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Minotaur I User's Guide

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This Minotaur I User's Guide is intended to familiarize potential space launch vehicle users with the Minotaur I launch system, its capabilities and its associated services. All data provided herein is for reference purposes only and should not be used for mission specific analyses. Detailed analyses will be performed based on the requirements and characteristics of each specific mission. The launch services described herein are available for US Government sponsored missions via the United States Air Force (USAF) Space and Missile Systems Center, Detachment 12, Rocket Systems Launch Program (RSLP).

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PAGE

PAGE

PAGE

LIST OF FIGURES

PAGE

PAGE

LIST OF TABLES

LIST OF APPENDICES

A	Ampere
AADC	Alaska Aerospace Development Corporation
AC	Air Conditioning
ACS	Attitude Control System
AFRL	Air Force Research Laboratory
AODS	All-Ordnance Destruct System
ATP	Authority to Proceed
C/CAM	Collision/Contamination Avoidance Maneuver
CCAFS	Cape Canaveral Air Force Station
CDR	Critical Design Review
CFE	Customer Furnished Equipment
CG,cg	Center of Gravity
CLA	Coupled Loads Analysis
CLF	Commercial Launch Facility
cm	Centimeter
CVCM	Collected Volatile Condensable Materials
dB	Decibels
deg	Degrees
DPAF	Dual Payload Attach Fitting
ECS	Environmental Control System
EGSE	Electrical Ground Support Equipment
EMC	Electromagnetic Compatibility
EME	Electromagnetic Environment
EMI	Electromagnetic Interference
FLSA	Florida Spaceport Authority
FM	Frequency Modulation
ft	Feet
FTLU	Flight Termination Logic Unit
FTS	Flight Termination System
g	Gravitational Force
GACS	Ground Air Conditioning System
GFE	Government Furnished Equipment
GN2	Gaseous Nitrogen
GPB	GPS Position Beacon
GPS	Global Positioning System
GSE	Ground Support Equipment
GTO	Geosynchronous Transfer Orbit
HAPS	Hydrazine Auxiliary Propulsion System
HEPA	High Efficiency Particulate Air
Hz	Hertz
I/F	Interface
ICD	Interface Control Drawing
ILC	Initial Launch Capability
IMU	Inertial Measurement Unit
in	Inch
INS	Inertial Navigation System
ISO	International Standardization Organization
IVT	Interface Verification Test
kbps	Kilobits per Second
kg	Kilograms
km	Kilometer
lb	Pound
lbm	Pound(s) of Mass
LCR	Launch Control Room
LEV	Launch Equipment Vault
LITVC	Liquid Injection Thrust Vector Control
LOCC	Launch Operations Control Center
LRR	Launch Readiness Review
LSA	Lower Stack Assembly
LSE	Launch Support Equipment
m/s	Meters per Second
Mbps	Mega Bits per Second
mA	Milliamps
MACH	Modular Avionics Control Hardware
MDR	Mission Design Review
MHz	Megahertz
MIL-STD	Military Standard
MIWG	Mission Integration Working Group
mm	Millimeter
MMODS	Modular Mechanical Ordnance Destruct System
MPA	Multiple Payload Adapter
MRD	Mission Requirements Document
MRR	Mission Readiness Review
MSPSP	Missile System Prelaunch Safety Package
ms	Millisecond
N/A	Not Applicable
NCU	Nozzle Control Unit
NGC	Northrop Grumman Mission Systems
nmi	Nautical Mile
OD	Operations Directive
OD	Outside Dimension
ODM	Ordnance Driver Module
OR	Operations Requirements Document
OSP	Orbital Suborbital Program
P/L	Payload
PACS	Pad Air Conditioning System

PAF	Payload Attach Fitting
PCM	Pulse Code Modulation
PDR	Preliminary Design Review
PI	Program Introduction
PID	Proportional-Integral-Derivative
POC	Point of Contact
ppm	Parts Per Million
PRD	Program Requirements Document
PSD	Power Spectral Density
PSP	Prelaunch Safety Package
RCS	Roll Control System
RF	Radio Frequency
RGIU	Rate Gyro Interface Unit
RGU	Rate Gyro Unit
rpm	Revolutions per Minute
RSLP	Rocket Systems Launch Program
RWG	Range Working Group
S&A	Safe & Arm
scfm	Standard Cubic Feet per Minute
SEB	Support Equipment Building
sec	Second(s)
SINDA	Finite Element Thermal Analysis Tool Trade Name
SLC	Space Launch Complex
SLV	Space Launch Vehicle
SMC	Space and Missile Systems Center
SOC	Statement of Capabilities
SPL	Sound Pressure Level
SRM	Solid Rocket Motor
SRS	Shock Response Spectrum
SRSS	Softride for Small Satellites
SSI	Spaceport Systems International
STA	Station
TLV	Target Launch Vehicle
TML	Total Mass Loss
TVC	Thrust Vector Control
UDS	Universal Documentation System
UFS	Ultimate Factor of Safety
USAF	United States Air Force
V/M	Volts per Meter
VAB	Vehicle Assembly Building
VAFB	Vandenberg Air Force Base
W	Watt
WFF	Wallops Flight Facility
WP	Work Package
YFS	Yield Factor of Safety

1. INTRODUCTION

This User's Guide is intended to familiarize payload mission planners with the capabilities of the Orbital Suborbital Program (OSP) Minotaur I Space Launch Vehicle (SLV) launch service. This document provides an overview of the Minotaur I system design and a description of the services provided to our customers. Minotaur I offers a variety of enhanced options to allow the maximum flexibility in satisfying the objectives of single or multiple payloads.

The users handbook is not intended as a design document but rather it is to be used to select a launch vehicle that meets the requirements of the payload. This document describes typical environments seen on previous missions. Each spacecraft is unique and will require detailed analysis early in the program.

The primary mission of Minotaur I is to provide low cost, high reliability launch services to government-sponsored payloads. Minotaur I accomplishes this by using flight-proven components with significant flight heritage such as surplus Minuteman II boosters, the upper stage Pegasus motors, the Pegasus Fairing and Attitude Control System, and a mix of Pegasus, Taurus, and sub-orbital Avionics, all with a proven, successful track record. The philosophy of placing mission success as the highest priority is reflected in the success and accuracy of all Minotaur I missions to date. The use of flight-proven, surplus components has resulted in a highly responsive vehicle with one of the fastest timelines of all commercially available vehicles.

The Minotaur I launch vehicle system is composed of a flight vehicle and ground support equipment. Each element of the Minotaur I system has been developed to simplify the mission design and payload integration process and to provide safe, reliable space launch services. This User's Guide describes the basic elements of the Minotaur I system as well as optional services that are available. In addition, this document provides general vehicle performance, defines payload accommodations and environments, and outlines the Minotaur I mission integration process.

The Minotaur I system can operate from a wide range of launch facilities and geographic locations. The system is compatible with, and will typically operate from, commercial spaceport facilities and existing U.S. Government ranges at Vandenberg Air Force Base (VAFB), Cape Canaveral Air Force Station (CCAFS), Wallops Flight Facility (WFF), and Kodiak Island, Alaska. This User's Guide describes Minotaur I-unique integration and test approaches (including the typical operational timeline for payload integration with the Minotaur I vehicle) and the existing ground support equipment that is used to conduct Minotaur I operations.

2. MINOTAUR I LAUNCH SERVICE

2.1. Minotaur I Launch System Overview

The Minotaur I launch vehicle, shown in Figure 2-1, was developed by Orbital for the United States Air Force (USAF) to provide a cost effective, reliable and flexible means of placing small satellites into orbit. An overview of the system and available launch services is provided within this section, with specific elements covered in greater detail in the subsequent sections of this User's Guide.

Figure 2-1. OSP Minotaur I Launch Vehicle

Minotaur I has been designed to meet the needs of United States Government-sponsored customers at a lower cost than commercially available alternatives through the use of surplus Minuteman boosters. The requirements of that program stressed system reliability, transportability, and operation from multiple launch sites. Minotaur I draws on the successful heritage of three launch vehicles: Orbital's Pegasus and Taurus space launch vehicles and the Minuteman II system of the USAF. The upper two stages and avionics of Minotaur I are derived from the Pegasus and Taurus systems, providing a combined total of more than 35 successful space launch missions. Orbital's efforts have enhanced or updated Pegasus and Taurus avionics components to meet the payload-support requirements of the OSP program. Combining these improved subsystems with the long successful history of the Minuteman II boosters has resulted in a simple, robust, self-contained launch system that has been completely successful in its flights to date and is fully operational to support government-sponsored small satellite launches. The Minotaur I Launch system has been successfully demonstrated on four missions (as of January 2006).

The Minotaur I system also includes a complete set of transportable Launch Support Equipment (LSE) designed to allow Minotaur I to be operated as a self-contained satellite delivery system. To accomplish this goal, the Electrical Ground Support Equipment (EGSE) has been developed to be portable and adaptable to varying levels of infrastructure. While the Minotaur I system is capable of self-contained operation using portable vans to house the EGSE, it is typically launched from an established range where the EGSE can be housed in available, permanent structures or facilities. This has been the case for the first launches from VAFB.

The vehicle and LSE are designed to be capable of launch from any of the four commercial Spaceports (Alaska, California, Florida, and Virginia), as well as from existing U.S. Government facilities at VAFB and CCAFS. The Launch Control Room (LCR) serves as the actual control center for conducting a Minotaur I launch and includes consoles for Orbital, range safety, and customer personnel. Further description of the Launch Support Equipment is provided in Section 2.4.

2.2. Minotaur I Launch Service

The Minotaur I Launch Service is provided through the combined efforts of the USAF and Orbital, along with associate contractors including NGC and Commercial Spaceports. The primary customer interface will be with the USAF Space and Missile Systems Center, Detachment 12, Rocket Systems Launch Program (RSLP or SMC Det 12/RP). Orbital is the launch vehicle provider. For brevity, this integrated team effort will be referred to as "OSP". Where interfaces are directed toward a particular member of the team, they will be referred to directly (i.e., "Orbital" or "RSLP").

OSP provides all of the necessary hardware, software and services to integrate, test and launch a satellite into its prescribed orbit. In addition, OSP will complete all the required agreements, licenses and documentation to successfully conduct Minotaur I operations. All Minotaur I production and integration processes and procedures have been demonstrated and are in place for future Minotaur I missions. The Minotaur I mission integration process completely identifies, documents, and verifies all spacecraft and mission requirements. This provides a solid basis for initiating and streamlining the integration process for future Minotaur I customers.

2.3. Minotaur I Launch Vehicle

The Minotaur I vehicle, shown in expanded view in Figure 2-2, is a four stage, inertially guided, all solid propellant ground launched vehicle. Conservative design margins, state-of-the-art structural systems, a modular avionics architecture, and simplified integration and test capability, yield a robust, highly reliable launch vehicle design. In addition, Minotaur I payload accommodations and interfaces have been designed to satisfy a wide range of potential payload requirements.

2.3.1. Lower Stack Assembly

The Lower Stack Assembly (LSA) consists of the refurbished Government Furnished Equipment

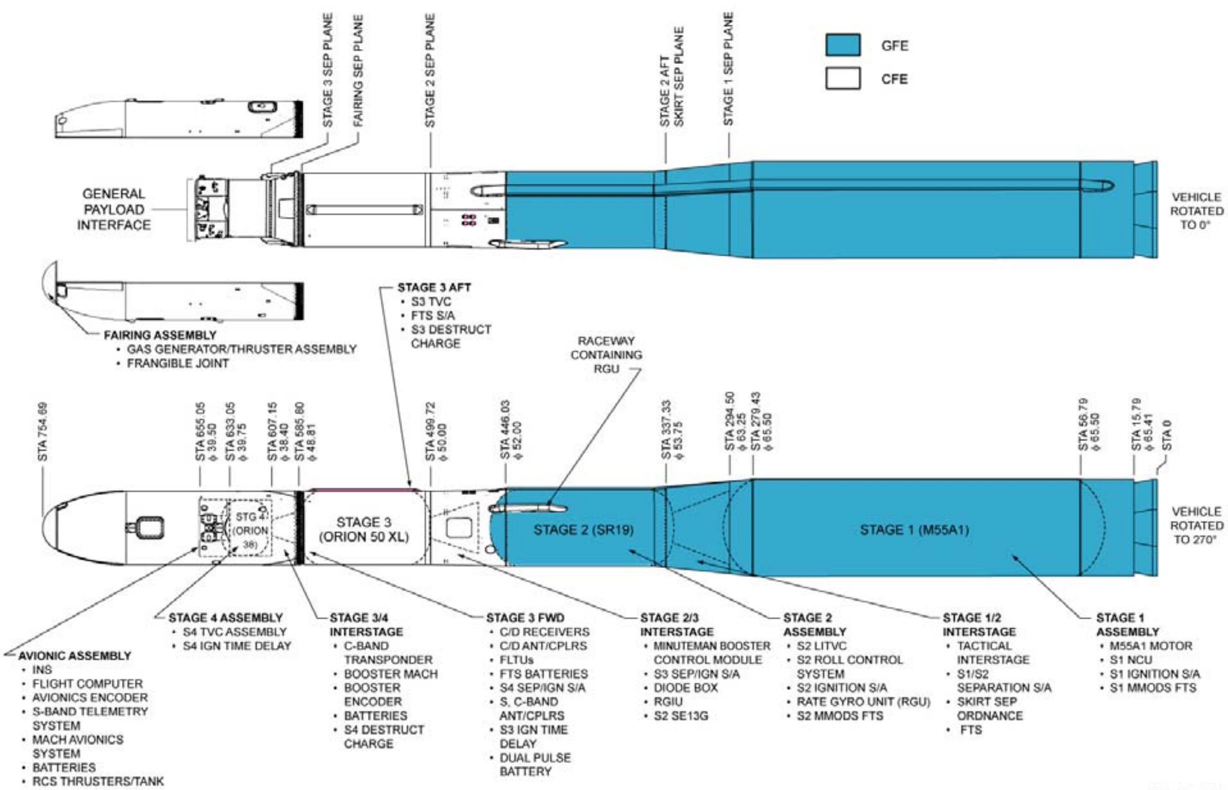


Figure 2-2. OSP Minotaur I Launch Vehicle Configuration

(GFE) Minuteman Stages 1 and 2. Only minor modifications are made to the boosters, including harness interface changes and conversion from All-Ordnance Destruct System (AODS) to Modular Mechanical Ordnance Destruct System (MMODS) Flight Termination System (FTS).

The first stage consists of the Minuteman II M55A1 solid propellant motor, Nozzle Control Units (NCU), Stage 1 Ignition Safe/Arm, S1/S2 Interstage and Stage 1 MMODS FTS. Four gimbaled nozzles provide three axis control during first stage burn. The Second stage consists of a refurbished Minuteman II SR19 motor, Liquid Injection Thrust Vector Control subsystem (LITVC), S2 ignition safe/arm device, a Roll Control System (RCS), and the Stage 2 MMODS FTS components. Attitude control during second stage burn is provided by the operational LITVC and hot gas roll control. A Rate Gyro Unit (RGU) is installed on the outer skin of the SR19 to enhance the vehicle control and increase launch availability.

2.3.2. Upper Stack Assembly

The Minotaur I Upper Stack is composed of Stages 3 and 4 which are the ATK Orion 50XL and 38 SRMs, respectively. These motors were originally developed for Orbital's Pegasus program and are used in a similar manner on the ground-launched Minotaur I vehicle. Common design features, materials and production techniques are applied to both motors to maximize reliability and production efficiency. The motors are fully flight qualified based on their heritage, conservative design, ground static fires and over thirty successful flights. Processing of the Minotaur I Upper Stack is conducted at the same processing facility as Pegasus, directly applying the integration and testing experience of Pegasus to the Minotaur I system (Figure 2-3).

Figure 2-3. Minotaur I Upper Stack Assembly Processing at Orbital's Vehicle Assembly Building at VAFB

Avionics — Minotaur I utilizes the Orbital standard avionics system which has heritage to the Pegasus and Taurus designs. However, the Minotaur I design was upgraded to provide the increased capability and flexibility required by the OSP contract, particularly in the area of payload accommodations. The flight computer, which is common to Pegasus and Taurus, is a 32-bit multiprocessor architecture. It provides communication with vehicle subsystems, the LSE, and the payload, if required, utilizing standard RS-422 serial links and discrete I/O. The Minotaur I design incorporates Orbital's Modular Avionics Control Hardware (MACH) to provide power transfer, data acquisition, Minuteman booster interfaces, and ordnance initiation. MACH has exhibited 100% reliability on OSP SLV and Target Launch Vehicle (TLV) flights and several of Orbital's suborbital launch vehicles. In addition, the Minotaur I telemetry system has been upgraded to provide up to 2 Mbps of real-time vehicle data with dedicated bandwidth and channels reserved for payload use.

Attitude Control Systems — The Minotaur I Attitude Control System (ACS) provides three-axis attitude control throughout boosted flight and coast phases. Stages 1 and 2 utilize the Minuteman Thrust Vector Control (TVC) systems. The Stage 1

TVC is a four-nozzle hydraulic system, while the Stage 2 system combines liquid injection for pitch and yaw control with hot gas roll control. Stages 3 and 4 utilize the same TVC systems as Pegasus and Taurus. They combine single-nozzle electromechanical TVC for pitch and yaw control with a three-axis cold-gas attitude control system resident in the avionics section providing roll control.

Attitude control is achieved using a three-axis autopilot that employs Proportional-Integral-Derivative (PID) control. Stages 1 and 2 fly a pre-programmed attitude profile based on trajectory design and optimization. Stage 3 uses a set of pre-programmed orbital parameters to place the vehicle on a trajectory toward the intended insertion apse. The extended coast between Stages 3 and 4 is used to orient the vehicle to the appropriate attitude for Stage 4 ignition based upon a set of pre-programmed orbital parameters and the measured performance of the first three stages. Stage 4 utilizes energy management to place the vehicle into the proper orbit. After the final boost phase, the three-axis cold-gas attitude control system is used to orient the vehicle for spacecraft separation, contamination and collision avoidance and downrange downlink maneuvers. The autopilot design is a modular object oriented software design, so additional payload requirements such as rate control or celestial pointing can be accommodated with minimal additional development.

Telemetry Subsystem — The Minotaur I telemetry subsystem provides real-time health and status data of the vehicle avionics system, as well as key information regarding the position, performance and environment of the Minotaur I vehicle. This data may be used by Orbital and the range safety personnel to evaluate system performance. The minimum data rate is 750 kbps, but the system is capable of data rates up to 2 Mbps.

Payload Fairings — The baseline Minotaur I fairing is identical to the Pegasus fairing design. However, due to differences in vehicle loads and environments, the Minotaur I implementation allows for a larger payload envelope. The Minotaur I payload fairing consists of two composite shell halves, a nose cap integral to one shell half, and a separation system. Each shell half is composed of a cylinder and ogive sections.

Options for payload access doors and enhanced cleanliness are available. A larger 61 inch diameter (OD) fairing is also available. Further details on the baseline fairing are included in Section 5.1 and for the larger fairing in Section 8.1.

With the addition of a structural adapter, either fairing can accommodate multiple payloads. This feature, described in more detail in Section 9.0 of this User's Guide, permits two or more smaller payloads to share the cost of a Minotaur I launch, resulting in a lower launch cost for each as compared to other launch options. OSP has access to several Multiple Payload Adaptor (MPA) designs that allow for a cost sharing benefit to programs with excess payload and/or mass capability.

2.4. Launch Support Equipment

The Minotaur I LSE is designed to be readily adaptable to varying launch site configurations with minimal unique infrastructure required. The EGSE consists of readily transportable consoles that can be housed in various facility configurations depending on the launch site infrastructure. The EGSE is composed of two primary functional elements: Launch Control and Vehicle Interface (Figure 2-4). The Launch Control consoles are located in a LCR, depending on available launch site accommodations. The Vehicle Interface EGSE is located in structures near the pad, typically called a Launch Equipment Vault (LEV). Fiber optic connections from the Launch Control to the Vehicle Interface consoles

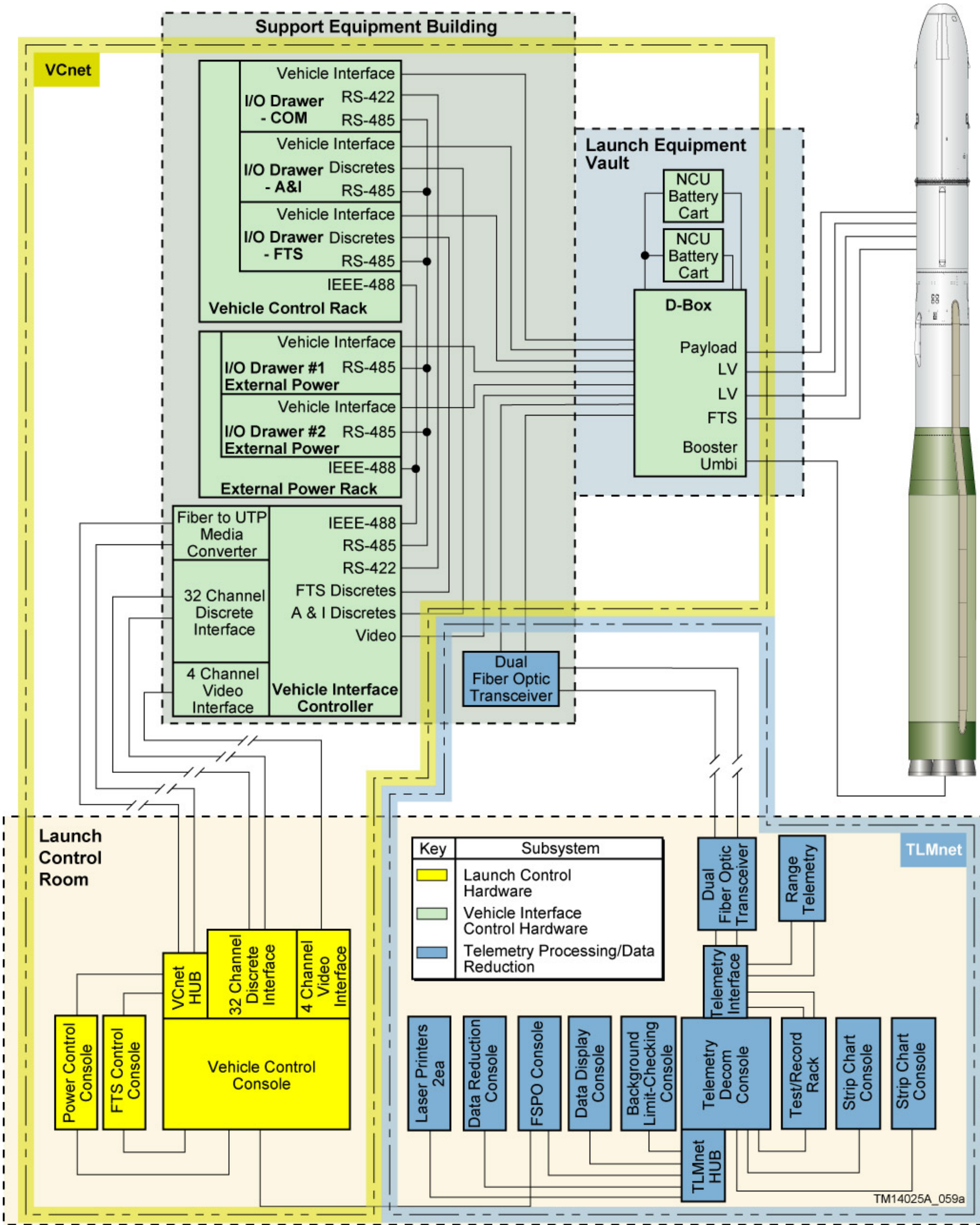


Figure 2-4. Minotaur I EGSE Configuration

are used for efficient, high bandwidth communications and to minimize the amount of cabling required. The Vehicle Interface racks provide the junction from fiber optic cables to the copper cabling interfacing with the vehicle.

The LCR serves as the control center during the launch countdown. The number of personnel that can be accommodated are dependent on the launch site facilities. At a minimum, the LCR will accommodate Orbital personnel controlling the vehicle, two Range Safety representatives (ground and flight safety), and the Air Force Mission Manager. A typical layout is shown in Figure 2-5. Mission-unique, customer-supplied payload consoles and equipment can be supported in the LCR and LEV, within the constraints of the launch site facilities or temporary structures utilized. Interface to the payload through the Minotaur I payload umbilicals and land lines provides the capability for direct monitoring of payload functions. Payload personnel accommodations will be handled on a mission-specific basis.

Figure 2-5. Minotaur I Launch Control Consoles Configuration

3. GENERAL PERFORMANCE

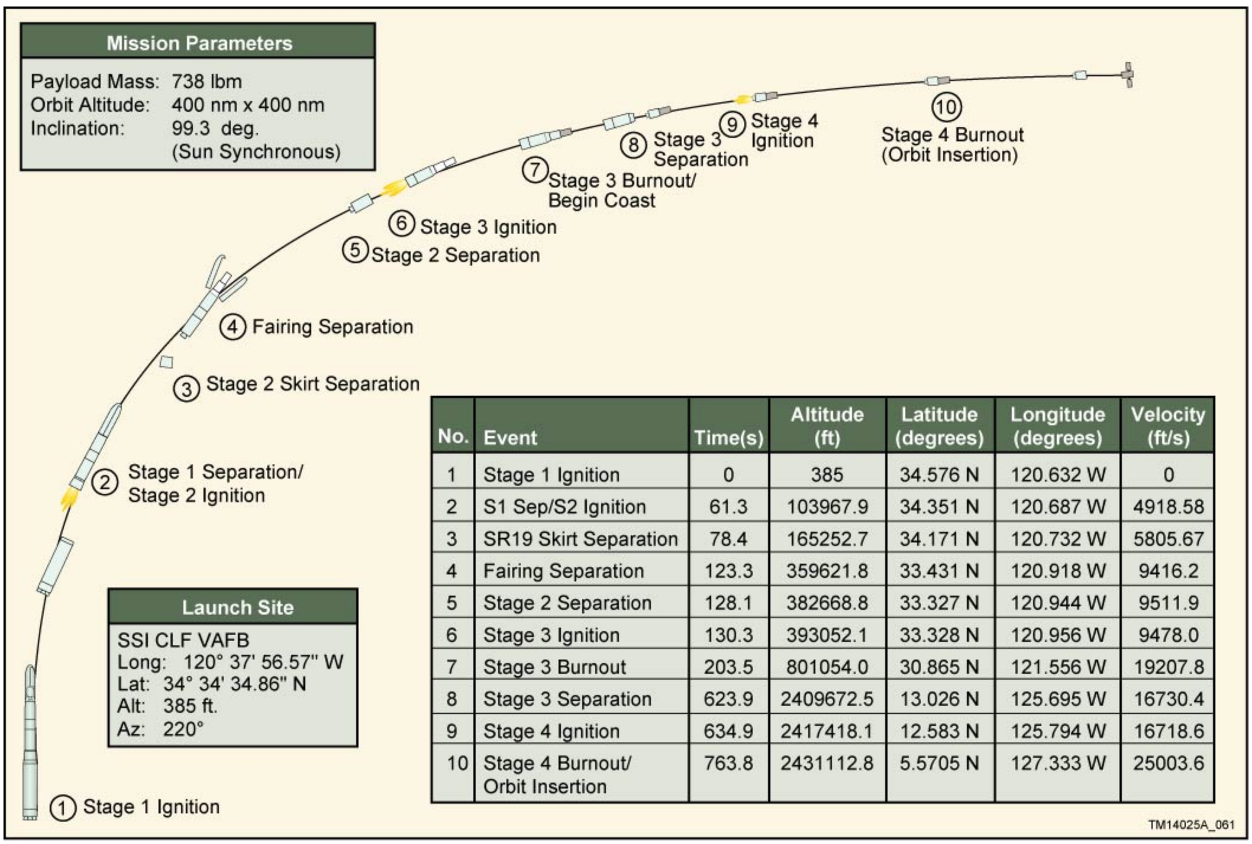
3.1. Mission Profiles

Minotaur I can attain a range of posigrade and retrograde inclinations through the choice of launch sites made available by the readily adaptable nature of the Minotaur I launch system. A typical mission profile to a sun-synchronous orbit is shown in Figure 3-1. High energy and Geosynchronous Transfer Orbit (GTO) missions can also be achieved. All performance parameters presented herein are typical for most expected payloads. However, performance may vary depending on unique payload or mission characteristics. Specific requirements for a particular mission must be coordinated with OSP. Once a mission is formally initiated, the requirements will be documented in the Mission Requirements Document (MRD). The MRD is the requirement kick off document that initiates the contractual agreement and flows the payload requirements to Orbital. The MRD establishes the data required to begin formal trajectory analysis as well as CLAs. Further detail will be captured in the Payload-to-Launch Vehicle Interface Control Drawing (ICD).

3.2. Launch Sites

Depending on the specific mission and range safety requirements, Minotaur I can operate from several East and West launch sites, illustrated in Figure 3-2. Specific performance parameters are presented in Section 4.

