Pegasus XL and Taurus

Performance Requirements and Capabilities

Under NASA SELVS-KSC Contract
Small Expendable Launch Vehicle Services (SELVS)
Kennedy Space Center (KSC)
Performance Requirements and Capabilities

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Dulles, Va. 20166
PEGASUS XL EXHIBIT 2
Mission and Performance Requirements and Capabilities

2 - 1 LAUNCH TIMING

A. Reference Data
   1. Reserved

B. Performance
   1. Payload Access
      Access to the payload through the fairing access doors shall be supported at the
      OCA location beginning with LV/OCA mate until one hour before Taxi.
   2. Multiple Launch Attempts
      Provided that sufficient time remains in the daily launch window defined in the ICD,
      the integrated LV/OCA shall be capable of supporting an additional launch attempt in
      the event of a launch abort on the first attempt.
   3. 24 Hour Turn-around
      The integrated LV/OCA shall be capable of a 24-hour turn-around and subsequent
      launch attempt in the event of a launch abort and return-to-base (RTB) that occurs
      prior to Fin Actuator System (FAS) thermal battery ignition.

2 - 2 PAYLOAD ORBIT AND MASS REQUIREMENTS

A. Reference Data
   1. Reference Earth Radius
      All altitudes in this section are specified as circular orbit altitudes above an equatorial
      Earth radius of 6378.14 km.
   2. Attach Hardware Mass
      Spacecraft masses in Table 2-1 do not include Contractor-provided attach and
      interface hardware. The mass of the entire separation system (both the aft and
      forward portions) has been accounted for on the launch vehicle side of the interface.
      For the purposes of the payload performance capabilities in Table 2-1, the 38” PAF is
      assumed. If the 23” PAF is used, the payload masses in Table 2-1 are reduced by 2
      kg.
3. Non-Standard Mission-Unique Hardware
All performance data provided in this section are for the baseline Pegasus XL launch vehicle, and do not include assessments for non-standard mission-unique hardware, unless otherwise noted.

4. Fairing Deployment Dynamic Pressure
The nominal payload fairing deployment criterion (dynamic pressure of 0.01 psf as calculated by Chapman's Equation) was used to generate all performance data in this section.

5. Guidance Reserve
The performance requirements in Table 2-1 were generated with a 220 ft/s guidance reserve.

6. Statistical Error Distribution for Insertion State Vector Dispersion Analyses
The statistical error distribution for insertion state vector dispersion analyses in this section assume gaussian distribution error sources, unless otherwise noted.

7. Insertion State Vector
The insertion state vector shall be composed of the following mean orbital elements: apogee radius, perigee radius, inclination, and right ascension of ascending node. The argument of periapsis shall also be included as an element of the insertion state vector if required by the LSTO.

B. Performance
1. SELVS Performance Regions
The launch vehicle service shall be capable of supporting launches within the SELVS performance regions shown in Figure 2-1 through Figure 2-4.
28.5 and 38 Degree Inclinations

Figure 2-1. SELVS Performance Region At 28.5° And 38° Inclinations

60 and 70 Degree Inclinations

Figure 2-2. SELVS Performance Region At 60° And 70° Inclinations

NOTE:
60 degree inclination requires a VAFB waiver
Figure 2-3. SELVS Performance Region At 90° And Sun-Synchronous Inclinations

Note: Pegasus XL does not currently have SELVS escape mission performance capability.

Figure 2-4. SELVS Performance Region For Escape Missions
2. Inclinations Supported
The launch service shall support missions to any desired inclination, consistent with launch range restrictions.

3. Minimum Launch Vehicle Performance
Minimum launch vehicle performance, in kg, to various reference circular orbits and inclinations shall be as shown in Table 2-1.

<table>
<thead>
<tr>
<th>Orbit Altitude (km)</th>
<th>Inclination (°)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>28.5</td>
</tr>
<tr>
<td>200</td>
<td>441</td>
</tr>
<tr>
<td>300</td>
<td>420</td>
</tr>
<tr>
<td>400</td>
<td>395</td>
</tr>
<tr>
<td>500</td>
<td>368</td>
</tr>
<tr>
<td>600</td>
<td>340</td>
</tr>
<tr>
<td>700</td>
<td>311</td>
</tr>
<tr>
<td>800</td>
<td>285</td>
</tr>
<tr>
<td>900</td>
<td>261</td>
</tr>
<tr>
<td>1,000</td>
<td>234</td>
</tr>
<tr>
<td>1,100</td>
<td>207</td>
</tr>
<tr>
<td>1,200</td>
<td>182</td>
</tr>
<tr>
<td>1,300</td>
<td>155</td>
</tr>
<tr>
<td>1,400</td>
<td>129</td>
</tr>
<tr>
<td>1,500</td>
<td></td>
</tr>
<tr>
<td>1,600</td>
<td></td>
</tr>
<tr>
<td>1,700</td>
<td></td>
</tr>
<tr>
<td>1,800</td>
<td></td>
</tr>
<tr>
<td>1,900</td>
<td></td>
</tr>
<tr>
<td>2,000</td>
<td></td>
</tr>
</tbody>
</table>

(1) - Requires VAFB Waiver.

Table 2-1. Minimum Launch Vehicle Performance to Various Circular Orbits

4. Minimum Guidance Reserve Requirement
All SELVS mission trajectories shall include a nominal guidance reserve of no less than 180 ft/s, unless otherwise agreed to by NASA.

5. Insertion Accuracies
Required right ascension of ascending node (RAAN) and inclination insertion accuracies for SELVS missions are shown in Table 2-2. Insertion accuracies in Table 2-2 for RAAN are applicable only for missions with a required RAAN expressed in the LSTO. Insertion accuracy for RAAN is valid only for inclinations between 80 degrees and 100 degrees.
<table>
<thead>
<tr>
<th>Required 3-Sigma Insertion Error Limit Magnitudes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inclination</td>
</tr>
<tr>
<td>Right Ascension of Ascending Node</td>
</tr>
</tbody>
</table>

Note: 1. Applicable only for missions with inclinations between 80 degrees and 100 degrees.

**Table 2-2. Standard Insertion Accuracy Requirements**

"Intentionally Deleted"

6. Insertion Accuracy Analysis Dispersion Sources

"Intentionally Deleted"
"Intentionally Deleted"
PAYLOAD FAIRING AND PAYLOAD FAIRING ACCESS DOORS

A. Reference Data

1. Minimum Lateral Frequency

A minimum lateral frequency of 20 Hz is assumed to determine the payload static envelopes. The payload is assumed to be hard mounted at the payload to payload attach fitting (PAF) interface plane.

2. Interface Plane

Deflections of the PAF interface plane have been accounted for on the launch vehicle side of the interface plane. The payload side of the PAF interface plane is assumed to be rigid.

3. Static Envelope

The static envelopes account for fairing and payload structural deflections. Payload dimensions due to manufacturing/design and tolerance stack-up shall be accounted for within the static envelope. The payload shall not extend aft of the payload/launch vehicle interface plane, unless otherwise approved in the ICD. Other static envelope violations are subject to approval in the ICD.

B. Performance

1. 38.810" Diameter Payload Interface Payload Fairing Static Envelope

The payload fairing static envelope with a 38.810" diameter payload interface shall be as shown in Figure 2-5. If nitrogen cooling is required by the payload, the payload envelope shall be locally reduced by 1" along cooling tube routing.
Figure 2-5. Payload Fairing Static Envelope with 38.810" Diameter Payload Interface
2. 23.250" Diameter Payload Interface Payload Fairing Static Envelope

The payload fairing static envelope with a 23.250" diameter payload interface shall be as shown in Figure 2-6. If nitrogen cooling is required by the payload, the payload envelope shall be locally reduced by 1" along cooling tube routing.

![Figure 2-6. Payload Fairing Static Envelope with 23.250" Diameter Payload Interface](image)

3. Payload Fairing Access Doors

Two 33.0 cm x 21.6 cm (13.0 in x 8.5 in), graphite, RF-opaque payload fairing access doors shall be provided. The 13.0 in dimension will be located along the x-axis. The doors shall be positioned according to NASA requirements within the zones defined in
Figure 2-7. The entire fairing access opening will be within the specified range. Fairing door centerlines will be at least 55° apart. The edge of the fairing doors will be at least 5" from fairing joints. The interface plane in the launch vehicle x-coordinate shall be as shown in Figure 2-7 for both separating and non-separating payloads.

### Notes:

<table>
<thead>
<tr>
<th></th>
<th>38&quot; Payload Interface Plane</th>
<th>23&quot; Payload Interface Plane</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Pegasus Station X</td>
<td>Pegasus Station X</td>
</tr>
<tr>
<td></td>
<td>(cm/in)</td>
<td>(cm/in)</td>
</tr>
<tr>
<td>Separable</td>
<td>1485.4/584.8</td>
<td>1509.4/594.3</td>
</tr>
<tr>
<td>Non-Separable</td>
<td>1475.4/580.9</td>
<td>1501.9/591.3</td>
</tr>
</tbody>
</table>

1. Entire Access Hole Must Be Within Specified Range.
2. Two 8.5" x 13.0" Doors per Mission Are Standard.
3. Door Centerlines Must Be at Least 55° Apart.
4. Edge of Doors Cannot Be Within 5" of Fairing Joints.

Figure 2-7. Payload Fairing Access Doors Placement Zones

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2-4 PAYLOAD SEPARATION ATTITUDE ACCURACY AND RATES

A. Reference Data

1. Separation System Reference Configuration

   The separation system configuration in this section assumed a 38" or 23" separation
system with bolt cutters located at 0° and two MS27484/MS27474 connectors, at 0° and 180° respectively.

2. Statistical Error Distribution for Tip-Off Rate Dispersion Analyses
The statistical error distribution for tip-off rate dispersion analysis assume gaussian distributed error sources, unless otherwise noted.

3. Payload Tip-Off
Payload tip-off refers to the angular velocity imparted to the payload upon separation due to an uneven distribution of torques and forces.

B. Performance

1. Payload Pre-Separation Pointing Accuracy
The Contractor shall provide a total payload pre-separation pointing accuracy of ±1 degree per axis. However, for sun-pointing missions, the launch service shall provide a total payload pre-separation pointing accuracy of ±2 degrees per axis. NASA will specify either an inertially-fixed or spin-stabilized attitude in the ICD.

2. Payload Deployment
The launch vehicle shall provide the desired initial payload attitude prior to payload separation. If required, this capability shall also be used to incrementally reorient the launch vehicle for the deployment of multiple spacecraft with independent-attitude requirements.

3. Payload Separation Tip-Off Rate Analysis Dispersion Sources
The Contractor shall perform a mission-specific tip-off analysis for each payload.
"Intentionally Deleted"
2 - 5  PAYLOAD SPIN/DESPIN

A. Reference Data
   1. Spin
      Spin refers to rotation about the longitudinal axis in either direction (clockwise (CW) or counter-clockwise (CCW)).

B. Performance
   1. Spin Capability
      The launch service shall provide a spin capability from a range of zero degree/sec up to 360 degrees/sec.

"Intentionally Deleted"
2. Standard Separating Mechanical Interfaces

NASA may select from one of two different separation systems, as a standard service. The 38" separable payload interface is defined by Figure 2-9 and Figure 2-10. The 23" separable payload interface is defined by Figure 2-11 and Figure 2-12.

3. Band and Clamp Shoe Configuration

The band and clamp shoes shall remain attached to the avionics structure or adapter cone by retention springs and lanyards. The Marmon clamp or equivalent shall be retracted by springs and shall be tethered to the lower section of the separation system during payload deployment.

4. Separation Ring Bolt Holes

The separation ring to which the payload attaches shall be supplied with through holes. The Contractor shall provide attachment bolts to this interface and can be inserted from either the launch vehicle or the payload side of this interface. The Contractor shall provide the integration ring and all necessary attachment hardware.
Figure 2-9. 38" Separable Payload Interface.

Figure 2-10. Flange for 38.81" Diameter Bolt Circle Interface.
Figure 2-11. 23" Separable Payload Interface

Figure 2-12. Flange for 23.25" Diameter Bolt Circle Interface.
2. Spin-Rate

The launch vehicle shall be capable of controlling the spin-rate of any payload prior to separation to within ±1.5 degrees/sec.

2 - 6 MECHANICAL INTERFACE

A. Reference Data

1. Reference System Weights

The weight of hardware separated with the payload is 4.0 kg (8.7 lbm) for the 38" (38.81" diameter bolt circle) system, 2.7 kg (6.0 lbm) for the 23" (23.25" diameter bolt circle) system, which includes bolts, nuts, washers, and harnesses connecting the separation system to the payload.

2. Payload Separation Velocity

The payload is released by matched push-off springs with sufficient energy to produce the typical nominal relative separation velocities shown in Figure 2-8.

![Figure 2-8. Payload Separation Velocities Using the Standard Separation System.]

B. Performance

1. Hardware and Integration Services

The Contractor shall provide all hardware and integration services necessary to attach non-separating and separating payloads to the launch vehicle. All attachment hardware shall contain locking features consisting of locking nuts, inserts, or fasteners. The Contractor shall provide identical bolt patterns for both separating and non-separating mechanical interfaces.
5. Separation System Debris

The payload shall be protected from debris generated by the separation system. A debris shield shall contain a majority of all bolt cutter shrapnel and prevent direct impingement of all escaped gas and debris from the separation ordnance on the payload. The separation system shall function in a manner that prevents any recontact with the payload, including Contractor-provided attach hardware on the payload, by the upper stage or any element of the separation system once separation has been initiated.

6. Interface Drill Tool

The Contractor shall provide a drill tool to the payload for precision drilling of the payload/launch vehicle interface. The Contractor shall coordinate delivery of the tool to NASA.

7. Payload Separation System Fit Check

The Contractor shall support a fit check of the upper ring of the flight separation system with the payload. The upper ring separation connector bracketry shall be installed so that the separation connector harnesses can be fit checked. The Contractor shall provide all hardware, procedures, and personnel to support the separation system fit check. The payload will provide the facility and the ground support equipment, procedures and personnel to support handling of the payload.

8. Launch Site Installation

The Contractor shall provide the launch site integration of the separation system to the payload and the launch vehicle. The payload is responsible for providing all necessary ground support equipment, procedures, and personnel associated with handling of the payload.

9. Test Payload Attach Fitting (TPAF) for Payload System Testing Support

The payload may request TPAF support for payload system testing. The Contractor shall provide an existing tensioned TPAF including a standard payload separation system (23" or 38" as required by the ICD), a standard 23" payload cone and all required attach hardware for payload vibration testing, thermal vacuum testing, spin balancing or other testing. The Contractor shall provide the existing TPAF four weeks after receipt of payload request. The payload may use the existing TPAF for a period of up to three weeks. Longer durations can be specified by mutual agreement.

10. Test Payload Attach Fitting (TPAF) for Dynamic Release Test Support

The Contractor shall provide a single existing TPAF including a standard payload separation system (23" or 38" as required by the ICD), a standard 23" payload cone and all required attach hardware. The Contractor shall provide two technical support periods of up to three consecutive days for each period to tension (or de-tension) the
separation system and integrate (or de-integrate) it with the payload and the payload cone. Technical support and all required expendable hardware/ordnance shall be provided by the Contractor to perform two consecutive dynamic release tests.

2 - 7 ELECTRICAL INTERFACE

A. Reference Data

1. Electrical Interface

The standard electrical interface shall be defined as the separation plane of the electrical power, control and telemetry connector and the pyrotechnic connector.

2. Standard Electrical Connectors - Coupling Rings

The ICD shall document that the standard electrical connectors provided to the payload will have the coupling rings removed prior to integration.

B. Performance

1. Electrical Interfaces

The pin allocations for the standard and non-standard payload electrical interfaces provided by the launch vehicle shall be defined in Table 2-5. The electrical interface shall be capable of providing an interface between the payload and the payload GSE (or ASE aboard the OCA) during launch processing until six minutes before launch. A block diagram of the standard launch vehicle electrical interface capabilities shall be as shown in Figure 2-13.

<table>
<thead>
<tr>
<th>Function</th>
<th>Number of Wires</th>
<th>Peak Voltage (Vdc)</th>
<th>Continuous Current (Amps)</th>
<th>Resistance from ASE to Payload (Ohms)</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power or Data (1)</td>
<td>10 (5 TSP)</td>
<td>60</td>
<td>3.0</td>
<td>2.5</td>
<td>90%</td>
</tr>
<tr>
<td>Payload Sep Sense</td>
<td>6</td>
<td>60</td>
<td>3.0</td>
<td>N/A</td>
<td>N/A</td>
</tr>
</tbody>
</table>

(1) Data Lines Shall Be Twisted Shielded Pairs

Table 2-5. Electrical Interface Minimum Wire Requirements
Figure 2-13. Payload Electrical Interface Block Diagram

2. GSE/ASE Installation
The Contractor shall assist in the installation of payload-provided ASE aboard the OCA, and shall provide a Launch Panel Operator (LPO) to operate the payload ASE during launch operations.

3. Interface Hardware
The standard payload electrical and pyrotechnic connectors and harness configurations shall be as shown in Figure 2-14 and Figure 2-15, respectively. The Contractor shall provide the payload side of the interface connectors for the payload flight harness (payload side - MS27474T-16F-42S (42 pin) for power, control and telemetry and MS27474T-14F-18S (18 pin) for pyrotechnic commands) one year prior to launch. The payload will be responsible for integrating these connectors to the payload flight harness forward of the interface plane. The Contractor shall then integrate the payload flight harness to the separation system two months prior to launch. The Contractor shall provide the matching interface connectors (launch vehicle side MS27484T-16F-42P for power, control and telemetry and MS27484T-14F-18P for pyrotechnic commands) and the associated electrical harnesses aft of the interface plane for non-separating and separating payloads. All interface wires shall be shielded for EMI protection.
4. Payload Auxiliary Power

The Contractor shall provide power during ground operations and captive flight directly from the carrier aircraft.
5. Payload Command and Control

The Contractor shall provide discrete sequencing commands to the payload. These commands shall be opto-isolated pulses of programmable lengths in multiples of 40 milliseconds. The Contractor shall provide eight command line pairs, each capable of multiple pulses, for the payload. The payload is responsible for supplying the voltage source (≤ 40 VDC) and limiting the current to 500 milliamps (mA) nominal in a fashion similar to using a dry contact relay as shown in Figure 2-13.

6. Payload Status Monitoring

The Contractor shall provide payload discrete telemetry during ground processing (limited to launch vehicle on-times), checkout, captive carry and launch. Four discrete telemetry signals shall be accommodated in the launch vehicle telemetry stream. This telemetry interface shall include a signal and ground for each discrete transmitted on dedicated twisted shielded wire pairs. The launch vehicle side of the interface shall limit the current of a 5.0 VDC signal to 10 mA. The payload is responsible for optical isolation at the payload side of the interface.

7. Payload Pyrotechnic Initiator Driver Unit

The Contractor shall provide one pair and four single 75 ms pulses at 5 amps. Two of the four single output pulses will be used for the standard separation system. The remaining two single output pulses and 1 dual output will be available for payload use.

8. Provisions To Prevent Static Charge On The Payload Umbilical Connectors

The Contractor shall establish and maintain controls to prevent charge build-up on all electrical connectors during ground processing. Prior to launch vehicle/OCA separation, all power interfaces between the launch vehicle and the OCA shall be deadfaced.

2 - 8 PAYLOAD ENVIRONMENT

A. Reference Data and Assumptions

1. Payload Separation System Assumption

The payload environments defined below apply to the launch vehicle with a single payload using a standard separation system (23" or 38" clamp band). The payload environments associated with the use of alternative separation systems, a non-separating payload interface or multiple payload attach fittings will differ from those presented below.

