LOCKHEED MARTIN

Proposal Information Package

MARS ONE

Mars One 2018 Lander Payload

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REVISION HISTORY

Revision	Date	Change Description / Authority
New	May 22, 2014	Initial Release

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1 INTRODUCTION

Mars One will launch a lander to Mars based on a successful NASA heritage lander design in May of 2018. This Proposal Information Package document is being supplied with the Mars One 2018 Precursor Lander mission Request for Proposals for payloads. It is applicable only to the Mars 2018 Precursor Lander mission.

1.1 PURPOSE AND SCOPE

This document describes general lander performance, resources and constraints that payload proposers must meet to propose viable instruments. To be selected, proposed payloads must be able to be accommodated within the resources described herein with margin. Proposals will be reviewed with respect to compatibility with the capabilities described herein and may not be selected if the design and design maturity suggest the instrument cannot be developed within cost and schedule and resources.

2 GENERAL DESCRIPTION OF MISSION AND SPACECRAFT

2.1 MISSION

The launch period for the lander launch opportunity opens May 16, 2018. The earliest Mars arrival is March 17, 2019 when the spacecraft directly enters the Mars atmosphere from the arrival hyperbola. The latest arrival is April 15, 2019 with a similar approach methodology. The lander will land in the northern hemisphere of Mars, nominally near 45 degrees north latitude, at a site predetermined to have the highest probability of accessible ice for the water extraction demonstration. The season of arrival for the lander is middle northern spring with a heliocentric longitude Ls=356 assuming a launch on the first day of the launch period. If the launch occurs on the last day of the 20 day period, arrival will happen somewhat later with an Ls=11. Once on the surface, the lander is expected to survive for up to one Earth year before the approach of winter reduces available energy to the point where the lander can no longer maintain energy balance.

2.2 SPACECRAFT

The spacecraft is shown in Figure 2.2-1. The lander and the instruments are encapsulated in an aeroshell to protect them during hypersonic entry at Mars. Most instruments will be located on the top deck of the lander, along with several flight system components such as antennas, etc.

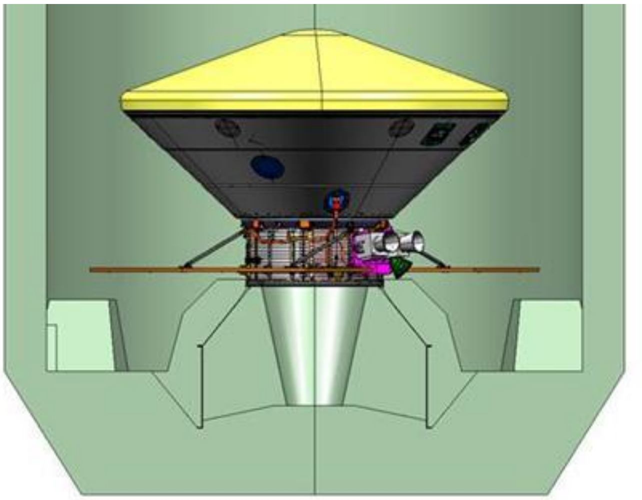


Figure 2.2-1. Spacecraft in launch configuration. Lander and payloads are inside aeroshell.

3 CONSTRAINTS IMPOSED BY MISSION AND SPACECRAFT DESIGN

3.1 PAYLOAD ACTIVITIES BY MISSION PHASE

3.1.1 Launch

The lander launch period opens on May 16, 2018 and lasts until June 5th 2018. As of this writing, it is expected that the lander will be launched on a US launch vehicle. The launch site will be limited to the sites that can support those vehicles, currently either the Western Test Range or the Eastern Test Range. Payload providers may be expected to support launch site processing operations. During launch, the payload, like the spacecraft itself, will be subjected to all the launch environments described in section 3.6.2. No payload activities will take place during launch.

3.1.2 Cruise

Cruise to Mars is approximately ten months. During that time, there are no payload activities, thought a post launch checkout and a pre landing checkout may be executed. Payloads must not require any cruise operations, and in general will be powered off. Cruise heater circuits may be considered if necessary but every attempt should be made to design instruments to the cruise environments without the need for replacement heater power.

3.1.3 Entry, Descent and Landing

The arrival period depends on the launch date, but the earliest possible landing is March 17, 2019. The latest arrival is April 15th 2019. No payload activities are allowed during EDL except for descent imaging. No other payloads will be powered on during EDL. During EDL, driving environments include repressurization to Mars ambient and parachute opening loads, as well as vibration loads during terminal descent. All these environments are specified in section 3.6.4.

3.1.4 Landed Mission

Immediately upon landing, the spacecraft will deconfigure from flight and configure for surface operations. The landed solar array deployment can take up to 40 minutes under worst case timelines. After deployment of the arrays, all payload pyro circuits will be fired and then limited imaging will occur to ascertain spacecraft status. Even before touchdown, the lander engines will displace significant amounts of regolith. Top deck payloads should expect to have some dust settle on them. Under deck imagers may be hit by small pieces of the surface displaced by the engines.

No instrument operations will be conducted the sol of landing (Sol 0) after initial limited imaging. Sol 1-7 comprise the characterization period. Flight system performance (power, thermal, telecom) will be characterized. Instruments are deployed as appropriate, if not already done on sol 0, and will also be characterized during this period. Instrument characterization will include imaging of the surface accessible by the sample acquisition system and depending on the sample acquisition system, the surface may be interacted with.

Once characterization is complete, water extraction begins. This process is a tactical surface operation, where each sol's activities will depend on the previous sols activities' success or failure. Imaging will be used to support sample acquisition as well as other activities based on longer term strategic planning. Other payload activities, if any, will be scheduled as appropriate and needed. All payload data is stored in memory until it is relayed to an orbiter for transmission back to Earth. These relay periods are irregular, and depend on the orbiter being used. Direct-to Earth X-band serves as a lower data return backup to data relay via orbiters.

3.2 LANDING SITE LIMITATIONS

3.2.1 Latitude Accessibility

The Mars One 2018 precursor lander is targeted at a nominal landing latitude of 45 degrees north, though latitudes as far south as 42 and as far north as 50 may be considered. The currently favored regions are southeast of Viking 2 in Utopia Planitia and east through Arcadia Planitia. Modeling suggests these regions have the greatest probability of near surface ice, which seems to be confirmed by images of recent craters exposing what appears to be ice. Several regions under consideration are shown in Figure 3.2-1.

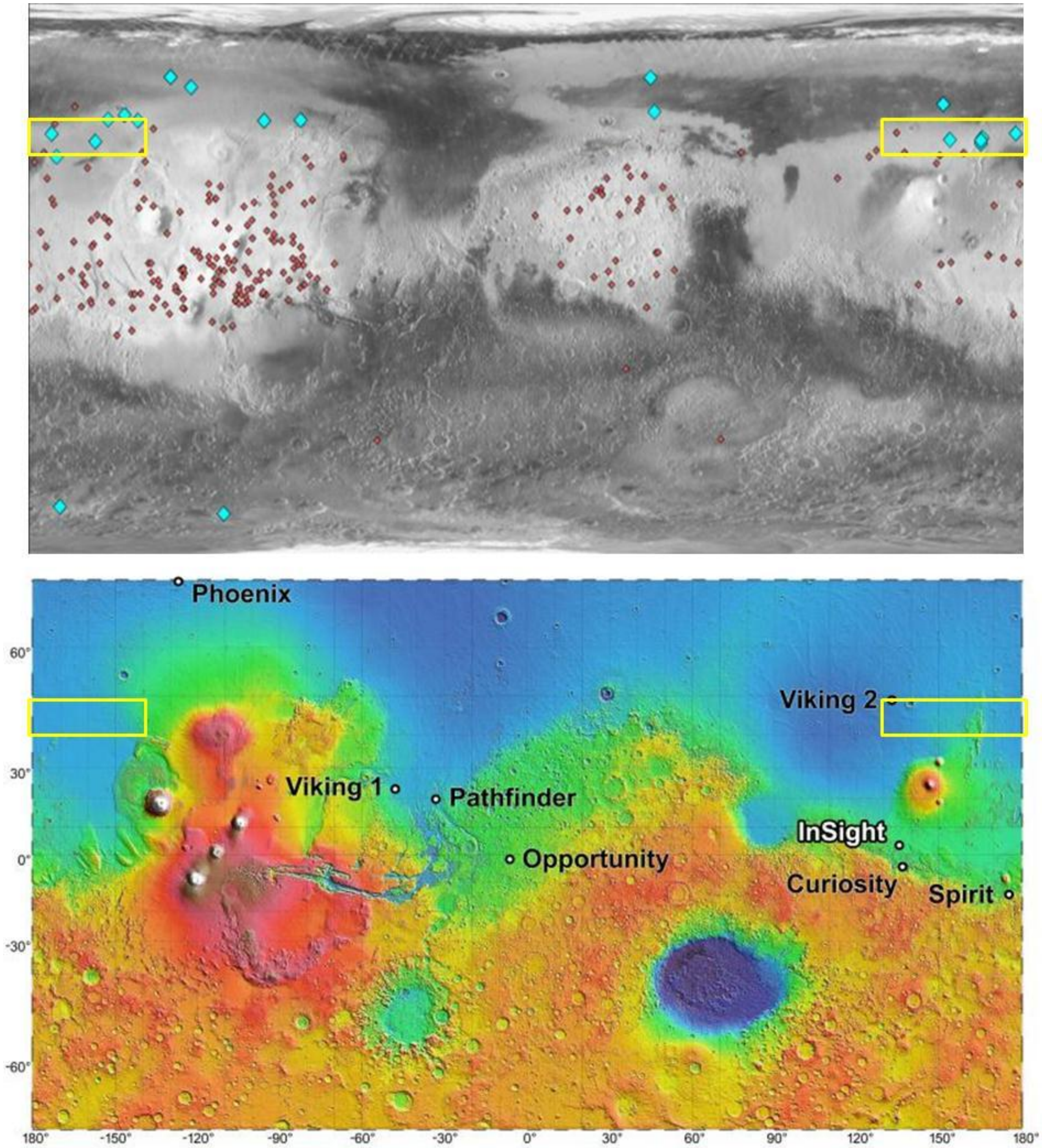


Figure 3.2-1. Upper image show infrared imagery with light colored diamonds indicating fresh icy craters (Image credit NASA/JPL-Caltech). Dark diamonds indicate fresh craters without signs of ice. Lower image shows color MOLA relief with US lander landing sites (Image credit NASA/JPL-Caltech/Arizona State University). Yellow box indicates Mars One Precursor landing regions under consideration.

3.2.2 Descent and Landing

The Mars One 2018 precursor lander is unguided. Therefore, the landing footprint will be on the order of 150x30 km. One candidate location is shown in Figure 3.2-2 showing what may be ice polygon terrain.

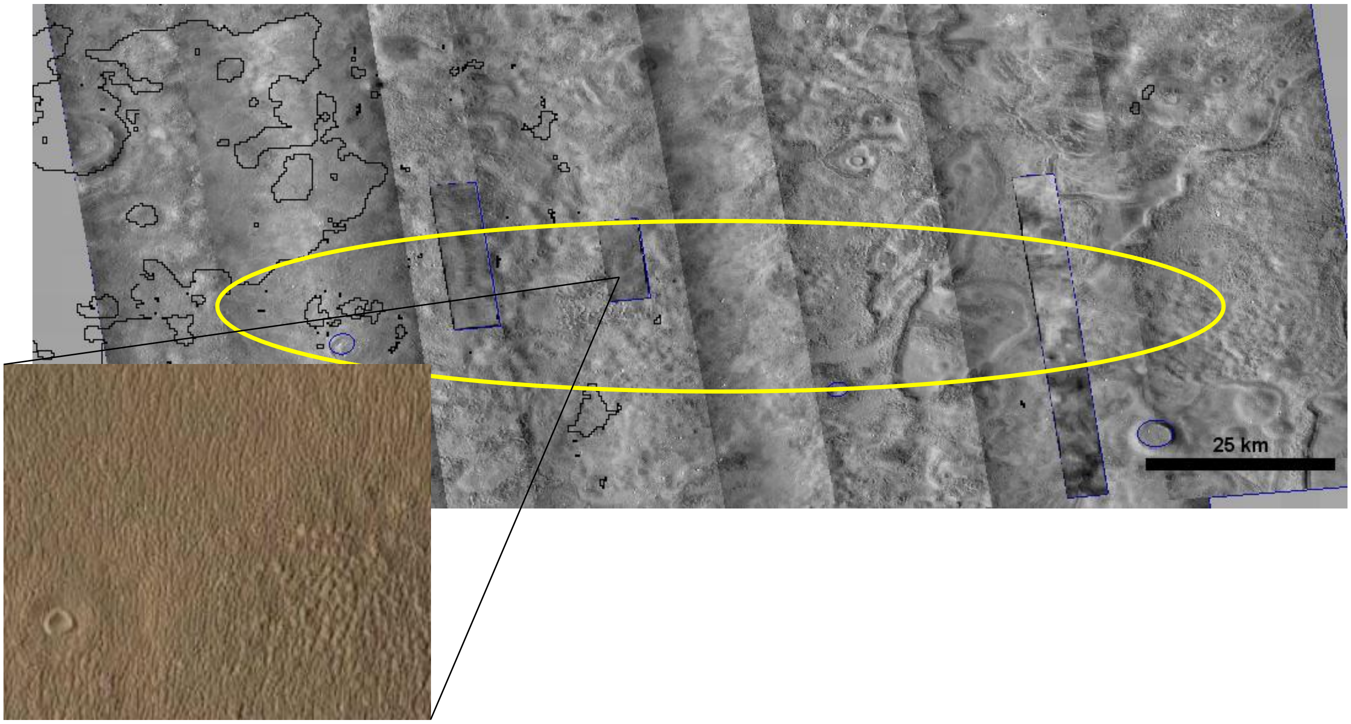


Figure 3.2-2. One possible landing ellipse near 45N, 210E showing polygonal terrain usually indicative of near surface ice. (Image credit NASA/JPL-Caltech/Arizona State University)

3.2.3 Landing Site Elevation

The Mars One 2018 precursor lander is planned to target a potential future site for crewed missions. To maximize landed mass, lower altitudes are desired. To that end, the nominally planned elevation is less than or equal to – 4.0 km with respect to the MOLA geoid. All performance analyses, including derivation of payload mass allocations, assume a -4.0 km altitude.

3.3 RESOURCES AVAILABLE FOR PAYLOAD OPERATIONS

Payload	Not-To-Exceed Mass (kg)	Available Volume	Daytime Power Switches	Nighttime Power Switches	Data Interface (input/output is spacecraft relative)	TTL Discrete Ins/Outs	Temp Sensors	Pyro Circuits
Water Extraction	10	1, 1a	1	1	Asynch RS-422 input Asynch RS-422 output	Limited	2	2
Sample Acquisition	15	2, 2a	1	1	Asynch RS-422 input Asynch RS-422 output		2	2
Solar Array Demonstration	6	3	1 if needed	1 if needed	From spares if needed		2	2
Camera System	5	4, 4a, 4b	1	1	Synchronous RS-422 input Asynch RS-422 output		2	2
Optional Payload A	2	5	1 if needed	2 shared among all optional payloads	Spare Pool 2 Asynch RS-422 inputs 2 Asynch RS-422 outputs 2 Synch RS422 inputs 2 LVDS Inputs 2 LVDS Outputs"		2	2
Optional Payload B	2	6	1 if needed				2	2
Optional Payload C	2	7	1 if needed				2	2
Optional Payload D	2	8	1 if needed				2	2
TOTAL	44		8	6			16	16

Table 3.3-1. Payload resource allocations.

Table 3.3-1 summarizes the resources available to payloads on the Mars One 2018 Precursor Lander. The first column identifies the payload. The second column is the Not-to-Exceed mass allocated to that payload. The third column is the volume identification as discussed further in section 3.3.2. The next column contains the number of daytime power switches, discussed in section 3.4.3.3.1. The next column contains the number of allocated Nighttime Power Switches, described in section 3.4.3.3.2. The water extraction, camera system, and sample acquisition system have been allocated data interface in the sixth column. They are described in the first three subsections of section 3.4.3.1. Note that inputs and outputs are from the spacecraft point of view. There are a limited number of TTL discretes, not enough to assign to every payload. They are described in section 3.4.3.1.5 and 3.4.3.1.6. Any payload proposing use of discretes much provide justification. Payloads should have alternate methods of data transfer for any discretes they propose in case the project prioritizes these discretes to other needs. The next column assigns up to two temperature sensors to each payload. The two types are described in sections 3.4.3.2.1 and 3.4.3.2.2. Payload may propose their preference. After selection, it may be necessary to change types. Finally, the last column shows that each payload is allocated two pyro circuits described in section 3.4.3.3.3. It is important to note that there is no guarantee that a payload will be assigned any resources beyond what is specifically identified in Table 3.3-1.

Two types of analog interfaces available only to the solar array experiment are described in sections 3.4.3.2.3 and 3.4.3.2.4. Only two current analog channels exist and are available only to the solar array experiment. Nominally, voltage analog channels are only allocated to the solar array experiment. Any other payload that proposes analog voltage channels must provide justification. There is no guarantee that a payload will be assigned any analog voltages.

3.3.1 Mass

Mass is allocated to each payload as listed in Table 3.3-1. Note this is the allocated mass for each payload and must include all payload hardware, targets, and intra-instrument cabling. Proposers should be aware that the mass allocations in Table 3.3-1 represent the mass they must not exceed as launch. At a minimum, proposers must show sufficient margin from the current best estimate (CBE) to the allocation to meet the guidelines in AIAA-S-120-2006 for mass margin. Experience is that planetary mission payloads almost always exceed these needs. Mars One may make more or less mass available as necessary to meet the projects prioritized objectives.

3.3.2 Landed Payload Volume

The following section describes the payload volume allocations for the Mars One 2018 lander. Each payload is given a rectangular box that is completely available to the payload. Also provided is the larger section each payload is assigned to, which may include certain flight system components and cabling. Excursions outside the rectangular area may be considered on a case by case basis. For payloads that will be deployed, an attempt has been made to provide additional information with respect to post landed configurations and clearances.

Figure 3.3-1 shows the preliminary Mars One lander deck layout. Payload volumes available to proposers are numbered. The lander is in the stowed configuration, but the backshell and heatshield are not shown for clarity.

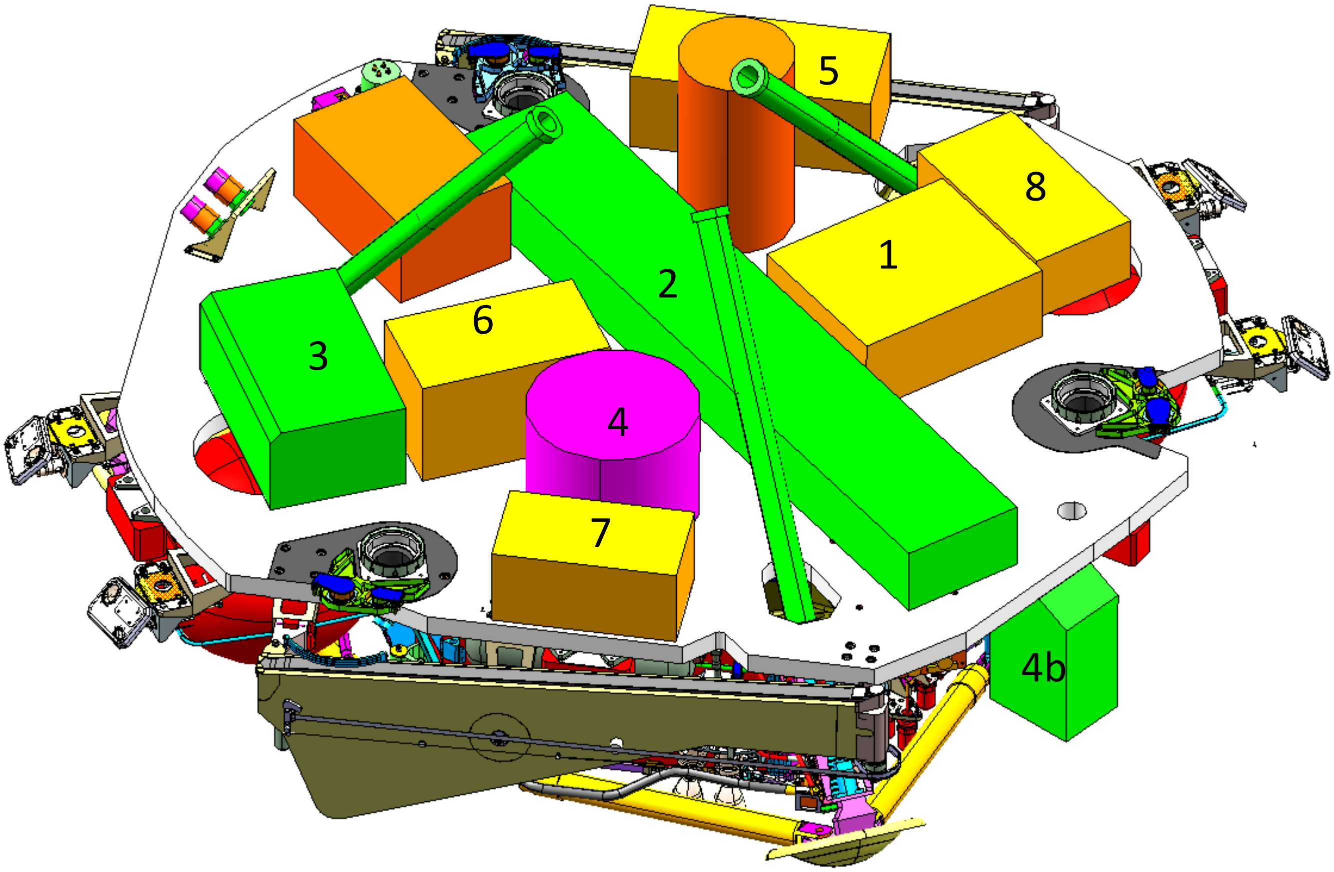


Figure 3.3-1. Volumes allocated to payloads per Section 3.3.2

Figure 3.3-2 shows a top down view of the deck. Again, the volumes available to proposers are numbered.