Standard Minotaur I launches are conducted from facilities at VAFB, CA. This facility is capable of accommodating inclinations between 60° and 120°, although inclinations below 72° would require an out-of-plane dogleg, thereby reducing payload capability. Initial Minotaur I missions were launched from Space Launch Complex-8 (SLC-8) - the California Spaceport facility operated



No.	Event	Time(s)	Altitude (ft)	Latitude (degrees)	Longitude (degrees)	Velocity (ft/s)
1	Stage 1 Ignition	0	385	34.576 N	120.632 W	0
2	S1 Sep/S2 Ignition	61.3	103967.9	34.351 N	120.687 W	4918.58
3	SR19 Skirt Separation	78.4	165252.7	34.171 N	120.732 W	5805.67
4	Fairing Separation	123.3	359621.8	33.431 N	120.918 W	9416.2
5	Stage 2 Separation	128.1	382668.8	33.327 N	120.944 W	9511.9
6	Stage 3 Ignition	130.3	393052.1	33.328 N	120.956 W	9478.0
7	Stage 3 Burnout	203.5	801054.0	30.865 N	121.556 W	19207.8
8	Stage 3 Separation	623.9	2409672.5	13.026 N	125.695 W	16730.4
9	Stage 4 Ignition	634.9	2417418.1	12.583 N	125.794 W	16718.6
10	Stage 4 Burnout/ Orbit Insertion	763.8	2431112.8	5.5705 N	127.333 W	25003.6

Figure 3-1. Minotaur I Typical Mission Profile

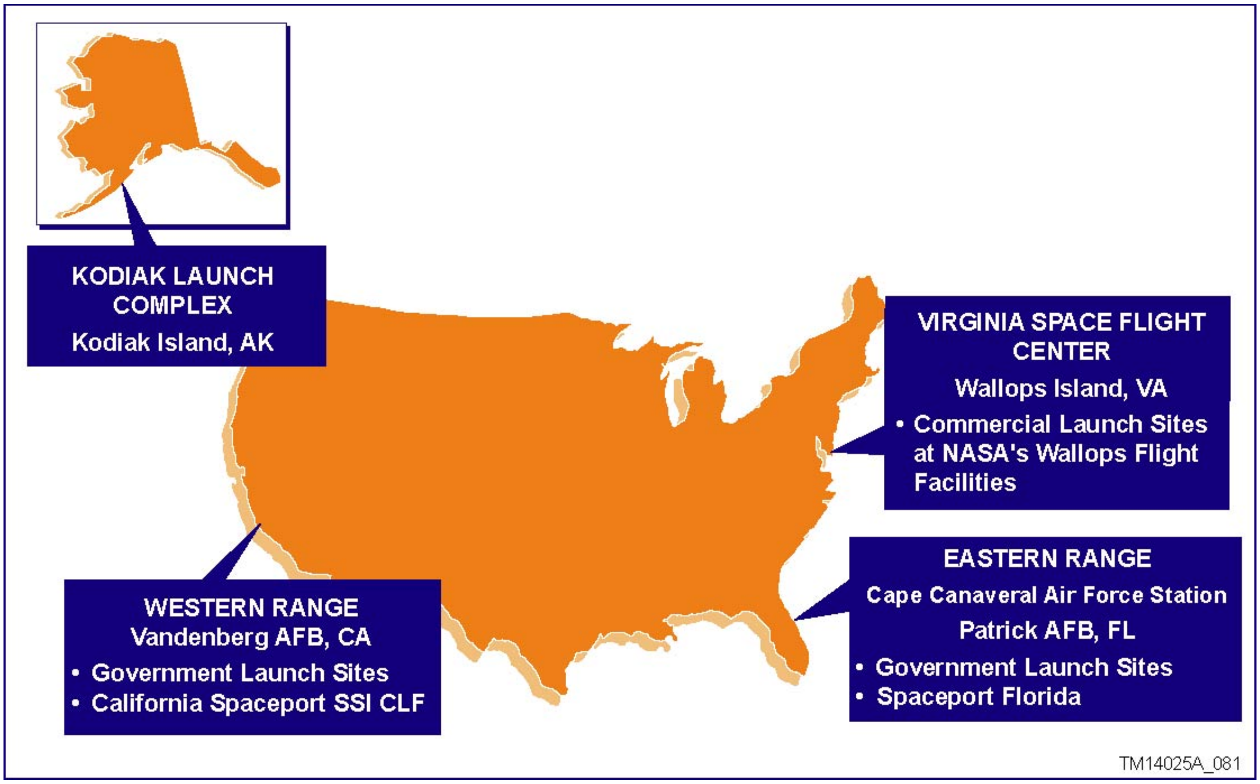


Figure 3-2. Minotaur I Launch Site Options

by Spaceport Systems International (SSI), on South VAFB.

As a non-standard service Minotaur I can operate from several other East and West launch sites including Kodiak Island, AK, Cape Canaveral, FL, and Wallops Island, VA. Additional details on conducting launches from these locations are provided in Section 8.5.

3.3. Performance Capability

Minotaur I performance curves for circular and elliptical orbits of various altitudes and inclinations are detailed in Figure 3-3 through Figure 3-10 for launches from all four Spaceports.

These performance curves provide the total mass above the standard, non-separating interface. The mass of any Payload Attach Fitting (PAF), isolation system, or separation system is to be accounted for in the payload mass allocation. Figure 3-11 illustrates the stage vacuum impact points for a typical sun-synchronous trajectory from VAFB.

3.4. Injection Accuracy

Minotaur I injection accuracy is summarized in Figure 3-12. Better accuracy can be provided dependent on specific mission characteristics. For example, heavier payloads will typically have better insertion accuracy, as will higher orbits. An enhanced option for increased insertion accuracy is also available (Section 8.2.1). It utilizes the flight-proven Hydrazine Auxiliary Propulsion System (HAPS) developed on the Pegasus program.