2. Payload Acoustic & Random Vibration Environment

Powered flight acoustic and random vibration from all three motor burns and captive carry data was used to create an overall envelope for all phases of a Pegasus XL.
launch vehicle mission. The typical captive carry duration is approximately 90 minutes.

3. Payload Shock Environment
The flight limit levels are derived from ground separation test data and analytical predictions for the vehicle and payload separation systems.

4. Payload Acceleration Environment
The axial accelerations are analytically predicted based on vehicle and payload mass properties and predicted motor performance, and verified with flight accelerometer data.

5. Payload Transient Acceleration Environment
The drop transient acceleration assumes the payload first fundamental lateral frequency is greater than 20 Hz.

6. Temperature Profile of Inner Fairing Surface Adjacent to the Payload
The temperature profile was derived using the worst case heating trajectory, the minimum tolerance TPS thickness, and worst case warm initial temperatures.

B. Performance

1. Payload Acoustic Environment
The acoustic levels during OCA take-off, captive carry and powered flight shall not exceed the flight limit levels shown in Figure 2-16. The +6dB spectrum for 75 seconds is recommended for payload standard acoustic testing to account for fatigue duration effects to encompass at least two launch attempts and powered flight.
2. Payload Random Vibration Environment

The maximum payload interface random vibration curve that encompasses launch vehicle/OCA takeoff, captive carry and powered flight environments shall not exceed the levels shown in Figure 2-17. A +3dB spectrum for 75 seconds in each axis is recommended for payload standard vibration testing to account for fatigue duration effects to encompass at least two launch attempts and powered flight.
Figure 2-17. Maximum Payload Interface Random Vibration Environment

3. Payload Shock Environment

The maximum shock response spectrum at the base of the payload from all launch vehicle events shall not exceed the levels shown in Figure 2-18.

Figure 2-18. Maximum Shock at the Base of the Payload

4. Payload Acceleration Environment

The load conditions experienced during launch operations using the OCA shall not exceed the levels shown in Table 2-6. The accelerations listed are design limit loads. The axial accelerations for each stage shall not exceed the levels shown in Figure 2-19. The maximum duration for post-stage burnout quasi-static accelerations shall be no greater than 6 seconds.
The launch vehicle shall have no significant sustained sinusoidal vibration environments during captive carry or powered flight.

<table>
<thead>
<tr>
<th>Environment (g's)</th>
<th>Static</th>
<th>Quasi-Static*</th>
<th>Static</th>
<th>Quasi-Static*</th>
<th>Static</th>
<th>Quasi-Static*</th>
</tr>
</thead>
<tbody>
<tr>
<td>Taxi, Abort &amp; Captive Flight</td>
<td>±1.0</td>
<td>N/A</td>
<td>±0.7</td>
<td>N/A</td>
<td>+3.6-1.0</td>
<td>N/A</td>
</tr>
<tr>
<td>(Man-Rated Events)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Aerodynamic Pull-Up</td>
<td>-3.7</td>
<td>±1.0</td>
<td>±0.2</td>
<td>±1.0</td>
<td>+2.33</td>
<td>±0.2</td>
</tr>
<tr>
<td>Stage Burn Out</td>
<td>See Fig. 2-19</td>
<td>±1.0</td>
<td>±0.2</td>
<td>±1.0</td>
<td>±0.2</td>
<td>±0.2</td>
</tr>
<tr>
<td>Post-Stage Burnout</td>
<td>±0.2</td>
<td>±1.0</td>
<td>±0.2</td>
<td>±2.0</td>
<td>±0.2</td>
<td>±2.0</td>
</tr>
</tbody>
</table>

*Static Equivalent of Mixed Dynamic Loads

Table 2-6. Design Limit Loads

![Figure 2-19. 3σ High Maximum Axial Acceleration (G's)](image)

Does Not Include Random Vib (See Figure 4-26)

5. Payload Drop Transient Acceleration

The launch vehicle/OCA separation transient design limit load environment shall not exceed the levels shown in Table 2-7.

<table>
<thead>
<tr>
<th>Location</th>
<th>Ax</th>
<th>Ay</th>
<th>Az</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload Base</td>
<td>±0.5 g</td>
<td>±0.5 g</td>
<td>±3.85 g (interface)</td>
</tr>
</tbody>
</table>

Table 2-7. Launch Vehicle/OCA Separation Transient Design Limit Load Environment
6. Payload Thermal and Humidity Environment

The Contractor shall provide filtered, conditioned air until LV/OCA separation.

Diffusers shall be provided at the fairing air conditioning inlet to reduce impingement velocities on the payload. The indirect airflow within the payload envelope shall not exceed 250 ft³/min with a maximum velocity of 6.9 ft/sec from launch vehicle/payload mate through payload fairing closeout. From payload fairing closeout through drop, the indirect airflow shall not exceed 320 ft³/min with a maximum velocity of 8.9 ft/sec.

The payload temperature and humidity environments for vehicle assembly, flight line and captive carry operations shall be as shown in Table 2-8. The VAB shall be capable of accepting NASA provided ducting to route, clean, filtered conditioned air from an external source to the clean tent for localized cooling capability.

<table>
<thead>
<tr>
<th>Event</th>
<th>Temperature Range</th>
<th>Control</th>
<th>Relative Humidity (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>°C</td>
<td>°F</td>
<td></td>
</tr>
<tr>
<td>Pre-Payload Fairing Installation</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>-Outside VAB Clean Tent</td>
<td>21.1±1.7</td>
<td>70 ± 3</td>
<td>A/C</td>
</tr>
<tr>
<td>-Inside VAB Clean Tent</td>
<td>21.1±1.7</td>
<td>70 ± 3</td>
<td>Filtered A/C</td>
</tr>
<tr>
<td>Post-Payload Fairing Installation (GSE)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>-VAB</td>
<td>21.1±1.7</td>
<td>70 ± 3</td>
<td>Filtered A/C</td>
</tr>
<tr>
<td>-Roll-Out to OCA (VAFB)</td>
<td>12.8±5.6</td>
<td>55 ± 10</td>
<td>Filtered Ambient</td>
</tr>
<tr>
<td>-LV/OCA Mate/Hot Pad</td>
<td>23.3±5.6</td>
<td>74 ± 10</td>
<td>Filtered A/C</td>
</tr>
<tr>
<td>OCA AACS (Ground)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>-Taxi</td>
<td>23.3±5.6</td>
<td>74 ± 10</td>
<td>Filtered A/C</td>
</tr>
<tr>
<td>OCA AACS (Altitude)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>-Abort/Contingency</td>
<td>23.3±5.6</td>
<td>74 ± 10</td>
<td>Filtered A/C</td>
</tr>
<tr>
<td>Notes: (1). Temperature and relative humidity are not selectable by the Payload at the VAB. (2). GSE A/C performance is dependent upon ambient conditions. Inlet temperature is selectable and controlled to within ±2°C (±4°F) of Set Point. (3). AACS ground performance is dependent upon ambient conditions (dew point). Inlet temperature is selectable and controlled within ±2°C (±4°F) of Set Point.</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 2-8. Payload Thermal and Humidity Environments

The component that exhibits the maximum temperature inside the payload fairing, with a view factor to the payload, is the inner surface of the fairing. Worst case transient temperature profile shall not exceed the levels shown in Figure 2-20. The fairing inner surface emissivity shall not exceed 0.1. The external surfaces of the upper stage motor shall not have direct view factors to the payload.
7. Payload Electromagnetic Environment

All vehicle power, control and signal lines inside the payload fairing shall be shielded and terminated.

The Contractor shall provide a fairing with a minimum of 20 dB attenuation between 1 and 10,000 MHz.

The frequencies and maximum radiated signal levels from the launch vehicle antennas that are located near the payload during powered flight shall not exceed the levels shown in Table 2-9. Launch vehicle antennas located inside the fairing shall be inactive until after fairing deployment. The frequencies and maximum radiated signal levels from the OCA emitters and receivers during powered flight shall not exceed the levels shown in Table 2-10.

<table>
<thead>
<tr>
<th>Source</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
<th>5</th>
<th>6</th>
<th>7</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>Command &amp; Destruct</td>
<td>Tracking</td>
<td>Tracking</td>
<td>Instrument</td>
<td>Booster</td>
<td>Telemetry</td>
<td>Telemetry</td>
</tr>
<tr>
<td>Role</td>
<td>Receive</td>
<td>Transmit</td>
<td>Receive</td>
<td>Transmit</td>
<td>Transmit</td>
<td>Receive</td>
<td>Transmit</td>
</tr>
<tr>
<td>Band</td>
<td>UHF</td>
<td>C-Band</td>
<td>C-Band</td>
<td>S-Band</td>
<td>S-Band</td>
<td>L-Band</td>
<td>S-Band</td>
</tr>
<tr>
<td>Frequency (MHz)</td>
<td>425.0 or 416.5</td>
<td>5.756</td>
<td>5.850</td>
<td>2,250.5</td>
<td>1,370.5</td>
<td>1,575.42</td>
<td>1,227.60</td>
</tr>
<tr>
<td>Bandwidth</td>
<td>180 kHz at 60 dB</td>
<td>14 MHz at 3 dB</td>
<td>750 kHz at 3 dB</td>
<td>315 kHz at 3 dB</td>
<td>20.45 MHz</td>
<td>12 MHz</td>
<td></td>
</tr>
<tr>
<td>Power Output</td>
<td>N/A</td>
<td>400 W Peak</td>
<td>N/A</td>
<td>5 W</td>
<td>5 W</td>
<td>N/A</td>
<td>8 W</td>
</tr>
<tr>
<td>Sensitivity</td>
<td>-107 dBm</td>
<td>N/A</td>
<td>-70 dBm</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
</tr>
<tr>
<td>Modulation</td>
<td>FM</td>
<td>Pulse Code</td>
<td>Pulse Code</td>
<td>FM/FM</td>
<td>PCM/FM</td>
<td>PRN Code</td>
<td>N/A</td>
</tr>
</tbody>
</table>

Table 2-9. Pegasus XL RF Emitters and Receivers
Table 2-10. OCA RF Emitters and Receivers

2 - 9 CONTAMINATION CONTROL

A. Reference Data and Assumptions

1. Reserved

B. Performance

1. Contamination Control Plan

The Contractor's generic payload contamination control plan shall be based on, and comply with, MIL-STD-1246C requirements. The Contractor shall develop and implement a mission-unique contamination control plan to accomplish any standard and non-standard contamination control requirements identified for each payload. Any mission-unique contamination control plan shall also comply with the requirements of MIL-STD-1246C.

2. Payload/Vehicle Integration Environment

The Contractor's integration facility shall be maintained at all times as a visibly clean work area.

The Contractor shall provide FED-STD-209E Class 100,000 cleanroom environment for payload integration operations. Air shall be supplied to the cleanroom through a bank of High-Efficiency Particulate Air (HEPA) filters. Particulate size vs. time data shall be recorded in accordance with FED-STD-209E. The Contractor shall certify the cleanliness level of the cleanroom no earlier than 30 days and no later than 5 days prior to payload arrival.

Carbon filters shall be provided in the cleanroom to remove volatile hydrocarbons of molecular weight 70 or greater from the fairing air supply, with better than 95% efficiency.

The Contractor shall provide HEPA filtered, FED-STD-209E Class 100,000 air to the
PLF during encapsulation, transport to the OCA and LV/OCA mate. The Contractor shall provide HEPA filtered, FED-STD-209E Class 100,000 air to the PLF during post LV/OCA mate integration operations, during LV/OCA taxi, captive carry and launch. The Contractor shall certify FED-STD-209E Class 100,000 air prior to any connection to the fairing. This certification shall be made after each system has been running a minimum of 30 minutes.

3. Fairing Environment

Following fairing closeout, the payload fairing environment shall be purged with conditioned, filtered, FED-STD-209E Class 100,000 air. All HEPA filters shall have a minimum 99.97% effective in removing particles of ≥0.3 microns in size. The hydrocarbon content of any fairing air supply shall be continuously monitored after fairing mate. Carbon filters shall be provided to remove volatile hydrocarbons of molecular weight 70 or greater from the fairing air supply, with better than 95% efficiency.

The Contractor shall continuously record and monitor the launch vehicle fairing supply air for hydrocarbons via a probe installed into the fairing inlet duct. The AACS system shall incorporate an inline HEPA filter to remove particulates, and an inline charcoal filter to absorb volatile organic compounds (VOCs). Recording and monitoring of the fairing inlet air shall be accomplished by a ground-based continuous monitor through OCA/Launch Vehicle mate. A second, airborne hydrocarbon monitor shall record and monitor air from the AACS supply duct from OCA/Launch Vehicle mate through captive carry.

4. Fairing Internal Surface Cleaning

The Contractor shall clean, certify and maintain the cleanliness of the fairing inner surface to MIL-STD-1246C, Level 750A.

5. Clean Room Garments

The Contractor shall provide cleanroom garments and cleaning supplies. Clean room garments provided by the Contractor shall be cleaned by the Contractor.

6. Cooling Supply

The Contractor shall provide unlimited MIL-P-27401C Grade B GN2 for spot cooling of a payload component from payload mate through launch vehicle/OCA taxi. The system shall be capable of regulating the flow between 0 and 3 SCFM. The Contractor shall provide 200 lbm of GN2 during launch vehicle/OCA taxi and captive carry.

7. Materials

The Pegasus payload fairing inner exposed surface shall meet NRP-1124 outgassing standards of Total Mass Loss (TML) ≤ 1.0%, and Collected Volatile Condensable Material (CVCM) ≤ 0.1%.
8. MLI Thermal Insulation

MLI thermal insulation blankets shall be sealed such that venting of blanket material debris does not occur in a direction towards the payload.

2 - 10 COLLISION/CONTAMINATION AVOIDANCE MANEUVER (CCAM)

A. Reference Data
   1. Reserved

B. Performance
   1. Following payload separation, the launch vehicle shall perform a collision/contamination avoidance maneuver (CCAM) to minimize payload contamination and any chance of re-contact with the separated payload.

2 - 11 PAYLOAD ENVIRONMENT INSTRUMENTATION

A. Reference Data
   1. Reserved

B. Performance
   1. Phases
      The instrumentation to support mission success determination (Article H-8) shall be provided by the launch vehicle and shall encompass ground, captive carry and flight mission phases. The payload environment instrumentation shall be transmitted, received and recorded in accordance with SOW 2.4.1.2.3.

   2. Loads And Dynamics
      The Contractor shall provide an accelerometer (wideband) system to gather and transmit low frequency and random vibration data near the payload interface to determine compliance to accepted levels. At a minimum, measurements shall be made using accelerometers defined in Table 2-11.

   3. Temperature
      The Contractor shall provide the temperature sensors defined in Table 2-11.
<table>
<thead>
<tr>
<th>Environment</th>
<th>Type</th>
<th>Number and Locations</th>
<th>Range, Sensitivity, and Bandwidth</th>
</tr>
</thead>
<tbody>
<tr>
<td>Loads &amp; Dynamics</td>
<td>Piezo Accelerometer</td>
<td>Payload UF, X-Axis</td>
<td>±250g, 100 mV/g, 5-2000 Hz</td>
</tr>
<tr>
<td></td>
<td>Piezo Accelerometer</td>
<td>Payload UF, Y-Axis</td>
<td>±250g, 100 mV/g, 5-2000 Hz</td>
</tr>
<tr>
<td></td>
<td>Piezo Accelerometer</td>
<td>Payload UF, Z-Axis</td>
<td>±250g, 100 mV/g, 5-2000 Hz</td>
</tr>
<tr>
<td></td>
<td>Servo Accelerometer</td>
<td>Payload UF, Z-Axis</td>
<td>±10g, 240 mV/g, 0-500 Hz</td>
</tr>
<tr>
<td></td>
<td>Servo Accelerometer</td>
<td>Avionics Structure, X-Axis</td>
<td>±10g, 240 mV/g, 0-500 Hz</td>
</tr>
<tr>
<td></td>
<td>Servo Accelerometer</td>
<td>Avionics Structure, Y-Axis</td>
<td>±10g, 240 mV/g, 0-500 Hz</td>
</tr>
<tr>
<td>Temperature</td>
<td>RTD</td>
<td>Av, Sect/Fairing Air (0&quot;)</td>
<td>-85 - 174°C</td>
</tr>
<tr>
<td></td>
<td>RTD</td>
<td>Av, Sect/Fairing Air (180&quot;)</td>
<td>-85 - 174°C</td>
</tr>
<tr>
<td></td>
<td>Thermocouple</td>
<td>Failing Nose (External)</td>
<td>-73 - 194°C</td>
</tr>
<tr>
<td></td>
<td>Thermocouple</td>
<td>Failing Ogue (External)</td>
<td>-73 - 194°C</td>
</tr>
<tr>
<td></td>
<td>Thermocouple</td>
<td>Fairing Cylinder (External)</td>
<td>-73 - 194°C</td>
</tr>
<tr>
<td></td>
<td>RTDs</td>
<td>Various Av, Sect, Components</td>
<td>-85 - 174°C</td>
</tr>
<tr>
<td></td>
<td>TG</td>
<td>Fairing AACS&quot; Inlet</td>
<td>0 - 50°C</td>
</tr>
</tbody>
</table>

*Airborne Air Conditioning System

Table 2-11. Payload Environment Instrumentation
ATTACHMENT D-P
NON-STANDARD SERVICES

PEGASUS XL LAUNCH VEHICLE

The following non-standard services shall be performed only as requested and authorized by NASA in accordance with Article G-3 Task Ordering Procedures and SOW Section 3.0.

1.0 MISSION–UNIQUE HARDWARE MODIFICATIONS

The Contractor shall design, manufacture, test, and implement the following, if authorized:

1.1. RESERVED

1.2. FAIRING MODIFICATIONS:

1.2.1. Additional Access Doors

All access doors (Reference Figure D-1) specified for this non-standard service are 8.5" x 13" and are oriented with the 13" dimension along the launch vehicle X-axis. Additional 8.5" x 13" access doors shall have an impact on payload performance to orbit of approximately 1 kg each (orbit specific).

1.2.1.1. Cylinder Doors in Standard Locations (Authorize by: L-15 Months)

One additional fairing access door, with the following restrictions:

For doors in the existing approved cylindrical section, the same restrictions apply as the standard service doors, plus:

- The additional door must have at least 55° centerline to centerline circumferential displacement if at the similar station location on the same fairing half; and
- The additional door must have at least 24-inch nearest edge to nearest edge displacement if at dissimilar station locations.

1.2.1.2. Ogive Doors in Standard Locations (Authorize by: L-15 Months)

Doors in the following specific locations in the ogive section of the fairing as long as the restrictions listed in Non-Standard Service Section 1.2.1.1, are not violated with payload fairing doors:
Figure D-1. Payload Fairing Access Door Placement Zone

Station 137.94 of the fairing (vehicle station 652.74) with clocking restrictions the same as the cylinder section; and

Station 132.47 of the fairing (vehicle station 647.27) with clocking restricted to 45°, 71°, 225°, or 300°.

1.2.1.3. Ogive Doors in Non-standard Locations (Authorize by: L-18 Months)

The Contractor shall provide doors in ogive locations other than those specified in Non-Standard Service Section 1.2.1.2. The Contractor shall perform an analysis to verify structural integrity of the fairing.