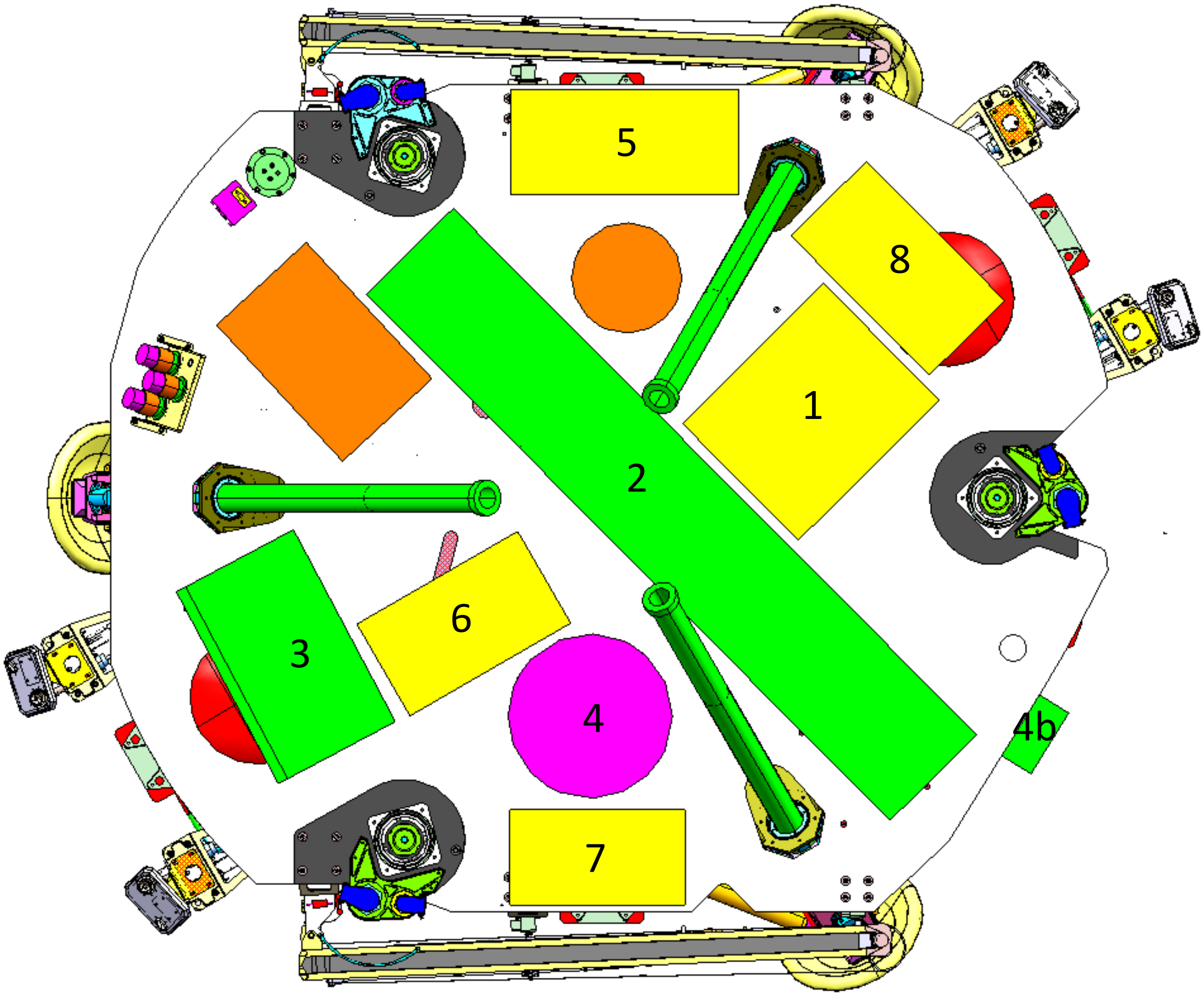


Figure 3.3-2. Top down view of volumes allocated to payloads.

Figure 3.3-3 shows the same view but with the larger dynamic envelopes as translucent shapes around the boxes.

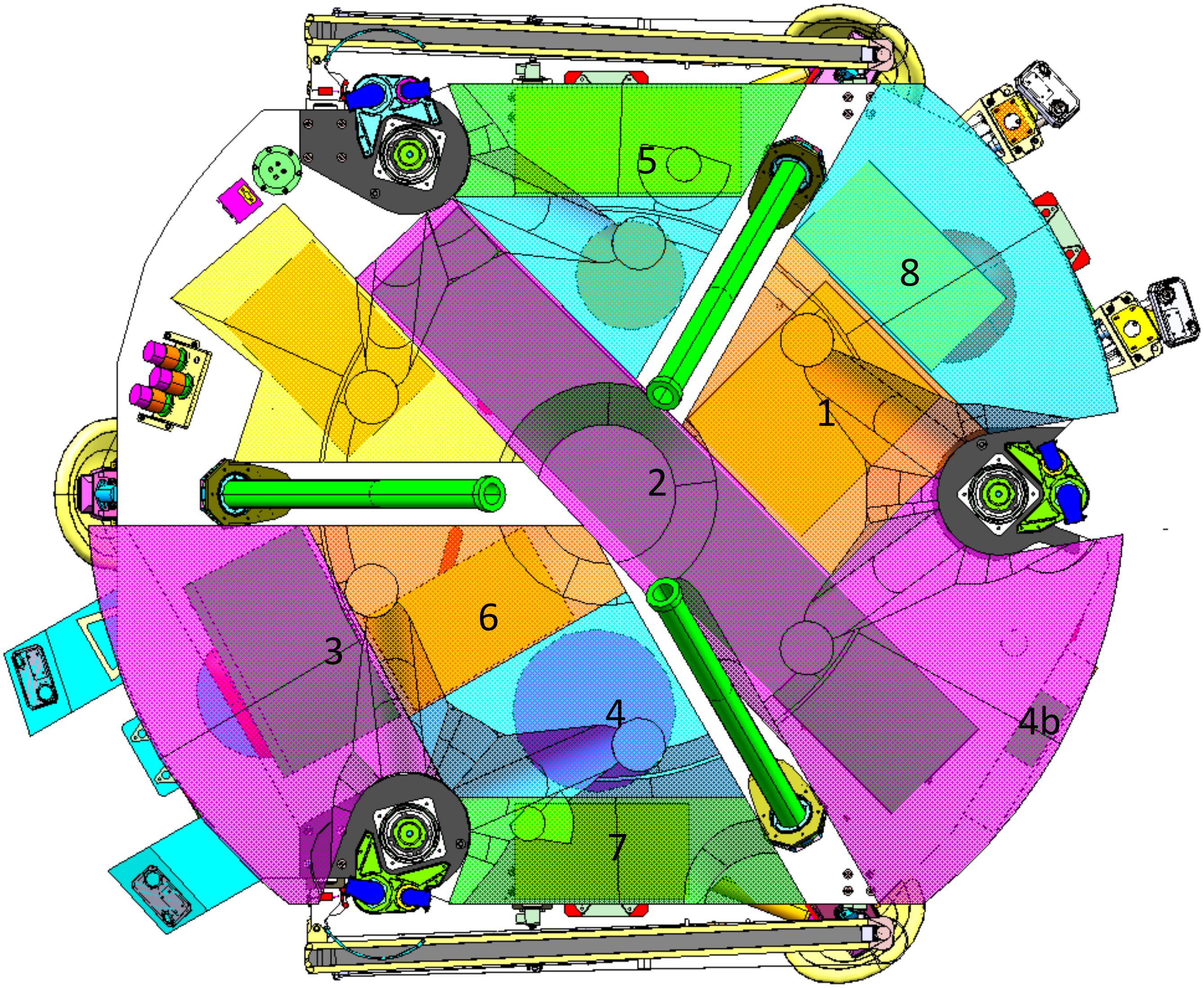


Figure 3.3-3. Allocated payload volumes with dynamic envelopes.

Figure 3.3-4 shows the lander in the landed configuration. Note the sample acquisition system assigned to payload volume 2. Depending on the design, it will be able to access a region similar to the one shown in Figure 3.3-4. Presently, the only payload that the sample acquisition is required to reach is payload box 1, the water extraction experiment. Also important to note is that payload volume 4b, allocated to an underdeck camera head, is located such that it should have an easy view of the surface workspace. Finally, payload volume 4 is currently allocated to the camera system. From there, likely after a deployment, it should meet all its field of view requirements discussed in section 3.3.2.3.

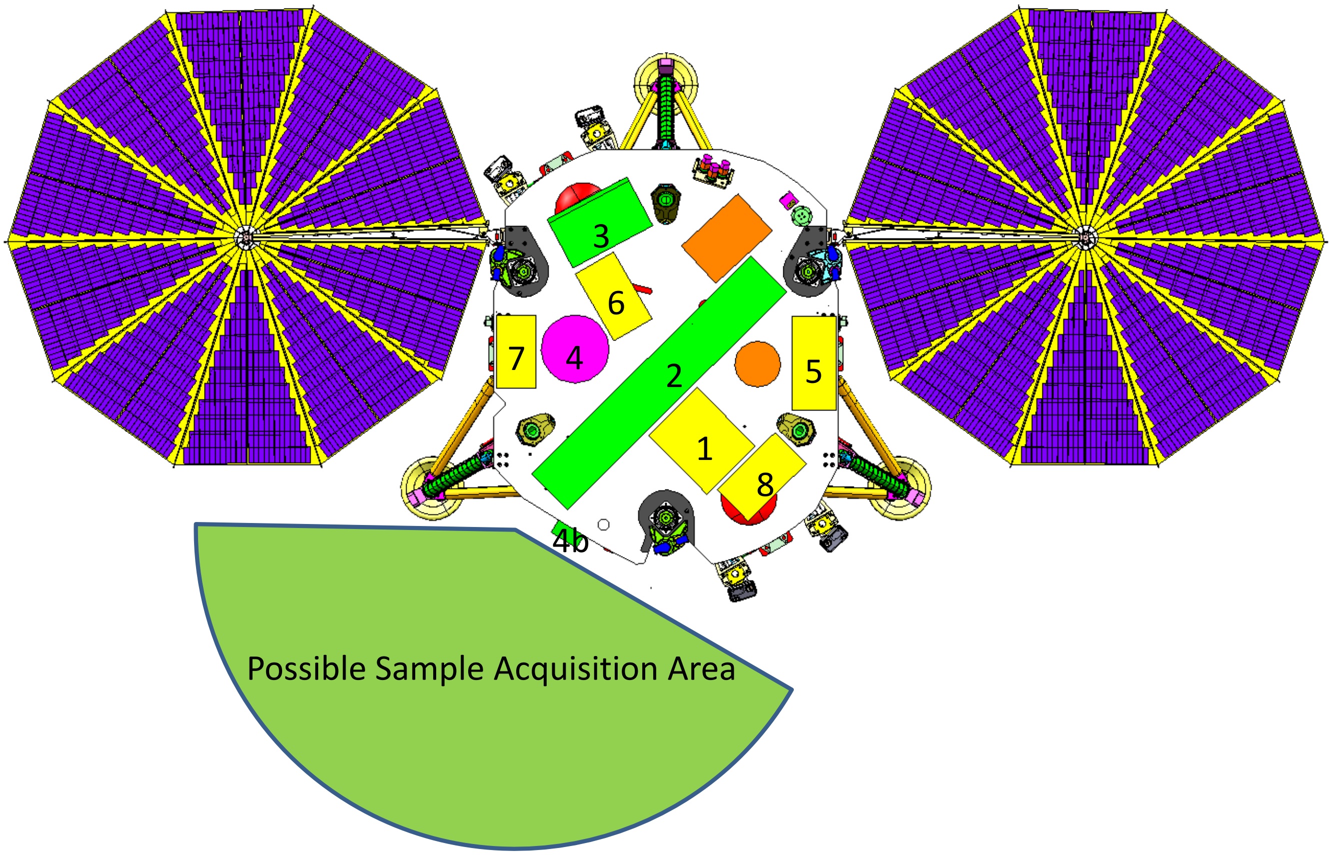


Figure 3.3-4. Landed configuration with potential dig area and allocated payload volumes shown.

3.3.2.1 Water Extraction Payload Volume

The water extraction payload stowed volume is shown in figure 3.3-5. A rectangular box 200mm X 130 mm X 370 mm is the nominal allocated volume. Minor excursions outside this box are possible and will be evaluated on a case by case basis. In no case may any excursion exceed the larger envelope found in the file Water-Exp-Env-040214.stp. Note this volume represents the stowed volume and can be exceeded on a case by case basis by a deployment after landing if necessary. It is expected that a delivery port for samples will be required somewhere on the top face of this payload.

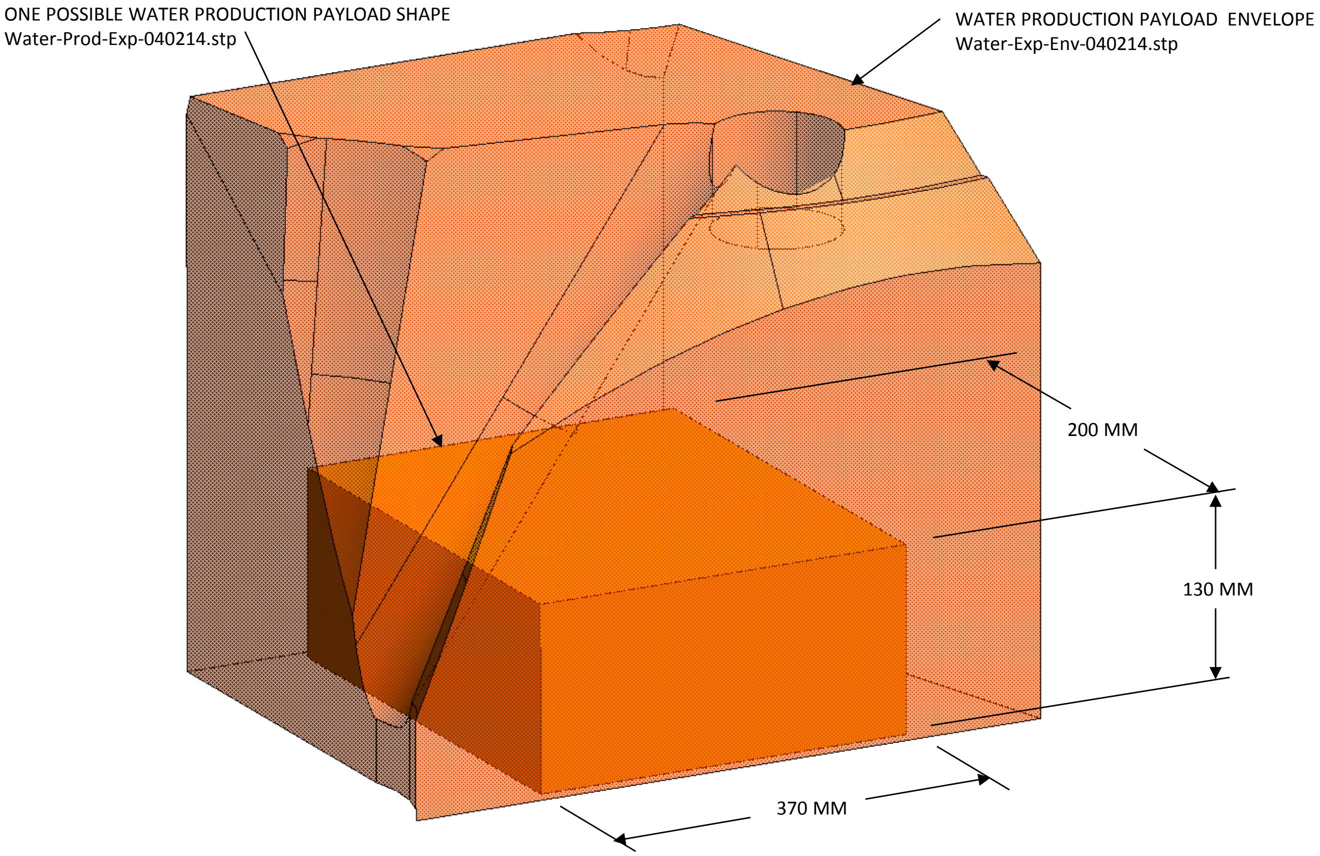


Figure 3.3-5. Water extraction payload volume. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

The water extraction experiment is expected to require operations for more hours per sol than the nominal 7 hours the lander is expected to be awake. Because no energy is allocated for payload heater power, the water extraction experiment may elect to house its temperature sensitive electronics inside the lander thermal enclosure. This enclosure is thermally regulated, see the environmental section later. Figure 3.3-6 shows the volume allocated to an optional water extraction payload electronics box.

Payload-Elect-Mars1-040314.stp

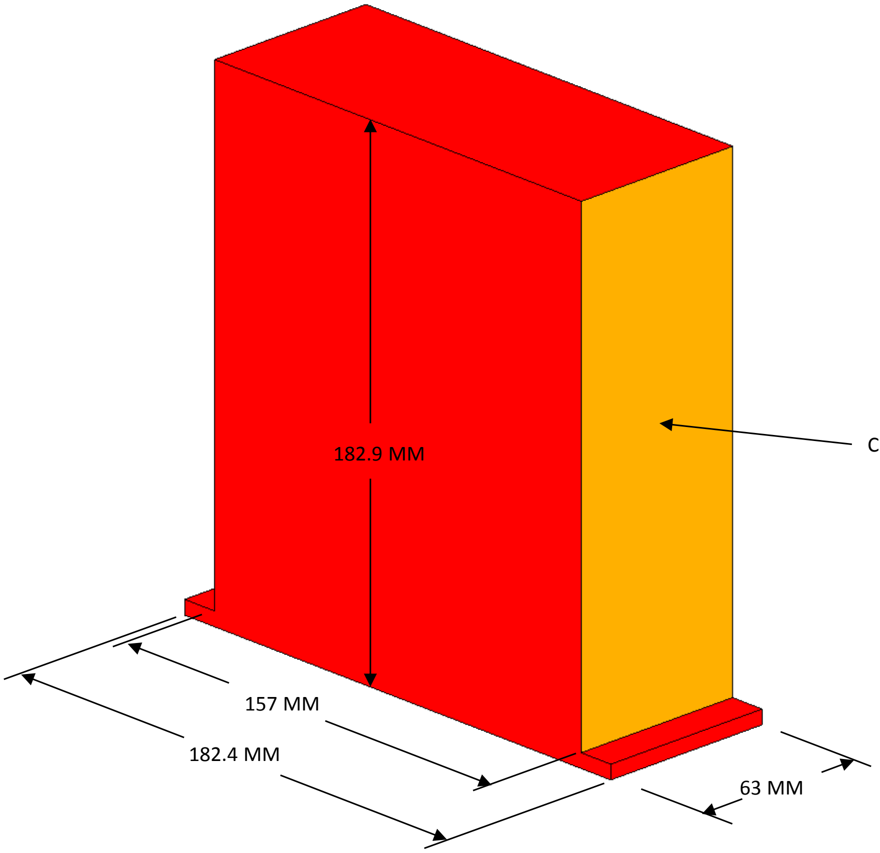


Figure 3.3-6. Water extraction experiment optional electronics volume for lander thermal enclosure.

3.3.2.2 Sample Acquisition System Payload Volume

The sample acquisition payload stowed volume is shown in Figure 3.3-7. A rectangular box 230mm X 130 mm X 1400 mm is the nominal allocated volume. Minor excursions outside this box are possible and will be evaluated on a case by case basis. In no case may any excursion exceed the larger envelope found in the file SAS-Env-040214.stp. Note this volume represents the stowed volume and can be exceeded upon deployment after landing. It is also expected that the sample acquisition system will need access to the edge of the deck. The edge of the deck can be altered slightly to fit a sample acquisition system, and the end of the rectangular box may reach the edge of the deck, subject to the constraints in SAS-Env-040214.stp. Note the deck itself can be found in file Science-Deck-040314.stp.