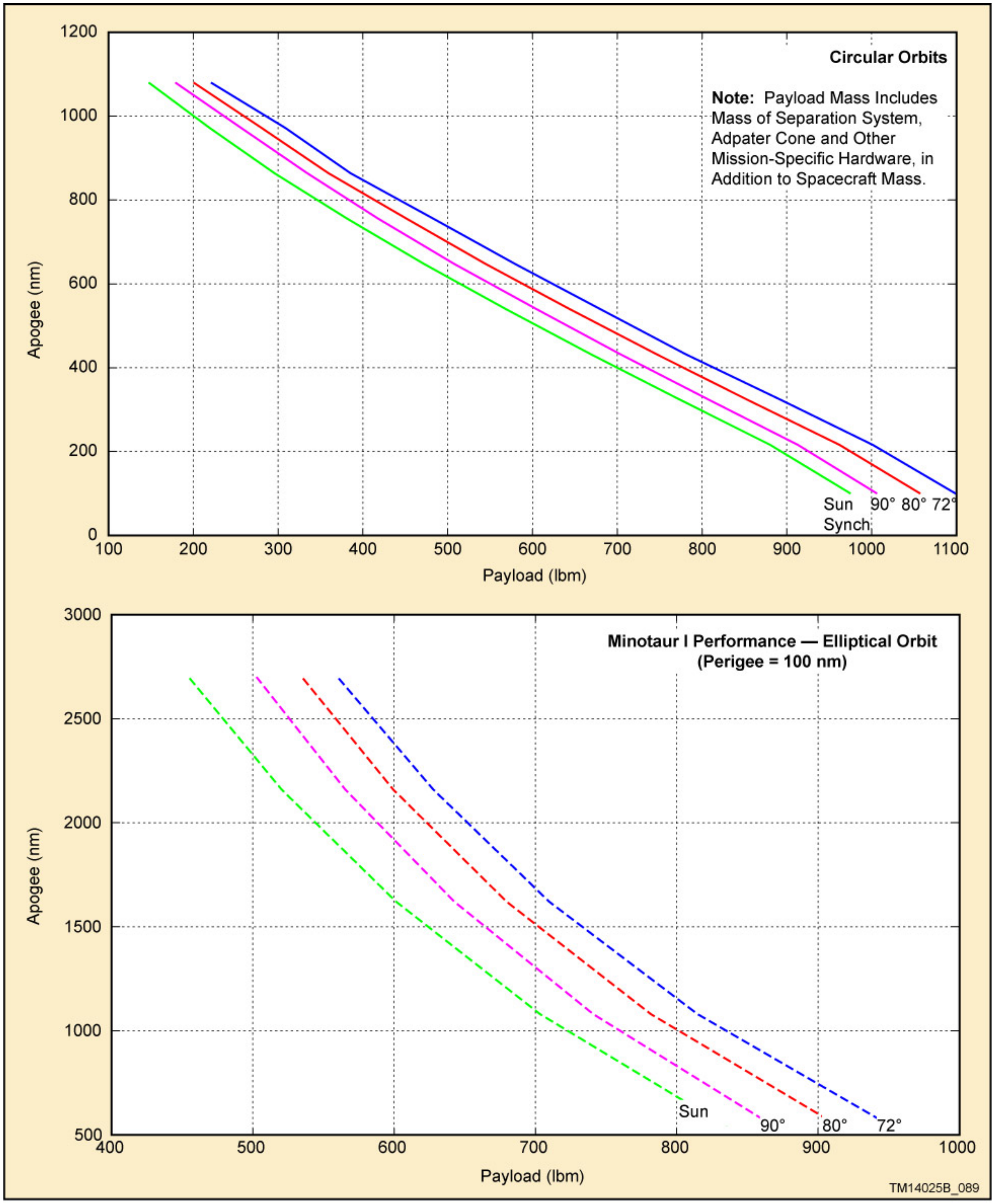


Figure 3-3. Minotaur I Performance (English Units) – California Spaceport

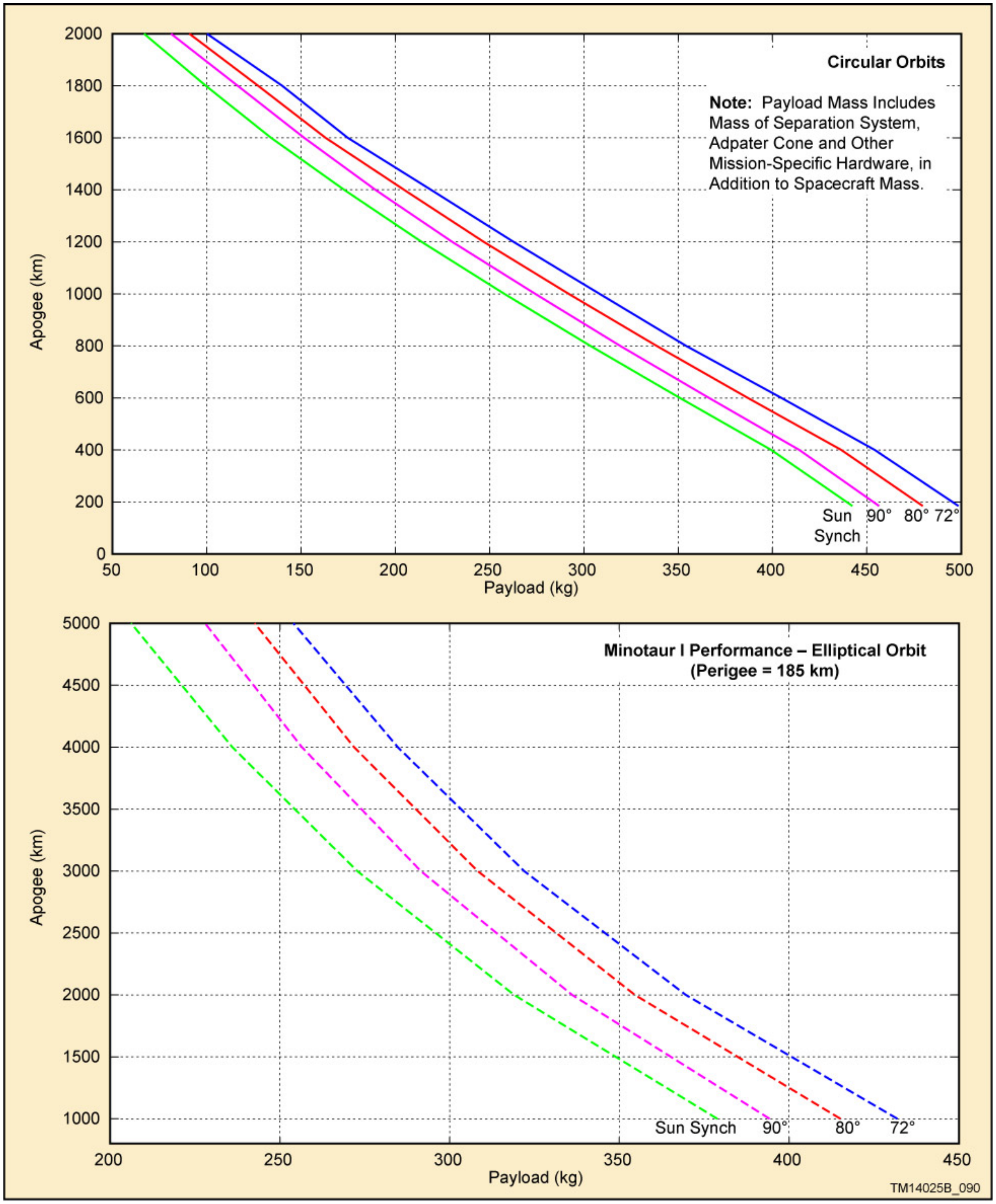


Figure 3-4. Minotaur I Performance (Metric Units) – California Spaceport

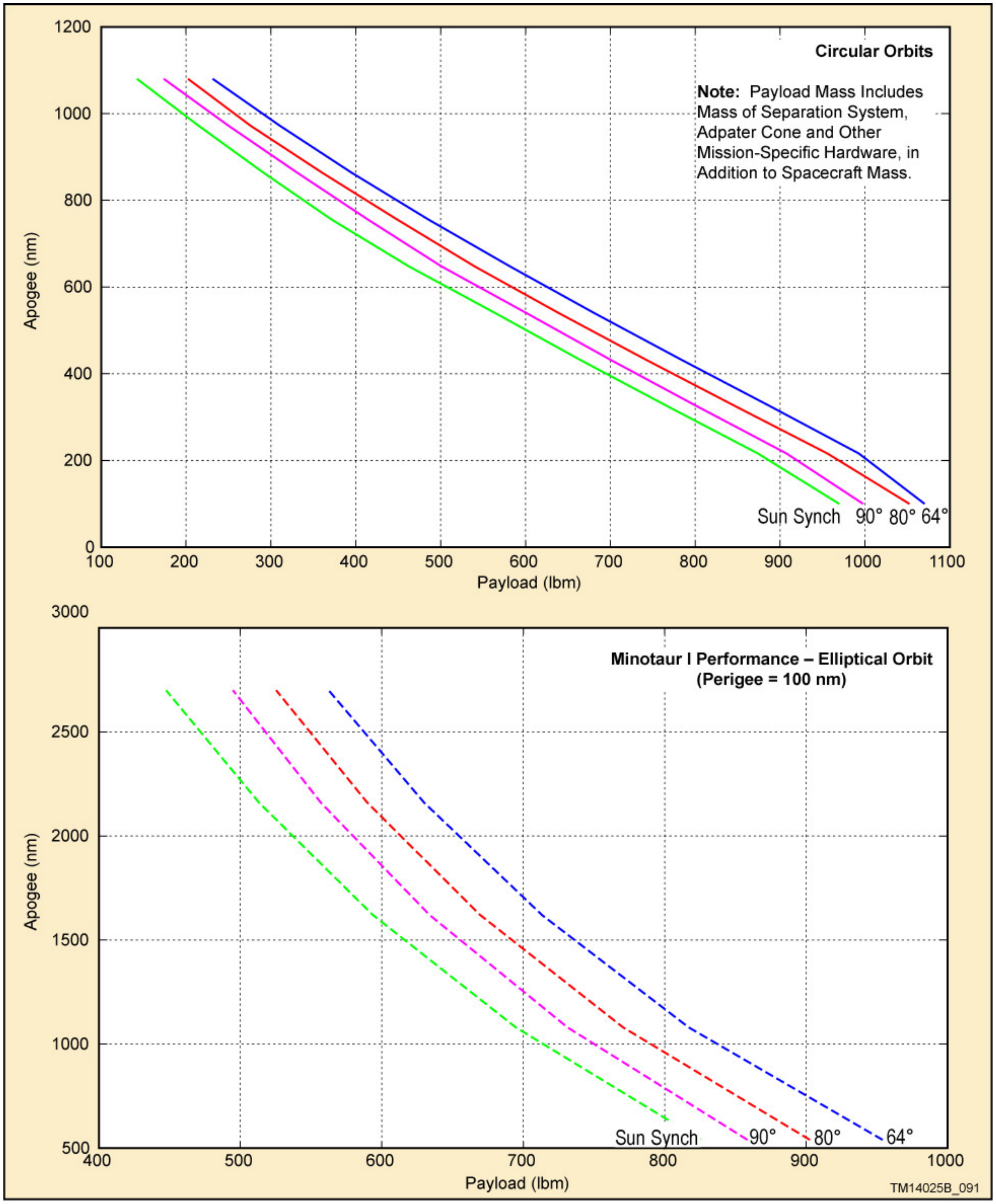


Figure 3-5. Minotaur I Performance (English Units) – Kodiak Launch Complex

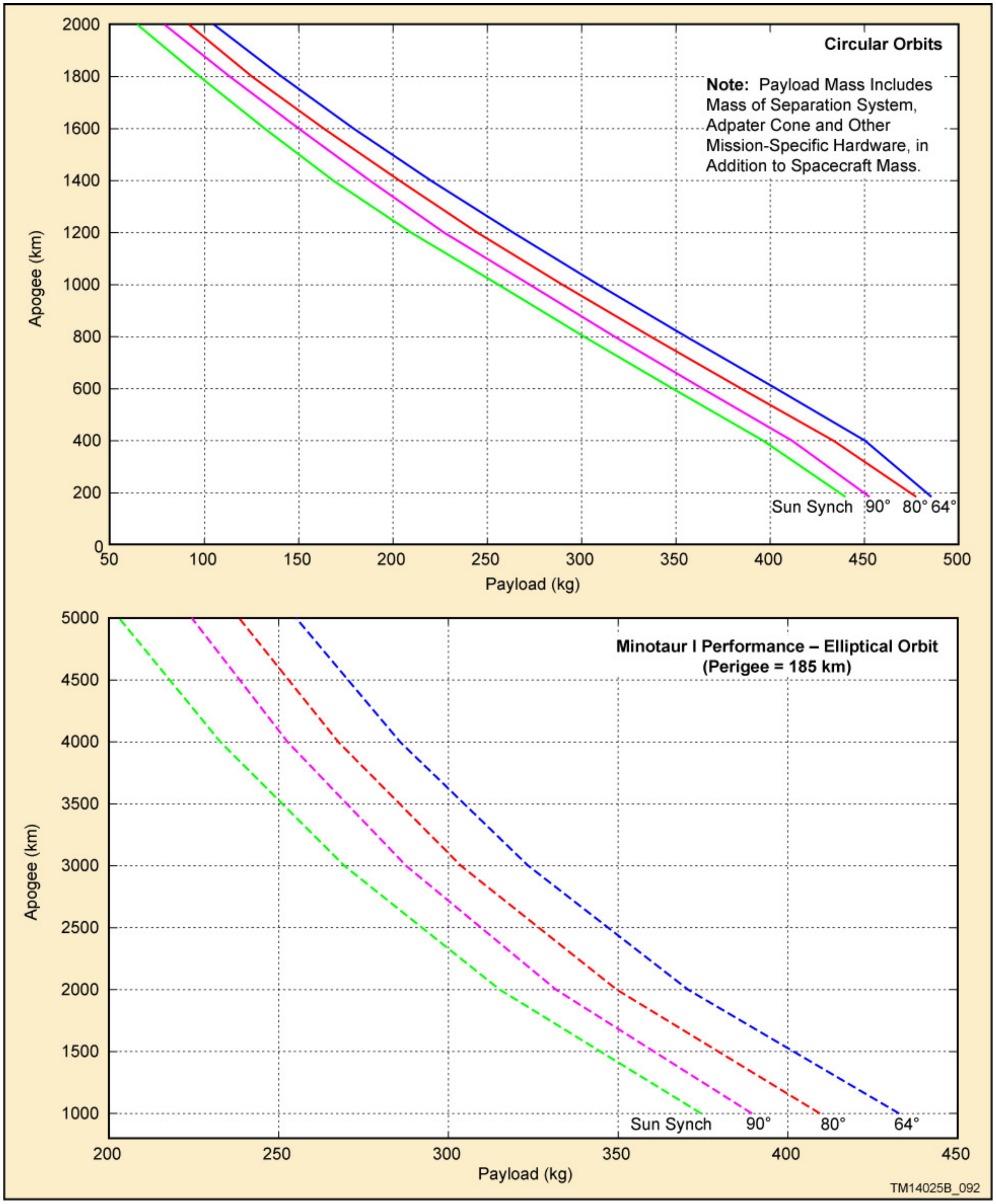


Figure 3-6. Minotaur I Performance (Metric Units) – Kodiak Launch Complex

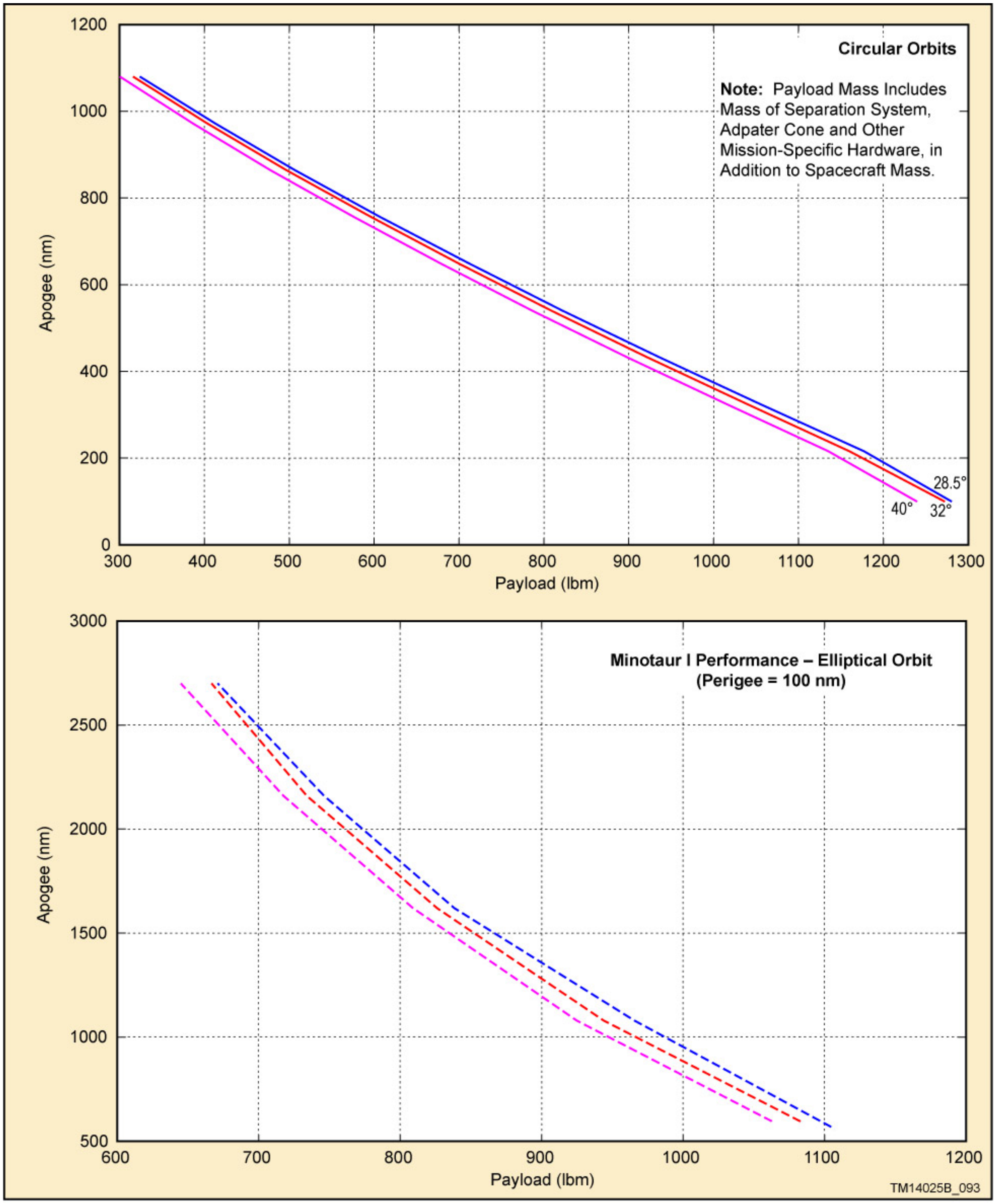


Figure 3-7. Minotaur I Performance (English Units) – Spaceport Florida

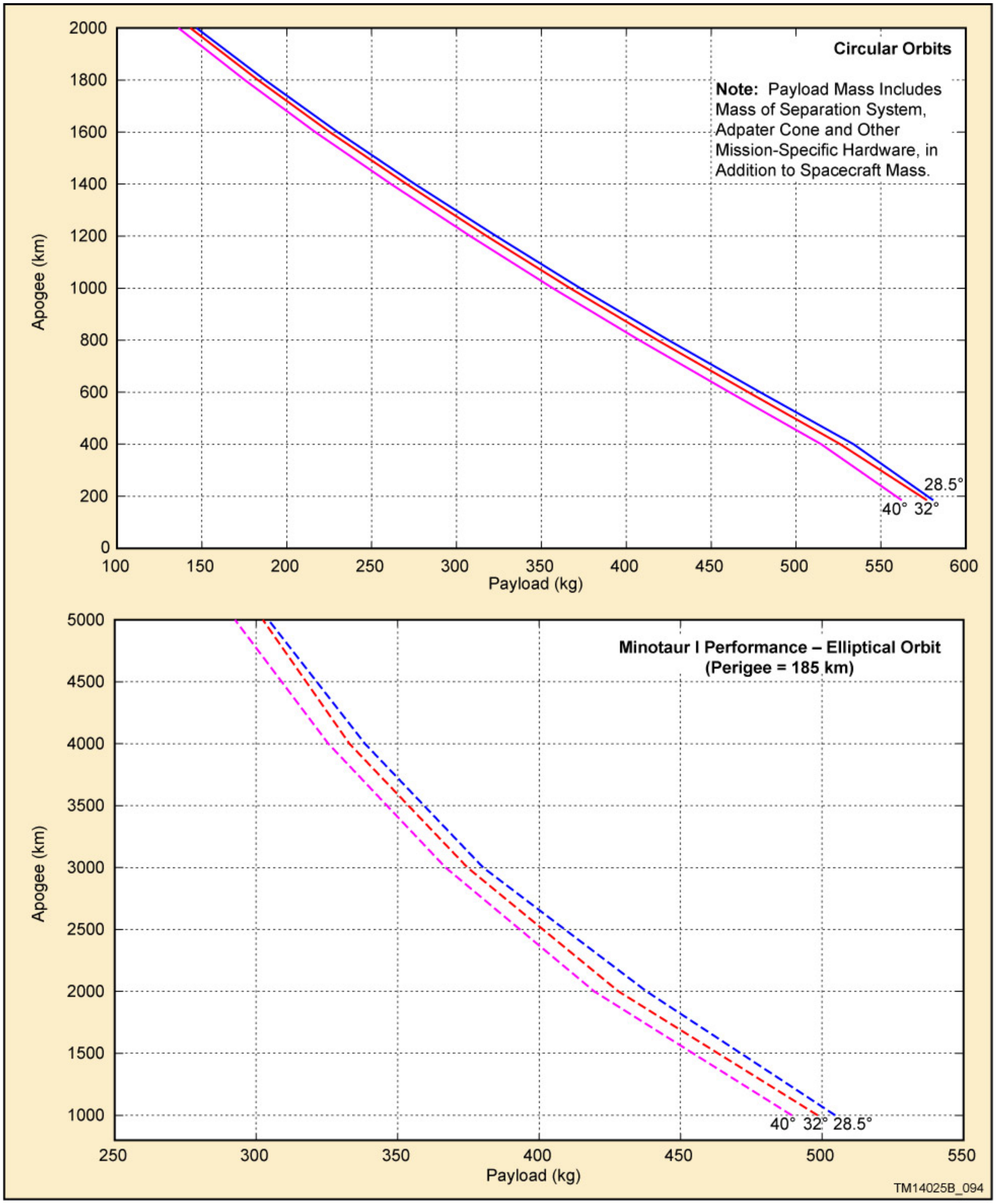


Figure 3-8. Minotaur I Performance (Metric Units) – Spaceport Florida

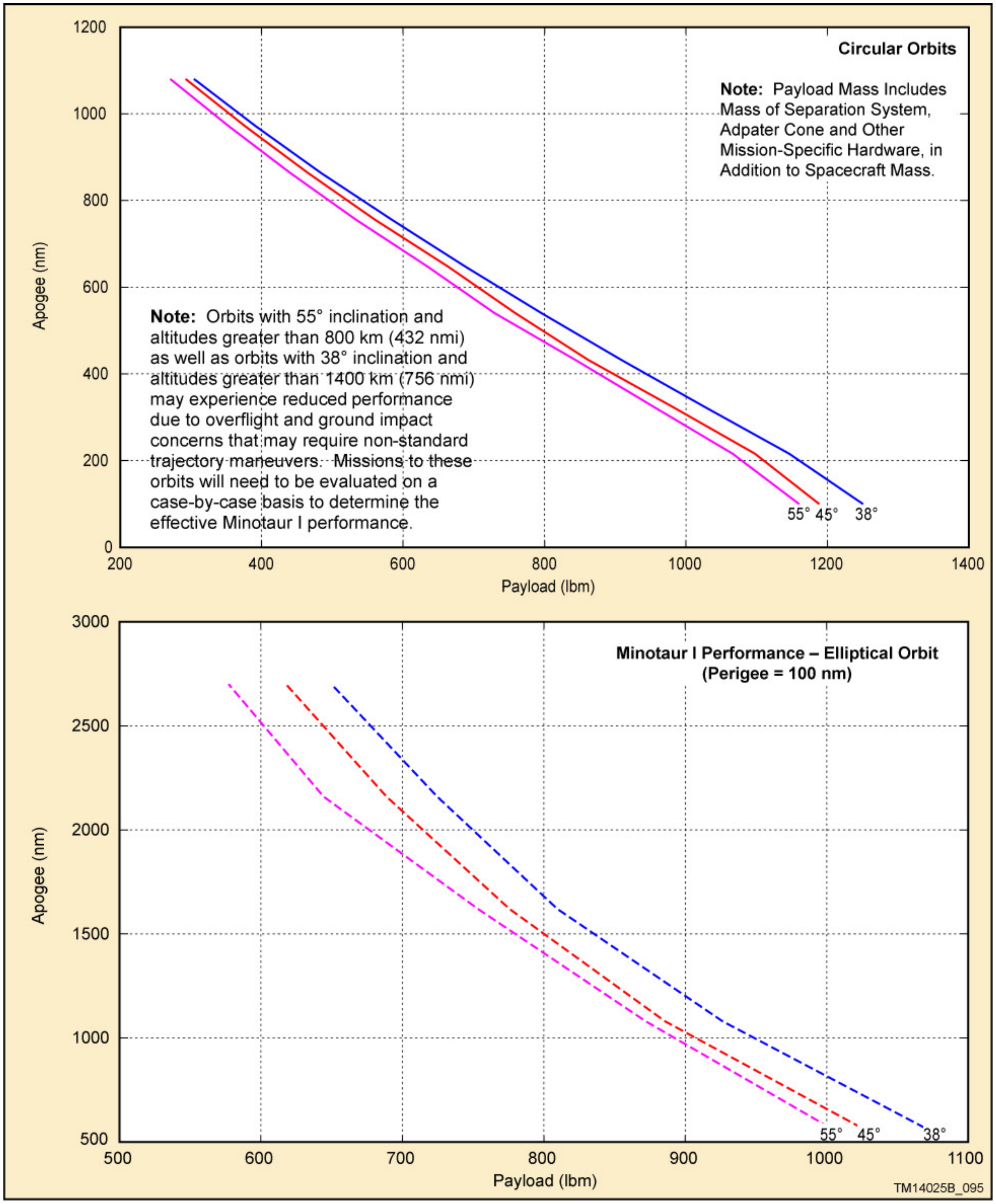


Figure 3-9. Minotaur I Performance (English Units) – Virginia Spaceflight Center

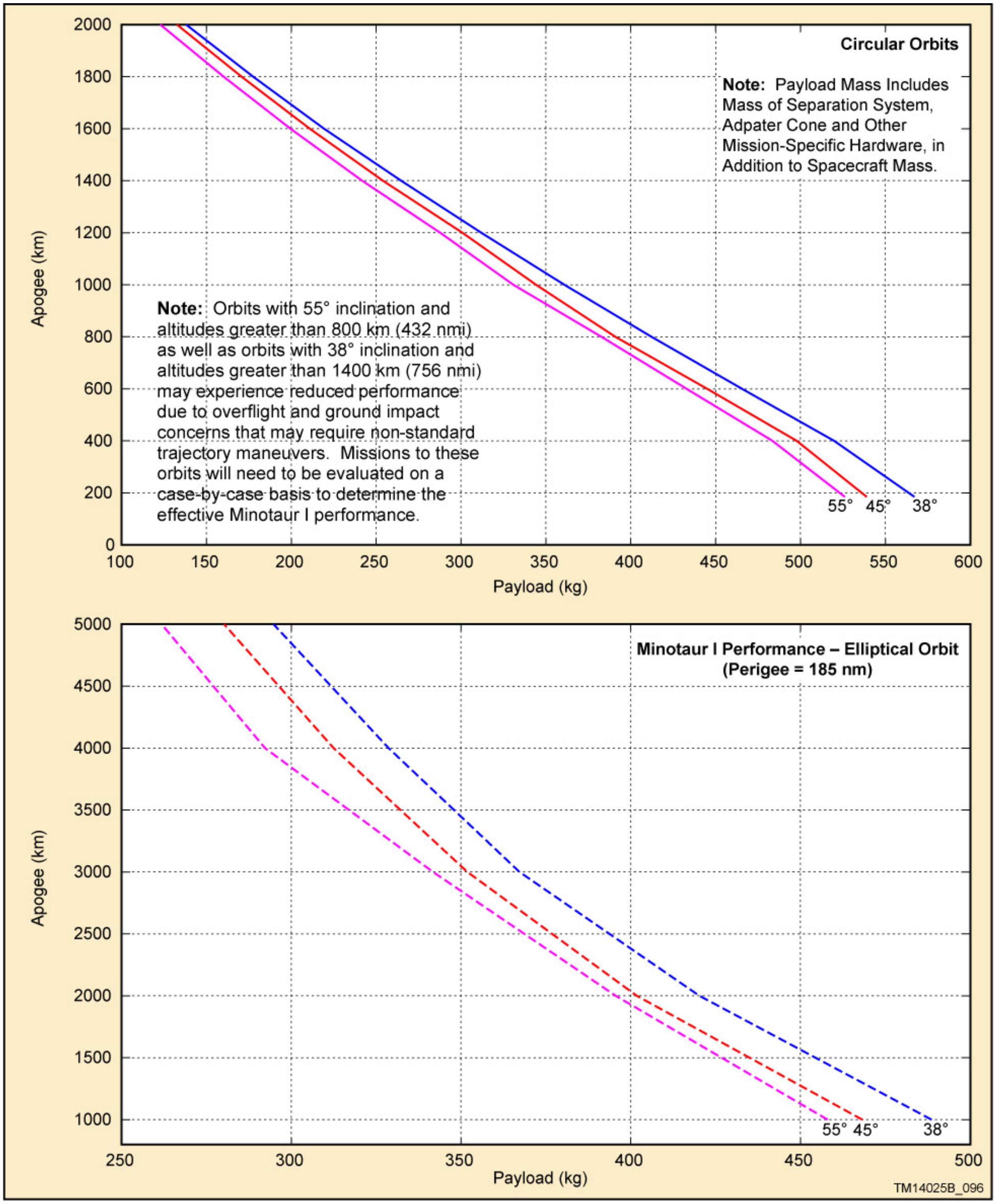


Figure 3-10. Minotaur I Performance (Metric Units) – Virginia Spaceflight Center

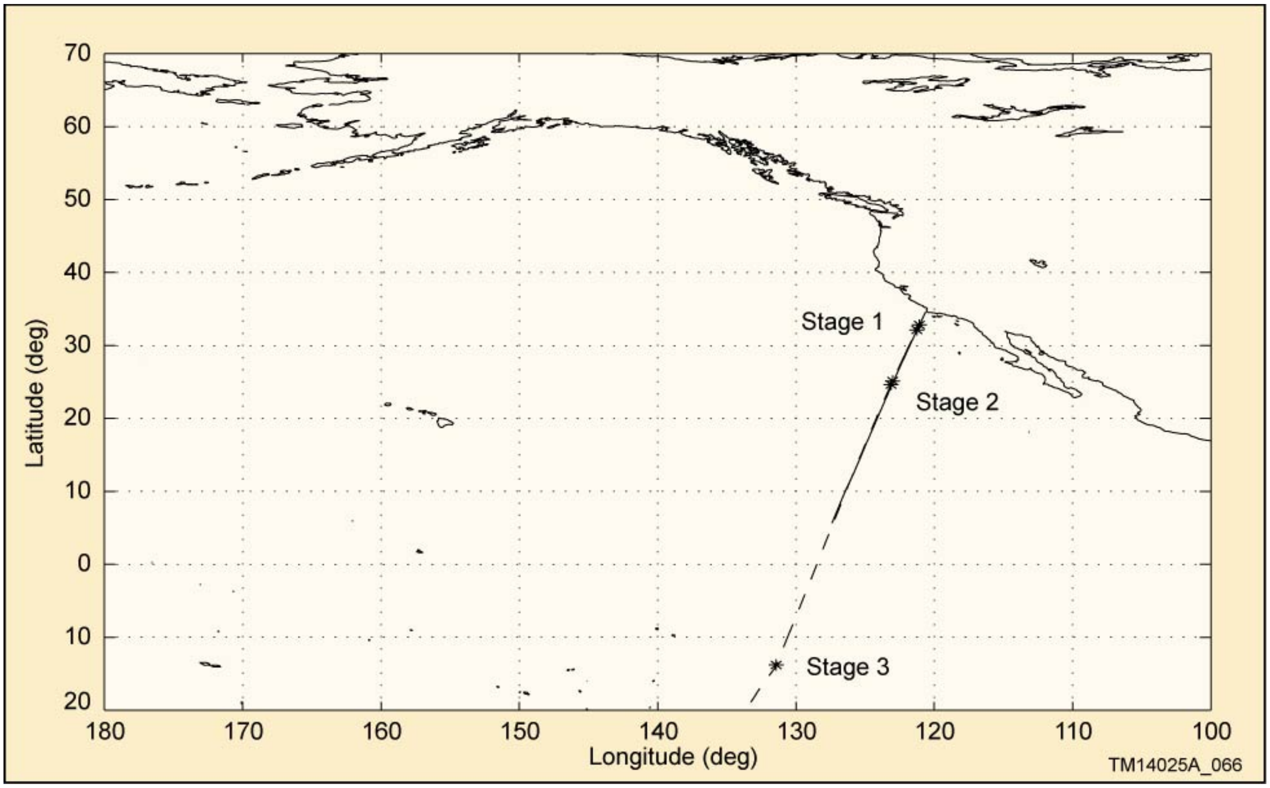


Figure 3-11. Stage Impact Points for Typical Sun-Synchronous Launch From VAFB

Error Type	Tolerance (Worst Case)
Altitude (Insertion Apse)	±10 nmi (18.5 km)
Altitude (Non-Insertion Apse)	±50 nmi (92.6 km)
Inclination	±0.2°

Figure 3-12. Injection Accuracies to Low Earth Orbits

Error Type		Angle	Rate
3-Axis	Yaw	±1.0°	±0.5°/sec
	Pitch	±1.0°	±0.5°/sec
	Roll	±1.0°	±0.5°/sec
Spinning	Spin Axis	±1.0°	≤10 rpm
	Spin Rate		±3°/sec

Figure 3-13. Typical Pre-Separation Payload Pointing and Spin Rate Accuracies

3.5. Payload Deployment

Following orbit insertion, the Minotaur I Stage 4 avionics subsystem can execute a series of ACS maneuvers to provide the desired initial payload attitude prior to separation. This capability may also be used to incrementally reorient Stage 4 for the deployment of multiple spacecraft with independent attitude requirements. Either an inertially-fixed or spin-stabilized attitude may be specified by the customer.

The maximum spin rate for a specific mission depends upon the spin axis moment of inertia of the payload and the amount of ACS propellant needed for other attitude maneuvers. Figure 3-13 provides the typical payload pointing and spin rate accuracies.

3.6. Payload Separation

Payload separation dynamics are highly dependent on the mass properties of the payload and the particular separation system utilized. The payload tip-off and overall separation velocity will be provided in a mission specific analysis based on the properties of the separation system. An OSP provided separation system is available as an option for the Minotaur I vehicle. This separation system is discussed in detail Section 8.1.1.

3.7. Collision/Contamination Avoidance Maneuver

Following orbit insertion and payload separation, the Minotaur I Stage 4 will perform a Collision/Contamination Avoidance Maneuver (C/CAM). The C/CAM minimizes both payload contamination and the potential for recontact between Minotaur I hardware and the separated payload. OSP will perform a recontact analysis for post separation events.

A typical C/CAM begins soon after payload separation. The launch vehicle performs a 90° yaw maneuver designed to direct any remaining Stage 4 motor impulse in a direction which will increase the separation distance between the two bodies. After a delay to allow the distance between the spacecraft and Stage 4 to increase to a safe level, the launch vehicle begins a "crab-walk" maneuver to impart a small amount of delta velocity, increasing the separation between the payload and the fourth stage of the Minotaur I.

Following the completion of the C/CAM maneuver as described above and any remaining maneuvers, such as downlinking of delayed telemetry data, the ACS valves are opened and the remaining ACS nitrogen propellant is expelled.

4. PAYLOAD ENVIRONMENT

CAUTION
Environments presented here bound a generic mission and should not be used in mission specific analyses. Mission specific levels are provided as a standard service and documented or referenced in the mission ICD.

This section provides details of the predicted environmental conditions that the payload will experience during Minotaur I ground operations, powered flight, and launch system post-boost operations.

Minotaur I ground operations include payload integration and encapsulation within the fairing, subsequent transportation to the launch site and final vehicle integration activities. Powered flight begins at Stage 1 ignition and ends at Stage 4 burnout. Minotaur I post-boost operations begin after Stage 4 burnout and end following payload separation. To more accurately define simultaneous loading and environmental conditions, the powered flight portion of the mission is further subdivided into smaller time segments bounded by critical flight events such as motor ignition, stage separation, and transonic crossover.

The environmental design and test criteria presented have been derived using measured data obtained from previous Pegasus, Taurus and Minotaur I missions, motor static fire tests, other system development tests and analyses. These criteria are applicable to Minotaur I configurations using the standard 50 in. diameter fairing. The predicted levels presented are intended to be representative of a standard mission. Satellite mass, geometry and structural components vary greatly and will result in significant differences from mission to mission.

Dynamic loading events that occur throughout various portions of the flight include steady state acceleration, transient low frequency acceleration, acoustic impingement, random vibration, and pyroshock events. Figure 4-1 identifies the time phasing of these dynamic loading events and environments and their significance. Pyroshock events are not indicated in this figure, as they do not occur simultaneous with any other significant dynamic loading events. In addition, dynamic loading associated with S4 ignition are insignificant.

4.1. Steady State and Transient Acceleration Loads

Design limit load factors due to steady state acceleration are presented in Table 4-1. Transient lateral accelerations at Stage 2 ignition are defined in Figure 4-2. Note that the levels shown in Figure 4-2 do not include any load reduction due to an SRSS isolation system (see Section 4.1.1). Transient accelerations were obtained by performing 10,000 case Monte Carlo runs of the coupled loads analysis and represent the 97.72% with 50% confidence of the load limits defined by the maximum predicted acceleration. This

Item	*Liftoff*	*Transonic*	*Supersonic/ Max Q*	*Balance of S1 Burn*	*S2 Ignition*	*S2 Burn*	*S3 Ignition*	*S3 Burn*	*S4 Burn*
Typical Flight Duration	3 sec	17 sec	30 sec	10 sec	N/A	70 sec	N/A	70 sec	70 sec
Steady State Loads	Yes	Yes	Yes	Yes	No	Yes	No	Yes	Yes
Transient Loads	Yes	Yes	Yes	No	Yes	No	Yes	No	No
Acoustics	Yes	Yes	Yes	Yes	No	No	No	No	No
Random Vibration	Yes	Yes	Yes	Yes	No	Yes	No	Yes	Yes

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Figure 4-1. Phasing of Dynamic Loading Events

TABLE 4-1. PAYLOAD CG NET STEADY STATE ACCELERATION FOR A NOMINAL 800 LB STIFF PAYLOAD

Event	Axial (G's)	Lateral (G's)
Liftoff	0 ±4.6	0 ±1.6
Transonic	3.2 ±0.5	0.2 ±1.2
Supersonic	3.8 ±0.5	0.4 ±1.1
S2 Ignition	0 ±6.6	See Fig 4-2
S3 Ignition	0 ±6.1	See Fig 4-2
S3 Burnout	See Fig 4-3	0.3 ±0.0
S4 Burnout	See Fig 4-3	0.3 ±0.0

approach is based on the NASA General Environmental Verification Specification (GEVS), Rev A, Jun 96.

During powered flight, the maximum steady state accelerations are dependent on the payload mass. The maximum level can potentially occur in either Stage 3 or 4 burn. Figure 4-3 depicts the axial acceleration at burnout for each stage as a function of payload mass.

During upper stage burnout, prior to staging, the transient loads are relatively benign. There are significant transient loads that occur at both Stage 2 and Stage 3 ignition. During the transient portion of these ignition events, the steady state axial loads are relatively nonexistent.

As dynamic response is largely governed by payload characteristics, a mission specific Coupled Loads Analysis (CLA) will be performed, with customer provided finite element models of the payload, in order to provide more precise load predictions. Results will be referenced in the mission specific ICD. It is standard for two CLAs to be performed for each mission. A preliminary CLA is based on the analytical payload model. A final CLA is based on the updated test verified payload model. For preliminary design purposes, Orbital can provide initial Center-of-Gravity (CG) netloads given a payload's mass properties, CG location and bending frequencies.

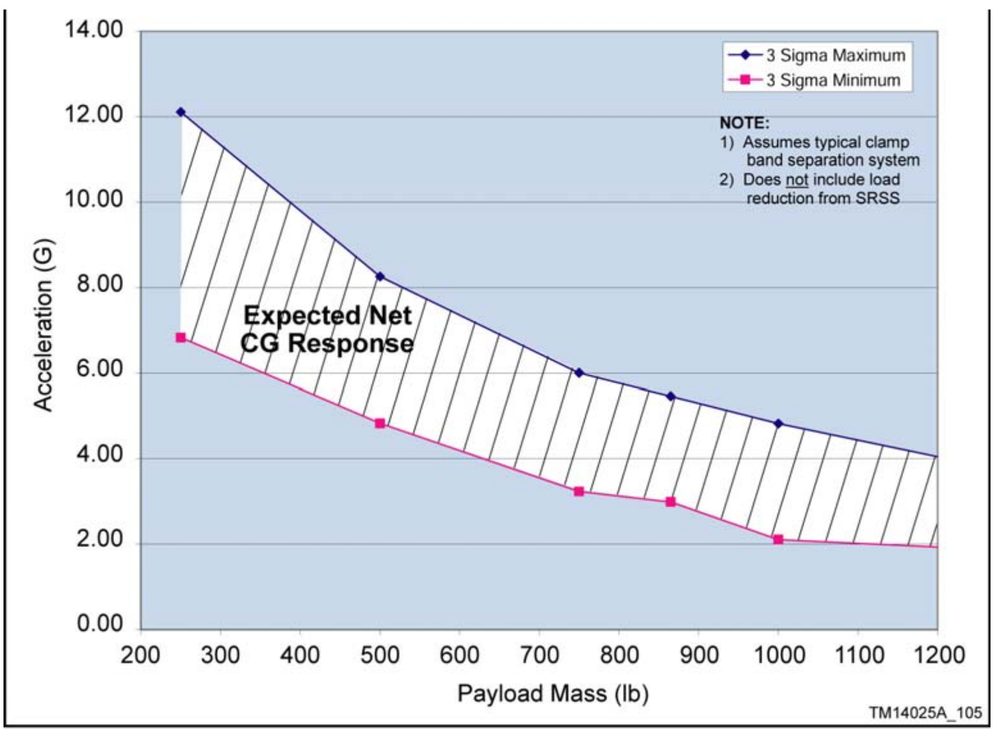


Figure 4-2. Payload CG Net Transient Lateral Acceleration Envelope at Stage 2 Ignition

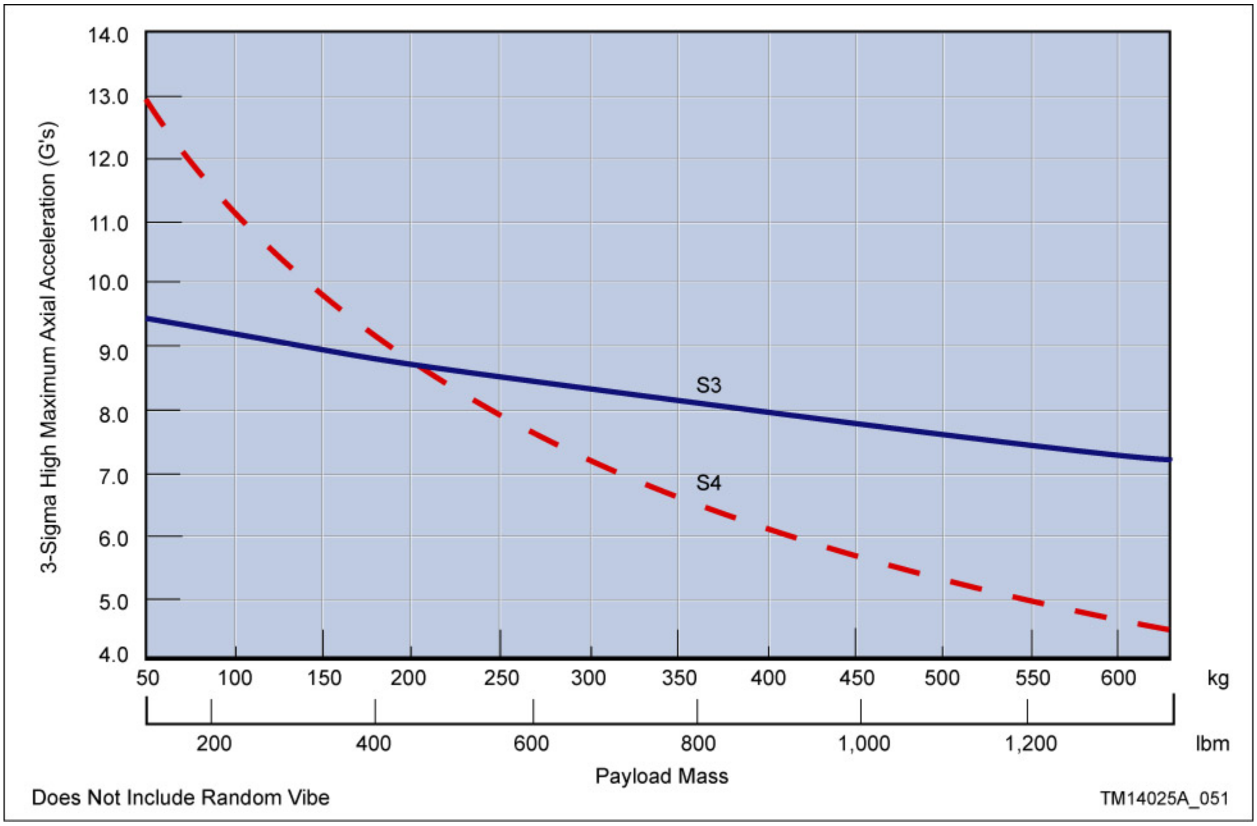


Figure 4-3. Minotaur I 3-Sigma High Maximum Acceleration as a Function of Payload Weight

4.1.1. Standard Payload Isolation System

OSP utilizes a flight-proven payload load isolation system as a standard service. This Soft Ride for Small Satellites (SRSS) was developed by Air Force Research Laboratory (AFRL) and CSA Engineering. It has been successfully demonstrated on previous Minotaur I missions in the typical configuration shown in Figure 2-2. This mechanical isolation system has demonstrated the capability to significantly alleviate the transient dynamic loads that occur during flight. The isolation system can provide relief to both the overall payload center of gravity loads and component or subsystem responses. Typically the system will reduce transient loads to approximately 50% of the level they would be without the system. In addition, the system generally reduces shock and vibration levels transmitted between the vehicle and spacecraft. The exact results can be expected to vary for each particular spacecraft and with location on the spacecraft. The isolation system does impact overall vehicle performance (by approximately 20-50 lb [9-22.3 kg]) and the available payload dynamic envelope (by up to 3.0 in (10.16 cm) axially and approximately 1.0 in (2.54 cm) laterally).

4.2. Payload Vibration Environment

The in-flight random vibration curve shown in Figure 4-4 encompasses all flight vibration environments.

The maximum shock response spectrum at the base of the payload from all launch vehicle events will not exceed the flight limit levels in Figure 4-5 (separating shock). For missions that do not utilize an Orbital supplied payload separation system, the shock response spectrum at the base of the payload from vehicle events will

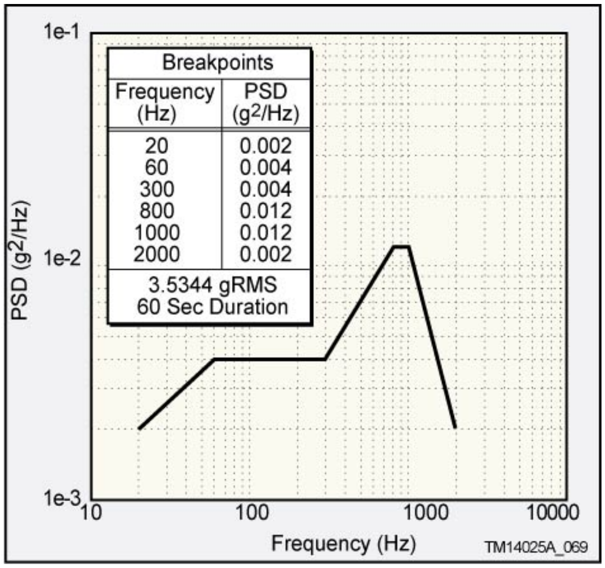


Figure 4-4. Payload Random Vibration Environment During Flight

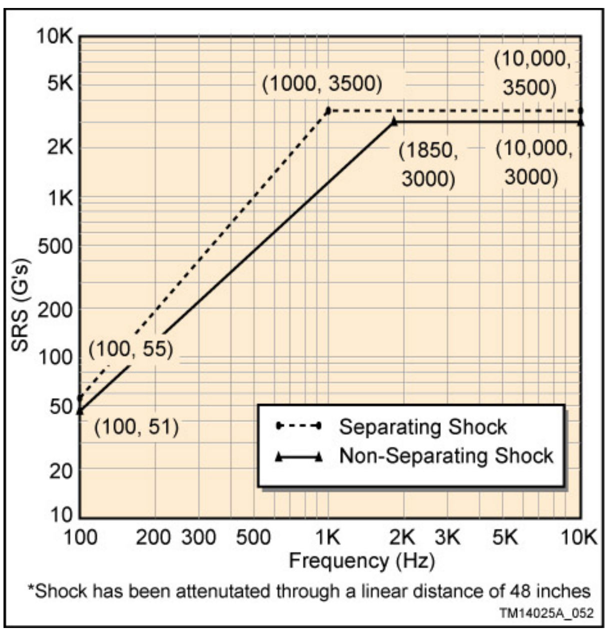


Figure 4-5. Maximum Shock Environment – Launch Vehicle to Payload

not exceed the levels in Figure 4-5 (non-separating shock).

4.3. Payload Shock Environment

If a payload-provided separation system is used, then the shock delivered to the Stage 4 vehicle interface must not exceed the limit level characterized in Figure 4-6. Shock above this

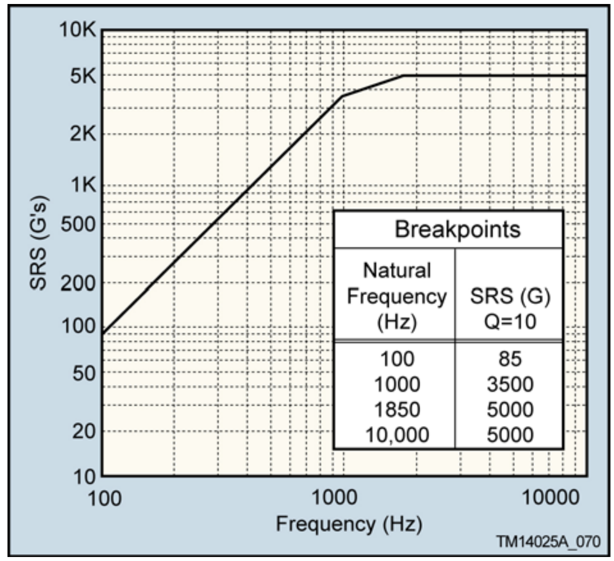


Figure 4-6. Maximum Shock Environment – Payload to Launch Vehicle

level could require requalification of components or an acceptance of risk by the launch mission customer.

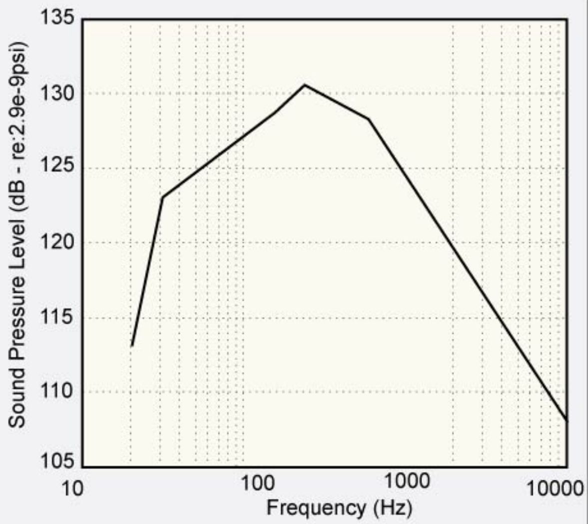
4.4. Payload Acoustic Environment

The acoustic levels during lift-off and powered flight will not exceed the flight limit levels shown in Figure 4-7. If the vehicle is launched over a flame duct, the acoustic levels can be expected to be lower than shown. This has been demonstrated with flight data.

4.5. Payload Structural Integrity and Environments Verification

The primary support structure for the spacecraft must possess sufficient strength, rigidity, and other characteristics required to survive the critical loading conditions that exist within the envelope of handling and mission requirements, including worst-case predicted ground, flight, and orbital loads. It must survive those conditions in a manner that assures safety and that does not reduce the mission success probability.

For each mission, a CLA will be performed, with customer provided finite element models of the payload, in order to provide precise load



Breakpoints		Breakpoints Cont'd	
Natural Frequency (Hz)	SPL (dB-re:2.9e-9psi)	Natural Frequency (Hz)	SPL (dB-re:2.9e-9psi)
20	113	500	128.6
25	118	630	128
32	123	800	126.3
40	123.8	1000	124.7
50	124.7	1250	123
63	125.5	1600	121.3
80	126.3	2000	119.7
100	127.2	2500	118
125	128	3150	116.3
160	128.8	4000	114.7
200	129.7	5000	113
250	130.5	6300	111.3
315	129.9	8000	109.7
400	129.3	10,000	108

140 dB OASPL TM14025A_071

Figure 4-7. Payload Acoustic Environment During Liftoff and Flight

predictions. Flight events that will be analyzed by the CLA include liftoff, the transonic portion of flight, supersonic flight, maximum dynamic pressure (max q), Stage 2 ignition and Stage 3 ignition. The worst-case lateral accelerations occur during Stage 2 ignition. Therefore, for the Stage 2 ignition event a 10,000 case Monte Carlo statistical analysis is performed. Responses for 10,000 randomly distributed sets of Stage 2 ignition forces are calculated and the results of the 97.72% and 99% percentile values are presented. Figure 4-8 illustrates the Design, Qualification Test and Acceptance Test factors that are required. If the payload intends to utilize MIL-HDBK-340A factors then the 99% Stage 2 Ignition loads must

Reference	Design Limit Loads (MPE) (Note 1)		Conventional Structures		Composite/Sandwich/ Bonded Structures	
			Qual Factor	Accept/ Proof Factor	Qual Factor	Accept/ Proof Factor
MIL-HDBK-340A, 1 Apr 99 (Ref: MIL-STD-1540D)	99% w/90% Conf.	Qualified	1.25	N/A	1.25	1.1
		Flightproof	N/A	1.1	N/A	1.1
		Protoflight	N/A	1.25	N/A	1.25
NASA GEVS, Rev A, Jun 96	97% w/50% Conf.	Qualified	1.25 (Note 2)	N/A	1.25 (Note 2)	1.25
		Protoflight	N/A	1.25 (Note 2)	N/A	1.25 (Note 2)
		Qual By Analysis	2.0 (Yield) 2.6 (Ulti)	N/A	N/A	N/A

Notes: (1) – MPE. Multiplied by test factors to get test results.
(2) – Analysis required showing positive margins to ultimate at 1.4xLimit Load
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Figure 4-8. Factors of Safety Payload Design and Test

be used. If NASA GEVS (Rev A) factors are planned then the 97.72% Stage 2 Ignition loads will apply.

4.5.1. Recommended Payload Testing and Analysis

Sufficient payload testing and/or analysis must be performed to ensure the safety of ground crews and to ensure mission success. The payload design should comply with the testing and design factors of safety in Figure 4-8. At a minimum, it is recommended that the following tests be performed:

a. ***Structural Integrity*** — Static loads, sine vibration, or other tests should be performed that combine to encompass the acceleration load environment presented in Section 4.1.

b. ***Random Vibration*** — The flight level environment is defined in Figure 4-4. Recommended test levels are defined in Figure 4-9.