1.2.2. Access doors of non-standard sizes

1.2.2.1. 4.5 Inch Circular Doors in Cylinder (Authorize by: L-15 Months)
Circular 4.5” diameter doors within the authorized cylinder section of the fairing per Non-Standard Service Section 1.2.1.1. These doors have non-structural closeouts and shall be analyzed on a case by case basis for placement limitations prior to location approval. There shall be no performance penalty with 4.5” circular doors.
1.3. ALTERNATE PAYLOAD ATTACH FITTINGS

1.3.1. Different Size or Different Payload Interface PAF's

1.3.1.1. 17" PAF (Authorize by: LSTO)

The 17" PAF interface is a circle of 24 equally spaced 0.251" holes 17.0 inches in diameter. The 17" PAF (Reference Figure D-2) is comprised of a 17" Marmion clamp (or equivalent) separation system on a 38" to 17" adapter cone. The distance from the payload interface plane to the ogive mate line for the payload static envelope is reduced by 22.15" inches when this option is exercised. The 17" PAF shall use the same debris shield design as the 23" PAF.

The Contractor shall perform an analysis of the payload using actual payload mass properties to determine the launch vehicle random vibration and drop transient environmental specifications

"Intentionally Deleted"

The electrical interface requirements in Pegasus Exhibit 2, Section 2-7, shall be applicable to the 17" PAF, with the exception that the connector types will be mission-unique. The electrical connector types shall be
identified in the ICD. Non-Standard Service Section 3.8 (40-pin Pass Through Harness) shall also be available as an option that may be exercised in combination with the 17” PAF Non-Standard Service. The electrical connector for Non-Standard Service Section 3.8, if exercised, shall also be identified in the ICD.

Non-separable interface attachment shall use the same hole pattern. If a non-separable interface attachment is used, then all four signal output pyro discretes offered under the standard electrical service shall be available for payload use.

The impact on the launch vehicle performance to orbit when using this 17” PAF is 18.5 pounds.

1.3.1.2. Reserved
Figure D-2. 17" Separable Payload Interface.
1.3.1.3. 10" PAF (Authorize by: LSTO)

The 10" PAF consists of a 38" to 23" adapter cone and a 23" to 10" adapter cone with an integral Marmon clamp (or equivalent) separation system. The adapter cone is 6" in height and reduces the available payload volume to less than the minimum cylindrical height of 34". The 10" PAF uses the same flight proven debris shield design as the 23" PAF.
This integral system requires separation system mating while the payload is attached to the separation system, which should be taken into account in the payload environmental analysis. Installation shock levels shall not exceed the normal separation system activation levels, however they may be repeated many times during the clamp band tension equalization process.

"Intentionally Deleted"

The electrical interface requirements in Pegasus Exhibit 2, Section 2-7, shall be applicable to the 10" PAF, with the exception that the pyro discretes described in Section 2-7 B., Item 5, and also in Figure 2-13, of Pegasus Exhibit 2 will be unavailable. Standard payload power, command, and telemetry services described by Pegasus Exhibit 2, Section 2-7, will be available through a single standard 42-pin connector described in that section. The standard 18-pin pyro discretes connector will be unavailable. The electrical interface Non-Standard Service (Sections 1.9 and 3.6-3.8) are unavailable in combination with the 10" PAF Non-Standard Service.

There shall be a 19.0 pound performance impact on the launch vehicle performance to orbit when using this PAF.
1.3.1.4. 3-point PAF (Authorize by: LSTO)

Figure 1.3.1.4-1 is drawing of the payload static envelope and payload stayout zones when using a 3-point PAF.

---

Figure D-4. 3-Point PAF Static Envelope
The 3-point PAF is comprised of a flangeless avionics section with aluminum brackets attached at the 0°, 120° and 240° clocking with integral titanium cups and bolt catchers for the launch vehicle end of a non-explosive actuated separating nut/bolt separation system. This system is designed for use with MicroStar bus satellites only. Environmental levels for this interface are within normal launch vehicle environments. There shall be no performance impact for use of this design. The available payload volume for this design exceeds the volume available for any other PAF (Reference Figure D-4).

The separation plane is at the top of a standard avionics section, providing 3.95" greater envelope length than the 38" PAF (Reference Figure D-3). The 3-Point PAF does not require debris shielding by the launch vehicle as the non-explosive actuator that initiates separation is fully encapsulated by the payload.

"Intentionally Deleted"
The electrical interface requirements in Pegasus Exhibit 2, Section 2-7, shall be applicable to the 3-point PAF, with the following exceptions:

The standard 42-pin connector shall be replaced with a 22 pin G&H 1179 "zero force" connector.

The dual-serial electrical interface (4 pins) required for sending primary and redundant firing codes to the payload shall be included in the standard electrical interface. The payload shall have 18 pins to select functions from the standard electrical service.

Non-Standard Service Section 3.8 (40-pin Pass Through Harness) shall be unavailable in combination with the 3-point PAF Non-Standard Service.

1.3.2. Low tip-off rate PAF's

1.3.2.1. Reduced Clamp Band Tension PAF (Authorize by: LSTO)

To reduce clamp band separation impulse and thus proportionally reduce tip-off, the Contractor shall perform analysis to verify system structural capability and couple loads analysis of the clamp band with a reduced clamp band tension. If required, the Contractor shall conduct testing to validate the analysis. This tip-off reduction technique shall be performed with the 38", 23", 17" and 10" PAF. The tip-off reduction technique is not valid for the 3-point PAF. The impact to dispersion sources shall be provided by the Contractor at the time of Launch Service Proposal submission. The reduced clamp band tension option does not affect the standard or non-standard electrical interfaces on any of the PAFs.

1.3.2.2. Low Tip-off Rate 38" PAF (Authorize by: L-15 Months)

The low tip-off 38" separation system uses two aluminum interface rings that are clamped by dual, semi-circular stainless steel clamp bands with aluminum clamp shoes. This option uses two low impulse separation connectors.

"Intentionally Deleted"

The electrical interface requirements in Pegasus Exhibit 2, Section 2-7, shall be applicable to the 38" Low-Tip-Off PAF, with the exception that the connector types shall be 1179-700-101 (launch vehicle side) and 1179-800-101 (payload side). Electrical Interface Non-Standard Service Section 3.8 and Non-Standard Service Section 1.9 are available with the following notes:

1. Cannot combine NSS Section 3.8 with NSS Sections 1.9B or 1.9C. Can combine NSS Section 3.8 with NSS Section 1.9A.
2. Cannot combine NSS Section 1.9B and NSS Section 1.9C if the full capability of each option is desired, and the full capability of Exhibit 2 Section 2-7 standard services are utilized (i.e., could not have full standard service, full NSS Section 1.9B, and full NSS Section 1.9C). Options must be compatible with the total number of pins (56) available across the interface.

"Intentionally Deleted"

1.4. OPTIONAL UPPER STAGE HARDWARE

1.4.1. Hydrazine Auxiliary Propulsion System (HAPS) (Authorize by: L-24 Months)

An integral liquid fourth stage called the Hydrazine Auxiliary Propulsion system (HAPS) shall be provided. HAPS is a 4th stage integral to an extended vehicle avionics structure. HAPS is a monopropellant hydrazine propulsive system, which functions in blow down mode. HAPS consists of a titanium propellant
tank with an AF-E-332 bladder, three 45 lbf nominal Rocket Engine Assemblies (REA), and a redundantly initiated pyrotechnic isolation valve.

"Intentionally Deleted"

The use of the HAPS will increase the length of the avionics structure, thereby decreasing the static envelope for the payload by 3.8 inches.

1.5. RESERVED
1.6. RESERVED
1.7. RESERVED

1.8. PAYLOAD CONNECTOR COVERS (Authorize by: L-15 Months)

The Contractor shall provide connector covers for the payload side of the separation system to cover the 42-pin and 18-pin interface connectors. The connector covers provided shall be spring loaded and attach to the standard umbilical support brackets. A delrin bracket on the launch vehicle side of the separation system shall be used to hold the cover open until the two halves of the separation system are physically separated. At payload separation, the cover shall close to cover the connector. This non-standard service shall include connector covers for any of the payload non-standard service payload electrical interfaces.

1.9. ENHANCED ELECTRICAL INTERFACE

A. Increased Capacity Payload GSE Interface (Authorize by: L-18 Months)

The standard electrical interface of 5 TSP from payload ASE on the OCA to payload shall be modified to provide 2 MIL-STD-1553 connections (2 pins for shield pass-through and 4 pins for dedicated Contractor provided 1553 bus impedance controlled cable) and retain 2 TSP (4 pins) for payload use. The key parameters for these commands are specified in Table 1.9-1.
### Table 1.9-1. Payload-to-ASE/GSE Electrical Interface Minimum Requirements

<table>
<thead>
<tr>
<th>Function</th>
<th>Number</th>
<th>Peak Voltage ((V_{\text{dc}}))</th>
<th>Continuous Current (Amps)</th>
<th>Resistance from ASE to Payload ((\Omega))</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power or Data</td>
<td>2 TSP</td>
<td>60</td>
<td>3.0</td>
<td>2.5</td>
<td>90%</td>
</tr>
<tr>
<td>MIL-STD-1553B</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>


The harness shall be routed through a floor panel just below the LPO rack. On the OCA side, the rack-mounted panel that fits into the LPO station in the space reserved for payload applications shall plug into the floor panel connector. NASA will be responsible for supplying the rack-mounted panel to the LPO station and all ASE to the interface with the Contractor provided cable. The connector type, contact arrangement, and physical dimensions shall be specified in the Interface Control Document for the specific mission.

### B. Launch Vehicle Command & Control of Payload (Authorize by: L-18 Months)

The launch vehicle shall provide the means for step or pulse commanding of payload separation system ordnance, payload relays, and payload opto-isolated discretes from 10 minutes before launch until payload separation. The standard electrical service shall be modified to add 2 single pyro events to the 6 pyro events already provided in the standard service and retain one payload separation sense loopback. Hence, the Non-Standard Service shall use the 18 pin pyro connector to provide 8 pyro events or commands (16 pins) and 1 payload separation sense loopback (2 pins). Table 1.9-2 defines the key electrical parameters for these functions.

### Table 1.9-2. Requirements for Launch Vehicle Command & Control of Payload

<table>
<thead>
<tr>
<th>Function</th>
<th>Transient Characteristic</th>
<th>Voltage ((0\text{-pk, VDC}))</th>
<th>Current ((0\text{-pk, Amps}))</th>
<th>Impedance ((\Omega))</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ordnance</td>
<td>75ms pulse width max</td>
<td>NA</td>
<td>5.0</td>
<td>Payload Input: 1.0</td>
<td>&gt;90%</td>
</tr>
<tr>
<td>Relay Discretes</td>
<td>40ms min to infinite pulse width</td>
<td>40 VDC max at payload input</td>
<td>0.5 max</td>
<td>Vehicle Output Open: &gt;1 Meg Closed: &lt;1 Meg</td>
<td>&gt;90%</td>
</tr>
<tr>
<td>Opto-Isolated Discretes</td>
<td>&quot;On&quot; for 3.0 seconds min</td>
<td>NA</td>
<td>20mA source max</td>
<td>NA</td>
<td>&gt;90%</td>
</tr>
</tbody>
</table>

C. Telemetry (Authorize by: L-20 Months)

Telemetry throughput exists in the Stage 3 MUX telemetry stream to downlink data from 8 analog and 8 discrete points at 25 Hz. The remaining available sensors are sampled at 5 Hz.

1. Analog

For Non-HAPS Missions, 16 High-range analog inputs shall be provided on the Stage 3 MUX. These points can be set to sample a range of 0 to 5 V or -79 to 162 mV with 8 bit resolution. For a -15 to 40 V range, an interface box
will be provided to scale and bias the signal to the 0 - 5 V range. Input impedance shall be greater than 100 KΩ, and shielding coverage shall be greater than 90%. These points can be sampled at 5 Hz or 25 Hz.

2. Discrete

Ten discrete inputs shall be provided. The discrete is actuated by “switch closure” on the payload side, allowing current to flow through a 10 KΩ pull-up resistor and MCT9001 Opto-isolator to ground. With no active components on the payload side, current flow shall be 0.5 mA. These points can be sampled at 5 Hz or 25 Hz. Up to four, similar discretes shall also be provided in the flight computer, sampled at 5 Hz.

<table>
<thead>
<tr>
<th>Function</th>
<th>Measurement Resolution</th>
<th>Voltage (0-pk, VDC)</th>
<th>Current (0-pk, Amps)</th>
<th>Payload Input Impedance (Ω)</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Analog</td>
<td>20 mV</td>
<td>-15 to +40</td>
<td>N/A</td>
<td>100 K</td>
<td>&gt;90%</td>
</tr>
<tr>
<td>Discrete</td>
<td>N/A</td>
<td>+5</td>
<td>10 mA</td>
<td>100 K (Switch Open)</td>
<td>&gt;90%</td>
</tr>
</tbody>
</table>

* Payload shall provide an open collector output that requires being tied to the 5 VDC launch vehicle bus via a pull-up resistor.

Table 1.9-3. *Payload Telemetry Monitoring Requirements*
2.0 MISSION–UNIQUE SUPPORT

2.1. RESERVED

2.2. SPECIAL CONTAMINATION CONTROL OPTIONS:

The following contamination control requirements shall be met on a non-standard basis:

A. Fairing Environment: (Authorize by: L-12 Months)

The Contractor shall purge the payload fairing environment with level 10,000 per FED-STD-209E (or better) air. This includes installing, operating, monitoring, and cleaning special HEPA-and carbon-filtered conditioned-air supply systems during four phases of integrated operations:

1. Inside the integration facility (Vehicle Assembly Building);
2. During transport to Hot Pad;
3. During Hot Pad ground operations;
4. During Orbital Carrier Aircraft mated operations.

B. Fairing Internal Surface Cleaning: (Authorize by: L-12 Months)

The Contractor shall clean, certify, and maintain internal surfaces of the payload fairing to level 500A or 600A per MIL-STD-1246C. This includes all efforts to support increased levels of precision cleaning of the internal fairing surfaces prior to payload encapsulation; additional surface cleanliness measurements to verify surface cleanliness; and additional handling controls to maintain cleanliness. This Non-Standard Service shall always be exercised in concert with Non-Standard Service Section 2.2.C below.

C. Payload/Vehicle Integration Environment: Class 10,000 (Authorize by: L-10 Months)

The Contractor shall ensure that the payload/vehicle integration environment shall be cleaned, certified, and maintained at FED-STD-209E Class 10,000, to support payload mate through fairing closeout operations. This includes as minimum:

- preparation and verification of a FED-STD-209E Class 10,000 cleanroom,
- assembly of the payload separation system,
- final assembly of the payload fairings and other components, and
- integration of the launch vehicle and payload in the cleanroom.

D. Hydrocarbon Monitoring (Authorize by: L-12 Months)

The Contractor shall provide continuous monitoring and recording of hydrocarbon levels during all integrated payload/vehicle operations. This service includes: the installation, calibration, and frequent round-the-clock monitoring of fixed and portable hydrocarbon (VOC) detectors in the Vehicle Assembly Building. The Contractor shall provide computer-controlled
contamination data recorders and alarming systems, for continuous capture of hydrocarbon level data and remote warning of excessive levels.

E. Instrument Purge System: (Authorize by: L-18 Months)

The Contractor shall provide a single instrument purge system using MIL-STD 27401C Grade B GN2 purge gas. This instrument purge system shall use a 0.5 micron filter (or better), and provide from 0 to 25 SCFM. This instrument purge system shall be provided from final fairing closeout until launch or reconnect to NASA-provided ground purge following abort and return to base from which the CCA departed. The Contractor shall provide the purge gas and any equipment (e.g. hoses, fittings, regulators) to reestablish the purge should the OCA be diverted to an alternate landing site. After final fairing closeout, the payload is limited to a maximum of 354 kg (780 lbm) of GN2 throughout captive carry. The purge system shall provide a quick disconnect fitting that is disconnected at fairing separation. The quick disconnect system shall not exert more than 25 lbf on the payload fitting. The Contractor shall certify the purge system cleanliness to MIL-STD-1246C Level 100A prior to use. The Contractor shall certify and maintain the instrument purge system hydrocarbon levels at or below 5.0 ppm. The Contractor shall monitor, maintain, and replace purge gas supplies as they are used, while maintaining all cleanliness requirements.

F. Reserved

2.3. ENVIRONMENTAL ASSESSMENT DATA BOOK (Authorize by: L-20 Months)

For those missions in which the payload contains a radioactive component, such as a Radioisotope Heating Unit (RHU) or Radioisotope Thermoelectric Generator (RTG), an Environmental Assessment Data Book is required from the launch vehicle Contractor to support payload reviews. The Contractor shall develop a comprehensive environmental assessment data book for any mission that incorporates radioactive sources into the design of the payload. The Data Book shall contain a detailed description of the launch vehicle, the launch site infrastructure, the trajectory profile and instantaneous impact point history, and descriptions of possible accident scenarios, their environments, and probabilities of occurrence. The Contractor shall implement the criteria of KSC HBK 1860.1C “Radiation Protection For Ionizing Radiation” and perform the necessary coordination to ensure compliance with EWR 127-1 Section 3.9. In addition, the Contractor shall coordinate with the US Air Force Range Safety Office at either CCAS or VAFB, or with the NASA Safety Office at Wallops Flight Facility. The Contractor shall incorporate the environmental findings into the data book, which defines the potential physical and environmental hazards for each radioactive source. The environmental assessment data book will also address the potential for hazards to the general public in the event of a launch-related accident or incident where the payload does not reach orbit. The Contractor shall work closely with NASA to develop a credible accident response plan, which provides NASA management with casualty expectations under various payload or launch vehicle failure scenarios.
2.4. Payload Fit Check Support (Authorize by: L-18 Months)

The Contractor shall provide a facility to perform a fit check between the payload and the flight fairing. NASA will transport the payload or a full-scale model of the payload to this facility. All personnel, procedures, and any specialized handle equipment (e.g. lifting sling, turn over cradles) required to handle the payload will be provided by NASA. The Contractor shall provide for NASA’s use standard handling equipment (e.g. forklifts, cranes) to unpack, move, or position the payload. In addition, all required GSE to perform the fit check shall be provided by the Contractor. All personnel required for the operations concerned with launch vehicle hardware handling shall be provided by the Contractor. The Contractor shall use the integrated procedures for payload/fairing operations that are developed for use at the launch site. Unless otherwise directed, the payload’s flight PAF shall be used for performing the fit check.

Up to ten critical clearance locations, as identified by the NASA, shall be measured. The accuracy of the measurements shall be dependent on the position of the critical clearance locations.
3.0 OTHER NON-STANDARD SERVICES

3.1. PAYLOAD CONTINGENCY SERVICES

In the event of a payload contingency that requires the payload to be deintegrated from the launch vehicle flight hardware, the Contractor shall be capable of performing all the necessary activities and services for each or a combination of the following conditions described below.

3.1.1 After Vehicle Rollout, but prior to Vehicle/OCA mate (Authorized as Required)

The Contractor shall provide the support necessary for rolling the vehicle back to the VAB (Building 1555), removing the payload fairing, demating the spacecraft, re-mating the spacecraft, and rolling the vehicle back to the Hot Pad. Safe-to-Mate checks will be repeated, as well as a Single Flight Simulation, as part of the process to re-validate the payload electrical interfaces.

3.1.2 After Vehicle/OCA mate, but prior to OCA Wheels Up (Authorized as Required)

The Contractor shall provide the support necessary for returning the Launch vehicle back to the Hot Pad, demating the Launch vehicle from the OCA, transporting the vehicle back to the VAB (Building 1555), removing the payload fairing, demating the spacecraft, re-mating the spacecraft, transporting the vehicle back to the Hot Pad, re-performing the CST, and re-executing the launch checklist, up to the point of takeoff. Safe-to-Mate checks will be repeated, as well as a Single Flight Simulation, as part of the process to re-validate the payload electrical interfaces.

