermal environment for the aft spacecraft.

The DPC-S structure is made of lightweight carbon fiber reinforced composite sandwich structure and attaches to the forward interface of the 4,394 mm (173 in) truss adapter.

To allow DPC-S jettison after forward spacecraft deployment, the DPC-S contains a separation joint in the forward area of the cylinder, near the conic interface. This pyrotechnic, frangible joint-type separation system contains all combustion byproducts and debris. After separation, balanced springs push the DPC-S forward section away from the aft spacecraft, ensuring adequate clearance is maintained between the DPC-S and the aft spacecraft. The Centaur then turns to the required separation attitude and commands aft spacecraft separation.

Figure 8.1.3-1: Dual Payload Carrier – Short Preliminary Payload Envelopes



8.1.4 Secondary Payload Carrier

The Atlas Secondary Payload Carrier (SPC) is a flexible and modular approach to secondary payload manifesting. It can be flown on either 4-m or 5-m PLF configurations (400 or 500 series vehicles). It is designed to carry one to four secondary payloads up to 200 kg each. Figures 8.1.4-1 and 8.1.4-2 depict the SPC in either the 4-m or 5-m PLF.

Figure 8.1.4-1: Secondary Payload Carrier – Attaches to C Adapter

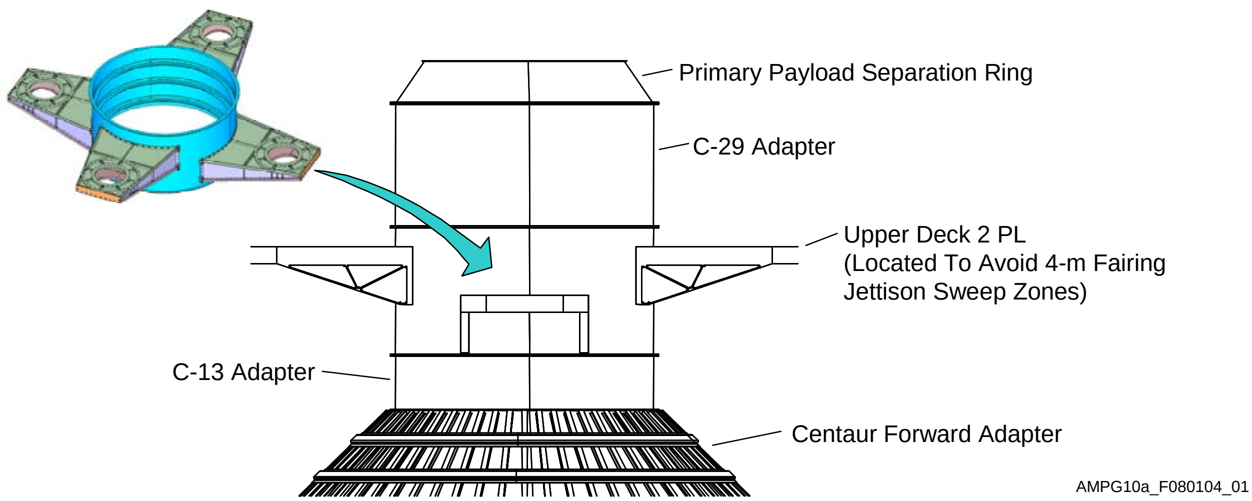
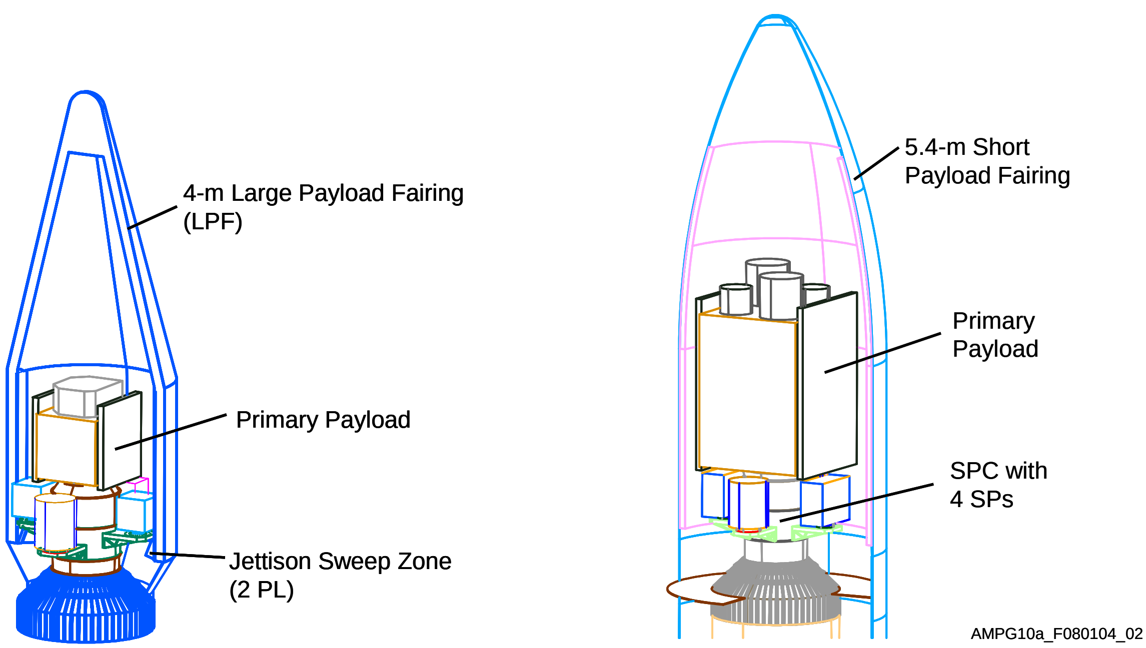


Figure 8.1.4-2: Secondary Payload Carrier – Fits in either 4-m or 5-m PLF



8.2 ATLAS V HEAVY-LIFT ENHANCEMENTS

8.2.1 Heavy Lift Payload Truss

A 3,302-mm (130-in.) truss has been designed and built for the upper stage acoustic, modal, and static load testing. The truss was designed as a load fixture to input loads to the Centaur similar to a flight truss. Figure 8.2.1-1 shows the struts, which were made from aluminum tubes. A flight truss would have struts made from graphite/epoxy tubes and the forward ring would be an aluminum structural shape. A truss similar to this one with a spacecraft interface from 2,972 mm (117 in) to 4,394 mm (173 in) could be available as a mission-unique option for heavy payload requirements (Figure 8.2.1-2). This mission-unique interface is being designed to carry a heavy-lift spacecraft up to 20,400 kg (45,000 lb). The 914-mm (36-in) high truss interfaces with the Centaur forward adapter at the same 12 mounting locations as the 4,394-mm truss. Similar to the 4,394-mm truss, the struts would be graphite epoxy with titanium end fittings, and the forward and aft brackets and the forward ring would be aluminum alloy. Spherical bearings are in each end of each strut to allow only tension or compression loads in each strut. This truss would be used inside the 5-m PLF on either the Atlas V 500 or Atlas V HLV configurations for heavy payloads more than 9,072 kg (20,000 lb).

Figure 8.2.1-1: 3,302-mm Test Truss

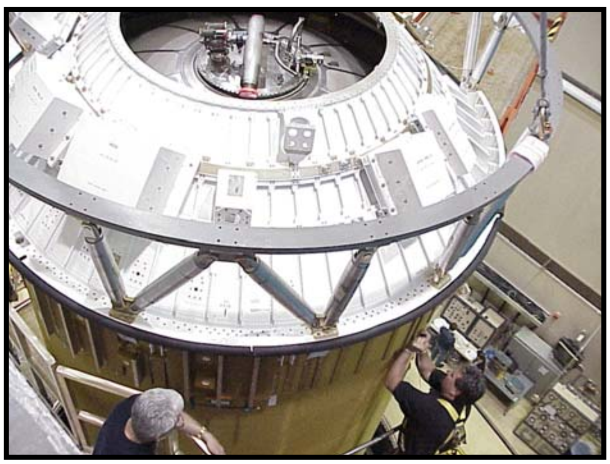
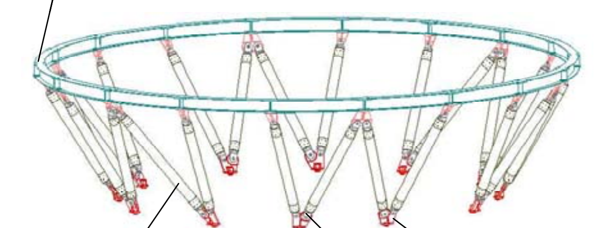


Figure 8.2.1-2: 4,394-mm Diameter Payload Truss



8.3 ATLAS EVOLUTION

8.3.1 Overview

Since 1957, Atlas has been incrementally enhanced, improving performance, reliability, and responsiveness while reducing operational cost. By segmenting the development steps into focused incremental upgrades, Atlas has been able to continuously and seamlessly upgrade the system while maintaining 100% mission success on all eight new flight configurations since 1990, as shown in Figure 8.3.1-1.

Atlas family is designed around common elements
a) Common booster, Centaur, launch complex
b) High production rates to reduce cost
c) Shared infrastructure cost
d) High demonstrated reliability

Leveraging off of this successful history, Atlas has developed an Atlas evolutionary (i.e., spiral) development strategy for future space transportation capabilities that spans the existing Atlas V performance range while more than tripling the maximum performance capability to support future needs. This evolution is split into two primary phases. Phase 1 enhances the Centaur upper stage, while Phase 2 enhances the booster (Figure 8.3.1-2).

A key feature of the current Atlas V family is the use of common vehicle elements to support performance requirements spanning 9 to 28 mT to LEO. Use of common elements improves production efficiency, reliability, and cost while providing the ability to tailor the overall launch vehicle configuration with combinations of these common elements to meet mission requirements. The continued Atlas evolution plan maintains this commonality and incremental performance capability to support existing and future customer requirements. As phases 1 and 2 are implemented, the new configurations (wide body Centaur, wide body booster) become the new common elements of the Atlas fleet, retiring the existing Centaur and booster (maintaining the production rate and reliability benefits).

Figure 8.3.1-1 Recent Atlas Evolution Resulted in 8 of 8 First Flight Successes

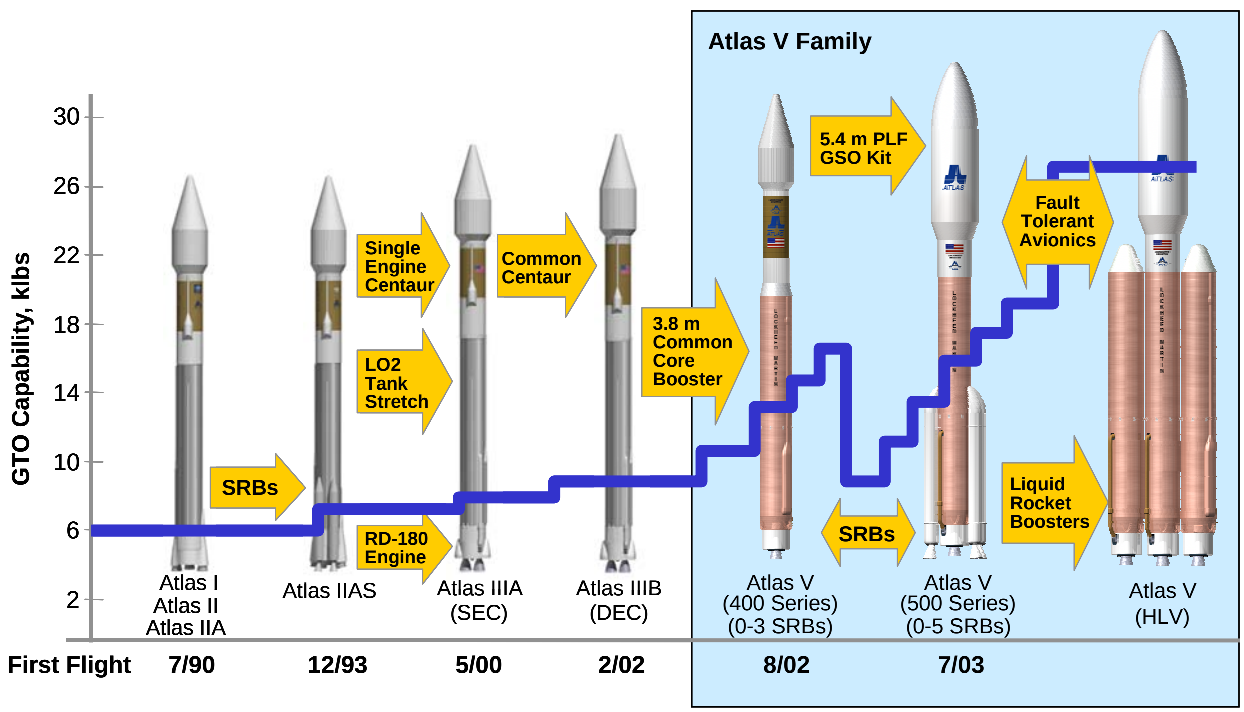
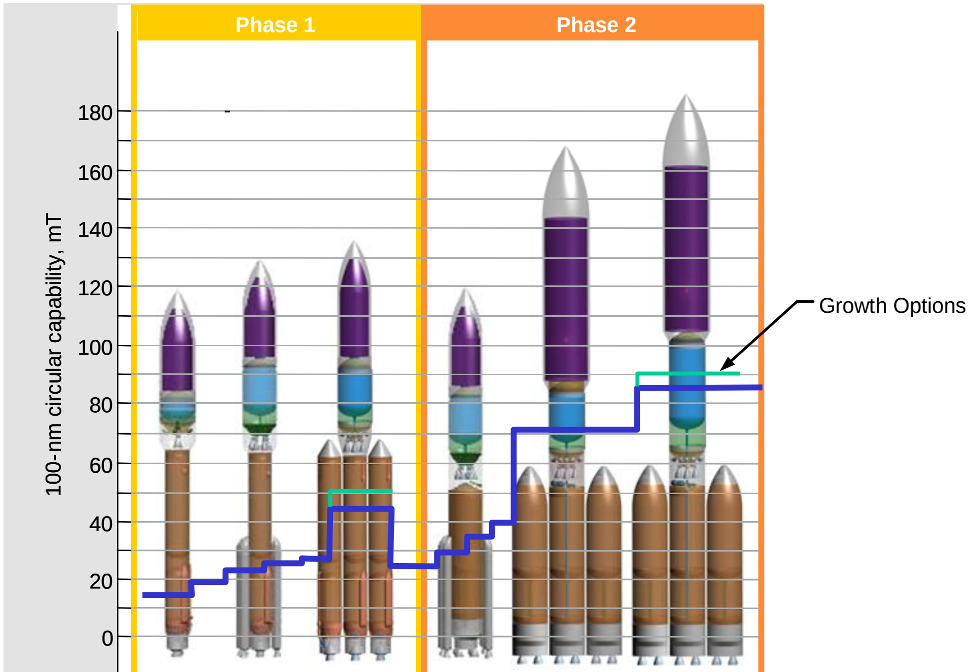


Figure 8.3.1-2 Atlas Evolution Supports Existing and Future Customer Requirements



	Phase 1						Phase 2				
Solids	0	2	3	4	5	0	0	2	4	6	0
LRBs	0	0	0	0	0	2	–	–	–	–	2
100 nm C	14.7	19.4	22.1	24.5	26.6	43.7	26.2	30.9	35.6	39.8	71.3
220 nm C	12.2	17.5	20.3	22.7	25.3	40.6	27.5	29.0	33.7	37.9	68.5
GTO	5.9	8.7	10.2	11.6	12.9	20.9	12.6	15.3	17.9	20.2	35.3
C3=0	3.9	6.4	7.6	8.9	10.1	16.2	9.6	11.8	13.8	15.6	27.5
C3=20	2.2	4.1	5.1	6.2	7.2	11.7	6.8	8.5	10.1	11.5	20.5
C3=80	—	.3	.9	1.4	2.0	4.4	1.7	2.6	3.5	4.2	9.0

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8.3.2 Phase 1: Wide Body Centaur

Phase 1 of Atlas evolution focuses on modifying the Centaur vehicle. With the advent of the high thrust and high Isp of the RD-180 engine and high performance SRBs, much larger Centaur propellant masses can be accommodated. Centaur propellant mass can be increased to six times the present Atlas V levels. The additional propellants are accommodated by increasing the Centaur diameter to 5.4m (18 ft) to match the diameter of the Atlas V-500 series PLF, as shown in Figure 8.3.2-1.

This development is accommodated using existing tank structural technology and maintains existing site infrastructure, avionics, engines, pneumatics, and other subsystems. Once implemented, this wide-body Centaur will replace the existing Atlas V Centaur and become the new upper stage common element for the fleet.

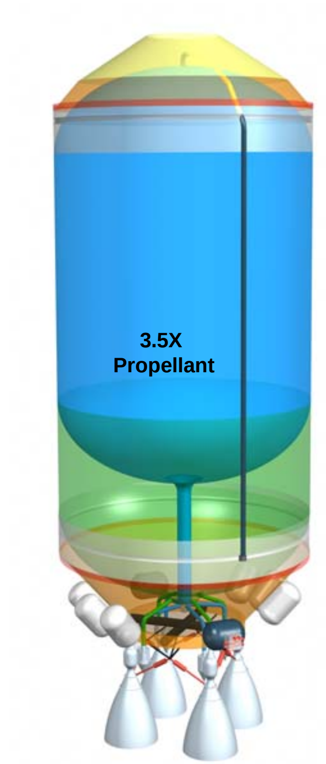
To satisfy a wide range of mission needs, the wide-body Centaur is being designed to simply accommodate multiple propellant volumes through the addition of tank length. This allows the propellant mass to vary from 1.5 to 6 times the current Atlas V Centaur's mass.

The 5.4-m diameter Centaur can accommodate one or more RL10s as needed. Multiple RL-10 engines provide the thrust required by the larger versions of the wide-body Centaur while also enabling Centaur engine-out capability. Centaur engine-out capability more than doubles the launch vehicle system reliability.

The wide body Centaur is being designed specifically to accommodate a wide range of mission performance requirements. The baseline wide body Centaur will accommodate GTO and GSO missions. Through the addition of a mission-peculiar kit consisting of solar power and passive thermal shielding, mission durations in excess of a year are possible.

Like the existing Atlas V, the phase 1 vehicles can be flown without SRMs, with up to five SRMs, or in the HLV configuration. The no-solids configuration effectively matches the cost/performance of the Atlas V 401 while providing the larger 5.4-m PLF envelope. Vehicles with five SRMs and four RL10s exceed the current Atlas V HLV performance. The phase 1 HLV configuration further amplifies performance, supporting 44 mT to LEO.

Figure 8.3.2-1 Wide Body Centaur Provides first Atlas Evolution Phase



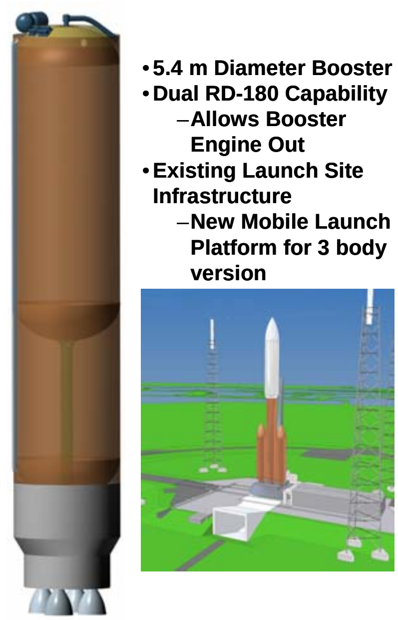
8.3.3 Phase 2: Wide Body Booster

Phase 2 vehicles retain the phase 1 Centaur and apply the same principles of modularity (combinations of common elements), mission flexibility, and fabrication methodology to a 5.4-m (18-ft) diameter booster. The aft thrust structure, plumbing, and support systems are modular to support either one or two RD-180 engines. Adding a second RD-180 provides additional thrust to support heavy mission requirements, while providing engine-out capability during booster phase of flight. Figure 8.3.3-1 illustrates the wide body booster configuration.

Like phase 1 Centaur development, the booster tank volume is being designed to be variable (0.9 to 1.7 times Atlas V booster volume) to accommodate diverse mission needs. Through simplified tank design, the booster can accommodate up to six SRMs, resulting in a “single stick” performance in excess of 39 mT to LEO, 40% larger than the current Atlas V HLV.

A phase 2 three-body HLV configuration provides 86 mT to LEO, approaching Saturn V performance.

Figure 8.3.3-1 Wide Body Booster Provides Second Atlas Evolution Phase



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, intermediate, and heavy space lift requirements.

A.1 VEHICLE DEVELOPMENT

The Atlas program began in the mid-1940s with studies exploring the feasibility of long-range ballistic missiles. The Atlas launch vehicle family has evolved through various United States Air Force (USAF), National Aeronautics and Space Administration (NASA), and commercial programs from the first Research and Development (R&D) launch in 1957 to the current Atlas V configurations (Figure A.1-1). More than 565 Atlas vehicles have flown since December 2004.

Atlas Booster — Versions of Atlas boosters were built specifically for manned and unmanned space missions, including the pioneering Project Mercury manned launches that paved the way toward the Apollo lunar program. The addition of the high-energy Centaur upper stage in the early 1960s made lunar and planetary missions possible. In 1981, the Atlas G booster improved Atlas/Centaur performance by increasing propellant capacity and upgrading engine thrust. This baseline was developed into the successful Atlas I, II, Atlas IIA, IIAS, IIIA and IIIB launch vehicles.

Atlas V continues the evolution of the Atlas launch vehicle family. Today, as the world's most successful launch vehicle, the Atlas V is offered in a comprehensive family of configurations that efficiently meet spacecraft mission requirements (Figure A.1-2).

Atlas V, the nation's next-generation space launch vehicle, is the most flexible, robust, and reliable launch vehicle system offered by Lockheed Martin Corporation. The Atlas V system is capable of delivering a diverse array of spacecraft, including projected government missions to Low Earth Orbit (LEO), heavy lift Geosynchronous Orbits (GSO), and numerous Geostationary Transfer Orbits (GTO). To perform this variety of missions, Lockheed Martin combines a Common Core Booster™ (CCB) powered by a single RD-180 engine with a standard Atlas V 4-m Long Payload Fairing (LPF), Extended Payload Fairing (EPF) or Extended EPF (XEPF) to create the Atlas V 400 series. For larger and heavier spacecraft, the Atlas V 500 series combines the CCB with a 5-m diameter Payload Fairing (PLF) available in three lengths of short, medium and long. The Atlas V Heavy Lift Vehicle (HLV) has been designed to meet heavy spacecraft mission requirements, combining three CCBs with the 5-m Long PLF. The Atlas V 400 and 500 series include a Common Centaur that can be configured with either a single or dual engine, depending on mission requirements.

The Atlas V system provides increased reliability over its predecessors. The increased reliability is achieved through a simplified design that incorporates fault avoidance, fault tolerance, and reduction of Single Point Failures (SPF). Production Engineering and Test Engineering have been involved in all phases of the design process leading to producibility and testability improvements that have yielded streamlined and repeatable manufacturing processes. The ultimate result is a reduction in nonconformances and defects, which translates into more reliable processes and hardware.

The robustness of the Atlas V system is enhanced by the use of common system elements assembled into a family of vehicles that satisfy a wide range of mission requirements while providing substantial performance margins. In addition to common elements, the Atlas V system features improved structural capability allowing it to withstand worst-case day-of-launch winds. The result is increased launch availability.

APPENDIX B— MISSION SUCCESS AND PRODUCT ASSURANCE

B.1 PRODUCT DELIVERY SYSTEM

Lockheed Martin Space Systems Company (LMSSC) operates an AS9100 (Quality Management Systems-Aerospace-Requirements) registered quality management system. LMSSC is internationally accredited through the British Standards Institute (BSI) under registration number FM 35743. The registration was conferred in October 2004, and includes LMSSC's Denver, CO; San Diego, CA; Harlingen, TX; Cape Canaveral Air Force Station, and Vandenberg Air Force Base facilities. Scope of the registration includes 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 AS9100 quality standard is revalidated at 6-month intervals by BSI, an independent, third party registrar. In addition, AS9100 compliance is monitored and certified onsite by the U.S. government's Defense Contract Management Agency (DCMA), which also maintains insight into LMSSC processes.

AS9100 is executed through LMSSC's internal command media, which is described in the *Product Delivery System Manual* (PDSM). AS9100 is a basic quality management system that provides a framework of operating requirements. The PDSM addresses each requirement and provides an overview of the requirement and flowdown references to individual procedures. LMSSC also includes mission success as an additional requirement because of the deep focus on and commitment to mission success principles.

The PDSM management requirements and section numbers are:

4.0 Management Processes

4.1 Management Responsibility

4.2 Measuring and Monitoring

4.3 Document Control

4.4 Mission Success

4.5 Records Management

5.0 Project (Product Life Cycle) Processes

5.1 Develop and Acquire New Business

5.2 Design and Develop Product

5.3 Manufacture and Test Product

5.4 Assembly and Integration

5.5 Inspection and Testing

5.6 Nonconforming Material Control

5.7 Preparation for Shipment/Storage

5.8 Operations and Maintenance

5.9 Contract Closure

6.0 Support Processes

6.1 Resources

6.2 Customer Property

6.3 Measuring and Monitoring Equipment

6.4 ESH Management System

LMSSC is maintaining AS9100 registration to the latest released version of the standard.

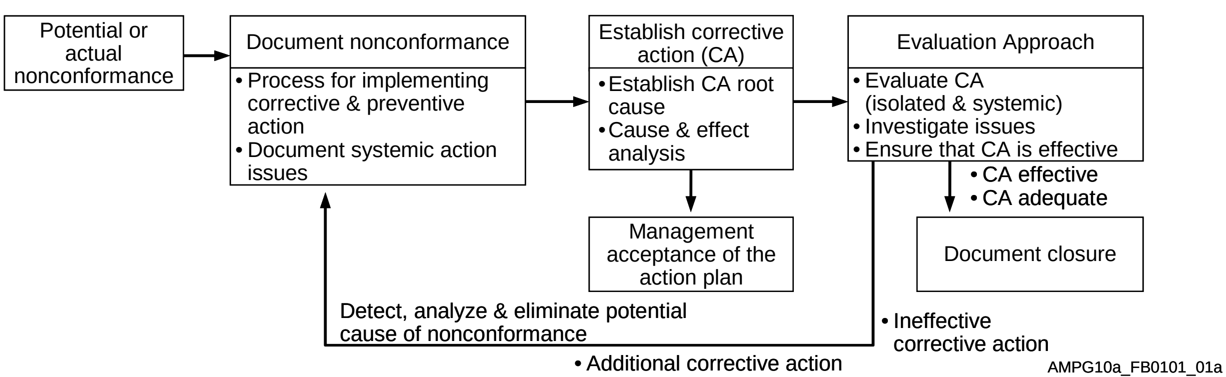
B.1.1 Product Assurance

Product Assurance (PA) ensures the quality of products and processes to achieve maximum effectiveness and continued stakeholder confidence.

Quality is ensured through physical examination, measurement, test, process monitoring, and/or other methods as required to determine and control the quality of all deliverable airborne, ground equipment products, software and services. Independent verification includes mandatory inspection points; witness, monitor, and surveillance; statistical methods; data analysis; trending; sampling plans; work instruction, build, and test documentation review; associated data reviews; process control; assessments; acceptance tooling; and/or other techniques suited to the products or processes being verified. Responsibilities are defined in the following paragraphs.

Corrective/Preventive Action — The corrective and preventive action process (Figure B.1.1-1) ensures visibility and resolution of anomalous conditions affecting or potentially affecting products, processes, or systems. This process encompasses customer concerns, internal activities, and supplier or subcontractor issues. The corrective and preventive 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 or preventive action is implemented and verified for effectiveness.

Figure B.1.1-1: Corrective and Preventive Action Process



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 (Figure B.1.3-1).

Parts, Materials, and Processes Control Board — The Parts, Materials, and Processes (PMP) Control Board (PMPCB) provides the method for ensuring the use of proven parts, materials, and processes across the 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 direction to order parts, assign component selection priorities, and assure mitigation of device lot failures.

Change Control — All engineering changes are managed and approved by an Engineering Review Board (ERB) and implemented by the Change Control Board (CCB). 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 requirements. 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 (paper and/or electronic) encompass purchasing, handling, machining, assembling, fabricating, processing, inspecting, testing, modifying, installing, and any other treatment of product, facilities, standards, or equipment. 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. Hardware is only 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, Procurement Quality Assurance Representative (PQAR) are assigned to certain suppliers and critical hardware to ensure hardware conformance to requirements during suppliers' 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 stamping, tags, routing cards, move tickets, build records, and/or other control devices. The overall product acceptance program is controlled by adherence to a program Product Acceptance Plan which details acceptance processes at Harlingen, San Diego, Denver, suppliers, Cape Canaveral Air Force Station (CCAFS) and Vandenberg Air Force Base (VAFB).

Identification and Stamp Control — Inspection, fabrication, workmanship (including physical or electronic stamps), signatures, 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 often necessary to determine if the nonconformance can be eliminated through rework, scrapping, or by returning hardware 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 Engineering organization, who is responsible for product design. All MRB members are required to be certified and approved by program Product Assurance.

Evaluation of Suspect Hardware — Hardware that may be affected by a potentially generic or systemic problem is controlled and dispositioned via the Suspect Product Evaluation Process. This process utilizes the nonconformance system to control the use of suspect items until the condition can be verified and an appropriate disposition is completed.

Software Quality — The Software Quality program is designed to ensure software products (code and documentation) are compliant with program, corporate, and customer requirements. Specifically, the program provides the methods necessary to:

1. Ensure approved processes and procedures are followed in design, development and testing of software products;

2. Ensure requirements are clear, quantifiable, and testable;

3. Ensure subcontracted software is developed according to approved processes and procedures;

4. Ensure Software Anomaly Reports (SAR) are tracked to closure and only approved changes are incorporated into controlled baselines.

The 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 independent technical evaluations, software testing, documentation verification, and management reviews necessary to achieve this goal.

Record Retention — All Product Assurance records are retained as required by contract, policy and procedures, and specific program direction. A secure product assurance data center is used as the central repository for Atlas quality-related data.

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 and/or reduce program costs, to ensure continued mission success, and to drive the program toward improvement. Examples of metrics maintained are:

1. Escapements

2. Recurring nonconformances

3. Supplier liability

4. Foreign object incidents

5. Aging of nonconformance documents

6. Open Items/Clean Vehicle

7. Training – Certification

8. Mishaps/Near Misses

9. Customer Satisfaction and Feedback

10. Software Quality Performance Metrics

11. System Safety Metrics

Product Assurance (Quality, System Safety and Software Quality) metrics are reviewed each month at the Atlas Product Assurance Operations Review (PAOR).

Calibration — A calibration system 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.1.2 Atlas Launch Operations Product Assurance

Atlas Launch Operations Product Assurance supports launch sites during launch vehicle processing, ground systems maintenance and installations, and checkout of modifications.

Product Assurance is the focal point for coordination of facility issues with VAFB and CCAFS 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.

All launch site operations are an extension of factory operations and are covered by the same requirements for reporting, control, and problem resolution.

B.1.2.1 Payload Operations

Payload Operations is responsible for 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 spacecraft integration at commercial and government processing facilities. This effort is inclusive of mating to payload adapters, encapsulation, radio frequency checkout, integration testing, and transportation to the launch pad.

B.1.2.2 Safety Operations

Safety operations include system safety engineering efforts and launch site operations safety support. To assure safety of personnel and hardware, our Safety Operations Group provides onsite support from hardware receipt through launch. Operational support includes consultation/coordination of industrial, material handling, fire protection, and environmental/chemical safety requirements; enforcement of Lockheed Martin as well as Range Safety policy; and oversight of selected activities.

System Safety reviews potential spacecraft design hazards/risks, assesses launch vehicle/spacecraft interfaces, and evaluates mission-specific ground processing operations to identify accident risks and develop the appropriate hazard controls. System Safety is responsible for coordination of technical issues with the 45th/30th Space Wing and timely completion of the Range Safety review and approval process (refer to Section 4.3).

B.1.3 Mission Success — Independent Oversight

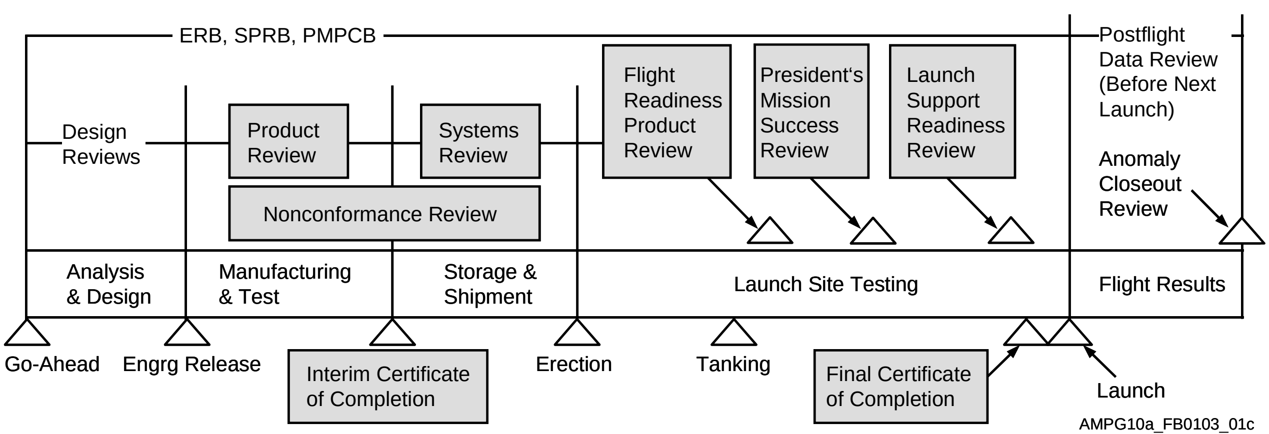
The Mission Success organization provides an evaluation for program and LMSSC management, independent from Engineering and Product Assurance, to ensure all potential mission impacts at Lockheed Martin and/or suppliers are resolved prior to launch. This is accomplished through their participation in program activities, such as tabletops and reviews, and through the reporting of all functional failures to the Mission Success organization. Mission Success Engineers (MSE) coordinate technical evaluations with appropriate subject-matter experts to establish mission impact. Failures with significant mission impact are presented to the Space Program Reliability Board (SPRB) to ensure complete analysis and effective remedial and corrective action is taken to mitigate mission impact. The SPRB is chaired by the Program Vice President, co-chaired by the Mission Success Manager, and is made up of technical experts who ensure complete investigation and resolution of all hardware concerns potentially affecting mission success. Mission Success is tasked to ensure all applicable flight constraints are resolved before launch.

Alerts — The Government-Industry Data Exchange Program (GIDEP) review process includes evaluating GIDEP alerts for impacts to hardware. All known alerts will be reviewed for impacts before each flight.

Audit — The Mission Success Organization also includes Internal Audit. Documented procedures are established and maintained for planning, performing, reporting, and follow-up of internal audits. The Space Systems Company Internal Audit Program coordinates with other audit organizations including corporate internal audit and customer audit teams to maximize effectiveness.