Test Type	Test Purpose	Test Level
Random Vibration: The Flight Limit Level Is Characterized In Figure 4-4	Qualification	Flight Limit Level +6dB
	Acceptance	Flight Limit Level
	Protoflight	Flight Limit Level +3dB

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Figure 4-9. Recommended Payload Testing Requirements

c. ***Acoustics*** — Depending on the payload configuration, the payload organization may elect to perform acoustic testing on the payload, or sub-sections of the payload, in addition to, or in-lieu of, random vibration testing. The acoustic levels are defined in Figure 4-7.

The payload organization must provide Orbital with a list of the tests and test levels to which the payload was subjected prior to payload arrival at the integration facility.

4.6. Thermal and Humidity Environments

The thermal and humidity environment to which the payload may be exposed during vehicle processing and pad operations are defined in the sections that follow and listed in Figure 4-10.

4.6.1. Ground Operations

The payload environment will be maintained by the Ground or Pad Air Conditioning Systems (GACS or PACS). The GACS provides conditioned air to the payload in the Vehicle Assembly Building (VAB) after fairing integration. The PACS is used at the launch pad after vehicle stacking operations. Air Conditioning (AC) is not provided during transport or lifting operations without the enhanced option that includes High Efficiency Particulate Air (HEPA) filtration. The conditioned air enters the fairing at a location forward of the payload, exits aft of the payload and is maintained up to 5 minutes prior to launch (for the 61 in. fairing, the conditioned air can be maintained until liftoff). Baffles are provided at the air conditioning inlet to reduce impingement velocities on the payload if required.

Fairing inlet conditions are selected by the customer, and are bounded as follows:

- Dry Bulb Temperature: 55-85 °F (13-29 °C) controllable to ±4 °F (±2 °C) of setpoint
- Dew Point Temperature: 38-62 °F (3-17 °C)

Event	Temp Range		Control	Humidity (%)	Purity Class (Note 3)
	Deg C	Deg F			
Pre-Payload Fairing Installation • Outside VAB Clean Tent • Inside VAB Clean Tent	 23 ±5 23 ±5	 74 ±10 74 ±10	 AC Filtered AC	 45 ±15 45 ±15	 None 100 K (M6.5)
Post-Payload Fairing Installation (GSE) • VAB (GACS) • Transportation to Launch Pad (Optional) • Lifting Operations (Optional)	PLF Inlet 23 ±5 Ambient (Note 4) Ambient	PLF Inlet 74 ±10 Ambient (Note 4) Ambient	 Filtered AC Filtered Ambient Filtered AC	 45 ±15 <60 (Note 1) Ambient	 100 K (M6.5) 100 K (M6.5) Optional
PACS (Ground)	PLF Inlet 13 - 29	PLF Inlet 55 - 85	Filtered AC Filtered AC	(Note 2)	100 K (M6.5)

Notes:

1. GSE AC Performance is Dependent Upon Ambient Conditions. Temperature Is Selectable and Controlled to Within ±4 °F (±2 °C) of Set Point. Resultant Relative Humidity is Maintained to 45 ±15%.
2. PACS Performance is Dependent Upon Ambient Conditions (Dew Point). Temperature is Selectable and Controlled Within ±4 °F (±2 °C) of Set Point. Resultant Relative Humidity is Maintained to 45 ±15%.
3. Class 10K (M5.5) Can Be Provided Inside the VAB Clean Tent and at the Payload Fairing Air-Conditioning Inlet on a Mission Specific Basis As a Non-Standard Service.
4. Temperature Control During Transport and Lifting Ops Available as a Non-Standard Service.

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Figure 4-10. Payload Thermal and Humidity Environments

- Relative Humidity: determined by drybulb and dewpoint temperature selections and generally controlled to within ±15%. Relative humidity is bound by the psychrometric chart and will be controlled such that the dew point within the fairing is never reached.
- Max Flow: 500 cfm

4.6.2. Powered Flight

The maximum fairing inside wall temperature will be maintained at less than 200 °F (93 °C), with an emissivity of 0.92 in the region of the payload. As a non-standard service, a low emissivity coating can be applied to reduce emissivity to less than 0.1. Interior surfaces aft of the payload interface will be maintained at less than 250 °F (121 °C). Figure 4-11 shows the worst case transient temperature profile of the inner fairing surface adjacent to the payload during powered flight.

This temperature limit envelopes the maximum temperature of any component inside the payload fairing with a view factor to the payload with the exception of the Stage 4 motor. The maximum Stage 4 motor surface temperature exposed to the payload will not exceed 350 °F (177 °C), assuming no shielding between the aft end of the payload and the forward dome of the motor assembly. Whether this temperature is attained prior to payload separation is dependent upon mission timeline.

The fairing peak vent rate is typically less than 0.6 psi/sec. Fairing deployment will be initiated at a time in flight that the maximum dynamic pressure is less than 0.01 psf.

4.6.3. Nitrogen Purge (non-standard service)

If required for spot cooling of a payload component, Orbital will provide GN2 flow to localized regions in the fairing as a non-standard service. This option is discussed in more detail in Section 8.3.2.

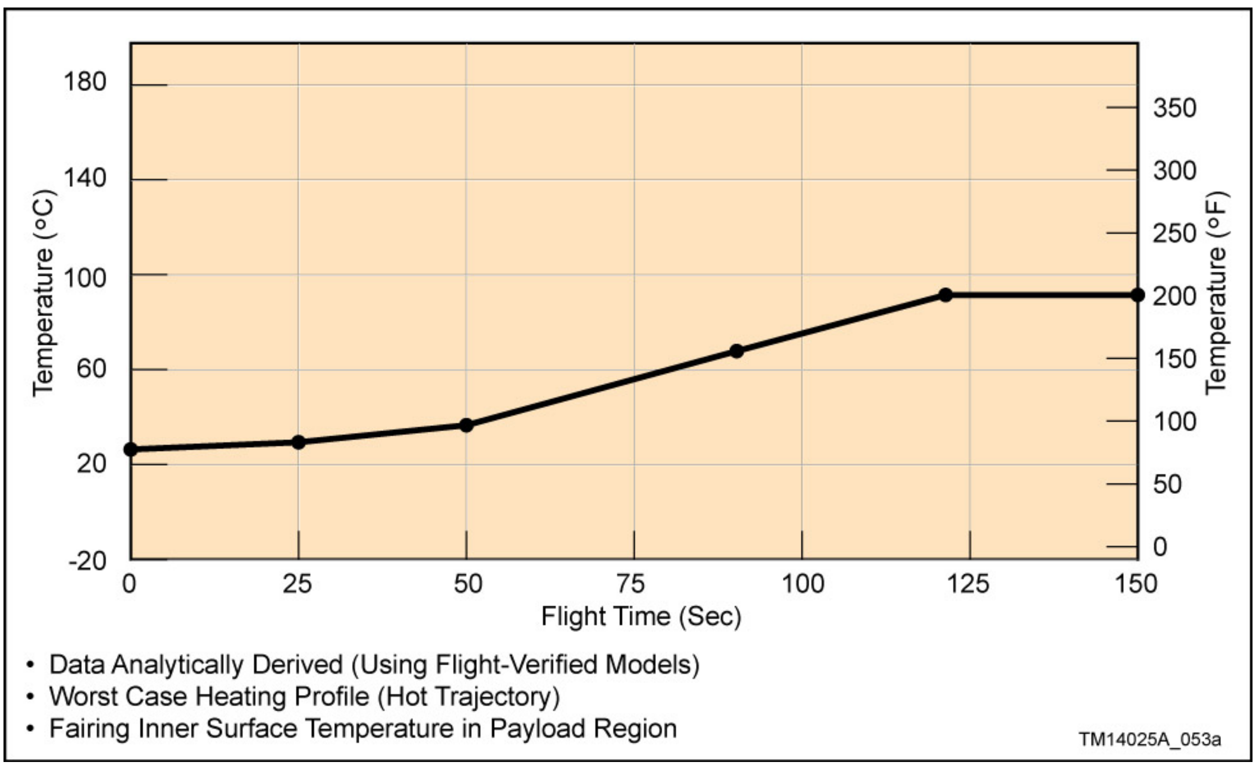

Figure 4-11. Minotaur I Worst-Case Payload Fairing Inner Surface Temperatures During Ascent (Payload Region)

4.7. Payload Contamination Control

The Minotaur I vehicle, all payload integration procedures, and Orbital's contamination control program have been designed to minimize the payload's exposure to contamination from the time the payload arrives at the payload processing facility through orbit insertion and separation. The payload is maintained in a Class 100,000 cleanliness environment at all times. Prior to encapsulation in the payload fairing, the cleanliness level is ensured through the use of clean tents in all payload processing areas. The Vehicle Assembly Building is maintained as a visibly clean, temperature and humidity controlled work area at all times. The Minotaur I assemblies that affect cleanliness within the encapsulated payload volume include the fairing, and the payload cone assembly. These assemblies are cleaned such that there is no particulate or non-particulate matter visible to the normal unaided eye when inspected from 2 to 4 feet under 50 ft-candle incident light (Visibly Clean Level II). If required, the payload can be provided with enhanced contamination control as a non-standard service. (See Section 8.3.3.)

Minotaur I contamination control is based on industry standard contamination reference documents, including the following:

- MIL-STD-1246C, "Product Cleanliness Levels and Contamination Control Program"
- FED-STD-209E, "Airborne Particulate Cleanliness Classes in Clean Rooms and Clean Zones."

4.8. Payload Electromagnetic Environment

The payload Electromagnetic Environment (EME) results from two categories of emitters: 1) Minotaur I onboard antennas and 2) Range radar. All power, control and signal lines inside the payload fairing are shielded and properly terminated to minimize the potential for Electromagnetic Interference (EMI). The Minotaur I payload fairing is Radio Frequency (RF) opaque, which shields the payload from most external RF signals while the payload is encapsulated. Details of the analysis can be provided upon request.

Figure 4-12 lists the frequencies and maximum radiated signal levels from vehicle antennas that are located near the payload during ground operations and powered flight. Antennas located inside the fairing are inactive until after fairing deployment. The specific EME experienced by the payload during ground processing at the VAB and the launch site will depend somewhat on the specific facilities that are utilized as well as operational details. However, typically the field strengths experienced by the payload during ground processing with the fairing in place are controlled procedurally and will be

SOURCE	1	2	3	4	5
Function	Command Destruct	Tracking Transponder	Tracking Transponder	Launch Vehicle	Instrumentation on Telemetry (Optional)
Receive/Xmit	Receive	Transmit	Receive	Transmit	Transmit
Band	UHF	C-Band	C-Band	S-Band	S-Band
Frequency (MHz)	425.0	5765	5690	2288.5	2269.5
Bandwidth	N/A	N/A			
Power Output	N/A	400 W (peak)	N/A	10 W	10 W
Sensitivity	-107 dB		-70 dB		
Modulation	Tone	Pulse Code	Pulse Code	PCM/FM	PCM/FM
Field Strength at PL Interfaces	Receive Only	3.016 V/M Average 67.436 V/M 0.5 μsec		99.316 V/M	

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Figure 4-12. Minotaur I Launch Vehicle RF Emitters and Receivers

less than 2 V/m from continuous sources and less than 10 V/m from pulse sources. Range transmitters are typically controlled to provide a field strength of 10 V/m or less. This EME should be compared to the payload's RF susceptibility levels (MIL-STD-461, RS03) to define margin.

5. PAYLOAD INTERFACES

This section describes the available mechanical, electrical and Launch Support Equipment (LSE) interfaces between the Minotaur I launch vehicle and the payload.

5.1. Standard Payload Fairing (50 Inch Diameter)

The standard payload fairing consists of two graphite composite halves, with a nosecap bonded to one of the halves, and a separation system. Each composite half is composed of a cylinder and an ogive section. The two halves are held together by two titanium straps, both of which wrap around the cylinder section, one near its midpoint and one just aft of the ogive section. Additionally, an internal retention bolt secures the two fairing halves together at the surface where the nosecap overlaps the top surface of the other fairing half. The base of the fairing is separated using a frangible joint. Severing the frangible joint allows each half of the fairing to then rotate on hinges mounted on the Stage 3 side of the interface. A contained hot gas generation system is used to drive pistons that force the fairing halves open. All fairing deployment systems are non-contaminating.

5.1.1. Payload Dynamic Design Envelope

The fairing drawing in Figure 5-1 shows the maximum dynamic envelopes available for the payload during powered flight. The dynamic envelopes shown account for fairing and vehicle structural deflections only. The payload contractor must take into account deflections due to spacecraft design and manufacturing tolerance stack-up within the dynamic envelope. Proposed payload dynamic envelope violations must be approved by OSP via the ICD.

No part of the payload may extend aft of the payload interface plane without specific OSP approval. These areas are considered stay out zones for the payload and are shown in Figure 5-1. Incursions to these zones may be approved on a case-by-case basis after additional verification that the incursions do not cause any detrimental effects. Vertices for payload deflection must be given with the Finite Element Model to evaluate payload dynamic deflection with the Coupled Loads Analysis (CLA). The payload contractor should assume that the interface plane is rigid; Orbital has accounted for deflections of the interface plane. The CLA will provide final verification that the payload does not violate the dynamic envelope.

5.1.2. Payload Access Door

Orbital provides one 8.5 in. x 13.0 in. (21.6 cm x 33.0 cm), graphite, RF-opaque payload fairing access door. The door can be positioned according to user requirements within the zone defined in Figure 5-2. The position of the payload fairing access door must be defined no later than L-8 months. Additional access doors can be provided as a non-standard service.

5.1.3. Increased Volume Payload Fairing (61 Inch Diameter - Non-Standard)

An increased volume payload fairing is available as an enhancement for the Minotaur I vehicle. This larger fairing is discussed in more detail in Section 8.1.4.

5.2. Payload Mechanical Interface and Separation System

Minotaur I provides for a standard non-separating payload interface and several optional Orbital-provided payload separation systems. Orbital will provide all flight hardware and integration services necessary to attach non-separating and separating payloads to Minotaur I. Ground handling equipment is typically the responsibility of the payload contractor. All attachment hardware, whether Orbital or customer provided, must contain locking features consisting of locking nuts, inserts or fasteners.

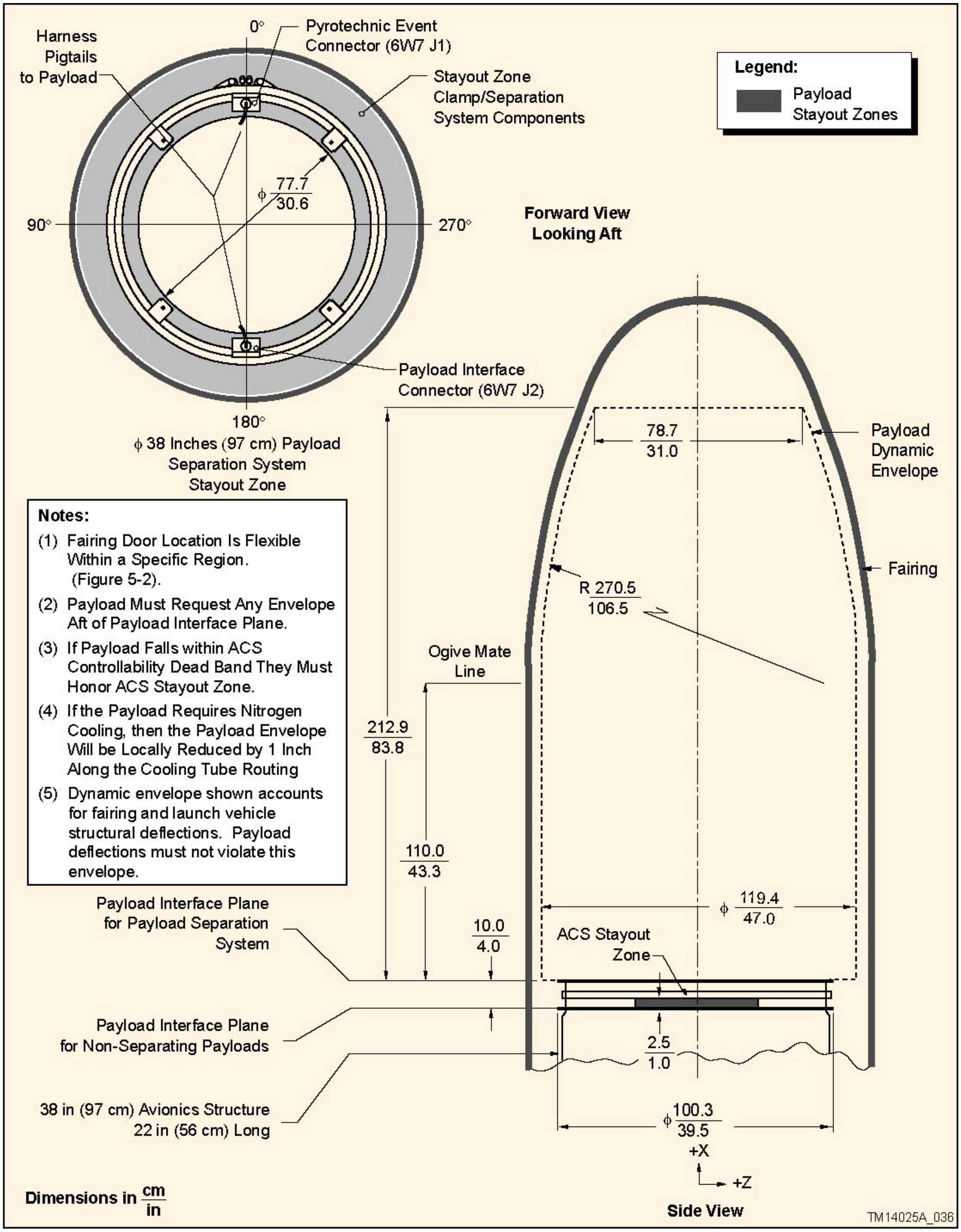


Figure 5-1. Payload Fairing Dynamic Envelope With 38 in. (97 cm.) Diameter Payload Interface

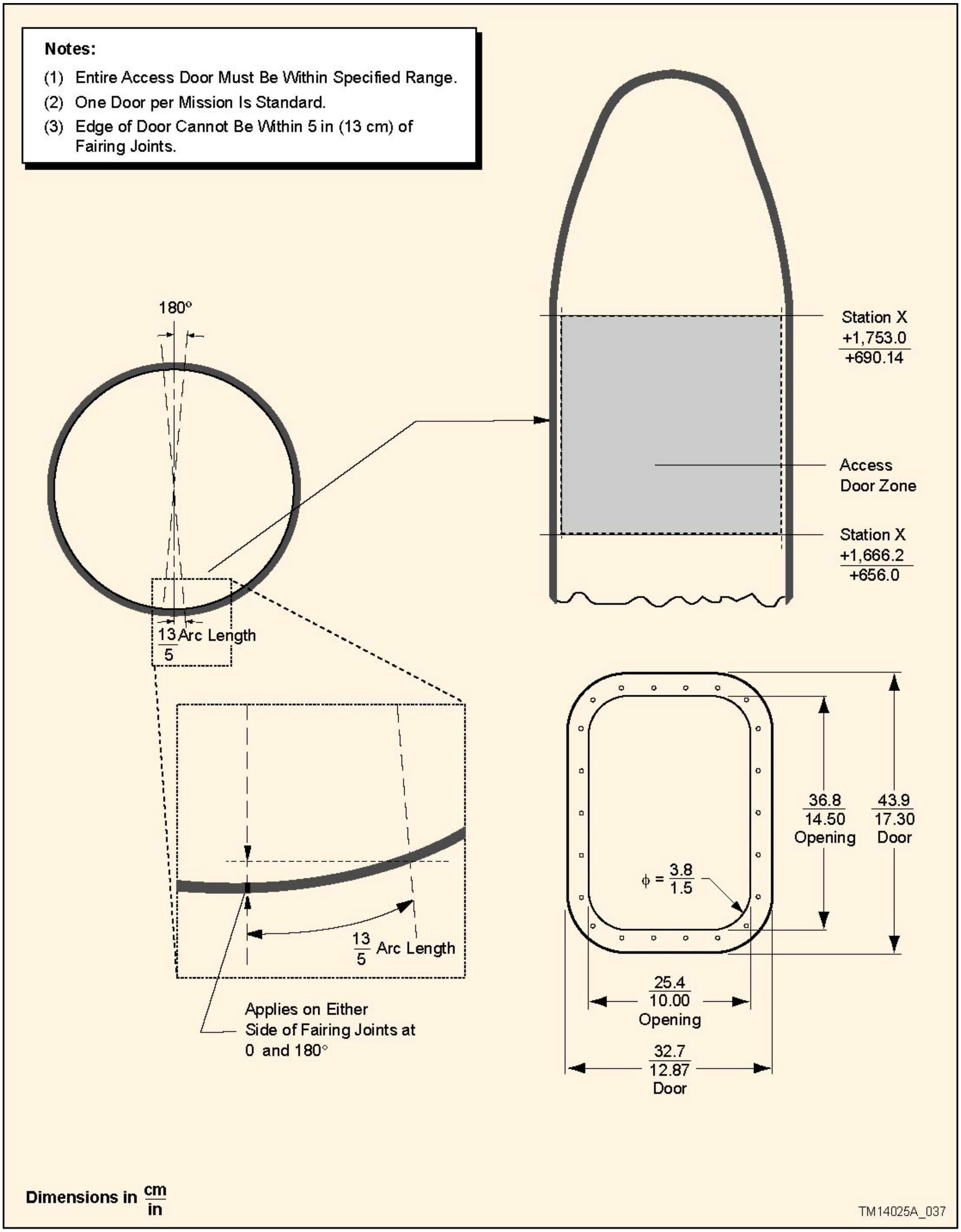


Figure 5-2. Payload Fairing Access Door Placement Zone

5.2.1. Standard Non-Separating Mechanical Interface

Figure 5-3 illustrates the standard, non-separating payload mechanical interface. This is for payloads that provide their own separation system and payloads that will not separate. Direct attachment of the payload is made on the Avionics Structure with sixty #10 fasteners as shown in Figure 5-3.

The Minotaur I avionics section is designed to accommodate a 1500 lbm (680 kg) payload with a cg 30 in. (76.2 cm) above the fixed interface flange. Therefore, as an initial guideline, payload mass times its cg location above this fixed interface needs to be less than or equal to a mass moment of 45,000 lbm-in (51820 kg-cm). The payload mass and cg location must include the Payload Attach Fitting (PAF) hardware (adapter cone, separation system, isolation system, etc.), in addition to the actual spacecraft mass properties.

5.2.2. Separating Mechanical Interface

Three flight-qualified, optional separation systems are available for the OSP SLV. These systems are discussed in detail in Section 8.1.1.

5.3. Payload Electrical Interfaces

The existing, standard design for the payload electrical interface supports battery charging, external power, discrete commands, discrete telemetry, analog telemetry, serial communication, payload separation indications, and up to eight (8) redundant ordnance events in support of payload processing and operational requirements.

If an Orbital-provided separation system is provided, Orbital will provide the cabling to the separable interface plane. If the separation system is not provided by Orbital, the payload provider will be responsible for providing the cabling from the payload interface to the payload separation plane. Orbital will provide mating halves of the interface connectors to the payload provider.

5.3.1. Standard Payload/Minotaur I Electrical Interface, With A Payload Provided Separation System

The standard Minotaur I electrical interface is located at the Payload/Minotaur I mating surface. Note that this interface is not the separation plane, it is defined for payloads that provide their own separation system. The Payload Umbilical connector, 4W33 J2, provides 60 wires from the ground to the spacecraft via a dedicated payload umbilical within the vehicle, as shown in Figure 5-4. The length of the internal umbilical is approximately 25 ft (7.62 m). The cabling from the LEV to the launch vehicle is approximately 130 ft (39.6 m). This umbilical is a dedicated pass through harness for ground processing support. It allows the payload command, control, monitor, and power to be easily configured per each individual user's requirements. The umbilical wiring is configured as a 1:1 from the Payload/Minotaur I interface through to the payload EGSE interface in the Launch Equipment Vault, the closest location for operating customer supplied payload EGSE equipment.

It is a Launch Vehicle requirement that the payload provide two (2) separation loopback circuits on the payload side of the separation plane. These should ideally be wired into different separation connectors for redundancy. These breakwires are used for positive separation indication telemetry and initiation of the CCAM maneuver.

The interface connector IDs, connector part numbers, descriptions and configurations are as specified in Table 5-1. Note also that Orbital will supply the mating halves of the three interface connectors to the payload provider to increase the level of confidence in Payload mate operations.

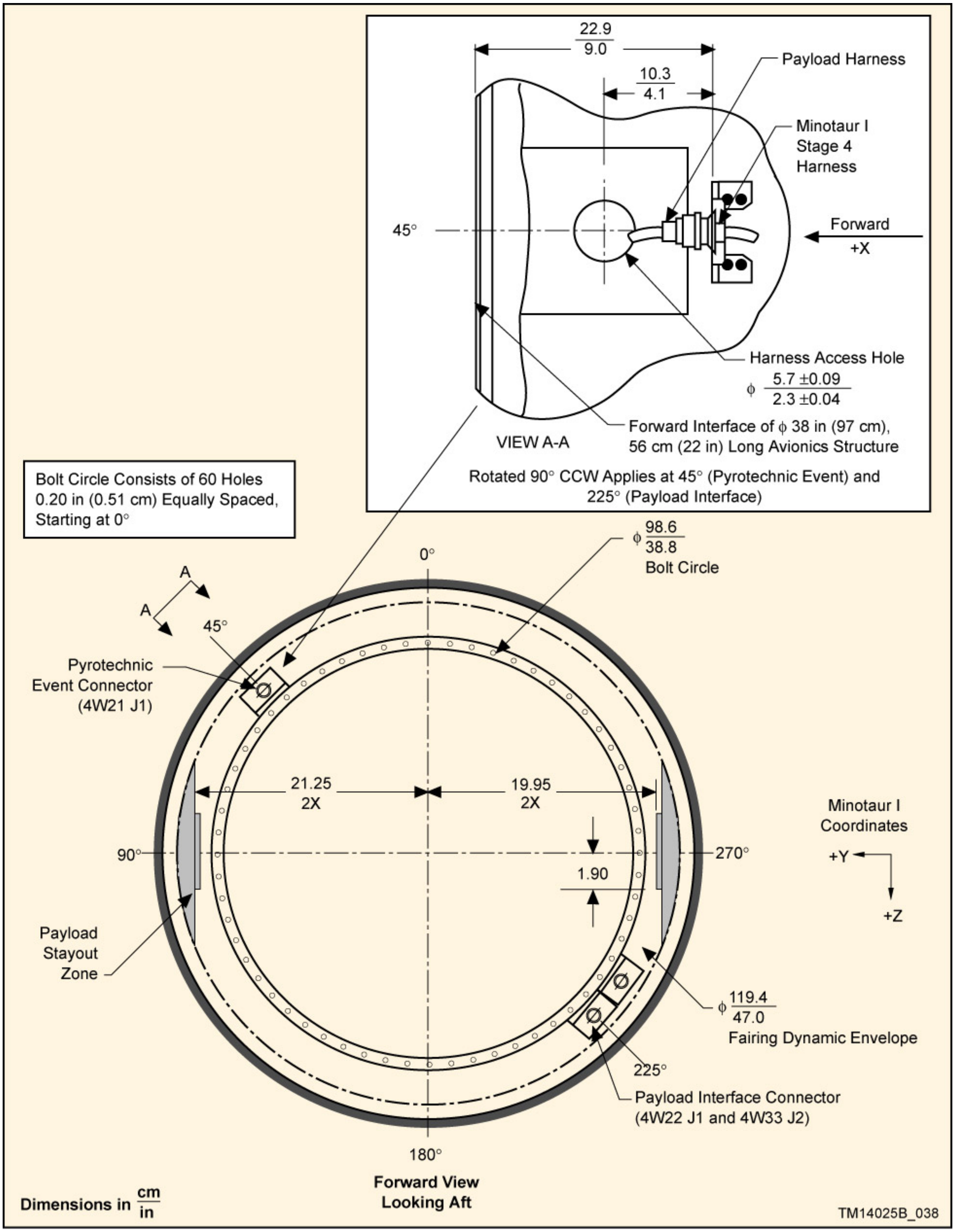


Figure 5-3. Non-Separable Payload Mechanical Interface

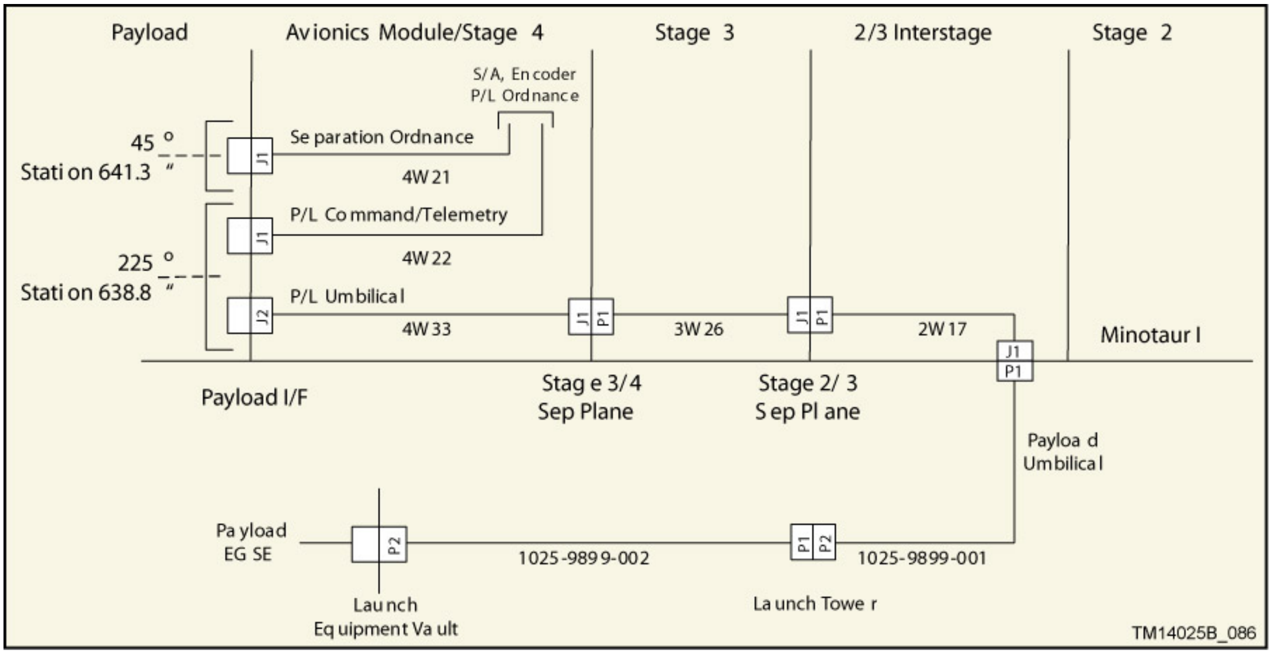


Figure 5-4. Payload Electrical Interface Block Diagram Without Orbital Supplied Separation System

Figure 5-5 details the pin outs for the standard interface umbilical. All wires are twisted, shielded pairs, and pass through the entire umbilical system, both vehicle and ground, as 1:1 to simplify and standardize the payload umbilical configuration requirements while providing maximum operational flexibility to the payload provider.