3.1.3 Ferry-Back-To-VAFB from CCAS and subsequent return to CCAS (Authorized as Required)

The Contractor shall provide the support necessary for returning the Launch vehicle back to the Hot Pad at VAFB from the CCAS Hot Pad. This may be followed by additionally-authorized operations at VAFB, such as demating of the Launch vehicle from the OCA (reference Non-Standard Service Sections 3.1.1 or 3.1.2). After such additionally authorized activities have been completed, the Contractor shall provide for only ferry flight activities back to CCAS, commencing with the opening of the ferry-flight checklist at VAFB and ending with OCA engine shutdown at CCAS. Additional authorization may be required for repeated activities at CCAS.

3.1.4 Ferry-Back-To-VAFB from WFF and subsequent return to WFF (Authorized as Required)

The Contractor shall provide the support necessary for returning the Launch vehicle back to the Hot Pad at VAFB from the WFF Hot Pad. This may be followed by additionally authorized operations at VAFB, such as demating of the Launch vehicle from the OCA (reference Non-Standard Service Sections 3.1.1 or 3.1.2). After such additionally authorized activities have been completed, the Contractor shall provide for only ferry flight activities back to
WFF, commencing with the opening of the ferry-flight checklist at VAFB and ending with OCA engine shutdown at WFF. Additional authorization may be required for repeated activities at WFF.

3.1.5 Ferry-Back-To-VAFB from Kwajalein Missile Range and subsequent return to Kwajalein Missile Range (Authorized as Required)

The Contractor shall provide the support necessary for returning the Launch vehicle back to the Hot Pad at VAFB from the KMR Hot Pad. This may be followed by additionally authorized operations, such as demating of the Launch vehicle from the OCA (reference Non-Standard Service Sections 3.1.1 or 3.1.2). After such additionally authorized activities have been completed, this non-standard service will provide for only ferry flight activities back to KMR, commencing with the opening of the ferry-flight checklist at VAFB and ending with OCA engine shutdown at KMR. Additional authorization may be required for repeated activities at KMR.

3.2. 14 TO 29 DAYS SCIENTIFIC LAUNCH PERIOD WITH ANNUAL REOPENING

(Authorize by: LSTO)

The Contractor shall provide all launch services described in the SOW to accommodate a scientific launch period as small as 14 days but not greater than 29 days in duration. The Contractor's price shall remain valid for missions in which the payload scientific launch period reopens no more than one-year after the closing of the original scientific launch period.

3.3. SPECIAL TEST CONNECTOR FOR PAYLOAD (Authorize by: L-12 Months)

The service includes a one-time effort to design and implement a test connector for use by the Payload into the avionics section and avionics harness. The standard electrical interface, 4W1 harness and avionics assembly drawings shall be modified to include the following configuration and shall be implemented as a no-cost, non-standard electrical service after initial use. The MS27474T-16F-42S (socket receptacle) test connector shall be mounted on the 270° RCS bracket and shall be accessible through the RCS door until final fairing closeout. The Contractor shall provide the mating connector (MS27484T-16F-42P) to the payload. The following pins shall be provided at the test connector mounted on the 270° RCS bracket: 16 pins for 8 TSP (See Table 3.3-1); or 10 pins for 5 TSP and 6 pins for 2 MIL-STD-1553 connections (2 pins for shield passthrough and 4 pins for dedicated MIL-STD-1553 bus impedance-controlled cable) (See Table 3.3-2). The test connector shall be disconnected by the payload and capped (resistive termination of 1553 channels) by the Contractor during fairing closeout at the hotpad.

<table>
<thead>
<tr>
<th>Function</th>
<th>Number</th>
<th>Peak Voltage (V_{pe})</th>
<th>Continuous Current (Amps)</th>
<th>Resistance from Connector to Payload (Ω)</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power or Data</td>
<td>8 TSP</td>
<td>60</td>
<td>3.0</td>
<td>2.5</td>
<td>90%</td>
</tr>
</tbody>
</table>

Table 3.3-1. Payload Test Connector Electrical Interface Minimum Requirements (8 TSP)
Table 3.3-2. **Payload Test Connector Electrical Interface Minimum Requirements (5 TSP and 2 MIL-STD-1553B Channels)**

<table>
<thead>
<tr>
<th>Function</th>
<th>Number</th>
<th>Peak Voltage $V_{dc}$</th>
<th>Continuous Current (Amps)</th>
<th>Resistance from Connector to Payload ($\Omega$)</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power or Data</td>
<td>5 TSP</td>
<td>60</td>
<td>3.0</td>
<td>2.5</td>
<td>90%</td>
</tr>
</tbody>
</table>

3.4. RESERVED

3.5. STAGE 2 ONBOARD CAMERA (Authorized by: L-15 Months)

The Contractor shall provide a real-time second stage video system and image data. This system is completely self contained and has a dedicated battery, RF system for transmission of signal, and two cameras for forward and aft views of the launch vehicle. The cameras shall be capable of switching views as commanded by the flight computer to capture critical staging events and fairing separation. The onboard camera shall be able to switch views from the LPO control station while in captive carry. The Contractor shall evaluate Sun angles and time of day lighting levels to insure acceptable video quality from the start of countdown until second stage separation.

There shall be a 6 pound performance impact on the launch vehicle performance to orbit using this onboard camera.

3.6. SERIAL TELEMETRY (Authorize by: L-18 Months)

The Contractor shall provide a polled payload Serial Telemetry Interface to incorporate payload telemetry data into the launch vehicle telemetry downlink. The RS-485/RS-422 interface employs a serial link between the payload and the launch vehicle flight computer. The flight computer shall interrogate the payload at a nominal 40 millisecond rate and receive payload data that will be interleaved into the downlink telemetry stream. The telemetry data volume shall not exceed 675 bytes/sec. This service shall include any additional harnessing or connectors required to make the system operational.

3.7. STATE VECTOR TRANSMISSION FROM LAUNCH VEHICLE (Authorize by: L-12 Months)

The Contractor shall utilize the serial telemetry link with the payload to transmit a state vector from the flight computer directly to the satellite. This state vector shall be in a format specified in the launch vehicle documentation. Accuracy of the state vector shall be that of the launch vehicle inertial navigation system. The Contractor shall provide a serial RS-485/RS-422 data link with the payload to transmit vector state of vehicle prior to payload deployment.

3.8. 40-PIN PASS-THROUGH HARNESS (Authorize by: LSTO)

The Contractor shall incorporate 20 twisted shielded pairs of wires from the payload interface plane to the OCA as defined in Table 3.8-1 below.
<table>
<thead>
<tr>
<th>Function</th>
<th>Number</th>
<th>Peak Voltage (Vdc)</th>
<th>Continuous Current (Amps)</th>
<th>Resistance from ASE to Payload (Ω)</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power or Data</td>
<td>20 TSP</td>
<td>60</td>
<td>3.0</td>
<td>2.5</td>
<td>90%</td>
</tr>
</tbody>
</table>

Table 3.8-1. Payload-to-ASE/ GSE Electrical Interface Minimum Requirements

This option may not be exercised in combination with option identified in Non-Standard Service Section 1.9.A, "Increased Capacity Payload GSE Interface", due to electrical connector limitations.

3.9. ULTRA-HIGH FREQUENCY (UHF) UPLINK (Authorize by: L-16 Months)

The Contractor shall provide a dedicated UHF uplink (receive only) between the OCA and range to allow communication from the NASA GSE on the ground to the NASA ASE on the OCA during captive carry flight (or during pre-launch testing). Support includes calibration and installation of the UHF receivers in the OCA, integration with the NASA ASE, end-to-end ground testing of the receivers and uplink support during the vehicle captive carry flight to the drop point. The harness shall be routed through to a floor panel just below the LPO rack. On the OCA side, the rack-mounted panel that fits into the LPO station in the space reserved for payload applications will plug into the floor panel connector. NASA will be responsible for supplying the rack-mounted panel to the LPO station and all ASE to the interface with the Contractor provided cable to the payload. Connector type, contact arrangement, and physical dimensions shall be specified in the Interface Control Document.

3.10. PAYLOAD ISOLATION SYSTEM (Authorize by: L-20 Months)

The Contractor shall provide a Payload Isolation System that shall lower the fundamental frequencies of the payload to avoid dynamic coupling with the vehicle fundamental frequencies at vehicle/OCA separation.

3.11. TPAF (38", 23" OR 17") (Authorize by: L-18 Months)

The Contractor shall design and fabricate a TPAF. The TPAF shall have the appropriate electrical and mechanical interfaces to simulate a flight PAF. The diameter of the TPAF will be specified by NASA upon ordering this non-standard service.
4.0 SECONDARY/CO-MANIFESTED PAYLOADS SERVICES, COMPATIBILITY ASSESSMENTS, AND HARDWARE

This section summarizes the services, support, and hardware required to launch a NASA Secondary/Co-Manifested Payloads. Additional studies or payload compatibility assessments can be ordered by NASA as outlined in Articles H-14, H-15, or H-16.

4.1. SECONDARY/CO-MANIFESTED PAYLOADS SERVICES

The Contractor shall allow for sufficient clearance between the secondary payload envelope and the launch vehicles so that no fit checks of the secondary payload are required.

Weight and CG constraints

The performance available for a secondary payload shall include assessments for all mission unique flight hardware necessary for that configuration. This shall include but not be limited to the mass of DPAF hardware, additional separation systems, or harnessing. Figure D-5 shall define the mass vs. c.g. envelope for the secondary payload in the DPAF configuration. In this configuration, the c.g. is constrained by the shear capability of the 17" bolted interface. In the load-bearing configuration, the secondary c.g. location must be less than 26" from the interface plane.

Coupled Frequency

The Contractor shall analyze the coupled frequencies as part of the Payload Accommodation Study (Non-Standard Service Section 4.2.3). Table 4.1-1 outlines the following payload characteristics are required for secondary payload.

<table>
<thead>
<tr>
<th>Secondary Payload Characteristic</th>
<th>Non-Separating DPAF</th>
<th>Non-Separating Load Bearing</th>
<th>Separating DPAF</th>
<th>Separating Load Bearing</th>
</tr>
</thead>
<tbody>
<tr>
<td>Coupled Frequency</td>
<td>Total Payload Stack Frequency &gt; 20 Hz</td>
<td>Total Payload Stack Frequency &gt; 20 Hz</td>
<td>Frequency &gt; 20 Hz</td>
<td>Total Payload Stack Frequency &gt; 20 Hz</td>
</tr>
</tbody>
</table>

Table 4.1-1. Secondary Couple Frequency
Payload Volume

The secondary payload volume for the DPAF configuration is defined in Figure D-6. The height of the DPAF shall be adjusted to tailor the height of the secondary payload as long as the envelope and environmental requirements for the primary payload are not violated. The secondary payload lateral dimensions are defined by the secondary payload separation tip-off motion. Adequate separation clearance margin is required between the payload and the DPAF structure. For the load bearing secondary configuration, the volume available for a secondary payload is highly dependent on the volume and separation system of the primary payload. For the load bearing secondary

Figure D-6. Dual Payload Attach Fitting (DPAF) Configuration.

Figure D-7. Load Bearing Secondary Configuration
configuration, a typical configuration is shown in Figure D-7. Actual
dimensions, clearance margins, and payload attachment fittings are unique for
each mission and shall be determined by the Contractor’s Secondary Payload
Accommodation Study.

Electrical Interface

The secondary payload shall have access to the remaining electrical services
described in Pegasus Exhibit 2 Section 2-7.

Load bearing secondary payloads will be responsible for furnishing a pass-
through harness to support the primary payload electrical interface with the
launch vehicle. This includes double shielded pyro event channels. The
Contractor shall provide G&H 1179 zero force connectors at the flight
separation planes, at the top and bottom of the secondary payload.

DPAF missions will use either an additional PDU or an ordnance delay to
provide the required separation between the primary and redundant bolt cutter
initiation.

The Contractor shall provide battery trickle charge capability through an
existing fairing access door until fairing closeout. NASA will be responsible for
providing charging equipment and cabling. In general, secondary payloads
shall have access for trickle charging up to approximately three hours before
OCA takeoff (approximately L-4 hours). Final fairing closeouts (access door
installation) would then be performed after primary payload pre-closeout
activities are complete.

Secondary Payload Mission Services

The secondary payload services shall apply to either a Co-manifested or
NASA Secondary Payload. These services shall provide for all integration
services required for delivering a secondary payload to orbit. These services
shall include as a minimum:

- additional payload integration activities (e.g., the preparation of, or
  modifications to, the payload ICD)
- coordination of additional payload range requirements
- CDRLs
- MIWG meetings
- additional mission team support, program, project, and engineering
  management
- field site support required to execute the launch

4.1.1. Co-Manifested Payload Mission Service (Authorize by: LSTO)

This service applies to any NASA Secondary Payloads that are Co-manifested
(Reference Article H-14) with a NASA Primary Payload. The Contractor shall
provide these services for separating and non-separating secondary payloads
as described in Non-Standard Service Sections 4.1. Secondary payload
technical capability and compatibility assessment studies are described and
ordered separately as a Non-Standard Service in Sections 4.2. The hardware
necessary to attach the primary and secondary payloads to the launch vehicle are described and ordered separately as a Non-Standard Service in Sections 4.3.

4.1.2. NASA Secondary Payload Service (Authorize by: LSTO)

This service applies to any NASA Secondary Payloads that are carried in surplus space without interfering with a non-NASA Primary Payload (Reference Article H-16). The Contractor shall provide this service for separating and non-separating secondary payloads as described in Non-Standard Service Sections 4.1. Secondary payload technical capability and compatibility assessment studies are described and ordered separately as a Non-Standard Service in Sections 4.2. The hardware necessary to attach the secondary payloads to the launch vehicle are described and ordered separately as a Non-Standard Service in Sections 4.3.

4.2. SECONDARY PAYLOAD COMPATIBILITY ASSESSMENTS

4.2.1. Payload Compatibility Assessment (Authorize as Required)

The Contractor shall perform all necessary compatibility analyses to verify that the NASA Secondary Payload does not impact the primary payload unacceptably. The Contractor shall document and submit these analyses, which shall include as minimum:

- combined coupled load analysis
- combined thermal analysis
- critical clearance analysis
- interface failure modes and effect analysis
- safety assessment
- mass properties report
- orbital performance estimate
- GN&C analysis

NASA will be responsible for providing any existing documentation or information about the NASA Secondary Payload, which may be necessary to complete this assessment.

4.2.2. Secondary Payload Mission Feasibility Study (Authorize as Required)

The Contractor shall conduct a study that assesses the excess performance and payload volume available to accommodate a NASA Secondary Payload. This performance assessment shall include an estimate of the performance decrement associated with any particular mission-unique hardware. The Contractor shall provide draft configuration drawings, which identifies payload volume available and any potential configurations.

4.2.3. Secondary Payload Accommodation Study (Authorize as Required)

The Contractor shall develop the design, implementation scheme, and general payload requirements for accommodating NASA Secondary Payloads. The
general payload requirements shall include, as a minimum: the mechanical interfaces, available mass, volume, and expected environments.

The Contractor shall prepare and submit parametric performance studies and load analysis. The Parametric Performance Studies shall examine the excess performance available as a function of the potential payload mass and orbit requirements. The Loads Analysis shall examine the secondary payload dynamic load environments, for various viable combinations of primary and secondary payload characteristics (e.g., mass properties, fundamental bending mode frequencies, and axial center of gravity position using given primary payload configuration).

For NASA Primary Payloads, NASA will supply the primary payload model used for the load analysis. The Contractor shall use a range of simplified secondary payload models that shall be integrated with the launch vehicle system model, and acceleration responses shall be evaluated. The results of this evaluation shall be presented to NASA.

4.3. SECONDARY PAYLOAD HARDWARE

4.3.1. Co-Manifested Payloads which use a DPAF

The Contractor shall provide a separation system to separate the secondary payloads. The separation event shall be sequenced, controlled, and verified by the launch vehicle. After the primary payload has separated, the 38” separation system shall separate the adapter cone from the cylinder. As identified in the ICD, the launch vehicle shall be orientated so that the 17” separation system can be initiated and place the secondary payload in the desired orbit. After each separation event, a CCAM shall be performed to avoid recontact with primary payload, adapter cone, and launch vehicle. The Contractor shall design, fabricate, and install all the components for one of the DPAF systems described.

4.3.1.1. DPAF - 23” Primary PAF, 17” Secondary PAF (Authorized by: LSTO)

In addition to the requirements outlined in Non-Standard Service Section 4.3.1, the Contractor shall include as a minimum: a 23” separation system (described in Pegasus Exhibit 2), adapter cone, 38” separation system (described in Pegasus Exhibit 2), structural cylinder to attach the 38” separation system to the launch vehicle, and 17” separation system/PAF (described in Non-Standard Service Section 1.3.1.1). Figure D-6 illustrates a typical DPAF - 23” Primary PAF, 17” Secondary PAF layout.

4.3.1.2. DPAF - 17” Primary PAF, 17” Secondary PAF (Authorized by: LSTO)

In addition to the requirements described in Non-Standard Service Section 4.3.1; the Contractor shall include as a minimum: a 17” separation system (described in Non-Standard Service Section 1.3.1.1), adapter cone, 38” separation system (described in Pegasus Exhibit 2), structural cylinder to attach the 38” separation system to the launch vehicle, and 17” separation system/PAF (described in Non-Standard Service Section 1.3.1.1).
4.3.2. Secondary Payload Hardware

4.3.2.1. DPAF for Separating Secondary - 17" PAF (Authorized by: LSTO)

This service includes all DPAF hardware necessary to accommodate a separating NASA secondary payload on a non-NASA mission. This DPAF utilizes a modified version of the 17" PAF attached to the aft end of the avionics structure to support the secondary satellite. Secondary interfaces are identical with the 17" PAF (Reference Non-Standard Service Section 1.3.1.1) interfaces. The secondary payload available volume is limited by tip-off rates and by the PAF chosen by the primary payload. Last, the Contractor shall comply with the requirements as outlined in Non-Standard Service Section 4.3.1 as well.

4.3.2.2. DPAF for Non-Separating Secondary – 17" PAF (Authorized by: LSTO)

This service includes all DPAF hardware necessary to accommodate a non-separating NASA secondary payload on a non-NASA mission. The DPAF described in Non-Standard Service Section 4.3.1 and Sections 4.3.1.1 or 4.3.1.2 shall be capable of supporting non-separating secondary payloads. The 38" separation system is moved down to the aft end of the cylinder. Upon initiation, the 38" separation system shall separate the cylinder/cone hardware form the launch vehicle. This shall leave the secondary payload exposed but still attached to the avionics section of the launch vehicle. After each separation event, a CCAM shall be performed to avoid recontact with primary payload, adapter cone, and cylinder.

4.3.3. Load Bearing Payloads PAF (Authorized by: LSTO)

The Contractor shall provide a separation system to separate the secondary payloads. The separation event shall be sequenced, controlled, and verified by the launch vehicle. After each separation event, a CCAM shall be performed to avoid recontact with primary payload and launch vehicle. The Contractor shall design, fabricate, and install one of the various secondary payload interfaces described below.

3-Point Load Bearing Secondary Payloads Interface

Contractor shall provide a dual-configuration 3-point DPAF that can support either a 23" or 17" PAF primary payload on top of the MicroStar-class secondary payload. The 3-point DPAF shall require a second Pyro Driver Unit to initiate separation of the three internally redundant non-explosive actuators on the 3-point DPAF cone. The Marmon clamp (or equivalent) interface bolt cutters and the adapter cone non-explosive actuators are initiated by the launch vehicle via the pass-through cable. The 3-point DPAF impact on launch vehicle performance to orbit shall be based upon the configuration chosen.
38" Load-Bearing Secondary Payloads Interface

The launch vehicle shall support a load-bearing secondary payload using the standard 38" PAF as the aft launch vehicle/payload interface. The usable payload volume defined for the 38" PAF shall be divided between the primary and secondary payloads. The primary payload attaches to the secondary payload using a 10", 17", 23", or 30" PAF. The launch vehicle pyro activation signals are routed through the secondary satellite to initiate primary satellite separation.