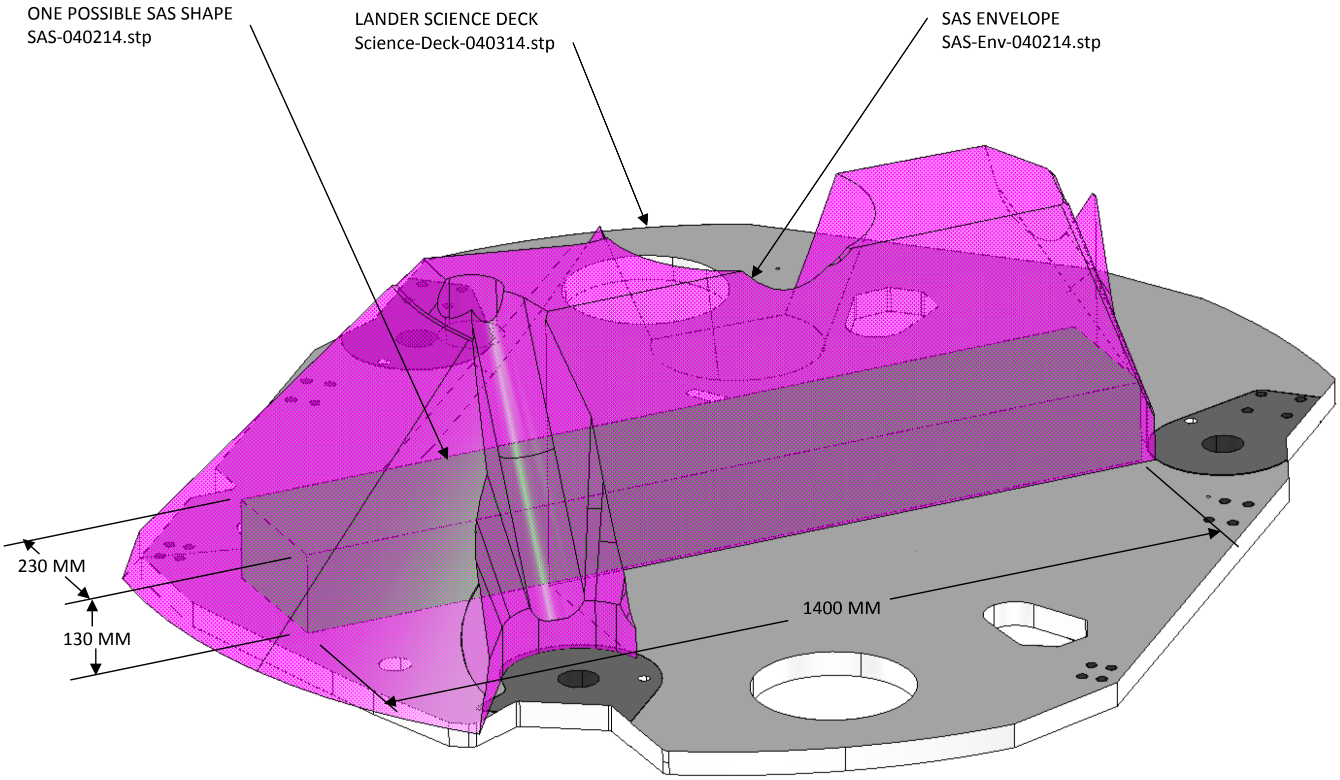


Figure 3.3-7. Sample acquisition system volume. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

The sample acquisition system must be able to reach the surface. The edge of the deck can be as close to the surface as 838 mm while tipped 6.7 degrees towards the surface work space and as high as 1074 mm while tipped as much as 6.6 degrees away from the work space. The arm must be able to reach the surface and dig to the depth required in the RFP.

The sample acquisition system is expected to be used throughout the entire mission lifetime. Because no energy is allocated for payload heater power, the sample acquisition system may elect to house its temperature sensitive electronics inside the lander thermal enclosure. This enclosure is thermally regulated, see the environmental section later. Figure 3.3-8 shows the volume allocated to an optional sample acquisition payload electronics box.

Payload-Elect-Mars1-040314.stp

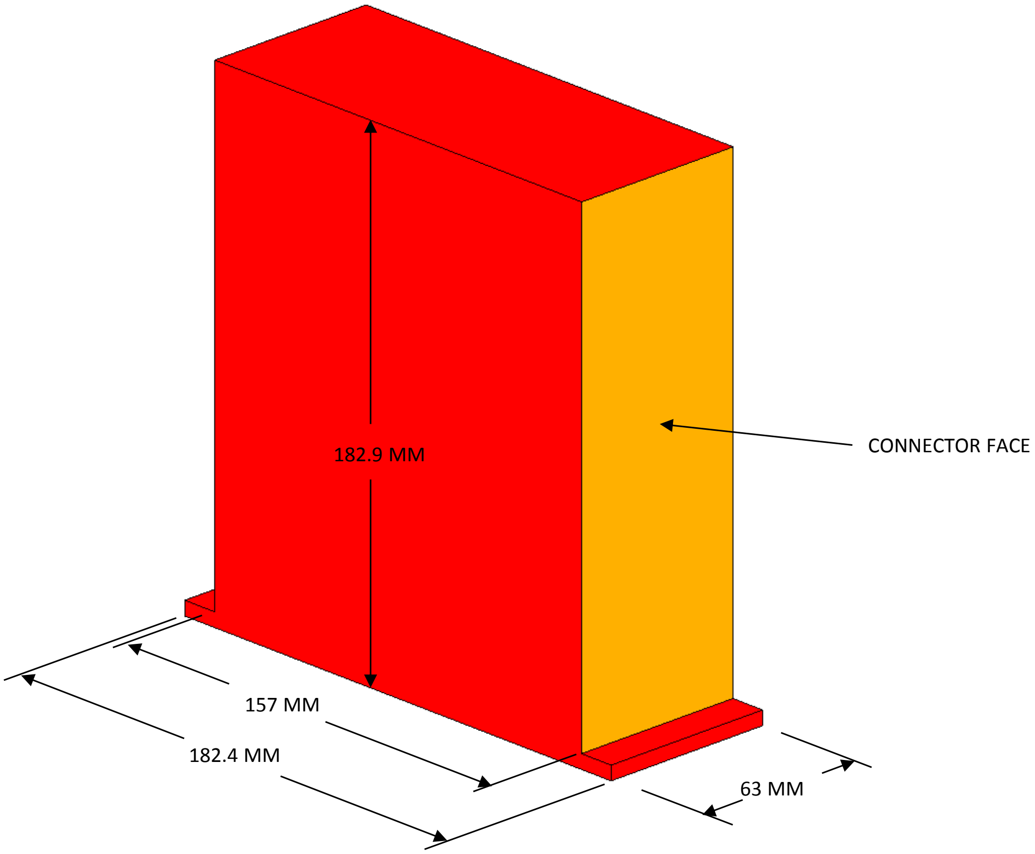


Figure 3.3-8. Sample acquisition system optional electronics volume for lander thermal enclosure.

3.3.2.3 Camera System Payload Volume

The camera payload stowed volume is shown in Figure 3.3-9. A cylinder 310 mm diameter X 225mm tall is the nominal allocated volume. The cylindrical volume assumes a camera head and deployment mast, however, any camera volume that does not exceed the larger envelope found in the file Camera-Env-040214.stp may be considered. Note this volume represents the stowed volume and is expected to be exceeded after deployment. The camera system must be able to image any other payload on the main deck (Science-Deck-040314.stp). It must also be able to image the work space from which the sample acquisition system will obtain samples. Note the edge of the deck can be as low as 838 mm above the surface while tipped 6.7 degrees towards the line bisecting the angle between any two legs and as high as 1074 mm above the surface while tipped as much as 6.6 degrees away in the direction of any one leg.

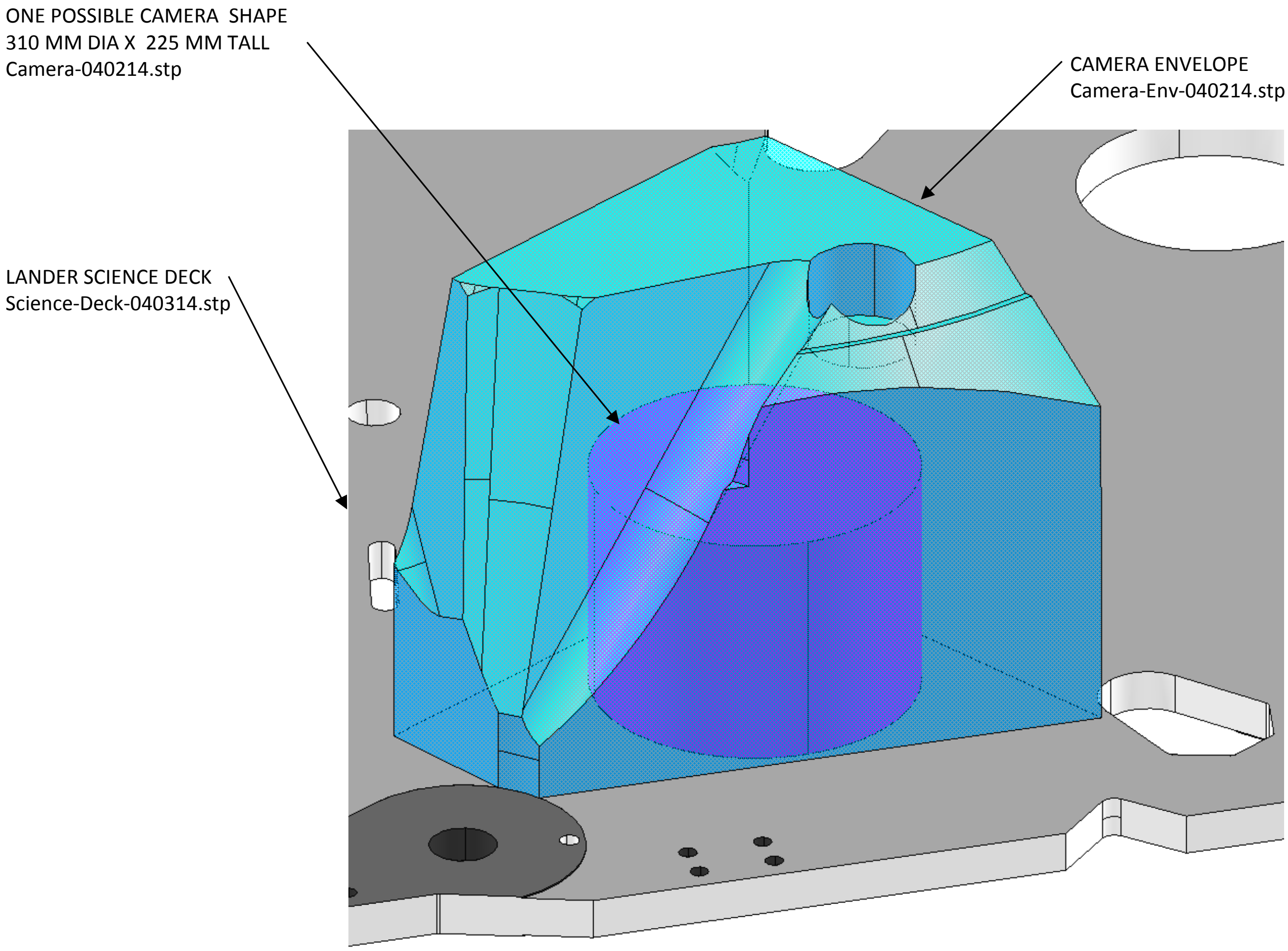


Figure 3.3-9. Camera system volume. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

The camera system will be used to produce digital elevation maps of the work space for planning the sample acquisitions. This implies some type of stereo imagery. While it is possible to include both heads in the already described camera volume, two other options exist. First, a provider may propose mounting a camera head to the sample acquisition system. There is no guarantee this will be possible. Another option is to mount an additional camera head under the deck where it can image the work space. A camera head mounted under the deck may also be used as a descent imager, if so proposed. The volume allocated to an under deck mounted camera head is shown in Figure 3.3-10.

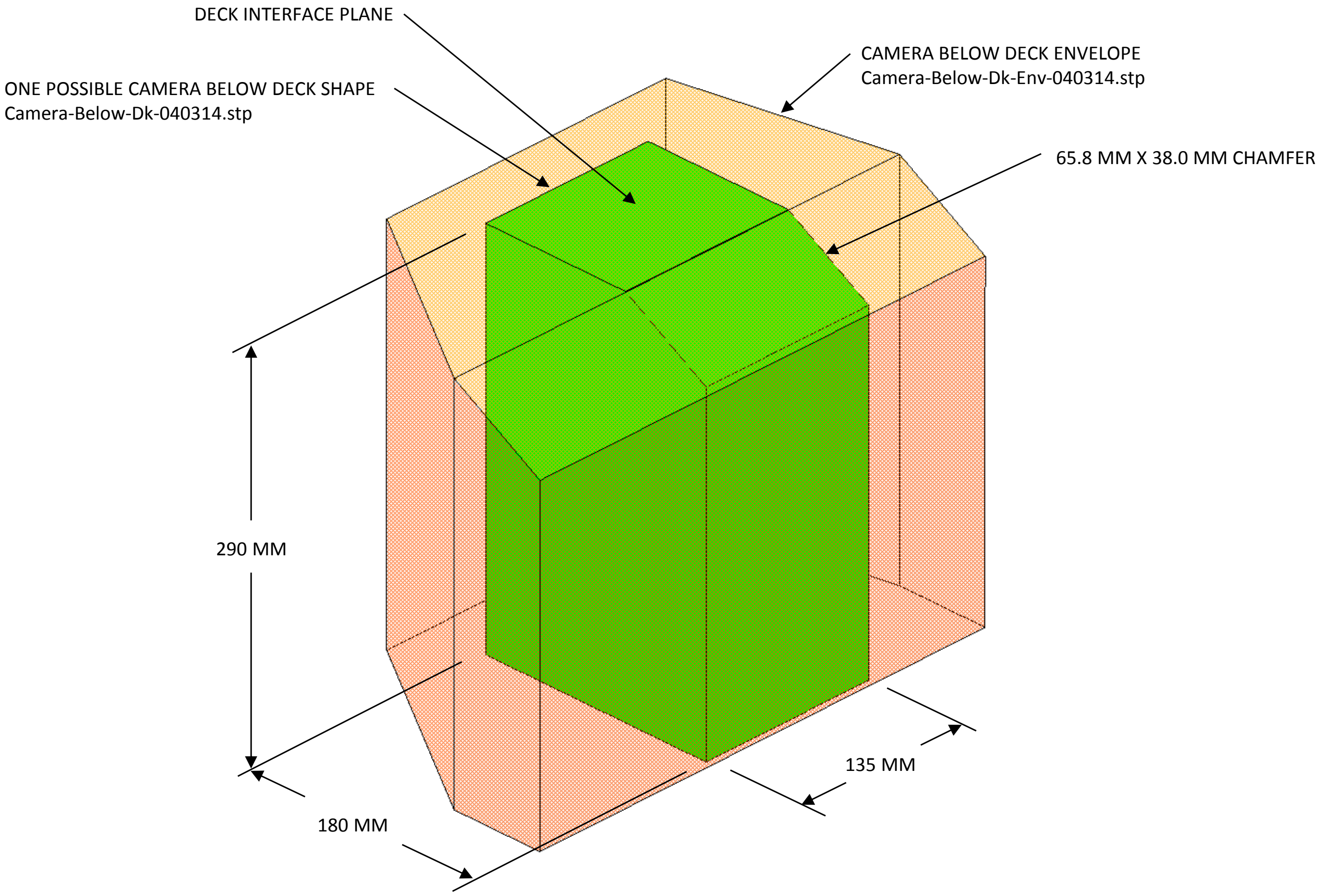


Figure 3.3-10. Camera system optional under deck camera head volume. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

The camera system is expected to operate throughout the landed mission. Because no energy is allocated for payload heater power, the camera system may elect to house its temperature sensitive electronics inside the lander thermal enclosure. This enclosure is thermally regulated, see the environmental section later. Figure 3.3-11 shows the volume allocated to an optional camera system electronics box.

Payload-Elect-Mars1-040314.stp

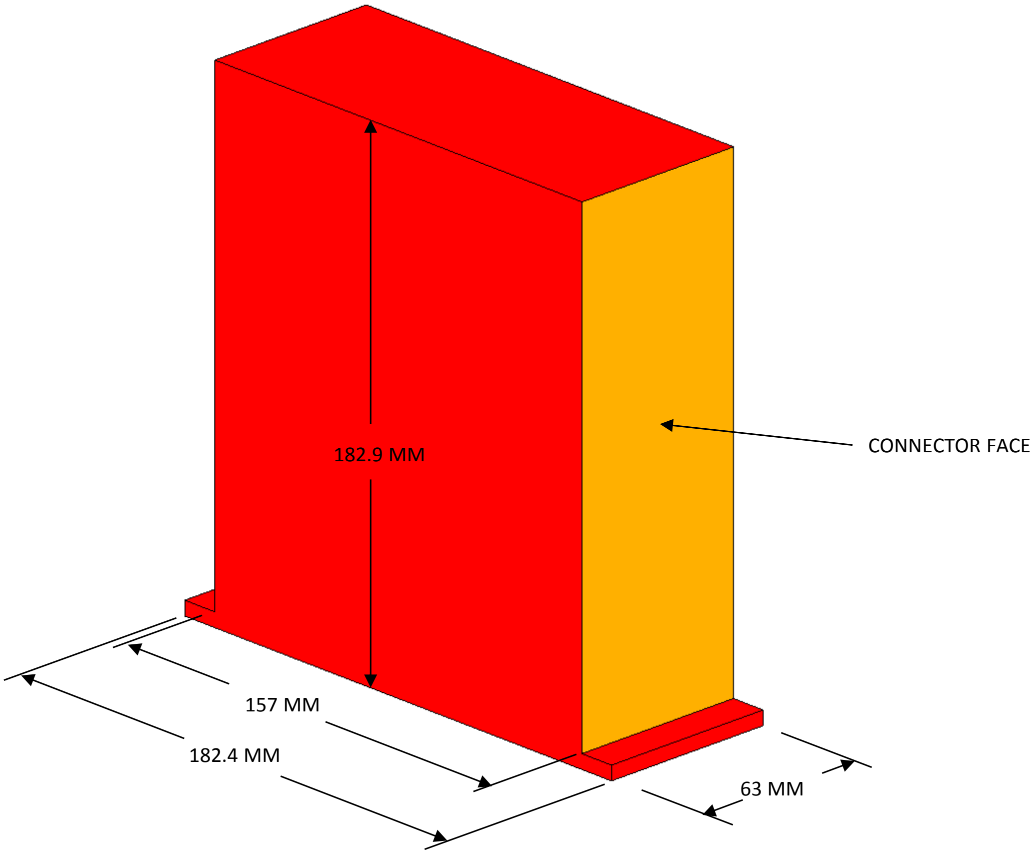


Figure 3.3-11. Camera system optional electronics box for lander thermal enclosure.

3.3.2.4 Solar Array Demonstration Payload Volume

The solar array demonstration payload stowed volume is shown in Figure 3.3-12. A rectangular box 170mm X 250 mm X 410 mm is the nominal allocated volume, with a chamfer that takes 20 mm off the top and outer edge faces. Minor excursions outside this box are possible and will be evaluated on a case by case basis. In no case may any excursion exceed the larger envelope found in the file Solar-Array-Demo-Env-040214.stp. This is the stowed volume requirement. It is expected that the deployed array will exceed this volume. The file Sci-Dk-Edge-Thrstr-and-Bal-Wt-Env-040214.stp includes the deck edge and other possible obstructions the solar array may encounter during a deployment. Metal edges can and should be assumed to be sharp.

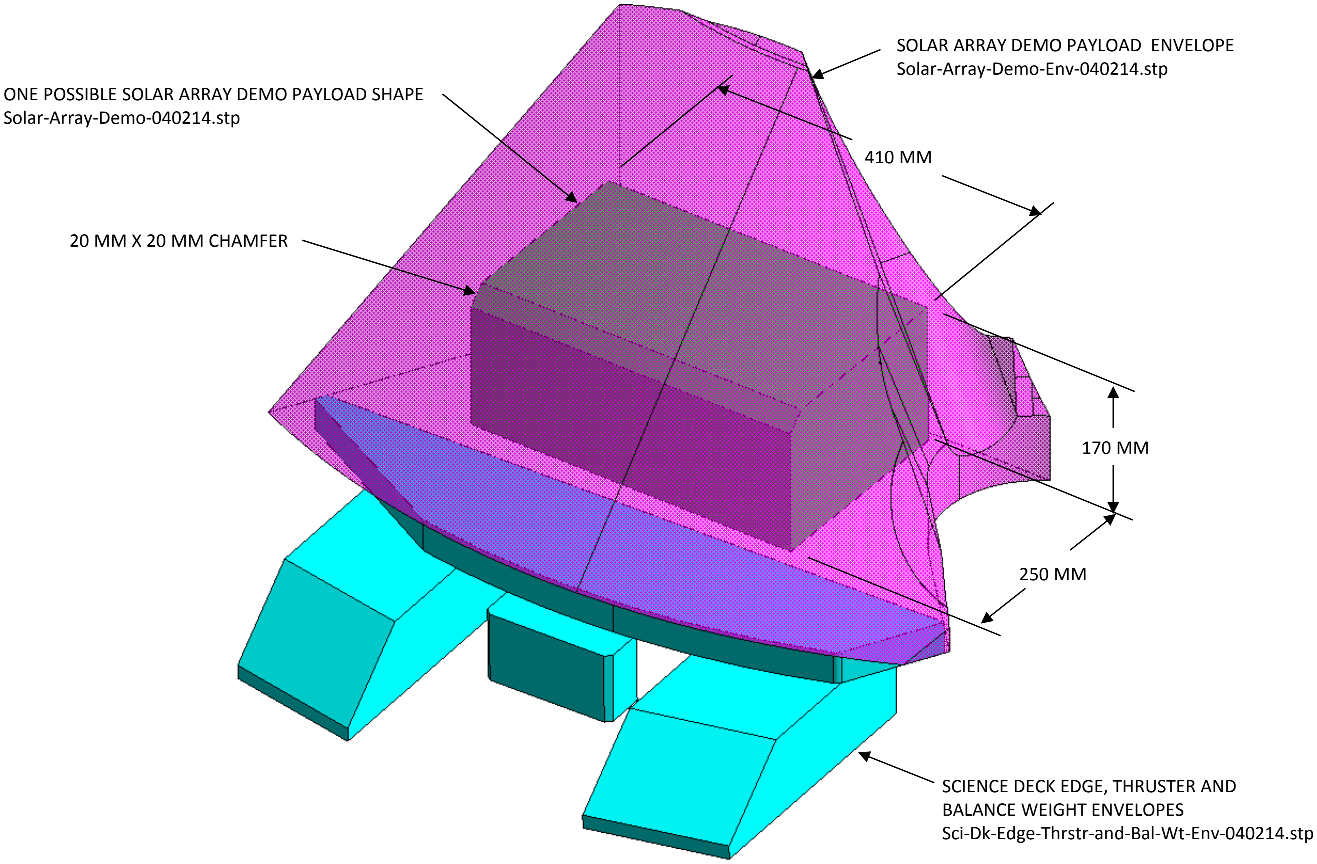


Figure 3.3-12. Solar array demonstration stowed volume allocation. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

3.3.2.5 Optional Payload A Payload Volume

Four additional payload volumes have been designated should Mars One elect to select additional payloads that further the Mars One mission objectives. The optional payload volume A stowed volume is shown in Figure 3.3-13 . A rectangular box 160mm X 200 mm X 430 mm is the nominal allocated volume. Minor excursions outside this box are possible and will be evaluated on a case by case basis. In no case may any excursion exceed the larger envelope found in the file Payload-A-Env-040214.stp. Note this volume represents the stowed volume and can be exceeded on a case by case basis by a deployment after landing if necessary.