5.3.2. Payload Interface Circuitry

Standard interface circuitry passing through the payload-to-launch vehicle electrical connections are shown in Figure 5-6. This figure details the interface characteristics for launch vehicle commands, discrete and analog telemetry, separation loopbacks, pyro initiation, and serial communications interfaces with the launch vehicle avionics systems.

5.3.3. Payload Battery Charging

Orbital provides the capability for remote controlled charging of payload batteries, using a customer provided battery charger. This power is routed through the payload umbilical cable. The payload battery charger should be sized to withstand the line loss from the LEV to the spacecraft, as defined in the Payload ICD.

5.3.4. Payload Command and Control

The Minotaur I standard interface provides discrete sequencing commands generated by the launch vehicle's Ordnance Driver Module (ODM) that are available to the payload as closed circuit opto-isolator command pulses of 5 A in lengths of 35 ms minimum. The total number of ODM discretes is sixteen (16) and can be used for any combination of (redundant) ordnance events and/or discrete commands depending on the payload requirements.

5.3.5. Pyrotechnic Initiation Signals

The standard payload interface provides sixteen (16) individual circuits to initiate payload deployment events through two dedicated Ordnance Driver Modules. The ODM provides a 5 A, 35 ms minimum, current limited pulse into a 1.5 ohm initiation device.

5.3.6. Payload Discrete Telemetry

Standard Minotaur I service provides five (5) discrete (bi-level) telemetry monitors through dedicated channels in the vehicle encoder. The payload provider must provide the 5 Vdc source and the return path. The current at the payload interface must be less than 10 mA. Payload

TABLE 5-1. PAYLOAD INTERFACE, WITH NO ORBITAL SEPARATION SYSTEM, CONNECTOR CHARACTERISTICS

	Payload Connectors	**Minotaur I Connectors**
Connector ID	**Defined by Payload Provider**	**4W21 J1**
Connector Part #	MS3467E37-50PN	D8174E37-0SN
Description	Connector, Circular, Plug	Connector, Circular, Jam Nut Recpt
# Pins	37	37
Type	Pins	Sockets
AWG	20	20
Supplied By	Orbital	Orbital
Connector ID	**Defined by Payload Provider**	**4W22 J1**
Connector Part #	MS27467T23F35PN	MS27468T23F35SN
Description	Connector, Circular, Grounded Plug	Connector, Circular, Jam Nut Recpt
# Pins	100	100
Type	Pins	Sockets
AWG	22	22
Supplied By	Orbital	Orbital
Connector ID	**Defined by Payload Provider**	**4W33 J2**
Connector Part #	MS27467T25F61PN	MS27468T25F61SN
Description	Connector, Circular, Grounded Plug	Connector, Circular, Jam Nut, Recpt
# Pins	61	61
Type	Pins	Sockets
AWG	20	20
Supplied By	Orbital	Orbital
Connector ID	**Defined by Payload Provider**	**-002 PL Umbi P2**
Connector Part #	D38999/24WJ61SA	D38999/26WJ61PA
Description	Connector, Circular, Jam Nut Recpt	Connector, Circular, Grounded Plug
# Pins	61	61
Type	Sockets	Pins
AWG	20	20
Supplied By	Orbital	Orbital

discrete telemetry requirements and sampling characteristics will be specified in the Payload ICD and should not change once the final telemetry format is released at approximately L-6 months.

5.3.7. Payload Analog Telemetry

Standard Minotaur I service provides 6 analog telemetry monitors through dedicated channels in the vehicle encoder. Analog telemetry signal characteristics are 20 mVp-p to 10 Vp-p, +/- 50 V. Payload analog telemetry requirements and signal characteristics will be specified in the Payload ICD and should not change once the final telemetry format is released at approximately L-6 months.

5.3.8. Payload Separation Monitor Loopbacks

Separation breakwire monitors are required on both sides of the payload separation plane. With the Orbital provided separation systems, Minotaur I provides three (3) separation loopbacks on the launch vehicle side of the separation plane for positive payload separation indication.

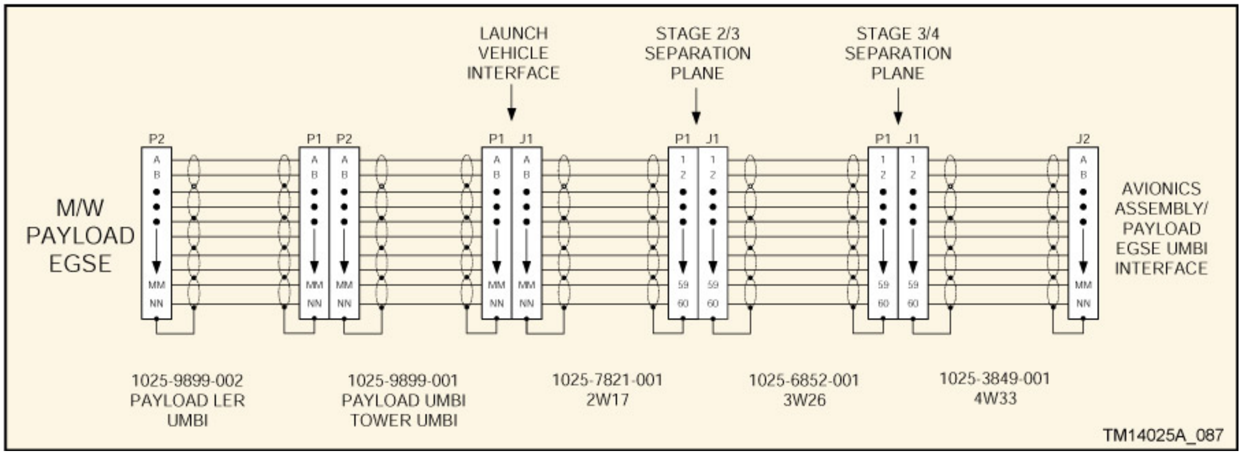


Figure 5-5. Payload Umbilical Pin Outs

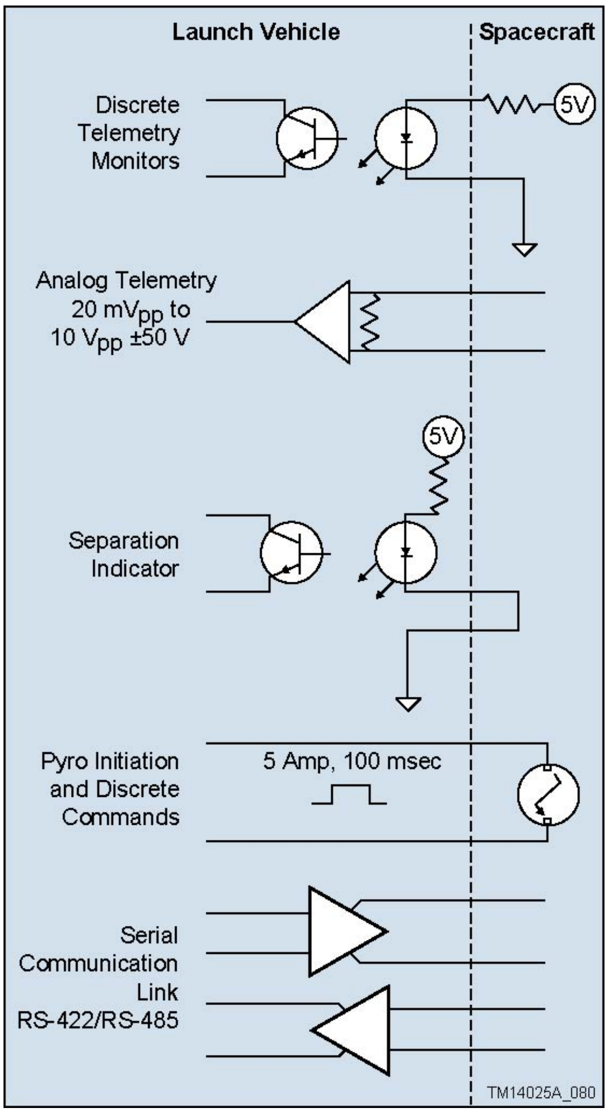


Figure 5-6. Typical Minotaur I Payload Electrical Interface Circuits

Minotaur I also requires two (2) separate loopbacks on the payload side of the separation plane. These are used for telemetry indication of separation and also the initiation of the Stage 4 CCAM maneuver.

5.3.9. Telemetry Interfaces

The standard Minotaur I payload interface provides a 16Kbps RS-422/RS-485 serial interface for payload use with the flexibility to support a variety of channel/bit rate requirements, and provide signal conditioning, PCM formatting (programmable) and data transmission bit rates. The number of channels, sample rates, etc. will be defined in the Payload ICD.

5.3.10. Non Standard Electrical Interfaces

Non-standard services such as serial command and telemetry interfaces can be negotiated between Orbital and the payload provider on a mission-by-mission basis. The selection of the separation system could also impact the payload interface design and will be defined in the Payload ICD.

5.3.11. Electrical Launch Support Equipment

Orbital will provide space for a rack of customer supplied EGSE in the LCR in the on-pad equipment vault. The equipment will interface with the launch vehicle/spacecraft through the dedicated payload umbilical interface. The payload customer is responsible for providing cabling from the EGSE location to the launch vehicle P2 umbilical interface in the LEV.

Separate payload ground processing harnesses that mate directly with the payload can be accommodated through the payload access door(s) as defined in the Payload ICD.

5.4. Payload Design Constraints

The following sections provide design constraints to ensure payload compatibility with the Minotaur I system.

5.4.1. Payload Center of Mass Constraints

Along the Y and Z axes, the payload c.g. must be within 1.5 in (3.8 cm) of the vehicle centerline and no more than 30 in. (76.2 cm) forward of the payload interface for the standard configuration (within the accuracy listed in Figure 5-7). Payloads whose c.g. extend beyond the 1.5 in. (3.81 cm) lateral offset limit will require Orbital to verify the specific offsets that can be accommodated.

Measurement	Accuracy
Mass	±1 lbm (±0.5 kg)
Principal Moments of Inertia	±5%
Cross Products of Inertia	±0.5 sl - ft^2(±0.7 kg - m^2)
Center of Gravity X, Y and Z Axes	±0.25 in (±6.4 mm)

TM14025A_045

Figure 5-7. Payload Mass Properties Measurement Tolerance

5.4.2. Final Mass Properties Accuracy

The final mass properties statement must specify payload weight to an accuracy of at least 1 lbm (0.5 kg), the center of gravity to an accuracy of at least 0.25 in (6.4 mm) in each axis, and the products of inertia to an accuracy of at least 0.5 slug-ft2 (0.7 kg-m2). In addition, if the payload uses liquid propellant, the slosh frequency must be provided to an accuracy of 0.2 Hz, along with a summary of the method used to determine slosh frequency.

5.4.3. Pre-Launch Electrical Constraints

Prior to launch, all payload electrical interface circuits are constrained to ensure there is no current flow greater than 10 mA across the payload electrical interface plane. The primary support structure of the spacecraft shall be electrically conductive to establish a single point electrical ground.

5.4.4. Payload EMI/EMC Constraints

The Minotaur I avionics share the payload area inside the fairing such that radiated emissions compatibility is paramount. OSP places no firm radiated emissions limits on the payload other than the prohibition against RF transmissions within the payload fairing. Prior to launch, Orbital requires review of the payload radiated emission levels (MIL-STD-461, RE02) to verify overall launch vehicle EMI safety margin (emission) in accordance with MIL-E-6051. Payload RF transmissions are not permitted after fairing mate and prior to an ICD specified time after separation of the payload. An EMI/EMC analysis may be required to ensure RF compatibility.

Payload RF transmission frequencies must be coordinated with Orbital and range officials to ensure non-interference with Minotaur I and range transmissions. Additionally, the customer must schedule all RF tests at the integration site with Orbital in order to obtain proper range clearances and protection.

5.4.5. Payload Dynamic Frequencies

To avoid dynamic coupling of the payload modes with the natural frequency of the vehicle, the spacecraft should be designed with a structural stiffness to ensure that the lateral fundamental frequency of the spacecraft <u>including isolation system and separation system, fixed at the spacecraft interface is typically greater than 12 Hz</u>. However, this value is effected significantly by other factors such as the coupled dynamics of the spacecraft, isolation system and/or separation system. Therefore, the final determination of compatibility must be made on a mission-specific basis.

5.4.6. Payload Propellant Slosh

Slosh models at 1, 3, 6g are required for payloads with liquid propellant. The model provided shall be either a NASTRAN or Craig/Bampton model. Data on first sloshing mode are required and data on higher order modes are desirable. The slosh model should be provided with the payload finite element model submittals.

5.4.7. System Safety Constraints

OSP considers the safety of personnel and equipment to be of paramount importance. EWR 127-1 outlines the safety design criteria for Minotaur I payloads. These are compliance documents and must be strictly followed. It is the responsibility of the customer to ensure that the payload meets all OSP, Orbital, and range imposed safety standards.

Customers designing payloads that employ hazardous subsystems are advised to contact OSP early in the design process to verify compliance with system safety standards.

6. MISSION INTEGRATION

6.1. Mission Management Approach

The Minotaur I program is managed through the US Air Force, Space and Missile Systems Center, Rocket Systems Launch Program (RSLP). RSLP serves as the primary point of contact for the payload customers for the Minotaur I launch service. A typical integrated OSP organizational structure is shown in Figure 6-1. Open communication between RSLP, Orbital, and the customer, emphasizing timely transfer of data and prudent decision-making, ensures efficient launch vehicle/payload integration operations.

6.1.1. RSLP Mission Responsibilities

The program office for all OSP missions is the RSLP. They are the primary Point of Contact (POC) for all contractual and technical coordination. RSLP contracts with Orbital to provide the Launch Vehicle and separately with commercial Spaceports and/or Government Launch Ranges for launch site facilities and services. Once a mission is identified, RSLP will assign a Mission Manager to coordinate all mission planning and contracting activities. RSLP is supported by NGC and other associate contractors for technical and logistical support, particularly utilizing their extensive expertise and background knowledge of the Minuteman booster and subsystems.

6.1.2. Orbital Mission Responsibilities

As the launch vehicle provider, Orbital's responsibilities fall into four primary areas:

a. Launch Vehicle Program Management
b. Mission Management
c. Engineering
d. Launch Site Operations

Orbital assigns a Mission Manager to manage the launch vehicle technical and programmatic interfaces for a particular mission. The Orbital Mission Manager is the single POC for all aspects of a specific mission. This person has overall program authority and responsibility to ensure that payload requirements are met and that the appropriate launch vehicle services are provided.

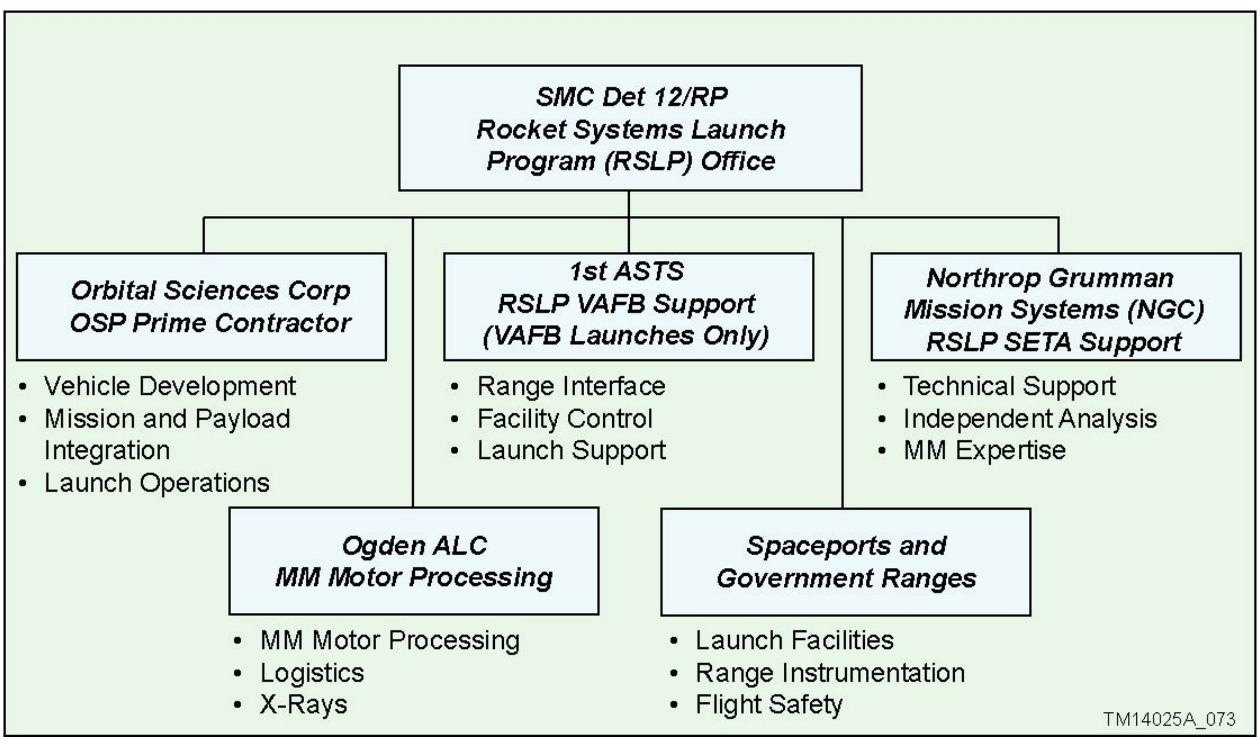


Figure 6-1. OSP Management Structure

The Orbital Mission Manager will jointly chair the Mission Integration Working Groups (MIWGs) with the RSLP Mission Manager. The Mission Managers responsibilities include detailed mission planning, payload integration services, systems engineering, mission-peculiar design and analyses coordination, payload interface definition, launch range coordination, integrated scheduling, launch site processing, and flight operations.

6.2. Mission Planning and Development

OSP will assist the customer with mission planning and development associated with Minotaur I launch vehicle systems. These services include interface design and configuration control, development of integration processes, launch vehicle analyses, facilities planning, launch campaign planning to include range services and special operations, and integrated schedules.

The procurement, analysis, integration and test activities required to place a customer's payload into orbit are typically conducted over a 20 month long standard sequence of events called the Mission Cycle. This cycle normally begins 18 months before launch, and extends to eight weeks after launch.

Once contract authority to proceed is received, the Mission Cycle is initiated. The contract option designates the payload, launch date, and basic mission parameters. In response, the Minotaur I Program Manager designates an Orbital Mission Manager who ensures that the launch service is supplied efficiently, reliably, and on-schedule.

The typical Mission Cycle interweaves the following activities:

a. Mission management, document exchanges, meetings, and formal reviews required to coordinate and manage the launch service.
b. Mission analyses and payload integration, document exchanges, and meetings.
c. Design, review, procurement, testing and integration of all mission-peculiar hardware and software.
d. Range interface, safety, and flight operations activities, document exchanges, meetings and reviews.

Figure 6-2 details the typical Mission Cycle for a representative launch and how this cycle folds into the Orbital vehicle production schedule with typical payload activities and milestones. A typical Mission Cycle is based on a 18 month interval between mission authorization and launch. This interval reflects the OSP contractual schedule and has been shown to be an efficient schedule based on Orbital's Taurus and Pegasus program experience. However, OSP is flexible to negotiate either accelerated cycles, which may take advantage of the Minotaur I/Pegasus multi-customer production sets, or extended cycles required by unusual payload requirements, such as extensive analysis, complex payload-launch vehicle integrated designs, tests or funding limitations, or Orbital support of spacecraft design reviews.

6.3. Mission Integration Process

6.3.1. Integration Meetings

The core of the mission integration process consists of a series of Mission Integration and Range Working Groups (MIWG and RWG, respectively). The MIWG has responsibility for all physical interfaces between the payload and the launch vehicle. As such, the MIWG creates and implements the Payload-to-Minotaur I ICD in addition to all mission-unique analyses, hardware, software, and integrated procedures. The RWG is responsible for the areas of launch site operations; range interfaces; safety review and approval; and flight design, trajectory, and guidance. Documentation produced by the RWG includes all required range and safety submittals.

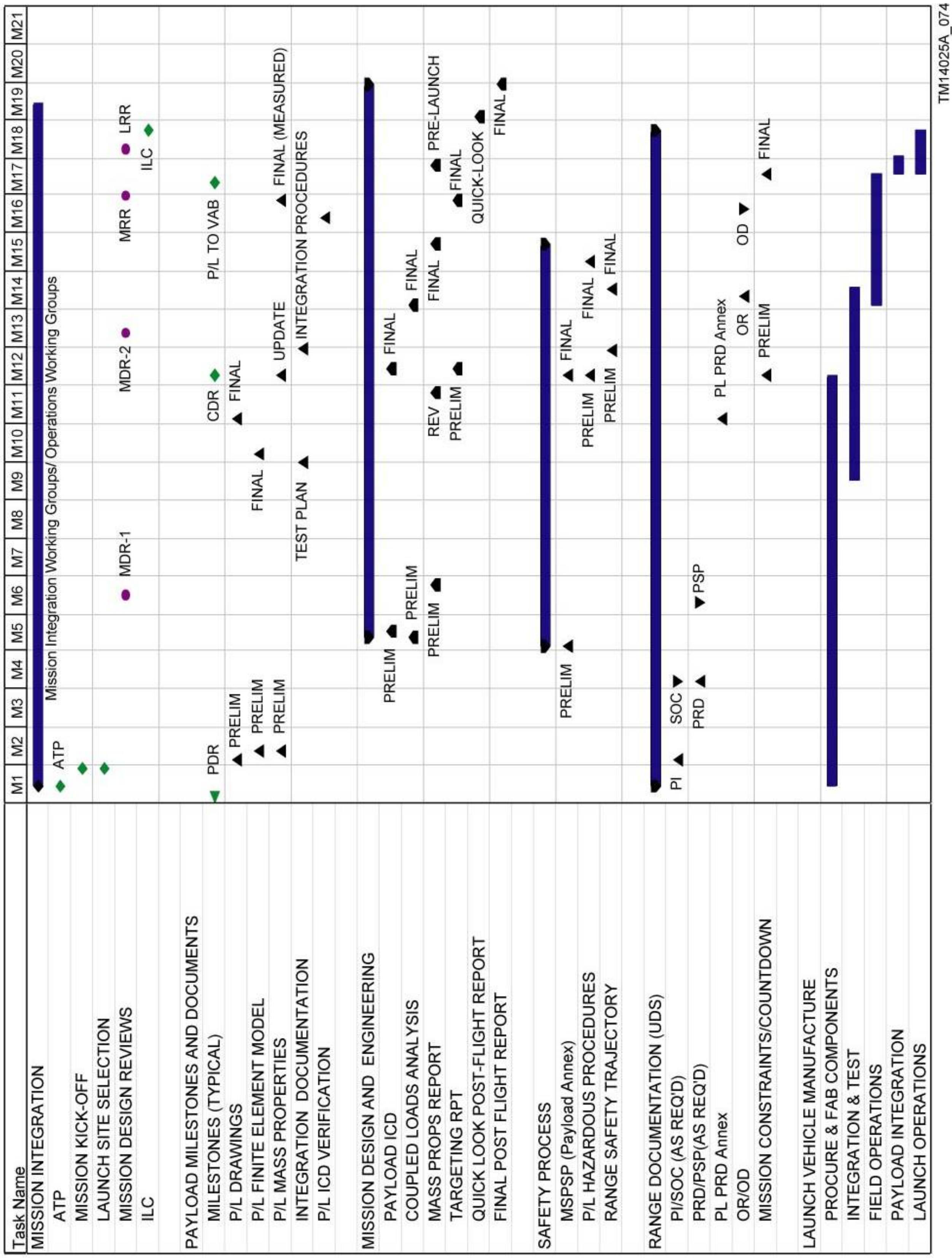


Figure 6-2. Typical Minotaur I Mission Integration Schedule

Working Group membership consists of the Mission Manager and representatives from Minotaur I engineering and operations organizations, as well as their counterparts from the customer organization. While the number of meetings, both formal and informal, required to develop and implement the mission integration process will vary with the complexity of the spacecraft, quarterly meetings are typical.

6.3.2. Mission Design Reviews (MDR)

Two mission-specific design reviews will be held to determine the status and adequacy of the launch vehicle mission preparations. They are designated MDR-1 and MDR-2 and are typically held 6 months and 13 months, respectively, after authority to proceed. They are each analogous to Preliminary Design Reviews (PDRs) and Critical Design Reviews (CDRs), but focus primarily on mission-specific elements of the launch vehicle effort.

6.3.3. Readiness Reviews

During the integration process, reviews are held to provide the coordination of mission participants and management outside of the regular contact of the Working Groups. Due to the variability in complexity of different payloads and missions, the content and number of these reviews can be tailored to customer requirements. As a baseline, Orbital will conduct two readiness reviews as described below.

Mission Readiness Review — Conducted within one month of launch, the Mission Readiness Review (MRR) provides a pre-launch assessment of integrated launch vehicle/payload/facility readiness prior to committing significant resources to the launch campaign.

Launch Readiness Review — The Launch Readiness Review (LRR) is conducted at L-1 day and serves as the final assessment of mission readiness prior to activation of range resources on the day of launch.

6.4. Documentation

Integration of the payload requires detailed, complete, and timely preparation and submittal of interface documentation. As the launch service provider, RSLP is the primary communication path with support agencies, which include—but are not limited to—the various Range support agencies and U.S. Government agencies such as the U.S. Department of Transportation and U.S. State Department. Customer-provided documents represent the formal communication of requirements, safety data, system descriptions, and mission operations planning. The major products and submittal times associated with these organizations are divided into two areas—those products that are provided by the customer, and those produced by Orbital.

6.4.1. Customer-Provided Documentation

Documentation produced by the customer is detailed in the following paragraphs.

6.4.1.1. Payload Questionnaire

The Payload Questionnaire is designed to provide the initial definition of payload requirements, interface details, launch site facilities, and preliminary safety data to OSP. The customer shall provide a response to the Payload Questionnaire form (Appendix A), or provide the same information in a different format, in time to support the Mission Kickoff Meeting. The customer's responses to the payload questionnaire define the most current payload requirements and interfaces and are instrumental in Orbital's preparation of numerous documents including the ICD, Preliminary Mission Analysis, and launch range documentation. Additional pertinent information, as well as preliminary payload drawings, should also be included with the response. Orbital understands that a definitive response to some questions may not be feasible. These items are defined during the normal mission integration process.

6.4.1.2. Payload Mass Properties

Payload mass properties must be provided in a timely manner in order to support efficient launch vehicle trajectory development and dynamic analyses. Preliminary mass properties should be submitted as part of the MRD at launch vehicle authority to proceed. Updated mass properties shall be provided at predefined intervals identified during the initial mission integration process. Typical timing of these deliveries is included in Figure 6-2.

6.4.1.3. Payload Finite Element Model

A payload mathematical model is required for use in Orbital's preliminary coupled loads analyses. Acceptable forms include either a Craig-Bampton model valid to 120 Hz or a NASTRAN finite element model. For the final coupled loads analysis, a test verified mathematical model is desired.

6.4.1.4. Payload Thermal Model for Integrated Thermal Analysis

An integrated thermal analysis can be performed for any payload as a non-standard service. A payload thermal model will be required from the payload organization for use in Orbital's integrated thermal analysis if it is required. The analysis is conducted for three mission phases:

a. Prelaunch ground operations;
b. Ascent from lift-off until fairing jettison; and
c. Fairing jettison through payload deployment.

Models must be provided in SINDA format. There is no limit on model size although turn-around time may be increased for large models.

6.4.1.5. Payload Drawings

Orbital prefers electronic versions of payload configuration drawings to be used in the mission specific interface control drawing, if possible. Orbital will work with the customer to define the content and desired format for the drawings.

6.4.1.6. Program Requirements Document (PRD) Mission Specific Annex Inputs

To obtain range support, a PRD must be prepared. A Minotaur I PRD has been submitted and approved by the Western Range. For launches from other Ranges, a Range-specific PRD will be created. This document describes requirements needed to generally support the Minotaur I launch vehicle. For each launch, an annex is submitted to specify the range support needed to meet the mission's requirements. This annex includes all payload requirements as well as any additional Minotaur I requirements that may arise to support a particular mission. The customer completes all appropriate PRD forms for submittal to Orbital.

6.4.1.6.1. Launch Operations Requirements (OR) Inputs

To obtain range support for the launch operation and associated rehearsals, an OR must be prepared. The customer must provide all payload pre-launch and launch day requirements for incorporation into the mission OR.

6.5. Safety

6.5.1. System Safety Requirements

In the initial phases of the mission integration effort, regulations and instructions that apply to spacecraft design and processing are reviewed. Not all safety regulations will apply to a particular mission integration activity. Tailoring the range requirements to the mission unique activities will be the first step in establishing the safety plan. OSP has three distinctly different mission approaches affecting the establishment of the safety requirements:

a. Baseline mission: Payload integration and launch operations are conducted at VAFB, CA.
b. Campaign/VAFB Payload Integration mission: Payload integration is conducted at VAFB and launch operations are conducted from a non-VAFB launch location.

c. Campaign/Non-VAFB Payload Integration mission: Payload integration and launch operations are conducted at a site other than VAFB.

For the baseline mission, spacecraft prelaunch operations are conducted at Orbital's VAB,

Building 1555, VAFB. For campaign style missions, the spacecraft prelaunch operations are performed at the desired launch site.

Before a spacecraft arrives at the processing site, the payload organization must provide the cognizant range safety office with certification that the system has been designed and tested in accordance with applicable safety requirements (e.g. EWR 127-1 Range Safety Requirements for baseline and VAFB Payload Integration missions). Spacecraft that integrate and/or launch at a site different than the processing site must also comply with the specific launch site's safety requirements. Orbital will provide the customer coordination and guidance regarding applicable safety requirements.

It cannot be overstressed that the applicable safety requirements should be considered in the earliest stages of spacecraft design. Processing and launch site ranges discourage the use of waivers and variances. Furthermore, approval of such waivers cannot be guaranteed.