The Contractor shall work with the secondary payload provider to establish bracket-mounting locations if an interface cone is not used. Appropriate envelope length will need to be reserved for the separation system or PAF used.

Non-Separating Secondary Payloads

By removing the separation capability from the MicroStar bus, the 3-point DPAF shall be capable of supporting a non-separating secondary payload. This would not be a change in the configuration from the 3-point DPAF and the interfaces shall be identical.

Inert Carrying Module Interface

The launch vehicle shall have the capability of flying small non-separating payloads in the Inert Carrying Module (ICM), which is attached to the Stage 3/avionics section interface and the Stage 3 motor case. The ICM is cylindrical with an internal diameter of 3.3" and is variable in length.

4.3.4. Secondary Payload Mass Simulator (Authorized by: L-18 Months)

The Contractor shall construct a metallic (minimal outgassing as specified in the ICD) mass simulator of the secondary payload. Mass properties shall match those of the secondary payload within the tolerance specified by the ICD. The mass simulator shall also adhere to the requirements (e.g., contamination, cleanliness, interface requirements) identified in the ICD. The mass simulator shall not exceed the maximum dimensions allotted to the secondary payload by the mechanical ICD. The mechanical interface connections shall meet the interface requirements of the secondary PAF. Any other analysis or documentation required for replacement of the secondary payload with the mass simulator shall be completed by the Contractor.
TAURUS EXHIBIT 2
Mission and Performance Requirements and Capabilities

2 - 1 LAUNCH TIMING

A. Reference Data

1. Payload Access

Prior to launch vehicle arming, the payload customer can access the payload through
the fairing doors up until T-10 hours before launch. If launch vehicle arming
operations occur in two distinct phases (over the course of two days), the final
payload access to the payload occurs approximately at T-20 hours.

2. Critical Countdown Event

Launch Vehicle Stage 0 Thrust Vector Control (TVC) system pressurization occurs
when a pyrotechnically-operated valve is opened to allow on-board pressurant gas
into the vehicle TVC system. This is a critical countdown event that occurs within T-2
minutes (typically T-16 seconds).

B. Performance

1. Payload Access

Access to the payload through fairing access doors shall be supported beginning with
encapsulation at the PPF until 10 hours before launch.

2. Multiple Launch Attempts

Provided that sufficient time remains in the daily launch window defined in the ICD,
the launch system shall be capable of supporting multiple launch attempts in the
event of a launch abort.

3. 24 Hour Turn-around

The launch system shall be capable of a 24-hour turn-around and subsequent launch
attempt in the event of a launch abort occurs prior to Launch Vehicle Stage 0 TVC
system pressurization.

2 - 2 PAYLOAD ORBIT AND MASS REQUIREMENTS

A. Reference Data

1. Reference Earth Radius

All altitudes in this section are specified as circular orbit altitudes above an equatorial
Earth radius of 6378.14 km.
2. Vehicle Configuration Numbering Convention

Vehicle configuration numbering shall be as shown in Figure 2-1. Vehicle configurations 1XX0, 3XX0, and 4XX0 are not offered under this contract. Vehicle configurations 2110 and 2210 (63" and 92" payload fairings) are standard services under this contract. Vehicle configurations 2130 and 2230 (STAR 37 upper stage) are non-standard services (NSS) requiring exercise of NSS SubCLIN 1.4.1 (refer to Attachment D-T).

![Vehicle Configuration Numbering Convention](image)

Figure 2-1. Taurus Configuration Numbering Convention.

3. Attach Hardware Mass

All performance data in Table 2-1 assumes that the launch vehicle supplies the standard 38" (or 37") clampband separation system and associated attach hardware. The mass of the entire separation system (both the aft and forward portions) has been accounted for on the launch vehicle side of the interface. If a non-standard payload adapter is required, additional mass for that adapter, beyond that of the standard separation systems, will be charged to the payload.

4. Non-Standard Mission-Unique Hardware

All performance data provided in this section are for the baseline launch vehicle, and do not include assessments for non-standard mission-unique hardware, unless otherwise noted.

5. Fairing Deployment Dynamic Pressure Assumption

The nominal payload fairing deployment criterion (dynamic pressure of 0.005 psf) was used to generate all performance data in this section.

6. Guidance Reserve Assumption

The performance requirements in Tables 2-1 and 2-2 were generated with a 150 ft/s guidance reserve.
7. Direct-Injection Escape Trajectory Assumption

The performance requirements in Table 2-2 were generated assuming optimal, direct-injection insertion into the escape orbit (i.e., coast lengths and periapsis locations were unconstrained, and a 90 degree launch azimuth from CCAS was assumed).

8. Statistical Error Distribution for Insertion State Vector Dispersion Sources

A gaussian statistical error distribution shall be used for the insertion state vector dispersion sources described in this section unless otherwise noted.

9. Insertion State Vector

The insertion state vector shall be composed of the following mean orbital elements: apogee radius, perigee radius, inclination, and right ascension of ascending node (RAAN). The argument of periapsis (AP) shall also be included as an element of the insertion state vector if required by the LSTO.

10. Launch Sites And Azimuth Restrictions

For the performance figures quoted in Table 2-1, standard service launches from North VAFB (NVAFB) take place at Site 576E and assume a minimum azimuth of 205° to prevent overflight of South VAFB. Launches from a South VAFB (SVAFB) site assume a minimum azimuth of 158° to prevent overflight of Santa Cruz. East Coast launches from Cape Canaveral Air Station (CCAS) take place at Launch Complex 46 (LC-46) and assume flight azimuths between 65° and 110°. Figure 2-2 shows the available flight azimuths from Eastern and Western Range. Each of these assumptions was used in the preparation of the data shown in Table 2-1.

![Launch Range Capabilities Diagram](image)

**Figure 2-2. Launch Sites And Azimuth Restrictions Assumed In Table 2-1**

**Performance Data**
B. Performance

1. SELVS Performance Regions

The launch vehicle service shall be capable of supporting launches within the SELVS performance regions shown in Figure 2-3 through Figure 2-6.

![Diagram showing payload mass at different circular altitudes for 28.5 and 38 degree inclinations.]

Figure 2-3. SELVS Performance Region at 28.5° and 38° Inclinations
60 and 70 Degree Inclinations

Figure 2-4. SELVS Performance Region at 60° and 70° Inclinations

90 Degree Inclination

Figure 2-5 A. SELVS Performance Region at 90° Inclination
Figure 2-5 B. SELVS Performance Region at Sun-Synchronous Inclinations

Figure 2-6. SELVS Performance Region for Escape Missions

2. Inclinations Supported
The launch service shall support missions to any desired inclination, consistent with launch range restrictions.

3. Minimum Launch Vehicle Performance To Circular Orbits
Minimum launch vehicle performance to various reference circular orbits and inclinations shall be as shown in Table 2-1.
Table 2-1. Minimum Launch Vehicle Performance to Various Circular Orbits

4. Minimum Launch Vehicle Performance To Escape Trajectories

Minimum launch vehicle escape trajectory performance shall be as shown in Table 2-2. Vehicle escape trajectory missions require exercise of NSS SubCLIN 1.4.1 (STAR 37 Upper Stage).
Table 2-2. Minimum Launch Vehicle Escape Performance

5. Minimum Guidance Reserve Requirement

All SELVS mission trajectories shall include a nominal guidance reserve of no less than 150 ft/s, unless otherwise agreed to by NASA.

6. Insertion Accuracies

Required right ascension of ascending node (RAAN) and inclination insertion accuracies for Taurus missions are shown in Table 2-3. Insertion accuracies in Table 2-3 for RAAN are applicable only for missions with a required RAAN expressed in the LSTO. Insertion accuracy for RAAN is valid only for inclinations between 80 degrees and 100 degrees.

<table>
<thead>
<tr>
<th></th>
<th>Required 3-Sigma Insertion Error Limit Magnitudes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inclination</td>
<td>0.25 deg</td>
</tr>
<tr>
<td>Right Ascension of Ascending Node</td>
<td>0.5 deg</td>
</tr>
</tbody>
</table>

Notes
1. Applicable only for missions with inclinations between 80 and 100 degrees.

Table 2-3. Standard Insertion Accuracy Requirements

Inclination and RAAN limits in Table 2-3 include effects of all dispersions sources listed in Table 2-4.

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A. Reference Data

1. Minimum Lateral Frequency

A minimum lateral frequency of 25 Hz, with no flexible appendages, is assumed to determine the payload static envelopes. The payload is assumed to be hard mounted at the payload to payload attach fitting (PAF) interface plane.

2. Interface Plane

Deflections of the PAF interface plane have been accounted for on the launch vehicle side of the interface plane. The payload side of the PAF interface plane is assumed to be rigid.

3. Static Envelope

The static envelopes account for fairing and payload structural deflections. Payload dimensions due to manufacturing/design and tolerance stack-up shall be accounted

---

Figure 2-7. 63" Payload Fairing Static Envelope with 38.810" Diameter Payload Interface

Figure 2-8. 92" Payload Fairing Static Envelope with 38.810" Diameter Payload Interface
for within the static envelope. The payload shall not extend aft of the payload/launch vehicle interface plane, unless otherwise approved in the ICD. Other static envelope violations are subject to approval in the ICD.

B. Performance

1. Standard Service Payload Fairings

The Contractor shall provide two standard service payload fairing options offering either a 63" or 92" diameter.

2. Payload Fairing Static Envelopes With 38.810" Diameter Payload Interface

The 63" payload fairing static envelope with a 38.810" diameter payload interface shall be as shown in Figure 2-7. The 92" payload fairing static envelope with a 38.810" diameter payload interface shall be as shown in Figure 2-8. If nitrogen cooling is required by the payload, the payload envelope shall be locally reduced by 1" along cooling tube routing.

3. Payload Fairing Access Doors

The Contractor shall provide two RF-opaque payload fairing access doors. The standard access doors shall be located according to NASA requirements in the cylindrical section of the fairing and will be 30.5 cm x 30.5 cm (12" x 12") (63" fairing) or 45.7 cm x 61.0 cm (18" x 24") (92" fairing) in size. The Contractor shall assess the structural impacts associated with the door locations relative to each other (clocking and axial station separation) on a mission-unique basis. Additional standard size doors or non-standard size doors are available as a non-standard service (Refer to Attachment D-T, Section 1.2).

2 - 4 PAYLOAD SEPARATION ATTITUDE ACCURACY AND RATES

A. Reference Data

1. Separation System Reference Configuration

The separation system configuration assumes a 37" or 38" separation system.

2. Statistical Error Distribution for Tip-Off Rate Dispersion Analyses

The statistical error distribution for tip-off rate dispersion analysis assumes gaussian distributed error sources unless otherwise noted.

3. Payload Tip-Off

Payload tip-off is the angular velocity imparted to the payload upon separation due to an uneven distribution of torques and forces.

B. Performance

1. Payload Pre-Separation Pointing Accuracy

The Contractor shall provide a total payload pre-separation pointing accuracy of ±1
degree per axis. However, for sun-pointing missions, the launch service shall provide a total payload pre-separation pointing accuracy of ±2 degrees per axis. NASA will specify either an inertially-fixed or spin-stabilized attitude in the ICD.

2. Payload Deployment

The launch vehicle shall provide the desired initial payload attitude prior to payload separation. If required, this capability shall also be used to incrementally reorient the launch vehicle for the deployment of multiple spacecraft with independent attitude requirements.

3. Payload Separation Tip-Off Rate Analysis Dispersion Sources

The Contractor shall perform a mission-unique tip-off analysis for payload.

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2 - 5  PAYLOAD SPIN/DESPIN

A. Reference Data

1. Spin
Spin is rotation about the longitudinal axis in either direction (clockwise (CW) or counter-clockwise (CCW)).

B. Performance

1. Spin Capability
The Contractor shall provide a spin capability ranging from 0 to 90 degrees/sec (15 RPM).

2. Spin-Rate < 1 RPM
The launch vehicle shall be capable of controlling spin-rates ranging from 0 to 6 degrees/sec to within ±0.4 degrees/sec for any payload prior to separation.

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3. Spin-Rate > 1 RPM, <15 RPM
The launch vehicle shall be capable of controlling spin-rates above 6 degrees/sec up to 90 degrees/sec to within ±1.5 degrees/sec for any payload prior to separation.

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2 - 6  MECHANICAL INTERFACE

A. Reference Data

1. General Description
The separation system is a marmar clamp band design that employs two aluminum interface rings that are clamped by dual, semi-circular stainless steel clamp bands with aluminum clamp shoes. Each of the two retention bolts is severed by a redundantly-initiated SDI bolt cutter. Upon band release, the movement and parking of each band is controlled by a set of four extraction spring assemblies and two band catchers that are designed to prevent recontact with the upper ring. Separation velocity is provided by up to eight matched spring actuators.

2. 38” and 37” Separation System Similarity
The two standard service mechanical interfaces (38” and 37” separation system) vary only in their forward interface ring geometry and have the same structural and separation performance. The 38” separation system is based on the 37” separation system (Figure 2-9) design.
B. Performance

1. Hardware and Integration Services

The Contractor shall provide all hardware and integration services necessary to attach non-separating and separating payloads to the launch vehicle. All attachment hardware shall contain locking features consisting of locking nuts, inserts, or fasteners. For the 38" interface, the Contractor shall provide identical bolt patterns for both separating and non-separating mechanical interfaces. Implementation of a non-separating payload requires use of a non-standard service as defined in Taurus Attachment D-T.

The separation ring to which the payload attaches shall be supplied with through holes. The Contractor shall provide attachment bolts to this interface and can be inserted from either the launch vehicle or the payload side of this interface. The Contractor shall provide the integration ring and all necessary attachment hardware.

2. Standard Separating Mechanical Interfaces

NASA may select a payload separation system with a 38.81" diameter payload bolted interface (known as the 38" separation system) or a 37.15" diameter payload bolted interface version of the separation system (known as the 37" separation system), as a standard service. The 38" and 37" separable payload interface flanges are defined in Figure 2-10.
3. Separation System Debris

The payload shall be protected from debris generated by the separation system. The payload separation system shall use a single, redundantly initiated bolt cutter to sever each of two retention bolts. To mitigate the ejection of any debris that may be generated from the bolt cut, the Contractor shall install a polymer "boot" over the end of the bolt cutter body and the adjacent length of the tensioning bolt.

The separation system shall function in a manner that prevents any recontact with the payload, including Contractor-provided attach hardware on the payload, by the upper stage or any element of the separation system once separation has been initiated.

4. Separation System Structural Capability

The 37" and 38" standard separation systems shall maintain the structural capability defined in Figure 2-11.

![Figure 2-11. 38" and 37" Separation System Structural Capability](image-url)
5. Tailorable Separation Springs For Payload CG Offset Compensation

The Contractor shall tailor separation spring energies, if required by NASA, to compensate for payload lateral center of gravity offsets.

6. Interface Drill Tool

The Contractor shall provide a drill tool to the payload for precision drilling of the payload/launch vehicle interface. The Contractor shall coordinate delivery of the tool to NASA.

7. Payload Separation System Fit Check

The Contractor shall support a fit check of the upper ring of the flight separation system with the payload. The upper ring separation connector bracketry shall be installed so that the separation connector harnesses can be fit checked. The Contractor shall provide all hardware, procedures, and personnel to support the separation system fit check. The payload will provide the facility and the ground support equipment, procedures and personnel to support handling of the payload.

8. Launch Site Installation

The Contractor shall provide the launch site integration of the separation system to the payload and the launch vehicle. The payload is responsible for providing all necessary ground support equipment, procedures, and personnel associated with handling of the payload.

9. Test Payload Attach Fitting (TPAF) for Dynamic Release Test Support

The Contractor shall provide an existing TPAF including a standard payload separation system (37" or 38" as required by the ICD), a standard payload cone and all required attach hardware. The Contractor shall provide two technical support periods of up to three consecutive days for each period to tension (or de-tension) the separation system and integrate (or de-integrate) it with the payload and the payload cone. Technical support and all required expendable hardware/ordnance shall be provided by the Contractor to perform two consecutive dynamic release tests.

2 - 7 ELECTRICAL INTERFACE

A. Reference Data

1. Electrical Interface

The standard electrical interface is defined as the separation plane of the electrical power, control and telemetry connector and the pyrotechnic connector.

B. Performance

1. Electrical Interfaces

The pin allocations for the standard and non-standard payload electrical interfaces provided by the launch vehicle shall be as defined in Table 2-5 and Figure 2-13. Table 2-5 describes the 89 standard conductors available from the payload to the payload GSE via the vehicle T-0 umbilical. The electrical interface shall be capable
of providing an interface between the payload and the payload GSE during launch processing until launch. As many of the 89 GSE pass-through wires as desired may be used as continuity sense loops allowing the payload to sense separation at either the T-0 umbilical interface or the vehicle/payload separation plane.

A block diagram of the standard launch vehicle electrical interface capabilities, including the 89 standard GSE pass-through wires, shall be as shown in Figure 2-13.

<table>
<thead>
<tr>
<th>Function</th>
<th>Number of Wires</th>
<th>Peak Voltage (Vdc)</th>
<th>Continuous Current (Amps)</th>
<th>Resistance from GSE to Payload (Ω)</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power</td>
<td>8</td>
<td>600</td>
<td>13</td>
<td>1</td>
<td>100%</td>
</tr>
<tr>
<td>Power, Data(^1), or Continuity Sense</td>
<td>81</td>
<td>600</td>
<td>10</td>
<td>10</td>
<td>100%</td>
</tr>
</tbody>
</table>

\(^1\) Data lines shall be twisted shielded pairs unless otherwise required by the payload

Table 2-5. Payload to GSE Electrical Interface Minimum Wire Requirements
Figure 2-13. Payload Electrical Interface Block Diagram.

2. GSE Installation

The payload umbilical is terminated at the Launch Equipment Van (LEV). The Contractor shall provide space in the LEV for payload-provided equipment and power supplies needed for pre-launch payload checkout and command. The Contractor shall provide a fiber optic link between the Launch Support Van (LSV) and the LEV to allow payload equipment in LEV to be controlled remotely from the payload console in the LSV. The Contractor shall provide any or all of the payload umbilical signals directly to the payload LSV console via fiber optic link from the LSV to the LEV. The Contractor shall provide space in the LSV payload console for payload-provided equipment and power supplies needed for pre-launch payload checkout and
command.

3. Interface Hardware

The Contractor shall implement the electrical interface across the separation plane via two 56-pin G\&H 1179 connector assemblies. Each assembly shall include a separation connector with strain relief backshell and all required mounting bracketry. The Contractor shall provide the payload side of the interface connectors for the payload flight harness 4 months prior to launch. The payload will be responsible for integrating these connectors to the payload flight harness forward of the interface plane. The Contractor shall then integrate the payload flight harness to the separation system 1 month prior to launch. The Contractor shall provide the matching interface connectors. All interface wires shall be shielded for EMI protection.

The launch vehicle flight harnesses and payload-provided harness shall be integrated with the flight separation system and shall be available no earlier than 1 month prior to launch.

4. Payload to GSE Dual-Serial Communication

The Contractor shall provide a dual serial RS-485/RS-422 data link, as shown in Figure 2-13, to the payload for remote monitoring of payload functional and safety inhibit status prior to launch. This data link is provided through the T-0 umbilical.

5. Payload Discrete Sequencing Commands

The Contractor shall provide discrete sequencing commands to the payload. These commands shall be optically-isolated pulses of programmable lengths in multiples of 40 milliseconds. The Contractor shall provide five command line pairs, each capable of multiple pulses, for the payload. The payload is responsible for supplying the voltage source (≤ 40 VDC) and limiting the current to 500 milliamps (mA) nominal in a fashion similar to using a dry contact relay.