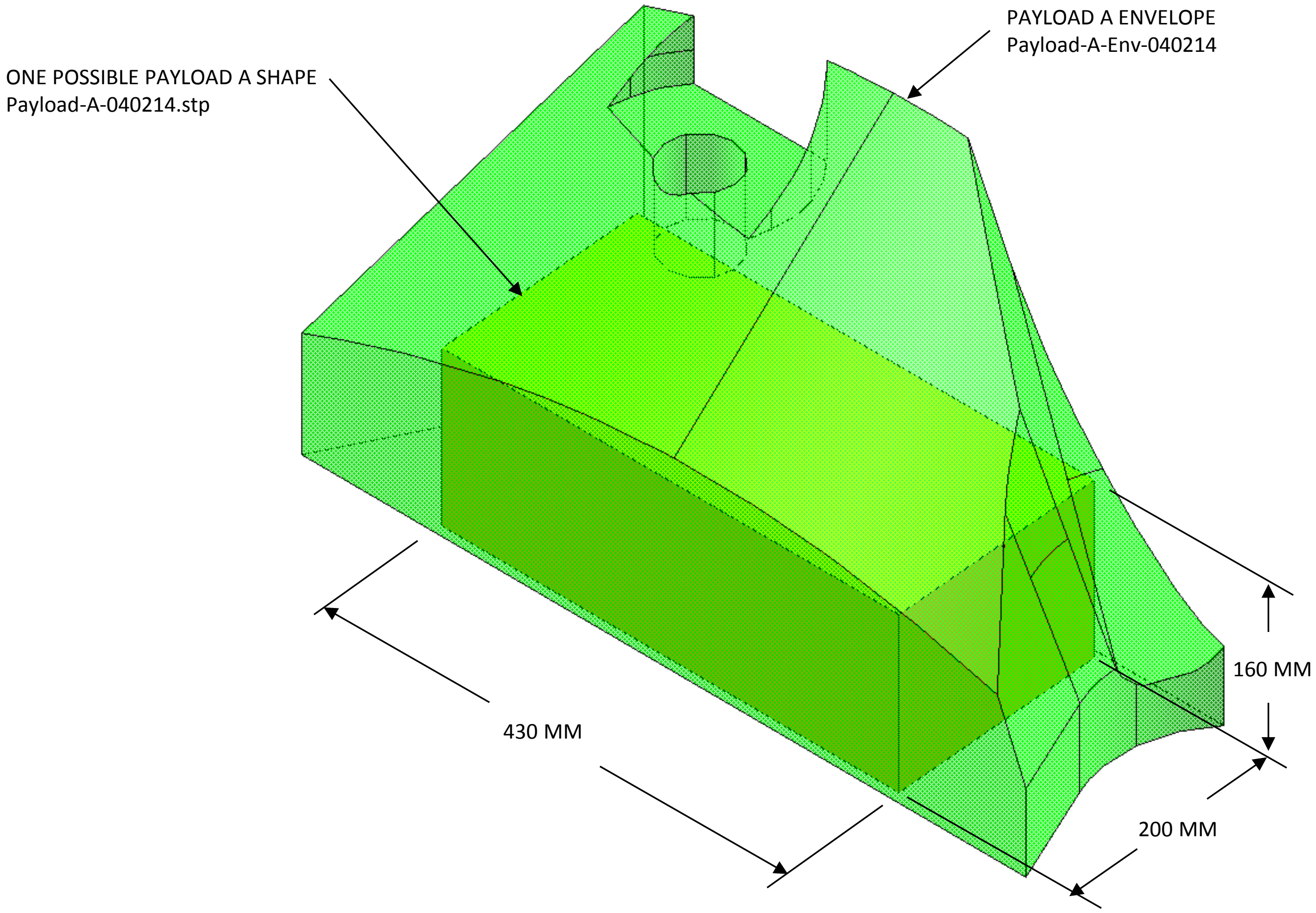


Figure 3.3-13. Optional Payload A allocated volume. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

3.3.2.6 Optional Payload B Payload Volume

Four additional payload volumes have been designated should Mars One elect to select additional payloads that further the Mars One mission objectives. The optional payload volume B stowed volume is shown in Figure 3.3-14. A rectangular box 210mm X 200 mm X 350 mm is the nominal allocated volume. Minor excursions outside this box are possible and will be evaluated on a case by case basis. In no case may any excursion exceed the larger envelope found in the file Payload-B-Env-040214.stp. Note this volume represents the stowed volume and can be exceeded on a case by case basis by a deployment after landing if necessary.

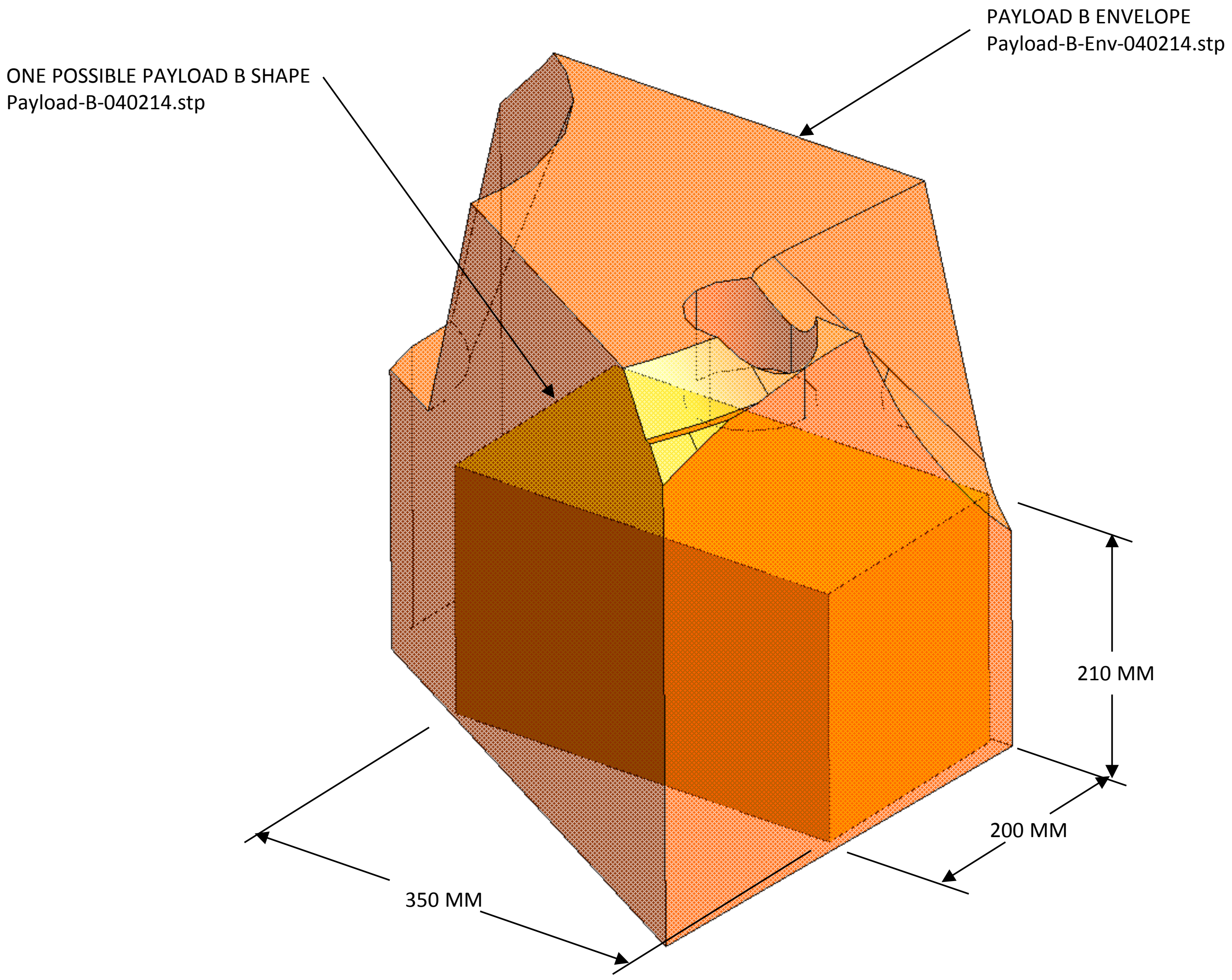


Figure 3.3-14. Optional Payload B allocated volume. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

3.3.2.7 Optional Payload C Payload Volume

Four additional payload volumes have been designated should Mars One elect to select additional payloads that further the Mars One mission objectives. The optional payload volume C stowed volume is shown in Figure 3.3-15. A rectangular box 180mm X 150 mm X 330 mm is the nominal allocated volume. Minor excursions outside this box are possible and will be evaluated on a case by case basis. In no case may any excursion exceed the larger envelope found in the file Payload-C-Env-040214.stp. Note this volume represents the stowed volume and can be exceeded on a case by case basis by a deployment after landing if necessary.

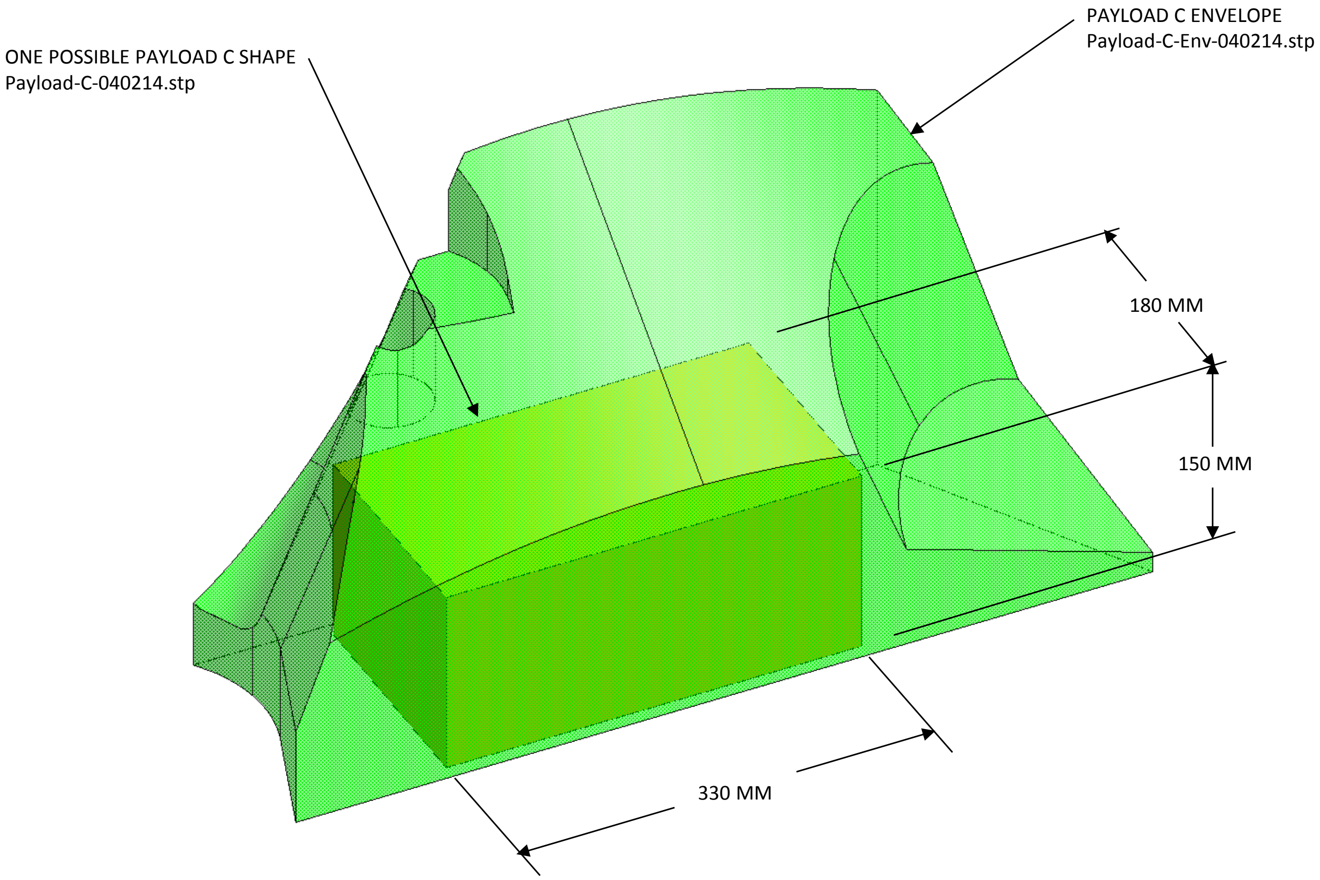


Figure 3.3-15. Optional Payload C allocated volume. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

3.3.2.8 Optional Payload D Payload Volume

Four additional payload volumes have been designated should Mars One elect to select additional payloads that further the Mars One mission objectives. The optional payload volume D stowed volume is shown in Figure 3.3-16. A rectangular box 130mm X 200 mm X 370 mm is the nominal allocated volume. Minor excursions outside this box are possible and will be evaluated on a case by case basis. In no case may any excursion exceed the larger envelope found in the file Payload-D-Env-040214.stp. Note this volume represents the stowed volume and can be exceeded on a case by case basis by a deployment after landing if necessary.

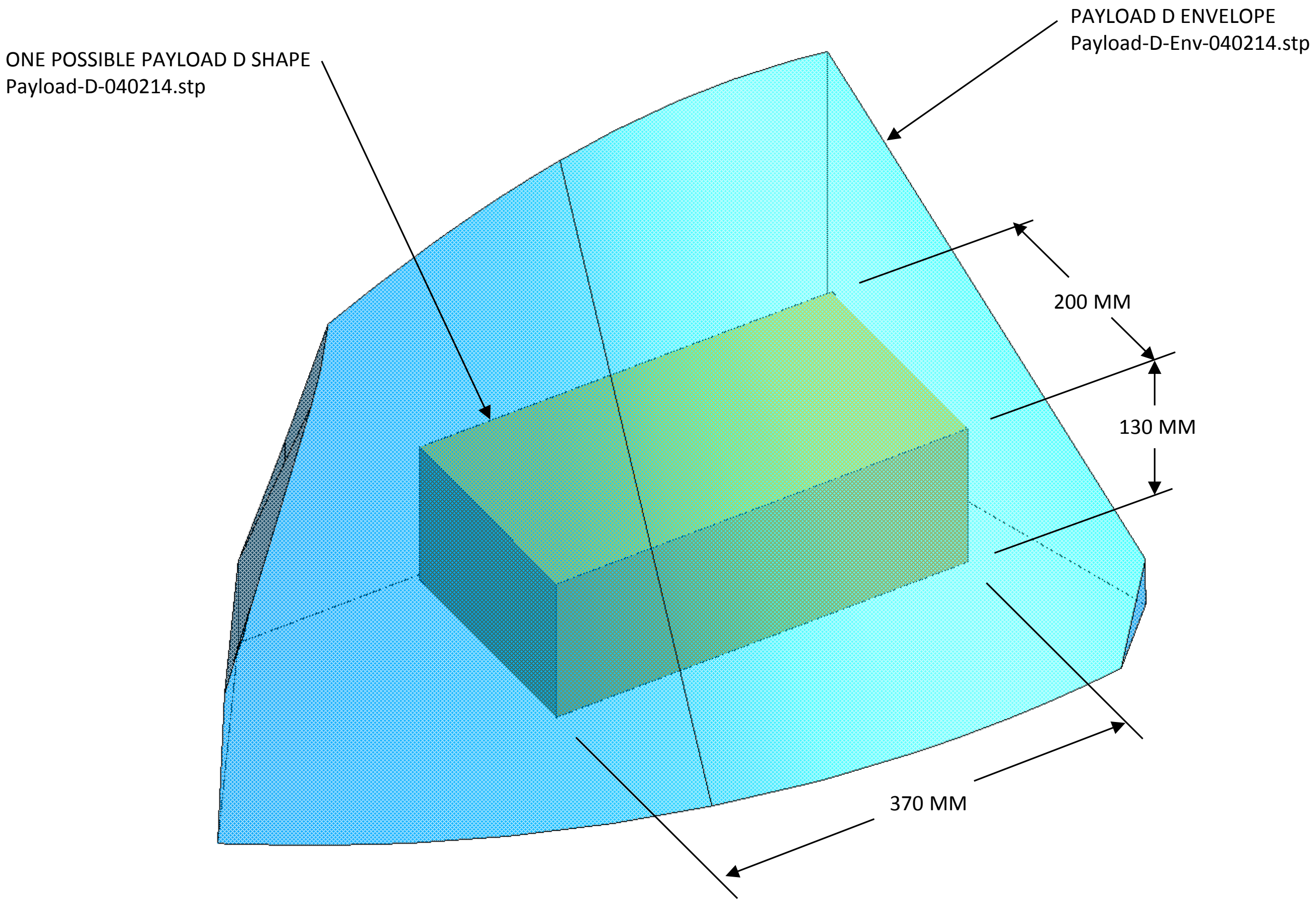


Figure 3.3-16. Optional Payload D allocated volume. Solid color shape represents nominal payload volume. Translucent shape represents additional space that may be negotiable depending on final flight system configuration.

3.3.3 Power Energy Available

The total energy available for all payloads is 700 Watt hours per sol, including any replacement heater power. Of this, 300 watt hours can be spent while the spacecraft is sleeping, most of this will be at nighttime. Nominally the lander will be awake for up to seven hours per sol, depending on energy collection loss due to dust build up on the solar arrays. Most sols, the energy is to be used for water extraction operations, that is, the camera, the sample acquisition system and the extraction system. Opportunistically, other activities can be undertaken.

Payloads cannot at any time exceed the maximum power that its assigned switch can tolerate. Standard daytime load switches can pass up to three amps from 34-22V, though to do this for very long will exceed the energy allocation. Nighttime latching switches can pass up to 1 amp from 34-22V. More than that could cause the fuse to blow. Again, this is the upper limit of the switch, energy allocations will not allow sustained periods where payload loads are at their switch limits.

3.3.4 Computational Resources

The lander computer is a RAD750 spaceflight computer. Payload software hosted on the RAD750 will be written by Lockheed Martin based on requirements supplied by payload providers. At a minimum, flight software on the lander will act as a pass-through command interface to the payloads. More comprehensive computing on the lander computer will be handled on a case by case basis. LM will manage CPU utilization and memory allocations for this payload software to ensure overall usage does not exceed spacecraft capabilities. Software resident on processors that are contained within the payloads is the responsibility of the payload provider.

The lander is nominally awake for 7 hours per sol, nominally 8 AM to 3 PM local solar time, though this will vary depending on comm pass scheduling. If a payload needs to operate outside this seven hour period, or for longer periods of time, the lander computer will not be available, and the payload must contain its own processor (or at least be able to continue to function without the lander computer).

3.3.5 Data Storage & Return

The lander can store far more data than can be returned with the baseline communications system. Generally, the spacecraft is capable of transmitting 50 Mbits per sol to an orbiter (averaged over many sols). It should be noted that at this time, there are no agreements in place as to exactly how much bandwidth will be made available by the NASA and ESA orbiters. As such, payloads should target the smallest amount of daily data production possible. Every attempt will be made to provide for maximum possible data return, however, the mission must be able to meet objectives with minimal data return. The table below shows the allocated daily data return assuming Direct-to-Earth only capability.