Reviews — The Mission Success organization participates in engineering, factory, data, and readiness reviews to support program management in determining hardware flight worthiness and readiness. A Mission Success Engineer (MSE) is assigned to each mission to ensure that vehicle noncompliance issues are appropriately resolved. The MSE also coordinates vehicle certificates of completion and flight constraint closeouts. The MSE follows the assigned vehicle throughout the production and launch sequence and reports on vehicle readiness in a series of program reviews. Figure B.1.3-1 is a flow of the acceptance process, including progressive reviews and acceptance throughout design, production, test, and launch site operations.

Figure B.1.3-1: Mission Success Acceptance Process



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 spacecraft's compatibility 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 by the spacecraft before a proposal is offered for Atlas V Launch Services. These lists are generalized and apply to any candidate mission. If Lockheed Martin has experience with the spacecraft bus or spacecraft contractor, less information can be provided initially (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.1-7 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, determine such items as optimum mission trajectory, and verify compatibility of launch vehicle environments and interfaces.

Table C.1-1: Spacecraft Information Worksheet (1 of 3)

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		
Spacecraft Mass	______ kg	______ lbm
Minimum Spacecraft Lifetime	______ yr	______ yr
Final Orbit Apogee	______ km	______ nmi
Final Orbit Perigee	______ km	______ nmi
Final Orbit Inclination	______ deg	______ deg
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	______ deg	______ deg
Right Ascension of Ascending Node Requirement	______ deg	______ deg
Apogee Accuracy Requirement	______ ±km	______± nmi
Perigee Accuracy Requirement	______ ±km	______ ±nmi
Inclination Accuracy Requirement	______ ±deg	______ ±deg
Argument of Perigee Accuracy Requirement	______ ±deg	______ ±deg
Right Ascension of Ascending Node Accuracy Requirement	______ ±deg	______ ±deg

Table C.1-1: Spacecraft Information Worksheet (2 of 3)

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.
Pre-separation RF Transmission Requirement	______ band	______ band
ELECTRICAL INTERFACE		
Spacecraft Electrical Drawing (near LV interface)		
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.
Fundamental Natural Frequency—Longitudinal	______ Hz	______ Hz
cg—Thrust Axis (Origin at Separation Plane)	______ mm	______ in.

Table C.1-1: Spacecraft Information Worksheet (3 of 3)

Spacecraft Design Parameter	SI Units	English Units
CONTAMINATION REQUIREMENTS		
Fairing Air Cleanliness	______ Class	______ Class
Maximum Deposition on Spacecraft Surfaces (particulate vs. molecular)	______ 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	______ deg	______ deg
Max Allowable Tip-Off Rate	______ deg/s	______ deg/s
Coefficients of Inertia—Ixx (x=Thrust Axis)	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Ixx Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Iyy	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Iyy Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Izz	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Izz Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Ixy	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Ixy Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Iyz	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Iyz Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Ixz	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Ixz Tolerance	______ ±kg m^2	______ ±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 and Flight Constraints Pre-separation Function*	• Acceleration Constraints (Pitch, Yaw, Roll) • Attitude Constraints • Spinup Requirements • Prearm • Arm • Spacecraft Equipment Deployment Timing and Constraints
Separation Parameters (Including Tolerances)*	• Desired Spin Axis • Angular Rate of Spacecraft • Orientation (Pitch, Yaw and Roll Axis) • Origin of Coordinate System (Location) • Acceleration Constraints
Any Special Trajectory Requirements	• Boost Phase • Coast Phase • Free Molecular Heating Constraints • Thermal Maneuvers • Separation Within View of Telemetry and Tracking Ground Station • Telemetry Dipout Maneuvers • Real Time Retransmission
Note: * Information Desired at Initial Meeting Between Lockheed Martin and Customer after Contract Award	

Table C.1-3: Spacecraft Characteristics (1 of 3)

Type of Data	Scope of Data
Configuration Drawings* Apogee Kick Motor*	• Drawings Showing the Configuration, Shape, Dimensions and Protrusions near the Mechanical Interface (Ground Launch and Deployment Configurations) • Coordinates (Spacecraft Relative to Launch Vehicle) • Special Clearance Requirements • Manufacturer's Designation • Thrust • Specific Impulse • Burn Action Time • Propellant Offload Limit
Mass Properties (Launch and Orbit Configurations)*	• Weight — Specify Total, Separable & Retained Masses • Center of Gravity — Specify in 3 Orthogonal Coordinates Parallel to the Booster Roll, Pitch and Yaw Axes for Total, Separable and Retained Masses • Changes in Center of Gravity Due to Deployment of Appendages • Propellant Slosh Models
Moments and Products of Inertia (Launch and Orbit Configurations)	• Specify About the Axes Through the Spacecraft Center of Gravity That Are Parallel to the Atlas Roll, Pitch and Yaw Axes for Total, Separable, and Retained Masses

Table C.1-3: Spacecraft Characteristics (2 of 3)

Type of Data	Scope of Data
Structural Characteristics	• Spring Ratio of Structure • Elastic Deflection Constants • Shear Stiffness • Dynamic Model • Bending Moments and Shear Loads at Atlas/Centaur/Spacecraft Interface • Limitations, Include Acoustic, Shock, Acceleration, Temperature and Bending Moments
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 and Coordinate System • Loads Transformation Matrix • Note: Models Must Include Rigid Body and Normal Modes
Handling Constraints	• Spacecraft Orientation During Ground Transport • Spacecraft Handling Limits (e.g., Acceleration Constraints)
Spacecraft Critical Orientations:	• Location and Direction of Antennas during Checkout, Prelaunch and Orbit • Location, Look Angle and Frequency of Sensors • Location and 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 and 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 • Ground Support Equipment (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 Safety Data Submittal—Identify Each Item/Operation Applicable to PPF, HPF, or Launch Site as Applicable	

Table C.1-3: Spacecraft Characteristics (3 of 3)

Type of Data	Scope of Data
Thermal Characteristics	• Spacecraft Thermal Math Model (Ref Sect. C.3.3) • Emissivity • Conductivity • Resistivity • Thermal Constraints (Maximum and Minimum Allowable Temperatures) • Heat Generation (e.g., Sources, Heat Flux, Time of Operation)
Contamination Control	• Requirements for Ground-Supplied Services • In-Flight Conditions (e.g., During Ascent and After PLF Jettison) • Surface Sensitivity (e.g., Susceptibility to Propellants, Gases, and Exhaust Products)
RF Radiation	• Characteristics (e.g., Power Levels, Frequency, and Duration for Checkout and Flight Configuration) • Locations (e.g., Location of Receivers and Transmitters on Spacecraft) • Checkout Requirements (e.g., Open-Loop, Closed-Loop, Prelaunch, Ascent Phase)
Note: * Information Desired at Initial Meeting Between Lockheed Martin and Customer After Contract Award	

Table C.1-4: Interface Requirements (Mechanical)

Type of Data	Scope of Data
Mechanical Interfaces *	• Base Diameter of Spacecraft Interface* • Structural Attachments at Spacecraft 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 and Location • Timeline for Pressure/Fuel System Operation • Spacecraft/Adapter Venting Requirements
PLF Requirements	• Heating Constraints • Venting Characteristics (e.g., Quantity, Timing and Nature of Gases Vented from Spacecraft) • RF Reradiation System (RF Band, Spacecraft Antenna Location, etc) • PLF Separation (e.g., Altitude, Cleanliness, Shock, Aeroheating and Airload Constraints) • Acoustic Environment Constraints • Special Environmental Requirements
Preflight Environment	• PLF Separation (e.g., Altitude, Cleanliness, Shock, Aeroheating and Airload Constraints) • Acoustic Environment Constraints • Special Environmental Requirements • Requirements – Cleanliness – Temperature and Relative Humidity – Air Conditioning – Air Impingement Limits • Monitoring and 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 and Customer After Contract Award	

Table C.1-5: Interface Requirements (Electrical)

Type of Data	Scope of Data
Power Requirements (Current, Duration, Function Time and Tolerances)*	• 28-Vdc Power • Other Power • Overcurrent Protection
Command Discrete Signals*	• Number* • Sequence • Timing (Including Duration, Tolerance, Repetition Rate, etc) • Voltage (Nominal and Tolerance) • Frequency (Nominal and Tolerance) • Current (Nominal and Tolerance) • When Discretes Are for EED Activation, Specify: – Minimum, Maximum and Nominal Fire Current; – Minimum and Maximum Resistance; – Minimum Fire Time; – Operating Temperature Range and – 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*	• Spacecraft Measurements Required To Be Transmitted by Atlas Telemetry: – Quantity – Type of Measurements (e.g., Temperature, Vibration, Pressure, etc); – Details Concerned with Related System Including Operating Characteristics – (Response Definition of System) and Locations and Anticipated Time of Operation; Impedance, Capacitance, Operating Range and Full-Scale Range of Each Measurement • Signal Conditioning Requirements (e.g., Input Impedance, Impedance Circuit Load Limits, Overcurrent Protection and Signal-to-Noise Ratio) • Discrete Events (Bilevel) • Analog Measurements • Transducers Required To Be Furnished by Launch Vehicle Contractor • Minimum Acceptable Frequency Response for Each Measurement • Minimum Acceptable System Error for Each Measurement (Sampling Rate Is Also Governed by This Requirement) • Period of Flight for Which Data from Each Measurement Are of Interest (e.g., from Liftoff to Spacecraft Separation)* • Atlas Flight Data Required by Spacecraft Contractor
Bonding	• Bonding Requirements at Interface (MIL-B-5087, Class R for Launch Vehicle) • Material and Finishes at Interface (for Compatibility with Launch Vehicle Adapter)
EMC	• Test or Analyze Spacecraft Emissions and Susceptibility • EMC Protection Philosophy for Low-Power, High-Power and Pyrotechnic Circuits • Launch Vehicle and Site Emissions (Provided by Lockheed Martin)
Grounding Philosophy	• Structure (e.g., Use of Structural As Ground and Current Levels) • Electrical Equipment (e.g., Grounding Method for Signals and Power Supplies) • Single-Point Ground (e.g., Location and Related Equipment)
Interface Connectors	• Connector Item (e.g., Location and 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 and Launch Operations (1 of 2)

Type of Data	Scope of Data
Spacecraft Launch Vehicle Integration	• Sequence from Spacecraft Delivery Through Mating with the Launch Vehicle • Handling Equipment Required • Lockheed Martin-Provided Protective Covers or Work Shields Required • Identify the Space Envelope, Installation, Clearance, and Work Area Requirements • Any Special Encapsulation Requirements Support Services Required
Spacecraft Checkout AGE & Cabinet Data	• List of All AGE and Location Where Used (e.g., Storage Requirements on the Launch Pad) • Installation Criteria for AGE Items: – Size and Weight – Mounting Provisions – Grounding and Bonding Requirements – Proximity to the Spacecraft When In Use – Period of Use – Environmental Requirements – Compatibility with Range Safety Requirements and Launch Vehicle Propellants – Access Space to Cabinets Required for Work Area, Door Swing, Slideout Panels, etc – Cable Entry Provisions & Terminal Board Types in Cabinets and/or Interface Receptacle Locations and Types – Power Requirements and 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 Spacecraft Is Not Compatible with Launch Vehicle Propellants and What Safety Measures Will Be Required • Environmental Monitoring and Verification Requirements
Space Access Requirements	• Access for Spacecraft Mating and Checkout • Access During Transportation to the Launch Pad and Erection Onto the Atlas V • Access for Checkout and Achieving Readiness Prior to Fairing Installation • Access After Fairing Installation; State Location, Size of Opening and Inside Reach Required • Access During the Final Countdown, if Any • AGE Requirements for Emergency Removal
Umbilicals	• Ground Servicing Umbilicals by Function and Location in Excess of Atlas/Centaur Baseline • Structural Support Requirements and Retraction Mechanisms • Installation (e.g., When and by Whom Supplied and Installed)
Commodities Required for Both Spacecraft, AGE & Personnel	• Gases, Propellants, Chilled Water and Cryogenics in Compliance with Ozone-Depleting Chemicals Requirements • Source (e.g., Spacecraft or Launch Vehicle) • Commodities for Personnel (e.g., Work Areas, Desks, Phones)
Miscellaneous	Spacecraft Guidance Alignment Requirements

Table C.1-6: Test and Launch Operations (2 of 2)

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 • Launch Vehicle / Spacecraft RF Interface Test • Environmental Demonstration Test
Launch Operations	• Detailed Sequence & Time Span of All Spacecraft-Related Launch Site Activities Including: – AGE Installation, – Facility Installation and Activities, – Spacecraft Testing and Spacecraft Servicing • Recycle Requirements • Launch Operations Restrictions Including: – Launch Site Activity Limitations, – Constraints on Launch Vehicle Operations, – Security Requirements – Personnel Access Limitations and Safety Precautions • Special Requirements Include Handling of Radioactive Materials, Security and Access Control • Support Requirements To Include Personnel, Communications and Data Reduction • Launch and 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	

Table C.1-7: Ground Equipment and Facility Requirements (Electrical)

Type of Data	Scope of Data
Spacecraft Electrical Conductor Data	S/C System Schematic Showing All Connectors Required Between S/C Equipment, & S/C 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 Requirements (e.g., Function, Location, Physical Characteristics, Beam Width & Direction & Line-of-Sight) • Frequency & Power Transmission • Operation
Cabling	• All Cabling, Ducting, or Conduits To Be Installed in the Mobile Service Tower; • Who Will Supply, Install, Checkout & Remove
Monitors & Controls	• Specify Which Signals from Spacecraft Are To Be Monitored During Readiness & Countdown; • Specify Signal Power Source (Spacecraft, Atlas V, Centaur) • Transmission Method (e.g., Spacecraft Telemetry, Launch Vehicle Telemetry, Landline, or Launch Vehicle Readiness Monitor) • Location of Data Evaluation Center, Evaluation Responsibility, Measurement Limits and Go/No-Go Constraints; • Identify Where in the Operational Sequence Measurements Are To Be Monitored and Evaluated; • Specify Frequency and Duration of Measurements • Video Output Characteristics of Telepaks (if Available) for Closed-Loop Prelaunch Checkout at the Launch Pad; • Data To Include Location and Type of Interface Connector(s), and Characteristics of Signal at Source; • This Includes Voltage Level, Output Impedance, Output Current Limitation, Maximum Frequency of Data Train and Output Loading Requirements

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. Lockheed Martin will work with the customer to resolve the incompatibility should the spacecraft not meet any of these requirements.

Table C.2-1: Spacecraft Design Elements to Be Certified by Analysis or Test

Spacecraft Design Requirement	Comment
Mechanical	
• Payload Fairing Envelope	Appendix D
• Payload Adapter Envelope	Appendix D
• Payload Adapter Interface	Appendix E
Electrical	
• Two or Fewer Separation Commands	Section 4.1.3
• 16 or Fewer Control Commands (28-V Discretes or Dry Loop)	Section 4.1.3
• Instrumentation Interface, 2 or Fewer Inputs for S/C 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 S/C Data	Section 4.1.3
• Two Umbilical Connectors at S/C Interface	Figure 4.1.3-1
Structure & Loads	
• Design Load Factors	Table 3.2.1-1 and Figure 3.2.1-1
• First Lateral Modes Above 10 Hz & First Axial Mode Above 15 Hz	Section 3.2.1
• Spacecraft Mass vs cg Range	Appendix E
• Design FS per Applicable Range Safety Documentation & MIL-STD-1522 (or Submit Deviations for Review)	
Environment	
• Spacecraft Test Requirements	Section 3.3
• Quasi-Sinusoidal Vibration	Figures 3.2.3-1
• Acoustic Levels in the PLF	Figures 3.2.2-1 Through 3.2.2-3
• Shock Induced by PLF Jettison & Spacecraft Separation	Figure 3.2.4-1 Through 3.2.4-3
• Payload Compartment Pressures & Depressurization Rates	Figures 3.2.6-1 Through 3.2.6-4
• Gas Velocity Across S/C Components 9.75 m/s (32 ft/s)	4-m PLF
• Gas Velocity Across S/C Components 10.67 m/s (35 ft/s)	5-m PLF
• Electric Fields	Figures 3.1.2.1-1 Through 3.1.2.2-3
• Spacecraft Radiation Limit	Figure 3.1.2.4-1
• EM Environment at Launch Range	TOR-95(5663)-1 & Section 3.1.2.3
Safety	
• All Spacecraft Propellant Fill and Drain Valves, All Pressurant Fill and Vent Valves Readily Accessible When Spacecraft Is Fully Assembled and Serviced in Launch Configuration (Encapsulated and on Launch Pad)	It Is Advisable To Accommodate Normal Servicing/Deservicing and Potential Emergency Backout Situation for New Spacecraft Design
• Requirements in Range Safety Regulation	
Miscellaneous	
• See Atlas Launch Services Facilities Guide for Spacecraft Propellants and 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.

Table C.3-1: Spacecraft Inputs to Integration Process

Spacecraft Data Input	Typical Need Date	Comments
Interface Requirements Document	Program Kickoff	See Section C.1
Initial Target Specification	Program Kickoff	Spacecraft Weight, Target Orbit, Separation Attitude; See Section C.3.4
Range Safety Mission Orientation Briefing Input	Program Kickoff	Top-Level Description of Spacecraft & Mission Design
Prelim Spacecraft MSPSP	Program Kickoff	See Section C.3.6.1
Intact Impact Breakup Data	Program Kickoff	See Section C.3.6.4.2
In-Flight Breakup Data	Program Kickoff	See Section C.3.6.4.3
CAD Model	30 days After Program Kickoff	See Section C.3.1
Procedures Used at Astrotech	2 month Before S/C Arrival	See Section C.3.6.2
Procedures Used on CCAFS	4 month Before Launch	See Section C.3.6.2
Thermal Models	5 month Before Design Review	See Section C.3.3
Preliminary Launch Windows	5 month Before Design Review	Support Thermal Analysis; See Section C.3.3
Coupled Loads Model	6 month Before Design Review	See Section C.3.2
Spacecraft EMI/EMC Cert Letter	6 month Before Launch	See Section C.3.5
Spacecraft EED Analysis	6 month Before Launch	See Section C.3.5
Final Target Specification	90 days Before Launch	Date Depends on Mission Design; See Section C.3.4
Spacecraft Environment Qualification Test Reports	As Available	See Table C.2-1 for Environment Qualification Requirements

C.3.1 Computer-Aided Design Data Transfer Requirements

The Atlas program uses both the UNIX and MS Windows based operating systems and supports two Computer Aided Design (CAD) software programs: Parametric Technology Corporation (PTC) Pro-Engineer (Pro-E) and Structural Dynamics Research Corporation (SDRC) I-DEAS Master Series. CAD data should be provided according to these specified software formats. When CAD data does not come from these supported software platforms, Lockheed Martin prefers to receive solid model data translated through the Standard for the Exchange of Product Model Data (STEP) converter. An alternative to this is an Initial Graphics Exchange Specification (IGES) 4.0 or higher file from a three-dimensional (3-D) surface model or wireframe extracted from a solid model.

C.3.1.1 Prerequisites to Data Transfer

The following criteria should be met before transferring CAD data:

1. The spacecraft contractor should verify that the data files contain the desired results by reading them back onto the originating CAD system from the source file before transmittal to Lockheed Martin.

2. Provide entire representation of all external spacecraft components for best integration to the Atlas Launch Vehicle. All internal structures are not necessary and should be removed from model transfer files.

3. Write out STEP and IGES files as assemblies and not as a single part file.

If feasible, the entire directory should be compressed and transferred as a single file using (UNIX) Tar (tar cvf/dev/rmt0 part name), or (Windows) “WinZip” or equivalent.

C.3.1.2 Data Transfer

Compact Disk media and/or File Transfer Protocol (FTP) are the preferred transfer methods for all data files. An account can be established on a Lockheed Martin firewall server for FTP data transfers. Once the account is set up and a password is provided for access, up to 1.5GB of data can be transmitted at one time. An alternative method 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. Proprietary or sensitive data should be encrypted using PGP keys or equivalent. Because of security concerns email transfers are not recommended at this time. If CDROM or FTP transfer methods are not feasible, contact appropriate Lockheed Martin personnel to provide a coordinated and acceptable method of data transfer.

The following information must be sent with the CAD data regardless of transfer method:

1. Name and phone number of the contact person who is familiar with the model in case problems or questions arise

2. Spacecraft axis and coordinate system

3. Spacecraft access requirements for structure not defined on CAD model (i.e. fill and drain valve locations)

4. Multiview plot of model

5. 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 spacecraft dynamic model should have an upper frequency cutoff of 90 to 100 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 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 electronically or on a computer diskette with printed listings of all the files. The Geometric Mathematical Model (GMM) and Thermal Mathematical Model (TMM) size should be less than 600 nodes each.

The preferred GMM format is Thermal Radiation Analysis System (TRASYS) input format. Alternate formats are ESABASE or NEVADA input formats. The documentation of the GMM should include illustrations of all surfaces at both the spacecraft and component levels, descriptions of the surface optical properties, and the correspondences between GMM and TMM nodes.

The preferred TMM format is System-Improved Numerical Differencing Analyzer (SINDA). The TMM documentation should include illustrations of all thermal modeling; detailed component power dissipations for prelaunch, ascent, and on-orbit mission phases; steady-state and transient test case boundary conditions, 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 and Electromagnetic Compatibility Certification Letter and Electroexplosive Device 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 likelihood 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 for the combined spacecraft/launch vehicle and site environment.

C.3.6 Safety Data

To launch from Cape Canaveral Air Force Station (CCAFS) on the Eastern Range or Vandenberg Air Force Base (VAFB) on the Western Range, spacecraft design and ground operations must meet the applicable launch-site safety regulations. Refer to Section 4.3.1 for a listing of these regulations. Mission-specific schedules for development and submittal of the spacecraft safety data will be coordinated in safety working group meetings during the safety integration process. Refer to Section 4.3.2 for additional information on this process.

C.3.6.1 Missile System Prelaunch Safety Package

The Missile System Prelaunch Safety Package (MSPSP) is the data package that describes in detail the hazardous and safety-critical spacecraft systems/subsystems, their interfaces, and the associated Ground Support Equipment (GSE). In addition, the Spacecraft MSPSP provides verification of compliance with the applicable Range Safety requirements. The spacecraft MSPSP must be approved by Range Safety before the arrival of spacecraft elements at the launch site.

C.3.6.2 Spacecraft Launch Site Procedures

Before any procedures are performed at the launch site, hazardous spacecraft procedures must be approved by the Range Safety Office and/or the safety organization at the appropriate spacecraft processing facility (e.g., Astrotech, NASA, DoD). Since the approving authority must also concur with the nonhazardous designation of procedures, all spacecraft launch-site procedures must be submitted for review. Lockheed Martin’s System Safety group is the point of contact for submittal/coordination of all spacecraft data (refer to Section 4.3.2)

C.3.6.3 Radiation Protection Officer Data

Permission must be received from the Range Radiation Protection Officer (RPO) before spacecraft Radio Frequency (RF) emissions are allowed at the launch complex. The required RPO data includes descriptions of the equipment involved, the procedures that will be used, and information on the personnel who will be running the procedures.

C.3.6.4 Spacecraft Breakup Data Requirements

The spacecraft data described in the following three subsections is required for the Atlas V program to complete mission-specific analyses that satisfy 45th Space Wing/SEOE and 30th Space Wing/SESE requirements for submitting a request for Range Safety Flight Plan Approval (FPA).

C.3.6.4.1 Inadvertent Spacecraft Separation and Propulsion Hazard Analysis

This data set is related to inadvertent separation of the spacecraft during early ascent and the potential for launch area hazards that could exist in the event 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 data set is related to the ground impact of the spacecraft. The intact impact analysis assumes ground impact of a fully loaded, fueled, intact spacecraft. It also assumes propellants will 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 in an explosion; and the location, size, weight, and shape of each piece.

C.3.6.4.3 Destruct Action Analysis

This data set 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 because of commanded vehicle destruction and estimates of their size, weight, shape, and location on the spacecraft.

C.3.7 Spacecraft Propellant Slosh Modeling

Lockheed Martin models spacecraft propellant slosh as part of the launch vehicle attitude control system stability analysis. The spacecraft propellant tank geometry, tank locations, minimum and maximum tank fill levels, propellant densities and propellant slosh damping ratios are required to perform this analysis. The data will be documented in the mission specific ICD.

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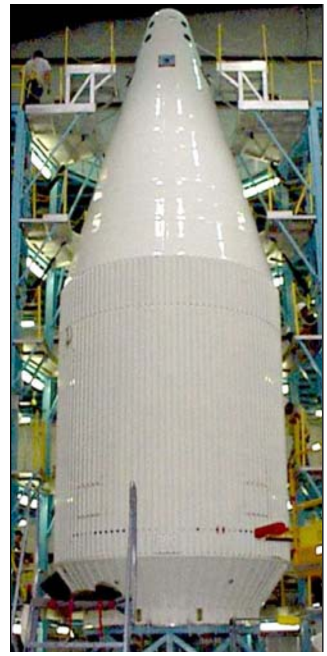
APPENDIX D— ATLAS V PAYLOAD FAIRINGS

D.1 ATLAS V 4-M PAYLOAD FAIRINGS

The Atlas V 4-m Large Payload Fairing (LPF), Extended Payload Fairing (EPF) and Extended EPF (XEPF) have a 4.2-m (165-in.) outer skin line diameter cylindrical section. The major sections of these payload fairings are the boattail, the cylindrical section, and the conical section that is topped by a spherical cap (Figures D.1-1 and D.1-2). The EPF was developed to support launch of larger volume spacecraft by adding a 0.9-m (36-in.) long cylindrical plug to the top of the cylindrical section of the LPF. The XEPF is a modified version of the EPF that incorporates an additional 0.9-m (36 in) high cylindrical plug to further increase the available payload volume. All of these sections consist of an aluminum skin, stringer, and frame construction with vertical, split-line longerons that allow the fairing to separate into bisectors for jettison. Electrical packages required for the fairing separation system are mounted on the internal surface of the boattail. Ducting for the Centaur upper-stage hydrogen tank venting system and cooling ducts for the equipment module packages are also attached to the boattail.

Figure D.1-2: Atlas V 4-m Payload Fairing



D.1.1 Static Payload Envelope

The usable volume for a spacecraft inside the Payload Fairing (PLF) is defined by the static payload envelope. The Atlas V 4-m PLF provides a 3,750-mm (147.64-in.) diameter envelope in the cylindrical section, as shown in Figure D.1.1-1, with additional volume available in the conical section of the PLF. On a mission-specific basis, the envelope may be increased to a 3,850-mm (151.57-in.) diameter in localized areas by modifying portions of the PLF structure. This envelope represents the maximum allowable spacecraft static dimensions (including) manufacturing tolerances) relative to the spacecraft/payload adapter interface. These envelopes include allowances for PLF static tolerances and misalignments, PLF and spacecraft dynamic deflections, and PLF out-of-round conditions, and were established to insure that a minimum 25-mm (1-in.) clearance between the spacecraft and the PLF is maintained. These envelopes were developed and are applicable for spacecraft that meet the stiffness and load requirements discussed in Section 3.2.1. Clearance layouts and analyses are performed for each spacecraft configuration and, if necessary, critical clearance locations are measured after the spacecraft is encapsulated inside the fairing to verify positive clearance during flight. Detailed views of the static payload envelope for the LPF, EPF and XEPF are shown in Figures D.1.1-2 to D.1.1-4.

Figure D.1.1-1: Atlas V 4-m PLF Static Payload Envelope, 3,850-mm Diameter

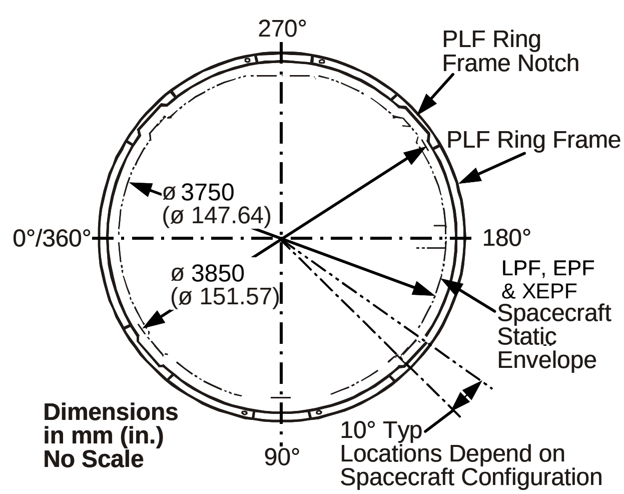
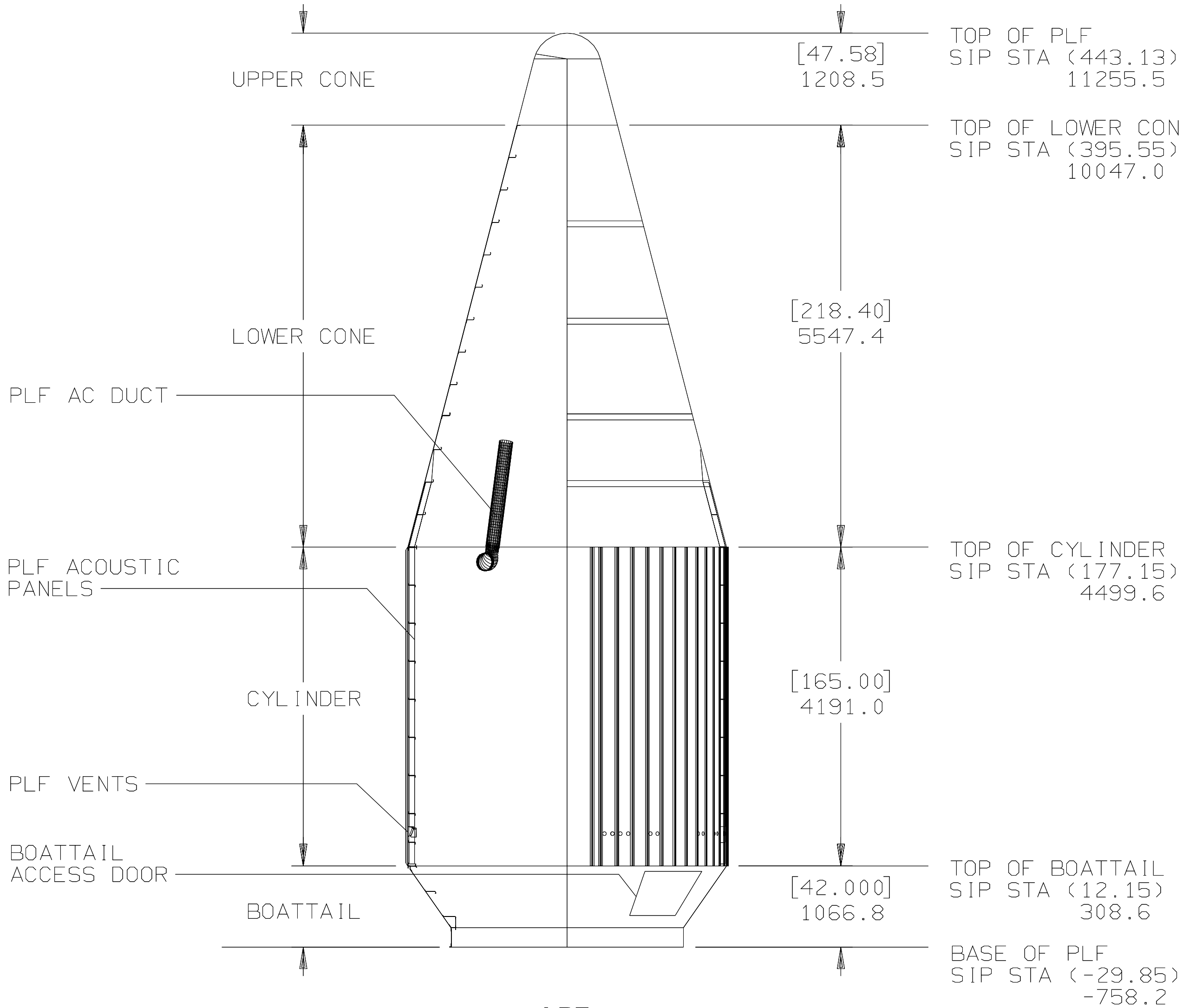


Figure D.1-1: Atlas V 4-m Payload Fairing (1 of 3)



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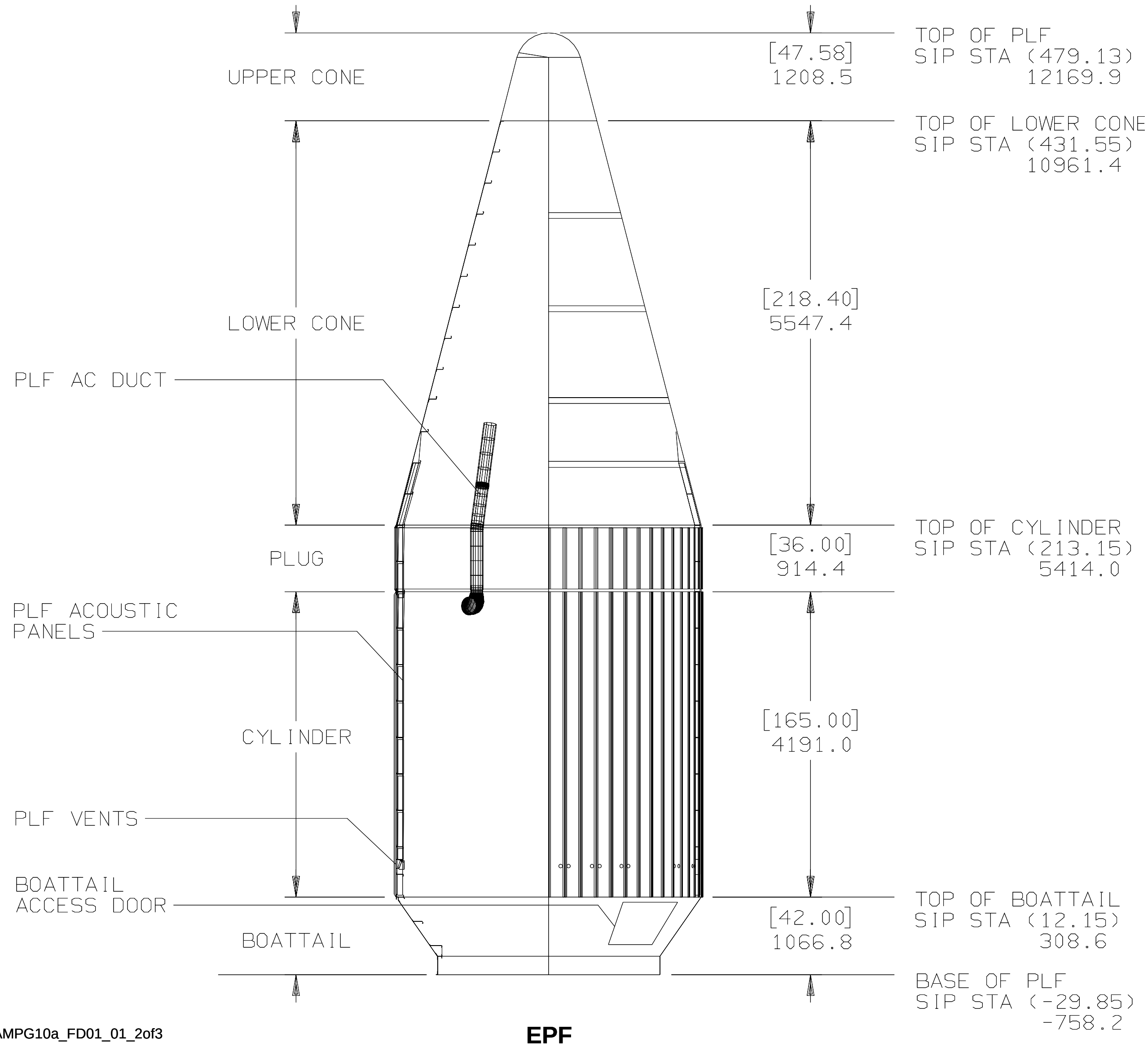
LPF