6. Payload Status Monitoring

The Contractor shall provide payload analog telemetry during ground processing (limited to launch vehicle power on-times), checkout, and launch. Six payload analog telemetry signals, per Figure 2-13, shall be accommodated in the launch vehicle telemetry stream. Supported sensor types shall include breakwires, bi-level discretes, and active and passive analog telemetry points. The standard service shall support configuring each channel on a mission-unique basis. The status of the payload telemetry points and separation breakwires shall be downlinked via the vehicle PCM telemetry stream during flight.

7. Payload Pyrotechnic Initiator Driver Unit

The Contractor shall provide four redundant pairs of 75 ms pulses at 5 amps for use by the Payload. Initiation of pyro events shall be sequenced by the Mission Data
Load (MDL) and commanded by the flight computer.

8. Provisions To Prevent Static Charge On The Payload Umbilical Connectors

The Contractor shall establish and maintain controls to prevent charge build-up on all electrical connectors during ground processing.

2 - 8 PAYLOAD ENVIRONMENT

A. Reference Data and Assumptions

1. Payload Separation System Assumption

The payload environments defined below apply to the standard service vehicle configuration employing the 63" diameter fairing and the standard 37" or 38" payload separation system. They apply equally to an otherwise standard vehicle configuration employing the 92" diameter fairing. The payload environments associated with the use of alternate separation systems, a non-separating payload interface, multiple payload attach fittings or alternative upper stages will differ from those presented below.

2. Payload Acoustic & Random Vibration Environment

Payload peak acoustic environments occur at lift-off, during transonic flight, and as the vehicle approaches maximum dynamic pressure. The transmission of this energy and its conversion into a vibration response at the payload interface is highly sensitive to spacecraft mass, geometry and structural design. The spectrum presented in Figure 2-16 is based upon analysis of the Taurus system with a full-sized rigid payload having typical absorption characteristics. Levels are expected to be within 6 dB of the peak levels shown in Figure 2-16 for 5.0 seconds at liftoff and for approximately 30 seconds during peak aeroacoustic periods (transonic through maximum dynamic pressure).

The worst-case payload random vibration environment is created by acoustic noise generated during lift-off, during transonic flight, and as the vehicle approaches maximum dynamic pressure.

3. Payload Shock Environment

The flight limit levels are derived from ground separation test data and analytical predictions for the vehicle and payload separation systems. The maximum predicted pyroshock levels associated with non-standard separation systems may vary significantly from those presented in Figure 2-18.

4. Payload Static Acceleration Environment

Acceleration levels for payloads are established by combining predicted steady state and transient accelerations as well as reviewing past flight measurements. Steady-state accelerations are predicted using the trajectory simulation codes POST and STEP.

5. Quasi-Static Acceleration Environment

Flight results and coupled loads analysis have been used to develop sinusoidal input
spectra for future payloads. Sinusoidal excitation occurs at lift-off and during the Stage 0 and Stage 1 resonant burn events.

The payload design load factors shown in Figure 2-19 assume a fixed-base payload first bending mode frequency above 25 Hz. The load factors in Figure 2-19 also assume payload axial natural frequencies such that the coupled vehicle/payload system resonant frequencies lie between 35 and 45 Hz or are greater than 75 Hz.

Resonant burn tends to generate the greatest axial response in payloads, with lift-off generating the next largest axial responses. Gust and buffet loading transients occur during the transonic portion of the flight adding an element of lateral excitation to the resonant burn induced axial excitation.

B. Performance

1. Payload Acoustic Environment

Composite maximum fairing interior noise levels (i.e. “maximum flight”) shall be as shown Figure 2-16. A +6 dB spectrum based on these max-flight levels for 60 seconds minimum duration is recommended for payload testing.

![1/3 Octave Band Sound Pressure Level (dB)](image)

<table>
<thead>
<tr>
<th>Frequency (Hz)</th>
<th>1/3 Oct Band SPL (dB)</th>
<th>Frequency (Hz)</th>
<th>1/3 Oct Band SPL (dB)</th>
</tr>
</thead>
<tbody>
<tr>
<td>20</td>
<td>121</td>
<td>400</td>
<td>127</td>
</tr>
<tr>
<td>25</td>
<td>121</td>
<td>500</td>
<td>124</td>
</tr>
<tr>
<td>31.5</td>
<td>122</td>
<td>600</td>
<td>122</td>
</tr>
<tr>
<td>40</td>
<td>123</td>
<td>800</td>
<td>120</td>
</tr>
<tr>
<td>50</td>
<td>124</td>
<td>1000</td>
<td>119</td>
</tr>
<tr>
<td>63</td>
<td>124</td>
<td>1250</td>
<td>119</td>
</tr>
<tr>
<td>80</td>
<td>125</td>
<td>1500</td>
<td>116</td>
</tr>
<tr>
<td>100</td>
<td>125</td>
<td>2000</td>
<td>113</td>
</tr>
<tr>
<td>125</td>
<td>126</td>
<td>2500</td>
<td>110</td>
</tr>
<tr>
<td>160</td>
<td>126</td>
<td>3150</td>
<td>107</td>
</tr>
<tr>
<td>200</td>
<td>126</td>
<td>4000</td>
<td>104</td>
</tr>
<tr>
<td>250</td>
<td>126</td>
<td>OASPL</td>
<td>137.2</td>
</tr>
</tbody>
</table>

**Figure 2-16. Maximum Flight Level Payload Acoustic Environments.**

2. Payload Random Vibration Environment

Figure 2-17 defines the envelope of the maximum random vibration response levels at the payload interface. The spectrum defined in Figure 2-17 is limited in applicability to payload components mounted adjacent to the payload interface. It should not be considered as a base drive input to the entire payload assembly, nor should it be added to the combined acceleration loads. A duration of 60 seconds in each axis is
recommended for vibration testing of payload components mounted adjacent to the payload interface.

![Graph showing PSD levels vs frequency](image)

**Figure 2-17. Maximum Flight Level Payload Interface Random Vibration Levels.**

3. VAPEPS Analysis

As a standard service, the Contractor shall perform mission-specific analyses using a customer provided statistical energy model of the payload in a Vibro-Acoustic Payload Environments Prediction System (VAPEPS) format. Alternatively, an Orbital generated model of the payload based upon customer supplied geometric, mass properties and cross sectional properties data may be used in the analysis if the Payload does not have a VAPEPS model. The VAPEPS model shall be used to account for differences between flight and test configurations and to evaluate vehicle and payload configuration changes as to their impact on payload acoustics and random vibration response.

The scope of this effort includes several analyses as they relate to predicting and refining the predicted fairing interior acoustic noise levels, the payload interface random vibration levels and the response of major payload structures. Early in the mission cycle, the Contractor will make an estimate of these levels based upon predictions and flight results obtained for previous missions.

The Contractor shall deliver the following data products to the customer:

1) Memorandum documenting mission-specific fill factor and resulting acoustic levels (deliverable 4 weeks after receipt of relevant spacecraft dimensions),
2) VAPEPS model report including input and output decks, and
3) Memoranda in which the composite acoustic and random vibration response are derived.

Items 2 and 3 are deliverable 12 weeks after receipt of the payload model in VAPEPS format or customer supplied geometric, mass properties and cross sectional properties data.

4. Payload Shock Environment

The maximum shock response spectrum at the base of the payload from all launch
vehicle events shall not exceed the levels shown in Figure 2-18.

![Figure 2-18. Maximum Flight Level Payload Interface Shock Response Spectrum.](image)

5. Payload Acceleration Environment

The payload design limit load factors provided in Figure 2-19 shall envelope the worst case combined steady state and transient accelerations experienced by the payload for a standard service launch vehicle. The load factors in Figure 2-19 apply to the center of gravity (CG) of the payload.

<table>
<thead>
<tr>
<th>Axis</th>
<th>Maximum Acceleration (g's)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Axial</td>
<td>+10.0/-3.0</td>
</tr>
<tr>
<td>Lateral</td>
<td>+/-3.5</td>
</tr>
</tbody>
</table>

![Figure 2-19. Payload Design CG Limit Load Factors.](image)

The levels provided in Figure 2-20 shall envelope the maximum flight level payload interface quasi-sinusoidal vibration levels associated with propulsion system resonances. The levels specified from 45 to 65 Hz address Stage 0 resonant burn excitation, while those specified from 65 to 75 Hz address resonant burn excitation associated with the standard Stage 1 motor. Notching with respect to these levels shall be acceptable based upon mission specific coupled loads analysis. The payload customer is responsible for determining whether sine vibration testing is required.

<table>
<thead>
<tr>
<th>Frequency (Hz)</th>
<th>Axial (g's)</th>
<th>Lateral (g's)</th>
<th>Sweep Rate (Hz/Min)</th>
</tr>
</thead>
<tbody>
<tr>
<td>45-65</td>
<td>1.5</td>
<td>0.5</td>
<td>24 (Linear)</td>
</tr>
<tr>
<td>65-75</td>
<td>1.0</td>
<td>0.3</td>
<td>12 (Linear)</td>
</tr>
</tbody>
</table>

![Figure 2-20. Maximum Flight Level Payload Interface Quasi-Sinusoidal Vibration Environment Due to Propulsion System Resonances.](image)

The levels provided in Figure 2-21 shall envelope the maximum flight level payload interface quasi-sinusoidal vibration levels associated with the ignition and lift-off forcing
function. Notching with respect to these levels shall be acceptable based upon mission specific coupled loads analysis. The payload customer is responsible for determining whether sine vibration testing is required.

<table>
<thead>
<tr>
<th>Frequency (Hz)</th>
<th>Axial (g's)</th>
<th>Lateral (g's)</th>
<th>Sweep Rate (Hz/Min)</th>
</tr>
</thead>
<tbody>
<tr>
<td>10-25</td>
<td>2.0</td>
<td>0.5</td>
<td>4 (Oct/Min)</td>
</tr>
</tbody>
</table>

Figure 2-21. Maximum Flight Level Payload Interface Quasi-Sinusoidal Vibration Environment Due to Ignition/Lift-Off Forcing Functions.

6. Payload Thermal and Humidity Environment

The Contractor shall maintain the thermal and humidity conditions described by Table 2-8 within the payload fairing beginning with encapsulation and for the remainder of ground operations through launch. Fairing conditions, measured at the payload cone, shall be selected by the payload and documented in the ICD. The Contractor shall control temperature and relative humidity such that the dew point is never reached.

<table>
<thead>
<tr>
<th>Event</th>
<th>Temperature Range</th>
<th>Control</th>
<th>Relative Humidity (%)</th>
</tr>
</thead>
</table>

Notes:
1. Dew Point Temperature: 3-17°C (38-62°F);
2. Fairing temperature and humidity are selectable by the payload (bounded by the psychometric chart);
3. Fairing temperature shall be controllable to ±3.3°C (±5°F) of payload set point.

Table 2-8. Payload Thermal and Humidity Environments

The maximum fairing inside wall temperature shall be maintained at less than 121°C (250°F), with an emissivity of 0.92. This temperature limit shall envelope the maximum temperature of any component inside the payload fairing with a view factor to the payload with the exception of the Stage 3 motor.

The maximum upper stage motor surface temperature exposed to the payload shall not exceed 177°C (350°F), assuming no shielding between the aft end of the payload and the forward dome of the motor assembly. Fairing deployment shall be initiated such that the maximum (3-σ) free molecular heating rate (FMH) is less than 1135 W/m² (360 BTU/ft²/hr).

The Contractor shall locate the conditioned air outlets for both the 63" and 92" fairings at the forward end of the cylindrical section. The outlets shall direct the air along or at a slight angle to the fairing wall. Fairing inlet air shall not impinge on any payload surface within Volume A in Figure 2-7 (63" fairing) or Volume B2 in Figure 2-8 (92" fairing). For payloads using the full static envelopes provided by the fairings, the
Contractor shall coordinate with the payload to ensure that sensitive payload equipment is not placed in the immediate vicinity of the duct outlet. Mission-unique inlet duct diffusers and/or deflectors may be requested as non-standard services.

7. Payload Electromagnetic Environment

All power, control, and signal lines inside the payload fairing shall be shielded and terminated.

The Contractor shall provide a fairing with a minimum of 20 db attenuation between 416.5 and 5840 MHz.

2 - 9 CONTAMINATION CONTROL

A. Reference Data and Assumptions

1. Payload Encapsulation Facility

Per SOW 2.4.2.1, the NASA-provided payload processing area to be used for encapsulation will be in accordance with FED-STD-209E Class 100,000 clean room environment and that the clean room and anteroom(s) will utilize HEPA filter units to filter the air.

B. Performance

1. Contamination Control Plan

The Contractor’s generic payload contamination control plan shall be based on, and comply with, MIL-STD-1246C requirements. The Contractor shall develop and implement a mission-unique contamination control plan to accomplish any standard or non-standard contamination control services identified for each payload. Any mission-unique contamination control plan shall also comply with the requirements of MIL-STD-1246C.

2. Fairing Environment

Following payload encapsulation, the payload fairing environment shall be continuously purged with conditioned, HEPA-filtered, air. All HEPA filters used shall have a minimum 99.97% effective rating in removing particles of ≥0.3 microns in size. Carbon filters shall be provided to remove volatile hydrocarbons of molecular weight 70 or greater from the fairing air supply, with better than 95% efficiency.

The Contractor shall continuously monitor the hydrocarbon content of any fairing air supply via probes installed into the fairing inlet duct. Hydrocarbon content shall be continuously monitored and recorded from payload encapsulation until T-0 separation of the A/C duct.

3. Payload Encapsulated Environment Inner Surface Cleanliness

The Contractor shall clean, certify and maintain the cleanliness of the payload encapsulated environment inner surfaces to MIL-STD-1246C, Level 750A.
The Contractor shall perform the following to meet the requirement of MIL-STD 1246C Level 750A:

a. the fairing inner walls shall be cleaned to Visibly Clean-Highly Sensitive (VC-HS) level per Contamination Control Requirements, JSC-SN-C-0005;
b. the fairing acoustic blankets will be cleaned and verified for both particulate and NVR;
c. the exposed surfaces of the fairing inner graphite walls, not covered by acoustic blankets, will be lined with aluminum foil then cleaned and verified for both particulate and NVR; and
d. the payload cone assembly will be lined with aluminum foil, cleaned and verified for both particulate and NVR.

4. Clean Room Garments
The Contractor shall provide cleanroom garments and cleaning supplies. Clean room garments provided by the Contractor shall be cleaned by the Contractor.

5. Cooling Supply
The Contractor shall provide MIL-P-27401C Grade B GN₂ for spot cooling, from payload encapsulation through launch vehicle stack. The system shall be capable of regulating the flow between 0 and 25 SCFM. The Contractor shall supply and maintain the GN₂.

6. Materials
The vehicle assemblies which effect cleanliness within the payload encapsulated environment are as follows: the fairing assembly, the acoustic blankets, and the payload cone assembly. The fairing and payload cone assemblies shall be graphite reinforced epoxy composite structures which shall have a total mass loss (TML) value of approximately 0.54% by weight and collected volatile condensable material (CVCM) value of approximately 0.02%. Other materials used within the payload encapsulated environment shall either meet the NASA Reference Publication 1124 outgassing requirements of less than 1.0% TML and less than 0.1% CVCM or shall be expressly identified and submitted to NASA for approval.

7. MLI Thermal Insulation
The Contractor shall seal MLI thermal insulation blankets such that venting of blanket material debris does not occur in a direction towards the payload.
2 - 10 COLLISION/CONTAMINATION AVOIDANCE MANEUVER (CCAM)

A. Reference Data

1. Reserved

B. Performance

1. Following payload separation, the launch vehicle shall perform a collision/contamination avoidance maneuver (CCAM) to minimize payload contamination and any chance of re-contact with the separated payload.

2 - 11 PAYLOAD ENVIRONMENT INSTRUMENTATION

A. Reference Data

1. Shock data will not be obtained due to EMI associated with the firing of pyrotechnic separation hardware which corrupts the transmission of the data at the moment of separation.

B. Performance

1. Phases

The instrumentation to support mission success determination (Article H-8) shall be provided by the launch vehicle and shall encompass ground and flight mission phases. The payload environment instrumentation shall be transmitted, received and recorded in accordance with SOW 2.4.1.2.3.

2. Loads And Dynamics

The Contractor shall provide the payload environment instrumentation as defined in Table 2-11 to determine compliance to accepted levels. Accelerometers shall be ranged to obtain steady state, transient, and random vibration responses.

<table>
<thead>
<tr>
<th>Environment</th>
<th>Type</th>
<th>Number and Locations</th>
<th>Range, Sensitivity, and Bandwidth</th>
</tr>
</thead>
<tbody>
<tr>
<td>Acoustics</td>
<td>Microphone</td>
<td>2, Facing Wall on Intersub</td>
<td>160 dB, 2.0 V/ms, 8,000 Hz Sample Rate</td>
</tr>
<tr>
<td>Acceleration</td>
<td>Accelerometers</td>
<td>5, Payload Interface Flange</td>
<td>7.5 g, 66 mW/kg, 4,000 Hz Sample Rate</td>
</tr>
<tr>
<td>Accl Pressure</td>
<td>Pressure Transducer</td>
<td>1, Facing Wall</td>
<td>20.0 psi, 0.125 V/ms, 5 Hz Sample Rate</td>
</tr>
<tr>
<td>Temperature</td>
<td>Thermocouples</td>
<td>5, Facing Wall</td>
<td>-70°C to +195°C, 10 mV/°C, 5 Hz Sample Rate</td>
</tr>
</tbody>
</table>

Table 2-11. Contractor Provided Payload Environment Instrumentation

3. Temperature

The Contractor shall provide the temperature sensors defined in Table 2-11.
ATTACHMENT D-T
NON-STANDARD SERVICES

TAURUS LAUNCH VEHICLE

The following non-standard services shall be performed only as requested and authorized by NASA in accordance with Article G-3 Task Ordering Procedures and SOW Section 3.0.

1.0 MISSION–UNIQUE HARDWARE MODIFICATIONS

The Contractor shall design, manufacture, test, and implement the following, if authorized:

1.1. RESERVED

1.2. FAIRING MODIFICATIONS:

1.2.1. Additional Access Doors (Authorized by: L-15 Months)

The Contractor shall provide additional access doors of standard size 30.5 cm x 30.5 cm (12" x 12") for the 63" fairing and 45.7 cm x 61.0 cm (18" x 24") for the 92" fairing. The Contractor shall verify, through analysis, the structural integrity of the fairing with the additional door in the desired location. Provided this analysis shows the additional door is feasible, the Contractor shall manufacture the additional door and the modified fairing for the mission. This analysis shall then be validated in the acceptance test of the flight fairing structure.

This additional door shall be located in the cylindrical section of the fairing. Doors in the ogive (63" fairing) and bi-conic (92" fairing) sections are feasible but shall incur additional costs due to the more complex geometry.

1.2.2. Access Doors of Non-Standard Sizes (Authorized by: L-15 Months)

The Contractor shall provide access doors of non-standard size. As with the additional doors, the Contractor shall perform an analysis to verify the structural integrity of the fairing with the non-standard door size. Provided this analysis indicates the non-standard size door is feasible, the Contractor shall manufacture the non-standard size door and the modified fairing. This analysis shall then be validated in the acceptance test of the flight fairing structure.

The non-standard size door shall be located in the cylindrical section of the fairing. Doors in the ogive (63" fairing) and bi-conic (92" fairing) sections are feasible but shall incur additional costs due to the more complex geometry.