Payload	Daily Data Volume Allocation
Camera System	5 Mbits
Sample Acquisition System	1 Mbit
Water Extraction System	1 Mbit
Solar Array Demonstration	100 Kbits
Each Additional Payload	1 Mbit (divided among 1 to 4 additional payloads)

Table 3.3-2. Daily Data Volume Allocations

Data storage capability for payload packets will exceed the daily data volume allocations above by orders of magnitude, and are therefore not driving constraints.

3.3.6 Lifetime

Analysis shows that the lander will maintain thermal and energy balance for approximately one Earth year after landing. However, because this is highly dependent on the Martian environment, the sample acquisition and water extraction experiments should be sized to complete operations and produce the quantity of water required in the RFP within 4 Earth months. This allows 4 months of margin for variability in environments and dust buildup on the arrays and 4 months margin for operational challenges that slow down the pace of daily operations. The total lifetime from delivery in May of 2017 to end of mission is at least 34 months, including 1 year of ATLO, 10 months of cruise, and one year on the surface of Mars.

3.4 PAYLOAD INTERFACES

3.4.1 Configuration

The launch, cruise and entry configuration is shown in Figure 3.4-1. No payload access is possible once the backshell is installed, nominally prior to shipping the spacecraft to the launch site. Optics covers, if necessary, are the responsibility of the payload. Also, any other protective measures, such as protection from flying debris and dust during touchdown, are the responsibility of the payload. The top deck will see substantial dust buildup over the life of the mission. Furthermore, sample delivery may leave substantial debris on the deck. Payloads are responsible for providing their own protection, if needed.

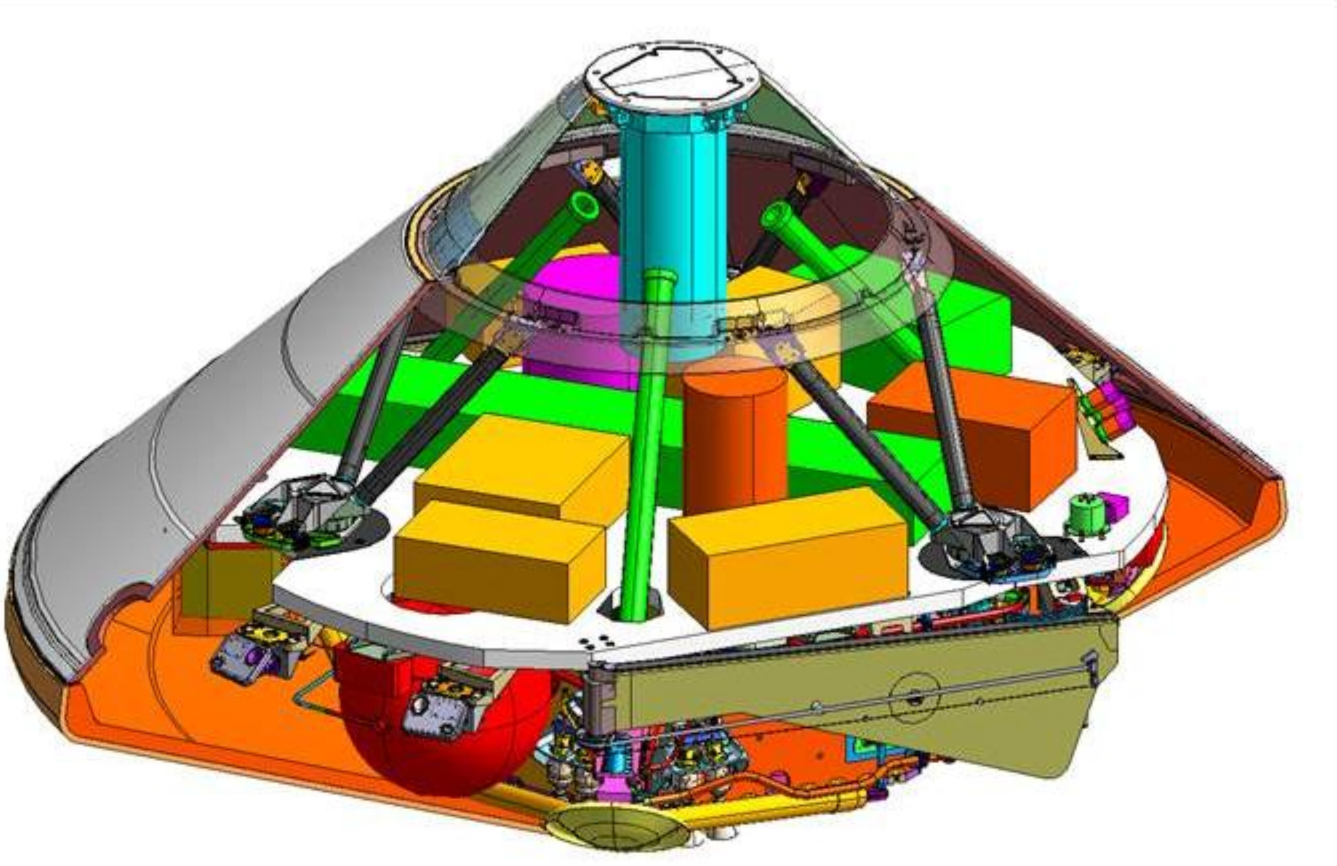


Table 3.4-1. Cutaway view showing payloads packaged inside aeroshell.

3.4.2 Thermal Control and Thermal Interfaces

3.4.2.1.1 Interface Constraints

All payload components shall be mounted to the top deck with the exception of those that have been allocated space inside the thermal enclosure and camera systems proposing hardware mounted in payload volume 4b. Payload suppliers are responsible for providing a thermally isolated interface between the component and the spacecraft bus. Total heat transfer to or from the spacecraft shall not exceed 1 W for each instrument. Exceptions require the consent of both Lockheed Martin and the payload supplier and must be documented in the corresponding ICD.

3.4.2.1.2 Payload Thermal Control Design

The payload supplier is responsible for providing thermal control measures for maintaining required component temperatures independent of spacecraft design. Thermo-electric devices such as heaters or coolers, if used, must be considered as included in the mass, power, and volume budgets of the instrument. Expected worst-case thermal environments from pre-launch shipment through landed operations are provided in section 3.6. These environments should be used for early component design purposes. Top deck temperatures are provided for the cruise phase of the mission. The backshell temperature represents the primary radiative environment for the payload. The backshell should be assumed to have a surface emissivity less than 0.1. The top deck, which represents the mounting surface, and thus both a radiative and a conductive interface, should be assumed to have an effective surface emissivity greater than 0.85.

3.4.2.1.3 Integrated Thermal Analysis and Design

Lockheed Martin will conduct an integrated thermal analysis to ensure compatibility with the lander bus and other payload components. Payload suppliers are responsible for providing simplified component-level thermal models to LM in a format compatible with the thermal analyzer software being used for the spacecraft system thermal model. This will be specified in the ICD. Results of the integrated thermal analysis will be provided to the payload suppliers for refinement of the component level thermal analysis and design. The payload suppliers will be responsible for maintaining the more detailed interface thermal environments resulting from the integrated thermal analysis in their refined design.

3.4.2.1.4 Environment and System Testing

Lockheed Martin will perform system level thermal vacuum testing. The spacecraft will be tested in the landed configuration. Design thermal environments will be simulated with a combination of IR and solar lamps. For system level testing, flight acceptance test margins in accordance with modified MIL-STD-1540C will be added to the minimum and maximum flight allowable temperatures. All components will be constrained within their protoflight limits. Individual payload components will not be qualified during the system level test. Component level thermal qualification testing shall be conducted by the payload suppliers in accordance with modified MIL-STD-1540C prior to delivery. Modifications to MIL-STD-1540C will be specified in the ICD.

3.4.3 Electrical Interfaces

The spacecraft offers three types of electrical interfaces, digital, analog and power, which includes pyro circuits. All interfaces are redundant. The digital and analog interfaces are block redundant, meaning there is an A side and a B side. The active side has one "string" of interfaces and the other string is not available unless the spacecraft swaps to the other side. This is done in certain fault responses or by ground command. The power interfaces, including the pyro circuits, are cross strapped and accessible from either side. Nominally, both power switches are enabled. This is also true of latching switches that are to remain on while the lander is asleep. Payload design may be block redundant, in which case the two redundant power switch can be closed separately from the primary. Primary and secondary pyro circuits are always fired simultaneous to ensure the actuation takes place even in the event of a failure of one circuit.

Nominally, the payloads are already allocated the resources shown in table 3.3.1. If a resource is allocated to a payload, they may choose not to use it. They may also request a resource not allocated to them, though there is no guarantee it will be made available. Justification should be provided when requesting a resource not allocated to a payload (e.g. cost savings by using a build-to-print design that requires an additional interface).

3.4.3.1 Digital Interfaces

3.4.3.1.1 Asynchronous RS-422 Command (Payload Input)

The spacecraft provides a low-speed command interface for the payload. This is an RS-422 Asynchronous Serial interface consisting of an NRZL differential command signal. Commands transmitted on this interface are transmitted as Most Significant Byte first, Least Significant Bit first. Commands can be sent in multiples of 32 bits (4 bytes) or 8 bits (single byte). (Note: single byte mode results in overhead to pad out the single byte to 32-bit boundaries.) This interface transmits individual bytes with start, stop, and parity bits for each byte of command data. Parity can be configured as even or odd. When the spacecraft is asleep, this interface is inactive. There are five of these low speed command interfaces available to payloads. The sample acquisition system, water extraction system and camera system are each allocated one of these interfaces, leaving two to be distributed to payloads at the discretion of Mars One.

3.4.3.1.2 Asynchronous RS-422 Telemetry (Payload Output)

The spacecraft provides a low-speed telemetry interface for the payload. This is an RS-422 Asynchronous Serial interface consisting of an NRZL differential data signal. Telemetry transmitted on this interface are transmitted as Most Significant Byte first, Least Significant Bit first. Telemetry can be sent in multiples of 32 bits (4 bytes) or 8 bits (single byte). (Note: single byte mode results in overhead to pad out the single byte to 32-bit boundaries.) This interface requires the payload to transmit individual bytes with start, stop, and parity bits for each byte of telemetry data. Parity can be configured as even or odd. When the spacecraft is asleep, the payload must not attempt to communicate on this interface. Additionally, the payload most not attempt to communicate on the interface to the inactive side of the spacecraft. There are six of these low speed telemetry interfaces available to payloads. The sample acquisition system and water extraction system are each allocated one of these interfaces, leaving four to be distributed to payloads at the discretion of Mars One.

3.4.3.1.3 Synchronous RS-422 Data (Payload Output)

The spacecraft provides a high-speed telemetry interface for the payload. This is an RS-422 Synchronous Serial interface, consisting of a set of three differential signals, Clock, Data, and Frame (active low), with a maximum clock frequency of 6 MHz. The clock duty cycle is 50% +/- 10%. Data transmitted on this interface must be in multiples of 32 bits. Data transmitted on this interface is transmitted as Most Significant Bit first, Most Significant Byte first. When the spacecraft is asleep, the payload must not attempt to communicate on this interface. Additionally, the payload most not attempt to communicate on the interface to the inactive side of the spacecraft.

3.4.3.1.4 Side Select Discrete (Payload Input)

The spacecraft provides 5V CMOS discrete single-ended outputs to the payload specifically to indicate which side of the spacecraft is active. This is for payload use in determining which port to communicate on. When the spacecraft is asleep, the side select is not asserted.

3.4.3.1.5 Discrete Outputs (Payload Input)

The spacecraft provides 5V CMOS discrete single-ended outputs to the payload. This allows the flight software operating in the C&DH to control discretes to the payload. These outputs default to logic low (0V) upon spacecraft power-up. When the spacecraft is asleep (mostly powered off to conserve energy), these discretes are set to logic low (0V). Because the number of these discretes available is extremely limited, they will only be made available in exceptional cases. Justification should be provided as to why this resource should be allocated. Note it is possible that requiring this resource could result in not being selected.

3.4.3.1.6 Discrete Inputs (Payload Output)

The spacecraft provides TTL discrete inputs to accept single-ended discrete signals from the payload. This allows the payload to set discrete values for the spacecraft to read and react to. Because the number of these discretes available is extremely limited, they will only be made available in exceptional cases. Justification should be provided as to why this resource should be allocated. Note it is possible that requiring this resource could result in not being selected.

3.4.3.1.7 Differential Input (Payload Output)

In addition to the TTL discrete inputs, the spacecraft also provides LVDS differential inputs. This allows the payload to set discrete values for the spacecraft to read and react to. Because the number of these discretes available is extremely limited, they will only be made available in exceptional cases. Justification should be provided as to why this resource should be allocated. Note it is possible that requiring this resource could result in not being selected.

3.4.3.1.8 Non-Standard LVDS Command (Payload Input)

The spacecraft provides an LVDS synchronous serial command interface to the payload, consisting of three differential signals – Clock, Data, and Frame (active low), with a maximum clock frequency of 33 MHz. The clock duty cycle is 50% +/- 10%. The data transmitted on this interface is defined to be Most Significant Byte first, Most Significant Bit first. The heritage spacecraft avionics contain two custom LVDS command interfaces. Up to two payloads may choose to use these interfaces, but will have to design their electronics to the unique requirements of this interface.

3.4.3.1.9 Non-Standard LVDS Telemetry (Payload Output)

The spacecraft provides an LVDS synchronous serial telemetry interface for the payload consisting of differential clock, data, and data strobe (DSTROBE, active low) lines. Data transmitted on this interface is defined to be Most Significant Byte first, Most Significant Bit first. The interface is defined for DSTROBE widths of one (1) to five (5) clock cycles, with one (1) clock cycle width corresponding with 20 bits of reported data, the Least Significant Bit of the data synchronized to the rising edge of the DSTROBE signal, and the other valid DSTROBE widths corresponding with 12 bits of reported data, also with the data's LSB synchronized to the rising edge of the DSTROBE signal. When the spacecraft is asleep, the payload must not attempt to communicate on this interface. The heritage spacecraft avionics contain two custom LVDS telemetry interfaces. Up to two payloads may choose to use these interfaces, but will have to design their electronics to the unique requirements of this interface.

3.4.3.2 Analog Interfaces

3.4.3.2.1 Temperature Sensor – AD590 (Payload Output)

The spacecraft provides AD590 inputs from the payload, temperature transducers that produce output currents proportional to absolute temperature. Each AD590 input accepts a single-ended voltage range of 0VDC to +10VDC with a resolution of 2.44mV. Spacecraft excitation sources for these sensors are provided. Each payload is allocated up to two temperature sensors. They can be either AD-590 or PRT's. Note there are not enough to assign all payloads AD-590's or all payloads PRT's, so it may be necessary to reassign temperature sensors after payload selection.

3.4.3.2.2 Temperature Sensor – PRT (Payload Output)

The spacecraft provides PRT passive temperature sensor inputs from the payload. Each passive input channel accepts a single-ended voltage range of 0VDC to +68VDC. The passive input scaling factor is 2.44 ohm / count (1kohm / Volt). Each payload is allocated up to two temperature sensors. They can be either AD-590 or PRT's. Note there are not enough to assign all payloads AD-590's or all payloads PRT's, so it may be necessary to reassign temperature sensors after payload selection.

3.4.3.2.3 Analog Voltage (Payload Output)

The spacecraft provides voltage measurement inputs from the payload. Each voltage input channel accepts a single-ended voltage range of 0VDC to +9.5VDC with a resolution of 2.44mV. Nominally these are only available to the solar array demonstration experiment.

3.4.3.2.4 Analog Current (Payload Output)

The spacecraft provides current sense inputs from the payload. The interface provides signal conditioning for input currents between 0mA to 2.0mA. Two of these interfaces are available and are allocated to the solar array demonstration experiment.

3.4.3.3 Power Interfaces

3.4.3.3.1 Daytime Load Switches

The spacecraft provides 28V (22VDC-34VDC), 3A load switches to provide power to the payload. These load switches are powered off by the spacecraft every time the spacecraft goes to sleep to conserve power. If a payload must remain powered while the spacecraft is asleep, the payload must be powered with a latching switch (below).

3.4.3.3.2 Nighttime Latching Switches

The spacecraft provides 28V (22VDC-34VDC), 1A latching switches to provide power to the payload. These latching switches remain powered while the spacecraft is asleep but provide less current than the load switches (above). The latching switches will be powered off if the spacecraft power drops below a safe level.

3.4.3.3.3 Pyro Circuits

The spacecraft provides Pyro circuits for use by the payload to activate pyrotechnic devices that use NASA Standard Initiators or equivalent. These switches provide up to 5 amps at 28 V (22VDC-34VDC). These circuits have configurable drive times up to 255 ms. Each payload may have up to two pyro circuits. It is important to note that the avionics box that houses these circuits is outside the lander thermal enclosure and will freeze the first night on Mars. Therefore, any payload pyro circuit will be fired autonomously after the landed arrays are deployed, within an hour of landing. The circuits will not be available for firing pyros after that.

3.5 PLANETARY PROTECTION

3.5.1 All Payloads

A planetary protection classification will be requested once payloads are selected. For preliminary design purposes, a COSPAR planetary protection classification of IV-C should be assumed. All spacecraft components, including payloads, are required to be cleaned to an average bioburden on the exposed and internal surfaces prior to launch of less than 300 viable spores/m2. The contamination requirement is to be cleaned to Particulate Cleanliness Level 100 per IEST-STD-CC1246D. The method that will be used to achieve this is to wipe with a mixture of isopropyl alcohol and water. The requirement will be verified by assay.

3.5.2 Components that will Contact Liquid Water

One objective of the Mars One 2018 Precursor lander is to demonstrate the production of liquid water. Any component that will contact liquid water and components that contact those components are required to perform an additional four orders of magnitude of bioburden reduction. Dry Heat Microbial Reduction is one approved method. It assumes that the payload is already cleaned to the level of 300 viable spores/m2 described in section 3.5.1. Nominally this is 50 hours at 111 degrees C or 5 hours at 125 degrees C after wipes described in section 3.5.1, but this duration is configuration specific, and could be much longer.

3.6 LANDER ENVIRONMENTS

3.6.1 Pre-Launch

Payload providers are responsible for ensuring that flight hardware is not exposed to environments during ground handling that are more extreme than flight. If it is necessary to violate this requirement, the environmental verification must show verification of compatibility. This includes transportation and storage. Once at the Denver integration facility, the ground environments in Table 3.6-1 are applicable.

Parameter	Low Limit	High Limit
Temperature	+5 degrees C	+40 degrees C
Temperature Change Rate	-5 degrees C/hour	+5 degrees C/hour
Pressure	7 x 10^4 N/m^2 (10.1 psi)	1 x 10^5 N/m^2 (14.7 psi)
Relative Humidity	30%	<70%

Table 3.6-1. Prelaunch environments.

3.6.1.1 Launch Site Environments

The Mars One 2018 Precursor mission launch vehicle has not yet been selected. The current planning is that the launch vehicle will be a U.S. launch vehicle. Currently, the US launch vehicles that can launch the lander are an Atlas V 401, a Delta IV and Falcon 9. Launch site environments for each can be found at the following:

Section 3.1 of the Atlas V users guide at http://www.ulalaunch.com/uploads/docs/AtlasVUsersGuide2010.pdf

Section 3.1 of the Delta IV users guide at
http://www.ulalaunch.com/uploads/docs/Launch_Vehicles/Delta_IV_Users_Guide_June_2013.pdf

As of this writing, SpaceX did not appear to have a user's guide posted. Ground environments are discussed in section 5.2.1 and 5.2.2 in the 2009 version (Rev 1) archived at http://decadal.gsfc.nasa.gov/pace-201206md/Launch%20Vehicle%20Information/Falcon9UsersGuide.pdf

Any vehicles launch site may be used.

3.6.2 Launch

The baseline lander design is compatible with an Atlas V 401. The Mars One 2018 Precursor mission launch vehicle has not yet been selected. The current planning is that the launch vehicle will be a U.S. launch vehicle. Currently, the US launch vehicles that can launch the lander are an Atlas V 401, a Delta IV and Falcon 9. Launch environments for each can be found at the following:

Section 3.2 of the Atlas V users guide at http://www.ulalaunch.com/uploads/docs/AtlasVUsersGuide2010.pdf

Section 3.2 of the Delta IV users guide at
http://www.ulalaunch.com/uploads/docs/Launch_Vehicles/Delta_IV_Users_Guide_June_2013.pdf

As of this writing, SpaceX did not appear to have a user's guide posted. Ground environments are discussed in section 5.2.3 in the 2009 version (Rev 1) archived at http://decadal.gsfc.nasa.gov/pace-201206md/Launch%20Vehicle%20Information/Falcon9UsersGuide.pdf

3.6.2.1 Launch Acoustic Environments

Payloads must be designed to be compatible with the acoustic environment inside the aeroshell. Those are listed in Table 3.6-2.