Figure D.1-1: Atlas V 4-m Payload Fairing (2 of 3)



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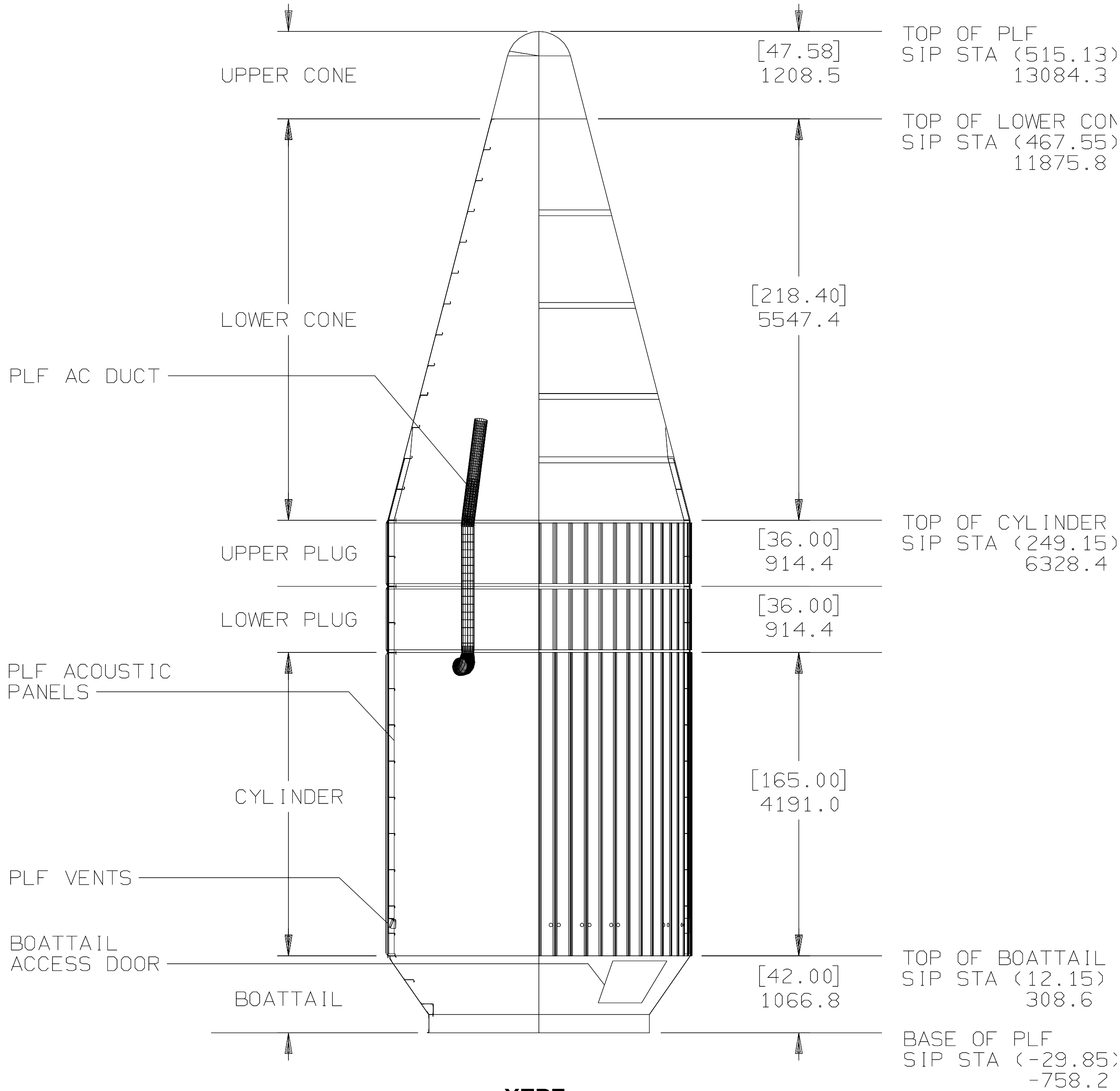
EPF

Figure D.1-1: Atlas V 4-m Payload Fairing (3 of 3)



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XEPF

Figure D.1.1-2: Atlas V 4-m LPF Static Payload Envelope (1 of 2)

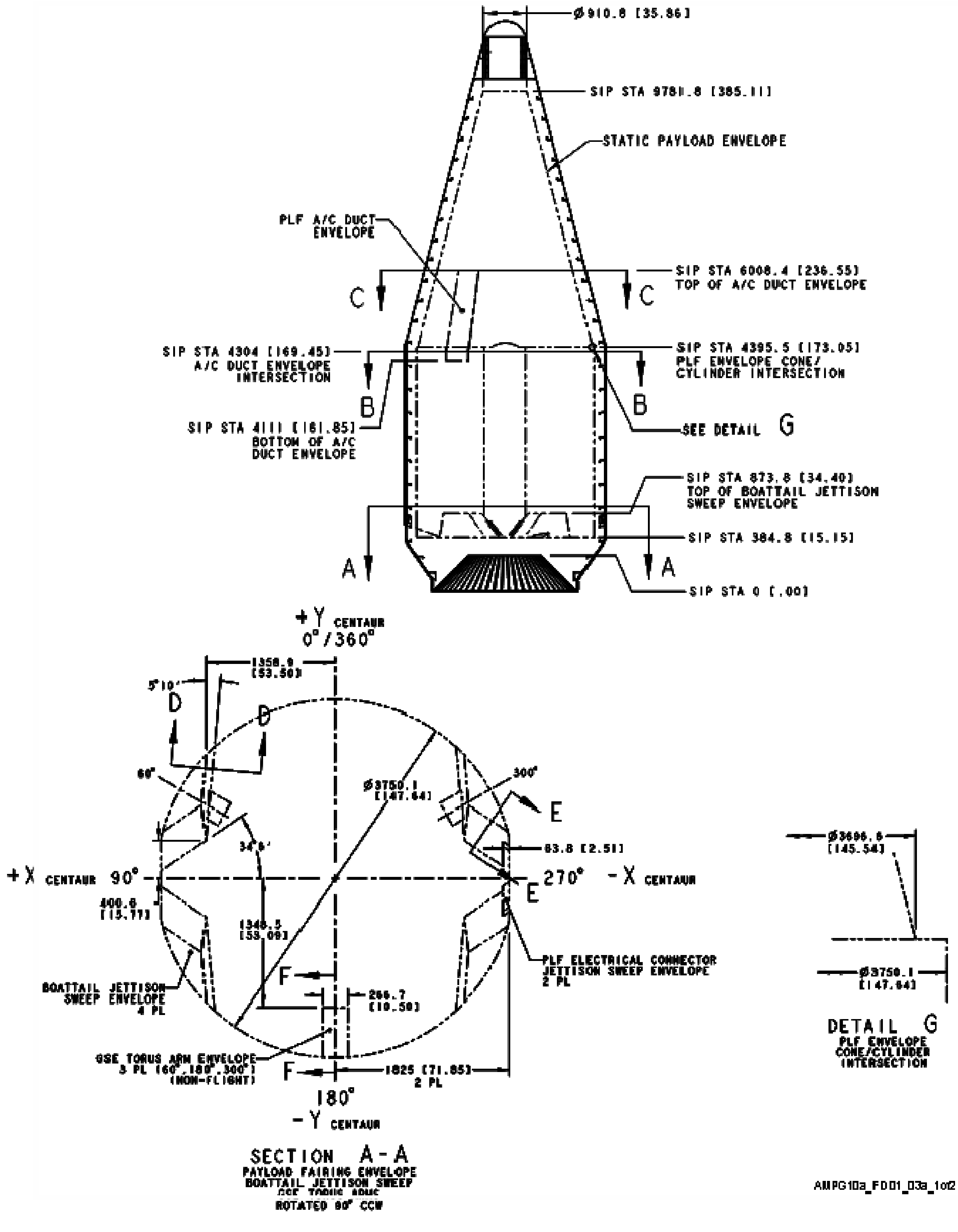


Figure D.1.1-2: Atlas V 4-m LPF Static Payload Envelope (2 of 2)

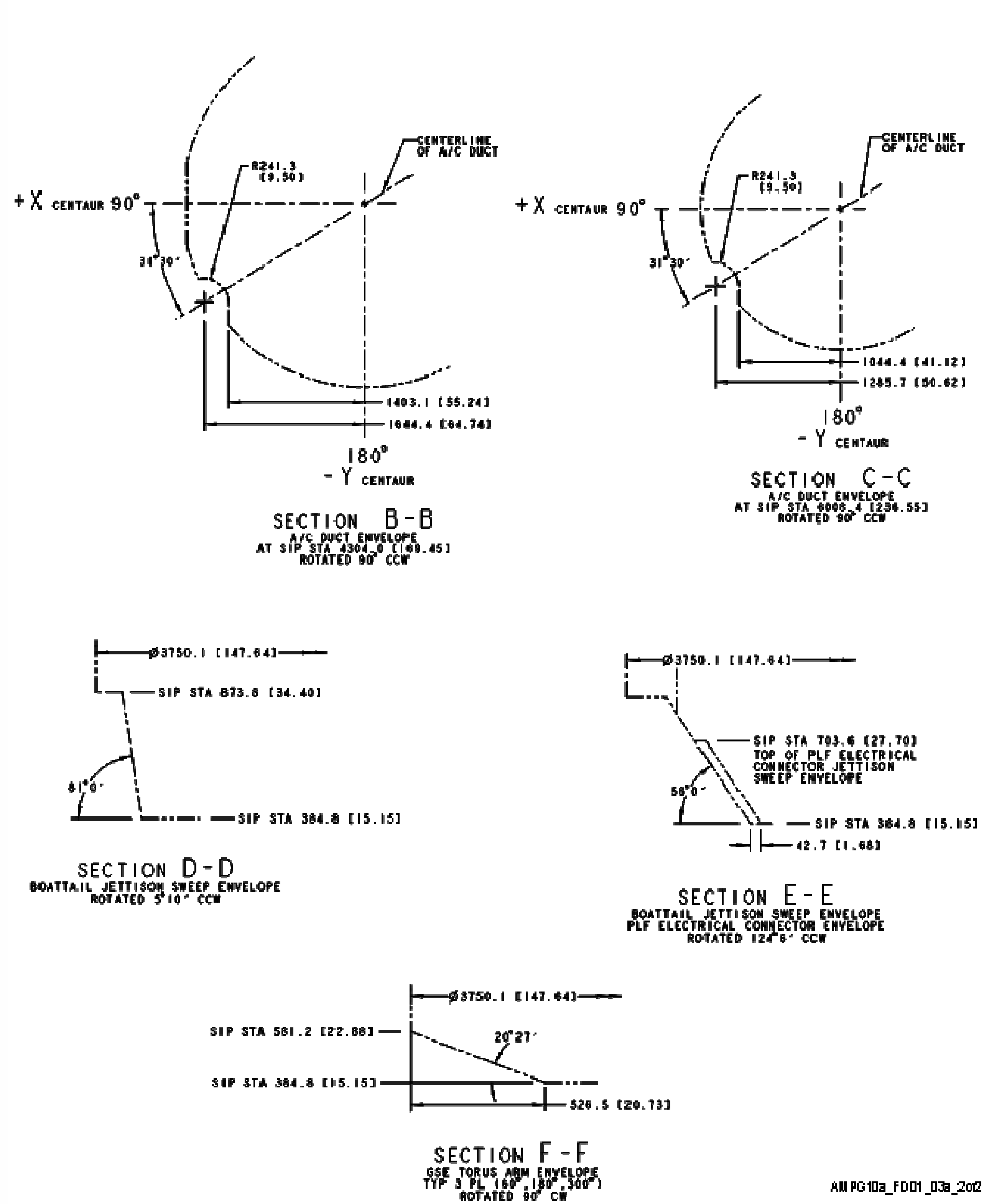


Figure D.1.1-3: Atlas V 4-m EPF Static Payload Envelope (1 of 2)

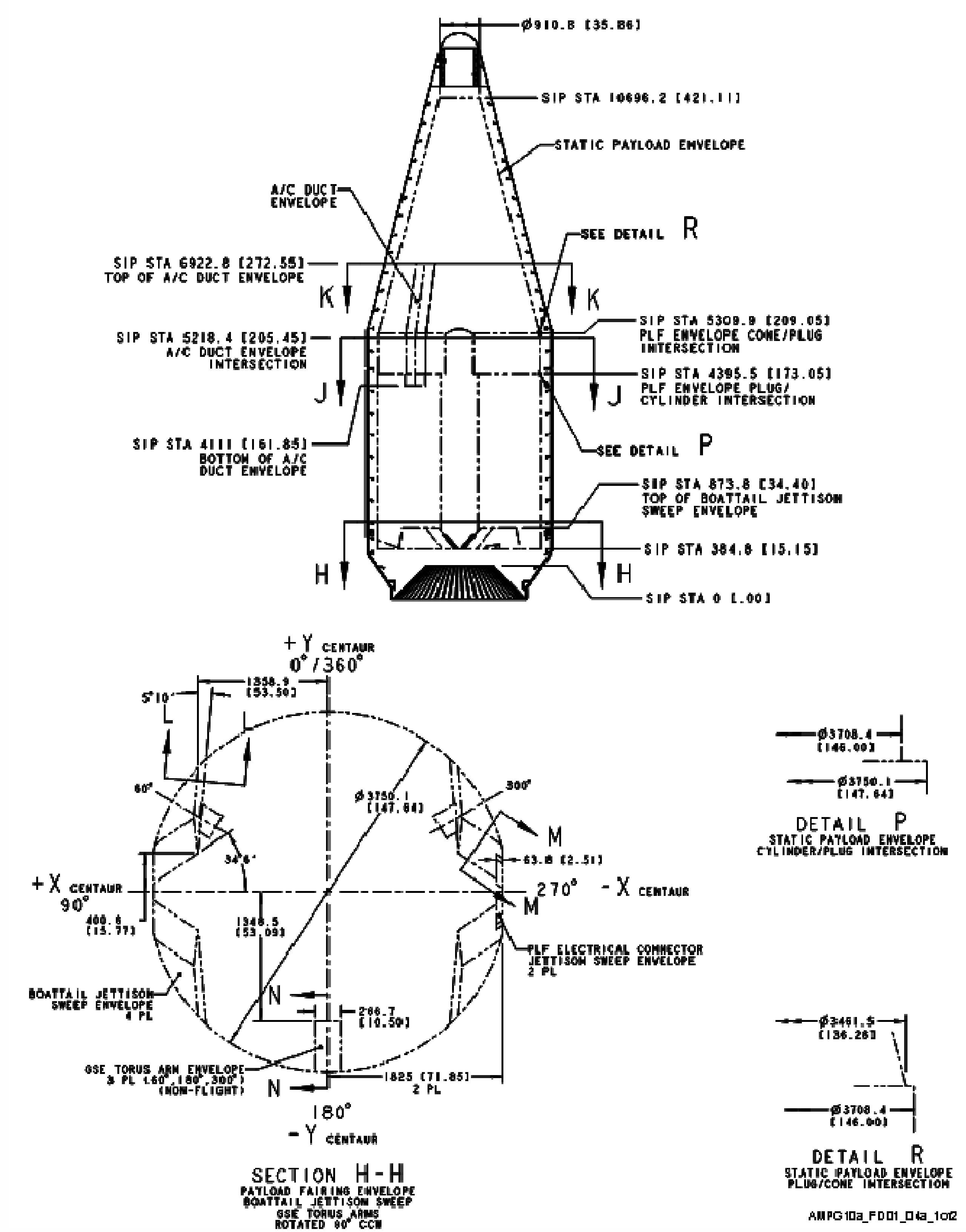


Figure D.1.1-3: Atlas V 4-m EPF Static Payload Envelope (2 of 2)

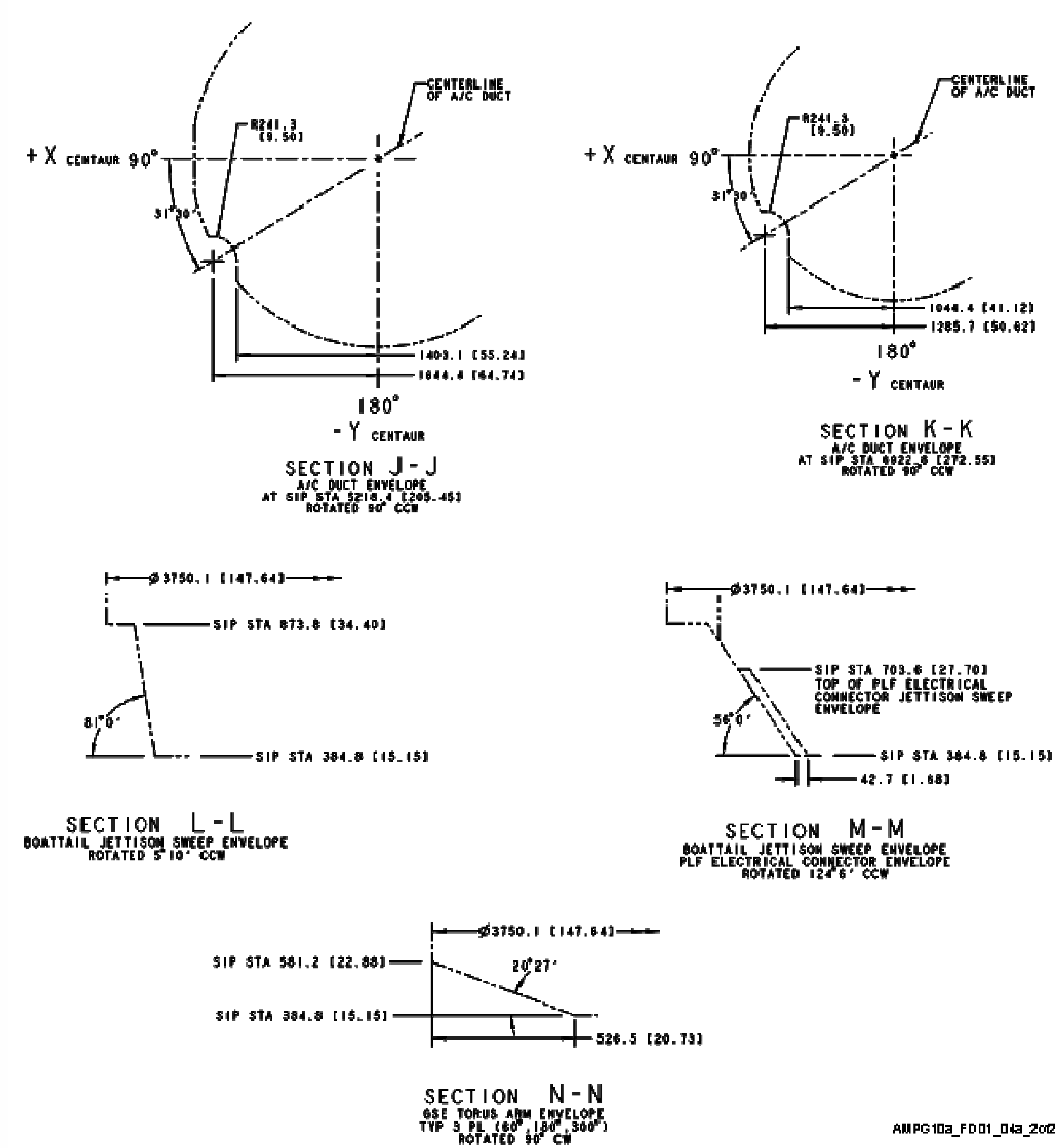


Figure D.1.1-4: Atlas V 4-m XEPF Static Payload Envelope (1 of 2)

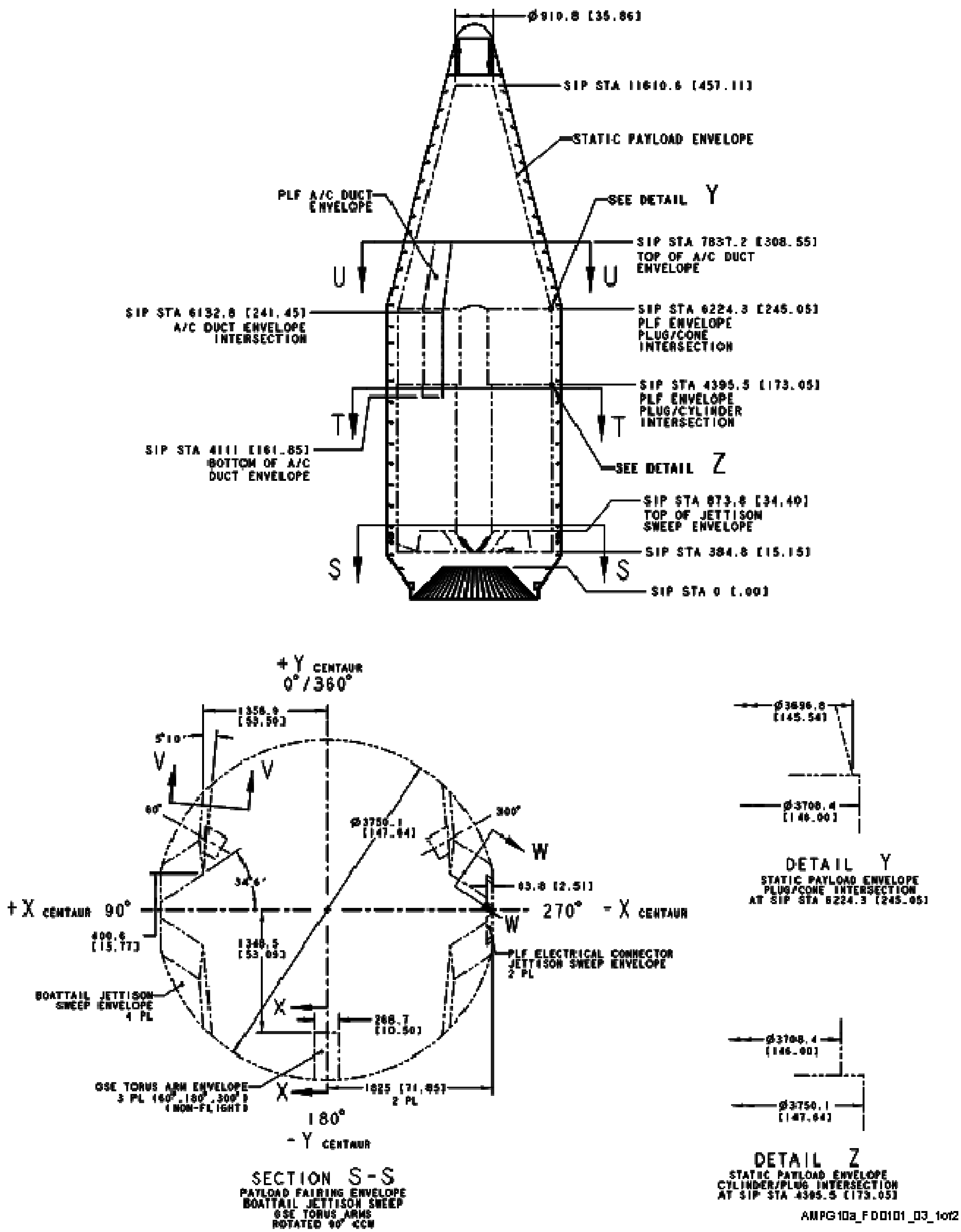
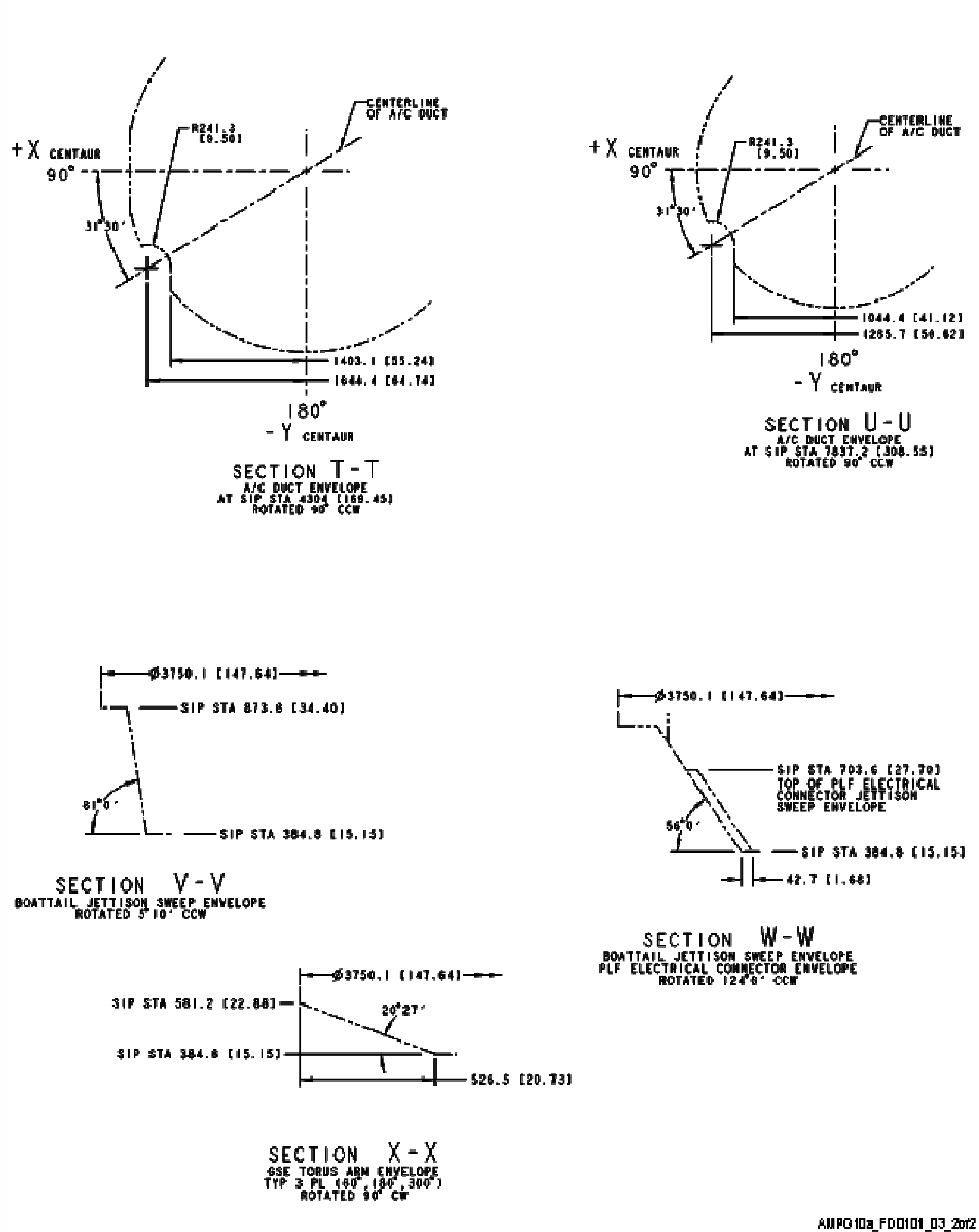


Figure D.1.1-4: Atlas V 4-m XEPF Static Payload Envelope (2 of 2)



For customers that request a dynamic payload envelope, the static payload envelopes shown in Figures D.1.1-1 and D.1.1-2 can be conservatively used for preliminary design purposes. These envelopes meet the requirements for dynamic payload envelopes of the Evolved Expendable Launch Vehicle Standard Interface Specification. The static payload envelopes were based on a combination of flight, jettison, and ground handling conditions, and the spacecraft dynamic deflections are only a consideration during flight conditions. Mission-specific modifications to these envelopes, either on a static or dynamic basis, are dependent upon the spacecraft configuration and dynamic behavior, and are considered based on analysis performed for each mission.

D.1.2 Payload Compartment Environmental Control

The Atlas V 4-m PLF is designed to provide a suitable acoustic, thermal, electromagnetic, and contamination controlled environment for the payload. The Atlas V vehicle acoustic panels (Figure D.1.2-1) are provided on the cylindrical section of the fairing and the first two bays of the cones to attenuate the sound pressure levels to acceptable limits (shown in Section 3.2.2).

For thermal control, the external surface of the conical section fairing is insulated with cork to limit temperatures to acceptable values (Sections 3.1.1 and 3.2.5). Noncontaminating, low emittance thermal control coatings are used on payload fairing internal surfaces to reduce incidental heat fluxes to the spacecraft. The acoustic panels located in the cylindrical section of the PLF also serve to reduce heating in the payload compartment. Thermal shields (Fig. D.1.2-2) may be added in the conical section of the fairing, above the acoustic panels, to provide additional thermal control as a mission-specific option. During prelaunch activities, conditioned air is provided through the air-conditioning duct located in the upper cylindrical and lower conical portion of the fairing. This duct directs conditioned air to provide thermal and humidity control upward into the conical section to avoid direct impingement on the spacecraft. Vent holes and housings are mounted on the lower part of the cylindrical section of the PLF to allow air from the air-conditioning system to exit the fairing and to allow depressurization during ascent. A secondary environmental control system may be added to provide additional cooling or to direct cooling air to specific points on the payload. This mission-specific option has been used on several missions and design approaches developed for these past missions can be adapted for future applications.

The metallic construction of the fairing provides some electromagnetic shielding for the spacecraft and serves to attenuate the external Radio Frequency (RF) environment when it is in place during ground operations. Electrically conductive seal materials are used between mating surfaces on the PLF to reduce entry paths for RF signals.

The PLF is fabricated and operated according to requirements of the Atlas Contamination Control Plan described in Section 3.1.3. This plan establishes rigorous procedures to ensure that all hardware that comes into contact with the payload meets cleanliness requirements and may be tailored to meet mission-specific payload requirements.

Figure D.1.2-1: Atlas V 4-m PLF Acoustic Panels

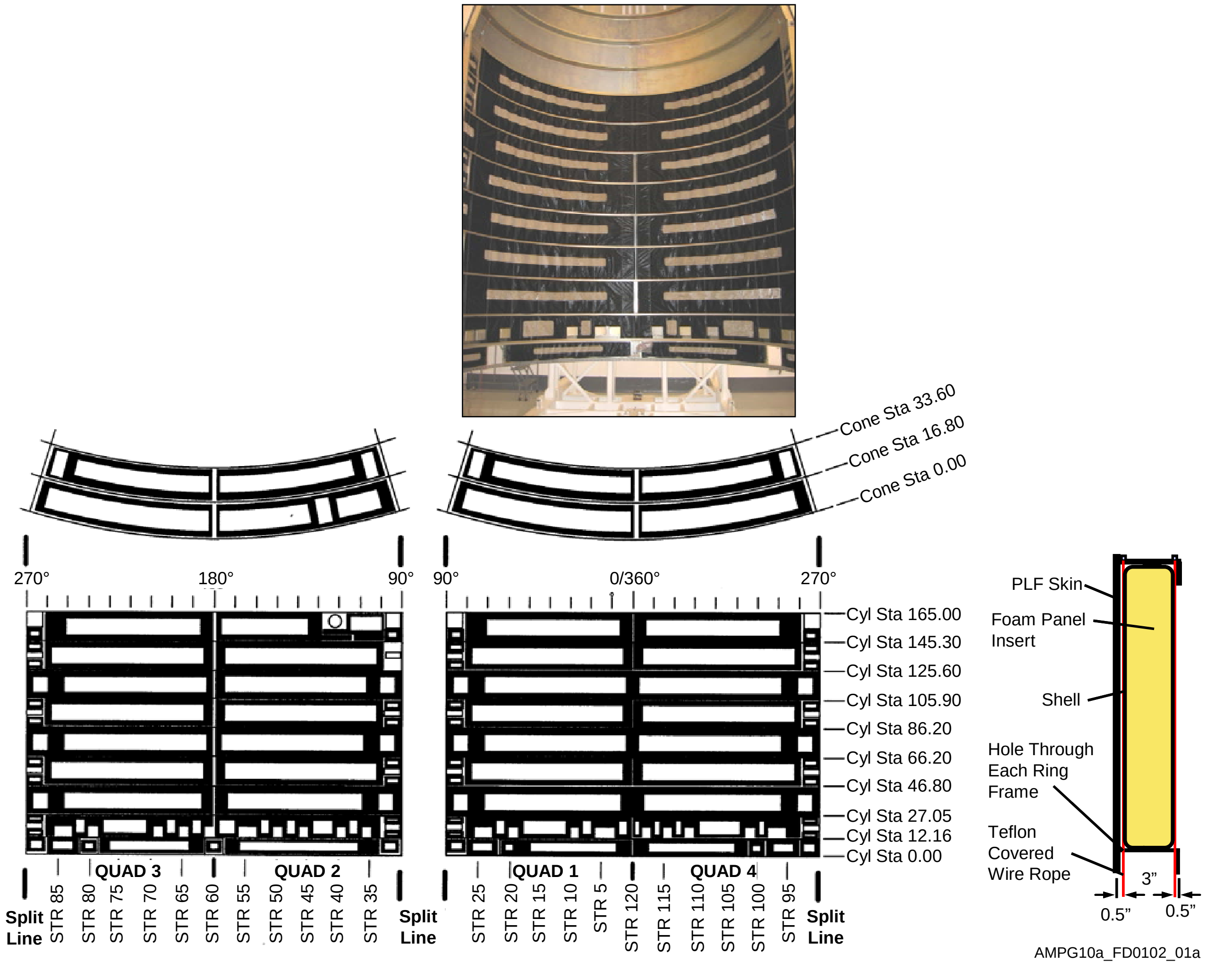
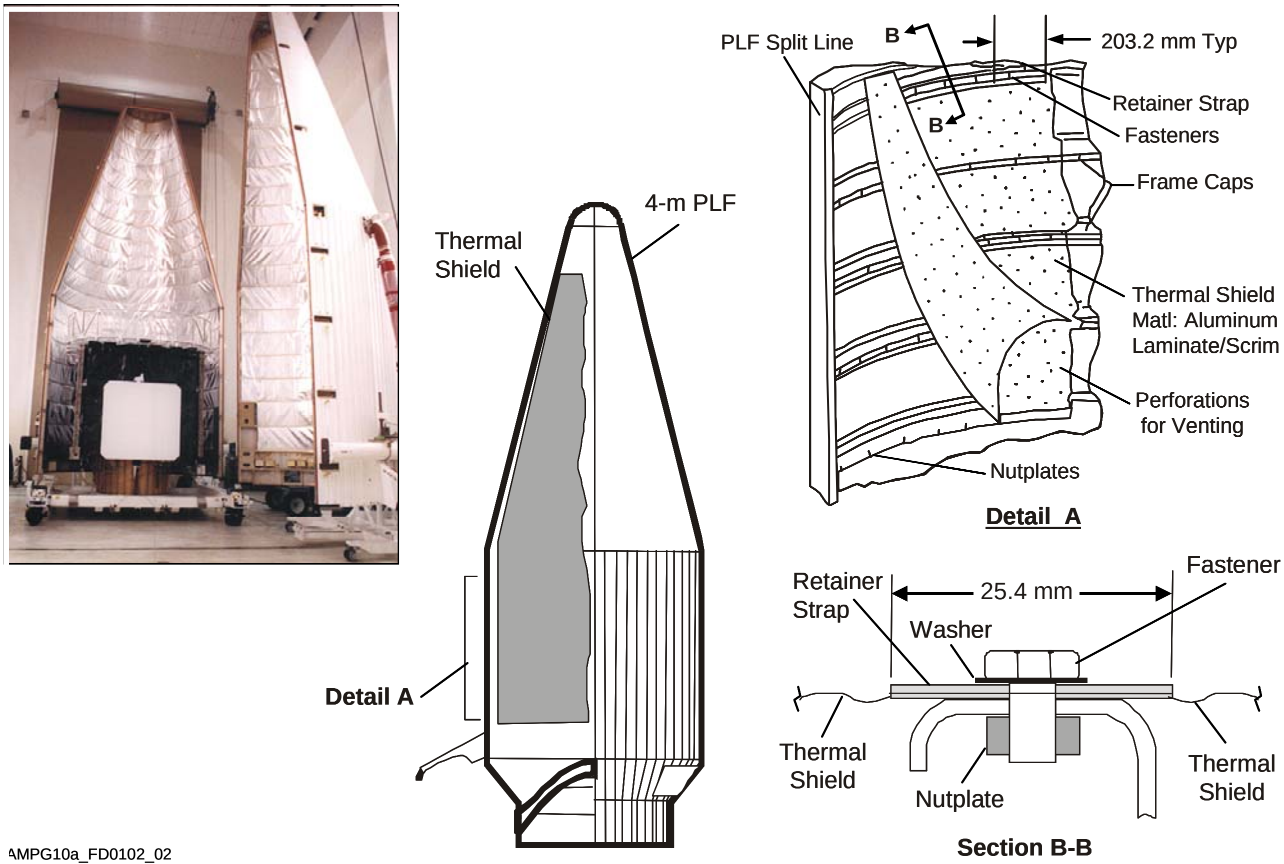


Figure D.1.2-2: Atlas V 4-m PLF Thermal Shields



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D.1.3 Payload Access

The four large doors in the boattail section of the 4-m PLF provide primary access to Centaur forward adapter packages and the encapsulated spacecraft. Work platforms can be inserted into the payload compartment through these doors 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 PLF. The available sizes and allowable locations for these doors are shown in Figure D.1.3-1. Access is permitted from the time of payload encapsulation until close-out operations prior to launch operations.