1.2.3. RF Mounting Provisions (Authorized by: L-20 Months)

The Contractor shall accommodate payload internal or external antennas or reradiators. This effort shall include payload-specific electrical modeling and analysis, antenna pattern testing, electrical component qualification testing, and mechanical and thermal design and analysis.
1.2.4. Provide S, C, X, and Ku-Band Reradiation Equipment

For this non-standard service, the Contractor shall provide the necessary hardware (e.g., transmitters/transponders, power dividers, antennas, and RF cables required for RF reradiation) to support the efforts as described in Non-Standard Service Section 1.2.3. Additional hardware shall be provided by the Contractor to support qualification testing. The re-radiation service transmitters shall provide between 5 to 10 Watt EIRP.

The Contractor provided hardware shall be capable supporting the following frequency bands:

1.2.4.1 Ku-Band (Authorized by: L-20 Months)
1.2.4.2 S, C, or X Band (Authorized by: L-20 Months)

1.2.5. RF-Transparent Doors (Authorized by: L-15 Months)

The Contractor shall provide a RF transparent door in the same size as the standard payload access door as described in SOW Section 2.1.B for the fairing in question. The standard cover for the access door shall be replaced with a fiberglass cover to provide RF transparency. Structural analysis shall be performed to verify the structural integrity of the fairing with the RF door in the desired location. Provided this analysis indicates a RF door is acceptable, the Contractor shall manufacture the RF transparent door and the modified fairing for the mission. This analysis shall then be validated in the acceptance test of the flight fairing structure.

The RF door shall be located in the cylindrical section of the fairing. Doors in the ogive (63" fairing) and bi-conic (92" fairing) sections are feasible but shall incur additional costs due to the more complex geometry.

In addition to mechanically providing an RF transparent door, the Contractor shall perform antenna pattern testing including field radiation and distribution testing to verify RF compatibility. EMC analysis shall be completed for the launch vehicle avionics and testing of all impacted avionics components shall be performed.

1.3. ALTERNATE PAYLOAD ATTACH FITTINGS (PAF)

1.3.1. Different Size or Different Payload Interface PAFs

1.3.1.1. Non-Separating Payload Interface (Authorized by: LSTO)

The non-separating interface shall be a 31.81" diameter bolt circle with 60, equally spaced, 0.257" diameter holes. The electrical interface requirements in Taurus Exhibit 2, Section 2-7, shall be applicable to this PAF with the exception that the electrical interface shall be via two, non-separating, MIL-C-38999 or equivalent (56-pin) connectors. The following services shall be included to support the non-separating interface:
Interface Drill Tool — The Contractor shall provide NASA or the NASA payload provider a drill tool for precision drilling of the payload/launch vehicle interface. Since other customers may require this tool, the Contractor shall coordinate usage of the drill tool with the other missions. Within the United States, delivery of the tool shall be coordinated and shipped by the Contractor. Delivery of the tool outside the United States will be coordinated and shipped by NASA.

Interface Fit Check — The Contractor shall support a fit check of the non-separating interface (payload cone structure) with the payload. The Contractor shall supply all the hardware, procedures, and personnel associated with the use of payload cone. NASA will provide a facility and the ground support equipment, procedures, and personnel associated with handling the payload.

1.3.1.2. High Capacity Payload Separation System Hardware (Authorized by: LSTO)

The Contractor can provide a high capacity payload separation system with a 38.81" (known hereafter as 38") diameter payload bolted interface. This system shall provide a significantly higher payload capability at the cost of a somewhat lower tip-off performance and a higher interface shock environment. The separation system shall be a marmon clamp band (or equivalent) design.

The electrical interface requirements in Taurus Exhibit 2, Section 2-7, shall be applicable to this PAF.

There shall be no performance impact associated with this non-standard service.

The features of the separation system are given in Figure 1.3.1.2-1 and Figure 1.3.1.2-2.

<table>
<thead>
<tr>
<th>Feature</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bolted Payload Interface</td>
<td>Standard Interface is a 38.81&quot; Diameter Bolt Circle with 60, Equally Spaced, 0.266&quot; Diameter Holes</td>
</tr>
<tr>
<td>Electrical Interface</td>
<td>Each (2) Separation Connector Will Provide 48 20AWG Contacts and 8 18AWG Contacts</td>
</tr>
<tr>
<td>Structural Capability</td>
<td>Capable of Handling the Payload Mass/C.G. Envelope Shown in Figure 5-2</td>
</tr>
<tr>
<td>Separation Energy</td>
<td>Will Deliver Up to 37.6 Joules (333 in-lb) of Energy to Separate the Payload and Launch Vehicle</td>
</tr>
<tr>
<td>Separation Spring Tailoring</td>
<td>In Order to Compensate for Payload Lateral C.G. Offsets, the Separation Spring Actuators Energies May Be Individually Tailored to Meet Stringent Tip-Off Requirements</td>
</tr>
<tr>
<td>for C.G. Compensation</td>
<td></td>
</tr>
<tr>
<td>Interface Ring Surface Treatment</td>
<td>Payload Side Interface Ring Has an Alodine 1200 Surface Finish. Other Surface Treatments on Non Critical Surfaces Are Available as Non-Standard Service</td>
</tr>
</tbody>
</table>

Figure 1.3.1.2-1. High Capacity Payload Separation System Design Features.
Figure 1.3.1.2-2. *High Capacity Payload Separation System Structural Capability.*

**High Capacity Payload Separation System Services** — The following services are included to support the high capacity separation system:

**Interface Drill Tool** — The Contractor shall provide NASA or the NASA payload provider a drill tool for precision drilling of the payload/launch vehicle interface. Since other customers may require this tool, the Contractor shall coordinate usage of the drill tool with the other missions. Delivery of the tool shall be coordinated by the Contractor. Within the United States, delivery of the tool shall be coordinated and shipped by the Contractor. Delivery of the tool outside the United States will be coordinated and shipped by NASA.

**Payload Separation System Fit Check** — The Contractor shall support a fit check of the upper ring of the flight separation system with the payload. The upper ring separation connector bracketry shall be installed so that the separation connector harnesses can be fit checked. The Contractor shall supply all the hardware, procedures, and personnel associated with the fit check. NASA will provide a facility and the ground support equipment, procedures, and personnel associated with handling the payload.

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1.3.1.3. 23" Payload Separation System Hardware (Authorized by: LSTO)

The Contractor shall provide a payload separation system with a 23.25" (known hereafter as 23") diameter payload bolted interface.

The separation system shall be a marmon clamp band (or equivalent) design. Separation velocity is provided by a set of four matched separation springs. An adapter cone shall be provided with the 23" separation system to interface with the standard 38" non-separating interface.

The electrical interface requirements in Taurus Exhibit 2, Section 2-7, shall be applicable to this PAF.

There shall be no performance impact associated with this non-standard service.

Figure 1.3.1.3-1 and Figure 1.3.1.3-2 detail the separation system design features.
<table>
<thead>
<tr>
<th>Feature</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bolted Payload Interface</td>
<td>Standard interface is a 23.25&quot; Diameter Bolt Circle with 32, Equally Spaced, 0.257&quot; Diameter Holes</td>
</tr>
<tr>
<td>Electrical Interface</td>
<td>Telemetry Connector: 42 22 AWG Contacts</td>
</tr>
<tr>
<td></td>
<td>Pyro Command Connector: 18 20 AWG Contacts</td>
</tr>
<tr>
<td>Structural Capability</td>
<td>Capable of Handling the Payload Mass/C.G.</td>
</tr>
<tr>
<td></td>
<td>Envelope Shown in Figure 5-4</td>
</tr>
<tr>
<td>Separation Energy</td>
<td>Will Deliver Up to 22.2 Joules (197 lb-ft) of Energy to Separate the Payload and Launch Vehicle</td>
</tr>
<tr>
<td>Interface</td>
<td>The Payload Side Interface Ring Has a Chemical</td>
</tr>
<tr>
<td>Ring Surface Treatment</td>
<td>Conversion Coat (per MIL-C-5541, Class III) Surface</td>
</tr>
<tr>
<td></td>
<td>Finish. Other Surface Treatments on Non-Critical Surfaces Are Available as a Non-Standard Service</td>
</tr>
</tbody>
</table>

**Figure 1.3.1.3-1. 23" Payload Separation System Design Features**

![Graph showing c.g. (m) vs S/C Mass (kg)]

Assumptions: Axial Accel = 2 g's (Tension), Lateral Accel = 3.5 g's, Load Peaking Factor = 1.0 (Where Load Peaking Is Defined as: Ratio of Maximum Predicted Line Load to Maximum Line Load Which Would Occur for an ideal Uniform Structure.)

**Figure 1.3.1.3-2. 23" Payload Separation System Structural Capability.**

**23" Payload Separation System Services** — The following services shall be included in the support of a 23" separation system and are described in detail in Non-Standard Service Section 1.3.1.2: interface drill tool, and fit check.

**"Intentionally Deleted"**
1.3.1.4 3-point PAF (Authorized by: LSTO)

The Contractor shall provide a non-explosive actuated separating nut/bolt 3-point attach MicroStar payload interface. This system is designed for use with the MicroStar bus satellites only and the mating parts of the separation system are integral with the satellites. The launch vehicle side of the 3-point attach hardware is comprised of a composite thrust tube with three aluminum interface brackets attached at the forward end of the tube. The interface brackets on the launch vehicle side have integral titanium cups and bolt catchers. The aft end of the thrust tube attaches to the non-separating interface of the launch vehicle. The performance impact shall be equal to the weight of the thrust tube minus the weight of the deleted standard separation system. Payload static envelope impacts shall be equal to the length of the thrust tube minus the height of the standard separation system, 13 cm (5.1"). The length, and thus the weight, of the thrust tube is dependent on the requirements of the payload/mission and shall be assessed on a mission unique basis. The impact of the 3-point attach to the payload environments shall be assessed on a mission unique basis

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The electrical interface requirements in Taurus Exhibit 2, Section 2-7, shall be applicable to the 3-point PAF, with the following exceptions:

The standard 42-pin connector shall be replaced with a 22 pin G&H 1179 \textit{"zero force"} connector.

The dual-serial electrical interface (4 pins) required for sending primary and redundant firing codes to the payload shall be included in the standard electrical interface. The payload shall have 18 pins to select functions from the standard electrical service.

1.4. OPTIONAL UPPER STAGE HARDWARE

1.4.1. Upper Stage Motor (Authorized by: LSTO)

The Contractor shall provide a spinning Thiokol STAR 37FM motor as a replacement for the standard Alliant built Orion 38.

This non-standard service shall be provided with all the engineering, documentation and equipment necessary to adapt the Thiokol motor to the Taurus vehicle, which includes spin testing and integration of the payload. The RCS system shall spin the satellite/motor up to a predetermined rate as specified in the ICD, separate the stage from the vehicle, and initiate the motor firing command from an onboard sequencer, which controls all ordnance functions. At the end of motor burn the system shall be despun to the specified ICD value, the payload is separated from the motor at the
appropriate time, and a CCAM is performed to mitigate recontact and thruster exhaust gas impingement.

As a minimum, the Contractor shall provide the following hardware:

- STAR 37FM Motor
- Electrical ordnance sequencer
- All ordnance including transfers, separation, ignition, PSS and FTS
- GSE, except for payload specific GSE
- Motor-to-launch vehicle and payload-to-motor separation systems
- Payload adapter
- Electrical harnessing and connectors
- S-band telemetry system
- RCS spin system
- Yo-Yo weight despins system and Yo-weight CCAM

The Contractor shall add all the necessary hardware to the motor to enable it to meet mission requirements. This hardware includes as a minimum: a telemetry system, nutation control, and a yo-yo weight despins system, as discussed above. The Contractor shall also provide an adapter between the STAR 37FM motor and the Taurus avionics structure. This adapter shall include a cylindrical support structure and flight termination system components, as well as a clampband and 38" separation system. This adapter shall be jettisoned with the burnt-out STAR 37FM motor.

1.4.2. Inertial Navigation System (INS) Upgrade (Authorized by: LSTO)

The Contractor shall have the capability to replace the existing INS (Litton LR-81) with an improved INS (Litton LN-100LG) that shall improve the insertion state accuracy.
"Intentionally Deleted"

1.5. RESERVED

1.6. RESERVED

1.7. RESERVED

1.8. PAYLOAD CONNECTOR COVERS (Authorized by: L-15 Months)

The Contractor shall provide a payload separation connector cover. The cover shall form a metal to metal seal onto the face of the open connector housing using a spring loaded mechanism. The seal shall prevent free-flying debris greater than 0.635 mm (0.025") in diameter from contaminating the exposed contacts of the payload half of the vehicle-to-payload electrical connector. This non-standard service shall include connector covers for any of the payload non-standard service payload electrical interfaces.

1.9. ENHANCED ELECTRICAL INTERFACE

A. Increased Capacity Payload GSE Interface (Authorized by: L-18 Months)

In addition to the standard payload to payload GSE interface, the Contractor shall provide up to four MIL-STD-1773 fiber optic lines or two MIL-STD-1553 coaxial lines, and an additional 10 twisted shielded wire pairs for monitoring and commanding the payload via payload GSE are desired. The essential electrical parameters for these functions are specified in SOW Exhibit 2, Table 2.5. This service includes modifying the existing cable design for four cables to add the additional fiber/coax/TSP wires as well as the modifications required to two junction boxes to pass these signals to the ground.
B. Launch Vehicle Command & Control of Payload (Authorized by: L-14 Months)

The launch vehicle shall provide the means for step or pulse commanding of payload separation system ordnance, payload relays, and payload opto-isolated discretes from ten minute before launch until payload separation. Up to eight twisted shielded pairs shall be designated for these commands, which may be utilized in any combination thereof. Figure 1.9-1 defines the essential electrical parameters for these functions.

The Contractor shall provide up to five discrete commands, one RS422/RS485 communication link, and four redundant pyro initiation signals. The non-standard service includes provision of additional commands required beyond the standard service within the availability limits of the launch vehicle. This service includes coordination and testing of additional signals beyond the normal vehicle scope.

<table>
<thead>
<tr>
<th>Function</th>
<th>Transient Characteristic</th>
<th>Voltage (0-pk, VDC)</th>
<th>Current (0-pk, Amps)</th>
<th>Impedance (Ohms)</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ordnance</td>
<td>75ms pulse width max</td>
<td>NA</td>
<td>5.0</td>
<td>Payload Input: 1.0</td>
<td>&gt;90%</td>
</tr>
<tr>
<td>Relay Discretes</td>
<td>40ms min to infinite pulse width</td>
<td>40 VDC max at payload input</td>
<td>0.5 max</td>
<td>Vehicle Output Open: &gt;1 Meg Closed: &lt;1Meg</td>
<td>&gt;90%</td>
</tr>
<tr>
<td>Opto-Isolated Discretes</td>
<td>&quot;On&quot; for 3.0 seconds min</td>
<td>NA</td>
<td>Payload provided 20mA source max</td>
<td>NA</td>
<td>&gt;90%</td>
</tr>
</tbody>
</table>

Figure 1.9-1. Requirements For Launch Vehicle Command & Control Of Payload

C. Telemetry (Authorized by: L-14 Months)

The launch vehicle shall provide for monitoring analog and discrete payload outputs during the period of pre-launch until payload separation. A 5 Hz minimum sampling rate is required. Up to eight twisted shielded pairs shall be provided. Figure 1.9-2 defines the key electrical parameters for these functions.

The Taurus standard service includes monitoring and downlinking up to 250 bytes per second worth of telemetry data via the RS422/RS485 serial link. In addition, the status of up to five discrete monitor points, five separation indicators, and six analog telemetry points can be downlinked with the launch vehicle MUX data. This non-standard service includes monitoring and downlinking additional telemetry points provided space is available in the telemetry stream and in the MUX. Data may be downlinked at 5/25 Hz via the MUX.
<table>
<thead>
<tr>
<th>Function</th>
<th>Measurement Resolution</th>
<th>Voltage (0-pk, VDC)</th>
<th>Current (0-pk, Amps)</th>
<th>Payload Input Impedance (Ohms)</th>
<th>Shielding Coverage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Analog</td>
<td>20 mV</td>
<td>-15 to +40</td>
<td>N/A</td>
<td>100 K</td>
<td>&gt;90%</td>
</tr>
<tr>
<td>Discrete</td>
<td>N/A</td>
<td>+5</td>
<td>10 mA</td>
<td>100 K (Switch Open)</td>
<td>&gt;90%</td>
</tr>
</tbody>
</table>

* Payload will provide an open collector output that requires being tied to the 5 VDC launch vehicle bus via a pull-up resistor.

**Figure 1.9-2. Payload Telemetry Monitoring Requirements**

**D. Launch Vehicle Instrumentation** (Authorized by: L-20 Months)

Additional launch vehicle environmental instrumentation shall be accommodated using the excess capacity available in the Wide Band Instrumentation System (WBIS) for the purposes of verifying vehicle performance or environments. The additional capacity of the system and the associated transducers and harnessing shall be provided consistent with a defined suite of measurements provided by the customer working within the bandwidth, channel count, encoder card count, and existing standard vehicle staging and payload interface connector pin count limitations. The additional capacity shall be 1.0 Mbyte/sec, and all channels are limited to 8 bit resolution. Forty channels shall be available, although these channels must be grouped in sets of four for DC coupled signals and sets of eight for AC coupled signals. The excess capacity of the WBIS shall be used to support the following types of instrumentation along with their associated objectives:

1) Low frequency accelerometers with a minimum frequency range from DC to 50 Hz distributed sufficiently on the vehicle to quantify and verify vehicle bending modes, structural loads and steady state acceleration.

2) High frequency accelerometers with a frequency range from 20 to 2000 Hz, minimum, distributed along the vehicle to ascertain random vibration and shock environments for all flight critical avionics and structures.

3) Strain gauges installed on all motor cases, engine nozzle mounts, gimbal blocks, interstage structures, payload fairing, and other critical structures for the purpose of verifying loads. Installation on engine nozzle mounts is not included within the scope of this non-standard service. All stages are solid rocket motors employing submerged flexseal bearings to support the nozzles for which strain measurements would be problematic to implement and of little inherent value given the flight history of these assemblies.

4) Temperature sensors installed on all motor cases, engine nozzle mounts, gimbal blocks, interstage structures, payload fairing, critical structures, and all flight critical avionics components for the purpose of verifying vehicle thermal environments.
5) Pressure transducers installed to monitor all pressure vessels, interstage and payload compartments pressures, some exterior static pressures, and some exterior dynamic pressures. Critical pressure vessel measurements are already provided through the standard telemetry system. No additional pressure sensors will be installed in existing pressure systems within the scope of this non-standard service.

6) Calorimeters or similar instrumentation installed to assess base heating environments in critical regions.

Transducers shall be installed in locations such that they do not present a ground operations hazard, interfere with the operation of existing vehicle hardware or GSE, or require the re-location of vehicle hardware. Transducers shall not be attached to the motor cases, although attachment to the domes shall be permissible. Aerodynamic fairings for externally mounted devices and associated thermal protection shall not be provided as part of this non-standard service.

Given the system limitations defined above, the WBIS system shall be configured for one flight to meet all or most of the objectives defined above as directed by NASA. The Contractor shall conduct system verification tests including: continuity checks, tap tests, piston phone checks and broadband vibration and acoustic tests of transducers at the highest level of assembly at which this is practical. In addition, the Contractor shall provide post-flight data processing and evaluation and provide comparisons with respect to previous flight data, and maximum predicted environments. The Contractor shall submit the data recorded during flight as a supplemental flight report, which shall be an attachment to CDRL C23.
2.0 MISSION–UNIQUE SUPPORT

2.1. RESERVED

2.2. SPECIAL CONTAMINATION CONTROL OPTIONS:

The following contamination control requirements shall be met.

A. **Fairing Environment** (Authorized by: L-16 Months)

The Contractor shall purge the fairing environment, from payload encapsulation through vehicle stack, with either FED-STD-209E Class 10,000 or Class 1,000 filtered air. The air supply system shall also incorporate carbon pre-filters. The Contractor shall accomplish continuous monitoring and verification of the supply air for particulate via probes installed into the fairing inlet duct.

B. **Fairing Internal Surface Cleaning** (Authorized by: L-12 Months)

The Contractor shall clean, certify, and maintain internal surfaces of the payload fairing to level 500A or 600A per MIL-STD-1246C. This includes all efforts to support increased levels of precision cleaning of the internal fairing surfaces prior to payload encapsulation; additional surface cleanliness measurements to verify surface cleanliness; and additional handling controls to maintain cleanliness.

C. **Payload/vehicle Integration Environment: Class 10,000** (Authorized by: L-16 Months)

The Contractor shall provide an Environmental Control System (ECS) that is cleaned, certified and maintained at Class 10,000 (or better) per FED-STD-209E to support the payload mate through fairing closeout operations. This system consists of the Environmental Control Unit (ECU), insulated flexible ducting to transport air to the payload volume, and a HEPA filter section for contamination control. The ECS shall be fully mobile and easily transportable. The ECS shall provide the temperature and humidity environment as specified in Exhibit 2, Section 2-8, B.6.

D. **Reserved**

E. **Instrument Purge System** (Authorized by: L-18 Months)

The Contractor shall provide a single instrument purge system using MIL-STD 27401C Grade B GN2. This instrument purge system shall use a 0.5 micron filter (or better), and shall be capable of providing from 0 to 25 SCFM. This instrument purge system shall be provided from payload encapsulation until launch. The purge system shall include a quick disconnect fitting that is disconnected at fairing separation. The Contractor shall certify the purge system cleanliness to MIL-STD-1246C Level 100A prior to use. The Contractor shall certify and maintain the instrument purge system hydrocarbon levels at or below 5.0 ppm. The Contractor shall monitor, maintain, and
replace purge gas supplies as they are used, while maintaining all cleanliness requirements

2.3. ENVIRONMENTAL ASSESSMENT DATA BOOK (Authorized by: L-18 Months)

For those missions in which the payload contains a radioisotope component, such as a Radioisotope Heating Unit (RHU) or Radioisotope Thermoelectric Generator (RTG), an Environmental Assessment Data Book shall be developed by the Contractor to support payload reviews. The data book shall contain a detailed description of the launch vehicle, the launch site infrastructure, the trajectory profile and instantaneous impact point history, and descriptions of possible accident scenarios, their environments, and probabilities of occurrence.

2.4. PAYLOAD FIT CHECK SUPPORT (Authorized by: L-20 Months)

The Contractor shall provide a facility to perform a fit check between the payload and the flight fairing. NASA will transport the payload or a full-scale model of the payload to this facility. All personnel, procedures, and any specialized handle equipment (e.g., lifting sling, turn over cradles) required to handle the payload will be provided by NASA. The Contractor shall provide for NASA’s use standard handling equipment (e.g., forklifts, cranes) to unpack, move, or position the payload. In addition, all required GSE to perform the fit check shall be provided by the Contractor. All personnel required for the operations concerned with launch vehicle hardware handling shall be provided by the Contractor. The Contractor shall use the integrated procedures for payload/fairing operations that are developed for use at the launch site. Unless otherwise directed, the payload’s flight PAF shall be used for performing the fit check.

Up to ten critical clearance locations, as identified by the NASA, shall be measured. The accuracy of the measurements shall be dependent on the position of the critical clearance locations.
3.0 OTHER NON-STANDARD SERVICES

3.1. PAYLOAD CONTINGENCY SERVICES

In the event of a payload contingency that requires the payload to be deintegrated from the launch vehicle flight hardware, the Contractor shall performing the necessary activities and services described below. This non-standard service is valid for standard launch site locations.

3.1.1 After Payload Encapsulation in PPF (Authorized as Required)

The Contractor has transported the Encapsulated Cargo Element (ECE) to the launch site, placed in the horizontal position, and moved inside the integration tent. The ECE has not been mechanically mated to the launch vehicle. However, the Contractor has mated the payload electrically to the launch vehicle but Flight Simulation 3 has not been conducted. The Contractor shall provide the support necessary to complete the following:

- Breakover ECE from horizontal to vertical, attach ECE Base GSE, and place on transporter.
- Transport ECE from launch site to PPF. Remove ECE from transporter and place on PPF floor.
- Return MGSE to PPF and demate fairing halves.
- Demate all electrical connections and remove payload from payload cone (integrated Contractor/NASA operation).
- Payload rework, as required (up to 7 days of NASA/payload operations).
- Install the payload on the payload cone and mate all electrical connections.
- Mate fairing halves and encapsulate payload.
- Load and transport ECE from PPF to launch site.
- Breakover ECE from vertical to horizontal and place on horizontal cart inside tent. Mate the payload electrically to the launch vehicle.

3.1.2 After Payload/Launch Vehicle Mate (Authorized as Required)

The Contractor has erected the payload/launch vehicle and final arming operations may have been completed. This non-standard service is based upon NASA or the payload initiating a hold and canceling the launch before Taurus Launch Vehicle Stage 0 TVC pressurization occurred. Thus, the Contractor shall provide the support necessary to complete the following:

- Reinstall weather protection on vehicle (requires bucket trucks).
- Move GSE back to launch site (AIT, Fairing MGSE, EGSE, etc).
- Construct scaffolding.
- Interstage: Perform mechanical, ordnance, and electrical disconnects.
- Demate upper and lower umbilical tower sections.
- Set up cranes and install aft ring (Upper Stages Lifting Sling (USLS)).
- Install forward USLS yoke.
- Demate, lift, lower, and breakover upper stack to horizontal.
- Remove forward USLS yoke and connecting rods.
- Remove upper umbilical tower.
- Reposition Integration Tent.
- Reverse ordnance closeouts.
- Remove TPS from various areas.
- Reinstall fairing handling GSE.
- Demate ECE from Avionics Skirt.
- Demate ECE from Stage 3.
- Breakover ECE from horizontal to vertical, attach ECE Base GSE, and place on transporter.
- Transport ECE from launch site to PPF. Remove ECE from transporter and place on PPF floor.
- Return MGSE to PPF and demate fairing halves.
- Demate all electrical connections and remove payload from payload cone (integrated Contractor/NASA operation).
- Payload rework, as required (up to 7 days of NASA/payload operations).
- Install payload on the payload cone and mate all electrical connections.
- Install fairing halves and encapsulate payload.
- Load and transport ECE from PPF to launch site.
- Breakover ECE from vertical to horizontal and place on horizontal cart inside tent.
- Reperform all nominal integration and testing operations from payload arrival at the site through launch.
- Reperform launch vehicle final arming operations.

3.2. STORE AND FORWARD TELEMETRY (Authorized by: L-15 Months)

The Contractor shall provide an auxiliary means of retrieving critical vehicle telemetry for West Coast launches after the vehicle travels over the horizon from VAFB. When utilized, the Store & Forward function saves critical flight data (e.g., IMU and guidance parameters, important temperatures and pressures, separation breakwire status) at a selectable 5 or 25 Hz sample rate (limited by available memory space). This data is then "replayed" through the RF system once the vehicle is in sight of a downrange tracking station (e.g., McMurdo Ground Station).
3.3. STAGE 2 ONBOARD CAMERA (Authorized by: L-15 Months)

The Contractor shall provide a real-time second stage video system. This system is completely self contained and has a dedicated battery, RF system for transmission of signal, and two cameras for forward and aft views of the rocket. The cameras shall switch views as commanded by the flight computer to catch critical staging events and fairing separation. The Contractor shall evaluate Sun angles and time of day lighting levels to ensure acceptable video quality from the start of countdown until second stage separation. The impact on the launch vehicle performance to orbit when using this non-standard service is approximately 12 lbs.

3.4. 14 TO 29 DAYS SCIENTIFIC LAUNCH PERIOD WITH ANNUAL REOPENING (Authorized by: LSTO)

The Contractor shall provide all launch services described in the SOW to accommodate a scientific launch period as small as 14 days but not greater than 29 days in duration. The Contractor’s price shall remain valid for missions in which the payload scientific launch period reopens no more than one-year after the closing of the original scientific launch period.

3.5. PAYLOAD ISOLATION SYSTEM (Authorized by: L-20 Months)

The Contractor shall provide a payload isolation system to facilitate reductions in payload transient and low frequency loads. The passive isolation system incorporates a flight-proven design and integration procedure. The entire system is specifically designed to attenuate critical payload responses at frequencies identified through Coupled Loads Analysis.

Additional considerations which shall be evaluated by the Contractor include as a minimum: reduction in payload/fairing clearance, a small forward offset of the payload (e.g., 25 mm (1.0")), and a slight reduction in performance associated with the additional mass (approximately 11.4 kg (25 lb) per set) of the system. Finally, the Contractor shall test all the flight isolation system components to the maximum predicted thermal environments, static and dynamic loads.

3.6. TEST PAYLOAD ATTACH FITTING (TPAF) FABRICATION (Authorized by: L-18
Months Before Required Use)

The Contractor shall design and fabricate a Taurus TPAF. The TPAF shall have the appropriate electrical and mechanical interfaces to simulate a flight PAF. The diameter of the TPAF will be specified by NASA upon ordering this non-standard service.
4.0 SECONDARY/CO-MANIFESTED PAYLOADS SERVICES, COMPATIBILITY ASSESSMENTS, AND HARDWARE

This section summarizes the services, support, and hardware required to launch a NASA Secondary/Co-Manifested Payloads. Additional studies or payload compatibility assessments can be ordered by NASA as outlined in Articles H-14, H-15, or H-16.

4.1. SECONDARY/CO-MANIFESTED PAYLOADS SERVICES

The Contractor shall allow for sufficient clearance between the secondary payload envelope and the launch vehicles so that no fit checks of the secondary payload are required.

**Weight and CG constraints**

The performance available for a secondary payload shall include assessments for all mission unique flight hardware necessary for that configuration. This shall include the mass of DPAF hardware, additional separation systems, and harnessing.

**Coupled Frequency**

The Contractor shall analyze the coupled frequencies as part of the Payload Accommodation Study (Non-Standard Service Section 4.2.3). Table 4.1-1 outlines the following payload characteristics that are required for secondary payload.

<table>
<thead>
<tr>
<th>Secondary Payload Characteristic</th>
<th>Non-Separating</th>
<th>Separating</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>DPAF Load Bearing</td>
<td>DPAF Load Bearing</td>
</tr>
</tbody>
</table>
| Coupled Frequency               | Total Payload Stack
Frequency >25 Hz | Total Payload Stack
Frequency >25 Hz |
|                                 | Frequency >25 Hz | Frequency >25 Hz |

**Table 4.1-1. Secondary Payload Couple Frequency**

**Payload Volume**

The secondary payload volume for the DPAF configuration shall be defined in Figures 4.3.1-1, 4.3.2-1, or 4.3.3-1. The height of the DPAF shall be adjusted to tailor the height of the secondary payload as long as the envelope and environmental requirements for the primary payload are not violated. The secondary payload lateral dimensions are defined by the secondary payload separation tip-off motion. The Contractor shall ensure adequate separation clearance margin between the payload and the DPAF structure. Actual dimensions, clearance margins, and payload attachment fittings are unique for each mission and shall be designed by the Contractor. Physical access to the secondary payload shall be available up through DPAF installation.

**Electrical Interface**

The secondary payload shall have access to the remaining electrical services described in Taurus Exhibit 2 Section 2-7.
The Contractor shall provide access for battery trickle charging through an existing payload connector following payload to payload cone mate. NASA will be responsible for providing charging equipment and cabling. The Contractor shall provide the capability for a secondary payload to remain on battery trickle charge via the umbilicals through the final launch countdown. Access to signals on the umbilicals shall be obtained through the final countdown.

The Contractor shall provide pyros, discretes, or commands, to initiate the secondary payload separation. The Contractor shall provide breakwire separation indications from both primary and secondary payloads to verify separation.

**Figure 4.3.1-1: Taurus 38” DPAF**

**Figure 4.3.2-1: Taurus 50” DPAF**
Secondary Payload Mission Services

The secondary payload services shall apply to either a Co-Manifested or NASA Secondary Payload. These services shall provide for all integration services required for delivering NASA Secondary Payloads to orbit. These services shall include as a minimum:

- additional payload integration activities (e.g., the preparation of, or modifications to, the payload ICD)
- coordination of additional payload range requirements
- CDRLs
- MIWG and ROWG meetings
- additional mission team support, program, project, and engineering management
- field site support required to execute the launch
4.1.1. Co-Manifested Payload Mission Service (Authorized by: LSTO)
This service applies to any NASA Secondary Payloads that are Co-manifested (Reference Article H-14) with a NASA Primary Payload. The Contractor shall provide these services for separating and non-separating secondary payloads as described in Non-Standard Service Sections 4.1. Secondary payload technical capability and compatibility assessment studies are described and ordered separately as a Non-Standard Service in Sections 4.2. The hardware necessary to attach the primary and secondary payloads to the launch vehicle are described and ordered separately as a Non-Standard Service in Sections 4.3.

4.1.2. NASA Secondary Payload Service (Authorized by: LSTO)
This service applies to any NASA Secondary Payloads that are carried in surplus space without interfering with a non-NASA Primary Payload (Reference Article H-16). The Contractor shall provide this service for separating and non-separating secondary payloads as described in Non-Standard Service Sections 4.1. Secondary payload technical capability and compatibility assessment studies are described and ordered separately as a Non-Standard Service in Sections 4.2. The hardware necessary to attach the secondary payloads to the launch vehicle are described and ordered separately as a Non-Standard Service in Sections 4.3.

4.2. SECONDARY PAYLOAD COMPATIBILITY ASSESSMENTS

4.2.1. Payload Compatibility Assessment (Authorized as Required)
The Contractor shall perform all necessary compatibility analyses to verify that the NASA Secondary Payload does not impact the primary payload unacceptably. The Contractor shall document and submit these analyses, which shall include as minimum:
- combined coupled load analysis
- combined thermal analysis
- critical clearance analysis
- interface failure modes and effect analysis
- safety assessment
- mass properties report
- orbital performance estimate
- GN&C analysis

NASA will be responsible for providing any existing documentation or information about the NASA Secondary Payload, which may be necessary to complete this assessment.

4.2.2. Secondary Payload Mission Feasibility Study (Authorized as Required)
The Contractor shall conduct a study that assesses the excess performance and payload volume available to accommodate a NASA Secondary Payload.
This performance assessment shall include an estimate of the performance decrement associated with any particular mission-unique hardware. The Contractor shall provide draft configuration drawings, which identifies payload volume available and any potential configurations.

4.2.3. Secondary Payload Accommodation Study (Authorized as Required)

The Contractor shall develop the design, implementation scheme, and general payload requirements for accommodating NASA Secondary Payloads. The general payload requirements shall include, as a minimum, the mechanical interfaces, available mass, volume, and expected environments.

The Contractor shall prepare and submit parametric performance studies and load analysis. The Parametric Performance Studies shall examine the excess performance available as a function of the potential payload mass and orbit requirements. The Loads Analysis shall examine the secondary payload dynamic load environments, for various viable combinations of primary and secondary payload characteristics (e.g., mass properties, fundamental bending mode frequencies, and axial center of gravity position using given primary payload configuration).

For NASA Primary Payloads, NASA will supply the primary payload model used for the load analysis. The Contractor shall use a range of simplified secondary payload models that shall be integrated with the launch vehicle system model, and acceleration responses shall be evaluated. The results of this evaluation shall be presented to NASA.

4.3. SECONDARY PAYLOAD HARDWARE

The Contractor shall provide a separation system to separate the secondary payloads. The separation event shall be sequenced, controlled, and verified by the launch vehicle. After the primary payload has separated, the 38" separation system shall separate the adapter cone from the cylinder. As identified in the ICD, the launch vehicle shall be orientated so that the separation system can be initiated and place the secondary payload in the desired orbit. After each separation event, a CCAM shall be performed to avoid recontact with primary payload, adapter cone, and launch vehicle. The Contractor shall design, fabricate, and install all the components for one of the DPAF systems described below.

4.3.1. 38” DPAF (Authorized by: LSTO)

Figure 4.3.1-1 depicts the secondary payload configuration utilizing the non-standard, 38” Dual Payload Attach Fitting (DPAF). The secondary payload shall be housed within the DPAF structure and is deployed out of this structure following primary payload separation. The DPAF shall be designed by the Contractor to fit within either the standard 63" or 92" payload fairing. The height of the DPAF shall be adjusted to tailor the height of the secondary payload as long as the envelope and environmental requirements for the primary payload are not violated. The envelope available for the secondary payload depends upon the primary payload characteristics and the separation dynamics of the secondary payload.
4.3.2. 50" DPAF (Authorized by: LSTO)

Figure 4.3.2-1 depicts the secondary payload configuration utilizing the non-standard 50" DPAF. The secondary payload shall be housed within the DPAF structure and can be deployed out of this structure following primary payload and DPAF forward cone separation. The 50" DPAF shall be designed by the Contractor to fit within either the standard 63" or 92" fairing. The height of the DPAF shall be adjusted to tailor the height of the secondary payload as long as the envelope and environmental requirements for the primary payload are not violated. The envelope available for the secondary payload depends upon the primary payload characteristics and the separation dynamics of the secondary payload.

4.3.3. 63" DPAF (Authorized by: LSTO)

A 63" DPAF is depicted in Figure 4.3.3-1. The 63" DPAF shall be designed by the Contractor to maximize the larger volume of the 92" fairing configuration. The secondary payload shall accommodated within with the DPAF structure and is deployed out of this structure following primary payload and forward payload cone separation. The height of the DPAF shall be adjusted to tailor the height of the secondary payload as long as the envelope and environmental requirements for the primary payload are not violated. The envelope available for the secondary payload depends upon the primary payload characteristics and the separation dynamics of the secondary payload separating payloads that use a DPAF.

4.3.4. Payloads Attach Fittings for Separating Secondaries - 38", 37", or 23" (Authorized by: LSTO)

The Contractor shall provide a 37", 38", or 23" separation systems for separating secondary payloads. The 37" or 38" systems are identical to those offered as a standard service for primary payloads as described in SOW Taurus Exhibit 2 Section 1.6. This service shall include the necessary hardware and shall be exercise in conjunction with the appropriate DPAF Non-Standard Service Section.

The 23" separation system shall be identical to the system offered as described in Non-Standard Service Section 1.3.1.3. This service shall include the necessary hardware and shall be exercise in conjunction with the appropriate DPAF Non-Standard Service Section.

4.3.5. Payload Attach Fittings for Non-Separating Secondaries (Authorized by: LSTO)

The Contractor shall offer the non-separating 38" payload interface described in Non-Standard Service Section 1.3.1.1. This service shall include necessary hardware and shall be exercised in conjunction with the 50" or 63" DPAFs detailed in Non-Standard Service Sections 4.3.2 or 4.3.3.
4.3.6. Secondary Payload Mass Simulator (Authorized by: L-20 Months)

The Contractor shall construct a metallic (minimal outgassing as specified in the ICD) mass simulator of the secondary payload. Mass properties shall match those of the secondary payload within the tolerance specified by the ICD. The mass simulator shall also adhere to the requirements (e.g., contamination, cleanliness, interface requirements) identified in the ICD. The mass simulator shall not exceed the maximum dimensions allotted to the secondary payload by the mechanical ICD. The mechanical interface connections shall meet the interface requirements of the secondary PAF. Any other analysis or documentation required for replacement of the secondary payload with the mass simulator shall be completed by the Contractor.