6.5.2. System Safety Documentation

For each Minotaur I mission, OSP acts as the interface between the mission and Range Safety. In order to fulfill this role, OSP requires safety information from the payloader. For launches from either the Eastern or Western Ranges, EWR 127-1 provides detailed range safety regulations. To obtain approval to use the launch site facilities, specified data must be prepared and submitted to the OSP Program Office. This information includes a description of each payload hazardous system and evidence of compliance with safety requirements for each system. Drawings, schematics, and assembly and handling procedures, including proof test data for all lifting equipment, as well as any other information that will aid in assessing the respective systems should be included. Major categories of hazardous systems are ordnance devices, radioactive materials, propellants, pressurized systems, toxic materials, cryogenics, and RF radiation. Procedures relating to these systems as well as any procedures relating to lifting operations or battery operations should be prepared for safety review submittal. OSP will provide this information to the appropriate safety offices for approval.

7. GROUND AND LAUNCH OPERATIONS

7.1. Minotaur I/Payload Integration Overview

The processing of the Minotaur I upper stack utilizes many of the same proven techniques developed for the Pegasus and Taurus launch vehicles. This minimizes the handling complexity for both vehicle and payload. Horizontal integration of the Minotaur I vehicle upper stages simplifies integration procedures, increases safety and provides excellent access for the integration team. In addition, simple mechanical and electrical interfaces reduce vehicle/payload integration times, increase system reliability and minimize vehicle demands on payload availability.

7.2. Ground And Launch Operations

Ground and launch operations are conducted in three major phases:

a. Launch Vehicle Integration — Assembly and test of the Minotaur I vehicle
b. Payload Processing/Integration — Receipt and checkout of the satellite payload, followed by integration with Minotaur I and verification of interfaces
c. Launch Operations — Includes transport of the upper stack to the launch pad, final integration, checkout, arming and launch.

7.2.1. Launch Vehicle Integration

7.2.1.1. Planning and Documentation

Minotaur I integration and test activities are controlled by a comprehensive set of Work Packages (WPs), that describe and document every aspect of integrating and testing Minotaur I and its payload. Mission-specific work packages are created for mission-unique or payload-specific procedures. Any discrepancies encountered are recorded on a Discrepancy Report and dispositioned as required. All activities are in accordance with Orbital's ISO 9001 certification.

7.2.1.2. Vehicle Integration and Test Activities

The major vehicle components and subassemblies that comprise the Minotaur I Upper Stack Assembly, including the Stage 3 and Stage 4 Orion motors, are delivered to Orbital's VAB located at VAFB, CA. There, the vehicle is horizontally integrated prior to the arrival of the payload. Integration is performed at a convenient working height, which allows relatively easy access for component installation, inspection and test.

The integration and test process ensures that all vehicle components and subsystems are thoroughly tested. Since the Minuteman motors are not available at the VAB, a high fidelity simulator consisting of actual Minuteman components is used.

7.2.1.2.1. Flight Simulation Tests

Flight Simulation Tests use the actual flight software and simulate a "fly to orbit" scenario using simulated Inertial Navigation System (INS) data. The Flight Simulation is repeated after each major change in vehicle configuration (i.e., Flight Simulation #2 after stage mate, Flight Simulation #3 with payload electrically connected (if required) and Flight Simulation #4 after the payload is mechanically integrated). After each test, a complete review of the data is undertaken prior to proceeding. The payload nominally participates in Flight Simulation #3 and #4.

7.2.2. Payload Processing/Integration

Payloads normally undergo initial checkout and processing at Air Force or commercial facilities at VAFB. The payload is then sent to the VAB for integration with the Minotaur I upper stack. After arrival at the VAB, the payload completes its own independent verification and checkout prior to beginning the integration process with Minotaur I. Following completion of Minotaur I and payload testing, the payload will be enclosed inside the fairing. The required payload environments will be maintained inside the fairing until launch. Any payload specific hazardous procedures should be coordinated through Orbital to the launch range no later than 120 days prior to first use (draft) and 30 days prior to first use (final).

7.2.2.1. Payload to Minotaur I Integration

The integrated launch processing activities are designed to simplify final launch processing while providing a comprehensive verification of the payload interface. The systems integration and test sequence is engineered to ensure all interfaces are verified.

7.2.2.2. Pre-Mate Interface Testing

If required, the electrical interface between Minotaur I and the payload is verified using a mission unique Interface Verification Test (IVT) to jointly verify that the proper function of the electrical connections and commands. These tests, customized for each mission, typically check bonding, electrical compatibility, communications, discrete commands and any off nominal modes of the payload. After completing the IVT, a Flight Simulation (Flight Sim #3) is performed with the payload electrically - but not mechanically - connected to Minotaur I to demonstrate the full sequence of events in a simulated flight scenario. Once Flight Sim #3 is successfully completed, the payload is mechanically mated to the launch vehicle. For payloads with simplified or no electrical interfaces to Minotaur I, it may be acceptable to proceed to payload mate immediately after the IVT. For pre-mate verification of the mechanical interface, the separation system can also be made available before final payload preparations.

7.2.2.3. Payload Mating and Verification

Following the completion of Flight Sim #3, the jumpers between the payload and Minotaur I are removed. Once the payload aft end closeouts are completed, the payload will be both mechanically and electrically mated to the Minotaur I. Following mate, the flight vehicle is ready for the final integrated systems test, Flight Simulation #4.

7.2.2.4. Final Processing and Fairing Closeout

After successful completion of Flight Simulation #4, all consumables are topped off and ordnance is connected. Similar payload operations may occur at this time. Once consumables are topped off, final vehicle / payload closeout is performed and the fairing is installed. The payload will coordinate with OSP access to the payload from payload mate until final closeout before launch.

7.2.2.5. Payload Propellant Loading

Payloads utilizing integral propulsion systems with propellants such as hydrazine can be loaded and secured through coordinated Orbital and contractor arrangements for use of the propellant loading facilities in the VAB or other non-Orbital facility. This is a non-standard service.

7.2.2.6. Final Vehicle Integration and Test

Due to operational constraints, the Lower Stack Assembly, consisting of the Minuteman motors, is processed by the Air Force at a separate facility. After testing by Orbital, it is delivered directly to the launch pad to await the arrival of the upper stack. After the vehicle is fully stacked at the pad, final tests are completed to verify vehicle integrity and all interfaces to the range are exercised.

A range interface test is performed to verify all the RF systems. A third mission simulation test is then performed on the integrated system. A launch dress rehearsal is performed without the vehicle powered to train and certify the launch team. Another dress rehearsal is then performed with all flight systems powered to certify the vehicle and launch procedures.

The flight termination system batteries are activated, conditioned, and installed in the vehicle and an end-to-end FTS test is performed to certify the FTS system. The ACS and separation systems are pressurized to final flight pressure, final vehicle preparations are accomplished and a safe and arm verification test is performed to assure proper S&A and AD switch operation. The vehicle and launch team are then ready for the final countdown and launch.

7.3. Launch Operations

7.3.1. Launch Control Organization

The Launch Control Organization is split into two groups: the Management group and the Technical group. The Management group consists of senior range personnel and Mission Directors/Managers for the launch vehicle and payload. The Technical Group consists of the personnel responsible for the execution of the launch operation and data review/assessment for the Payload, the Launch Vehicle and the Range. The Payload's members of the technical group are engineers who provide technical representation in the control center. The Launch Vehicle's members of the technical group are engineers who prepare the Minotaur I for flight, review and assess data that is displayed in the Launch Control Room (LCR) and provide technical representation in the LCR and in the Launch Operations Control Center (LOCC). The Range's members of the technical group are personnel that maintain and monitor the voice and data equipment, tracking facilities and all assets involved with RF communications with the launch vehicle. In addition, the Range provides personnel responsible for the Flight Termination System monitoring and commanding.

8. OPTIONAL ENHANCED CAPABILITIES

The OSP launch service is structured to provide a baseline vehicle configuration which is then augmented with optional enhancements to meet the unique needs of individual payloads. The baseline vehicle capabilities are defined in the previous sections and the optional enhanced capabilities are defined below.

8.1. Mechanical Interface and Separation System Enhancements

8.1.1. Separation Systems

Three flight qualified optional separation systems are available, depending on payload interface and size. The 38 in (97 cm) separable payload interface is shown in Figure 8-1; the 23 in (59 cm) separable payload interface is shown in Figure 8-2; the 17 in (43 cm) separable payload interface is shown in Figure 8-3. Each of these three systems are based on a Marmon band design.

The separation ring to which the payload attaches is supplied with through holes and the separation system is mated to the spacecraft during processing at the VAB. The weight of hardware separated with the payload is approximately 8.7 lbm (4.0 kg) for the 38 in (97 cm) system, 6.0 lbm (2.7 kg) for the 23 in (59 cm) system, and 4.7 lbm (2.1 kg) for the 17 in (43 cm) system. Orbital-provided attachment bolts to this interface can be inserted from either the launch vehicle or the payload side of the interface (NAS630xU, dash number based on payload flange thickness). The weight of the bolts, nuts, and washers connecting the separation system to the payload is allocated to the separation system and included in the payload mass.

At the time of separation, the payload is ejected by matched push-off springs with sufficient energy to produce the relative separation velocities shown in Figure 8-4. If non-standard separation velocities are needed, different springs may be substituted on a mission-specific basis as a non-standard service.

For this system, payload tip-off rates are generally under 4°/sec per axis. Orbital performs a mission-specific tip-off analysis for each payload.

8.1.1.1. Baseline Optional Separation System Electrical Interface

The Baseline Optional separation system is the 38 in. separation system. The electrical interface is located at the Payload/Minotaur I separation plane, reference Figure 8-5. There are two (2), 42 pin separation connectors, 6W7 J1 and 6W7 J2. The Payload Umbilical connectors provides 60 wires from the ground to the spacecraft interface via a dedicated payload umbilical within the vehicle. At the payload interface, cable assembly 6W7 splits the 60 umbilical wires out into two (2) 42 pin connectors, J1 and J2. The length of the internal umbilical is approximately 25 ft. The cabling from the LEV to the launch vehicle is approximately 130 ft. The Launch Equipment Vault is the closest location for operating customer supplied payload EGSE equipment. This umbilical is a dedicated pass through harness, which allows the payload command, control, monitor, and power to be easily configured per user requirements.

It is a Launch Vehicle requirement that the payload provide two (2) separation loopback circuits, one in each separation connector (6W7 J1 and J2) on the payload side of the separation plane. These breakwires are used for positive separation indication telemetry. The Minotaur I interface also contains three (3) loopbacks on the launch vehicle side for payload separation indication for the payload.

The separation plane connectors and their characteristics, are defined in Table 8-1.

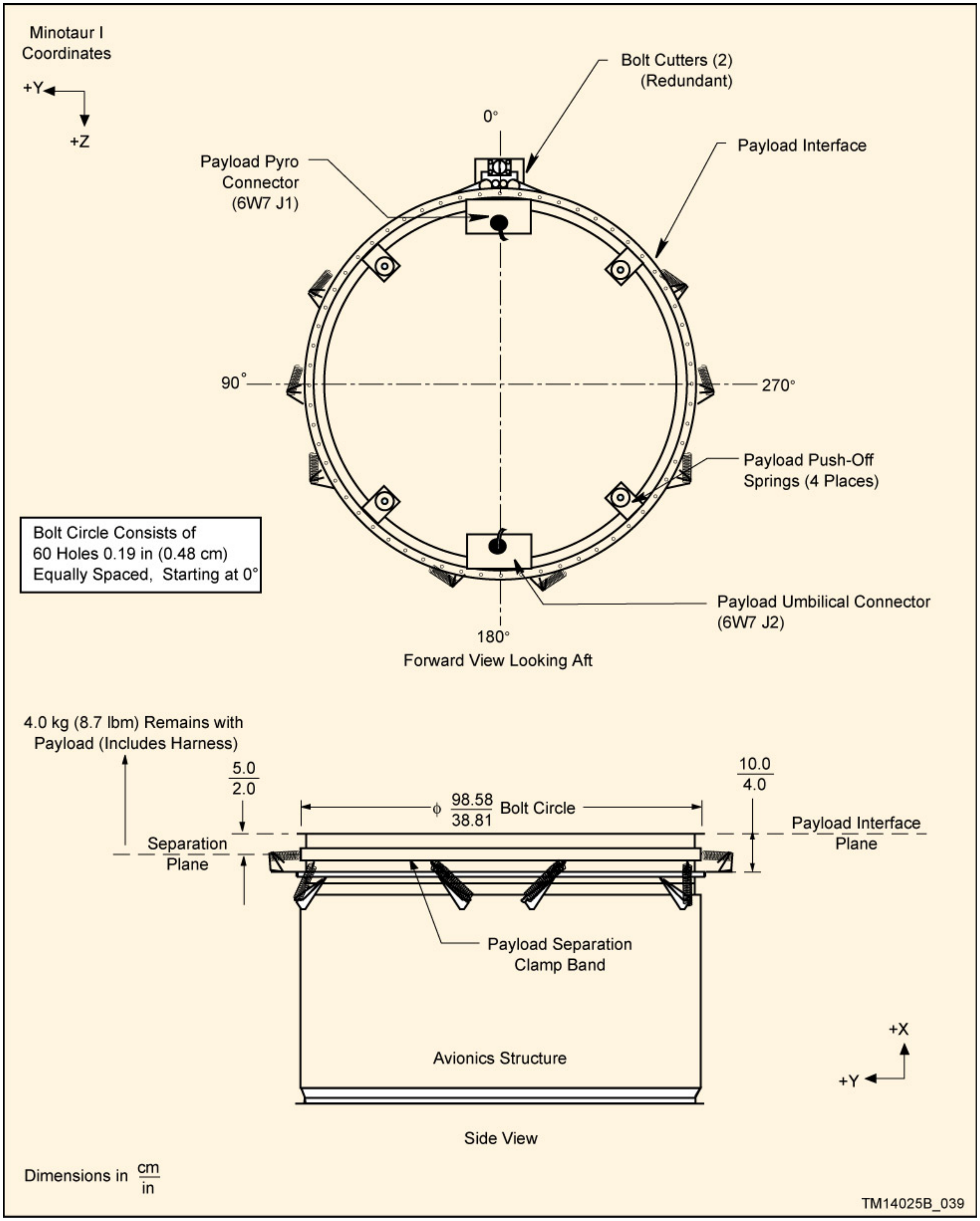


Figure 8-1. 38 in. (97 cm.) Separable Payload Interface

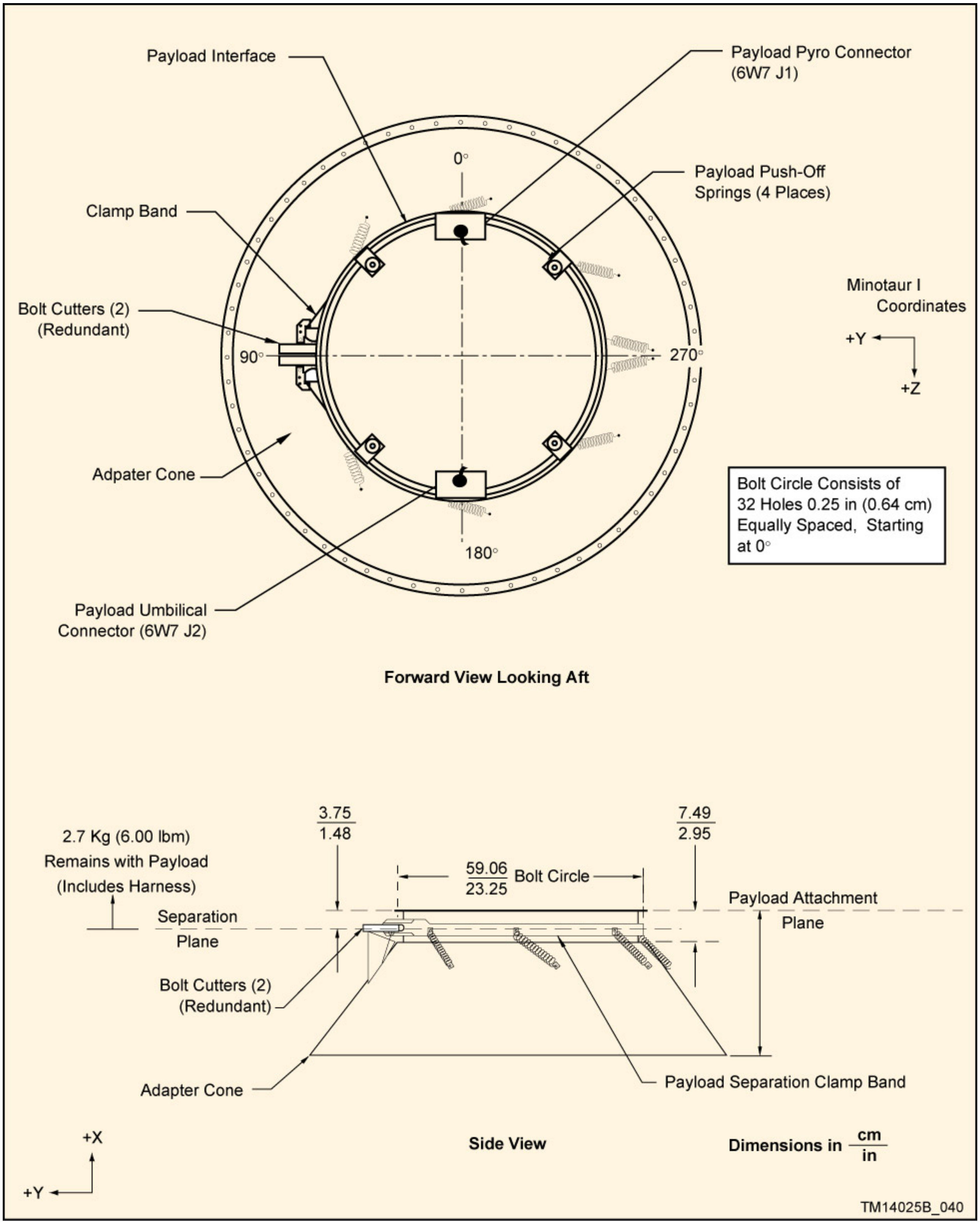


Figure 8-2. 23 in. (59 cm.) Separable Payload Interface

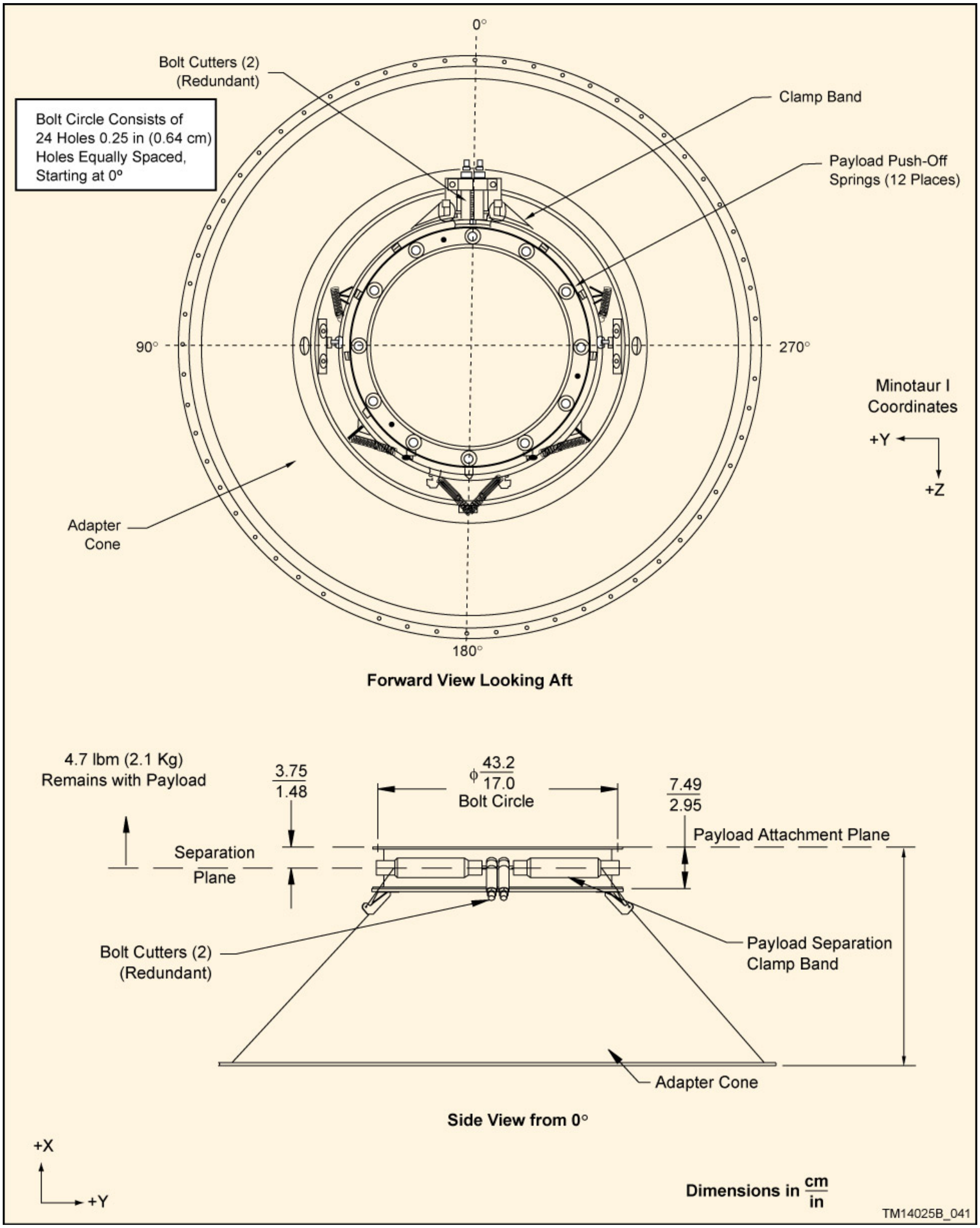


Figure 8-3. 17 in. (43 cm.) Separable Payload Interface

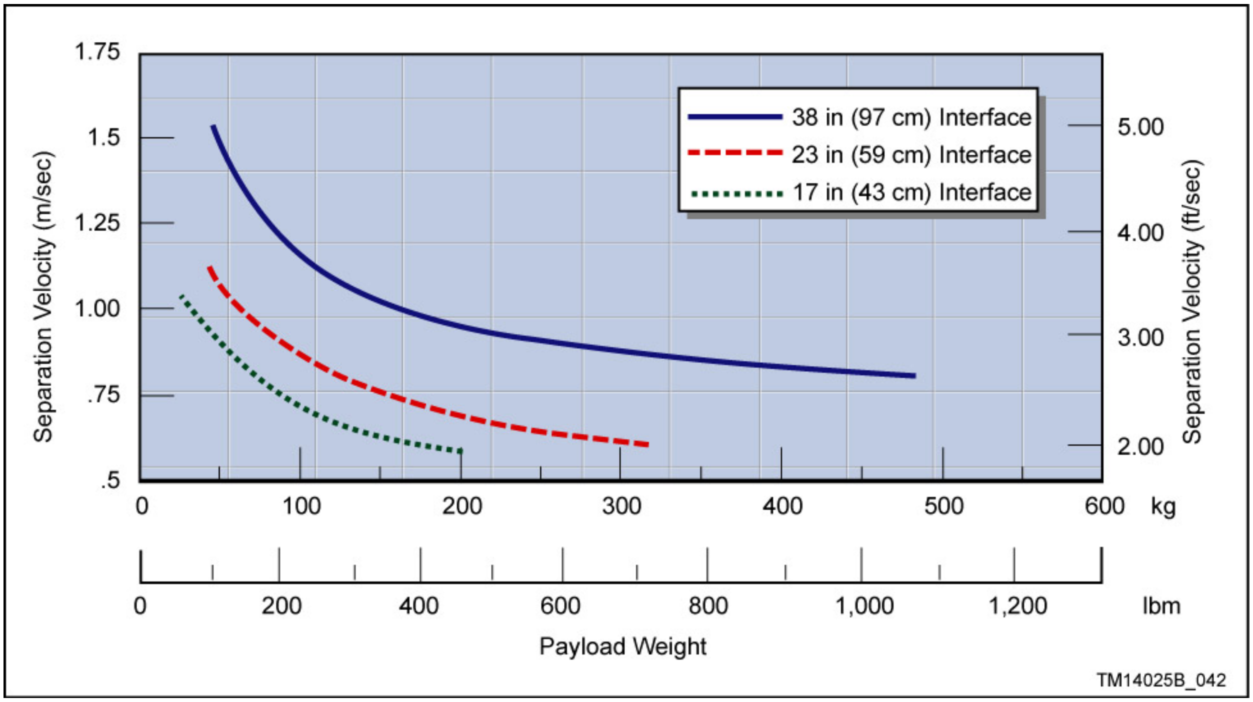


Figure 8-4. Payload Separation Velocities Using the Standard Separation System

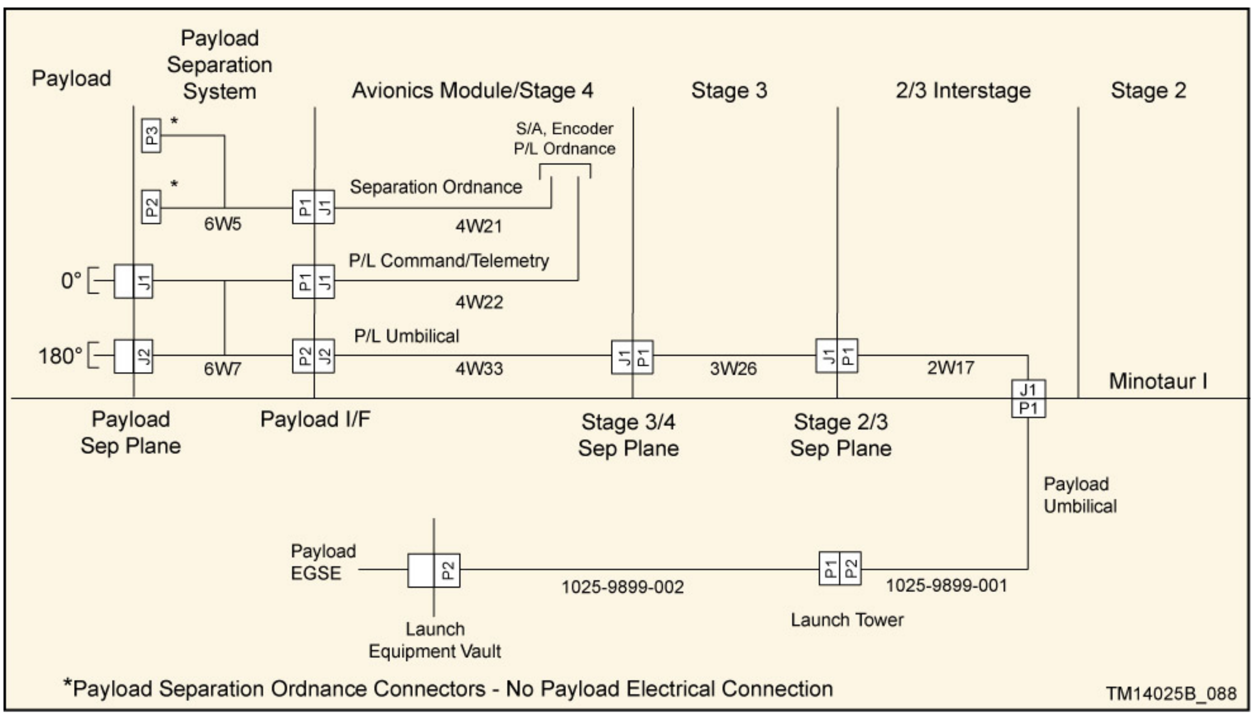


Figure 8-5. Payload Umbilical Interface With An Orbital Separation System Block Diagram

TABLE 8-1. PAYLOAD SEPARATION PLANE INTERFACE CONNECTOR CHARACTERISTICS

	Payload Connectors	**Minotaur I Connectors**
Connector ID	**Defined by Payload Provider**	**6W7 J1**
Connector Part #	MS27474T16F42SN	MS27484T16F42P
Description	Connector, Circular, Jam Nut Recpt	Connector, Circular, Straight Plug
# Pins	42	42
Type	Sockets	Pins
AWG	22	22
Supplied By	Orbital	Orbital
Connector ID	**Defined by Payload Provider**	**6W7 J2**
Connector Part #	MS27474T16F42SB	MS27484T16F42PB
Description	Connector, Circular, Jam Nut Recpt	Connector, Circular, Straight Plug
# Pins	42	42
Type	Sockets	Pins
AWG	22	22
Supplied By	Orbital	Orbital
Connector ID	**Defined by Payload Provider**	**-002 PL Umbi P2**
Connector Part #	D38999/24WJ61SA	D38999/26WJ61PA
Description	Connector, Circular, Jam Nut Recpt	Connector, Circular, Grounded Plug
# Pins	61	61
Type	Sockets	Pins
AWG	20	20
Supplied By	Orbital	Orbital

8.1.1.2. Alternate Separation System Electrical Interfaces

If any of the several alternate separating interfaces is selected as an optional service, the electrical interfaces supplied are located at the Payload/Minotaur I separation plane, just as with the 38 in. separation system. The connector configurations are similar, but not identical. The 23 in. separation system has one 42 pin connector and one 18 pin connector to provide telemetry, command and separation sensing circuitry. The 17 in. separation system has been flown both with and without separating connectors.