1/3 Octave Band Center Frequency, Hz	Sound Pressure Levels, dB		
	Flight Acceptance	Protoflight	Test Tolerances
31.5	117.5	120.5	± 5 dB
40	120	123	± 5 dB
50	123	126	± 3 dB
63	124.5	127.5	± 3 dB
80	125.5	128.5	± 3 dB
100	124.5	127.5	± 3 dB
125	124.7	127.7	± 3 dB
160	125.5	128.5	± 3 dB
200	126	129	± 3 dB
250	125.7	128.7	± 3 dB
315	125	128	± 3 dB
400	120.7	123.7	± 3 dB

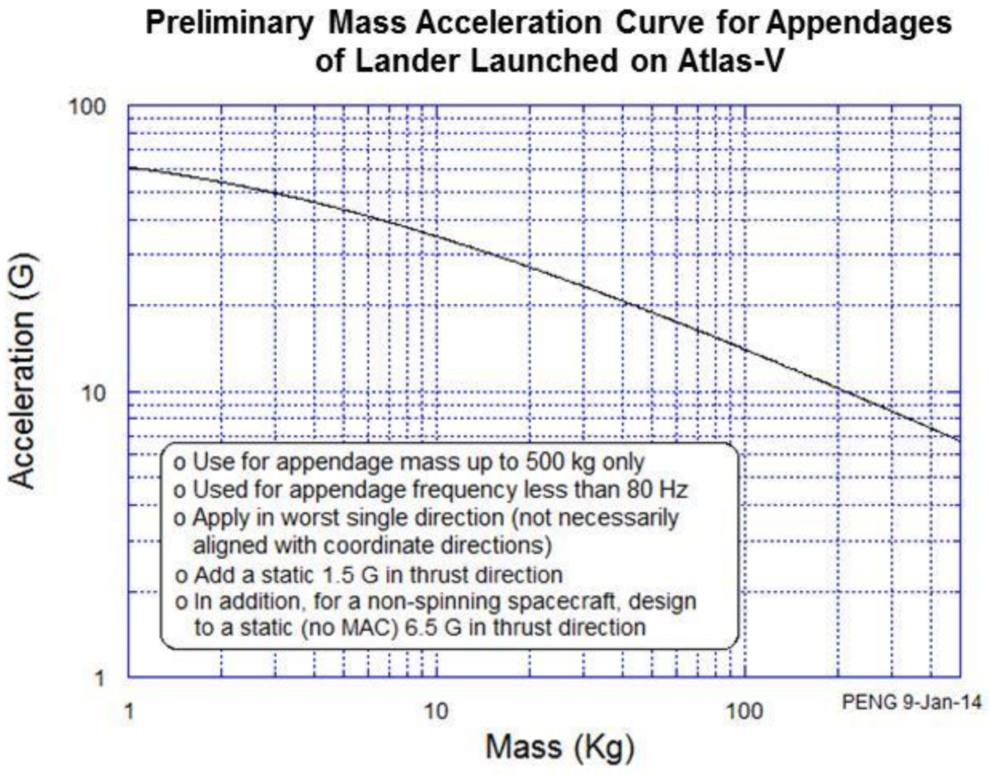
1/3 Octave Band Center Frequency, Hz	Sound Pressure Levels, dB		
	Flight Acceptance	Protoflight	Test Tolerances
500	114	117	± 3 dB
630	109	112	± 3 dB
800	104.5	107.5	± 3 dB
1000	102	105	± 3 dB
1,250	100	103	± 3 dB
1,600	98.3	101.3	± 3 dB
2,000	98.3	101.3	± 3 dB
2,500	97.2	100.2	± 3 dB
3,150	96.5	99.5	± 5 dB
4,000	96.7	99.7	± 5 dB
5,000	95.5	98.5	± 5 dB
6,300	95	98	± 5 dB
8,000	93.4	96.4	± 5 dB
10,000	91	94	± 5 dB
OASPL	135.0	138.0	
Duration	1 minute	1 minute	

Table 3.6-2. Acoustic environment inside the aeroshell.

3.6.2.2 Launch Limit Load Factors

Launch load factors for Atlas V are found in section 3.2.1 of the Atlas V user's guide. For the Delta IV they are found in section 3.2.4.1. In the 2009 version of the Falcon 9 Users guide they are described in section 5.2.3.1. The lander is currently designed for axial loads of +6.23/-1.8 g, while the lateral design load is +/-3.1 g. Payload designers should use the most conservative of these four sources.

The mass acceleration curve that should be used for preliminary design purposes is shown in Figure 3.6-1. Preliminary mass acceleration curve..



Preliminary Mass Acceleration Curve for Appendages of Lander Launched on Atlas-V

- o Use for appendage mass up to 500 kg only
- o Used for appendage frequency less than 80 Hz
- o Apply in worst single direction (not necessarily aligned with coordinate directions)
- o Add a static 1.5 G in thrust direction
- o In addition, for a non-spinning spacecraft, design to a static (no MAC) 6.5 G in thrust direction

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Figure 3.6-1. Preliminary mass acceleration curve.

3.6.2.3 Random Vibration

Random vibration for payloads mounted to the lander are shown in Table 3.6-3. Random vibration environmentsTable 3.6-3

Frequency **(Hz)**	Protoflight (MEFL +3dB) **(g^2/Hz)**	Acceptance (MEFL) **(g^2/Hz)**
20	0.020	0.010
80	0.080	0.040
500	0.080	0.040
2000	0.020	0.010
g_{rms}	**9.6**	**6.8**

Table 3.6-3. Random vibration environments

3.6.2.4 Pyro Shock

Pyro Shock loads for payloads mounted to the lander deck are shown in Table 3.6-4.

Frequency **(Hz)**	Protoflight (MEFL +3dB) **(g)**	Acceptance (MEFL) **(g)**
100	42	30
2000	2121	1500
10000	2121	1500

Table 3.6-4. Main deck pyro shock environments.

Components within 15 cm of lander-to-backshell separation brackets should design to the following pyro shock loads shown in Table 3.6-5.

Frequency **(Hz)**	Protoflight (MEFL +3dB) **(g)**	Acceptance (MEFL) **(g)**
100	42	30
3600	3100	2214
10000	3100	2214

Table 3.6-5. Separation bracket pyro shock environments.

Payload electronics that will be allowed inside the thermal enclosure (nominally only the sample acquisition system, the camera system and the water extraction system) should design those boxes to the pyro shock loads shown in Table 3.6-6

Frequency **(Hz)**	Protoflight (MEFL +3dB) **(g)**	Acceptance (MEFL) **(g)**
100	14	10
2000	700	500
10000	700	500

Table 3.6-6. Component deck pyro shock environments.

3.6.3 Cruise

3.6.3.1 Temperatures

Table 3.6-7 shows the minimum and maximum temperature payloads should expect to experience in cruise as a function of location.

Component Location	Minimum Launch/Cruise Predict	Maximum Launch/Cruise Predict
Science Deck (Top)	-45C	35C
Science Deck (Bottom, Camera Only)	-45C	35C
Payload Electronics on Component Deck	-20C	35C

Table 3.6-7. Payload cruise temperatures.

These are predicts and represent the least conservative allowable flight temperature (AFT) limits. Providers should assume that as the project matures, these values may be exceeded by an additional 5 degrees C and result in wider AFT limits. For flight acceptance, payloads should use an additional 5 degrees C of margin beyond conservative AFT limits. For protoflight/qualification limits, payload providers should use ten degrees C margin on the low limit and 15 degrees C margin on the high limit. Therefore, the protoflight/qualification limits should be 20 degrees colder than the minimums in Table 3.6-7 and 25 degrees warmer than the maximum in Table 3.6-7.

3.6.3.2 Ionizing Radiation

Ionizing radiation will be described in more thorough detail after selection. Payloads are protected by the aeroshell during cruise and the Martian atmosphere after landing. For design purposes, the total ionizing dose assuming 60 mil aluminum is under 2 krad.

3.6.4 Mars Aeroentry, Descent and Landing

3.6.4.1 Descent and Landing Loads

The driving static load from EDL is max deceleration during hypersonic entry. This load can be expected to be as high as -13 g. The maximum lateral static load occurs at touchdown and is +/- 2 g. The parachute inflation snatch load should be treated as quasi-static. This load can be as high as 66,750 N.

3.6.4.2 Thruster Pulsing

Payloads may experience vibration loads from thruster pulsing during the terminal descent. Vibration induced during powered descent is characterized by low-level sinusoids at the thruster pulse frequency of 10 Hz and at higher order harmonics. The duration of the powered descent phase is about 40 seconds. Payloads shall be designed to survive and function after (and if descent imaging, then during) exposure to the bounding sine levels shown in Table 3.6-8

Axes	Frequency (Hz)	Dwell Duration	Flight Acceptance	Protoflight/Qual
XYZ	10	60 seconds	0.7 g	1.0 g
	20,30,40,50,60	60 seconds	0.35 g	0.5 g
	70,80,90,100	60 seconds	0.1 g	0.14 g

Table 3.6-8. Terminal descent vibration environment.

3.6.4.3 Dynamic Pressure After Heatshield Jettison

Dynamic pressure during EDL after heatshield jettison and after backshell separation should be assumed to be as high as 110 Pa in any direction.

3.6.4.4 Martian Dust and Debris During Touchdown

During the touchdown event, dust and debris may be energized by the landing engines in such a way as to impact externally mounted hardware. This environment is expected but is not quantifiable. Externally mounted hardware should take whatever design precautions are needed to survive this environment.

In addition to being impacted by dust and debris during the landing event itself, it is possible that the dust cloud generated by touchdown can remain over the lander for some period of time after landing. That dust may settle out of the atmosphere onto instruments mounted on the deck. Again, any precautions needed to survive this environment should be designed into the instruments.

3.6.4.5 Landing Site Slopes and Tilt

The surface of Mars should not be expected to be perfectly flat and normal to zenith. The lander can tolerate landing slopes of at least 15 degrees, more if touchdown dynamics are favorable. Furthermore, the legs are designed to crush depending on how much touchdown load is applied to each. Therefore, payloads should expect up to a 15 degree angle between the deck normal and the gravity vector. Furthermore, the surface normal and the deck normal may differ by as much as 6.7 degrees. Payloads must be able to deploy (if applicable) and function in such tilts. The final touchdown geometry is expected to affect the amount of surface area accessible for sample acquisition.

3.6.5 Mars Surface Operations

3.6.5.1 Atmosphere Composition

Payloads must be designed to function in the Martian atmosphere. Based on Viking Lander measurements, the mole fractions of gases in the Martian atmosphere are approximately 95.5% $CO2$, 2.7% $N2$, 1.6% Ar, and 0.2% $O2$, CO, and other trace gases.

3.6.5.2 Surface Pressure

The minimum surface atmospheric pressure over the mission should be greater than or equal to 699 Pa and the maximum pressure is expected to be less than 892 Pa.

3.6.5.3 Surface Winds

Winds on Mars can vary greatly. For preliminary design purposes, payloads should be designed to tolerate a sustained wind speed of 10 m/s for cold cases. Zero wind should be used for hot cases

3.6.5.4 Surface Temperatures

Figure 3.6-2. Air and ground temperatures for cold case.Figure 3.6-2 shows the diurnal temperatures for surface and 1-m air temperatures for the coldest expected sol of the mission (nominally the first sol). The data table can be found in Appendix B.

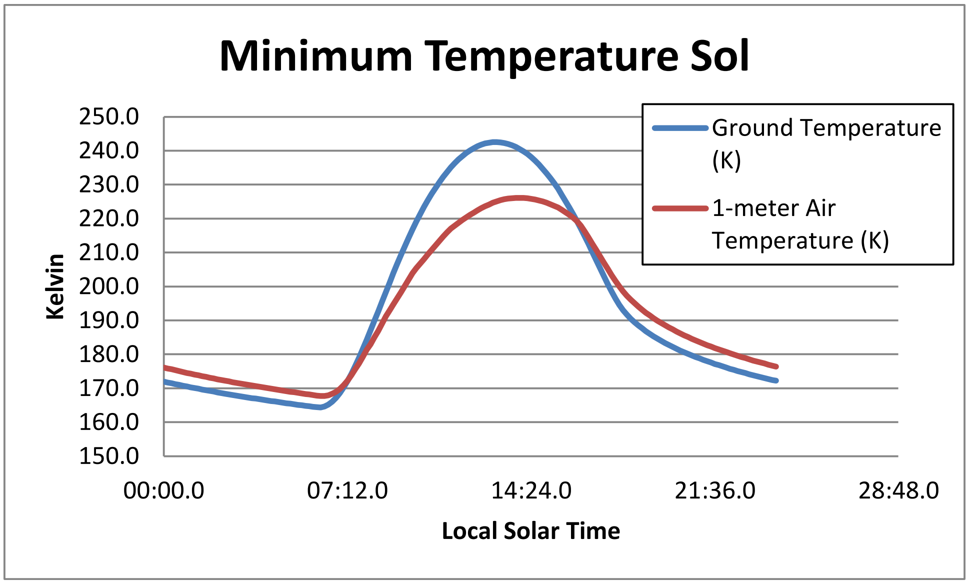

Figure 3.6-2. Air and ground temperatures for cold case.

Figure 3.6-3 shows the diurnal temperatures for the surface and 1-m air temperatures for the hottest expected sol of the mission (nominally sol 246). The data table can be found in Appendix B.

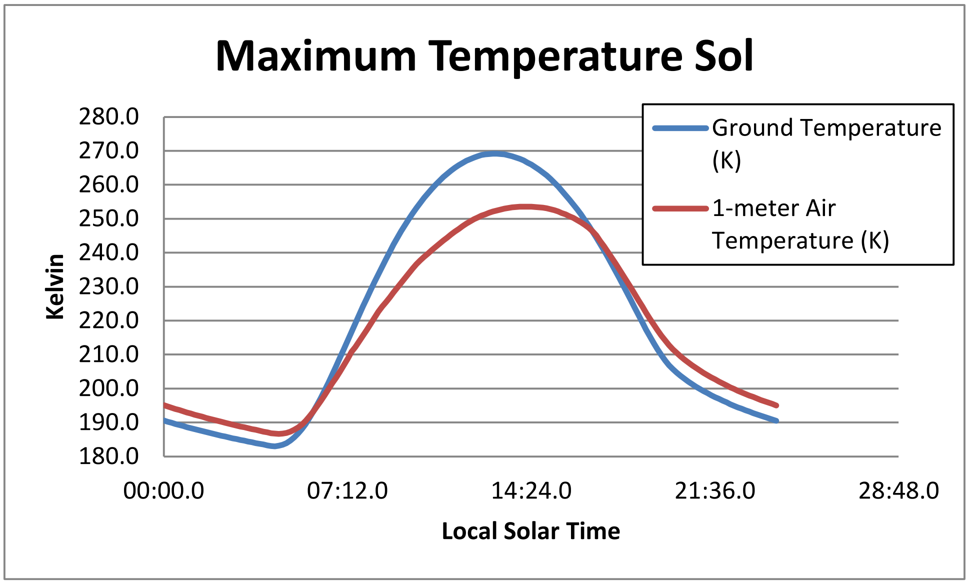

Figure 3.6-3. Air and ground temperatures for hot case.

Table 3.6-9 shows the expected minimum and maximum temperatures for payload components based on these thermal environments. Please note instrument surface finish will have some influence in determining actual temperature extremes.

Component Location	Minimum Landed Temperature	Maximum Landed Temperature
Science Deck (Top)	-115C	65C
Science Deck (Bottom, Camera Only)	-115C	45C
Payload Electronics on Component Deck	-35C	40C

Table 3.6-9. Payload surface temperatures.

These are predicts and represent the least conservative allowable flight temperature (AFT) limits. Providers should assume that as the project matures, an additional 5 degrees C of margin may be required on these values. For flight acceptance, payloads should use an additional 5 degrees C of margin beyond conservative AFT limits. For protoflight/qualification limits, payload providers should use ten degrees C margin on the low limit and 15 degrees C margin on the high limit. Therefore, the protoflight/qualification limits should be 20 degrees colder than the minimums in Table 3.6-9 and 25 degrees warmer than the maximum in Table 3.6-9.

3.6.5.5 Ionizing Radiation

Ionizing radiation described in section 3.6.3.2 describes radiation for the entire mission.

3.6.5.6 Dust and Soil Deposition

Mars is subject to local and global dust storms. Over time, this dust settles out of the atmosphere and will land on the lander and its payloads. This will cause a change in thermal properties. The dust coverage factor at the end of the mission will be 0.42. The maximum dust coverage factor for a vertical surface is 0.1.

For thermal properties, payloads should assume no change in IR surface emissivity due to dust. To derive solar absorptivity as a function of Dust Coverage Factor, absorptivity(now)=DCF*0.7+(1-DCF)*absorptivity(no dust).

The visible optical depth will impact thermal properties as well as the performance of the solar array demonstration experiment. Based on previous lander missions, the optical depth has varied from as low as 0.18 to as high as 2.34 over the lifetime of the Mars One 2018 mission duration.

3.6.5.7 Landing Site Soil Properties

The sample acquisition system must be designed to retrieve samples over the entire range of Mars surface properties, whatever the design (arm, dril, etc.). The candidate landing sites are in regions where ice is expected within the top 0.5 meters of the soil. Every attempt will be made to select landing sites that minimize rock coverage. While it is not possible to specify the soil properties that will be encountered at the landing site, the following references discuss the soil Phoenix encountered digging into ice on Mars:

- Shaw, Amy, "Characterization of Martian Surfaces using Mechanical and Spectrophotometric Models" (2012), Electronic Theses and Dissertations, Paper 977.
- Arvidson, R. E., et al. (2009), Results from the Mars Phoenix Lander Robotic Arm experiment, J. Geophys. Res., 114,E00E02, doi:10.1029/2009JE003408.
- Cull, S.,R. E. Arvidson, M. T. Mellon, P. Skemer, A. Shaw, and R. V. Morris (2010), Compositions of subsurface ices at the Mars Phoenix landing site, Geophys. Res. Lett., 37, L24203, doi:10.1029/2010GL045372.
- Bonitz, R. G., et al. (2008), NASA Mars 2007 Phoenix Lander Robotic Arm and Icy Soil Acquisition Device, J. Geophys.Res., 113, E00A01, doi:10.1029/2007JE003030.

The ability to retrieve icy samples and deliver them to the water extraction experiment will be critical to the success of the mission. Payloads should specify their capabilities and the verification of those capabilities as they relate to retrieving samples that contain water ice.

3.7 PAYLOAD INTEGRATION

This section describes the payload integration process. All delivery dates are contained in Table 5.2-1.

A payload accommodation team will be formed to negotiate, define and document the payload-to-spacecraft interface control documents (ICDs) during Phase B. Payload ICDs will be written by LM to define all interfaces (e.g. mechanical, electrical, configuration, environments, software, facility support) with the lander. Any unusual payload needs such as thermal, electrical, mechanical, contamination control, purge gases, etc., must be identified early and resolved in the ICD. The payload interface is standardized with respect to electrical specifications (power, data, EMI, connectors, etc.).

During development, thermal and structural models are traded between LM and payload providers to verify thermal environment compatibility and harness accommodations.

LM designs, fabricates, and operates spacecraft emulators to perform early interface verification testing of bit-byte ordering, basic low-level protocols, and communication passing between payload component and spacecraft.

Payload providers must deliver fit-check templates to verify payload-to-spacecraft physical compatibility (form and fit).

LM develops software simulation models of the payload interface using inputs from the payloads. These models are integrated into a software environment as well as a hardware test lab and are fundamental to verifying spacecraft to instrument interface compatibility. LM is responsible for writing all interface software resident on the spacecraft processor using inputs provided by the payloads. This software development is a highly interactive process that relies on definitions agreed to in the ICDs.

High-fidelity hardware payload EDU models are delivered by the payload providers to LM for testbed incorporation to perform software tests, verify command and data interfaces, data flow, and fault protection. ATLO procedures are dry run through the testbeds to uncover issues and decrease the probability of problems in ATLO. The EDU models will be used through ATLO and operations in the testbed for validating sequences and troubleshooting.

Payloads deliver tested and fully qualified flight units for spacecraft integration. Instruments are delivered to the S/C as fully integrated packages by June 15, 2017. After delivery, instruments undergo acceptance tests consisting of inspection, cleaning and microbiological assays to determine if bioburden levels are met and stand-alone testing to verify performance. All instrument-unique test equipment, purge equipment and sensor simulators as required, to facilitate stimulation and response verification of payload equipment and the spacecraft (end-to-end mission function) will be the responsibility of the payload supplier.

Facility support at Lockheed Martin required for standalone and integrated operations shall be specified in the ICD. Subsequent to payload delivery, the payload supplier will be responsible for all aspects of the standalone payload verification prior to spacecraft installation.

Following flight mounting, pin-out verification and flight connector mating, the library of payload software commands is sent to verify proper operation and telemetry feedback. Once each payload package passes form, fit, and function testing, the spacecraft is cleared for systems testing. These system tests include acoustic, thermal vacuum, EMI/EMC and pyro shock. To provide assurance for mission success, payload functions with the spacecraft will be verified. Instrument representatives must support these activities at Lockheed Martin and at the launch site either in person, for US produced payloads or via the Instrument Accommodation Engineer, who has been trained and assigned the responsibility to represent the instrument team for foreign produced payloads. There will be mandatory inspections of instruments and spacecraft configurations planned throughout the ATLO flow. Instrument teams are an integral part of ATLO and are expected to verify data following each test they are involved in and for providing any technical support personnel deemed necessary.

Trouble-shooting will be accomplished under documented control, including failure investigation, verification, and correction validation testing.

After initial installation on the lander and system testing there are no plans for subsequent removal of payload equipment prior to launch.

At the launch site, final mission sequence testing will be conducted. The payload will be required to provide resources to support this testing. Final cleaning and close-outs will be performed at the launch site (launch payload processing facility) except where access prevents launch site cleaning. In this case, final cleaning and close-outs will be performed at Lockheed Martin. The payload supplier shall provide any and all equipment required to support payload integration and launch site operations. The baseline plan is aeroshell installation is conducted in Denver prior to shipment to the launch site. There are currently no plans to access the science deck at the launch site, so payload closeout will occur prior to ship.

4 GROUND DATA SYSTEM/MISSION OPERATIONS SYSTEM

The Ground Data System consists of the hardware, software, facilities and networks used on Earth to operate the spacecraft, both during test on Earth and in flight and on Mars. The Mission Operations System includes the operations teams, processes and procedures for operating the spacecraft.

At the time of this writing, Mars One does not yet have agreements in place with NASA and/or ESA as to exactly what networks will be used for data return, so some requirements and constraints imposed by the ground data system are not yet available. As soon as these agreements are in place, these requirement and constraints will be provided to the selected payload providers.

Mars surface operations will be conducted at Lockheed Martin's Waterton facility in Colorado. The extent of instrument provider participation will depend on the instrument selected, but providers should expect to be requested to support integration, ground testing, and surface operations in Denver, with schedules to be negotiated depending on the activity of the payload. Sample Acquisition, Water Extraction, and Camera System Providers may be required to participate in surface operations for many months. Other payloads providers'' participation in operations will vary depending on the payload. To the extent possible, remote participation will be used. Prior to surface operations, operations testing and rehearsals will take place, both prior to launch and after launch.

4.1 GROUND DATA SYSTEM (GDS)

4.1.1 Hardware

Prior to delivery, each instrument provider is responsible for all hardware necessary to develop and test their instrument, with the exception that Lockheed Martin will provide spacecraft emulators to test the data interfaces between the instrument and the spacecraft. For providers that are required to participate in surface operations in Denver, workstations on an instrument LAN will be provided. For providers that will participate remotely, they must provide workstations for such remote participation.

4.1.2 Software

Prior to delivery, each instrument provider is responsible for all software necessary to develop and test their instrument, with the exception that Lockheed Martin will provide spacecraft emulators to test data interfaces between the instrument and the spacecraft. For providers that are required to participate in surface operations in Denver, workstations on an instrument LAN will be provided that have all the necessary software for operations installed on them. For providers that will participate remotely, they must provide workstations for such remote participation. If any project software is required to be operated on these workstations, any requirements on those workstations will be agreed to in the ICD.

As part of the software requirements development process, instrument providers will be required to participate in the development of operations planning software for their instrument. This software will be used both in ground test and as well as for surface operations to plan activities with the instrument. Providers should expect to provide any instrument unique software including but not limited to:

- Data compression/decompression requirements and/or algorithms
- Extraction of data needed for tactical planning
- Extraction of health and safety data for review
- Any specialized analysis software needed for tactical planning

4.1.3 Facilities

Instrument providers are responsible for providing their own facilities at their home institution. During integration and testing on Earth, all facilities are provided at Lockheed Martin's Waterton Colorado facility. For those providers required to participate in Denver for Mars surface operations, all facilities will be provided. Instrument providers operating remotely will be expected to provide their own facilities.

4.2 MISSION OPERATIONS

4.2.1 Pre-Launch Activities

Prior to launch, the instrument providers will be involved in the development of nominal and mission critical sequences which involve the lander and the lander instrument suite (such as the first-day landed activities), that need to be tested in ATLO. The building of spacecraft commands and blocks require provider participation whenever payload commands are included. Post-launch sequences necessary for payload health checks and in-flight calibrations will be approved prior to launch and generated post-launch. The instrument providers are participants in end-to-end ground system tests.

Mission planning, in particular landing site selection, begins prior to launch, in order to firmly plan for necessary spacecraft capability and mission design. Contingency planning (for example selection of backup landing sites) also begins at that time. Instrument providers will support development of contingency procedures for their instrument

4.2.2 Cruise Activities

Nominally no payload activities are permitted during cruise. Flight safe versions of ATLO checkout sequences may be used as post launch and/or prelanding checkouts to verify instrument aliveness. Any other cruise activities need to be proposed and justified. There is no guarantee that a requested cruise activity will be allowed.

If operational testing reveals a need for updates to any payload sequences or software, those may be uploaded prior to landing depending on the update.

4.2.3 Entry, Descent and Landing Activities

Except for the Camera System, all payloads will be powered off during EDL. If descent imagery is proposed, only power and commanding will be made available during EDL. The Camera System will not be allowed to utilize spacecraft data services during EDL.

4.2.4 Surface Activities

4.2.4.1 Strategic Planning

Instrument providers selected to support daily activities on Mars will participate in the strategic planning process. This process will define, on a weekly basis, which instrument activities will be conducted on which sol, and roughly how long they will take. This planning is required to support development of the background sequences.

4.2.4.2 Short Term Planning

Each sol, the strategic plan will be updated based on the previous sol's results and any new information. This may result in changes being fed into the tactical planning. Instrument providers selected to support daily activities on Mars will participate in this process.

4.2.4.3 Tactical Planning

Upon landing, the spacecraft will execute a sequence to deconfigure from flight and configure for surface operations. Once all flight system commands have been executed, limited sol 0 imaging will be conducted and all payload pyros will be fired. Once the data from this activity is confirmed on the ground, the mission operations team will begin planning the sol 1 activities. For the most part, these will have already been developed, but must be reviewed in light of the spacecraft state after landing.

The daily tactical process always begins with the downlink of data at the end of the Martian day. That data is reviewed to ensure all activities successfully executed and to identify any anomalies. Then that day's detailed sample acquisition, delivery, imaging and water extraction activities are planned. Depending on the instrument, the instrument provider may be required to support this tactical process onsite in Colorado. The day's activities are integrated, tested, altered if necessary to fit within resources, and finally uplinked to Mars. The process is repeated each Martian sol

The tactical process will begin on Mars time. The operations team will operate on a schedule that changes with the daylight hours on Mars and with the availability of data return from Mars. Initially, this process will be 7 days per week. Depending on progress, operations may later shift to an Earth time based schedule, and possibly to 5 days per week.

5 PAYLOAD MANAGEMENT/DELIVERABLES

5.1 INSTRUMENT PROVIDER RESPONSIBILITIES

The instrument provider is responsible for instrument design and development, fabrication, test, and calibration, and delivery of flight hardware, software requirements, and associated support equipment, within project schedule and payload resources. The instrument provider is also required to support planning and instrument operation development.

A Lockheed Martin payload integration engineer will represent the payload in Denver as a participant in the integrated product teams (IPTs) and to negotiate interfaces with the instrument. The specific responsibilities of the instrument provider include but are not limited to:

1. Develop an internal management plan.
2. Ensure that the design and fabrication of the instrument and any deployment/mobility devices (if applicable), development, and testing are appropriate to the objectives Mars One and meet the environmental and interface constraints.
3. Manage payload margin to ensure successful hardware integration and implementation of the experiment.
4. Be responsible for quality assurance and reliability, and for parts and materials selection.
5. Ensure that instrument development meets the approved schedules and cost plans.
6. Be the primary point of contact with the Project for the purpose of establishing requirements, ICDs, schedules and transfer of funds.
7. Ensure that the instrument(s) is properly calibrated.
8. Conduct payload reviews as required by section 5.2.1.
9. Participate in Software Working Group (SWG) meetings, as required by the proposed science mission use of spacecraft computational resources and services to resolve requirements and interface issues, and resolve resource allocations and operational timelines.
10. Support payload integration and system test procedure development and maintenance. Support instrument and GSE integration, and lander system testing at Lockheed Martin and the launch site. Remote support of tests is in general acceptable; however, on-site field engineering support is required for mandatory inspections and the following tests: initial power turn on (IPTO), functional electrical test (FET), first spacecraft functional tests at LM and the launch site, Thermal Vacuum Test, compatibility/EMI, and on-pad power-on/final payload close-out.
11. Support definition of mission database contents, including but not limited to: flight rules and constraints, sequences, payload telemetry, and commands.
12. Support integrated mission data/sequence development and flight software integration, using the Spacecraft Test Laboratory (STL).
13. Support planning and possibly executing mission operations, including end to end test support.
14. Support preparation as requested by Mars One any data for release.

5.2 DELIVERABLES

As described in the following sections, during Phase B, C & D, meeting both schedule and cost, the instrument providers shall:

1. Shortly after selection, sign a memorandum of agreement (MOA) or contract (as applicable) with the project, documenting resource allocations.
2. Provide and maintain required documentation (see Section 5.2.4).
3. Provide inputs and support development and maintenance of ICDs.
4. Participate in monthly management reviews (MMR’s).
5. Deliver flight hardware (including thermal blankets if required) to the LM which meets planetary protection requirements, with suitable shipping containers and any covers required. Also, deliver connectors to the LM for the harness side of the interface.
6. Deliver a fit check template (transfer tool), a single node analytical thermal model in accordance with section 3.4.2., and a payload interface simulator to the LM. Mass and/or thermal simulators will only be necessary in the event of a late delivery of the flight instrument.
7. Provide necessary instrument-unique ground support equipment (GSE) for standalone, integration, calibration and launch operations.

8. Provide an instrument end item data package (EIDP), as described in section 5.2.4.5.
9. Deliver the necessary software requirements and any other information or source code (see Section 5.2.3).
10. Provide timely information to establish and maintain controlled baselines for software interfaces, shared computational resources, mission data, and mission operations timelines and sequences.

Table 5.2-1 contains a detailed list of deliverables and associated dates.

DELIVERABLE	DESCRIPTION & I/O CAPABILITY	DUE DATE
Preliminary Interface Control Document Inputs	LM develops interface control documents that specify all requirements the payloads place on the spacecraft and all requirements that the spacecraft places on the payloads. This requires inputs from the instrument provider.	6/5/15
Preliminary Structural (CAD) & Thermal Models	Preliminary delivery of CAD models and thermal models in formats to be negotiated in the ICD.	7/17/15
Flight Software and Simulation Software Preliminary Requirements	Preliminary delivery of documentation of all requirements necessary for LM to write the spacecraft hosted flight software. It also includes command and telemetry definitions. This delivery also includes the necessary inputs for LM to write simulation software to allow software only testing.	7/17/15
Final Interface Control Document Inputs	Final ICD inputs to produce a final, signed ICD.	12/18/15
Final Structural (CAD) & Thermal Models	Final delivery of CAD models and thermal models in formats to be negotiated in the ICD.	12/18/15
Final Flight Software and Simulation Software Requirements	Final delivery of documentation of all requirements necessary for LM to write the spacecraft hosted flight software. It also includes command and telemetry definitions. This delivery also includes the necessary inputs for LM to write simulation software to allow software only testing.	12/18/15
Interface Circuit Diagram	An electrical schematic showing all circuits that will interface with spacecraft electrical circuits.	9/18/15
Fit Check Templates	Templates delivered to LM to perform early mechanical interface validation.	4/15/16
Operations Manual	This document contains operating instructions (how to run the instrument), flight rules, constraints, sequences, and any other information necessary to operate the payload.	6/15/17
Payload Integration and Test Procedure Inputs	LM will develop procedures to integrate and test the instruments. The instrument developers will need to work with the LM engineers and provide inputs to the development of those procedures.	12/1/16
Harness Connectors	LM will fabricate the spacecraft harness. The instrument providers must supply the spacecraft half of the connectors with connector savers in time for test lab and flight harness fabrication.	7/1/16
High Fidelity Engineering Develop Unit	This is a non-flight unit that is fit, form and function identical to the flight hardware. It is provided to check out integration and test procedures in the test labs prior to flight instrument integration with the spacecraft. These EDU's will also be used during operations development as well as Mars surface operations to validate flight products and payload interoperability.	3/1/17
Safety Certification and Plan	Instrument design specific document to be negotiated that describes safe handling and certifies safety of the delivered hardware.	6/15/17
Planetary Protection & Contamination Certification	Report including description of all planetary protection cleanings and any dry heat microbial reduction processes used if applicable.	6/15/17
Instrument Ground Support Equipment	All electrical and mechanical ground support equipment required to support instrument integration with the spacecraft, including any targets.	6/15/17
Flight Instrument and Thermal Blankets	The instrument that will fly to Mars on the lander	6/15/17
Interface Control Document Verification Report	Verification report for the ICD requirements assigned to the instrument for verification action.	6/15/17
End Item Data Package	"As-built" hardware documentation including final drawing, parts lists, final mass properties and center of gravity data. A complete manufacturing log. Total operating hours, number of mechanical cycles and remaining cycle life, and any other limited life data. A complete list of all tests performed and a compilation of test data and results for each test. An environmental test report verifying survival of all environmental requirements, including EMI testing.	6/15/17

Table 5.2-1 Payload deliverables and due dates.

5.2.1 Reviews

The payload providers (or their designate) will be expected to attend spacecraft design reviews, ground system reviews, and occasional informal reviews scheduled by integrated product teams (IPTs) with instrument issues/presentations to be made by the instrument representative.

The payload provider will conduct the instrument Preliminary Interface Requirements and Design Review (PDR). The PDR is scheduled as early as possible after the completion of the Functional Requirements Document (FRD), and the Preliminary Interface Control Document (ICD). Topics include: discussion of the FRD and a description of interfaces.

Likewise, the payload provider will conduct the instrument Manufacturing Readiness Review (MRR). The MRR follows the spacecraft CDR, at the completion of the payload detailed design and the final ICD. Topics include: status of hardware design, fabrication, test, and calibration, software design and test plans, plans for integration, description of support equipment, finalization of interfaces, command and telemetry requirements, and discussion of environmental and system tests.

Lastly, the payload provider will conduct an Instrument Delivery Review (IDR). This review is held just prior to instrument delivery to Lockheed Martin, and topics include how well the instrument complies with the Functional Requirements and the ICD, the results of environmental testing, and the completeness of the EIDP.

In addition to the milestone reviews above, monthly management reviews will be held.

5.2.2 Hardware Delivery

The instrument must be accompanied by all ground support equipment (GSE) needed to support system test. An EIDP shall accompany the flight hardware.

5.2.3 Software

Lockheed Martin will write all instrument flight software hosted on the lander computer. Instrument providers are responsible for any software resident within the payload itself. The software section of the ICD will contain all requirements necessary for LM to produce the fight software and a simulation software model that represents the instrument in testbeds where an instrument EM is not available. This section will also contain limitations on resources (such as memory and throughput) that the instrument must meet. Finally, instructions for operational sequences must also be supplied to LM.

The payload provider must provide either source code or detailed instructions necessary for any special payload code needs, such as compression or decompression algorithms, or other software capabilities needed to operate the instrument.

The software development flow is as follows:

- Initial flight and simulation software requirements are agreed to in Phase B.
- LM begins development of flight and simulation software.
- Final flight and simulation software requirements are signed prior to the final design review.
- LM provides spacecraft emulator for payloads to validate electrical data interfaces.
- Flight software build 2.0 with payload capability is tested using flight software and simulated instruments in software only environment.
- Flight software build 2.0 with payload software capability is tested using instrument EDU's in the STL.
- Flight software build 2.0 is installed on the spacecraft and instruments are integrated and tested.

5.2.4 Documentation

5.2.4.1 Contract/MOA's

Shortly after payload selection, the project will enter into an agreement with each provider for the implementation of their selected proposal. Each agreement will document the payload resource allocation (mass, power, volume and fiscal resources) and schedule between the Mars One and the payload provider. The agreement will take the form of a contract for non-government entities, and a memorandum of agreement (MOA) for government entities. If the Instrument provider is not within the US, a Technical Assistance Agreement (TAA) will need to be entered into.

5.2.4.2 FRD/Safety

The payload provider will be responsible for writing a functional requirements document (FRD) and supplying the necessary payload safety information to the lander contractor for the range safety plan and the payload safety reviews at the launch site.

5.2.4.3 ICD's

ICDs are negotiated directly with the lander engineering team in an integrated product team (IPT) environment. ICDs identify all payload interfaces, including but not limited to, the instrument envelope, mounting, mass, center of mass, electrical and mechanical connections, end circuits, power, pyro devices, features requiring access or clearance, purge requirements, environmental requirements, software requirements, testing, facility support, view angles, and clearances, thermal control, red and green tag lists, GSE interfaces/requirements, etc.

5.2.4.4 Payload Handling Requirements and Unit History Logbook

A payload handling requirements list will describe any special handling necessary to ensure the safety of the flight hardware. The unit history logbook will accompany the delivery of the flight hardware.

5.2.4.5 End-Item-Data-Package

The EIDP includes (but is not limited to) final drawings, documents, mass properties, qualification data, footprint drawings, final power, verification report, final parts and materials as built lists, planetary protection measures, and high resolution color photographs of the assembled instrument (with scale inserted).

6 APPENDIX A: ACRONYMS

AFT	Allowable Flight Temperatures
AIAA	American Institute of Aeronautics and Astronautics
ATLO	Assembly, Test and Launch Operations
C&DH	Command & Data Handling
COSPAR	Committee on Space Research
CMOS	Complimentary Metal Oxide Semiconductor
CAD	Computer Aided Design
CDR	Critical Design Review
CBE	Current Best Estimate
DCF	Dust Correction Factor
EMI	Electromagnetic Interference
EMC	Electromagnetic Compatibility
EIDP	End Item Data Package
EDU	Engineering Development Unit
EM	Engineering Model
EDL	Entry, Descent and Landing
ESA	European Space Agency
FET	Functional Electrical Test
GDS	Ground Data System
GSE	Ground Support Equipment
IR	Infrared
IPTO	Initial Power Turn On
IDR	Instrument Delivery Review
IPT	Integrated Product Team
ICD	Interface Control Document
JPL	Jet Propulsion Laboratory
LAN	Local Area Network
LM	Lockheed Martin
LVDS	Low Voltage Differential Signal
MRR	Manufacturing Readiness Review
MOLA	Mars Orbiter Laser Altimeter
MAC	Mass Acceleration Curve
MEFL	Maximum Expected Flight Level
MOA	Memorandum of Agreement
MMR	Monthly Management Review
NASA	National Aeronautics and Space Administration
PRT	Platinum Resistance Thermometer
PDR	Preliminary Design Review
PIP	Proposal Information Package
RFP	Request For Proposals
SWG	Software Working Group
S/C	Spacecraft
STL	Spacecraft Test Laboratory
TAA	Technical Assistance Agreement
TTL	Transistor-Transistor Logic
US	United States

7 APPENDIX B: SURFACE ENVIRONMENTS

7.1 MINIMUM TEMPERATURE SOL 0

Sol 0, Tau=0.52, Ls=355.91, Atmospheric Pressure at Surface=837.6 Pa

Mars Local True Solar Time	Ground Temperature (K)	1-meter Air Temperature (K)	Direct Solar Flux (W/m^2)	Diffuse Solar Flux (W/m^2)	IR Flux (W/m^2)	Sun Elevation Angle (deg)
00:00.0	172.0	176.1	0.0	0.0	20.3	0.0
00:09.0	171.7	175.8	0.0	0.0	20.2	0.0
00:18.0	171.5	175.6	0.0	0.0	20.1	0.0
00:27.0	171.2	175.3	0.0	0.0	20.1	0.0
00:36.0	171.0	175.0	0.0	0.0	20.0	0.0
00:45.0	170.7	174.8	0.0	0.0	19.9	0.0
00:54.0	170.5	174.5	0.0	0.0	19.8	0.0
01:03.0	170.3	174.3	0.0	0.0	19.7	0.0
01:12.0	170.1	174.0	0.0	0.0	19.7	0.0
01:21.0	169.8	173.8	0.0	0.0	19.6	0.0
01:30.0	169.6	173.6	0.0	0.0	19.5	0.0
01:39.0	169.4	173.3	0.0	0.0	19.4	0.0
01:48.0	169.2	173.1	0.0	0.0	19.3	0.0
01:57.0	169.0	172.9	0.0	0.0	19.3	0.0
02:06.0	168.8	172.7	0.0	0.0	19.2	0.0
02:15.0	168.6	172.4	0.0	0.0	19.1	0.0
02:24.0	168.4	172.2	0.0	0.0	19.1	0.0
02:33.0	168.2	172.0	0.0	0.0	19.0	0.0
02:42.0	168.0	171.8	0.0	0.0	18.9	0.0
02:51.0	167.8	171.6	0.0	0.0	18.9	0.0
03:00.0	167.6	171.4	0.0	0.0	18.8	0.0
03:09.0	167.5	171.2	0.0	0.0	18.7	0.0
03:18.0	167.3	171.0	0.0	0.0	18.6	0.0
03:27.0	167.1	170.8	0.0	0.0	18.6	0.0
03:36.0	166.9	170.6	0.0	0.0	18.5	0.0
03:45.0	166.7	170.4	0.0	0.0	18.5	0.0
03:54.0	166.6	170.2	0.0	0.0	18.4	0.0
04:03.0	166.4	170.0	0.0	0.0	18.3	0.0
04:12.0	166.2	169.9	0.0	0.0	18.3	0.0
04:21.0	166.1	169.7	0.0	0.0	18.2	0.0
04:30.0	165.9	169.5	0.0	0.0	18.1	0.0
04:39.0	165.8	169.3	0.0	0.0	18.1	0.0
04:48.0	165.6	169.2	0.0	0.0	18.0	0.0
04:57.0	165.4	169.0	0.0	0.0	18.0	0.0
05:06.0	165.3	168.8	0.0	0.0	17.9	0.0
05:15.0	165.1	168.6	0.0	0.0	17.8	0.0
05:24.0	165.0	168.5	0.0	0.0	17.8	0.0
05:33.0	164.8	168.3	0.0	0.0	17.7	0.0
05:42.0	164.7	168.1	0.0	0.0	17.7	0.0
05:51.0	164.5	168.0	0.0	0.0	17.6	0.0
06:00.0	164.4	167.8	0.0	0.0	17.6	0.0
06:09.0	164.3	167.7	0.0	1.4	17.5	2.3
06:18.0	164.6	167.7	0.0	6.3	17.4	2.3
06:27.0	165.2	167.9	0.0	11.9	17.4	3.5
06:36.0	166.0	168.3	0.2	19.5	17.4	5.1
06:45.0	167.2	169.0	0.8	27.8	17.4	6.7
06:54.0	168.6	169.8	2.2	35.9	17.4	8.3

07:03.0	170.2	170.9	4.6	43.8	17.4	9.8
07:12.0	172.1	172.2	8.0	51.8	17.4	11.4
07:21.0	174.2	173.8	12.4	59.8	17.5	12.9
07:30.0	176.6	175.6	17.6	67.1	17.6	14.4
07:39.0	179.0	177.3	23.5	73.5	17.7	15.9
07:48.0	181.6	179.3	30.0	79.8	17.8	17.4
07:57.0	184.3	181.4	37.0	85.7	17.9	18.9
08:06.0	187.1	183.0	44.4	90.7	18.1	20.4
08:15.0	189.9	185.0	52.1	95.6	18.2	21.8
08:24.0	192.7	186.9	59.9	100.5	18.4	23.2
08:33.0	195.5	189.1	67.9	105.2	18.7	24.6
08:42.0	198.4	191.3	75.9	108.9	18.9	26.0
08:51.0	201.2	193.0	83.8	112.3	19.2	27.3
09:00.0	203.9	194.7	91.7	115.5	19.4	28.6
09:09.0	206.6	196.5	99.5	118.7	19.7	29.9
09:18.0	209.3	198.3	107.1	121.7	20.0	31.1
09:27.0	211.8	200.1	114.5	124.7	20.3	32.3
09:36.0	214.4	202.0	121.7	127.5	20.6	33.4
09:45.0	216.8	203.9	128.5	130.1	20.9	34.5
09:54.0	219.1	205.4	135.1	131.8	21.3	35.5
10:03.0	221.3	206.7	141.3	133.4	21.6	36.5
10:12.0	223.4	208.1	147.1	135.0	21.9	37.5
10:21.0	225.4	209.4	152.6	136.4	22.2	38.4
10:30.0	227.3	210.8	157.6	137.7	22.5	39.2
10:39.0	229.0	212.1	162.2	138.9	22.8	39.9
10:48.0	230.7	213.4	166.4	140.0	23.2	40.6
10:57.0	232.3	214.7	170.1	141.0	23.5	41.2
11:06.0	233.7	216.0	173.3	141.9	23.9	41.8
11:15.0	235.1	217.0	176.1	142.6	24.3	42.2
11:24.0	236.3	217.9	178.3	143.2	24.6	42.6
11:33.0	237.4	218.8	180.1	143.6	24.9	42.9
11:42.0	238.4	219.6	181.4	144.0	25.2	43.1
11:51.0	239.3	220.4	182.1	144.2	25.5	43.2
12:00.0	240.1	221.1	182.4	144.2	25.8	43.3
12:09.0	240.8	221.8	182.1	144.2	26.0	43.2
12:18.0	241.3	222.4	181.4	144.0	26.3	43.1
12:27.0	241.8	223.1	180.1	143.6	26.6	42.9
12:36.0	242.1	223.6	178.3	143.2	26.9	42.6
12:45.0	242.4	224.2	176.0	142.6	27.2	42.2
12:54.0	242.5	224.6	173.3	141.8	27.4	41.7
13:03.0	242.5	225.1	170.0	141.0	27.7	41.2
13:12.0	242.5	225.4	166.3	140.0	27.9	40.6
13:21.0	242.3	225.7	162.1	138.9	28.1	39.9
13:30.0	242.0	225.9	157.5	137.7	28.3	39.2
13:39.0	241.6	226.0	152.5	136.4	28.5	38.3
13:48.0	241.1	226.1	147.0	134.9	28.7	37.5
13:57.0	240.5	226.1	141.2	133.4	28.8	36.5
14:06.0	239.8	226.1	135.0	131.7	29.0	35.5
14:15.0	239.0	226.0	128.4	130.0	29.1	34.5
14:24.0	238.1	225.9	121.6	127.5	29.2	33.4
14:33.0	237.1	225.6	114.4	124.6	29.3	32.2
14:42.0	235.9	225.4	107.0	121.7	29.4	31.1
14:51.0	234.7	225.1	99.4	118.6	29.4	29.8
15:00.0	233.3	224.7	91.6	115.5	29.5	28.6
15:09.0	231.9	224.3	83.7	112.2	29.5	27.3

15:18.0	230.4	223.8	75.7	108.9	29.5	25.9
15:27.0	228.8	223.2	67.7	105.1	29.5	24.6
15:36.0	227.0	222.6	59.8	100.4	29.5	23.2
15:45.0	225.2	221.9	52.0	95.5	29.5	21.8
15:54.0	223.3	221.1	44.3	90.6	29.5	20.3
16:03.0	221.3	220.3	36.9	85.6	29.4	18.9
16:12.0	219.2	219.2	29.9	79.7	29.3	17.4
16:21.0	217.0	217.9	23.4	73.4	29.2	15.9
16:30.0	214.8	216.2	17.5	67.0	29.1	14.4
16:39.0	212.5	214.5	12.3	59.7	28.9	12.9
16:48.0	210.1	212.6	8.0	51.7	28.6	11.4
16:57.0	207.7	210.7	4.6	43.7	28.4	9.8
17:06.0	205.4	208.9	2.2	35.8	28.1	8.2
17:15.0	203.0	207.0	0.8	27.7	27.8	6.7
17:24.0	200.7	205.2	0.1	19.4	27.5	5.1
17:33.0	198.5	203.4	0.0	11.9	27.2	3.5
17:42.0	196.5	201.7	0.0	6.2	26.9	2.3
17:51.0	194.6	200.0	0.0	1.3	26.6	2.3
18:00.0	193.0	198.6	0.0	0.0	26.3	0.0
18:09.0	191.7	197.3	0.0	0.0	26.0	0.0
18:18.0	190.5	196.1	0.0	0.0	25.7	0.0
18:27.0	189.5	195.1	0.0	0.0	25.5	0.0
18:36.0	188.5	194.1	0.0	0.0	25.3	0.0
18:45.0	187.6	193.1	0.0	0.0	25.0	0.0
18:54.0	186.8	192.3	0.0	0.0	24.8	0.0
19:03.0	186.0	191.5	0.0	0.0	24.6	0.0
19:12.0	185.3	190.7	0.0	0.0	24.4	0.0
19:21.0	184.6	190.0	0.0	0.0	24.2	0.0
19:30.0	184.0	189.2	0.0	0.0	24.1	0.0
19:39.0	183.3	188.6	0.0	0.0	23.9	0.0
19:48.0	182.7	187.9	0.0	0.0	23.7	0.0
19:57.0	182.2	187.3	0.0	0.0	23.6	0.0
20:06.0	181.6	186.7	0.0	0.0	23.4	0.0
20:15.0	181.1	186.2	0.0	0.0	23.3	0.0
20:24.0	180.6	185.6	0.0	0.0	23.1	0.0
20:33.0	180.1	185.1	0.0	0.0	23.0	0.0
20:42.0	179.7	184.6	0.0	0.0	22.8	0.0
20:51.0	179.2	184.1	0.0	0.0	22.7	0.0
21:00.0	178.8	183.6	0.0	0.0	22.6	0.0
21:09.0	178.4	183.2	0.0	0.0	22.4	0.0
21:18.0	177.9	182.7	0.0	0.0	22.3	0.0
21:27.0	177.5	182.3	0.0	0.0	22.2	0.0
21:36.0	177.2	181.9	0.0	0.0	22.1	0.0
21:45.0	176.8	181.4	0.0	0.0	22.0	0.0
21:54.0	176.4	181.0	0.0	0.0	21.8	0.0
22:03.0	176.1	180.7	0.0	0.0	21.7	0.0
22:12.0	175.7	180.3	0.0	0.0	21.6	0.0
22:21.0	175.4	179.9	0.0	0.0	21.5	0.0
22:30.0	175.1	179.6	0.0	0.0	21.4	0.0
22:39.0	174.8	179.2	0.0	0.0	21.3	0.0
22:48.0	174.4	178.9	0.0	0.0	21.2	0.0
22:57.0	174.1	178.5	0.0	0.0	21.1	0.0
23:06.0	173.8	178.2	0.0	0.0	21.0	0.0
23:15.0	173.6	177.9	0.0	0.0	20.9	0.0
23:24.0	173.3	177.6	0.0	0.0	20.8	0.0

23:33.0	173.0	177.3	0.0	0.0	20.7	0.0
23:42.0	172.7	177.0	0.0	0.0	20.6	0.0
23:51.0	172.5	176.7	0.0	0.0	20.5	0.0
24:00.0	172.2	176.4	0.0	0.0	19.6	0.0

7.2 MAXIMUM TEMPERATURE SOL 246

Sol 246, Tau=0.53, Ls=110.65, Atmospheric Pressure at Surface=760.6 Pa						
Mars Local True Solar Time	Ground Temperature (K)	1-meter Air Temperature (K)	Direct Solar Flux (W/m^2)	Diffuse Solar Flux (W/m^2)	IR Flux (W/m^2)	Sun Elevation Angle (deg)
00:00.0	190.6	195.1	0.0	0.0	32.8	0.0
00:09.0	190.2	194.7	0.0	0.0	32.6	0.0
00:18.0	189.9	194.4	0.0	0.0	32.4	0.0
00:27.0	189.6	194.0	0.0	0.0	32.2	0.0
00:36.0	189.3	193.6	0.0	0.0	32.0	0.0
00:45.0	188.9	193.3	0.0	0.0	31.9	0.0
00:54.0	188.6	193.0	0.0	0.0	31.7	0.0
01:03.0	188.3	192.6	0.0	0.0	31.5	0.0
01:12.0	188.0	192.3	0.0	0.0	31.4	0.0
01:21.0	187.8	192.0	0.0	0.0	31.2	0.0
01:30.0	187.5	191.7	0.0	0.0	31.0	0.0
01:39.0	187.2	191.4	0.0	0.0	30.9	0.0
01:48.0	186.9	191.1	0.0	0.0	30.7	0.0
01:57.0	186.6	190.8	0.0	0.0	30.6	0.0
02:06.0	186.4	190.5	0.0	0.0	30.4	0.0
02:15.0	186.1	190.2	0.0	0.0	30.2	0.0
02:24.0	185.9	189.9	0.0	0.0	30.1	0.0
02:33.0	185.6	189.6	0.0	0.0	29.9	0.0
02:42.0	185.4	189.4	0.0	0.0	29.8	0.0
02:51.0	185.1	189.1	0.0	0.0	29.7	0.0
03:00.0	184.9	188.8	0.0	0.0	29.5	0.0
03:09.0	184.6	188.6	0.0	0.0	29.4	0.0
03:18.0	184.4	188.3	0.0	0.0	29.2	0.0
03:27.0	184.2	188.1	0.0	0.0	29.1	0.0
03:36.0	183.9	187.8	0.0	0.0	29.0	0.0
03:45.0	183.7	187.6	0.0	0.0	28.8	0.0
03:54.0	183.5	187.3	0.0	0.0	28.7	0.0
04:03.0	183.3	187.1	0.0	0.0	28.6	0.0
04:12.0	183.1	186.9	0.0	0.0	28.4	0.0
04:21.0	183.0	186.7	0.0	1.8	28.3	2.3
04:30.0	183.1	186.6	0.0	5.6	28.2	2.3
04:39.0	183.4	186.7	0.0	9.8	28.1	3.3
04:48.0	183.9	186.9	0.1	15.4	28.0	4.7
04:57.0	184.7	187.3	0.4	22.2	27.9	6.1
05:06.0	185.6	187.8	1.2	28.8	27.9	7.5
05:15.0	186.7	188.5	2.7	35.5	27.8	9.0
05:24.0	188.0	189.4	4.9	42.2	27.8	10.4
05:33.0	189.5	190.4	8.1	49.2	27.8	11.9
05:42.0	191.2	191.7	12.0	56.3	27.8	13.4
05:51.0	193.1	193.1	16.7	62.5	27.9	14.9
06:00.0	195.1	194.7	22.2	68.4	28.0	16.4
06:09.0	197.2	196.3	28.2	74.3	28.1	17.9
06:18.0	199.4	198.0	34.9	79.6	28.2	19.4
06:27.0	201.7	199.8	42.0	84.5	28.4	21.0
06:36.0	204.1	201.5	49.5	89.4	28.6	22.6

06:45.0	206.6	203.2	57.4	94.3	28.8	24.1
06:54.0	209.0	205.0	65.6	98.6	29.0	25.7
07:03.0	211.5	206.8	74.1	102.3	29.2	27.3
07:12.0	214.0	208.8	82.7	106.0	29.5	28.8
07:21.0	216.6	210.8	91.5	109.6	29.8	30.4
07:30.0	219.1	212.2	100.3	113.3	30.2	32.0
07:39.0	221.7	213.8	109.3	117.0	30.4	33.6
07:48.0	224.2	215.5	118.2	119.9	30.7	35.2
07:57.0	226.6	217.3	127.1	122.3	31.1	36.8
08:06.0	229.0	219.0	136.0	124.8	31.5	38.4
08:15.0	231.3	220.8	144.7	127.1	31.9	40.0
08:24.0	233.7	222.7	153.4	129.5	32.3	41.5
08:33.0	235.9	224.1	161.8	131.8	32.8	43.1
08:42.0	238.1	225.4	170.2	134.1	33.2	44.7
08:51.0	240.3	226.8	178.3	136.3	33.5	46.2
09:00.0	242.4	228.3	186.2	138.5	33.9	47.8
09:09.0	244.4	229.7	193.8	140.3	34.4	49.3
09:18.0	246.3	231.1	201.2	141.6	34.8	50.8
09:27.0	248.1	232.5	208.2	142.8	35.3	52.3
09:36.0	249.9	233.9	215.0	144.0	35.8	53.8
09:45.0	251.6	235.3	221.4	145.1	36.3	55.2
09:54.0	253.2	236.6	227.5	146.2	36.8	56.6
10:03.0	254.7	237.7	233.3	147.2	37.3	58.0
10:12.0	256.2	238.8	238.7	148.1	37.7	59.3
10:21.0	257.5	239.8	243.7	149.0	38.2	60.6
10:30.0	258.8	240.8	248.3	149.9	38.6	61.8
10:39.0	260.1	241.7	252.4	150.6	39.0	62.9
10:48.0	261.2	242.7	256.2	151.3	39.5	64.0
10:57.0	262.3	243.6	259.6	151.9	39.9	65.0
11:06.0	263.3	244.5	262.5	152.4	40.4	65.9
11:15.0	264.2	245.3	265.0	152.9	40.8	66.6
11:24.0	265.0	246.2	267.0	153.3	41.3	67.3
11:33.0	265.8	247.0	268.6	153.6	41.7	67.8
11:42.0	266.5	247.8	269.7	153.8	42.2	68.2
11:51.0	267.1	248.5	270.4	153.9	42.6	68.4
12:00.0	267.6	249.2	270.6	153.9	43.1	68.5
12:09.0	268.1	249.8	270.4	153.9	43.5	68.4
12:18.0	268.4	250.4	269.7	153.7	43.9	68.2
12:27.0	268.7	250.9	268.6	153.5	44.3	67.8
12:36.0	269.0	251.3	267.0	153.2	44.6	67.3
12:45.0	269.1	251.7	264.9	152.9	45.0	66.6
12:54.0	269.2	252.1	262.4	152.4	45.3	65.8
13:03.0	269.2	252.4	259.5	151.9	45.6	65.0
13:12.0	269.1	252.7	256.2	151.3	45.9	64.0
13:21.0	269.0	253.0	252.4	150.6	46.2	62.9
13:30.0	268.7	253.2	248.2	149.8	46.5	61.8
13:39.0	268.4	253.3	243.6	149.0	46.7	60.5
13:48.0	268.1	253.5	238.6	148.1	47.0	59.3
13:57.0	267.6	253.5	233.2	147.1	47.2	58.0
14:06.0	267.1	253.6	227.5	146.1	47.5	56.6
14:15.0	266.6	253.6	221.4	145.0	47.7	55.2
14:24.0	265.9	253.6	214.9	143.9	47.9	53.7
14:33.0	265.2	253.5	208.1	142.7	48.0	52.3
14:42.0	264.5	253.4	201.1	141.5	48.2	50.8
14:51.0	263.6	253.2	193.7	140.2	48.4	49.3

15:00.0	262.7	253.0	186.1	138.4	48.5	47.8
15:09.0	261.7	252.8	178.2	136.2	48.6	46.2
15:18.0	260.6	252.5	170.1	134.0	48.7	44.7
15:27.0	259.5	252.1	161.7	131.7	48.8	43.1
15:36.0	258.2	251.7	153.2	129.4	48.8	41.5
15:45.0	257.0	251.3	144.6	127.0	48.8	39.9
15:54.0	255.6	250.8	135.8	124.7	48.8	38.4
16:03.0	254.2	250.2	127.0	122.3	48.8	36.8
16:12.0	252.8	249.6	118.1	119.8	48.8	35.2
16:21.0	251.3	249.0	109.1	116.9	48.7	33.6
16:30.0	249.7	248.2	100.2	113.2	48.7	32.0
16:39.0	248.0	247.3	91.4	109.5	48.6	30.4
16:48.0	246.2	246.3	82.6	105.9	48.4	28.8
16:57.0	244.4	245.1	74.0	102.2	48.3	27.2
17:06.0	242.6	243.4	65.5	98.5	48.1	25.7
17:15.0	240.7	241.9	57.3	94.2	47.8	24.1
17:24.0	238.7	240.2	49.4	89.3	47.4	22.5
17:33.0	236.7	238.6	41.9	84.4	47.1	21.0
17:42.0	234.6	236.8	34.8	79.5	46.7	19.4
17:51.0	232.5	235.1	28.1	74.2	46.3	17.9
18:00.0	230.3	233.3	22.1	68.3	45.9	16.4
18:09.0	228.1	231.5	16.7	62.4	45.4	14.9
18:18.0	226.0	229.6	12.0	56.1	45.0	13.3
18:27.0	223.8	227.8	8.0	49.1	44.5	11.9
18:36.0	221.6	225.9	4.9	42.1	44.1	10.4
18:45.0	219.4	224.1	2.6	35.4	43.6	8.9
18:54.0	217.4	222.3	1.2	28.7	43.1	7.5
19:03.0	215.4	220.6	0.4	22.1	42.6	6.1
19:12.0	213.4	218.8	0.1	15.3	42.1	4.7
19:21.0	211.6	217.2	0.0	9.8	41.7	3.3
19:30.0	209.9	215.6	0.0	5.6	41.2	2.3
19:39.0	208.3	214.1	0.0	1.7	40.7	2.3
19:48.0	206.9	212.8	0.0	0.0	40.3	0.0
19:57.0	205.7	211.6	0.0	0.0	39.9	0.0
20:06.0	204.7	210.5	0.0	0.0	39.5	0.0
20:15.0	203.7	209.5	0.0	0.0	39.1	0.0
20:24.0	202.9	208.5	0.0	0.0	38.7	0.0
20:33.0	202.1	207.7	0.0	0.0	38.4	0.0
20:42.0	201.3	206.8	0.0	0.0	38.0	0.0
20:51.0	200.6	206.1	0.0	0.0	37.7	0.0
21:00.0	199.9	205.3	0.0	0.0	37.4	0.0
21:09.0	199.2	204.6	0.0	0.0	37.1	0.0
21:18.0	198.6	203.9	0.0	0.0	36.8	0.0
21:27.0	198.0	203.3	0.0	0.0	36.5	0.0
21:36.0	197.5	202.7	0.0	0.0	36.2	0.0
21:45.0	196.9	202.1	0.0	0.0	36.0	0.0
21:54.0	196.4	201.5	0.0	0.0	35.7	0.0
22:03.0	195.9	200.9	0.0	0.0	35.4	0.0
22:12.0	195.4	200.4	0.0	0.0	35.2	0.0
22:21.0	194.9	199.9	0.0	0.0	35.0	0.0
22:30.0	194.5	199.4	0.0	0.0	34.7	0.0
22:39.0	194.0	198.9	0.0	0.0	34.5	0.0
22:48.0	193.6	198.4	0.0	0.0	34.3	0.0
22:57.0	193.2	197.9	0.0	0.0	34.1	0.0
23:06.0	192.8	197.5	0.0	0.0	33.8	0.0

23:15.0	192.4	197.1	0.0	0.0	33.6	0.0
23:24.0	192.0	196.6	0.0	0.0	33.4	0.0
23:33.0	191.6	196.2	0.0	0.0	33.2	0.0
23:42.0	191.2	195.8	0.0	0.0	33.0	0.0
23:51.0	190.9	195.4	0.0	0.0	32.8	0.0
24:00.0	190.5	195.0	0.0	0.0	32.6	0.0