D.1.4 Payload RF Communications

A reradiating system allows payload RF telemetry transmission and command receipt communications after the payload is encapsulated in the spacecraft until time of launch. The airborne system consists of an antenna, a mounting bracket, and cabling inside the PLF. Reradiating antennas are available in S-, C-, and Ku-bands. The pick-up antenna is mounted on a bracket at a location appropriate for the spacecraft configuration (Fig. D.1.4-1). This antenna acquires the spacecraft RF signal and routes it via RF cabling to PLF T-0 disconnect. A cable runs from the T-0 disconnect to a junction box that routes the signal to a customer-specified location.

D.1.5 Customer Logo

A customer-specified logo may be placed on the cylindrical section of the PLF. Logos up to 3.05 x 3.05 m (10 x 10 ft.) are provided as a standard service. The area of the PLF reserved for customer logos is shown in Figure D.1.5-1. The Atlas V program will work with the customer and provide layouts of the logo on the launch vehicle to assist in determining their proper size and location.

Figure D.1.3-1: Atlas V 4-m PLF Access Doors

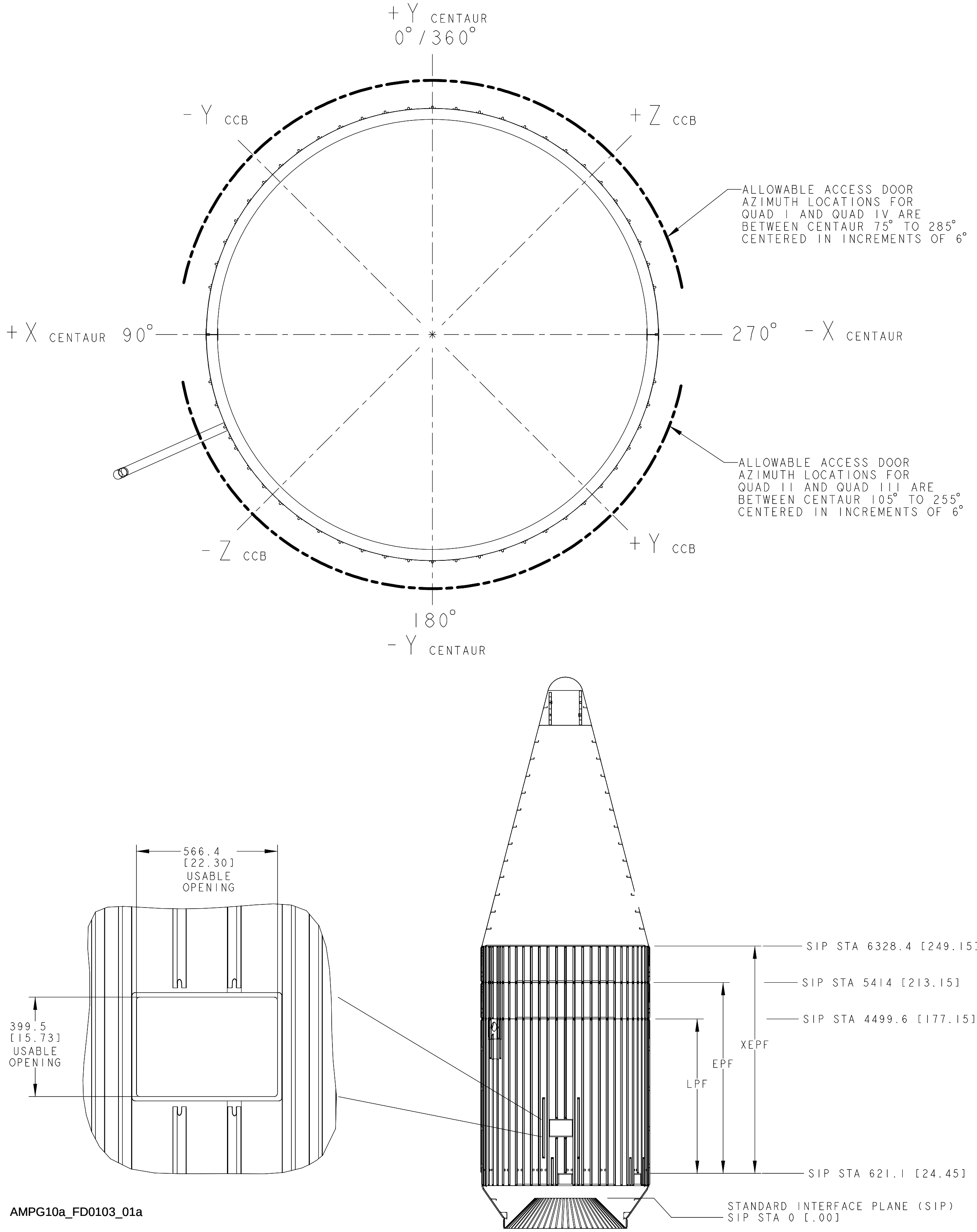

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Figure D.1.4-1: Atlas V 4-m PLF RF Reradiate Antenna Installation

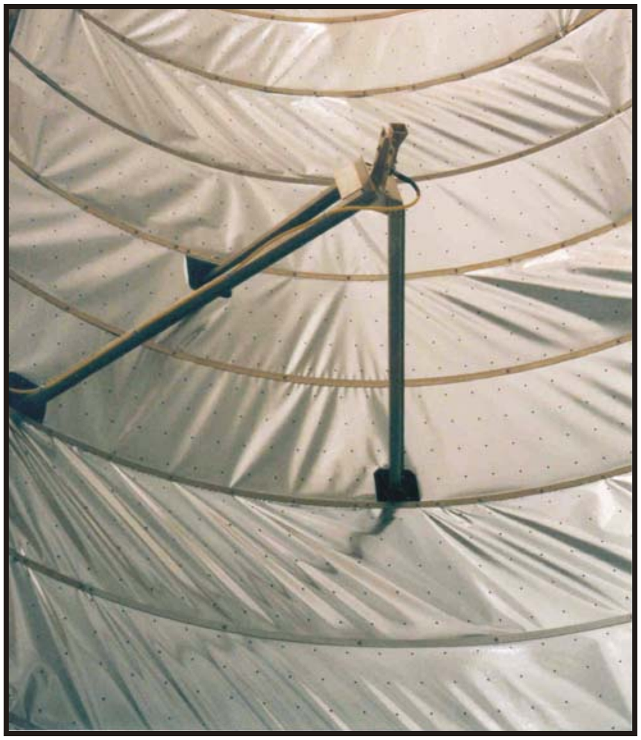
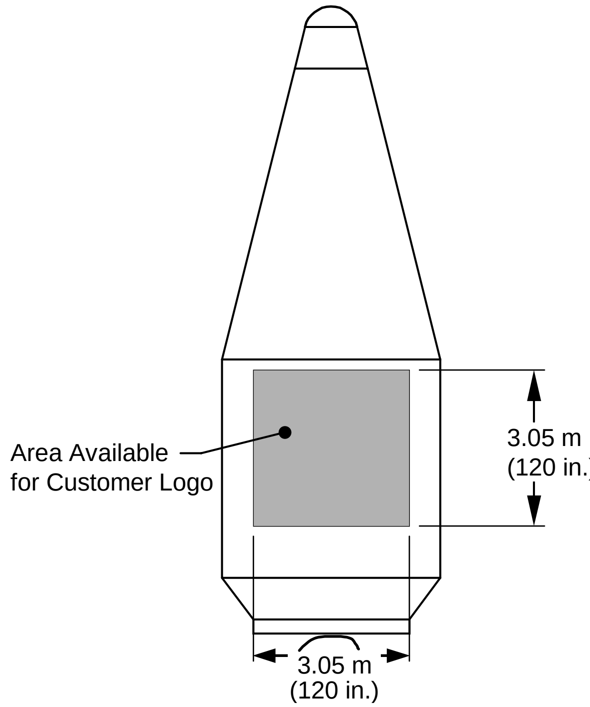


Figure D.1.5-1: Atlas V 4-m PLF Customer Logo Provisions



D.2 ATLAS V 5-METER PAYLOAD FAIRING

The Atlas V 5-m diameter payload fairing was developed along with the increased launch vehicle performance to accommodate growing spacecraft volume requirements. The Atlas V 5-m short, medium, and long length PLF have a 5.4-m (213.6-in.) outer skin line diameter cylindrical section. The 5-m PLF is a bisector payload fairing with a composite structure made from sandwich panels with composite facesheets and a vented aluminum honeycomb core. The PLF and its associated separation system are derived from an existing flight-proven system. There are two major components of the bisector payload fairing: the Base Module (BM) and the Common Payload Module (CPM). The BM is the lower section of the PLF that encapsulates the Centaur upper stage. The CPM is the upper section of the PLF that encapsulates the spacecraft. The CPM consists of a cylindrical section that transitions into a constant radius ogive nose section topped by a spherical nose cap as shown in Figure D.2-1. The ogive shape minimizes aerodynamic drag and buffet during booster ascent. The 5-m medium PLF adds a 2,743-mm (108-in.) Lower Payload Module (LPM) to the base of the CPM to increase the available payload volume. The fairing interfaces with the launch vehicle at the fixed conical boattail that is attached to the launch vehicle first stage. Clearance losses for payloads are minimized by the Centaur Forward Load Reactor (CFLR) system that stabilizes the top of the Centaur, thereby reducing the relative motion between the PLF and payload. The PLF sections provide mounting provisions for various secondary systems. Electrical packages required for the PLF separation system are mounted on the internal surface of the fairing. Payload fairing compartment cooling system provisions are in the ogive section of the fairing.

D.2.1 Atlas V 5-m Static Payload Envelope

The useable volume for a spacecraft inside the PLF is defined by the static payload envelope. The Atlas V 5-m PLF provides a 4,572-mm (180-in.) diameter envelope in the cylindrical section with additional volume available in the ogive section of the PLF. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. These envelopes were established to ensure that a minimum 25-mm (1-in.) clearance between the spacecraft and the PLF is maintained and to include allowances for PLF static tolerances and misalignments, spacecraft-to-PLF dynamic deflections, and PLF out-of-round conditions. These envelopes were developed and are applicable for spacecraft that meet the stiffness and load requirements discussed in Section 3.2.1. Clearance layouts and analyses are performed for each spacecraft configuration and, if necessary, critical clearance locations are measured after the spacecraft is encapsulated inside the fairing to ensure positive clearance during flight. Detailed views of the static payload envelope for the Atlas V 5-m short, medium and long PLFs are shown in Figure D.2.1-1.

For customers requesting a dynamic payload envelope, the static payload envelopes shown in Figure D.2.1-1 can be conservatively used for preliminary design purposes. These envelopes meet the requirements for dynamic payload envelopes of the Evolved Expendable Launch Vehicle Standard Interface Specification. The static payload envelopes were based on a combination of flight, jettison, and ground handling conditions, and the spacecraft dynamic deflections are only a consideration during flight conditions. Mission-specific modifications to these envelopes, either on a static or dynamic basis, are dependent upon the spacecraft configuration and dynamic behavior and are considered based on analysis performed for each mission.

Figure D.2-1: Atlas V 5-Meter Payload Fairing Configuration

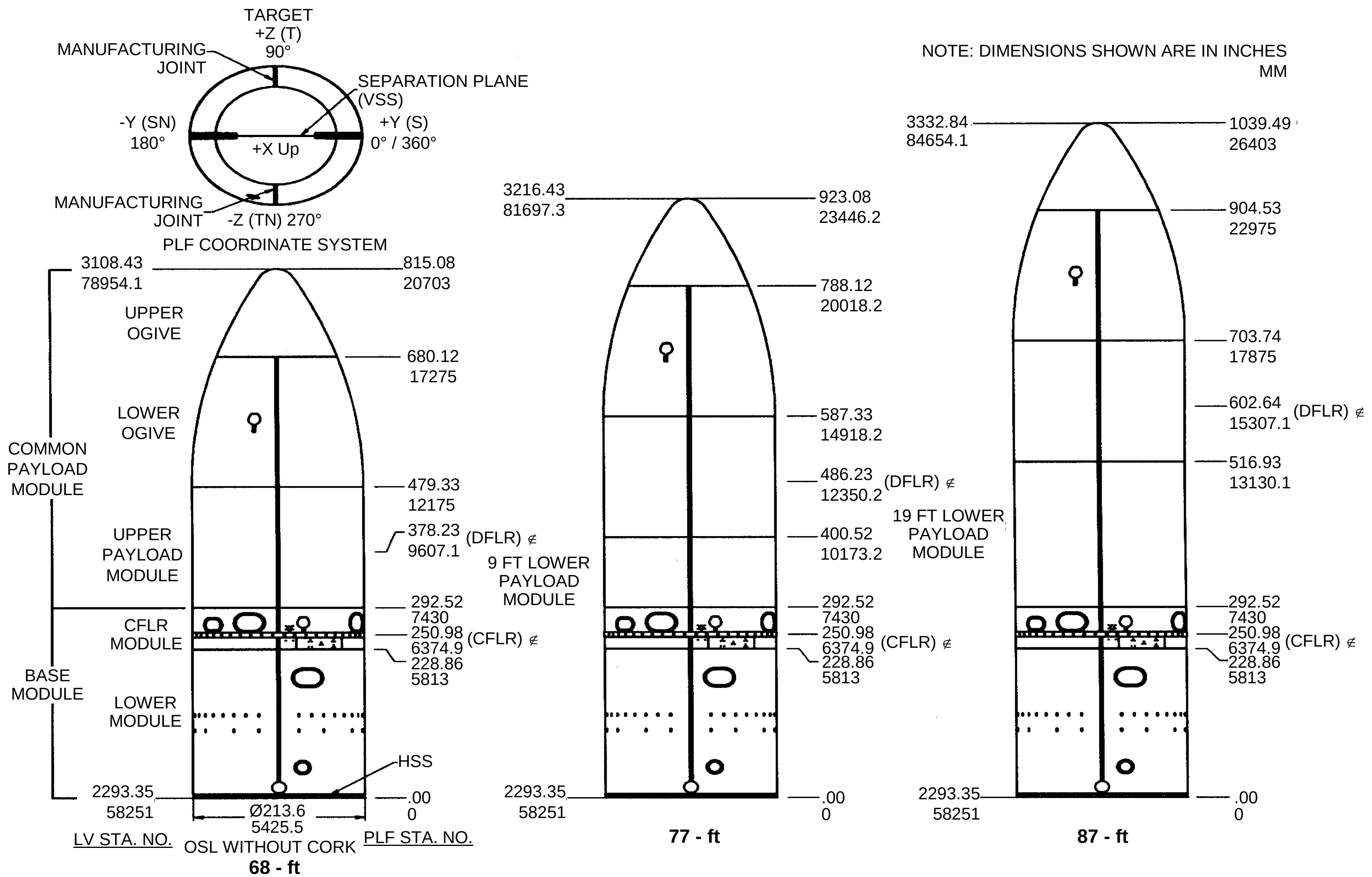
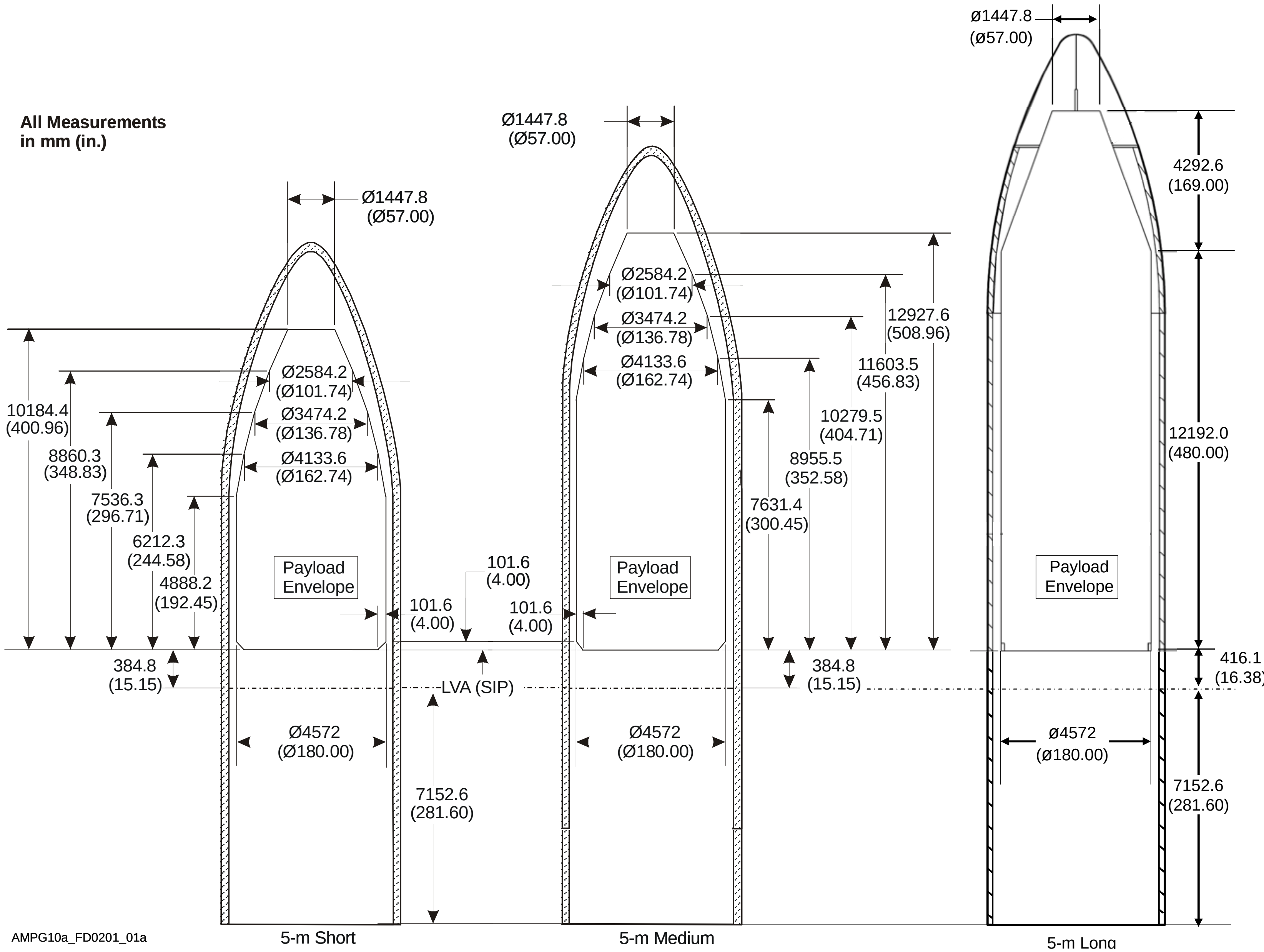


Figure D.2.1-1: Atlas V 5-m PLF Static Payload Envelope



D.2.2 Payload Compartment Environmental Control

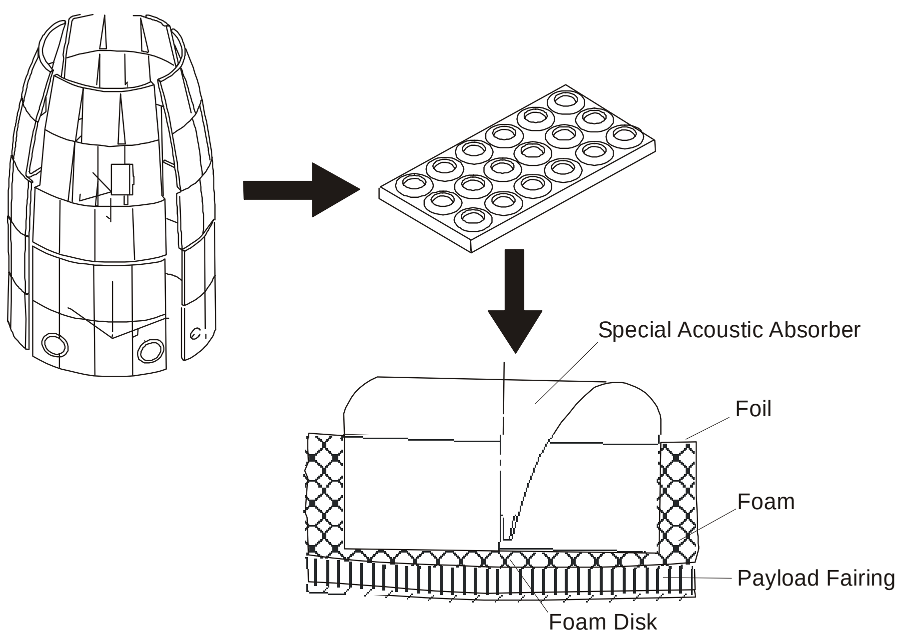
The Atlas V 5-m PLF is designed to provide a suitable acoustic, thermal, electromagnetic, and contamination-controlled environment for the payload. Fairing Acoustic Protection (FAP), shown in Figure D.2.2-1, is provided as a standard service to attenuate the sound pressure levels to acceptable limits (Section 3.2.2).

For thermal control, the external surface of the fairing is insulated with cork and painted white to limit temperatures to acceptable values as identified in Sections 3.1.1 and 3.2.5. The fairing acoustic protection panels also serve to reduce heating in the payload compartment. During prelaunch activities, conditioned air is provided through the air-conditioning inlet located in the ogive section of the PLF. This inlet directs conditioned air to provide thermal and humidity control for the payload compartment and prevents direct impingement of this flow on the spacecraft. Vent ports and vent port assemblies are mounted in the mid-section of the base module for air from the air-conditioning system to exit the PLF and to allow depressurization during ascent. A secondary environmental control system may be added as a mission-specific option to provide additional cooling or to direct cooling air to specific points on the payload.

Electrically conductive paint is used on the outside of the PLF to prevent electrostatic build-up on the PLF.

The PLF is fabricated and operated according to the requirements of the Atlas Contamination Control plan described in Section 3.1.3. This plan establishes rigorous procedures to ensure that all hardware that comes into contact with the payload meets cleanliness requirements and may be tailored to meet mission-specific payload requirements.

Figure D.2.2-1: Atlas V 5-m Fairing Acoustic Protection



D.2.3 5-m PLF Payload Access

The 5-m PLF has four large doors in the PLF base module to provide primary access to the Centaur forward adapter packages and the encapsulated spacecraft. The doors provide an opening of approximately 600 x 900 mm (24 x 36 in.). Work platforms can be inserted through these doors onto the CFLR deck to allow access to spacecraft hardware near the aft end of the payload module. If additional access to the spacecraft is required, additional doors can be provided on a mission-specific basis on the cylindrical and ogive section of the PLF. The available sizes and allowable locations for these doors are shown in Figures D.2.3-1 to D.2.3-3. Access is permitted from the time of payload encapsulation until closeout operations prior to launch operations.

D.2.4 Payload RF Communications

A PLF reradiating system provides the capability for payload RF telemetry transmission and command receipt communications after payload encapsulation through the time of launch. The airborne system consists of an antenna, a mounting bracket, and cabling inside the PLF. Reradiating antennas are available in the S-band, C-band, and Ku bands. The pickup antenna is mounted on a bracket at a location appropriate for the spacecraft configuration. This antenna acquires the spacecraft RF signal and routes it via RF cabling to the PLF T-0 disconnect. A cable runs from the T-0 disconnect to a junction box that routes the signal to a customer-specified location.

D.2.5 Customer Logo

A customer-specified logo may be placed on the cylindrical section of the PLF. Logos up to 3.05 x 3.05 m (10 x 10 ft.) are provided as a standard service. The area of the PLF reserved for customer logos is shown in Figure D.2.5-1. The Atlas V program will work with the customer and provide layouts of the logo on the launch vehicle to assist in determining their proper size and location.

Figure D.2.3-1: Atlas V 5-m Short PLF Mission Specific Access Doors (1 of 2)

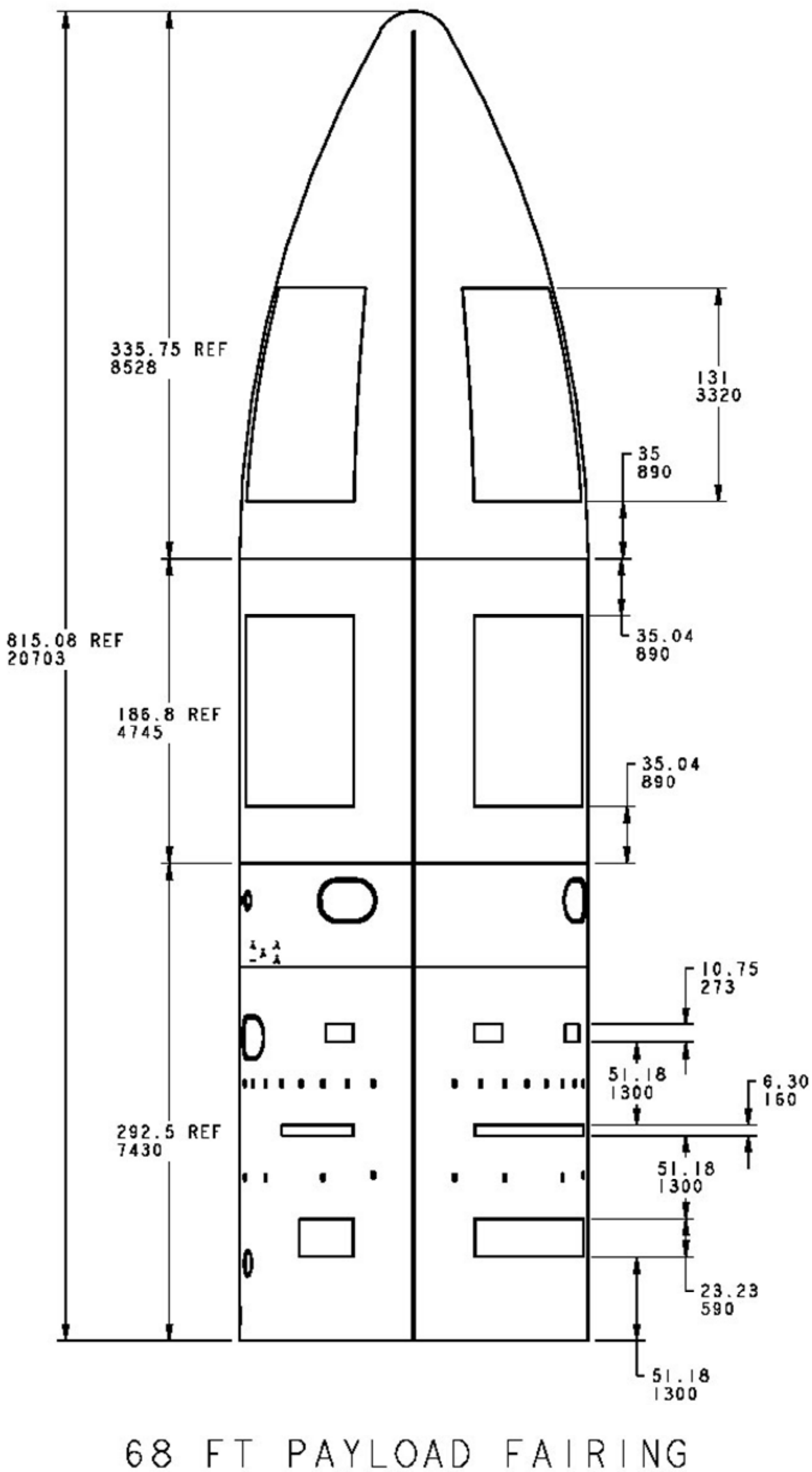


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Figure D.2.3-1: Atlas V 5-m Short PLF Mission Specific Access Doors (2 of 2)

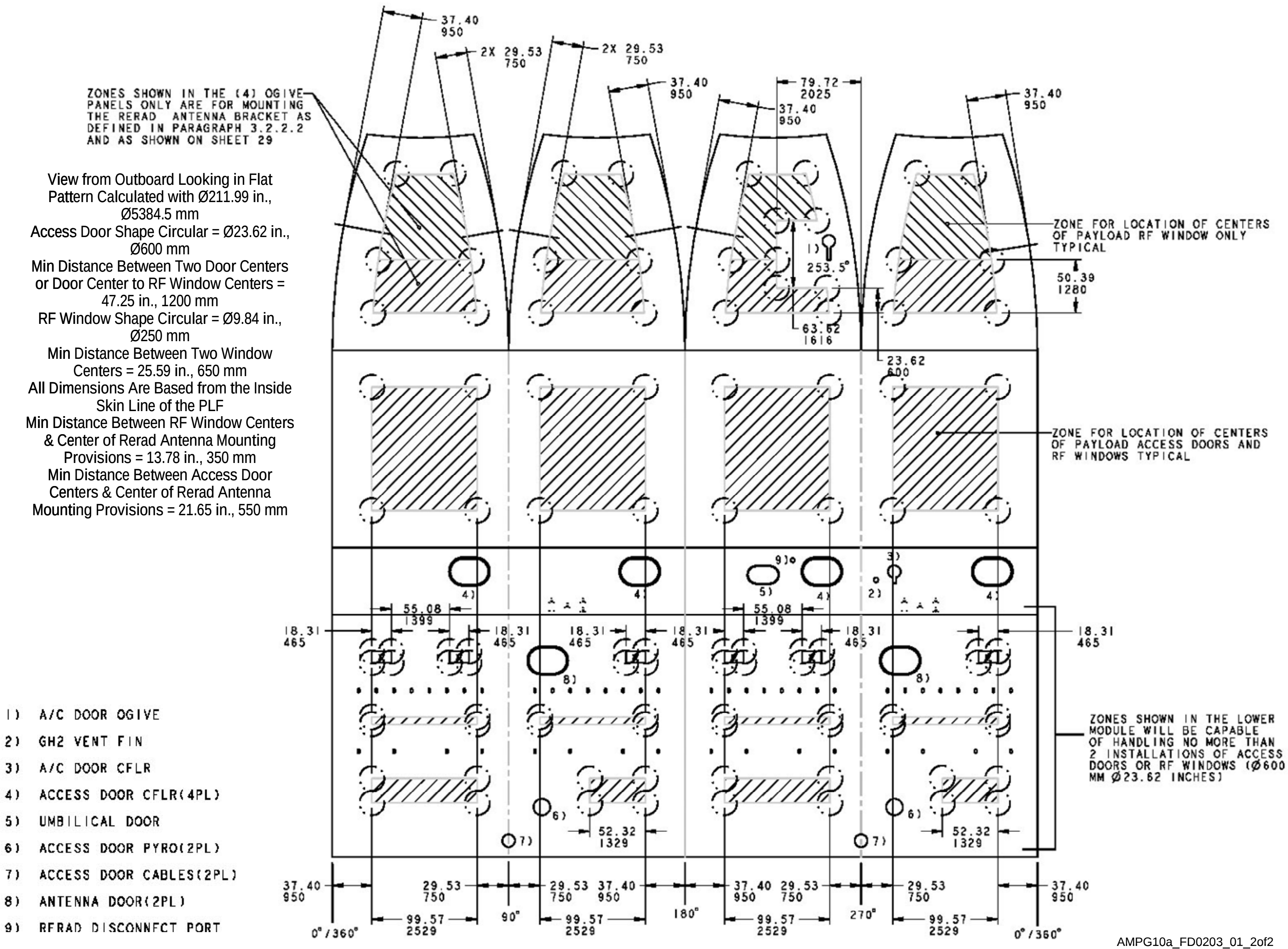

Figure D.2.3-2: Atlas V 5-m Medium PLF Mission Specific Access Doors (1 of 2)

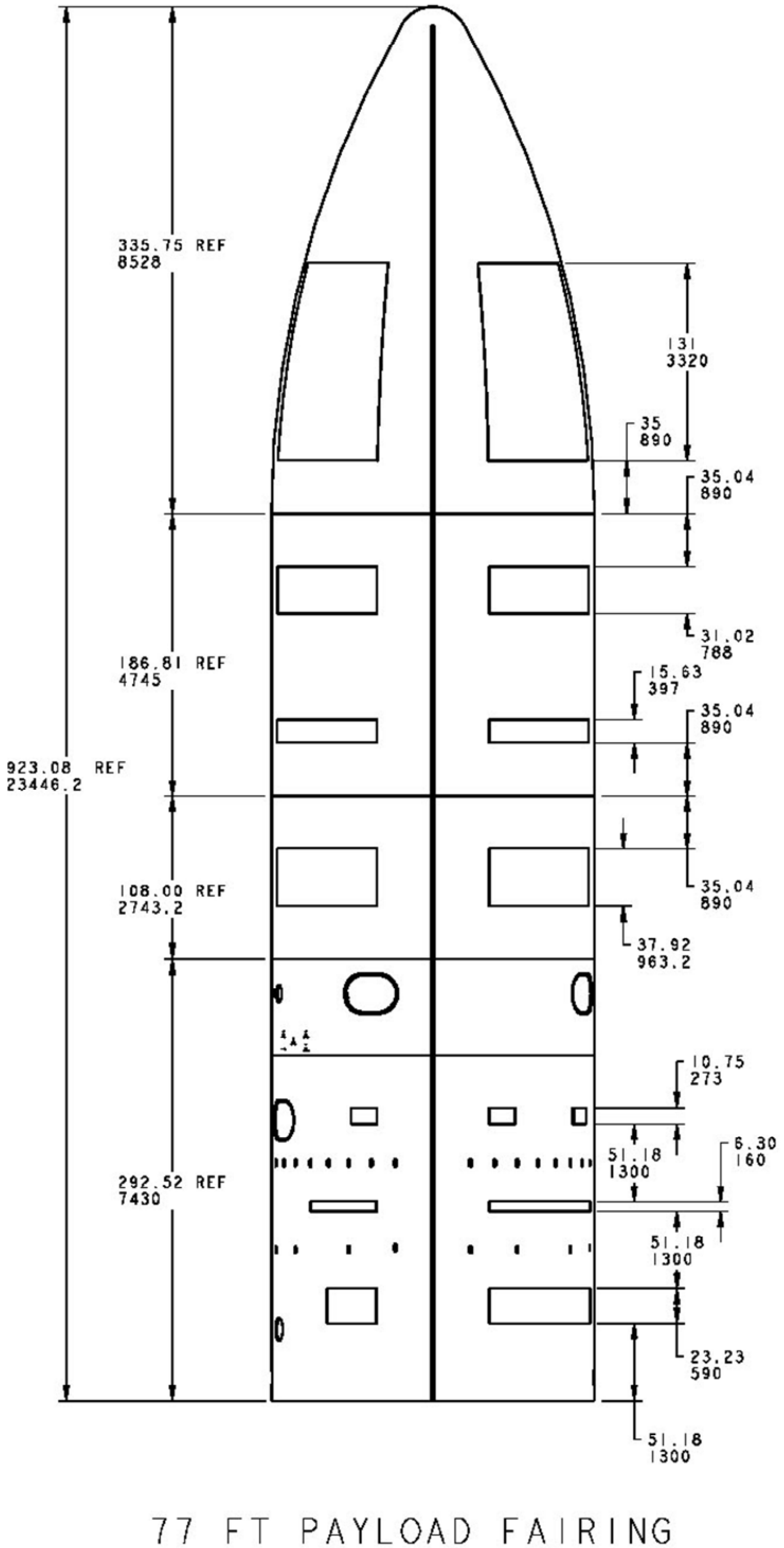

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Figure D.2.3-2: Atlas V 5-m Medium PLF Mission Specific Access Doors (2 of 2)

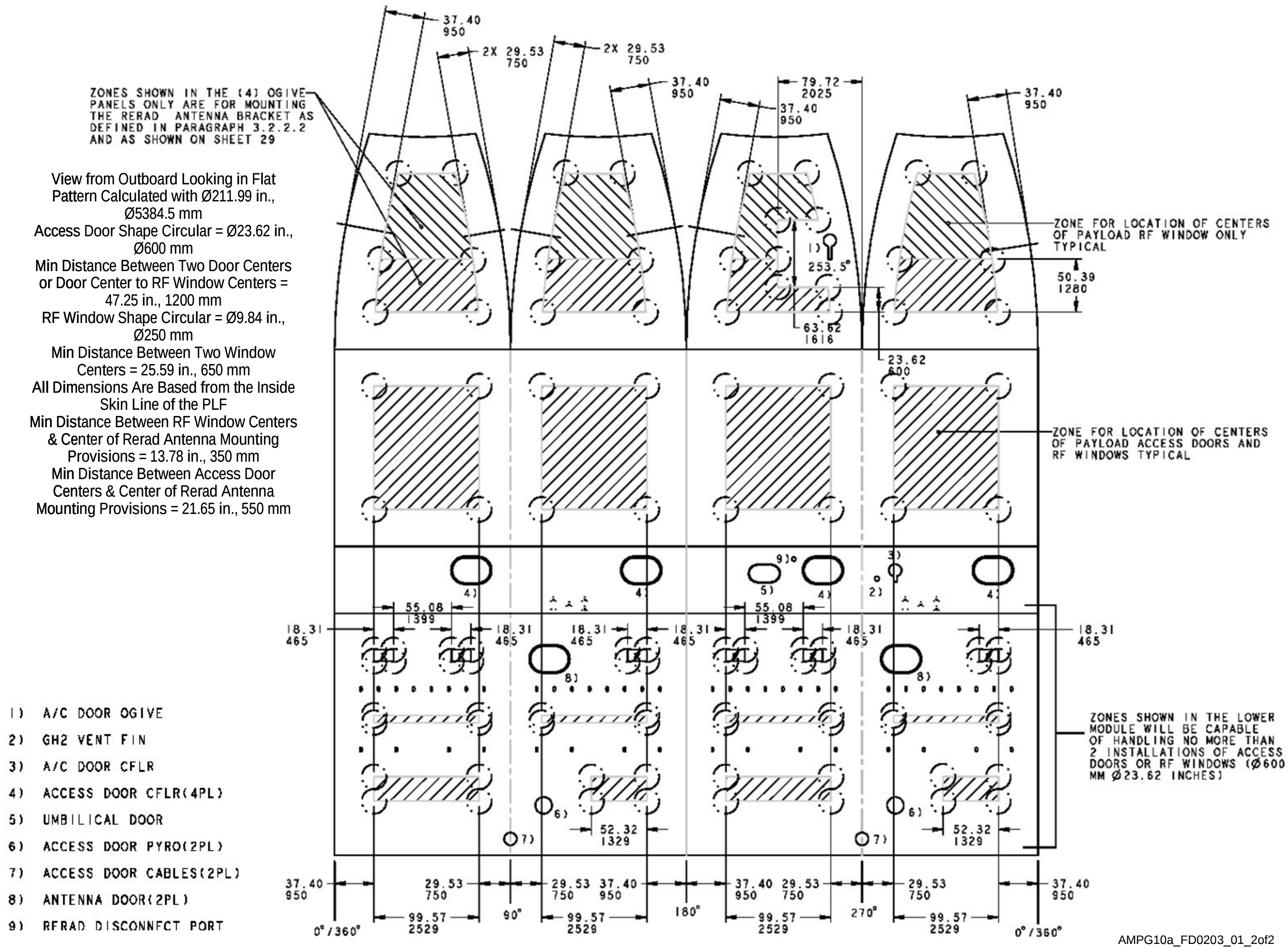

Figure D.2.3-3: Atlas V 5-m Long PLF Mission Specific Access Doors (1 of 2)

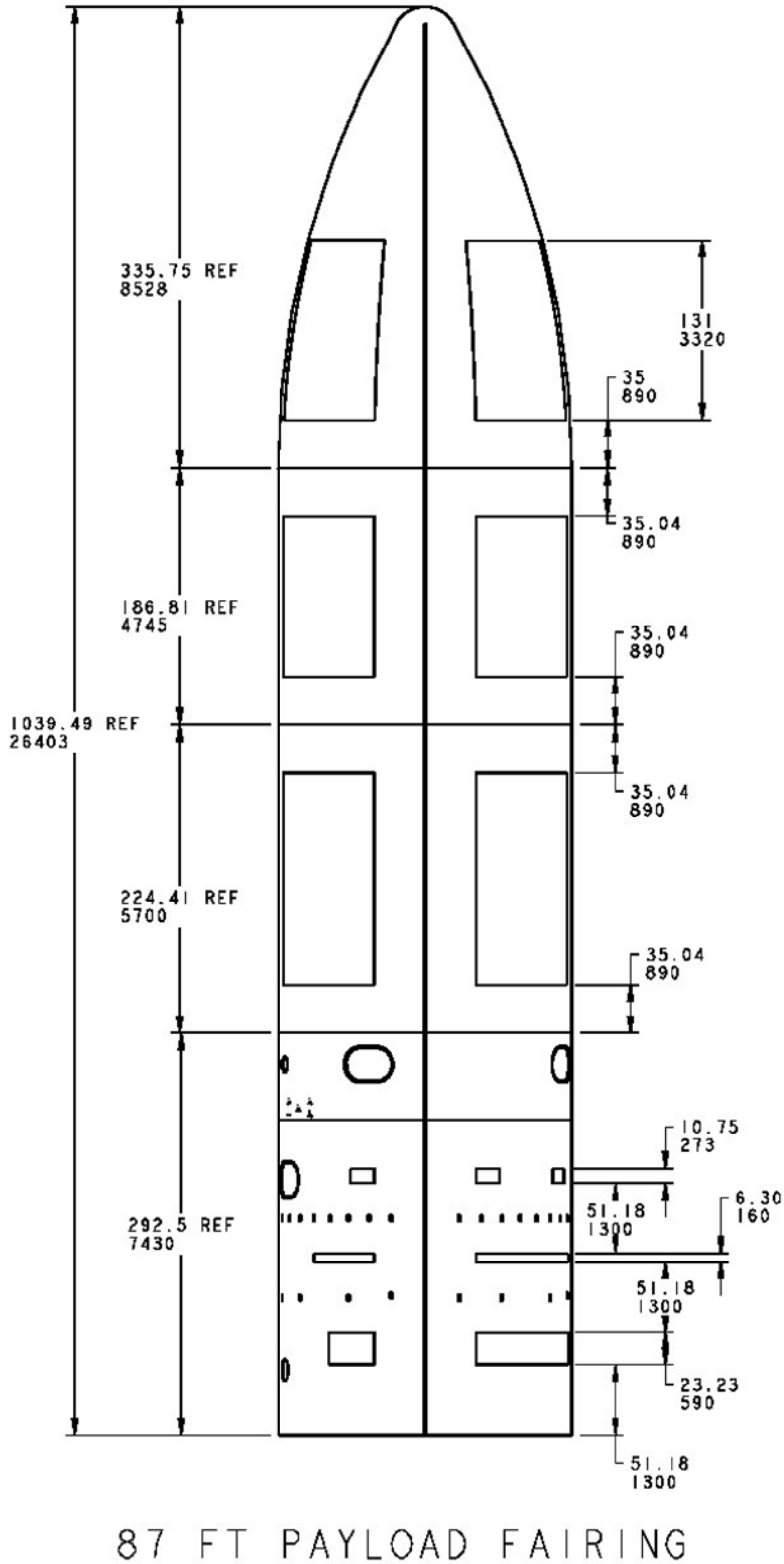

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Figure D.2.3-3: Atlas V 5-m Long PLF Mission Specific Access Doors (2 of 2)

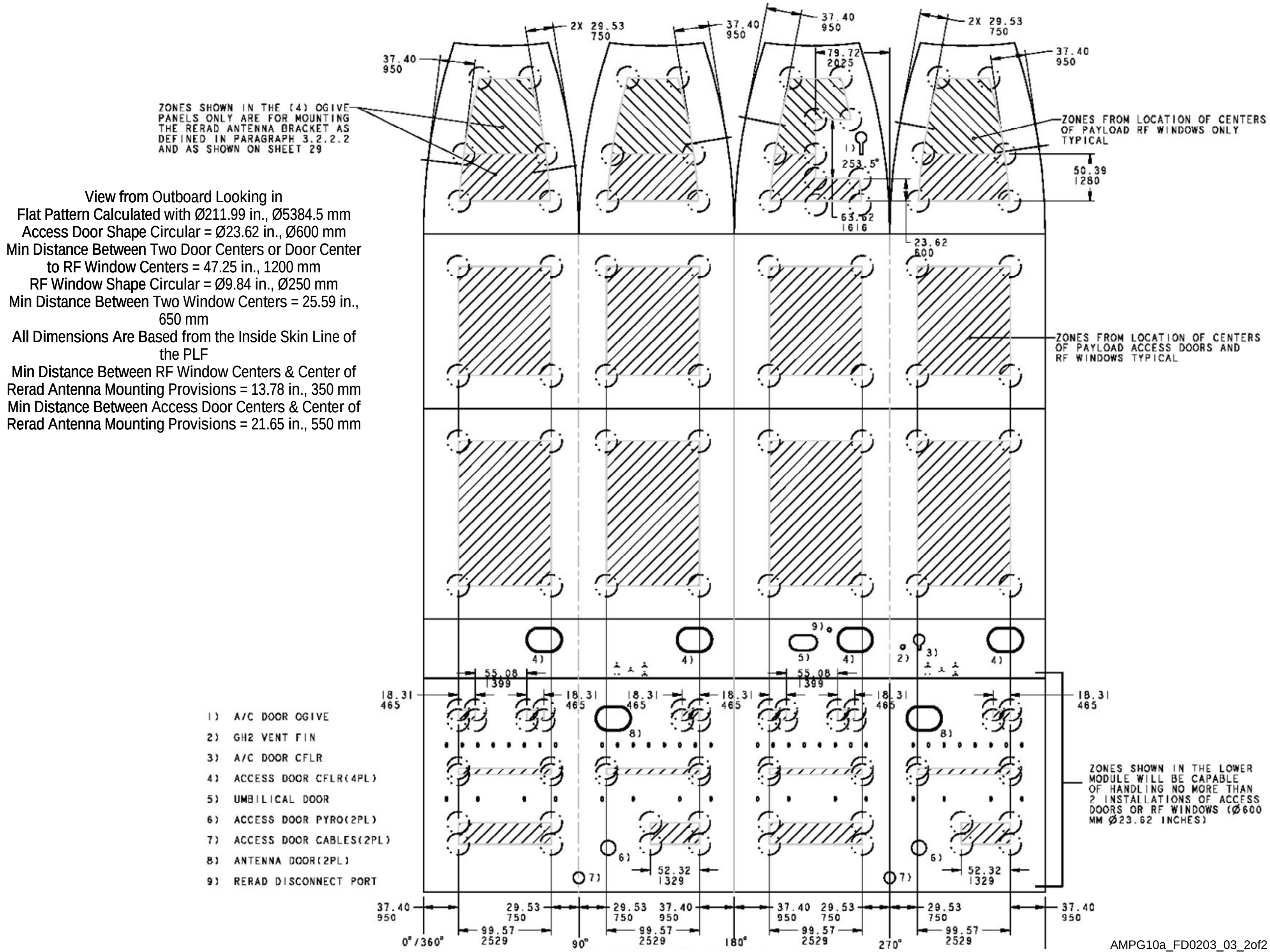
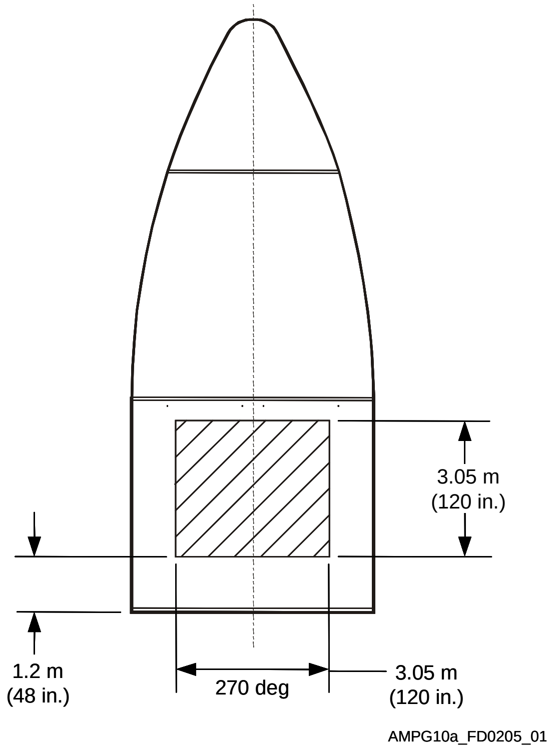


Figure D.2.5-1: Atlas V 5-m Customer Logo Provisions



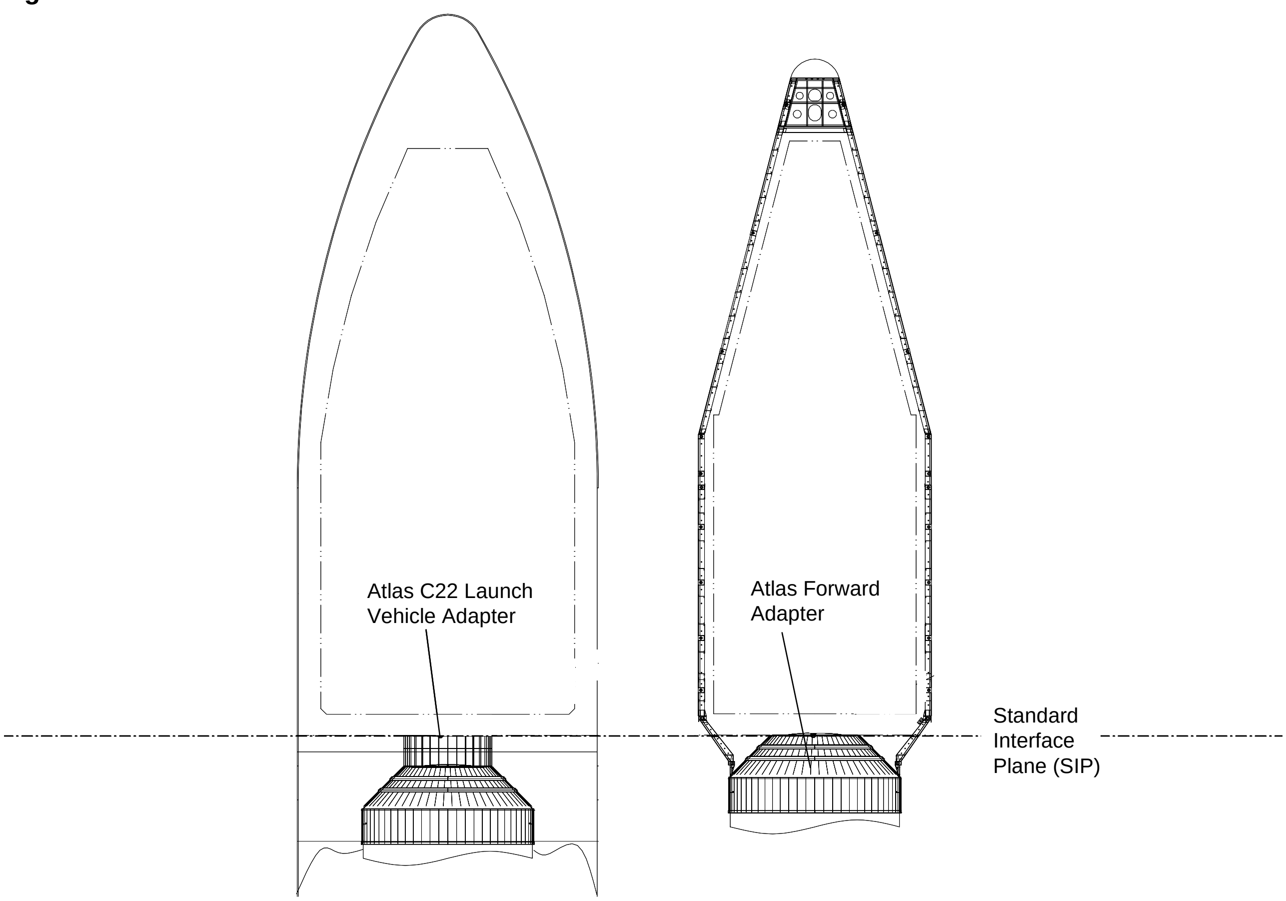
APPENDIX E— ATLAS V PAYLOAD INTERFACES

E.1 ATLAS V STANDARD INTERFACE PLANE

The Atlas V Standard Interface Plane (SIP) provides a standardized bolted interface for Atlas V and customer provided payload adapters. The SIP consists of a 1,570-mm (62.010-in.) diameter machined ring that contains 121 bolt holes. For vehicles using a 4-m Payload Fairing (PLF), this plane is at the top of the Atlas V Centaur forward adapter. For vehicles using a 5-m PLF, this plane is at the top of an Atlas V-provided C22 launch vehicle adapter that is mounted on top of the Centaur forward adapter as shown in Figure E.1-1. The C22 launch vehicle adapter is standard with the 5-m PLF to allow launch vehicle Ground Support Equipment (GSE) interfaces and is designed to provide an interface that is identical to that provided by the Centaur forward adapter. In this configuration, cost and performance impacts of the C22 launch vehicle adapter are considered to be a part of the basic launch vehicle service and are not counted against payload systems weight.

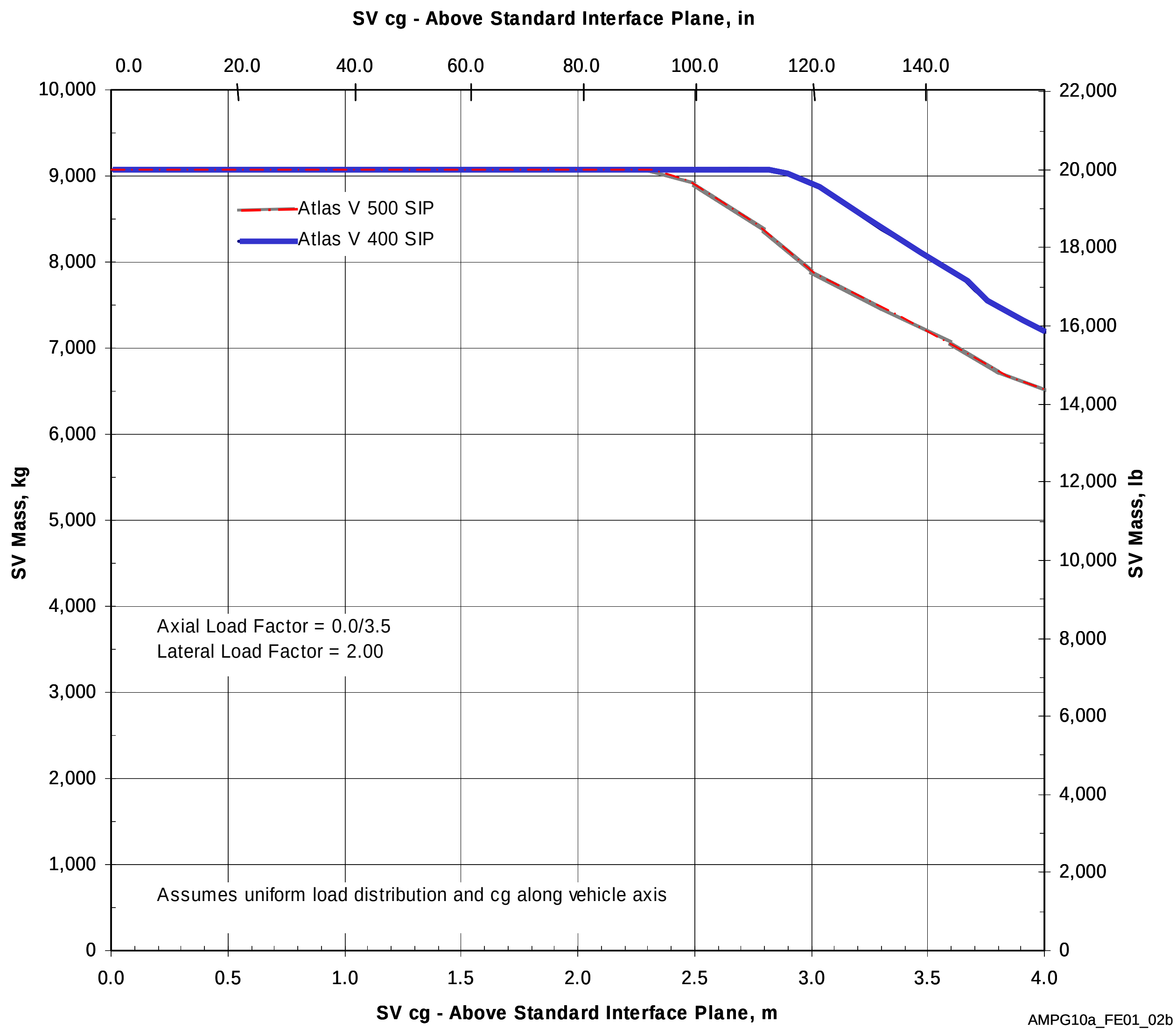
For customers that provide their own adapter, the SIP is the interface point between the spacecraft provided hardware and the launch vehicle. If a customer provided spacecraft adapter is used with a 4-m PLF, it must provide interfaces for ground handling, encapsulation, and transportation equipment. In particular, there needs to be provisions for torus arm fittings and an encapsulation diaphragm unless a launch vehicle supplied intermediate adapter is used. Information on these interface requirements is in Atlas Specification S/M-00-025, “Torus Arm And Ground Transport Vehicle Interface Control Document for Payloads Using the 4-meter Payload Fairing” which is available on request to Atlas V customers.

Figure E.1-1: Atlas V Standard Interface Plane



Standard Interface Plane Structural Capabilities — Allowable spacecraft weights and longitudinal Centers of Gravity (cg) for the SIP are shown in Figure E.1-2. The spacecraft mass and cg capabilities were determined using generic spacecraft interface ring geometry and quasi-static load factors (see Section 3.2.1). Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas V program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

Figure E.1-2: Atlas V Standard Interface Plane Structural Capability



Standard Interface Plane Definition — Configuration and dimensional requirements for the Atlas V SIP are shown in Figure E.1-3. The bolt hole pattern for this interface is controlled by Atlas V provided tooling. This tooling is made available to customers for fabrication of their matching hardware as a part of mission integration activities.

Static Payload Envelope — Usable volume for the spacecraft relative to the SIP 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. This envelope is designed to allow access to mating components for integration and installation operations and to ensure positive clearance between launch vehicle and spacecraft provided hardware. Clearance layouts are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-launch vehicle mate operations to ensure positive clearances. Detailed views of the static payload envelope for the SIP are shown in Figure E.1-4.

Figure E.1-3: Atlas V Standard Interface Plane Dimensional Requirements

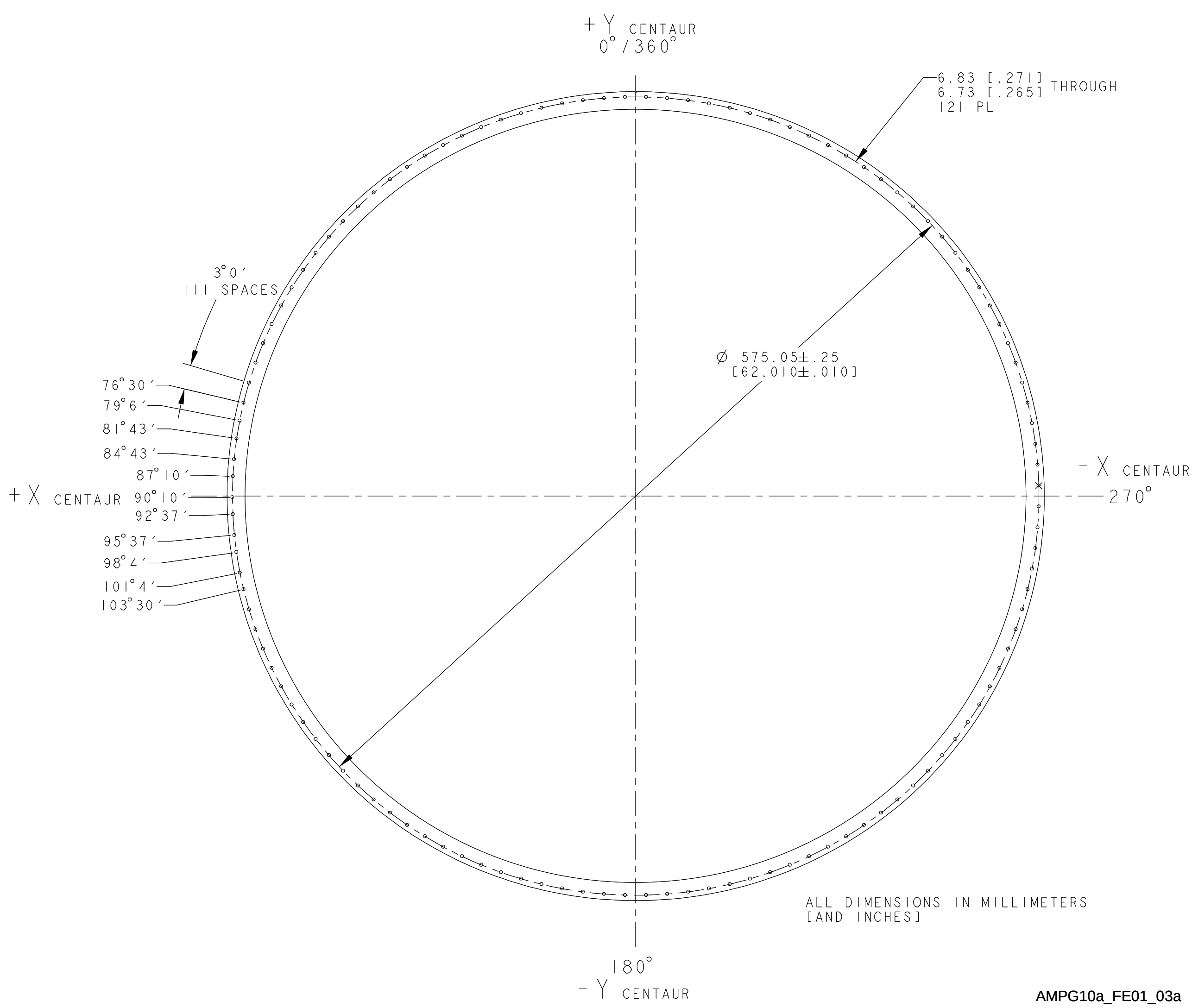
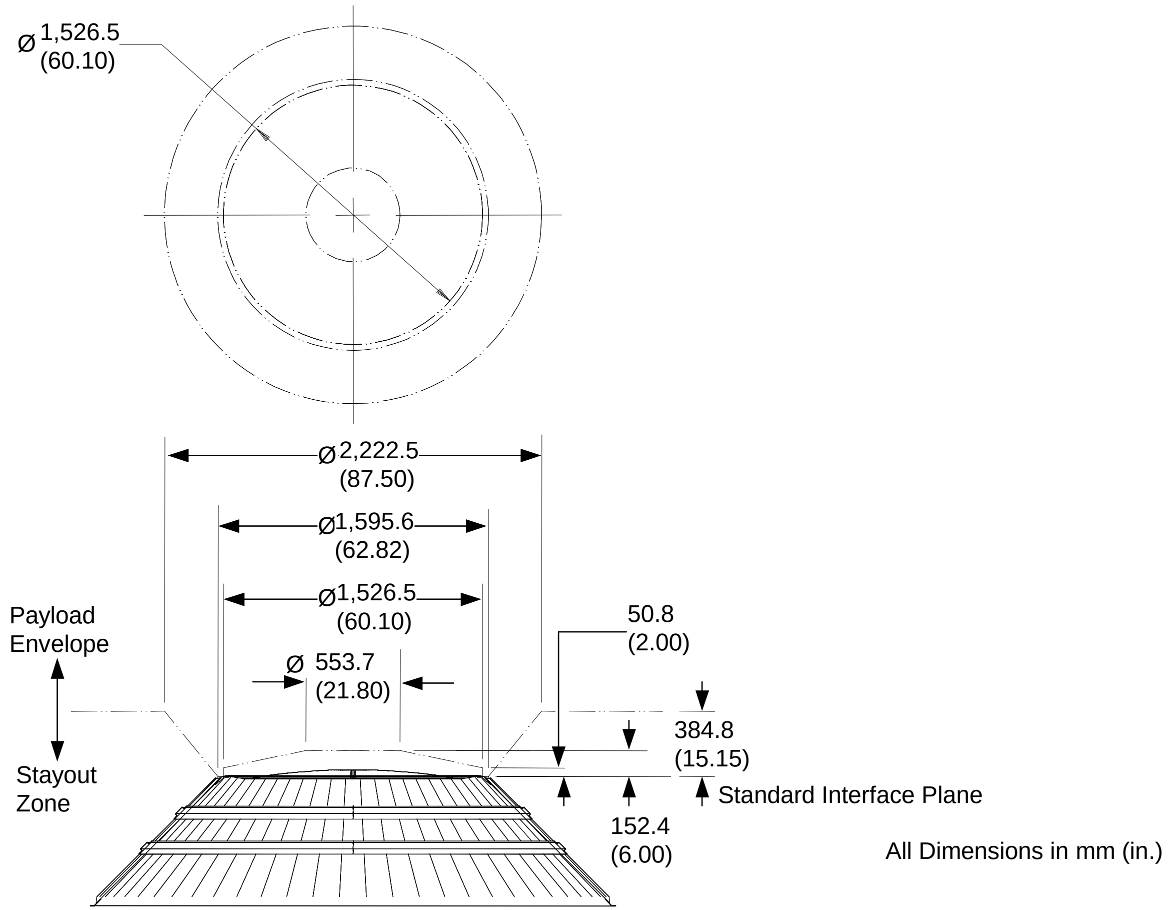


Figure E.1-4: Atlas V Standard Interface Plane Static Payload Envelope



E.2 ATLAS V LAUNCH VEHICLE ADAPTERS

Atlas V launch vehicle adapters were developed to provide a common interface for launch vehicle required ground support equipment that interfaces with payload adapter systems. Atlas V launch vehicle adapter characteristics are summarized in Table E.2-1. The launch vehicle adapter is a machined aluminum component in the form of a monocoque cylinder and is available in heights from 330.2 mm to 736.6 mm (13.00 in. to 29.00 in.). Standard configurations available are the C13, C15, C22 (shown in Figure E.2-1) and C29 launch vehicle adapters.

Table E.2-1: Atlas V Launch Vehicle Adapter Characteristics

Atlas V Launch Vehicle Adapters	
Construction	Integrally Machined Aluminum Construction
Mass Properties	PSW, See Section 2.5.1
Standard Height	
C13	330.2-mm (13.00-in.)
C15	384.8-mm (15.15-in.)
C22	558.8-mm (22-in.)
C29	736.3-mm (29-in)
Payload Capability (Figure E.2-2)	6,300 kg at 2.5 m (13,890 lb at 98.5 in.)

On the Atlas V 500 series vehicle, a C22 launch vehicle adapter is mounted to the top of the Atlas V Centaur forward adapter and provides an interface surface and hole-pattern at its forward end that is compatible with SIP requirements. The C22 adapter is standard with the 5-m PLF to allow clearance for launch vehicle ground support equipment. In this configuration, cost and performance impacts of the C22 launch vehicle adapter are considered to be a part of the basic launch vehicle service and are not counted against payload systems weight.

The Atlas V program has adapted a modified version of the launch vehicle adapter as a component of Atlas V standard payload adapters. This allowed creation of a modular series of payload adapters that have common interfaces to launch vehicle flight hardware and ground support equipment on a component that is separate from the mission-specific requirements of the spacecraft.

For customers that provide their own payload adapter and payload separation system, launch vehicle adapters are available as a mission-specific option. This allows the customer to raise the position of the SIP relative to the launch vehicle for additional clearance or to take advantage of standard GSE interfaces that are built into the launch vehicle adapter.

Launch Vehicle Adapter Structural Capabilities — Allowable spacecraft weights and longitudinal centers of gravity for launch vehicle adapters are shown in Figure E.2-2. These spacecraft mass and center of gravity capabilities were determined using generic spacecraft interface ring geometry and quasi-static load factors (see Section 3.2.1). Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas V program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables. Additional launch vehicle adapter capability is available on a mission unique basis.

Launch Vehicle Adapter Interfaces — Configuration and dimensional requirements for Atlas V launch vehicle adapters interface are shown in Figure E.2-3. The hole pattern for this interface is controlled by Atlas V provided tooling. This tooling is made available to customers for fabrication of their matching hardware as a part of mission integration activities. Alternative hole patterns for this interface can be incorporated on a mission-specific basis.

Figure E.2-1: Atlas Launch Vehicle Adapters

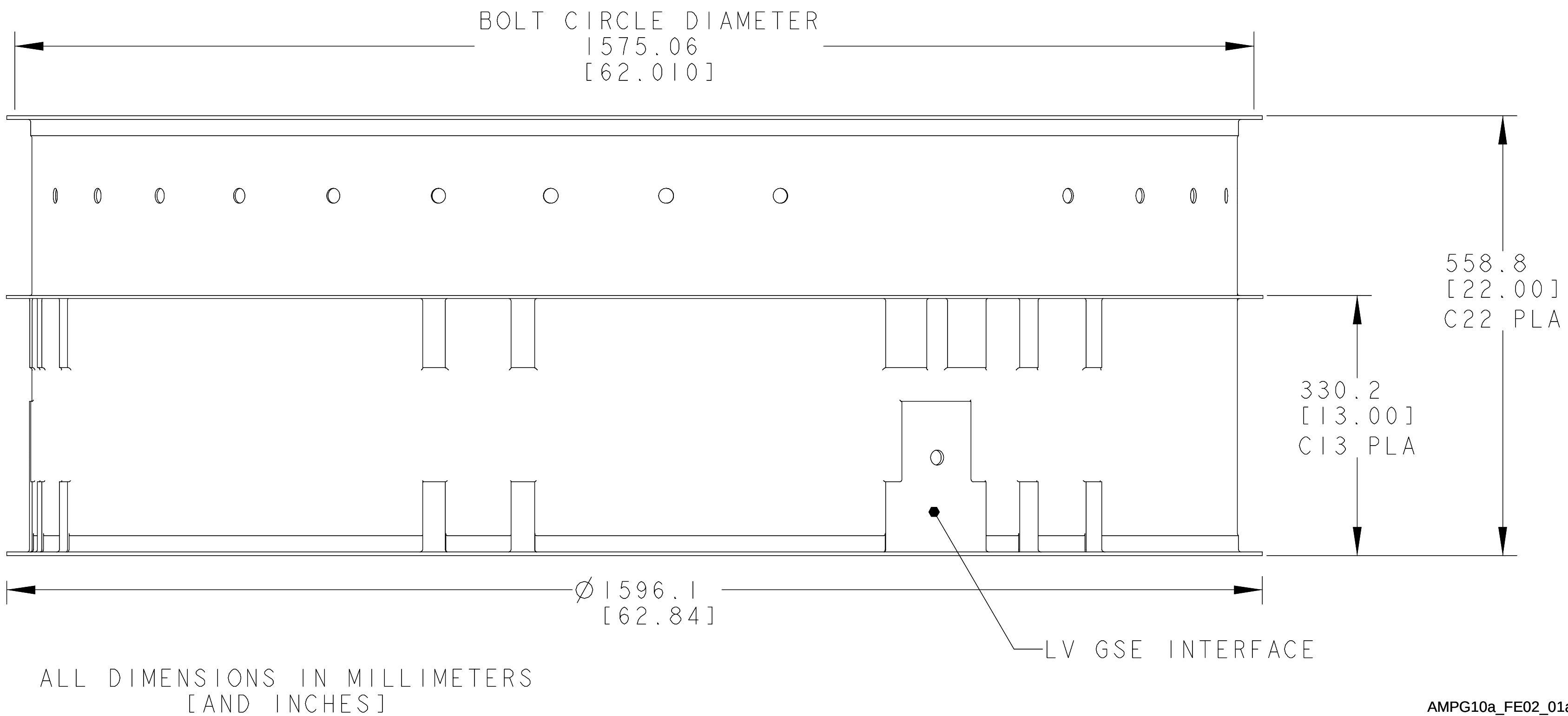


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Figure E.2-2: Atlas Launch Vehicle Adapter Structural Capability

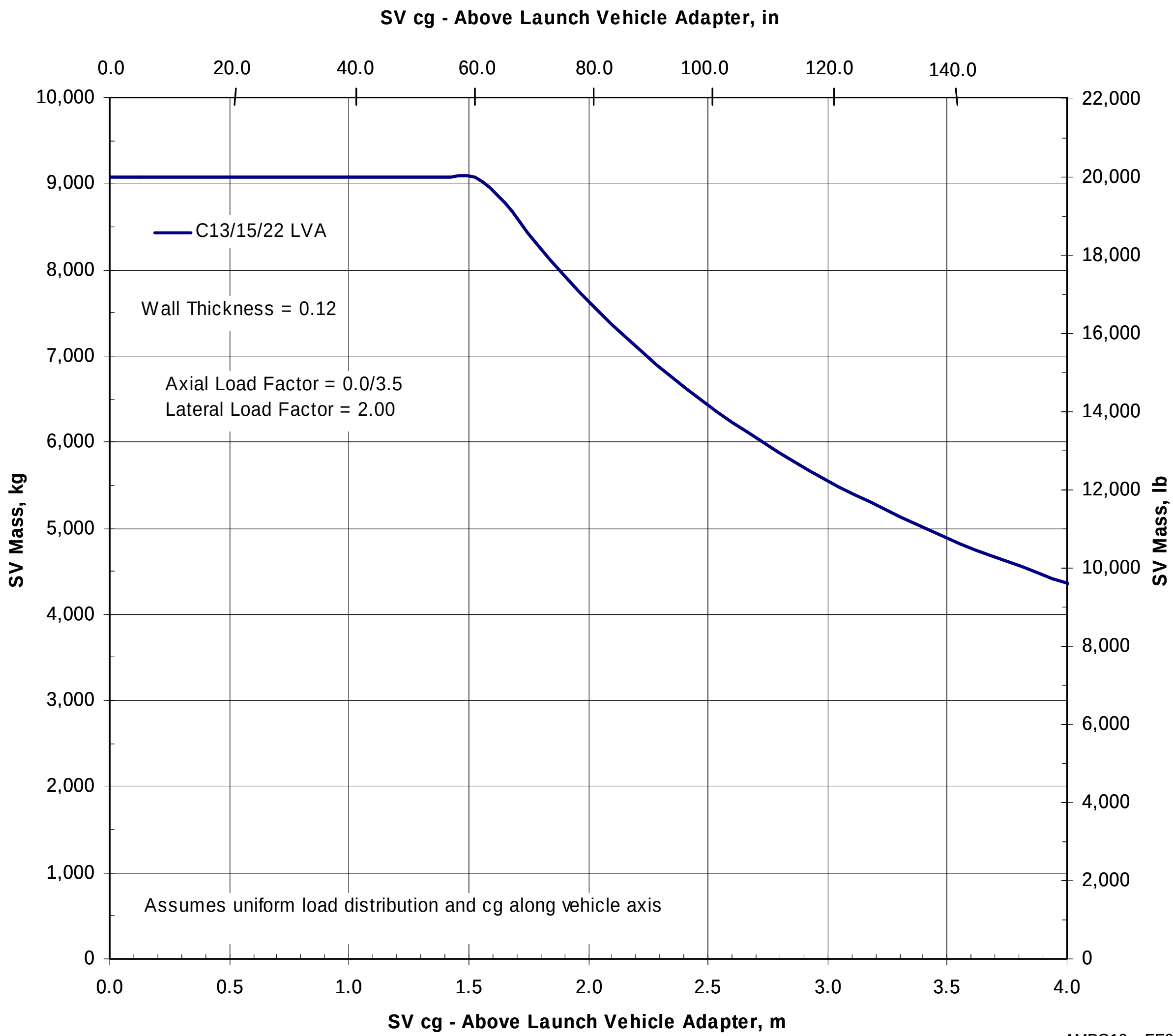

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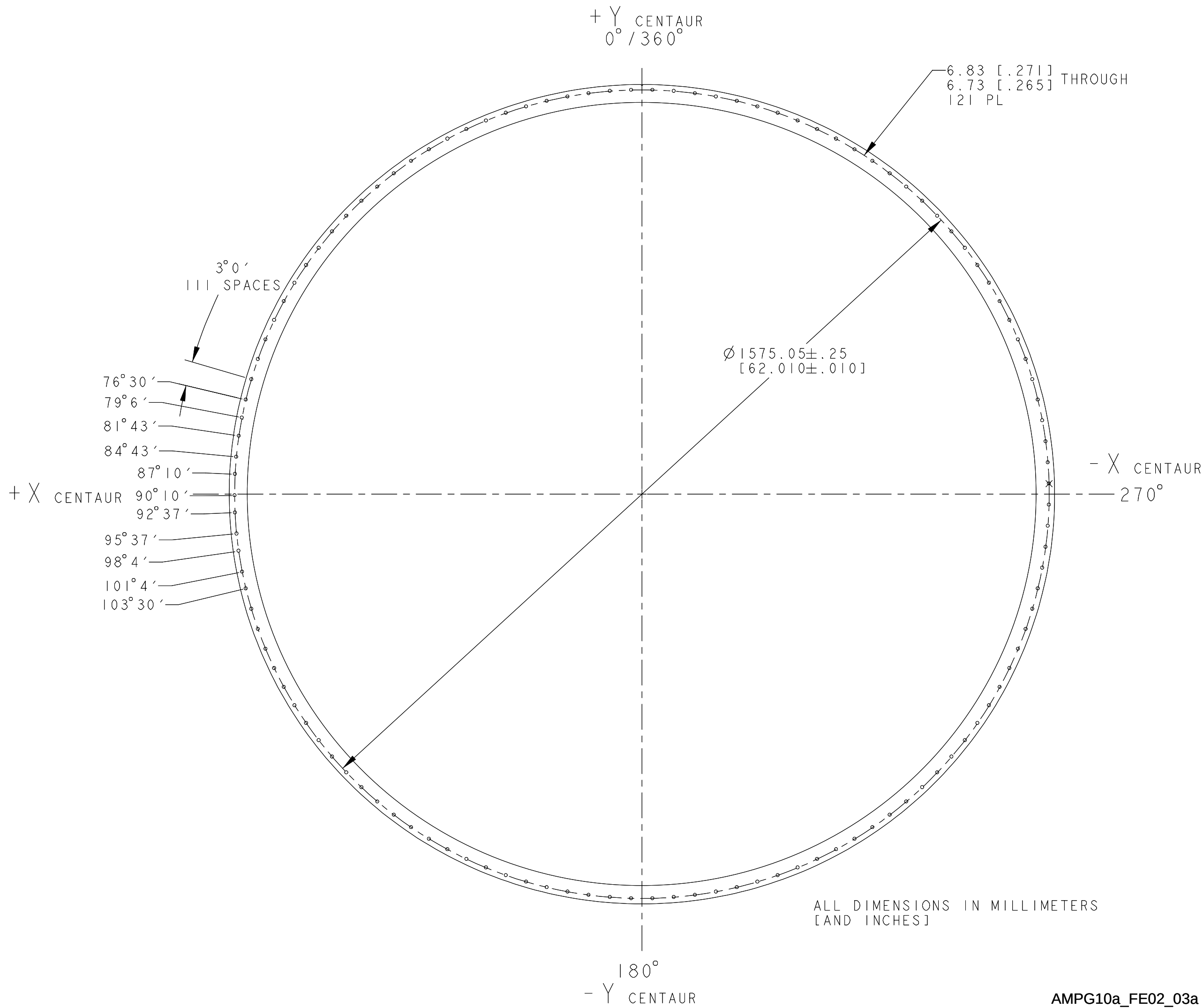
Figure E.2-3: Atlas Launch Vehicle Adapter Interface Requirements



E.3 ATLAS V TYPE A PAYLOAD ADAPTER

The Atlas V Type A payload adapter is designed to support spacecraft with an aft ring diameter of 937 mm (37 in.). Major characteristics of this payload adapter are summarized in Table E.3-1. This adapter is an aluminum skin, stringer, and frame construction with machined forward and aft rings that mate to the spacecraft and launch vehicle forward adapter (Fig. E.3-1). The forward ring has an outer diameter of 945.3 mm (37.215 in.) and forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with the Atlas V SIP requirements. The nominal height of the Type A payload adapter is 762 mm (30.00 in.). The Type A payload adapter supports all hardware that directly interfaces with the spacecraft including the payload separation system, electrical connectors, and mission-specific options, and it includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Table E.3-1: Atlas V Type A Payload Adapter Characteristics

Atlas V Type A Payload Adapter	
Construction	Aluminum Skin/Stringer/ Frame Construction
Mass Properties	PSW, See Section 2.5.1
Payload Capability (Figure E.3-3)	2,100 kg at 1.27 m (4,600 lb at 50 in.)
P/L Sep System	PSS37
Max Shock Levels	Section 3.2.4
Clampband Preload—Installation	23.7 ± 0.1 kN (5,328 ± 22 lb)
Clampband Preload—Flight	23.0 ± 0.5 kN (5,170 ± 112 lb)
Separation Springs	
Number	4
Force per Spring—Max	1 kN (225 lb)

Payload Separation System — The Atlas V Type A payload adapter uses a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Fig. E.3-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to extract, catch, and retain the clampband on

Figure E.3-1: Atlas V Type A Payload Adapter

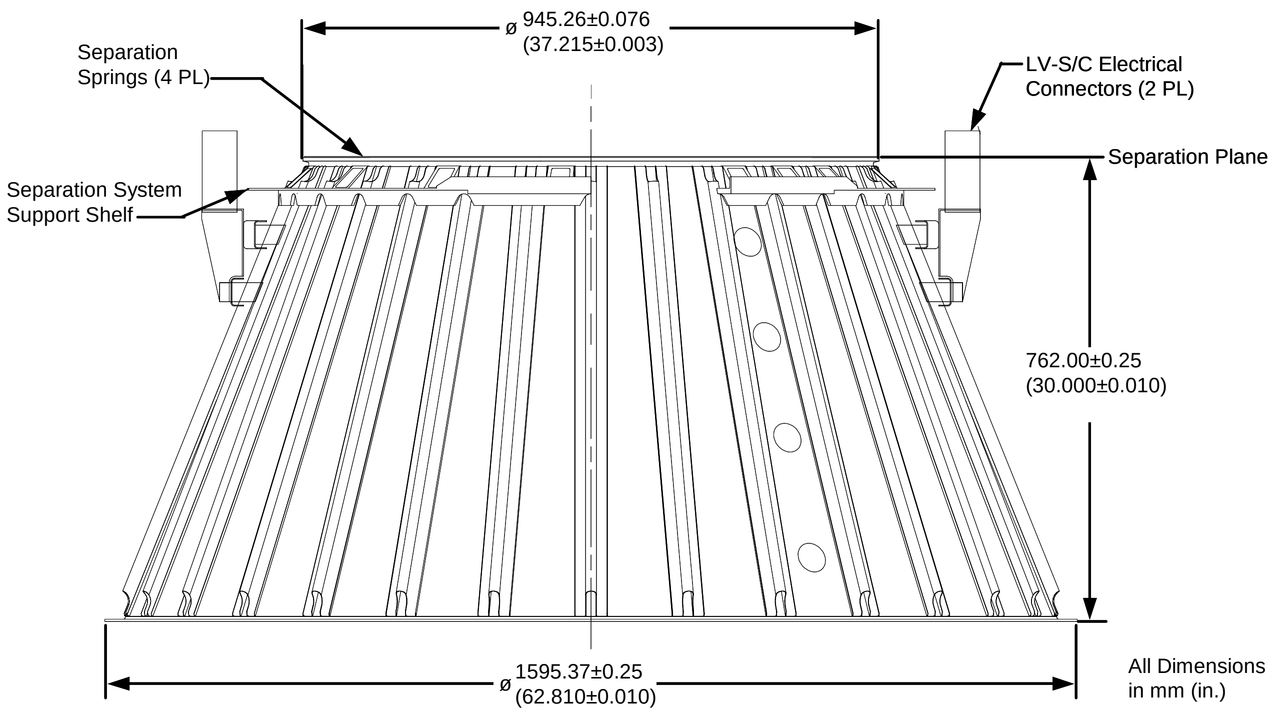
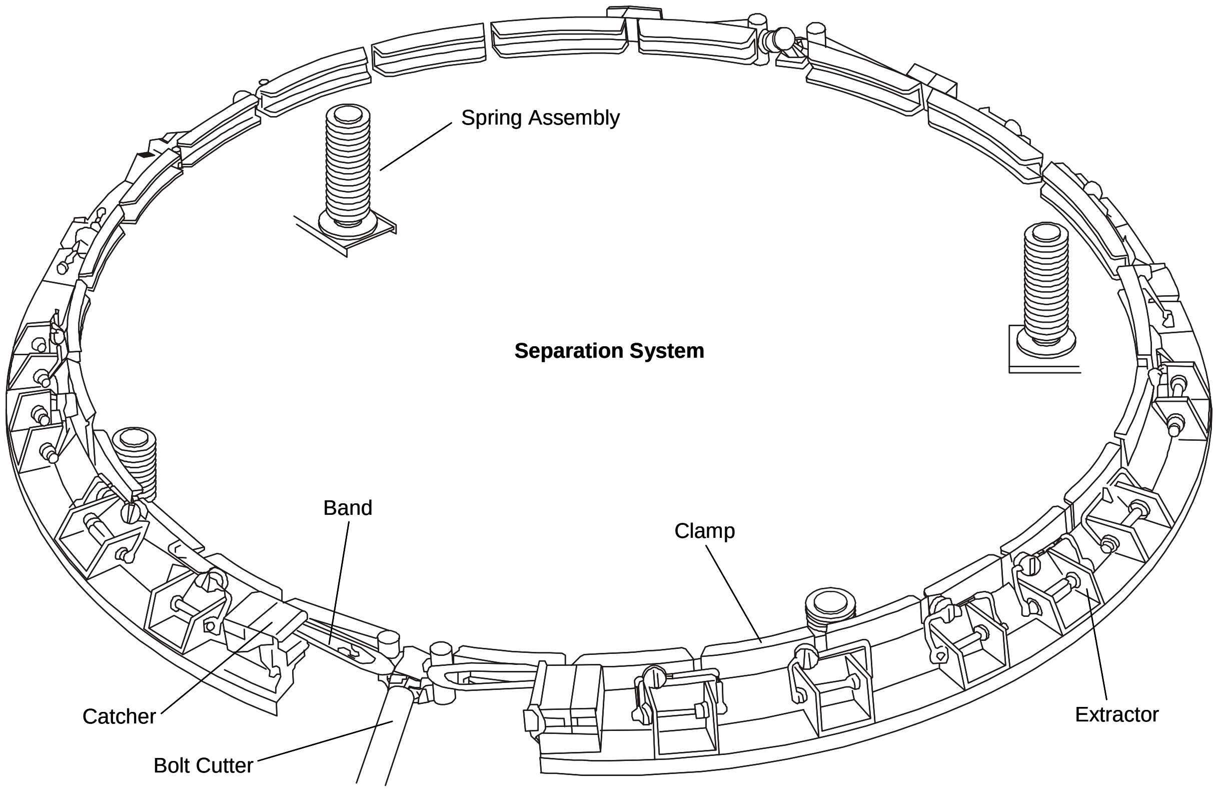


Figure E.3-2: PSS37 Payload Separation System

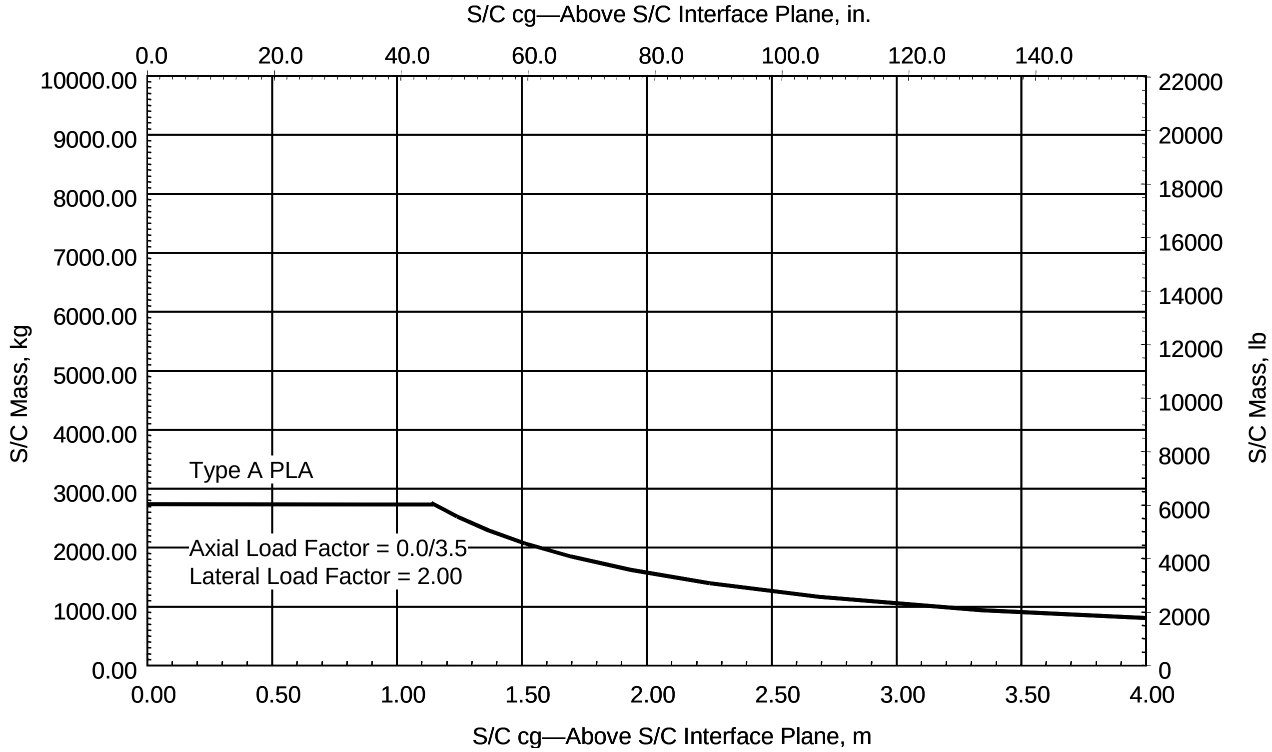


the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a two-piece stainless-steel retaining band that holds the clamp segments in place. The ends of the retaining band are held together by tension bolts. For separation, a pyrotechnically activated bolt-cutter severs the tension bolts, allowing the end of the clamp segments to move apart and release the payload adapter and spacecraft mating rings.

Separation spring assemblies provide the necessary separation energy after the clampband is released and 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.

Payload Adapter Structural Capabilities — Allowable spacecraft weights and longitudinal centers of gravity for the Atlas V Type A payload adapter/separation systems are shown in Figure E.3-3. Spacecraft mass and cg capabilities were determined using generic spacecraft interface ring geometry as shown in Figure E.3-4, and quasi-static load factors (see Section 3.2.1). Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas V program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

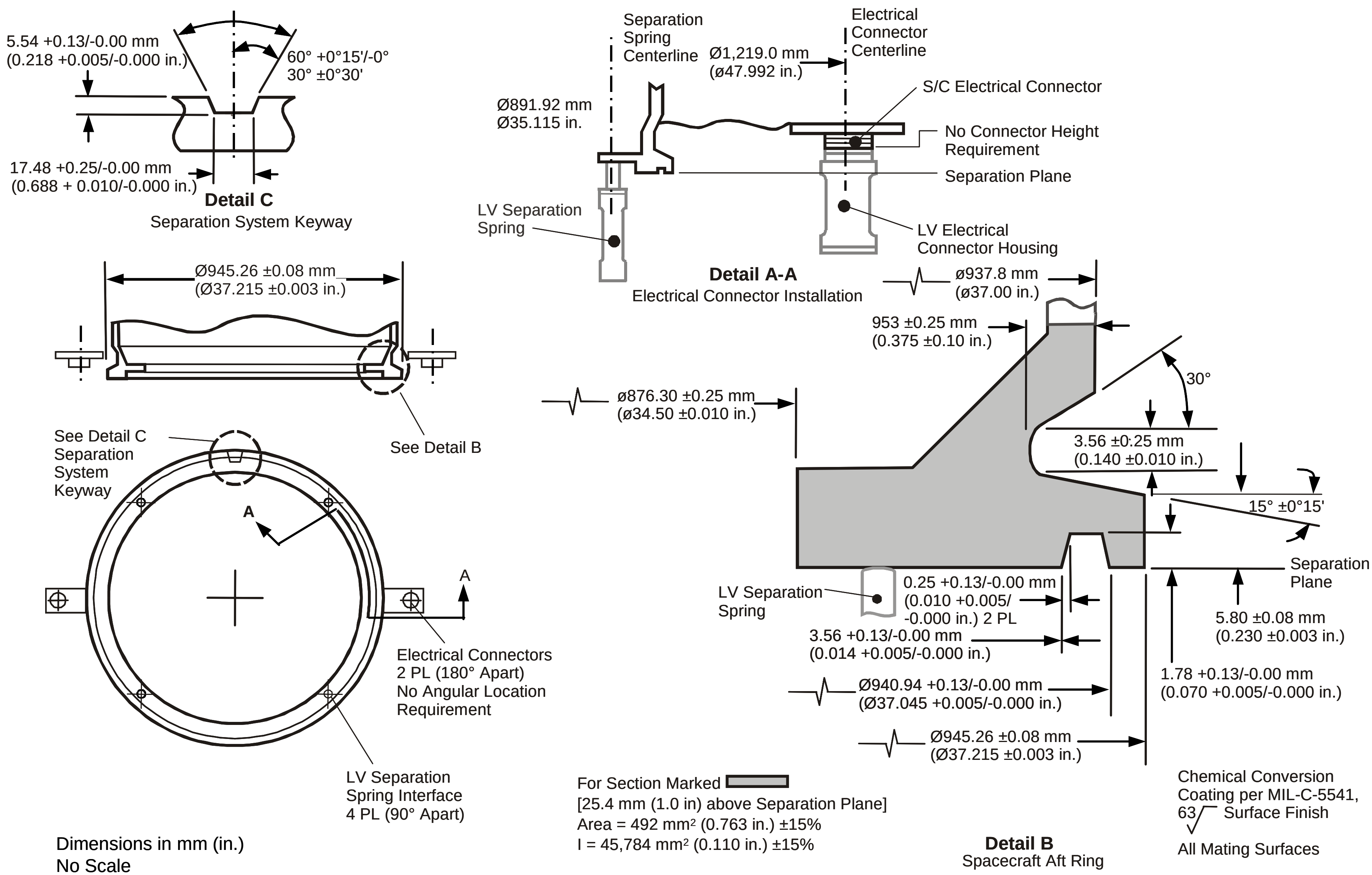
Figure E.3-3: Atlas V Type A Payload Adapter Structural Capability



Payload Adapter Interfaces — 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. Electrical bonding is provided across all interface planes associated with these components. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Interface requirements for these components are shown in Figures E.3-4 and E.3-5. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

Static Payload Envelope — The usable volume for the spacecraft relative to the payload adapter 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. This envelope is designed to allow access to mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and the movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the Atlas V Type A payload adapter are shown in Figure E.3-6.

Figure E.3-4: Spacecraft Interface Requirements



Dimensions in mm (in.)
No Scale

Figure E.3-5: Atlas V Type A Payload Adapter Interface Requirements

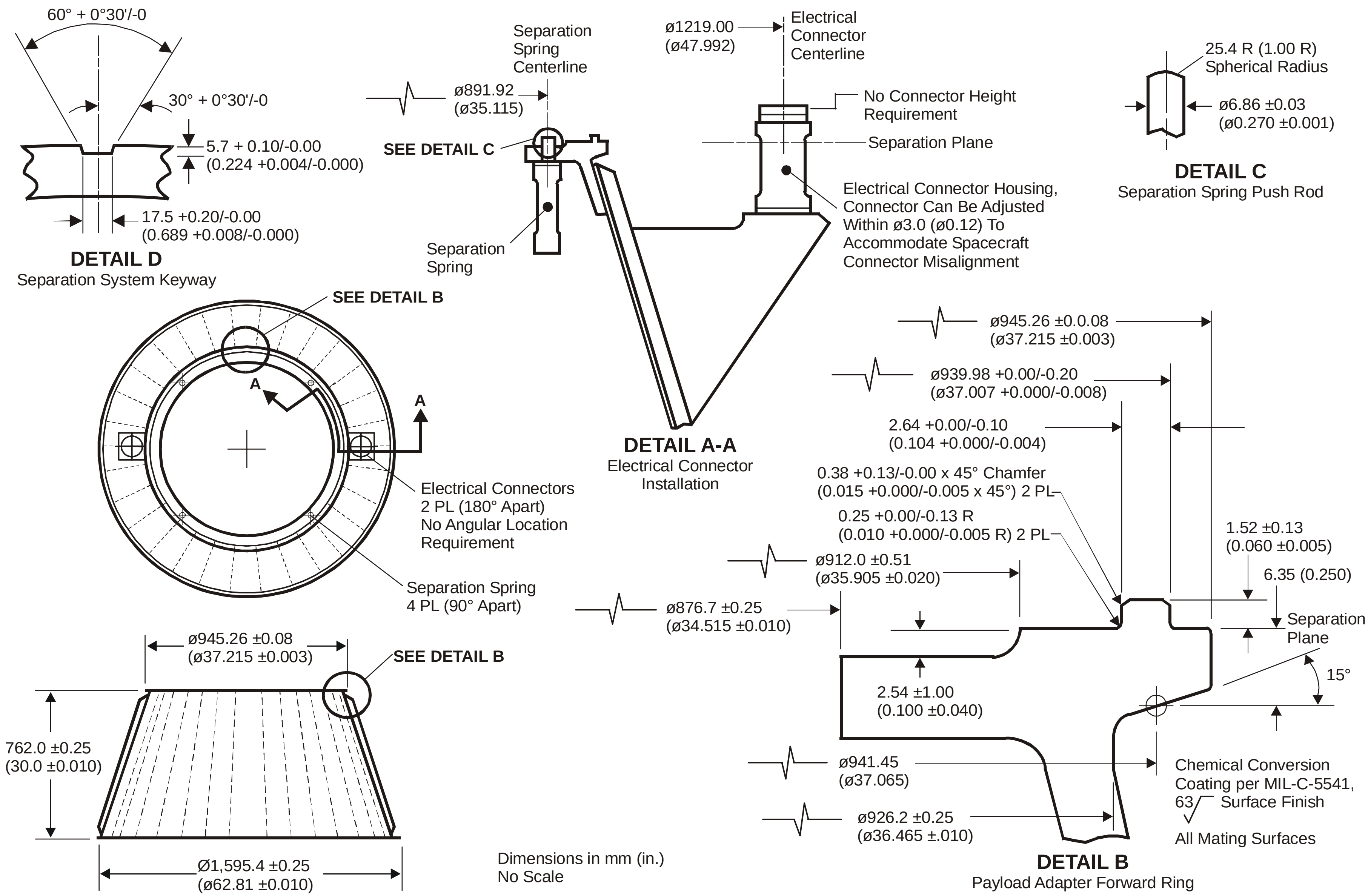
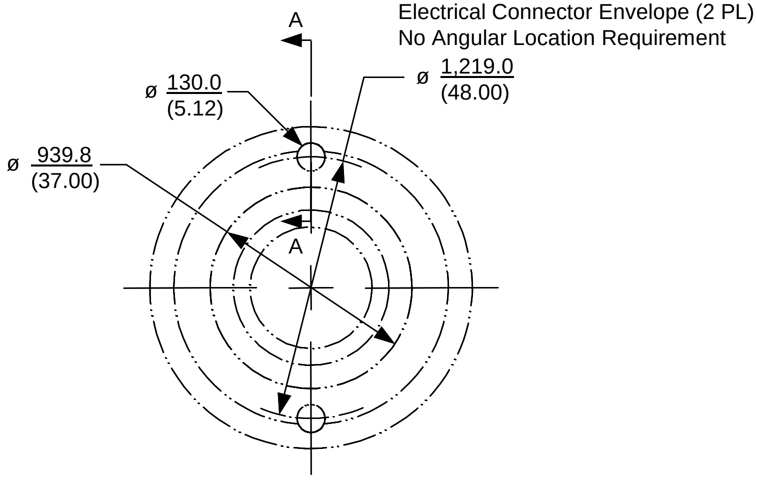
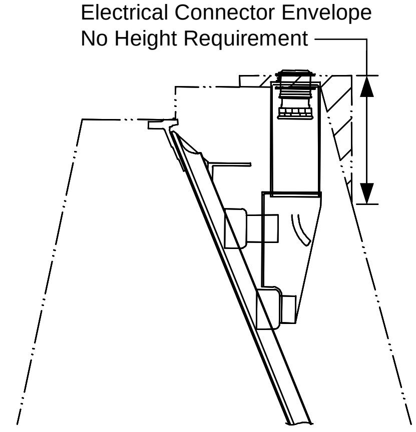
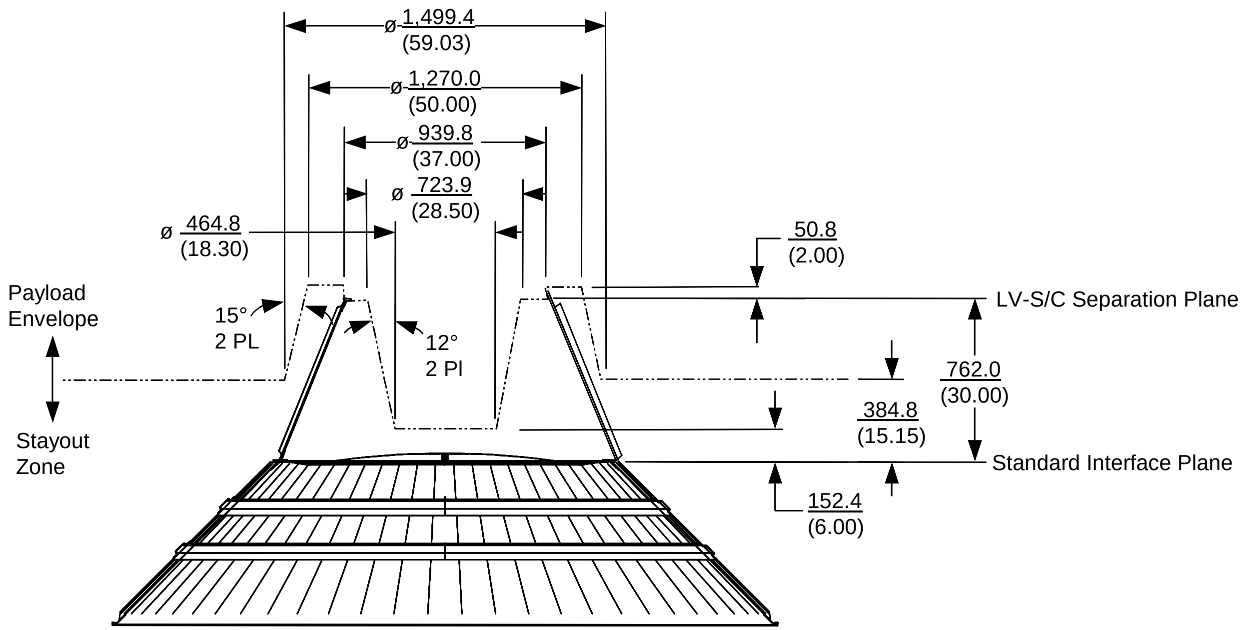


Figure E.3-6: Atlas V Type A Payload Adapter Static Payload Envelope





Section A-A
Electrical Connector Envelope



All Dimensions in mm (in.)

E.4 ATLAS V TYPE B1194 PAYLOAD ADAPTER

The Atlas V Type B1194 payload adapter is designed to support spacecraft with an aft ring diameter of 1,194 mm (47 in.). Major characteristics of this payload adapter are summarized in Table E.4-1. This payload adapter consists of two major sections: the payload separation ring and the launch vehicle adapter (Figure E.4-1). The payload separation ring is a machined aluminum component in the form of a 254 mm (10-in.) high truncated cone. The forward ring has an outer diameter of 1,215 mm (47.835 in.) and forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 evenly spaced bolt holes that allow it to be joined to the launch vehicle adapter. This symmetrical bolt hole pattern allows the payload separation ring and attached spacecraft to be rotated relative to the launch vehicle in 3-degree increments to meet mission-specific requirements. The payload separation ring supports all hardware that directly interfaces with spacecraft, including the payload separation system, electrical connectors, and mission-specific options.

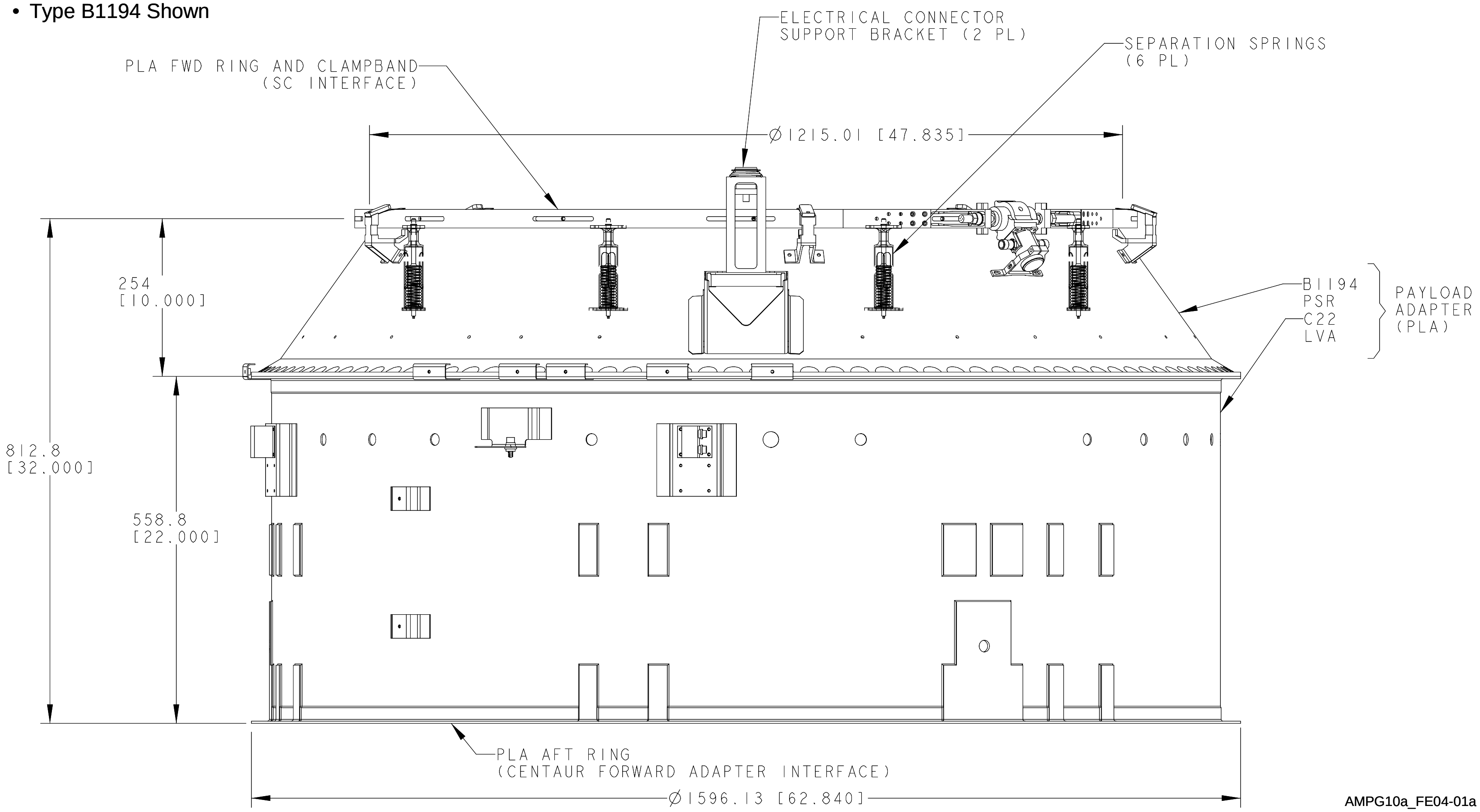
Table E.4-1: Atlas V Type B1194 Payload Adapter Characteristics

Atlas V Type B1194 Payload Adapter	
Construction	Two-Piece, Integrally Machined Aluminum Construction
Mass Properties	PSW, See Section 2.5.1
Payload Capability (Figure E.4-3)	6,300 kg at 2.4 m (13,890 lbs at 94.5 in.)
P/L Sep System	LSPSS1194
Max Shock Levels	Section 3.2.4
Clampband Preload - Installation	66.0 +0.5/ -0 kN (14,840 +112/-0 lb)
Clampband Preload - Flight	60.0 ± 0.5 kN (13,490 ± 112 lb)
Separation Springs	
Number	4–8
Force per Spring - Max	1 kN (225 lb)

The launch vehicle adapter is a machined aluminum component in the form of a monocoque cylinder. The forward ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 bolt holes spaced evenly every 3 degrees that allow it to be joined to the payload separation ring. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 bolt holes that match up with Atlas V SIP requirements. The nominal height of the launch vehicle adapter is 558.8 mm (22.00 in.), but this height may be varied from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.) to meet mission-specific requirements. The launch vehicle adapter includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Figure E.4-1: Atlas V Type B1194 Payload Adapter

- Type B1194 Shown

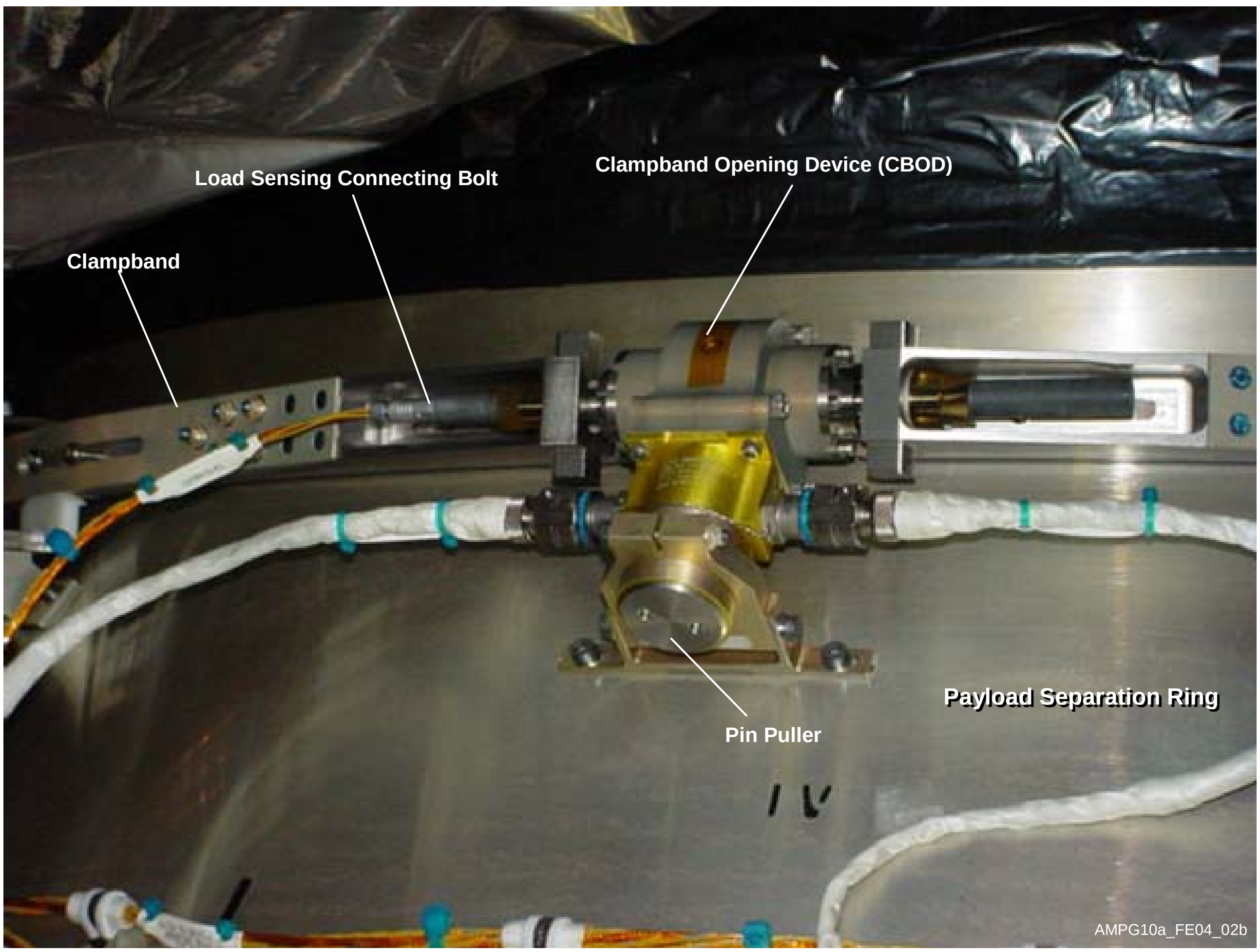


Payload Separation System — The Atlas Type B1194 payload adapter uses a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Figure E.4-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to catch and retain the clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a single-piece aluminum retaining band that holds the clamp segments in place.

The ends of the retaining band are held together by the low shock Clamp-Band Opening Device (CBOD). The CBOD includes release bolts that engage the ends of the clampband. These release bolts are threaded into a flywheel mechanism. During installation and flight, the flywheel is restrained against rotation by a restraining pin. For separation, a pyrotechnically activated pin-puller retracts this pin from the flywheel, allowing it to rotate and eject the release bolts. This system reduces shock in comparison to a conventional bolt-cutter system and is resetable allowing actual flight hardware to be tested during component acceptance testing.

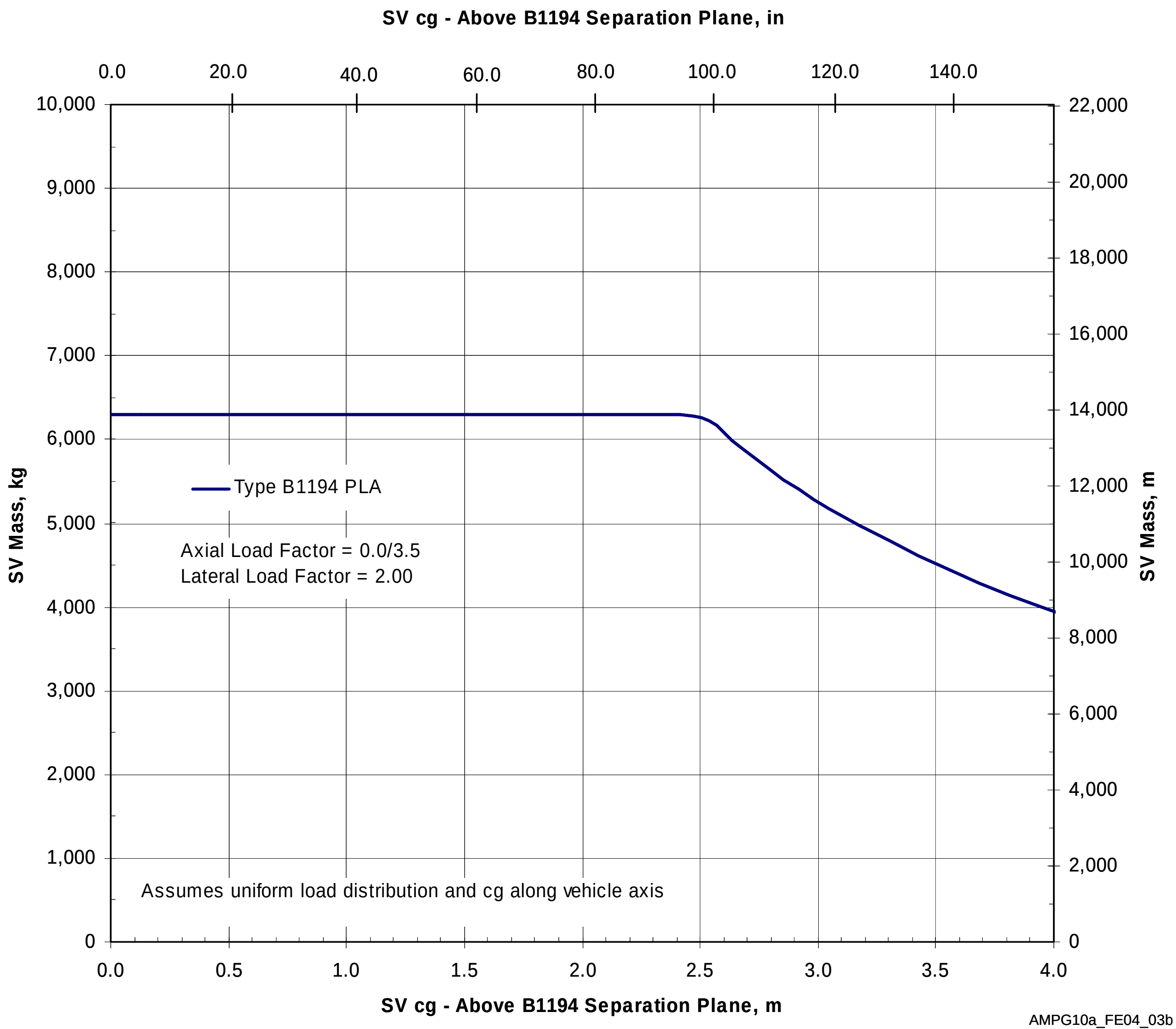
Separation spring assemblies provide the necessary separation energy after the clampband is released and are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through continuity loops installed in the spacecraft electrical connector and wired to the upper-stage instrumentation for monitoring and telemetry verification.

Figure E.4-2: LSPSS1194 Payload Separation System



Payload Adapter Structural Capabilities — Allowable spacecraft weights and longitudinal centers of gravity for the Type B1194 payload adapter/separation systems are shown in Figure E.4-3. These spacecraft mass and center of gravity capabilities were determined using generic spacecraft interface ring geometry as shown in Figure E.4-4, and quasi-static load factors (see Section 3.2.1). Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Additional structural testing is planned to increase the capabilities defined in Figure E.4-3. Coordination with the Atlas V program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

Figure E.4-3: Atlas Type B1194 Payload Adapter Structural Capability



Payload Adapter Interfaces — 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. A payload separation system holds the two rings together for the structural joint and provides the release mechanism for spacecraft separation. Electrical bonding is provided across all interface planes associated with these components. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Interface requirements for these components are shown in Figures E.4-4 and E.4-5. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

Static Payload Envelope — The usable volume for the spacecraft relative to the payload adapter 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. This envelope is designed to allow access to the mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration and, if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the B1194 payload adapter are shown in Figure E.4-6.

Figure E.4-4: Spacecraft Interface Requirements

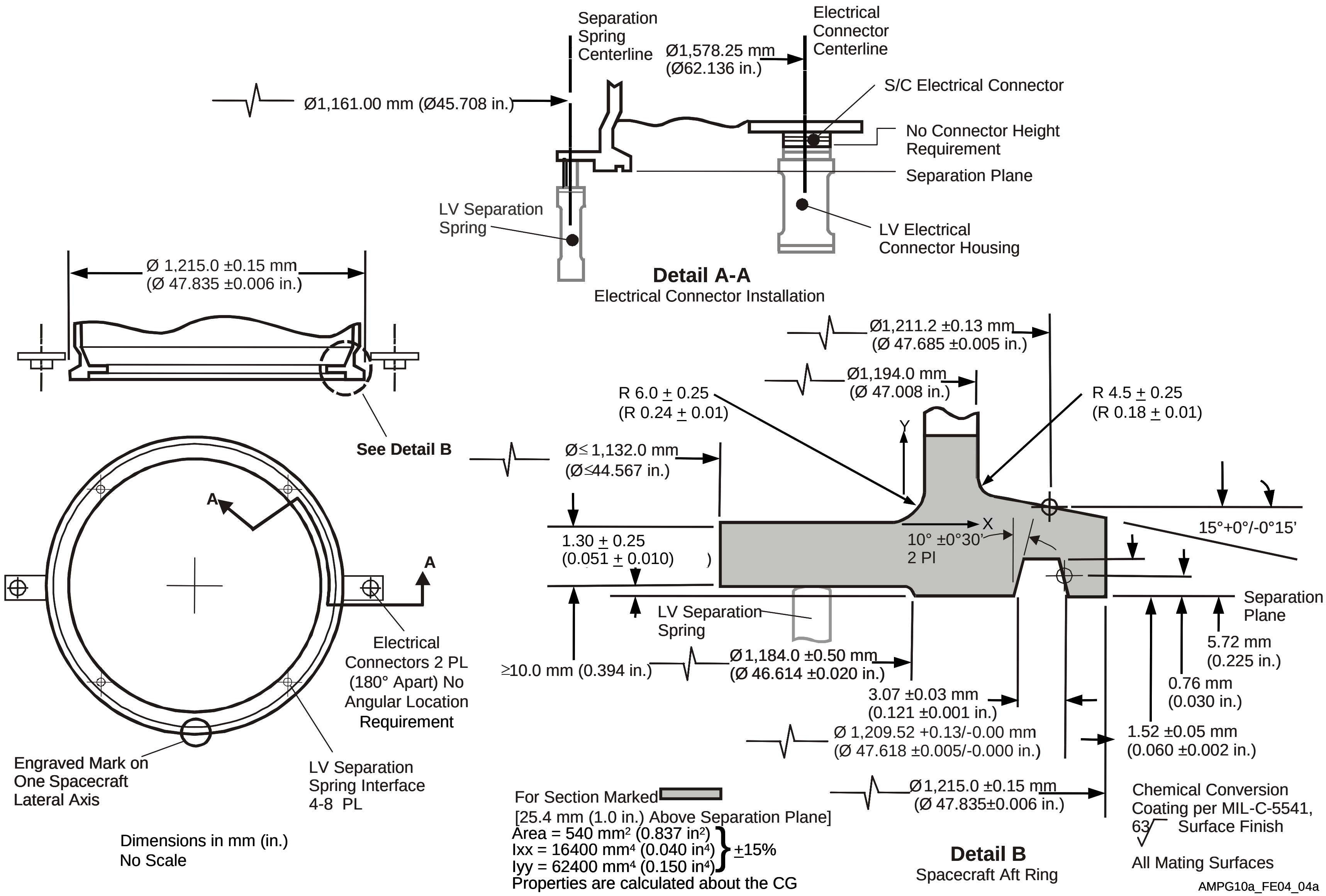


Figure E.4-5: Atlas V Type B1194 PSR Interface Requirements

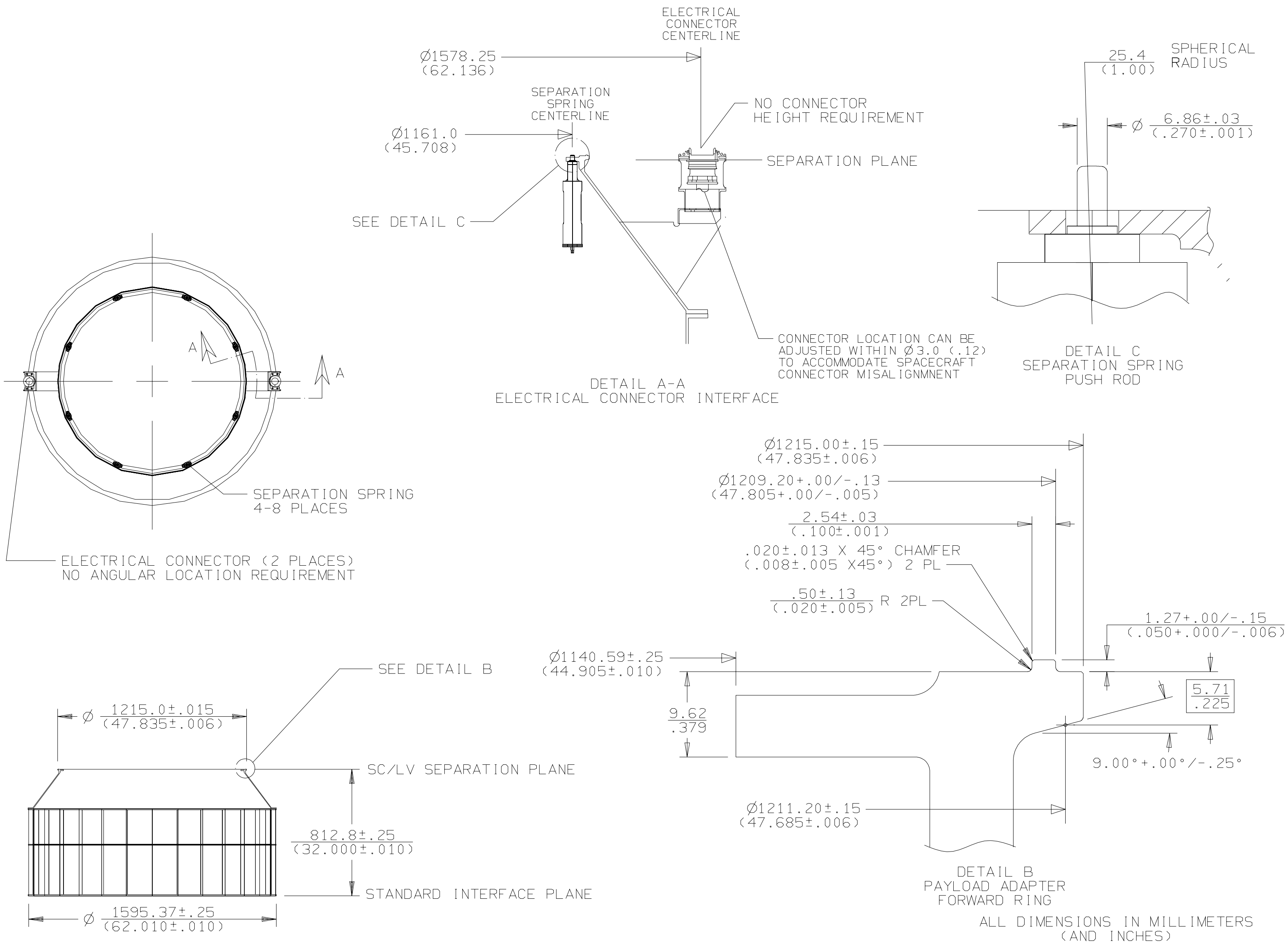


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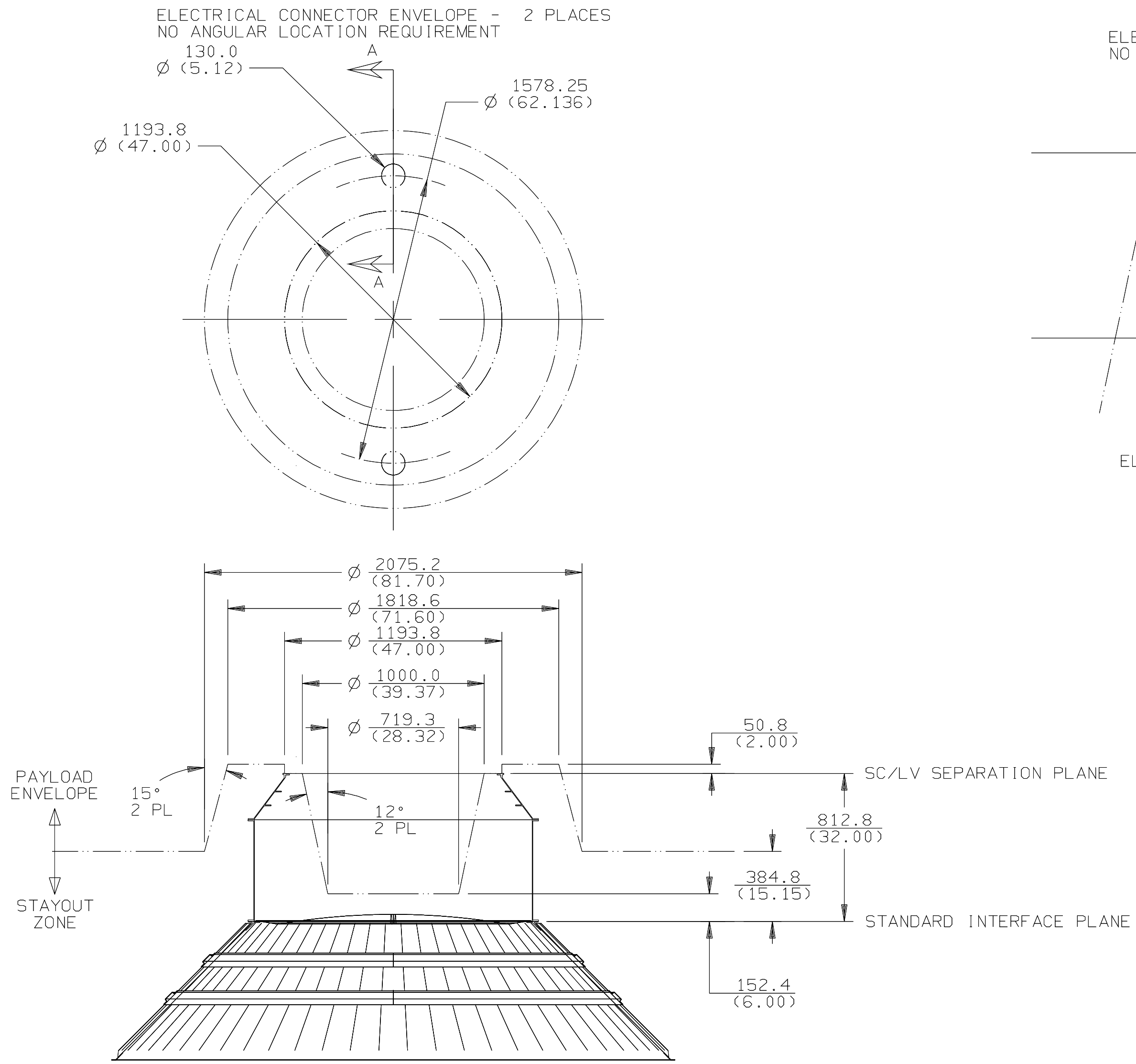
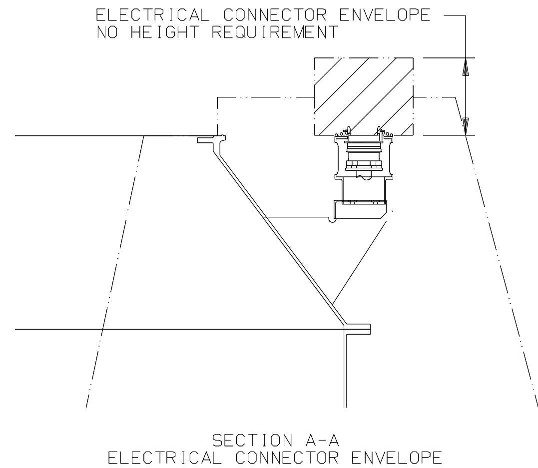
Figure E.4-6: Atlas V Type B1194 Payload Adapter Static Payload Envelope





ALL DIMENSIONS IN MILLIMETERS
(AND INCHES)

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E.5 ATLAS V TYPE D1666 PAYLOAD ADAPTER

The Atlas V Type D1666 payload adapter is designed to support spacecraft with an aft ring diameter of 1,666 mm (66 in.). Major characteristics of this payload adapter are summarized in Table E.5-1. This payload adapter consists of two major sections: the payload separation ring and the launch vehicle adapter (Fig. E.5-1). The payload separation ring is a machined aluminum component in the form of a 330.2-mm (13-in.) high truncated cone. The forward ring has an outer diameter of 1,666.1 mm (65.594 in.) and forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 evenly spaced holes that allow it to be joined to the launch vehicle adapter. This symmetrical hole pattern allows the payload separation ring and attached spacecraft to be rotated relative to the launch vehicle in 3-degree increments to meet mission-specific requirements. The payload separation ring supports all hardware that directly interfaces with spacecraft, including the payload separation system, electrical connectors, and mission-specific options.

Table E.5-1: Atlas V Type D1666 Payload Adapter Characteristics

Atlas V Type D1666 Payload Adapter	
Construction	Two-Piece, Integrally Machined Aluminum
Mass Properties	PSW, See Section 2.5.1
Payload Capability (Figure E.5-3)	6,000 kg at 2.0 m (13,225 lb at 78.8 in.)
P/L Sep System	LSPSS1666
Max Shock Levels	Section 3.2.4
Clampband Preload — Installation	44.0 +0.5/-0 kN (9,770 +112/-0 lb)
Clampband Preload — Flight	40.0 ± 0.5 kN (8,990 ± 112 lb)
Separation Springs	
Number	4-8
Force per Spring — Max	1 kN (225 lb)

The launch vehicle adapter is a machined aluminum component in the form of a monocoque cylinder. The forward ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 holes spaced evenly every 3 degrees that allow it to be joined to the payload separation ring. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with Atlas V SIP requirements. The nominal height of the launch vehicle adapter is 558.8 mm (22.00 in.), but this height may be varied from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.) to meet mission-specific requirements. The launch vehicle adapter includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Payload Separation System — The Atlas V Type D1666 payload adapter uses a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Figure E.5-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to catch and retain the clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a single-piece aluminum retaining band that holds the clamp segments in place. The clamp segments also hold shear pins at 22 locations that fit into shear slots on the spacecraft aft ring and payload adapter forward ring to increase the structural capability of this system. The shear slot pattern on the spacecraft and payload adapter rings for this interface is controlled by matched tooling.

The ends of the retaining band are held together by the low-shock Clamp-Band Opening Device (CBOD). The CBOD includes release bolts that engage the ends of the clampband. These release bolts are threaded into a flywheel mechanism. During installation and flight, the flywheel is restrained against rotation by a restraining pin. For separation, a pyrotechnically activated pin-puller retracts this pin from the flywheel, allowing it to rotate and eject the release bolts. This system reduces shock in comparison to a conventional bolt-cutter system and is resettable, allowing actual flight hardware to be tested during component acceptance testing.

Separation spring assemblies provide the necessary separation energy after the clampband is released and they 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.

Figure E.5-1: Atlas V Type D1666 Payload Adapter

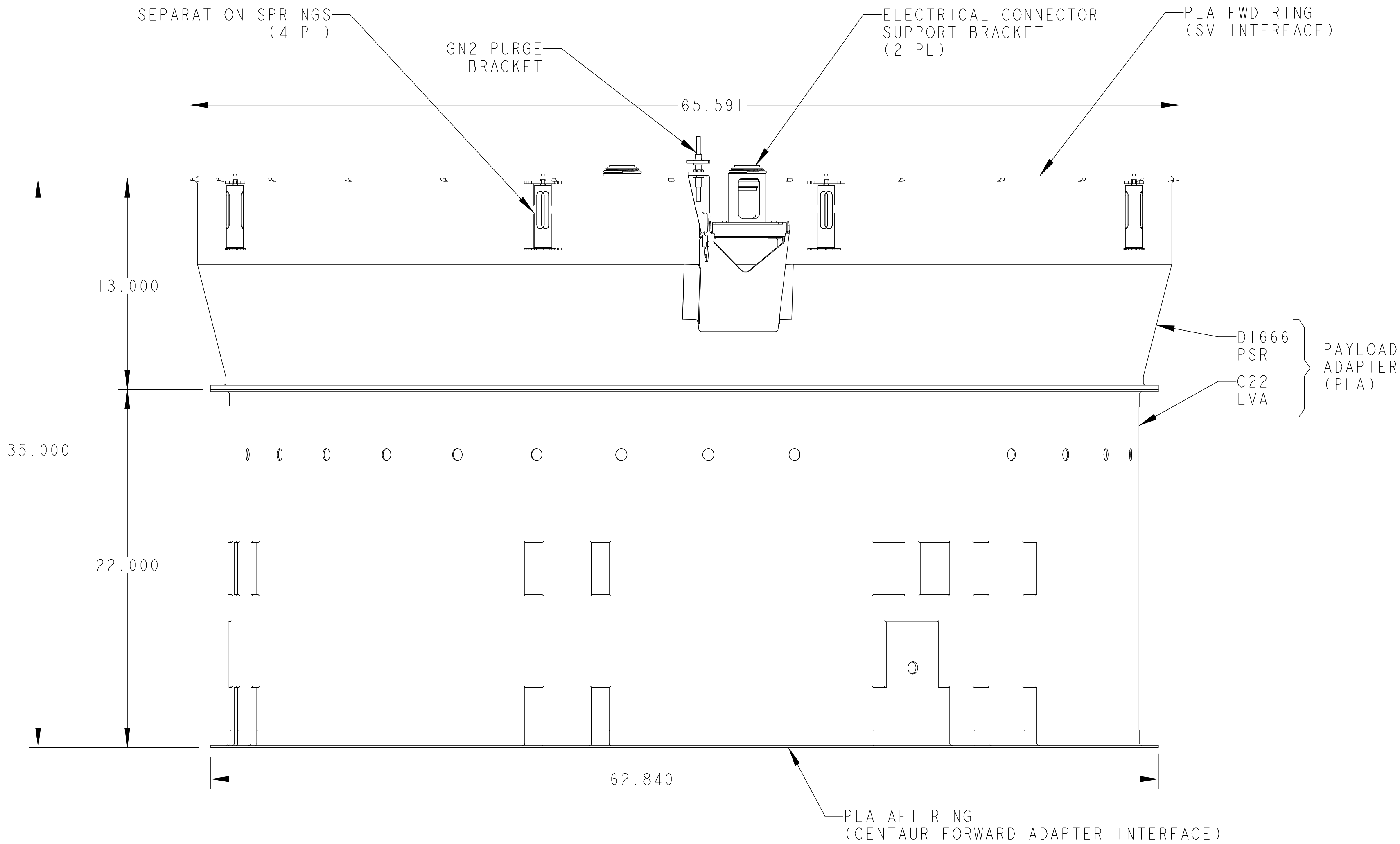

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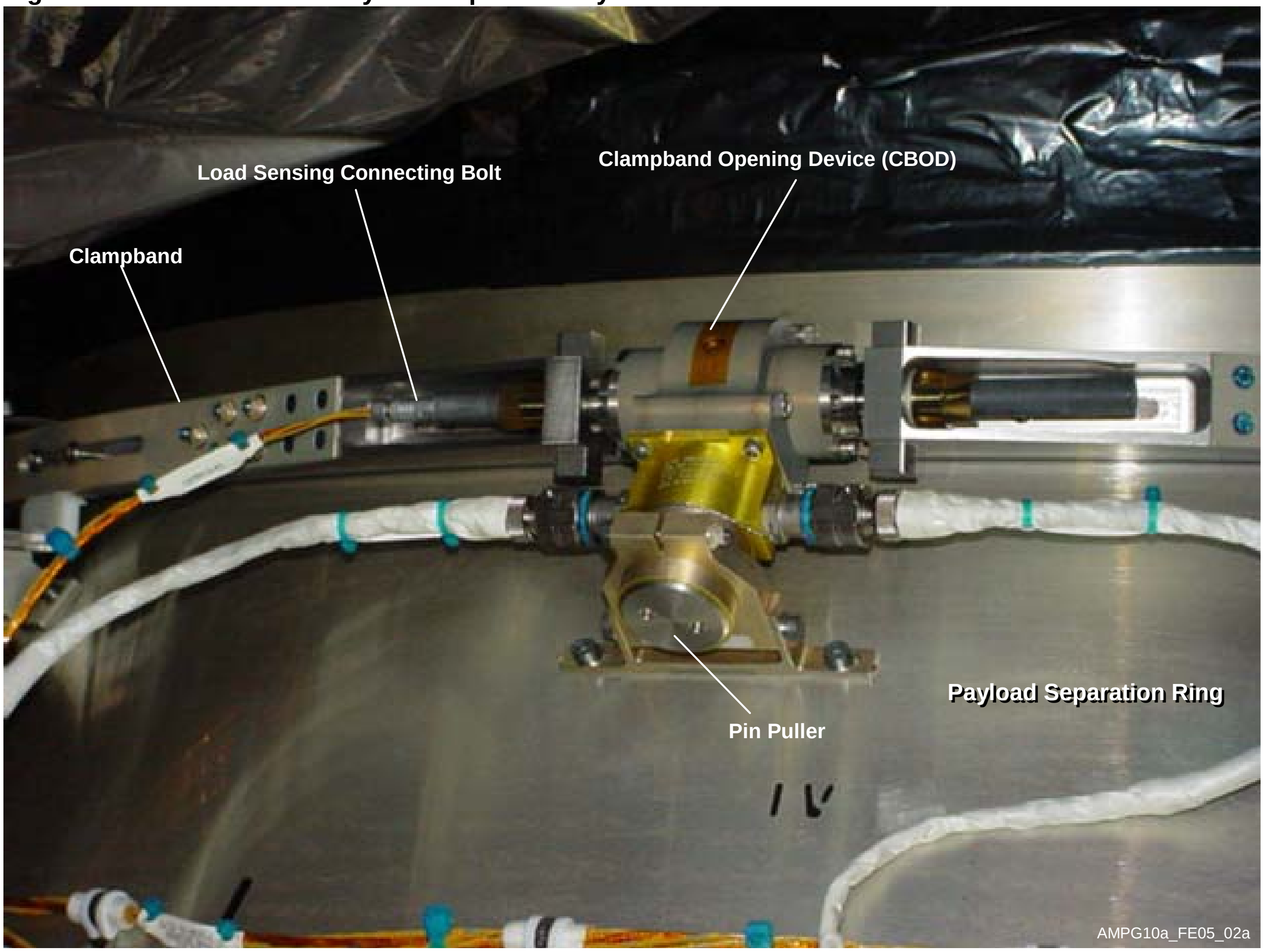
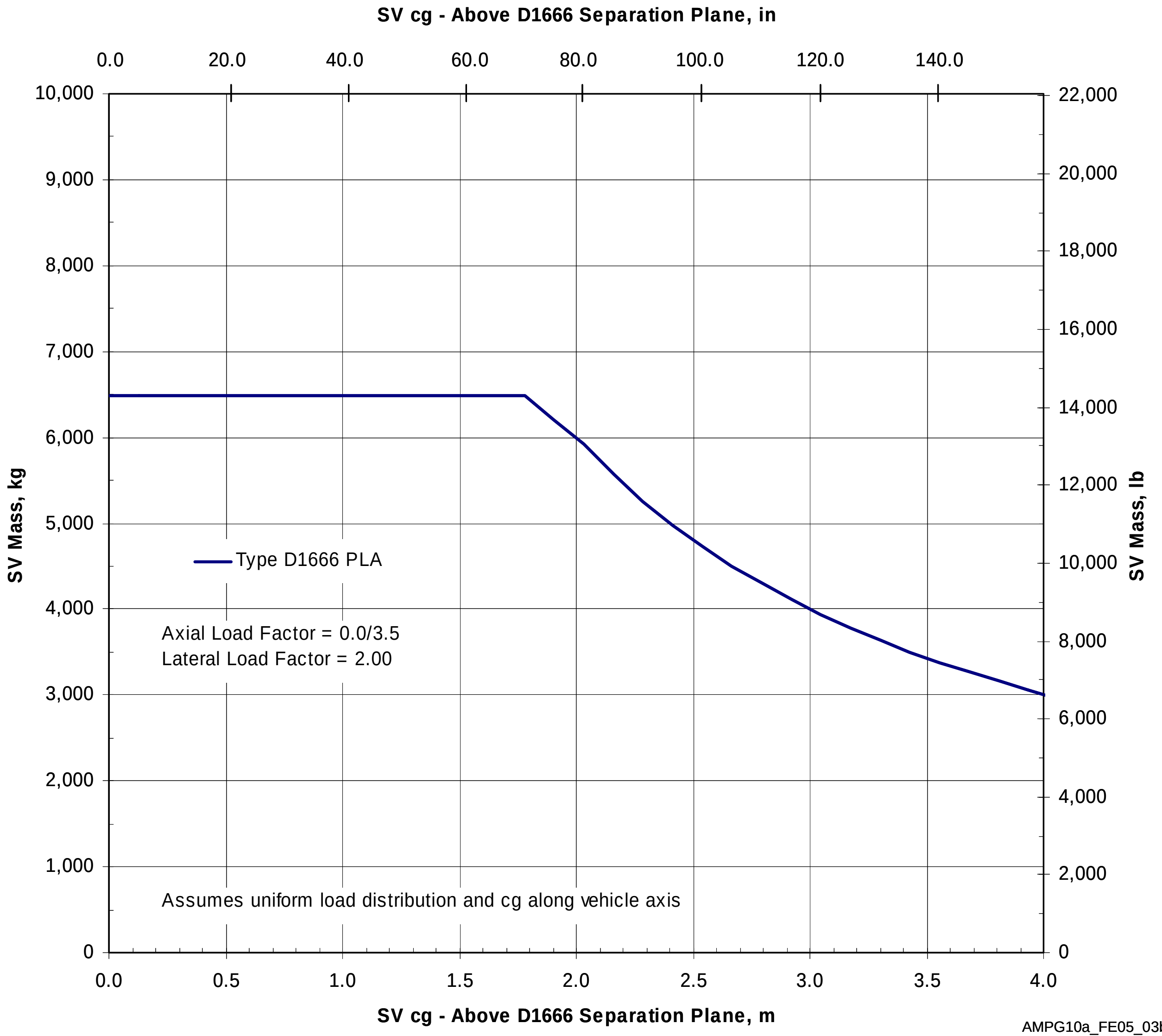
Figure E.5-2: LSPSS1666 Payload Separation System



Payload Adapter Structural Capabilities — Allowable spacecraft weights and longitudinal centers of gravity for the Type D1666 payload adapter/separation systems are shown in Figure E.5-3. These spacecraft mass and center of gravity capabilities were determined using generic spacecraft interface ring geometry, as shown in Figure E.5-4, and quasi-static load factors (see Section 3.2.1). Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas V program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

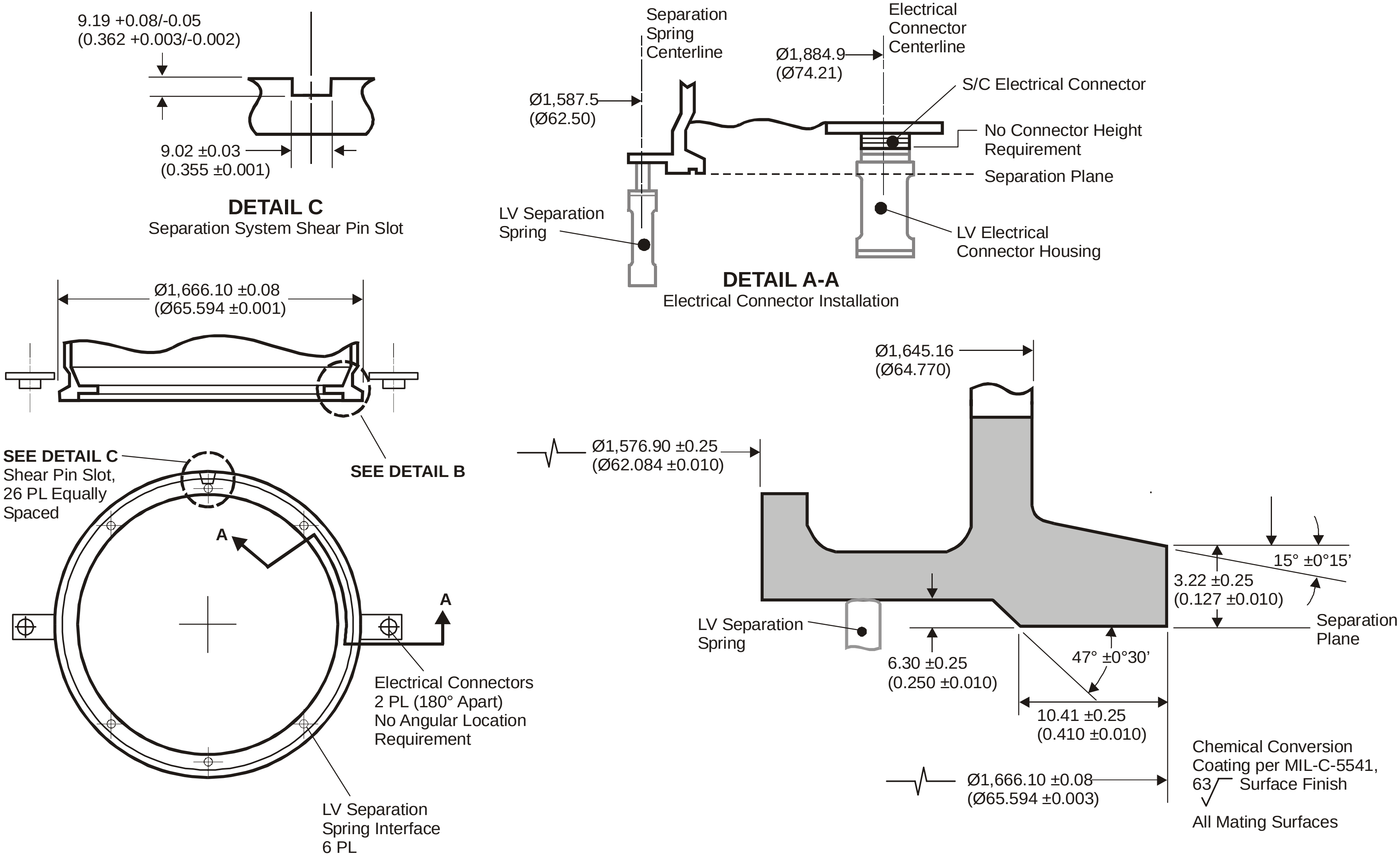
Figure E.5-3: Atlas V Type D1666 Payload Adapter Structural Capability



Payload Adapter Interfaces — 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. A payload separation system holds the two rings together for the structural joint and provides the release mechanism for spacecraft separation. Electrical bonding is provided across all interface planes associated with these components. Interface requirements for these components are shown in Figures E.5-4 and E.5-5. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Interface requirements for these components are shown in Figures E.5-4 and E.5-5. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

Static Payload Envelope — The usable volume for the spacecraft relative to the payload adapter 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. This envelope is designed to allow access to the mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the D1666 payload adapter are shown in Figure E.5-6.

Figure E.5-4: Type D Adapter Spacecraft Interface Requirements



Dimensions in mm (in.)
No Scale

Figure E.5-5: Atlas V Type D1666 Payload Adapter Interface Requirements

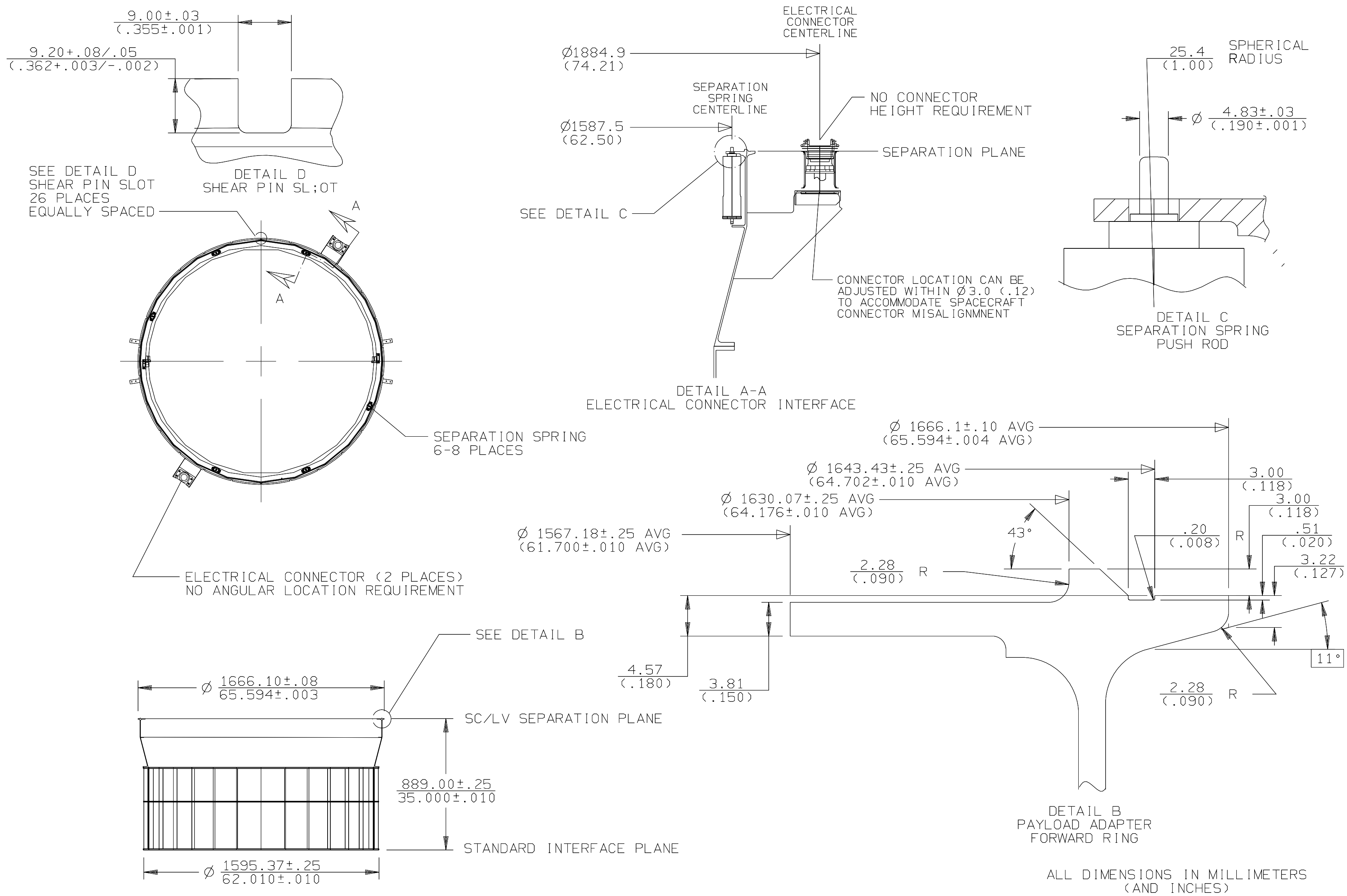


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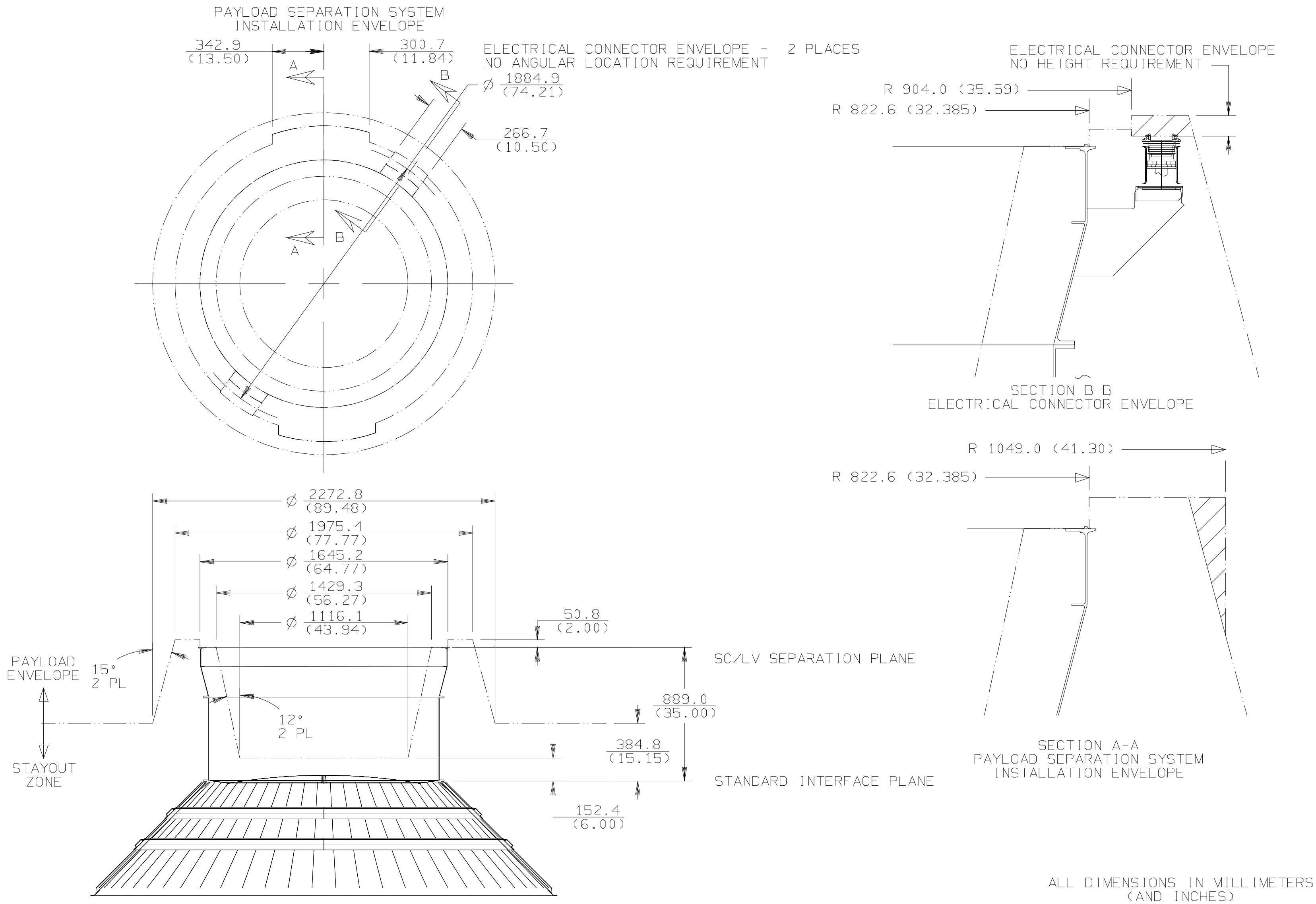
Figure E.5-6: Atlas V Type D1666 Payload Adapter Static Payload Envelope



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E.6 ATLAS V TYPE F1663 PAYLOAD ADAPTER

The Atlas V Type F1663 adapter is being developed to support a payload using a four-hard point interface on a diameter of 1,663 mm (65.50 in.). This payload adapter consists of two major sections: the payload separation ring and the launch vehicle adapter. The payload separation ring is a machined aluminum component in the form of approximately a 558.8-mm (22.00-in.) high truncated cone. The forward ring has four bearing surfaces that incorporate shear cones and bolt holes on an interface diameter of 1,663 mm (65.50 in.) that form the spacecraft separation plane. The payload separation ring supports all hardware that directly interfaces with spacecraft, including the payload separation system, electrical connectors, and mission-specific options.

The launch vehicle adapter is a machined aluminum component in the form of a monocoque cylinder. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 bolt holes that match up with Atlas V standard interface plane requirements. The nominal height of the launch vehicle adapter is 558.8 mm (22.00 in.), but this height may be varied from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.) to meet mission-specific requirements. The launch vehicle adapter includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Information about the payload adapter and spacecraft interface requirements is available upon request to the Atlas V program.

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GLOSSARY

3-D	Three Dimensional
3σ	3-Sigma
∆V	Delta Velocity
A	ampere(s)
Å	angstrom(s)
ac	Alternating Current
A/C	Air Conditioning
ADDJUST	Automatic Determination and Dissemination of Just-Updated Steering Terms
ADMS	Automated Data Monitoring System
ADS	Automatic Destruct System
AFM	Air Force Manual
AFR	Air Force Regulation
AFSCF	Air Force Satellite Control Facility
AFSCN	Air Force Space Control Network
AGE	Aerospace Ground Equipment
AGPO	Atlas Government Program Office
AHU	Air Handling Unit
Alt	Altitude
AMPG	*Atlas Mission Planner's Guide*
ARCU	Atlas Remote Control Unit
AS	Atlas Station
ASOC	Atlas V Spaceflight Operations Center
ATP	Authority To Proceed
ATS	Aft Transition Structure
AVE	Aerospace Vehicle Equipment
AWG	American Wire Gauge
Batt	Battery
BECO	Booster Engine Cutoff
BETO	Booster Engine Thrust Oscillation
BM	Base Module
BOS	Bottom of Steel
BPJ	Booster Package Jettison
BPS	Booster Propulsion System
BPSK	Binary Phase-Shift Keying
BRCU	Booster Remote Control Unit
BSI	British Standards Institute
Btu	British thermal unit(s)
°C	Degree(s) Celsius
C3	Earth Escape
CAD	Computer Aided Design
CBOD	Clampband Opening Device
CCAFS	Cape Canaveral Air Force Station
CCAM	Collision and Contamination Avoidance Maneuver
CCB	Common Core Booster™

CCB	Change Control Board
CCLS	Computer Controlled Launch Set
CCTV	Closed Circuit Television
CVAPS	Centaur Vent and Pressurization System
CDR	Critical Design Review
CFA	Centaur Forward Adapter
CFLR	Centaur Forward Load Reactor
cfm	cubic feet per minute
cg	center of gravity(ies)
CIB	Change Integration Board
C-ISA	Centaur Interstage Adapter
CLA	Coupled Loads Analysis
CLE	Centaur Longitudinal Event
CLS	Commercial Launch Services
cm	centimeter(s)
Cmd	Command
COLA	Collision Avoidance
COPV	Composite Overwrapped Pressure Vessel
C/O	Checkout
CPM	Common Payload Module
CRES	Corrosion Resistant Steel
CS	Centaur Station
CSF	Customer Support Facility
CT	Command Transmitter
CW	Continuous Wave
DAS	Data Acquisition System
dB	decibel(s)
dc	Direct Current
DCMA	Defense Contract Management Agency
DEC	Dual Engine Centaur
Deg	Degree
Dia	Diameter
DLF	Design Load Factor
DoD	Department of Defense
DOF	Degree of Freedom
DOP	Dioctyl Phthalate
DPC	Dual Payload Carrier
DPS-S	Dual Payload Carrier - Short
DPF	Defense System Communications Satellite (DSCS) Processing Facility
DPM	Director Program Management
DSCS	Defense System Communications Satellite
DSI	Direct Spark Ignition
DDSI	Dual Direct Spark Ignition
DSS	Dual Spacecraft System
DSN	Deep Space Network
DV	Delta Velocity

ECS	Environmentally Controlled System
ECU	Electronic Control Unit
EED	Electroexplosive Device
EELV	Evolved Expendable Launch Vehicle
EHA	Electrical Hydraulic Actuator
EM	Electromagnetic
EMA	Electromechanical Actuator
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EMK	Extended Mission Kit
EOC	Engineering Operations Center
EPA	Environmental Protective Agency
EPF	Extended Payload Fairing
ER	Eastern Range
ERB	Engineering Review Board
ERR	Eastern Range Regulation
ESMCR	Eastern Space and Missiles Control Regulation
EVCF	Eastern Vehicle Checkout Facility
EWR	Eastern/Western Range
°F	Degree(s) Fahrenheit
FAA	Federal Aviation Authority
FAB	Final Assembly Building
FAP	Fairing Acoustic Protection
FCA	Flow Control Assembly
FCDC	Flexible Confined Detonating Cord
FCS	Flight Control Subsystem
FM	Flight Model
FMH	Free Molecular Heating
FO	Fiber Optic
FOTS	Fiber Optics Transmission System
FPA	Flight Plan Approval
FPR	Flight Performance Reserve
ft	foot (feet)
ft^2	square feet
FTINU	Fault Tolerant Inertial Navigation Unit
FTS	Flight Termination System
g	gravity
G&C	Guidance and Control
GC^3	Ground, Command, Control, and Communication
GCS	Guidance Commanded Shutdown
GHe	Gaseous Helium
GIDEP	Government-Industry Data Exchange Program
G&N	Guidance and Navigation
GN&C	Guidance, Navigation and Control
GN_2	Gaseous Nitrogen
Gnd	Ground
GORR	Ground Operations Readiness Review

GOWG	Ground Operations Working Group
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
GSO	Geosynchronous Orbit
GSTDN	Ground Spaceflight Tracking and Data Network
GTO	Geosynchronous Transfer Orbit
GTV	Ground Transport Vehicle
He	Helium
HEPA	High Efficiency Particulate Air
HLV	Heavy Lift Vehicle
hp	horsepower
HPF	Hazardous Processing Facility
hr	hour(s)
HVAC	Humidity, Ventilation and Air Conditioning
Hz	hertz
I	Inclination
ICBM	Intercontinental Ballistic Missile
ICD	Interface Control Document
I/F	Interface
IFD	In-Flight Disconnect
IFR	Inflight Retargeting
IIP	Instantaneous Impact Point
ILC	Initial Launch Capability
IMS	Inertial Measurement System
in.	inch(es)
INU	Inertial Navigation Unit
IOC	Initial Operational Capability
IPF	Integrated Processing Facility
IRD	Interface Requirements Document
ISA	Interstage Adapter
ISO	International Organization for Standardization
I_{SP}	Specific Impulse
IST	Integrated System Test
ITA	Integrated Thermal Analysis
ITP	Integrated Test Plan
JPL	Jet Propulsion Laboratory
kg	kilogram(s)
klb	kilopound(s)
km	kilometer(s)
kN	kilonewton(s)
KPa	kilopascal(s)
KSC	Kennedy Space Center
kVA	kilovolt-ampere(s)
kV/m	kilovolt(s) per meter

LAAFB	Los Angeles Air Force Base
LAE	Liquid Apogee Engine
LAN	Local Area Network
lb	pound(s)
LC	Launch Complex
LCC	Launch Control Center
LD	Launch Director
LDA	Launch Site Authority
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LKEI	Khrunichev-Energia International Incorporated
LMCLS	Lockheed Martin Commercial Launch System
LMD	Lockheed Martin Mission Director
LMSSC	Lockheed Martin Space Systems Company
LN_2	Liquid Nitrogen
LO_2	Liquid Oxygen
LOB	Launch Operations Building
LOC	Launch Operations Center
LPF	Large Payload Fairing
LR	Load Ratio
LRB	Liquid Rocket Booster
LPM	Lower Payload Module
LRR	Launch Readiness Review
LSB	Launch and Service Building
LSPSS	Low Shock Payload Separation System
LV	Launch Vehicle
LVA	Launch Vehicle Adapter
LVRT	Launch Vehicle Readiness Test
m	meter(s)
m^2	square meter(s)
M	Million(s)
MAPS	Mission Air Purge System
MDU	Master Data Unit
MECO	Main Engine Cutoff
MES	Main Engine Start
MHz	MegaHertz
MILA	Merritt Island Launch Area
MLP	Mobile Launch Platform
MOC	Mission Operations Center
MP	Mission Peculiar
MPDR	Mission Peculiar Design Review
MR	Mixture Ratio
MRB	Material Review Board
MRS	Minimum Residual Shutdown
m/s	meters per second
MSE	Mission Success Engineers
MSPSP	Missile System Prelaunch Safety Package

MSR	Mission Support Room
MST	Mobile Service Tower
MVB	Main Vehicle Battery
N	Newton(s)
NASA	National Aeronautics and Space Administration
N_2H_4	Hydrazine
NIOSH	National Institute for Occupational Safety and Health
nmi	nautical mile(s)
NRZ-L	Nonreturn-to-Zero Level
NVR	Nonvolatile Residue
OASPL	Overall Sound Pressure Level
OCC	Operations Communication Center
OD	Outer Diameter
OIS	Orbit Insertion Stage
OML	On-orbit Maneuver Lifetime
Op	Operation(s)
ORCA	Ordnance Remote Control Assembly
OSHA	Occupational Health and Safety Administration
OVS	Operational Voice System
PA	Product Assurance
PAFB	Patrick Air Force Base
PAOR	Product Assurance Operations Review
PB	Preburner
PCA	Payload Controlled Area
PCM	Pulse Code Modulation
PDR	Preliminary Design Review
PDSM	*Product Delivery System Manual*
PEB	Payload Equipment Building
PFJ	Payload Fairing Jettison
PFM	Protoflight Model
PHSF	Payload Hazardous Servicing Facility
PIU	Pyroinhibit Unit
Pk	Peak
P/L	Payload
PLA	Payload Adapter
PLCP	Propellant Leak Contingency Plan
PLF	Payload Fairing
PLIS	Propellant-Level Indicating System
PLRR	President’s Launch Readiness Review
PM	Propellant Margin
PMP	Parts, Materials, and Processes
PMPCB	Parts, Materials, and Processes Control Board
PMR	Preliminary Material Review
PO	Park Orbit
PPF	Payload Processing Facility
PQAR	Procurement Quality Assurance Representative

PRD	Program Requirements Document
psf	pound(s) per square foot
psi	pound(s) per square inch
psig	pound(s) per square inch, gage
PSR	Payload Separation Ring
PSS	Payload Separation System
PST	Product Support Team
PSW	Payload Systems Weight
PTC	Payload Test Conductor
PTF	Pressure Test Facility
PU	Propellant Utilization
PVA	Perigee Velocity Augmentation
PVan	Payload Van
PVC	Polyvinyl Chloride
P&W	Pratt & Whitney
Pwr	Power
PYC	Pyrotechnic Control
Q	Dynamic Pressure
QPSK	Q Phase Shift Keying
RAAN	Right Ascension of Ascending Node
RC	Range Coordinator
RCS	Reaction Control System
RCU	Remote Control Unit
R&D	Research and Development
RDU	Remote Data Unit
RF	Radio Frequency
RFTS	Radio Frequency Test Set
RGU	Rate Gyro Unit
RH	Relative Humidity
RLCC	Remote Launch Control Center
Rm	Room
ROCC	Range Operations Control Center
RP	Rocket Propellant
RPM	Revolutions Per Minute
RPO	Radiation Protection Officer
RRGU	Redundant Rate Gyro Unit
RSSR	Range Safety System Report
RTS	Remote Tracking Stations
s	Second(s)
SAEF	Spacecraft Assembly and Encapsulation Facility
SAI	Safe/Arm Initiator
SAR	Safety Assessment Report
SAR	Software Anomaly Report
S/C	Spacecraft
SCAPE	Self-Contained Atmospheric-Protective Ensemble
SCTC	Spacecraft Test Conductor

SDO	San Diego Operations
sec	second(s)
SEC	Single Engine Centaur
SECO	Sustainer Engine Cutoff
Sep	Separation
SERB	Systems Engineering Review Board
SFTS	Secure Flight Termination System
SIL	Systems Integration Laboratory
SIP	Standard Interface Plane
SLC	Space Launch Complex
SLV	Space Launch Vehicle
SMD	Spacecraft Mission Director
SOC	Spacecraft Operations Center
SOW	Statement of Work
SPF	Spacecraft Processing Facility
SPL	Sound Pressure Level
SPRB	Space Program Reliability Board
SQP	Sequential Quadratic Programming
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSI	Spaceport Systems International
Sta	Station
STC	Satellite Test Center
STE	Special Test Equipment
STM	Structural Test Model
STV	Spacecraft Transport Vehicle
S/W	Software
Sync	Synchronous
TDRSS	Tracking and Data Relay Satellite System
Temp	Temperature
Tlm	Telemetry
TO	Transfer Orbit
TOPS	Transistorized Operational Phone System
TVC	Thrust Vector Control
TVCF	Transportable Vehicle Checkout Facility
Umb	Umbilical
UPS	Uninterruptible Power System
URCU	Upperstage Remote Control Unit
USAF	United States Air Force
UT	Umbilical Tower
V	volt(s)
Vac	volt(s), alternating current
VAFB	Vandenberg Air Force Base
Vdc	volt(s) direct current
VGP	Virtual Ground Plane
VIF	Vertical Integration Facility

V/m	volt(s) per meter
VTF	Vertical Test Facility
W	watt(s)
WDR	Wet Dress Rehearsal
WRR	Western Range Regulation
WSMCR	Western Space and Missiles Control Regulation
Xdcr	Transducer
Xe	Gaseous Xenon
XEPF	Extended EPF
$\omega\rho$	Argument of Perigee

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