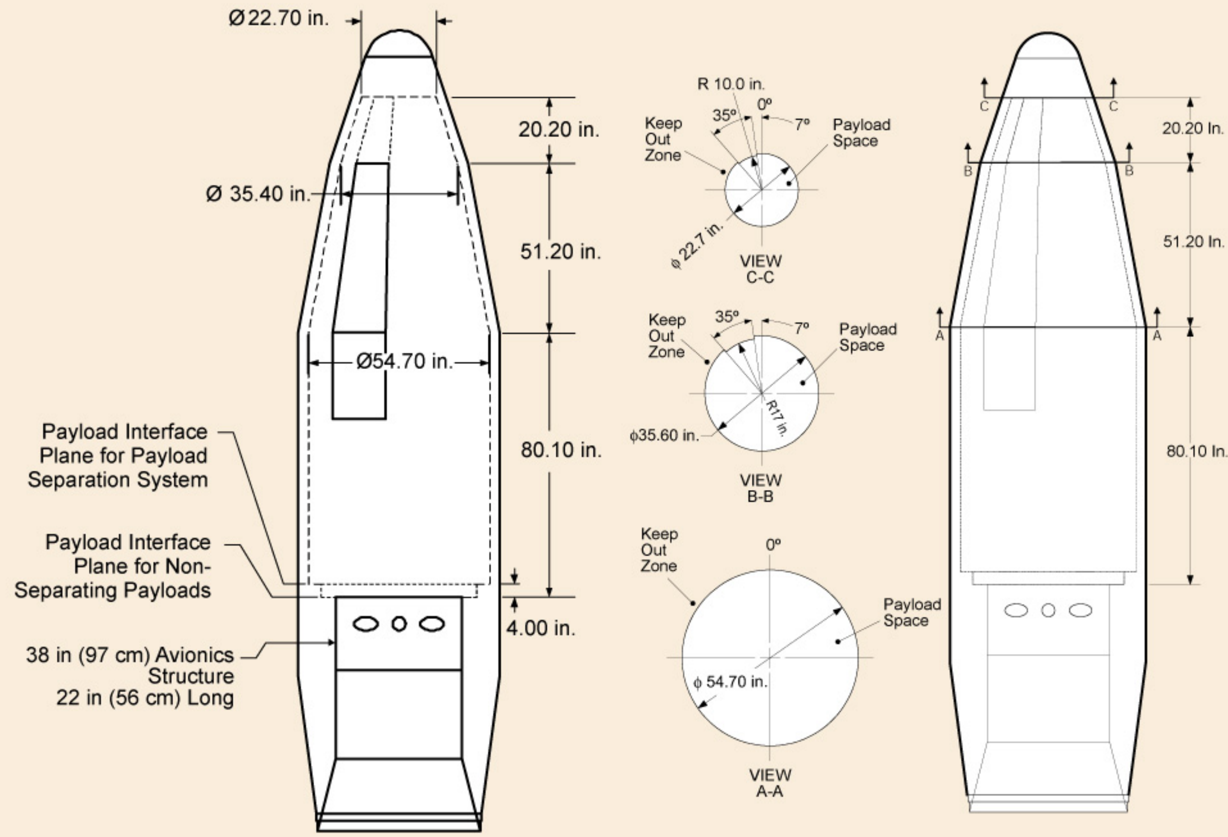
For each payload, Orbital will work with the Payload provider to insure that the payload interface requirements are met with the most reliable and cost effective solution possible. These requirements will be specifically documented in the Payload ICD.

8.1.2. Additional Fairing Access Doors

OSP provides one 8.5 in x 13.0 in (21.6 cm x 33.0 cm), graphite, RF-opaque payload fairing access door (per Section 5.1.2). Additional doors can be provided within the range specified in Section 5.1.2. Other fairing access configurations, such as small circular access panels, can also be provided as negotiated mission-specific enhancements.

8.1.3. Increased Payload Volume

To accommodate payloads larger than those that can be accommodated by the standard Pegasus-based fairing, a larger fairing design is being developed in conjunction with AFRL. A preliminary dynamic envelope is shown in Figure 8-6. A slight performance decrease will be incurred. Performance curves for circular and elliptical orbits of various altitudes and inclinations are detailed in Figures 8-7 through 8-14 for



NOTES:

(1) Payload access doors must be placed between grids. Axial and circumferential grids cannot be cut. One helical rib (diagonal rib) can be cut per door location. The doors must be located above the avionics structure and below the second conic. The fairing has been designed to handle four door locations.

(2) Payload must request any envelope aft of Payload Interface Plane.

(3) If the payload requires Nitrogen cooling, then the payload envelope will be locally reduced by 1 inch along the cooling tube routing.

TM14025A_075

Figure 8-6. Optional 61 in. Diameter Fairing

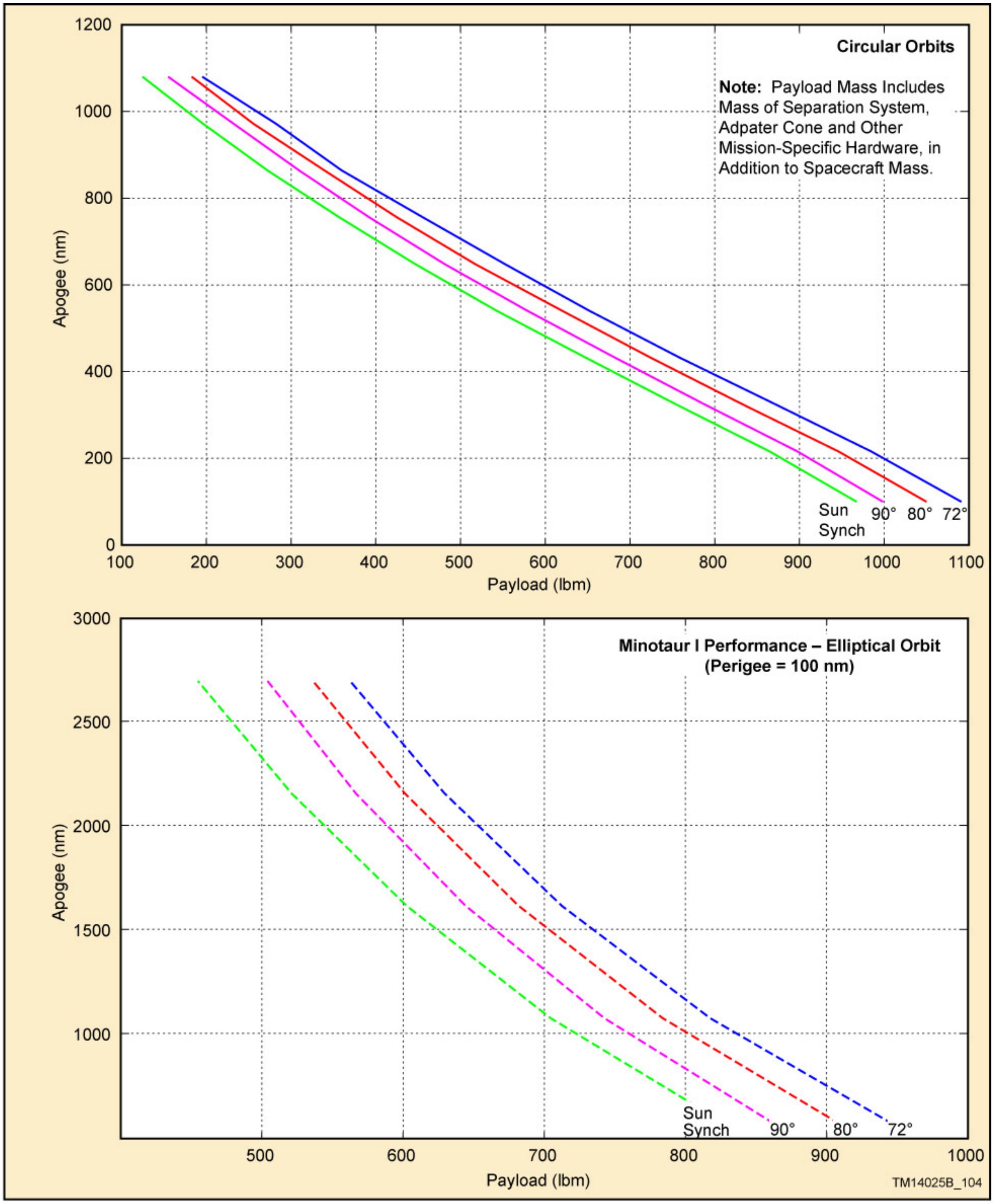


Figure 8-7. Minotaur I 61" Fairing Configuration Performance (English Units) – California Spaceport

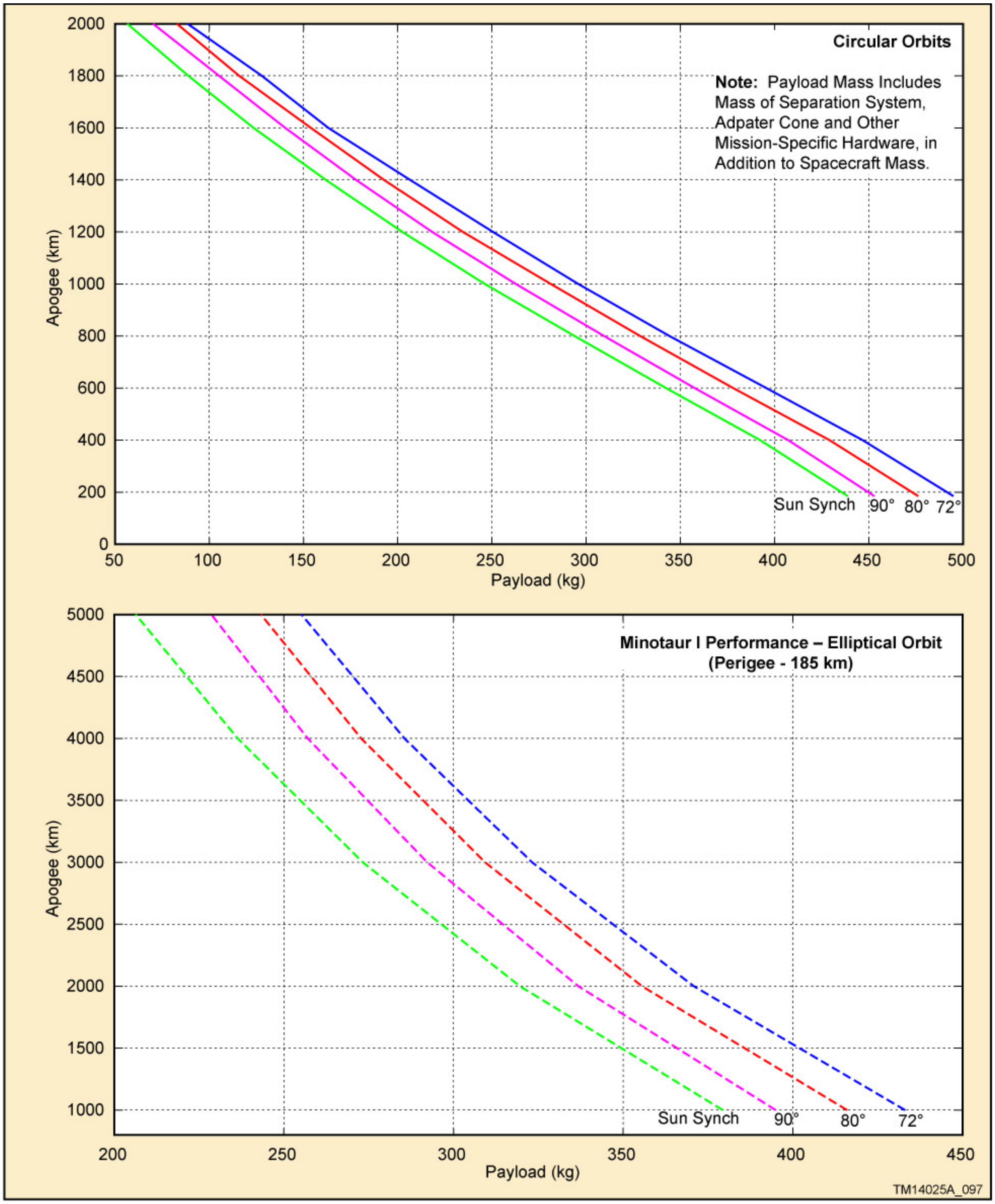


Figure 8-8. Minotaur I 61" Fairing Configuration Performance (Metric Units) – California Spaceport

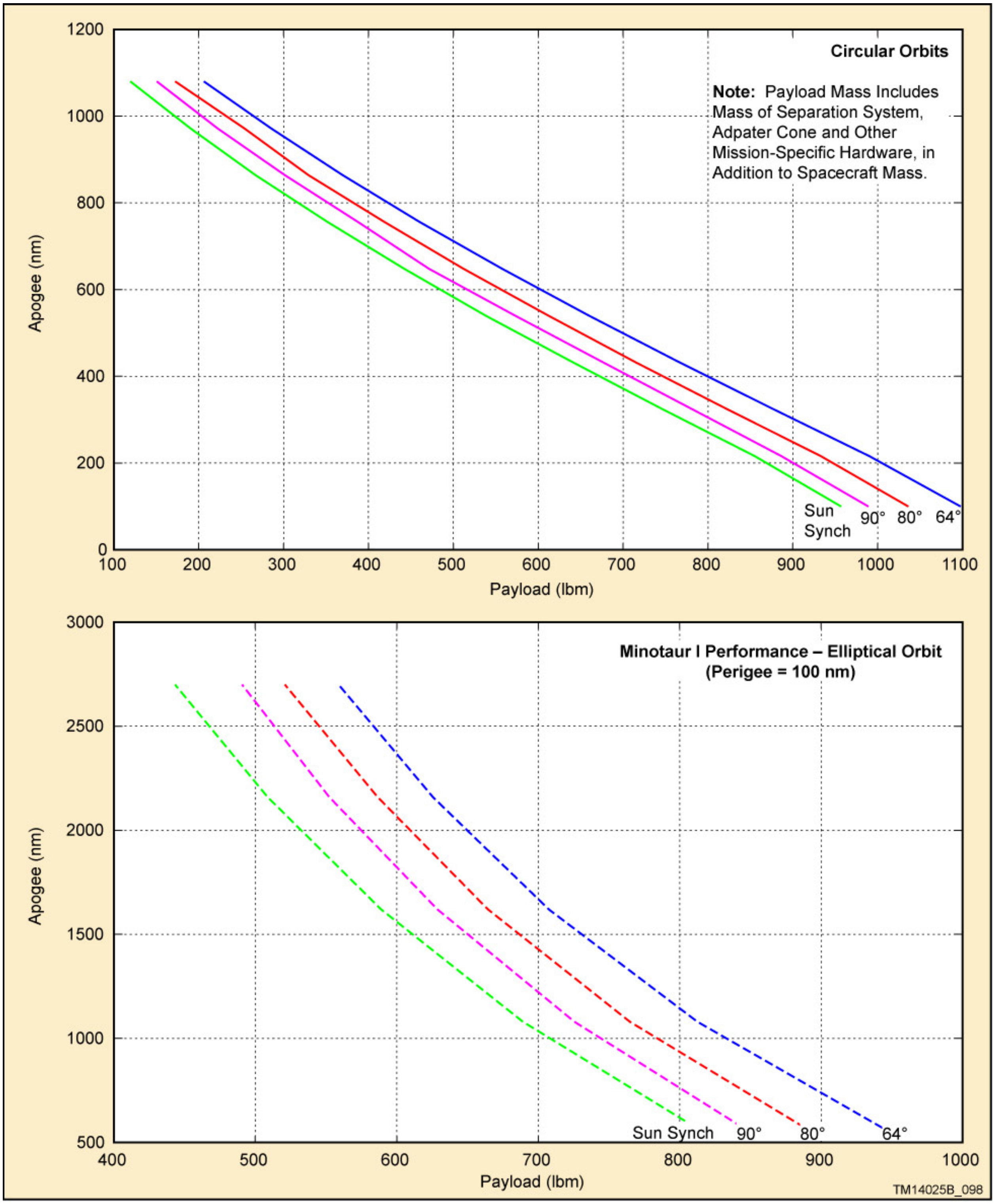


Figure 8-9. Minotaur I 61" Fairing Configuration Performance (English Units) – Kodiak Launch Complex

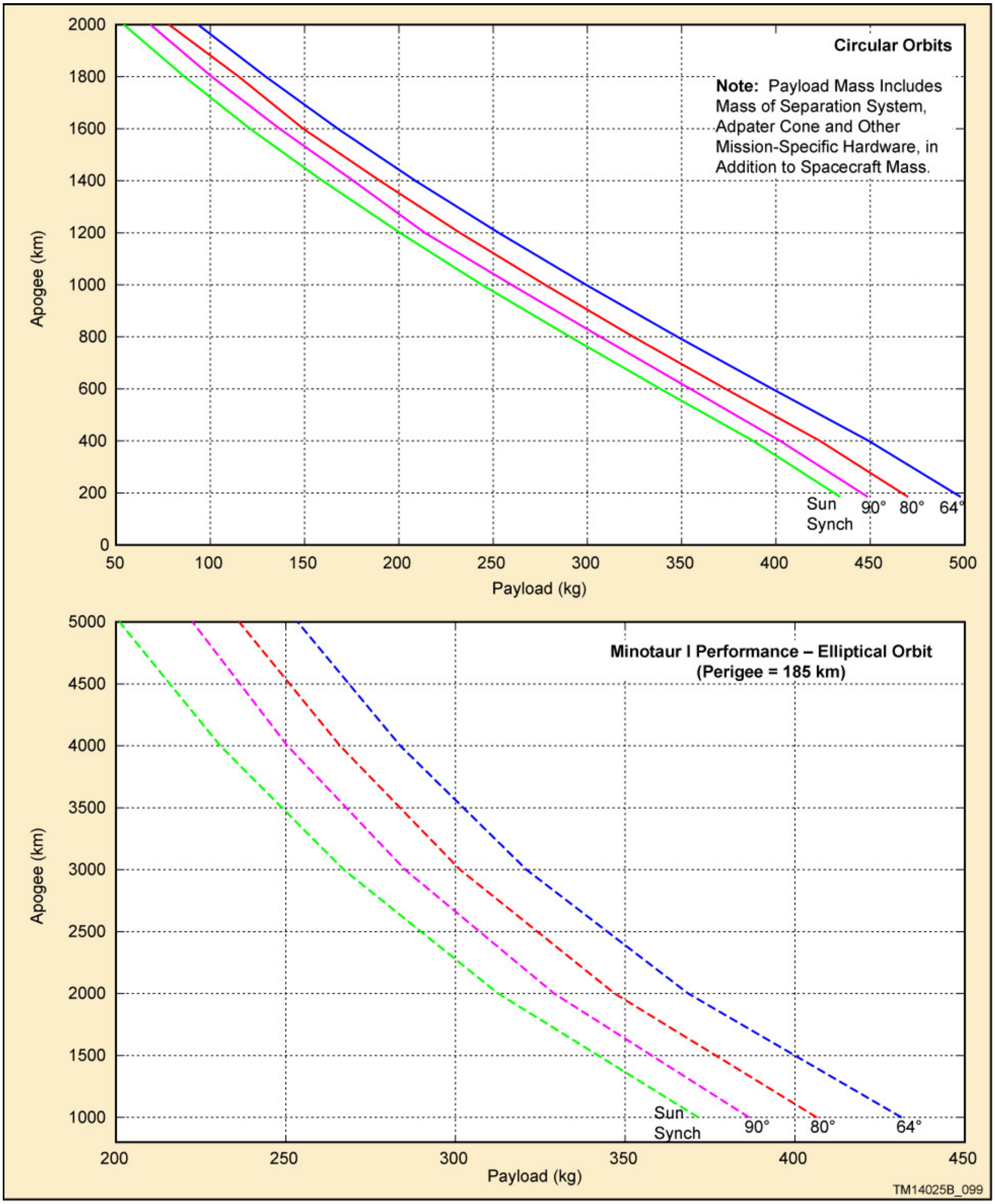


Figure 8-10. Minotaur I 61" Fairing Configuration Performance (Metric Units) – Kodiak Launch Complex

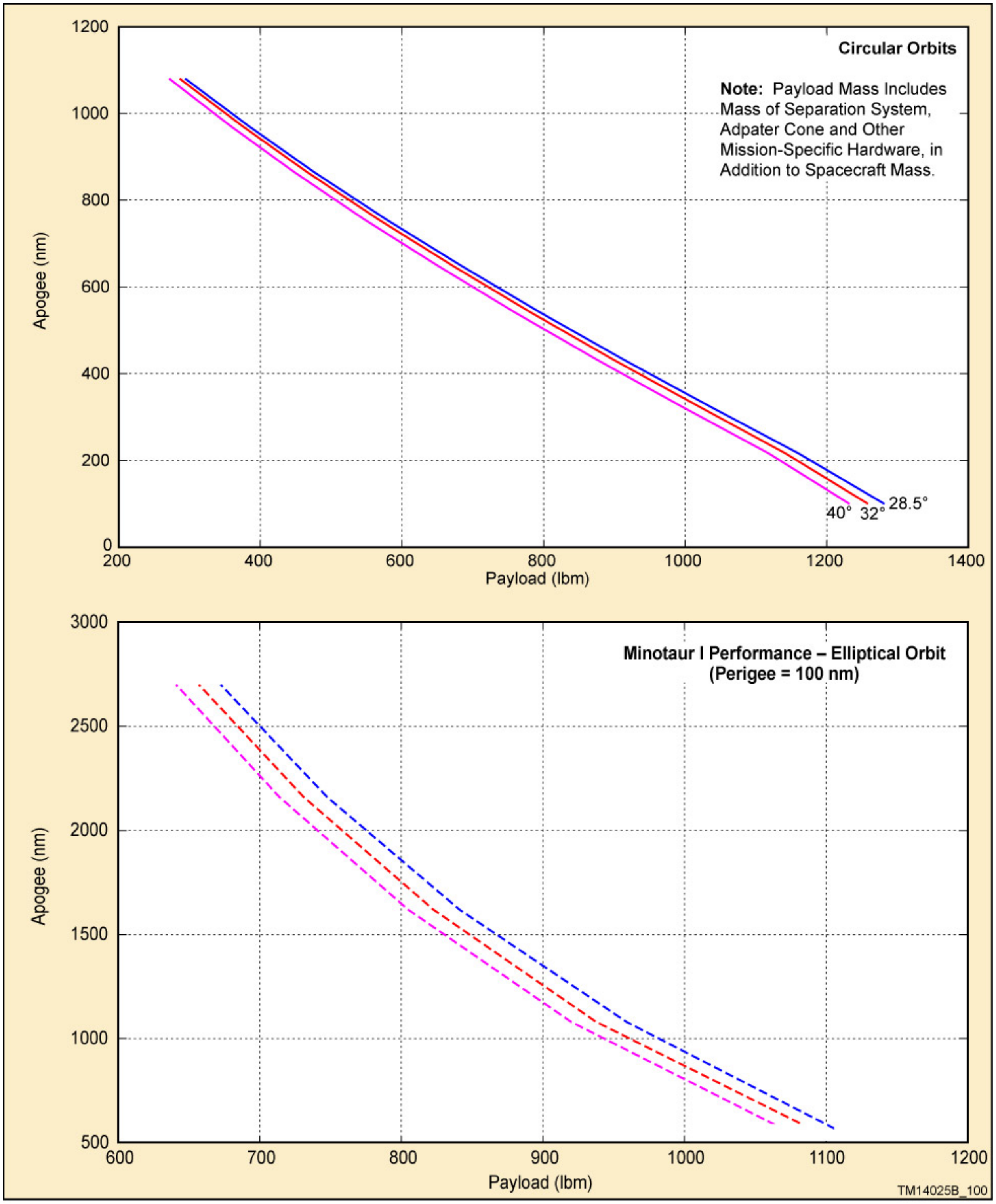


Figure 8-11. Minotaur I 61" Fairing Configuration Performance (English Units) – Spaceport Florida

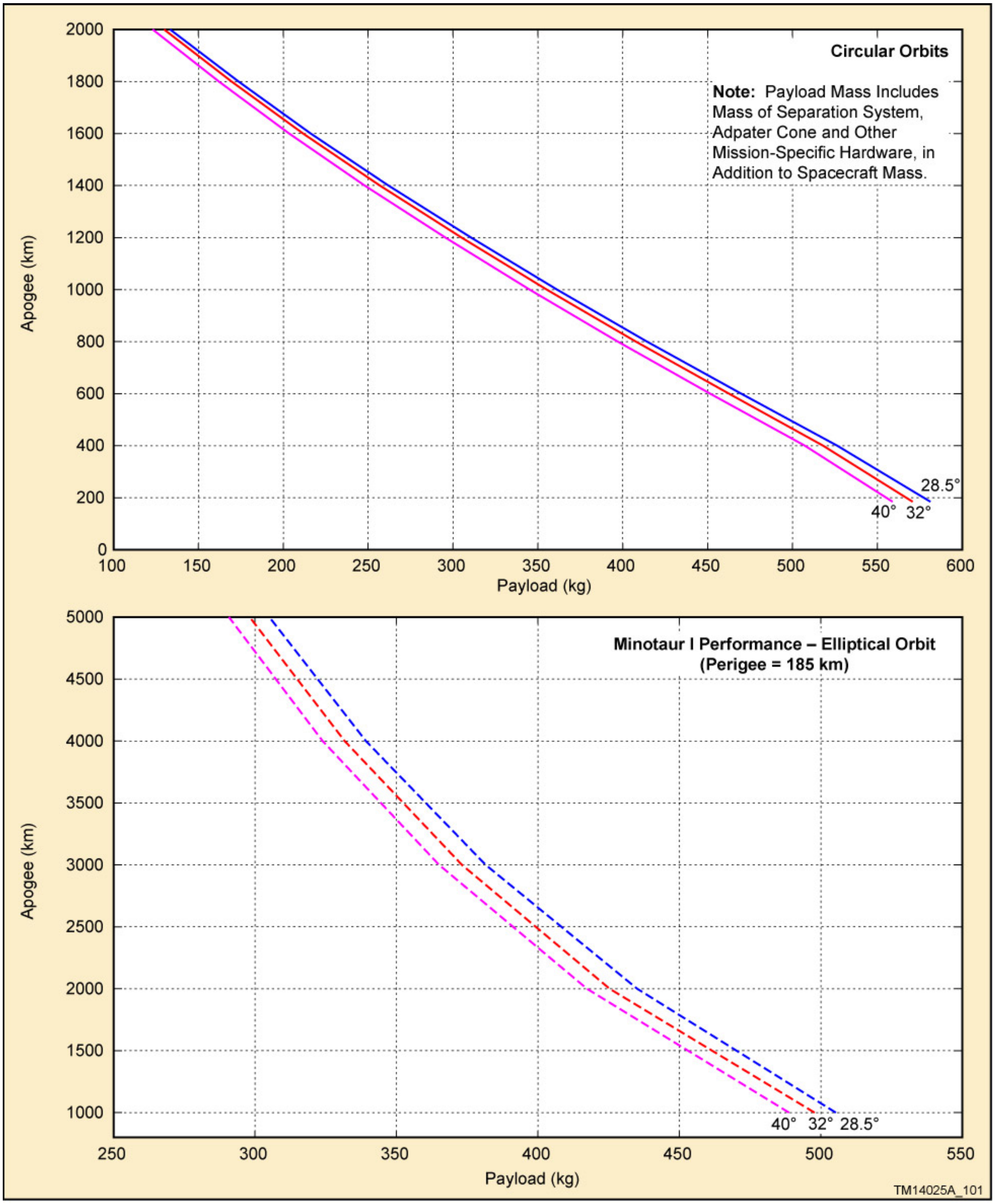


Figure 8-12. Minotaur I 61" Fairing Configuration Performance (Metric Units) – Spaceport Florida

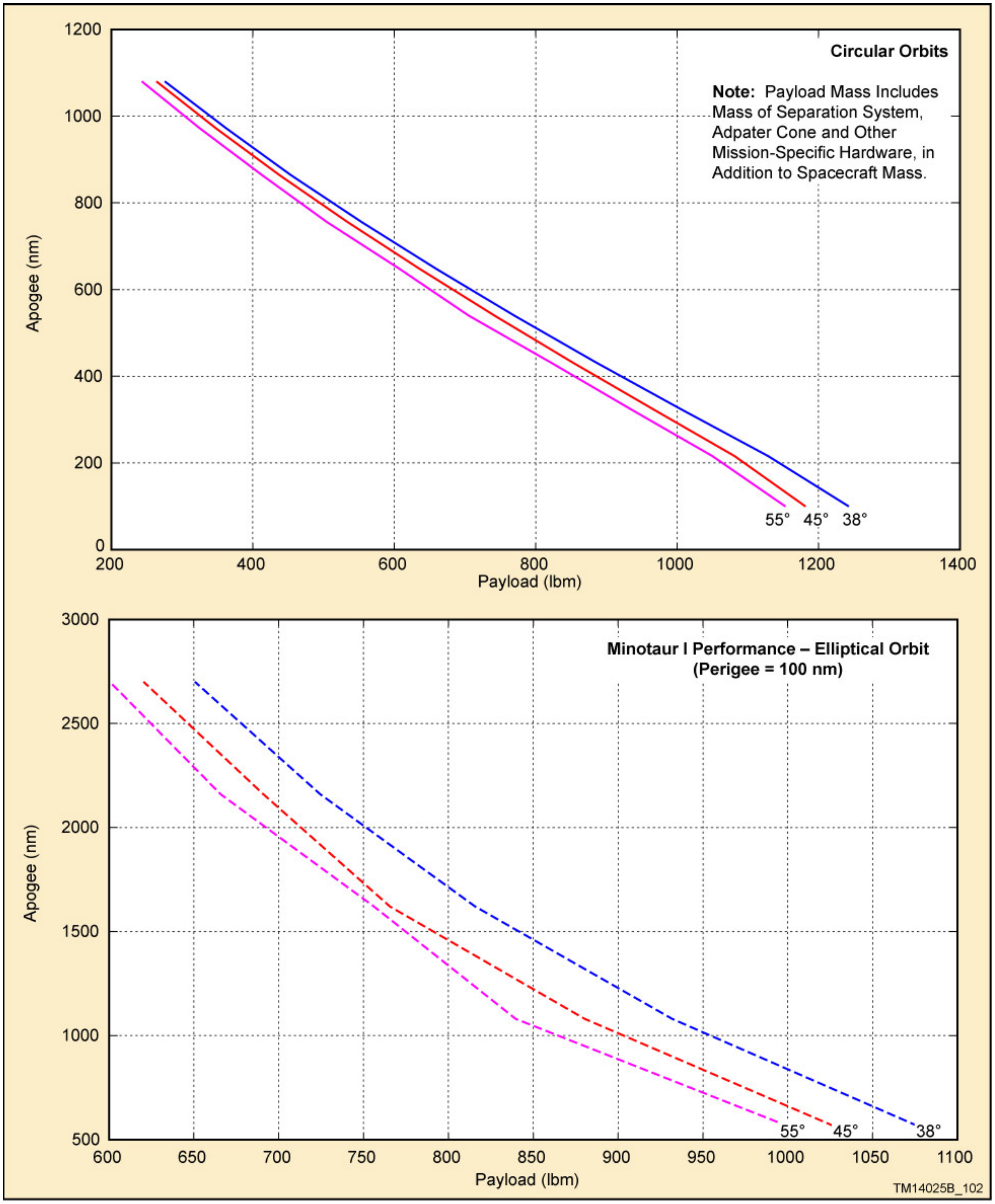


Figure 8-13. Minotaur I 61" Fairing Configuration Performance (English Units) – Virginia Spaceflight Center

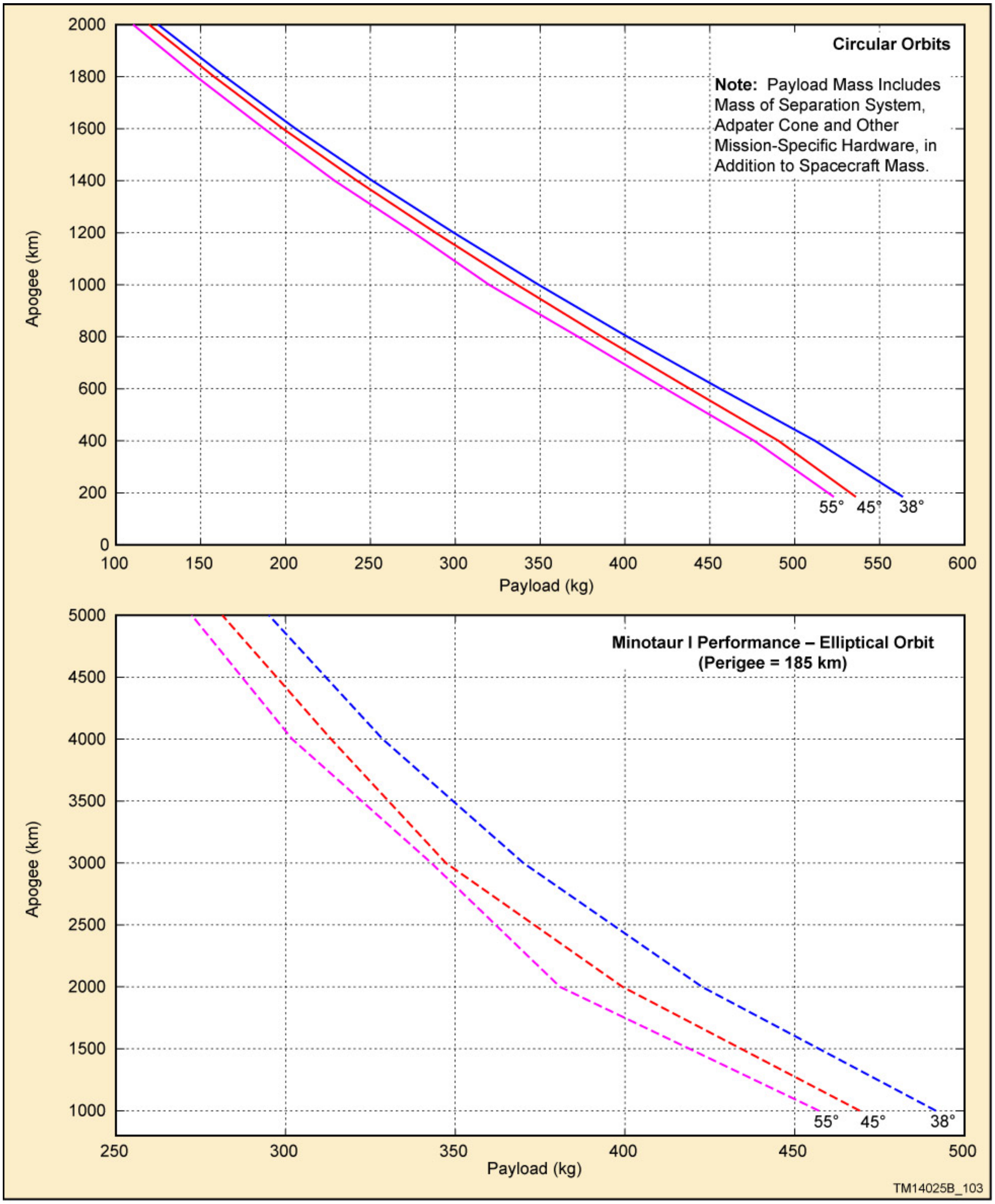


Figure 8-14. Minotaur I 61" Fairing Configuration Performance (Metric Units) – Virginia Spaceflight Center

launches from all four spaceports. These performance curves provide the total mass above the standard, non-separating interface. The mass of any Payload Attach Fitting (PAF), isolation system or separation system is to be accounted for in the payload mass allocation.

8.2. Enhancements

8.2.1. Insertion Accuracy

Enhanced insertion accuracy or support for multiple payload insertion can be provided as an enhanced option utilizing the Hydrazine Auxiliary Propulsion System (HAPS). Orbital insertion accuracy can typically be improved to ±10 nmi (±18.5 km) or better in each apse and ±0.05 deg in inclination. For orbits above 324 nmi (600 km), the HAPS can also increase payload mass by approximately 50 to 250 lbm (22.7 to 113 kg), depending on the orbit. Specific performance capability associated with the HAPS can be provided by contacting the OSP program office. HAPS also permits injection of shared payloads into different orbits. HAPS, which is mounted inside the Avionics Structure, consists of a hydrazine propulsion subsystem and a Stage 4 separation subsystem. After burnout and separation from the Stage 4 motor, the HAPS hydrazine thrusters provide additional velocity and both improved performance and precise orbit injection. The HAPS propulsion subsystem consists of a centrally mounted tank containing approximately 130 lbm (59 kg) of hydrazine, helium pressurization gas, and three fixed, axially pointed thrusters. The hydrazine tank contains an integral bladder which will support multiple restarts.

8.3. Environmental Control Options

8.3.1. Conditioned Air

Conditioned air can be provided within the fairing volume using an Environmental Control System (ECS) via a duct that is retracted in the last minutes prior to launch. Temperature and humidity can be regulated within the limits indicated in Figure 4-10. A filter is installed to provide a Class 100,000 environment, typically. Upon exercise of the Enhanced Cleanliness option, a certified HEPA filter is used in the input duct to assure the necessary low particulate environment (Class 100,000 or Class 10,000).

8.3.2. Nitrogen Purge

OSP can provide gaseous nitrogen purges to the payload after fairing encapsulation until lift-off. The system distribution lines are routed along the inner surface of the fairing. If required for spot cooling of a payload component, Orbital will provide GN2 flow to localized regions in the fairing. The GN2 will meet Grade B specifications, as defined in MIL-P-27401C and can be regulated to at least 5 scfm. The GN2 is on/off controllable in the launch equipment vault and in the launch control room.

The system's regulators are set to a desired flow rate during prelaunch processing. The system cannot be adjusted after the launch pad has been cleared of personnel.

Payload purge requirements must be coordinated with Orbital via the ICD to ensure that the requirements can be achieved.

8.3.3. Enhanced Contamination Control

Understanding that some payloads have requirements for enhanced cleanliness, OSP offers a contamination control option, which is composed of the elements in the following sections (and is also discussed in Section 4.7). Minotaur I customers can also coordinate combinations of the elements listed below to meet the unique needs of their payloads.

8.3.3.1. High Cleanliness Integration Environment (Class 10K or 100K)

With enhanced contamination control, a soft walled clean room can be provided to ensure a FED-STD-209 Class M6.5 (100,000) or Class M5.5 (10,000) environment during all payload processing activities up to fairing encapsulation. The soft walled clean room and anteroom(s) utilize HEPA filter units to filter the air and hydrocarbon content is maintained at 15 ppm or less. The payload organization is responsible for providing the necessary clean room garments for payload staff as well as vehicle staff that need to work inside the clean room.

8.3.3.2. Fairing Surface Cleanliness Options

The inner surface of the fairing and payload cone assemblies can be cleaned to cleanliness criteria which ensures no particulate matter visible with normal vision when inspected from 6 to 18 inches under 100 ft-candle incident light. The same will be true when the surface is illuminated using black light, 3200 to 3800 Angstroms (Visibly Clean Plus Ultraviolet). In addition, Orbital can ensure that all materials used within the encapsulated volume have outgassing characteristics of less than 1.0% TML and less than 0.1% CVCM. Items that don't meet these levels can be masked to ensure they are encapsulated and will have no significant effect on the payload.

8.3.3.3. High Cleanliness Fairing Environment

With the enhanced contamination control option, Orbital provides an ECS from payload encapsulation until just prior to vehicle lift-off. The ECS continuously purges the fairing volume with clean filtered air. Orbital's ECS incorporates a HEPA filter unit to provide FED-STD-209 Class M5.5 (10,000) air. Orbital monitors the supply air for particulate matter via a probe installed upstream of the fairing inlet duct prior to connecting the air source to the payload fairing.

8.4. Enhanced Telemetry Options

OSP can provide mission specific instrumentation and telemetry components to support additional payload or experiment data acquisition requirements. Telemetry options include additional payload-dedicated bandwidth and GPS-based precision navigation data.

8.4.1. Enhanced Telemetry Bandwidth

A second telemetry data stream capable of up to 2 Mbps data rate can be provided. Maximum data rates depend on the mission coverage required and the launch range receiver characteristics and configuration. This capability was successfully demonstrated on the inaugural Minotaur I mission.

8.4.2. Enhanced Telemetry Instrumentation

To support the higher data rate capability in Section 8.4.1, enhanced telemetry instrumentation can be provided. The instrumentation can include strain gauges, temperature sensors, accelerometers, analog data, and digital data configured to mission-specific requirements. This capability was successfully demonstrated on the inaugural Minotaur I mission.

8.4.3. Navigation Data

Precision navigation data using an independent GPS-receiver and telemetry link is available as an enhanced option. This option utilizes the Orbital-developed GPS Position Beacon (GPB) and provides a better than 100 m position accuracy with a nominal 1 Hz data rate. This capability was successfully demonstrated on the inaugural Minotaur I mission.

8.5. Alternate Launch Locations

As an alternative for missions requiring high inclination orbits (60° to 120°), launches can be conducted from Kodiak Island, AK. The launch facility at Kodiak Island, operated by the Alaska Aerospace Development Corporation (AADC) has been used for both orbital and suborbital launches.

For Easterly launch azimuths to achieve orbital inclinations between 28.5° and 60°, Minotaur I can be launched from facilities at Cape Canaveral, FL or Wallops Island, VA. Launches from Florida will nominally use the Florida

Spaceport Authority (FLSA) launch facilities at LC-46 on CCAFS, Cape Canaveral, FL. These will be typically for inclinations from 28.5° to 40°, although inclinations above 35° may have reduced performance due to the need for a trajectory dogleg. The Virginia Spaceflight Center facilities at the WFF may be used for inclinations from 30° to 60°. Southeasterly launches from WFF offer fewer overflight concerns than CCAFS.

Inclinations below 35° and above 55° are feasible, albeit with doglegs and altitude constraints due to stage impact considerations.

9. SHARED LAUNCH ACCOMMODATIONS

Minotaur I is uniquely capable of providing launches of multiple satellite payloads, leveraging SMC Det 12/RP and Orbital's extensive experience in integrating and launching multiple payloads. In addition to the five satellites successfully separated on the inaugural Minotaur I JAWSAT mission, multiple spacecraft configurations have been flown on many of Orbital's Pegasus and Taurus missions to date.

Two technical approaches are available for accommodating multiple payloads. These design approaches are:

Load-Bearing Spacecraft — aft spacecraft designed to provide the structural load path between the forward payload and the launch vehicle, maximizing utilization of available mass performance and payload fairing volume

Non Load-Bearing Spacecraft — aft spacecraft whose design cannot provide the necessary structural load path for the forward payload

9.1. Load-Bearing Spacecraft

Providing a load-bearing aft payload maximizes use of available volume and mass. The available mass for the aft payload is determined by the Minotaur I performance capability to orbit less the forward payload and attach hardware mass. All remaining mission performance, excluding a stack margin, is available to the aft payload. The load-bearing spacecraft interfaces directly to the avionics assembly interface and the forward payload via pre-determined interfaces. These interfaces include standard Orbital separation systems and pass-through electrical connectors to service the forward payload. Figure 9-1 illustrates this approach.

Two approaches may be taken for load-bearing spacecraft. The first approach is to use a design developed by other spacecraft suppliers, which must satisfy Minotaur I and forward payload

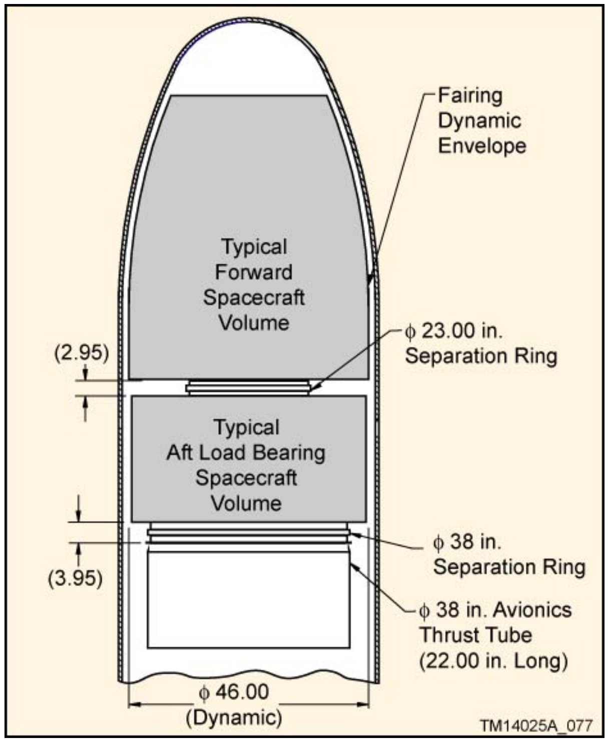


Figure 9-1. Typical Load Bearing Spacecraft Configuration

structural design criteria. The principal requirements levied upon load-bearing spacecraft are those involving mechanical and electrical compatibility with the forward payload. Structural loads from the forward payload during all flight events must be transmitted through the aft payload to the Minotaur I. Orbital will provide minimum structural interface design criteria for shear, bending moment, axial and lateral loads, and stiffness.

The second approach involves the use of an Orbital design using the MicroStar bus, successfully developed and flown for ORBCOMM spacecraft. The MicroStar bus features a circular design with an innovative, low-shock separation system. The spacecraft bus is designed to allow stacking of co-manifested payloads in "slices" within the fairing. The bus design is compact and provides exceptional lateral stiffness.

A variation on the load bearing spacecraft approach was flown on the JAWSAT inaugural mission, in which the primary payload (JAWSAT) was a Multiple Payload Adapter (MPA) from which four small satellites were separated (Figure 9-2). After separating the smaller "piggyback" satellites, the JAWSAT MPA was also separated as an autonomous satellite by utilizing the Orbital 23 in. separation system and adapter cone. An updated concept to provide greater payload options and primary payload volume (by mounting directly to the avionics assembly) is shown in Figure 9-3.

Integrated coupled loads analyses will be performed with test verified math models provided by the payload. These analyses are required to verify the fundamental frequency and deflections of the stack for compliance with the Minotaur I requirement of 12 Hz minimum. Design criteria provided by OSP will include "stack" margins to minimize interactive effects associated with potential design changes of each payload. OSP will provide the necessary engineering coordination between the spacecraft and launch vehicle.

Electrical pass-through harnesses will also need to be provided by the aft payload along with provisions for connectors and interface verification. The spacecraft supplier will need to provide details of the appropriate analyses and tests to OSP to verify adequacy of margins and show that there is no impact to the forward spacecraft or the launch vehicle.

9.2. Non Load-Bearing Spacecraft

For aft spacecraft that are not designed for withstanding and transmitting structural loads from the forward payload, several options are possible including AFRL and Orbital design concepts, as well as the flight-proven Dual Payload Attach Fitting (DPAF).

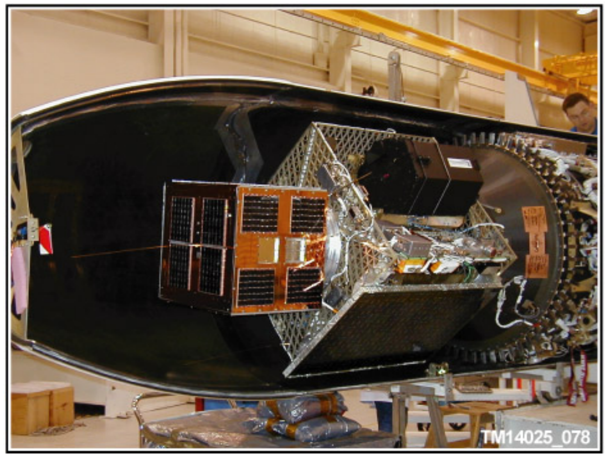


Figure 9-2. JAWSAT Multiple Payload Adapter Load Bearing Spacecraft

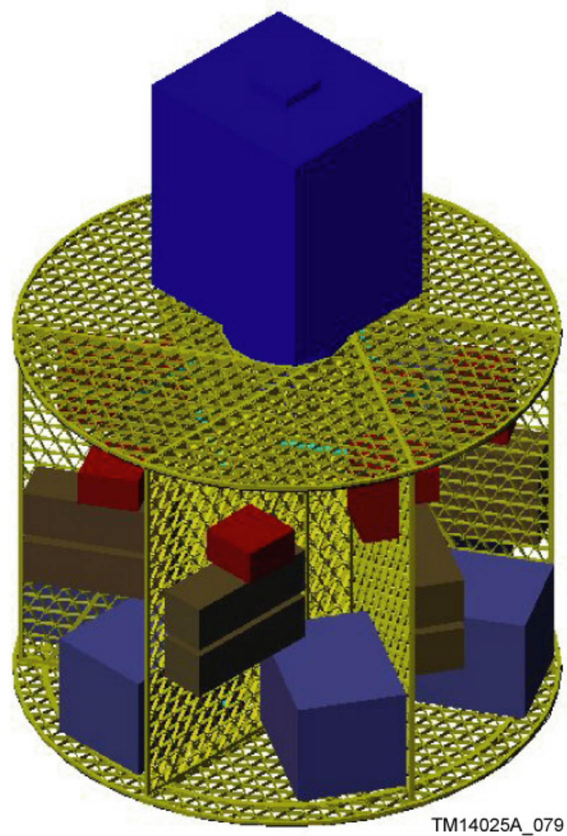


Figure 9-3. Five Bay Multiple Payload Adapter Concept

The DPAF structure (Figure 9-4) is an all graphite structure which provides independent load paths for each satellite. The worst-case "design payload" for the DPAF is a 425 lbm (193 kg) spacecraft with a 20 in. (51 cm) center of mass offset and first lateral frequency of typically 12 Hz. The DPAF is designed to accommodate this "design payload" at both the forward and aft locations, although the combined mass of the two payloads cannot exceed Minotaur I capabilities. The upper spacecraft loads are transmitted around the lower spacecraft via the DPAF structure, thus avoiding any structural interface between the two payloads.

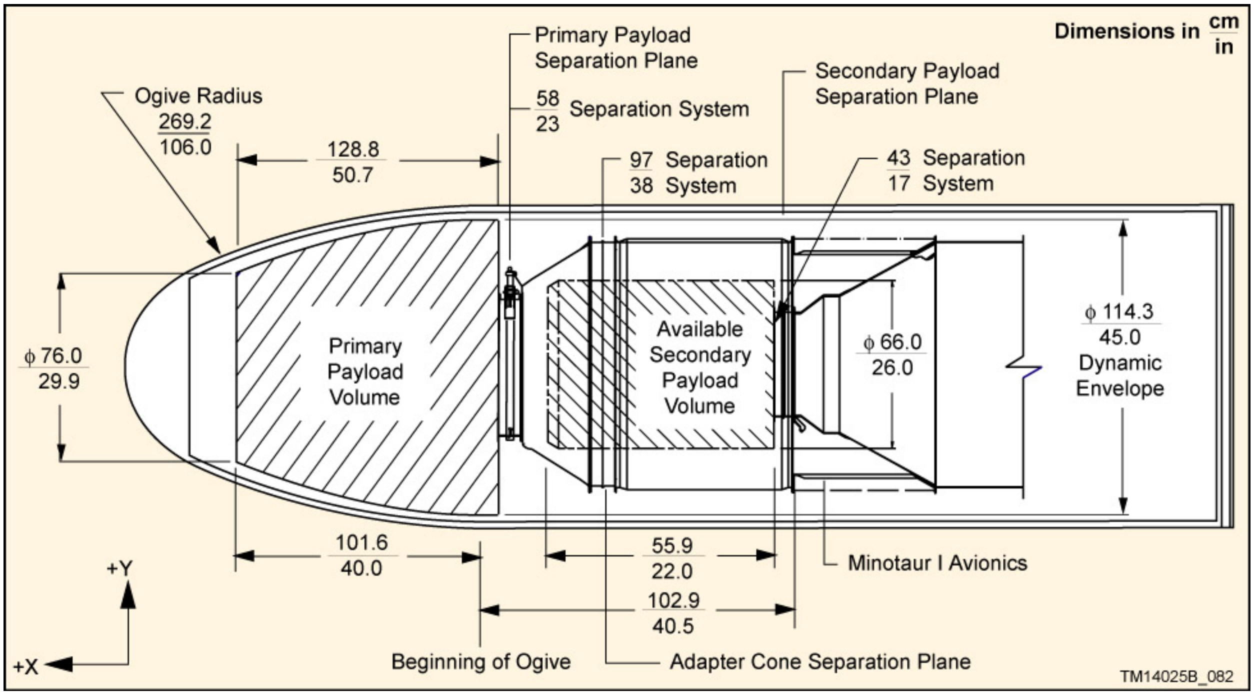


Figure 9-4. Dual Payload Attach Fitting Configuration

APPENDIX A

PAYLOAD QUESTIONNAIRE

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SATELLITE IDENTIFICATION	
FULL NAME:	
ACRONYM:	
OWNER/OPERATOR:	
INTEGRATOR(s):	

ORBIT INSERTION REQUIREMENTS*		
SPHEROID	❑ Standard (WGS-84, R_e = 6378.137 km) ❑ Other:	
ALTITUDE	Insertion Apse: ❑ nmi ____________ ± ________ ❑ km	Opposite Apse: ❑ nmi ____________ ± ________ ❑ km
or...	Semi-Major Axis: ❑ nmi ____________ ± ________ ❑ km	Eccentricity: __________ ≤ e ≤ __________
INCLINATION	____________ ± deg	
ORIENTATION	Argument of Perigee: ____________ ± ________ deg	Longitude of Ascending Node (LAN): ____________ ± ________ deg
	Right Ascension of Ascending Node (RAAN): ____________ ± ________ deg	...for Launch Date: ____________

* Note: Mean orbital elements

LAUNCH WINDOW REQUIREMENTS
NOMINAL LAUNCH DATE:
OTHER CONSTRAINTS (if not already implicit from LAN or RAAN requirements, e.g., solar beta angle, eclipse time constraints, early on-orbit ops, etc):

<table>
<tr><th colspan="4">GROUND SUPPORT EQUIPMENT</th></tr>
<tr><td colspan="4">Describe any additional control facilities (other than the baseline Support Equipment Building (SEB) and Launch Equipment Vault (LEV)) which the satellite intends to use:</td></tr>
<tr><td rowspan="4">SEB</td><td colspan="3">Describe (in the table below) Satellite EGSE to be located in the LSV.
[Note: Space limitations exist in the SEB, 350 ft umbilical cable length to spacecraft typical]</td></tr>
<tr><td><u>Equipment Name / Type</u></td><td><u>Approximate Size (LxWxH)</u></td><td><u>Power Requirements</u></td></tr>
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<tr><td colspan="3">Is UPS required for equipment in the SEB? Yes / No
Is Phone/Fax connection required in the SEB? Yes / No Circle: Phone / FAX</td></tr>
<tr><td rowspan="4">LEV</td><td colspan="3">Describe (in the table below) Satellite EGSE to be located in the LEV.
[Note: Space limitations exist in the SEB, 150 ft umbilical cable length to spacecraft typical]</td></tr>
<tr><td><u>Equipment Name / Type</u></td><td><u>Approximate Size (LxWxH)</u></td><td><u>Power Requirements</u></td></tr>
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<tr><td colspan="3">Is UPS required for equipment in the LEV? Yes / No
Is Ethernet connection between SEB and LEV required? Yes / No</td></tr>
</table>

EARLY ON-ORBIT OPERATIONS
Briefly describe the satellite early on-orbit operations, e.g., event triggers (separation sense, sun acquisition, etc), array deployment(s), spin ups/downs, etc:

SATELLITE SEPARATION REQUIREMENTS	
ACCELERATION	Longitudinal: = __________ g's Lateral: = __________ g's
VELOCITY	Relative Separation Velocity Constraints:
ANGULAR RATES (pre-separation)	Longitudinal: __________ ± __________ deg/sec Pitch: __________ ± __________ deg/sec Yaw: __________ ± __________ deg/sec
ANGULAR RATES (post-separation)	Longitudinal: __________ ± __________ deg/sec Pitch: __________ ± __________ deg/sec Yaw: __________ ± __________ deg/sec
ATTITUDE (at deployment)	Describe Pointing Requirements Including Tolerances:
SPIN UP	Longitudinal Spin Rate: __________ ± __________ deg/sec
OTHER	Describe Any Other Separation Requirements:

SPACECRAFT COORDINATE SYSTEM
Describe the Origin and Orientation of the spacecraft reference coordinate system, including its orientation with respect to the launch vehicle (provide illustration if available):

SPACECRAFT PHYSICAL DIMENSIONS		
STOWED CONFIGURATION	Length/Height: ____________ ❑ in ❑ cm Other Pertinent Dimension(s):	Diameter: ____________ ❑ in ❑ cm
	Describe any appendages/antennas/etc which extend beyond the basic satellite envelope:	
ON-ORBIT CONFIGURATION	Describe size and shape:	

If available, provide dimensioned drawings for both stowed and on-orbit configurations.

SPACECRAFT MASS PROPERTIES*				
PRE-SEPARATION			Inertia units: ❑ lb_m-in^2 ❑ kg-m^2	
	Mass: ____________	❑ lb_m ❑ kg	Ixx:____________	
	Xcg: ____________	❑ in ❑ cm		Iyy:____________
			Izz:____________	
	Ycg: ____________	❑ in ❑ cm		Ixy:____________
			Iyz:____________	
	Zcg: ____________	❑ in ❑ cm		Ixz:____________
POST-SEPARATION (non-separating adapter remaining with launch vehicle)			Inertia units: ❑ lb_m-in^2 ❑ kg-m^2	
	Mass: ____________	❑ lb_m ❑ kg	Ixx:____________	
	Xcg: ____________	❑ in ❑ cm		Iyy:____________
			Izz:____________	
	Ycg: ____________	❑ in ❑ cm		Ixy:____________
			Iyz:____________	
	Zcg: ____________	❑ in ❑ cm		Ixz:____________

* Stowed configuration, spacecraft coordinate frame

ASCENT TRAJECTORY REQUIREMENTS	
Free Molecular Heating at Fairing Separation: (Standard Service: = 360 Btu/ft^2/hr)	❑ Btu/ft^2/hr FMH = ________________ ❑ W/m^2
Fairing Internal Wall Temperature (Standard Service: = 200°F)	❑ deg F T = ________________ ❑ deg C
Dynamic Pressure at Fairing Separation: (Standard Service: = 0.01 lb_f/ft^2)	❑ lb_f/ft^2 q = ________________ ❑ N/m^2
Ambient Pressure at Fairing Separation: (Standard Service: = 0.3 psia)	❑ lb_f/in^2 P = ________________ ❑ N/m^2
Maximum Pressure Decay During Ascent: (Standard Service: = 0.6 psia)	❑ lb_f/in^2/sec Δ P = ________________ ❑ N/m^2/sec
Thermal Maneuvers During Coast Periods: (Standard Service: none)	

SPACECRAFT ENVIRONMENTS	
THERMAL DISSIPATION	Spacecraft Thermal Dissipation, Pre-Launch Encapsulated: __________ Watts Approximate Location of Heat Source:
TEMPERATURE	Temperature Limits During Ground/Launch Operations: Max __________ ❑ deg F ❑ deg C Min __________ ❑ deg F ❑ deg C (Standard Service is 55°F to 80°F)
	Component(s) Driving Temperature Constraint: Approximate Location(s):
HUMIDITY	Relative Humidity: Max ________ % Min ________ % ***or***, Dew Point: Max __________ ❑ deg F ❑ deg C Min __________ ❑ deg F ❑ deg C (Standard Service is 37 deg F)
NITROGEN PURGE	Specify Any Nitrogen Purge Requirements, Including Component Description, Location, and Required Flow Rate: (Nitrogen Purge is a Non-Standard Service)
CLEANLINESS	Volumetric Requirements (e.g. Class 100,000): ____________________ Surface Cleanliness (e.g. Visually Clean): ____________________ Other:
LOAD LIMITS	Ground Transportation Load Limits: Axial = _______ g's Lateral = _______ g's

ELECTRICAL INTERFACE			
Bonding Requirements:			
Are Launch Vehicle Supplied Pyro Commands Required?	Yes / No (Standard Service is 10 amps for 100 msec)	If Yes, magnitude: __________ amps for _________ msec	
Are Launch Vehicle Supplied Discrete Commands Required?	Yes / No	If Yes, describe:	
Is Electrical Access to the Satellite Required...	at Launch Site	After Encapsulation? Yes / No	Yes / No
Is Satellite Battery Charging Required...	at Launch Site?	After Encapsulation? Yes / No	Yes / No
Is a Telemetry Interface with the Launch Vehicle Flight Computer Required? If Yes, describe:			Yes / No
Other Electrical Requirements:			

Please complete attached sheet of required pass-through signals.

RF RADIATION	
Time After Separation Until RF Devices Are Activated: (Note: Typically, no spacecraft radiation is allowed from encapsulation until 30 minutes after liftoff.)	
Frequency: _________________ MHz	Power: _______________ Watts
Location(s) on Satellite (spacecraft coordinate frame): Longitudinal __________ ❑ in ❑ cm Longitudinal __________ ❑ in ❑ cm	 Clocking (deg), Describe: Clocking (deg), Describe:

REQUIRED PASS-THROUGH SIGNALS

Item #	Pin	Signal Name	From LEV	To Satellite	Shielding	Max Current (amps)	Total Line Resistance (ohms)
1							
2							
3							
4							
5							
6							
7							
8							
9							
10							
11							
12							
13							
14							
15							
16							
17							
18							
19							
20							
21							
22							
23							
24							
25							
26							
27							
28							

MECHANICAL INTERFACE	
DIAMETER	Describe Diameter of Interface (e.g. Bolt Circle, etc):
SEPARATION SYSTEM	Will Launch Vehicle Supply the Separation System? Yes / No If Yes approximate location of electrical connectors: special thermal finishes (tape, paint, MLI) needed: If No, provide a brief description of the proposed system:
SURFACE FLATNESS	Flatness Requirements for Sep System or Mating Surface of Launch Vehicle:
FAIRING ACCESS	Payload Fairing Access Doors (spacecraft coordinate frame): Longitudinal __________ ❑ in ❑ cm Clocking (deg), Describe: Longitudinal __________ ❑ in ❑ cm Clocking (deg), Describe: Note: Standard Service is one door
DYNAMICS	Spacecraft Natural Frequency: Axial _____________ Hz Lateral _____________ Hz Recommended: > TBD Hz > TBD Hz
OTHER	Other Mechanical Interface Requirements: