Mars Surveyor Program

Announcement of Opportunity

2001 Lander Mission

Proposal Information Package

Final

30 June 1997

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FIGURES

TABLES

1.0 Introduction

This Proposal Information Package (PIP) document is being supplied with the Announcement of Opportunity for the Mars Surveyor 2001 lander and orbiter payloads. There is a separate PIP for the Mars Surveyor 2001 orbiter. This PIP applicable only for the selection of the lander instruments and the rover instrument suite.

1.1 Purpose and Scope

The Mars Surveyor Project will launch two spacecraft to Mars in the 2001 launch opportunity - a lander and an orbiter. Figure 1.1.a presents a timeline of the 2001 Mars Surveyor Program orbiter and lander missions including launch/arrival dates for the opening of a 20 day launch period, trajectory properties, and the relationship to the Mars 1998 orbiter and 1996 Mars Global Surveyor.

This document is based on preliminary designs but supplies payload proposers the technical detail that is required to propose a viable payload for the 2001 lander and rover. The payload should meet the technical targets and requirements described herein for the mission, lander the rover designs.

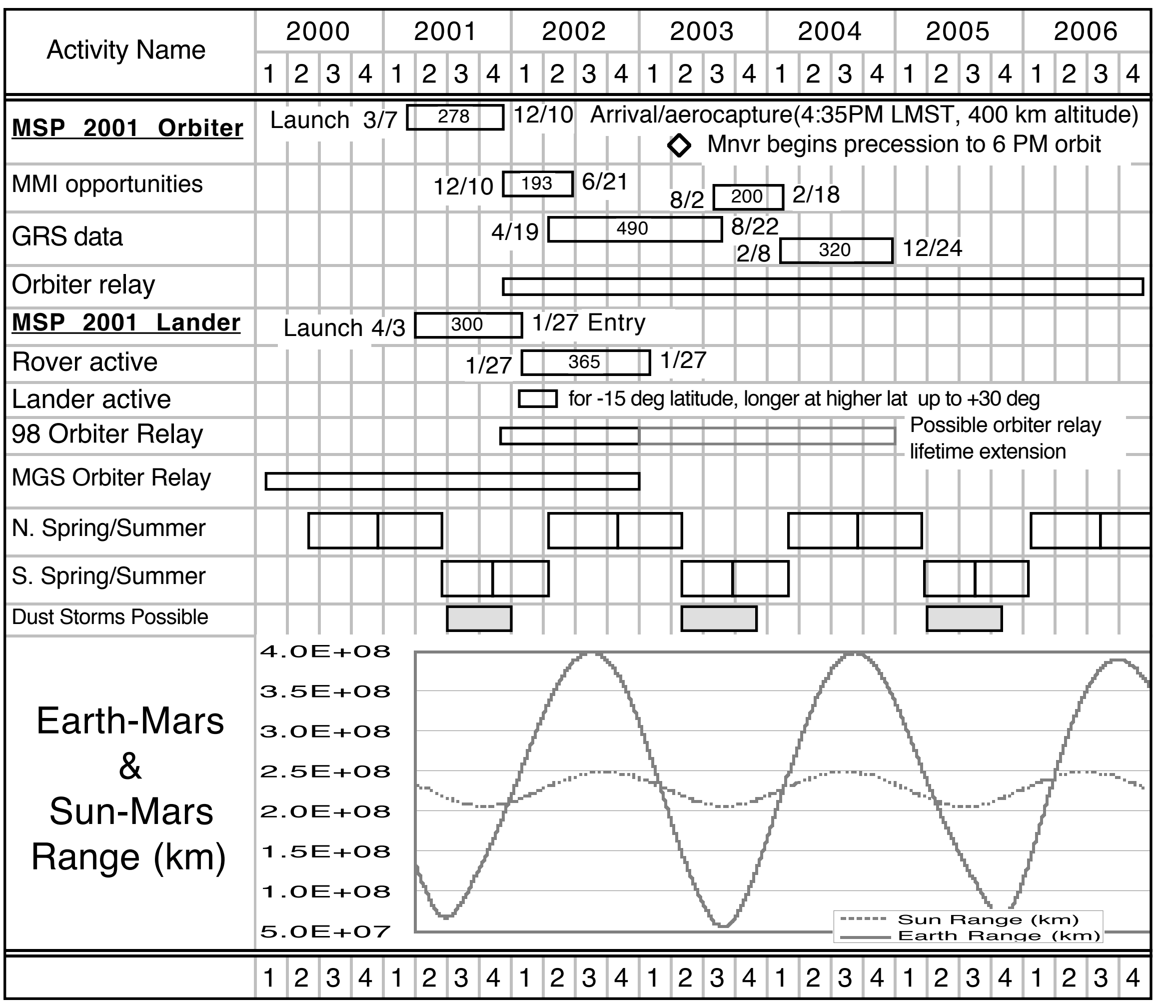


Figure 1.1.a MSP 2001 Mission Timeline

The Mars Surveyor lander and rover development requires a cost-capped, design-to-cost effort to enable Mars exploration at a minimal cost. The current lander and rover designs provide significant payload capabilities within the current budget. This design will evolve during the Phase B detailed definition phase (October, 1997 through the end of February, 1998).

Approximately 2.5 years are available from the authority to proceed with instrument development (11/1/97) to delivery of flight hardware to the spacecraft contractor.

1.2 Document Structure

Section 2 provides a general overview of the mission and spacecraft.

Section 3 provides a description of the payload resources and constraints imposed by the mission and spacecraft design. Section 3 also outlines areas in which science objectives may drive mission design, particularly in the choice of the landing site. Proposers should identify candidate landing sites at which the science objectives could be accomplished. The actual landing site will be selected by the Project and the science team and certified by NASA.

Section 4 describes the Ground Data System (GDS) and mission operations activities. The 2001 lander and orbiter use a common GDS.

Section 5 describes the payload management responsibilities and deliverables for the lander instruments.

Section 6 describes the payload management responsibilities and deliverables for the rover instrument suite.

2.0 General Description of the Lander Mission and Spacecraft

2.1 Mission

An overview of the mission is shown in figure 2.1.a.

The launch period for the 2001 lander launch opportunity opens April 5, 2001 and extends 20 days to April 24, 2001. The earliest Mars arrival is January 16, 2002 when the spacecraft directly enters the Mars atmosphere from the arrival hyperbola. The latest arrival is February 5, 2002 with a similar approach methodology. Assuming nominal launches at the beginning of the 20 day period for both the lander and the orbiter, the lander will arrive at Mars about 37 days after the orbiter has been captured into its mapping orbit. The lander will transmit data to Earth relayed through the 2001 orbiter. The season of arrival for the lander is late southern summer with an Ls=310 assuming a launch on the first day of the opportunity. If the launch occurs on the last day of the 20 day period, arrival will happen somewhat later with an Ls=321.

Figure 2.1.a Spacecraft Activities by Mission Phase

2.1.1 Launch

The lander will be launched on a McDonnell Douglas Delta II 7425. The launch phase begins at launch and ends at lander separation from the launch vehicle.

2.1.2 Cruise

The cruise phase begins after launch vehicle separation. After initial spacecraft attitude and communications are established, the flight system will deploy into cruise configuration and perform initial subsystems checkouts. The cruise phase lasts approximately 9 months. The lander is enclosed inside the aeroshell during cruise.

The first trajectory correction maneuver (TCM) takes place at Launch + 8 days. There are four TCMs planned altogether.

All communications during cruise is based upon usage of the low gain antenna (LGA) / medium gain antenna (MGA) mounted on the cruise stage.

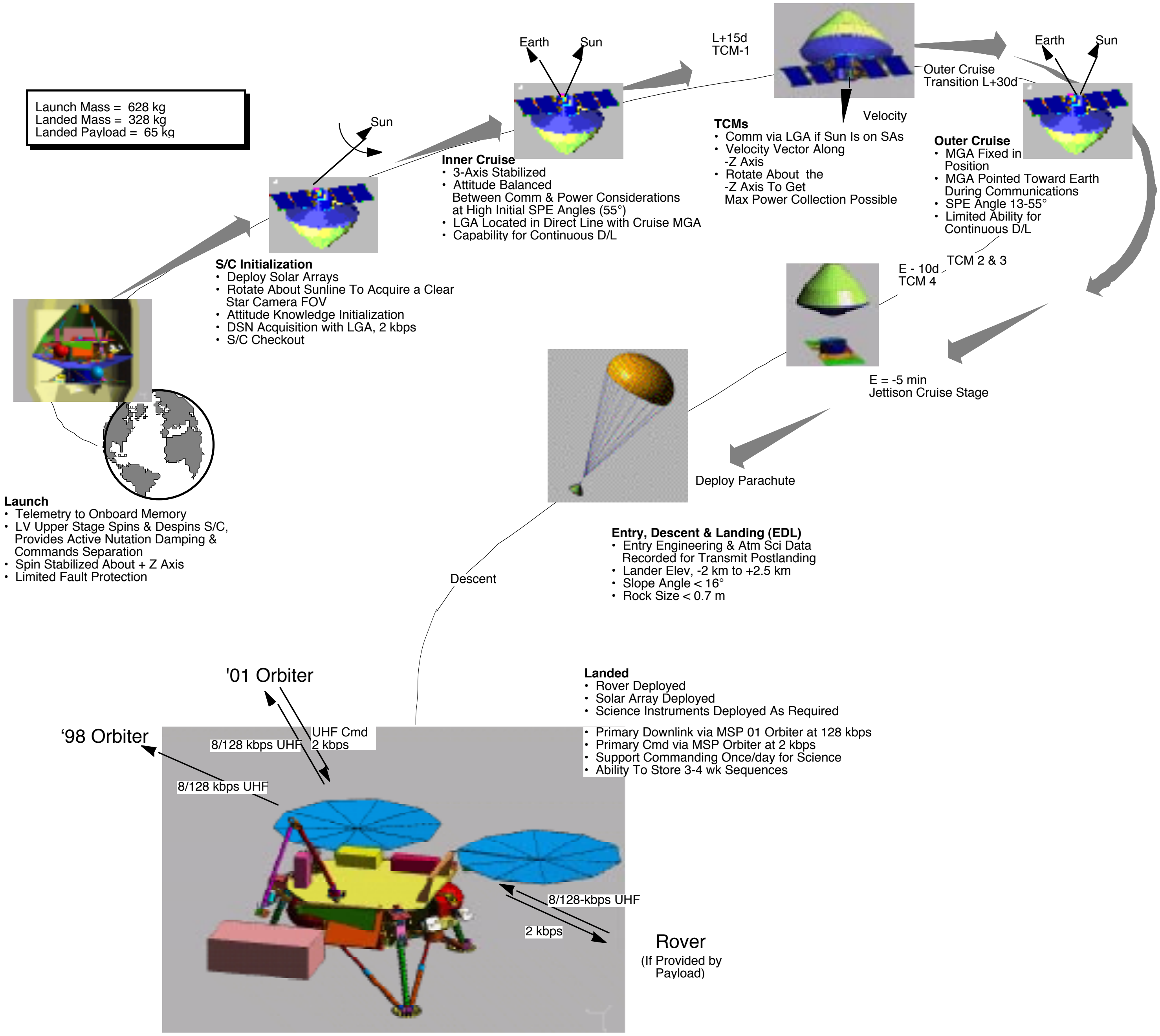

Launch Mass = 628 kg
Landed Mass = 328 kg
Landed Payload = 65 kg
Launch
• Telemetry to Onboard Memory
• LV Upper Stage Spins & Despins S/C, Provides Active Nutation Damping & Commands Separation
• Spin Stabilized About + Z Axis
• Limited Fault Protection
Sun
S/C Initialization
• Deploy Solar Arrays
• Rotate About Sunline To Acquire a Clear Star Camera FOV
• Attitude Knowledge Initialization
• DSN Acquisition with LGA, 2 kbps
• S/C Checkout
Earth
Sun
Inner Cruise
• 3-Axis Stabilized
• Attitude Balanced Between Comm & Power Considerations at High Initial SPE Angles (55°)
• LGA Located in Direct Line with Cruise MGA
• Capability for Continuous D/L
L+15d
TCM-1
Velocity
TCMs
• Comm via LGA if Sun Is on SAs
• Velocity Vector Along -Z Axis
• Rotate About the -Z Axis To Get Max Power Collection Possible
Outer Cruise Transition L+30d
Earth
Sun
Outer Cruise
• MGA Fixed in Position
• MGA Pointed Toward Earth During Communications
• SPE Angle 13-55°
• Limited Ability for Continuous D/L
TCM 2 & 3
E - 10d
TCM 4
E = -5 min
Jettison Cruise Stage
Deploy Parachute
Entry, Descent & Landing (EDL)
• Entry Engineering & Atm Sci Data Recorded for Transmit Postlanding
• Lander Elev, -2 km to +2.5 km
• Slope Angle < 16°
• Rock Size < 0.7 m
Descent
'01 Orbiter
'98 Orbiter
UHF Cmd
2 kbps
8/128 kbps UHF
8/128 kbps UHF
Landed
• Rover Deployed
• Solar Array Deployed
• Science Instruments Deployed As Required
• Primary Downlink via MSP 01 Orbiter at 128 kbps
• Primary Cmd via MSP Orbiter at 2 kbps
• Support Commanding Once/day for Science
• Ability To Store 3-4 wk Sequences
8/128-kbps UHF
2 kbps
Rover
(If Provided by Payload)

FIGURE 2.1.a

2.1.3 Entry, Descent and Landing

The entry, descent, and landing phase (EDL) begins one day before Mars arrival and ends when the lander has touched down onto the surface. The entry, descent and landing sequence of events is illustrated in Figure 2.1.3.a, and the flight profile is shown in Figure 2.1.3.b.

The aeroshell configuration has heritage from Pathfinder and Viking. The parachute system design is inherited from Pathfinder. A hydrazine monopropellant system and landing radar are used to effect terminal descent control and a soft landing.

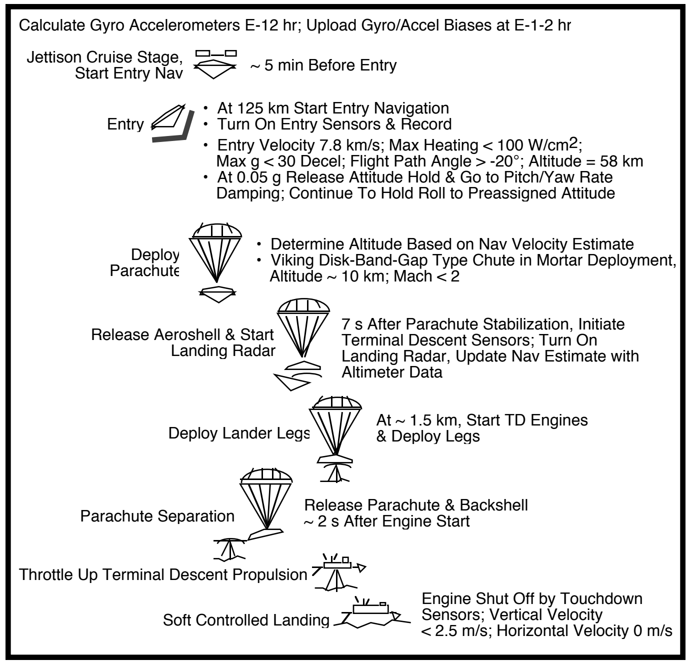


Figure 2.1.3.a EDL Sequence

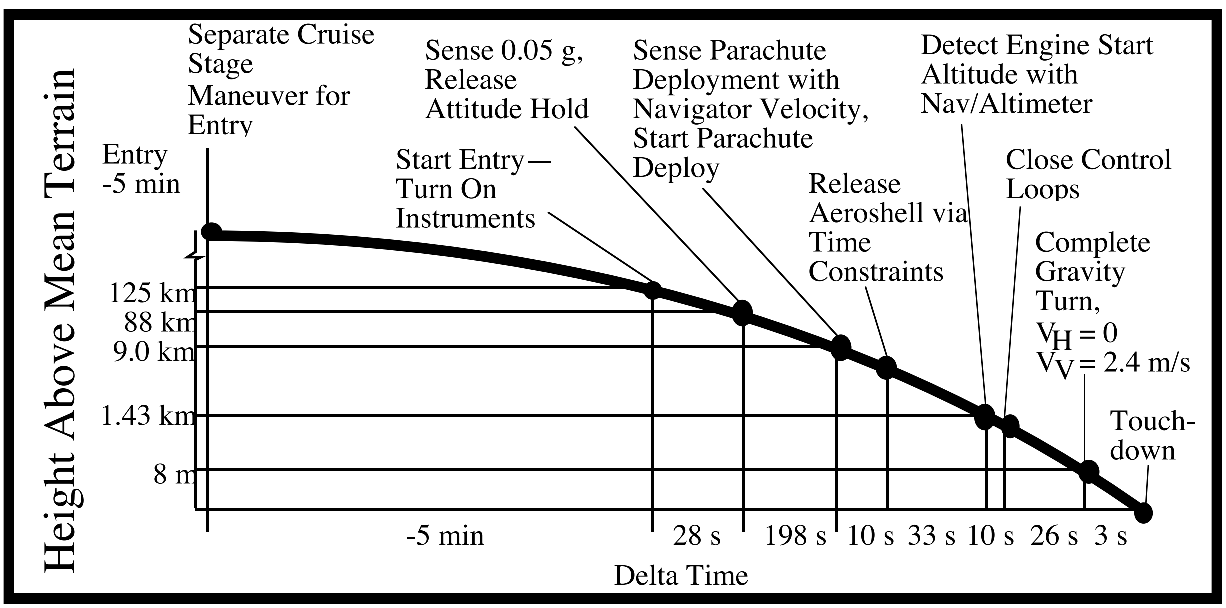


Figure 2.1.3.b Entry to Landing Flight Profile

Any type of landing on the surface of Mars will disturb the surface from its pristine state. Site contamination due to the lander's use of a monopropellant hydrazine terminal landing system is well characterized from Viking test data. Site contamination data is presented in Section 3.6.5.

2.1.4 Landed Mission

The landed mission begins with rover deployment, health checks and establishment of communication with the orbiter. Rover deployment will take precedence in the early lander operations and data return. Lander battery charging will begin as soon as possible. Routine operations will commence on the day following rover deployment .

2.2 Spacecraft

The lander flight system consists of a separable cruise stage and a propulsive terminal descent lander encapsulated in an aeroshell for mars entry (See Figure 2.2.a). Launch occurs with the lander positioned rightside up (i.e., heatshield down) per Figure 2.2.a. Following trans-Mars injection (TMI), the cruise stage solar arrays are deployed and the flight system cruises to Mars in a three-axis stabilized configuration. The lander payload does not have a view of space during cruise. Thermal control during cruise is provided by a capillary pumped (CPL) loop to reject heat from the lander thermal enclosure.

As the flight system approaches Mars, the available power is considerably reduced as the lander begins pre-entry activities. The cruise stage is jettisoned just before entry. The separated lander (inside the aeroshell) coasts to the entry point, enters and decelerates, and deploys the parachute. When the lander is near the ground, as determined by the landing radar, the lander separates from the parachute and performs a controlled descent to the landing site using a monopropellant blowdown propulsion system. The lander turns to a predetermined landing azimuth so the solar arrays will be aligned to collect maximum solar insolation each sol.

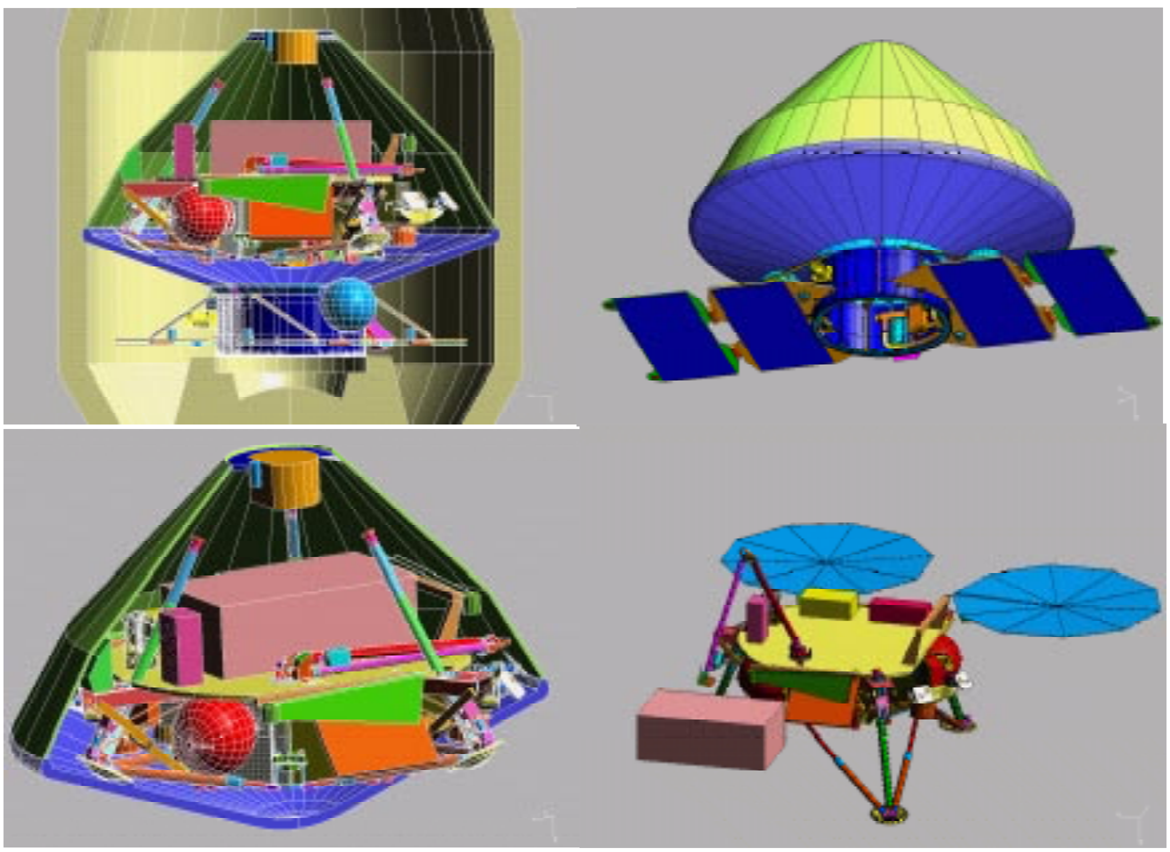


Figure 2.2.a Lander Flight System Configurations

The lander structure consists primarily of composite materials. The thermal enclosure provides temperature control to lander engineering equipment. The thermal enclosure structure is constructed of aluminum honeycomb with composite facesheets. Instruments are mounted on the top deck, along with the rover, UHF helix antenna, and rover deployment/sample delivery arm. Three square meters of deployable solar arrays deploy from the edges of the top deck. Figure 2.2.b provides dimensioned orthogonal views of the lander.

Figure 2.2.b, LANDER

The command and data handling subsystem (C&DH) uses redundant high performance reduced instruction set (RISC) flight computers in a central computer architecture. Payloads may use the available lander computing resources, but those that use their own microcontroller may enjoy more energy availability later in the mission. Standard interfaces are provided to the payload to facilitate early software development and checkout.

The landed power subsystem has a 3 m^2 photovoltaic array and two 12 Ah lithium ion batteries. The cruise solar array (jettisoned before entry) is a 2.3 m^2 photovoltaic array and uses the lander batteries for energy storage.

Telecommunications with earth are provided via a UHF link with one or more orbiters.

2.2.1 Descent Camera

The baseline descent camera optical system is composed of a 9 element refractive optics with an f/ratio of f/2, a focal length of 7.135, a FOV of 73.4 degrees, and an instantaneous FOV of 1.25 mrad (7.5 m @ 6 km, 12.5 cm @ 100 m altitude). The focal plane assembly utilizes a 1018 by 1008 photoactive CCD with a bandpass of 500 to 800 nm.

Image acquisition will be triggered by the entry and descent landing sequence at predetermined times, such as parachute deployment and jettison. The exact strategy for imaging will depend on the available throughput, the data storage capability, the number of images desired, and the editing and compression applied to the images. The data rate to the lander is expected to be 1 Mbps or greater, and storage is limited to 12.5 M bytes of non-volatile RAM and greater than 12.5 M bytes of volatile RAM.

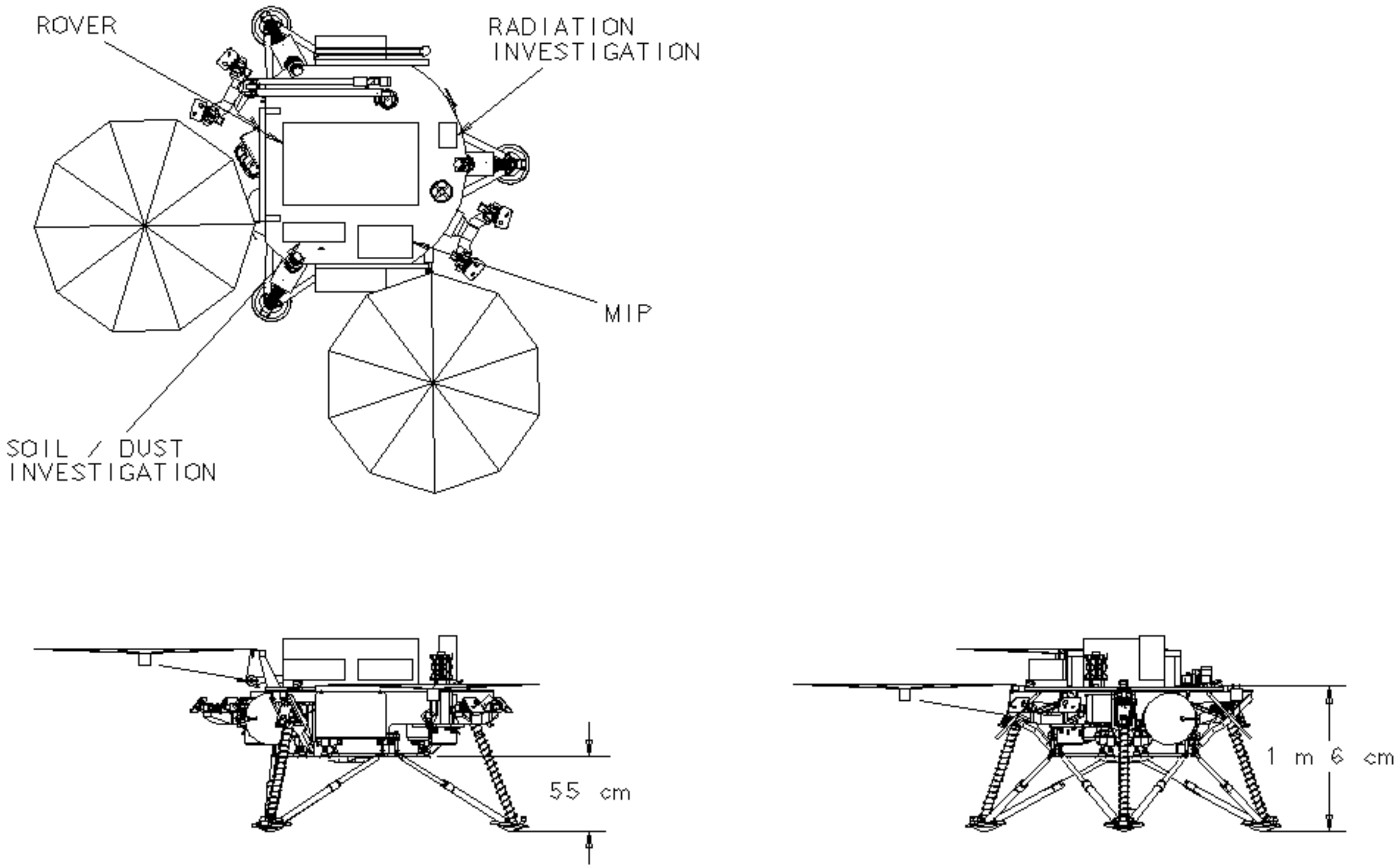


Fig 2.2.b - Lander
SCALE: .025

3.0 Constraints Imposed by Mission and Spacecraft Design

3.1 Payload Activities by Mission Phase

3.1.1 Launch Phase

The payload is launched in a powered off state.

3.1.2 Cruise Phase

The lander payload is totally enclosed in the aeroshell. There are no provisions for science collection during cruise. Instrument health checks are permitted within power and downlink capabilities . (See Section 3.3.3 for power allocations.) The spacecraft is designed to provide 100 bits per second downlink during cruise. All of this downlink is allocated for spacecraft health and welfare so instrument health checks would only be permitted if data volume is available above the needs of the spacecraft. No instrument operation is expected for the first fourteen days after launch, during trajectory correction maneuvers, within 30 days of Mars arrival and during other critical spacecraft events.

3.1.3 Entry, Descent and Landing Phase

Only descent imaging is planned after the aeroshell separates. (See Section 3.3.3 for power allocations.) Other than the descent imager, no power is allocated for science operations.

3.1.4 Surface Operations

The surface operations phase begins at landing and extends to the end of the mission. The first planned activities are lander functional checkout, lander deployments, return of critical descent images and rover deployments. No other science activities are planned (except survival heater power)until the rover is deployed.

Nominally, science operation will commence on the second sol of landed operations. One activity scheduled for sol 2 will be to transmit entry data to the orbiter. Other lander payload operations will begin based on priorities set by the project science working group (PSG). The proposers should outline the set of activities which are planned to take place daily, including any required instrument deployments, one-time activities (such as removal of covers), and any activities which are time-critical in nature.

During the 100 sol nominal mission, lander C&DH capability is available for up to 7 hours each sol. Due to thermal and energy availability considerations, the lander C&DH will be powered off for the remaining 17 hours of the Sol. Nighttime activities (except heater power) are only permitted by payloads with built in sequencing capabilities, see section 3.3.3 for more details.

3.2 Landing Site Limitations

3.2.1 Latitude Accessibility

For an entry from the hyperbolic approach trajectory, the landing site latitude and local solar time are constrained. However, rover engineering constraints will establish the actual landing latitude range. It is anticipated that the rover will survive and operate for up to one earth year between 15 S and 30 N (See Section 3.5.2.,c Energy). This will be the range of latitudes the lander will be designed to operate within as well (see Section 3.3.3.d-Surface Power to Payload).

The actual arrival time can be varied such that the rotation of the planet allows any longitude to be targeted. However, for an entry from the approach trajectory, the local solar time of day (and therefore lighting condition during descent) is constrained depending only on latitude and whether the selected entry trajectory is prograde or retrograde.

The local time of day at the moment of landing depends upon the latitude of the landing site and the proximity of the launch to the open or close of the 20 day period. To produce descent images, it is assumed that the landing will occur in daylight. Retrograde entries do not provide for daylight landings within the 30N to 15S range of latitudes. Hence, it is currently assumed that all entries will be in the prograde direction. For landing sites in-between 15S to 30N latitude ranges and for a launch at the opening of the launch period, the lander will arrive between 10:15 AM and 12:40 PM local surface time. For a launch at the close of the launch period, the lander will arrive between 9:45 AM and 11:45 AM local time.

3.2.2 Descent and Landing

The effects of atmospheric and navigation uncertainties will introduce errors into the landing site location. It is expected that landing will occur within 50 km (3σ)of the nominal desired location. Landing accuracy will be refined in Phase A/B.

3.2.3 Landing Site Elevation

The maximum altitude of the lander site is 2.5 km, to include al uncertainties. Lower altitudes are acceptable.

3.3 Resources Available for Payload Operations

3.3.1 Mass

The total allocated payload mass is not to exceed (NTE) 65 kg. The breakdown for each investigation, including all associated cabling, retention and release, health check, deployment and heater hardware and insulation, and contingency is:

INVESTIGATION	NTE MASS ALLOCATION (kg)
MIP	7.5
Radiation	4.0
Soil/Dust	8.5
Rover	45.0

3.3.2 Landed Payload Volume

The volume allocation for the payload is 284.9 liters and is suballocated as follows:

1) The main deck volume is limited by the envelope described in Figure 3.3.2.a. The investigation stowed volume allocation breakdown is nominally:

INVESTIGATION	DIMENSIONS (CM)	VOLUME (CC)
Radiation	12.5x18x33	7,425
MIP	25x24x40	24,000
Soil/Dust	15x15x45	10,125
Rover	39x60x104	243,360

2) Instrument devices to be deployed while on the surface can be mounted to the top deck using support structure attached to the main deck with stabilizers to the thermal enclosure base. These devices shall be included as part of the payload allocations.
3) Figure 3.3.2.a defines the stowed (undeployed) payload envelope which is constrained by the aeroshell configuration and rover deployment (Figure 3.3.2.b).

Figure 3.3.2.a Instrument Deck Allocation Configuration

Figure 3.3.2.b Lander Rover Deployment Configuration

3.3.3 Power/Energy Available

a. Power Conditioning

Payload power at payload power connector will be unregulated 28+8/-6 Vdc.

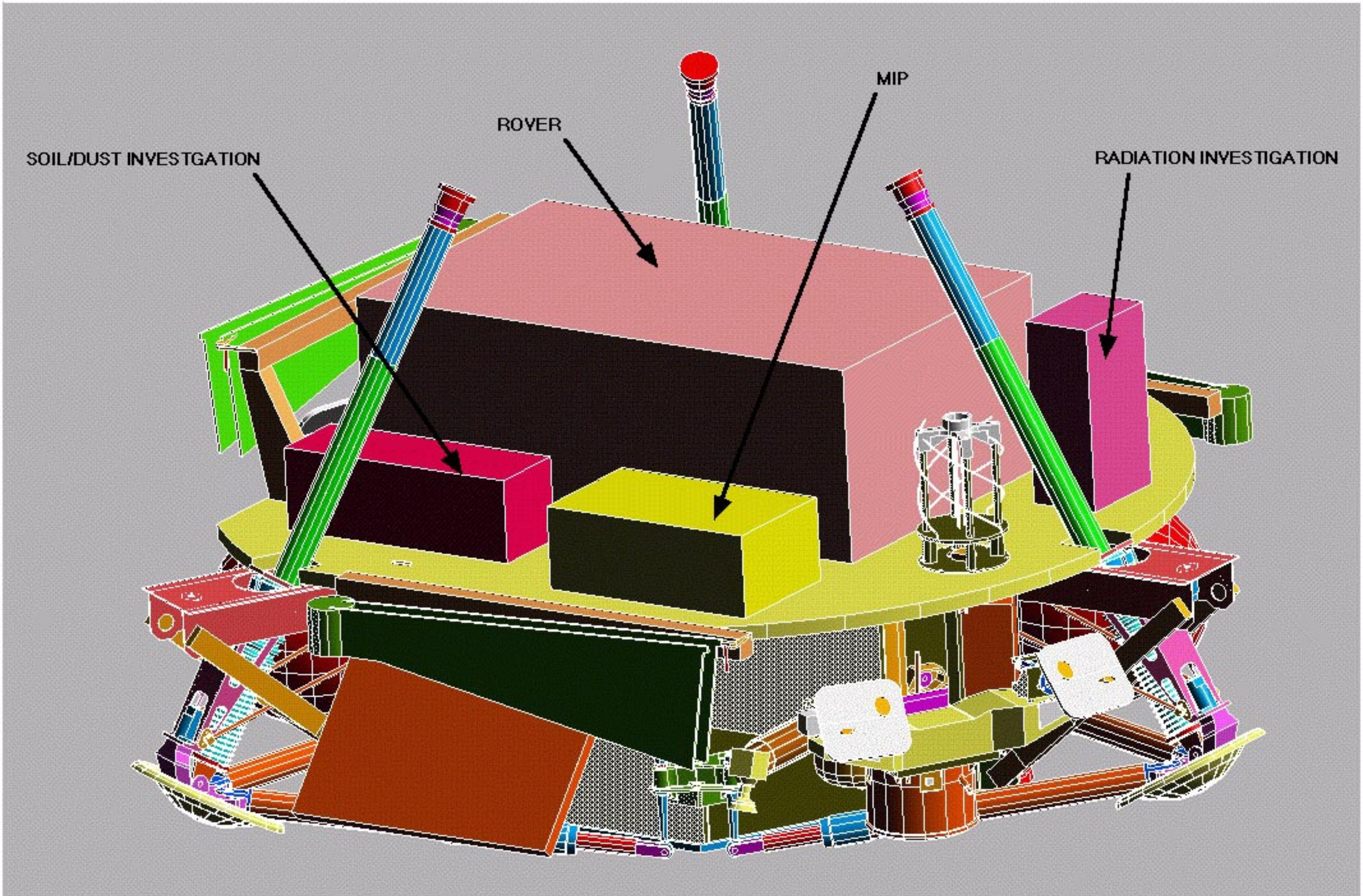
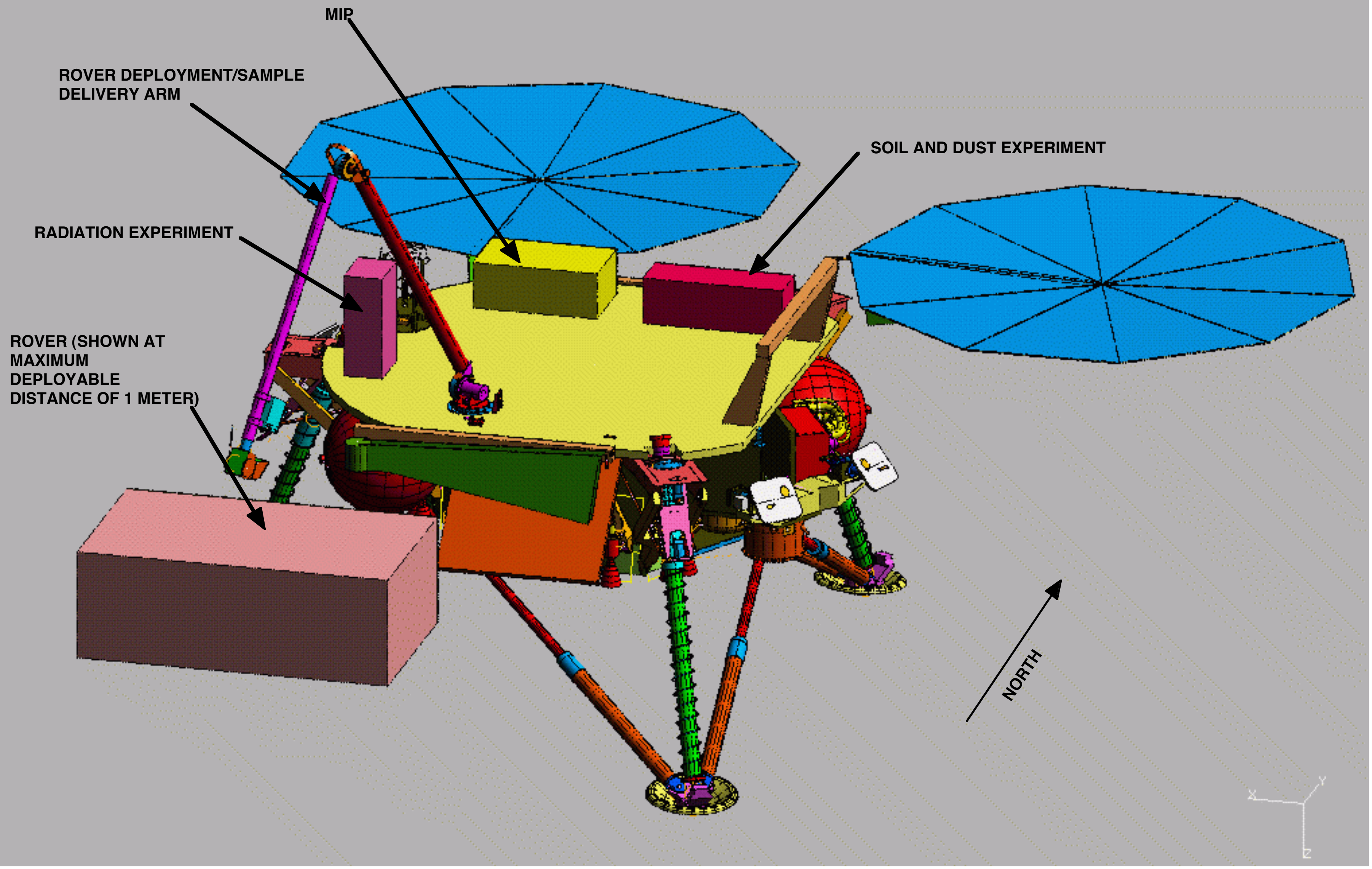


Figure 3.3.2.a Instrument Deck Allocation Configuration

Figure 3.3.2.b Lander Rover Deployment Configuration



b. Cruise Power

Cruise power to the payload is limited to heater power, bakeout, and health check as determined by power margin and sequencing. Maximum health checkout permitted will be 30 minutes. Power for this will be negotiated through the Interface Control Document (ICD) process. Power available to the payload during cruise, as margins are available and as can be sequenced without exceeding thermal constraints shall not exceed:

Cruise:

8W heater
10W health check (NTE 4 health checks for both inner and outer cruise)

c. Entry and Descent Power

Entry Preparations at Cruise Stage Separation to Entry:

8W heater
0W entry science (accelerometer data will be collected during entry and descent by the flight system)

Entry (after aeroshell separation):

5W heater
0W entry science (descent imagery will be collected by the flight system)

d. Surface Power to Payload

Average power targets (energy availability divided by daytime hours) during nominal daytime surface operations, in clear sky conditions (τ=0.5) and assuming a 16° (11° adverse tilt, 5° leg crush) tilt, are shown in Table 3.3.3.a. Power available during nighttime surface operations is shown in Table 3.3.3.b. These tables provide the average power available for all payload-specific functions including:

1) sensor and electronics power,
2) heater power,
3) deployment/mobility power

For the purpose of lander and payload surface operations, daytime is defined as when the sun is above a horizon mask of 20° (direct and diffuse sunlight are available), nominally 7 hours per sol. Nighttime is defined as when the sun is below a horizon elevation mask of 20° (direct sunlight is assumed not available for solar arrays; diffuse light is available as provided by Mars solar insolation models).

To conserve subsystem power, the lander will execute a nighttime power-down sleep mode. During sleep mode, the C&DH subsystem will not be continuously powered to accept payload data. However, it will be possible to periodically power-up the C&DH subsystem to accept payload data through lander sequencing. For payload equipment requiring extended nighttime operations consistent with power availability, it will be necessary for such payloads to provide their own control and data storage functions. This local control and data storage will be the responsibility of the payload and should be included in the proposed payload mass, volume, power and cost allocations.

When the temperature inside the thermal enclosure drops below the lower controlled limit (nominally at 100 sols at 15 south latitude, see Section 3.7.5.b) the lander enters “survival mode” where the lower limit of the thermal range cannot be controlled. Before the lander enters survival mode the payload systems will be powered down as required in order to conduct basic lander functions. Once the lander enters survival mode, no power is available to the payload for operations or heaters. In survival mode, temperatures inside the thermal enclosure will drop below qualification limits and will eventually approach Mars ambient.

Table 3.3.2.a Daytime Average TOTAL Payload Power Targets (Tau=0.5)

	Latitude (deg.)			
Sol (Ls, deg)	-15°	0°	15°	30°
1 (316)	25-30W	25-30W	25-30W	0-10W*
50 (344)	25-30W	25-30W	25-30W	25-30W
100 (9)	20-25W*	25-30W	25-30W	25-30W
150 (32)	0W*	25-30W	25-30W	25-30W
200 (55)	0W*	20-25W*	25-30W	25-30W
250 (77)	0W**	5-10W*	25-30W	25-30W
300 (99)	0W**	5-10W*	25-30W	25-30W

* If lander C&DH can be put into sleep mode, a total of 25-30W for 7 hours will be available

** If lander C&DH can be put into sleep mode, a total of 15-20W for 3-5 hours will be available

NOTE: Both conditions are valid ONLY if lander thermal limits are not exceeded

Table 3.3.2.b Nighttime Average TOTAL Payload Power Targets (Tau=0.5)

	Latitude (deg.)			
Sol (Ls, deg)	-15°	0°	15°	30°
1 (316)	5-10W	5-10W	5-10W	0W
50 (344)	5-10W	5-10W	5-10W	0-5W
100 (9)	0W	5-10W	5-10W	5-10W
150 (32)	0W	0-5W	5-10W	5-10W
200 (55)	0W	0W	5-10W	5-10W
250 (77)	0W	0W	5-10W	5-10W
300 (99)	0W	0W	5-10W	5-10W

e. Power Switches

Each instrument will be allocated one power switch to turn operational power on and off. A second switch will be allocated for survival heater power. The peak current for the switches is 2 Amps, while the peak current for the remote power controllers (RPC's) is 4.5 Amps. Peak currents in excess of these amounts may require parallel switches.

The power switching system consists of pre-mission configurable remote power controllers (RPC's) that are connected in series with individual switches. The switches provide load switching and selection of functions. The RPC's provide protection to the loads connected to it. Each RPC contains an over current trip function that will not allow excessive current. This over current protection is a trip function that is resetable. The RPC also provides current and ON/OFF status to the C&DH. This combination allows any load that has tripped the RPC to be reset as opposed to fuse protection that is final.

A limited number of latching relays may be available for payload use in lieu of using an RPC, if the payload so desires. However, fault protection on the latching relays includes a fuse, which if blown, is final.

3.3.4 Computational Resources

a. Lander Computer

The lander uses a high performance RISC processor computer that can provide the following computational services to the science payload: operating system, time/clock, sequencing, I/O interfaces with hardware, power switching, data packetization for downlink, and file system. Payloads must, however, provide their own data compression software. The lander computer will not compress payload data real time.. If the proposer elects to use the lander computer for computational services, the payload must meet the lander's flight software development cycle. The lander uses a spacecraft test lab (STL) to develop flight software and sequences. The flight software requirements are to be inserted into the ICDs. Deliverables are listed in Section 5.2.3.

b. Data Packetization

Data packetization services to put data into CCSDS format and Reed-Solomon encoding will be provided by the lander computer.

c. Processor Speeds

The on-board flight computer provides a variable speed capability for the payload that can be used as part of the sequence planning process. The computer speeds and their power deltas (chargeable to the payload power budget - Section 3.3.3.d) are as follows:

Computer Speed Increase* (MIPS)	Payload Power Charge (W)
23.5	12.4
9.5	6.2
2.5	3.1

* Lander engineering usage = 1.2 MIPS for normal surface operations

d. Volatile Memory

For those payloads that do make use of lander computing resources, a target of 1 Megabyte of volatile memory is available for payload program code. A target of 500 kilobytes are available for payload sequencing and sequence data. A target of 37 Megabytes of volatile memory are available to payloads for packetized data. It should be noted that the nominal communication data rate per sol is considerably less than this. An additional 37 Megabytes are available for temporary data storage. Algorithms must be programmed in ANSI Standard "C," Version 2, using VxWorks (a commercial real time operating system and development environment).

e. Non-Volatile Memory

1.5 Megabytes are available for program and sequence code. 16 Megabytes are available for untransmitted, packeted data upon night-time shutdown (on flash memory card), but this must be shared between science data and spacecraft telemetry. It is expected that 12.5 Megabytes will be available for payload use.

3.3.5 Data Return

The nominal communications link after landing will use the 2001 Mars Surveyor orbiter relay link. The UHF subsystem supports coverage throughout the 20 degree horizon mask. The data rate from the lander to the orbiter will be 8 and 128 kbps. Two passes per sol provides a nominal 50 Mbits/sol capability of which 40 Mbits/day is allocated to the rover and 10 Mbits/day is allocated to the lander. The relay link for commanding will be 2 kbps.

Science and telemetry data are transmitted using CCSDS formats. A total lander to Earth bit error rate of 10^{-6} for telemetry is achieved through concatenated Reed-Solomon and convolutional encoding techniques.

3.3.6 Lifetime

The expected surface lifetime of the lander, i.e., the time the lander is expected to operate before entering a survival mode, in which no science operations are possible, is shown in Table 3.3.6.a.

Table 3.3.6.a Expected Surface Lifetime as a Function of Latitude

Latitude	Lifetime with Arrival on Type 2 Trajectory
15° South	100 sols, with limited capability after day 100 until lower thermal limits exceeded (expected prior to day 150)
Equator	200 sols, with limited capability after day 200 until lower thermal limits exceeded (expected prior to day 250)
15° North	300+ sols
30° North	300+ sols, with limited capability 1st 50 days

Below the equator the lander batteries will freeze during the winter. Then, when sufficient solar energy is available to sustain daily operations, lander subsystems may be enabled and surviving systems can again operate. There is no guarantee that the lander's systems will survive these winters since the batteries and electronics are not qualified to the low temperatures they will experience.

3.3.7 Rover Deployment/ Sample Delivery Lander Robotic Arm (LRA)

The baseline system for deployment of the rover from the lander deck to the surface will utilize a reflight of the lander robotic arm (LRA) developed for the Mars 98 MVACS lander mission. After deployment of the rover, the LRA will be available to support lander based instruments and will constitute the mechanism for the retrieval of soil and dust for lander instruments.

Additional instrumentation can also be supported on the LRA, but the exact interface must be determined on an instrument by instrument basis. In particular, it must be noted that any mass allocation provided to the arm mounted instruments has to be within the range/mass capabilities of the lander robotic arm for rover deployment. The mass and power for such instruments will be charged to the appropriate investigation's allocation. Another issue that must be considered for instruments mounted to the LRA is routing through the rotating joints of any support or sensing cables. The LRA currently supports two 25 conductor cables for an arm mounted camera (the current baseline is the MSP '98 arm camera, with a mass of 0.350 kg), with additional conductors on the two LRA cables available for meteorology sensors. A summary list of the basic LRA capabilities follows:

-125C operating temperature
~10W during normal duty and ~35 W during heavy digging for total power expenditure
2W avg. for actuator heaters, with accommodations for a 25W scoop heater (if needed)
200 hr actuator life (goal)
50 cm trench depth in loose soil (accounting for a 45 degree angle of repose for cave-ins)
2 meter kinematic reach, providing a ~7 m^2 workspace (if mounted like MVACS)
1 cm absolute positioning and 0.5cm relative positioning of the active tool or sensor
Redundant joint position sensing and autonomous fault recovery software

A lander mounted cleaning brush
Motor current sensing to provide digging force information as well as arm and actuator safety

Joint 1:	shoulder azimuth	54 N-M
Joint 2:	shoulder pitch	146 N-M
Joint 3:	elbow pitch	86 N-M
Joint 4:	wrist pitch	15 N-M

Data is generated in an operator controlled manner, allowing data rates from ~ 85 bytes per command for engineering data, up to ~300Kbytes/hr for compressed digging data.
The software for the 98 mission is designed to fit within a 600Kbit EEPROM allocation.

These capabilities relate to significant digging forces when utilized properly, but are not straightforward to calculate due to the interactions of the joints. The cost due to any changes to the LRA mechanical, electrical, or software design will be charged to the requester.

3.4 Payload Interfaces

Payload interfaces applies to the Mars In-Situ Propellant Production (MIP), radiation monitor, and soil and dust instruments.

3.4.1 Configuration

a. Cruise Configuration

The cruise and entry configurations are shown in Figure 3.4.1.a. The configuration inside the aeroshell is retained through entry and parachute deployment. The heatshield is released once the system is stabilized on the parachute. No closeouts are available for payload access after integration of heatshield and backshell.

Figure 3.4.1.a Lander Spacecraft Cruise Configuration

b. Surface Access and Field of View

Surface access and field of view are depicted by Figure 3.3.2.a. Surface deployment devices (e.g. arm, mast or rover) can be mounted on the top deck only. Samples are delivered with the robotic arm (also used to deploy the rover). See section 3.3.7 for description of the robotic arm.

c. Payload Mounting

Mounting services for payload assemblies mounted to the top deck are provided by the lander contractor, and mounting services for payloads mounted to the robotic arm are provided by JPL. The proposer is responsible for the proper designs of the payload, booms, deployment devices, etc., to insure their adequacy for payload performance on the lander.

d. Optics Covers

Optics covers (hinged or deployable), if necessary for protection against surface environments or cruise/entry outgassing, are the responsibility of the payload proposer.

e. Surface Protection

The payload proposer shall provide protection for payload equipment vulnerable to damage caused by direct sunlight, dust, weather, or other environmental effects.

3.4.2 Thermal Control & Thermal Interfaces

a. Interface Constraints

All payload components shall be mounted to the top deck. 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.

b. 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

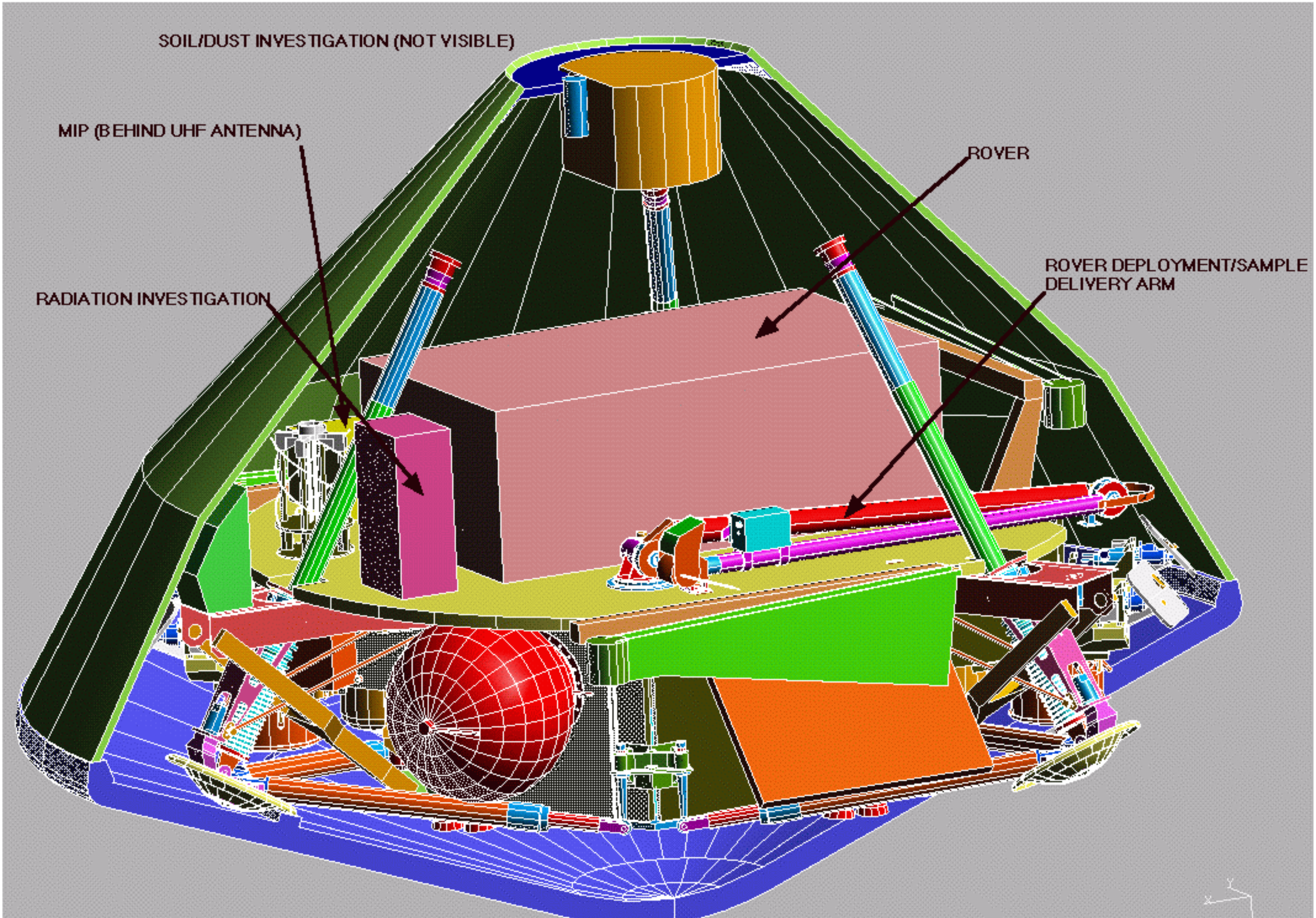


Figure 3.4.1.a Lander Spacecraft Cruise Configuration

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 3.7 Lander Environments. These environments should be used for early component design purposes. Top deck temperatures are provided for 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 greater than 0.89. 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 less than 0.1.

c. Integrated Thermal Analysis and Design

The spacecraft contractor will provide 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 the spacecraft contractor in a format compatible with the thermal analyzer software being used by the spacecraft contractor. This will be either TRASYS/SINDA ‘85 or TMG (thermal model generator provided as part of the I-DEAS CAD/CAM package) software. 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.

d. Environment and System Testing

The spacecraft contractor will perform system level thermal vacuum testing. The spacecraft will be tested in both cruise and landed (without solar panels) configurations. Design thermal environments for launch, cruise, and on-orbit phases will be simulated with a combination of IR and solar lamps. For system level testing, protoflight test margins in accordance with JPL standards (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 JPL standards (modified MIL-STD-1540C) prior to delivery.

3.4.3 Data Interface

a. Standard Interfaces

Standard data interface is RS-422. Each payload is allocated one low speed (9600 bps) RS-422 serial I/O channel. Each PI shall provide the interface circuitry based on signed ICDs. Each PI shall supply a payload interface simulator to the lander for payload checkout. Refer to section 5.2 for delivery schedule requirements

b. Status and Housekeeping

Each PI shall provide payload status and housekeeping data for collection and formatting by the lander computer. Each payload shall be provided two digital inputs, two digital outputs, and two analog temperature sensor outputs. Opto-couplers shall be used for grounding isolation on discrete inputs and outputs.

c. Time Tag Accuracy

The data time tag accuracy is within 30 milliseconds of spacecraft clock (3σ).

e. Electrical Connectors

Each PI is responsible for providing both sides of the standard connectors at the interface based on signed ICD's. The connector for the harness side of the interface needs to be delivered prior to the instrument to support spacecraft assembly (see section 5.2). Standard connectors are miniature "D" or micro "D". Other connector types will be treated on a case-by-case basis.

f. Workstations

Workstations provided by the ground system will be used for computer support equipment, software development, and hardware/software integration, and as data display/command generation areas.

3.5 Strawman Rover Capability and Payload Resources

The rover capabilities and resources available to the payload are those that reflect a preliminary design developed by the MSP 2001 Project to provide the basic functions of mobility, power, thermal control, computation, and communications. After NASA has selected the payload, the design of the rover will be negotiated with the selected PI and will be optimized around the selected payload and mission, insofar as is possible within the limits of available resources.

All proposed instrument deployment devices, sampling mechanisms, drills, sample containers, or other devices necessary for the instruments to accomplish their tasks must be clearly defined in the proposal. The proposed devices and the resources required for their operation are considered to be part of the total allocation for the payload. Because such devices significantly affect the basic design and operation of the rover, the MSP 2001 Project expects to approve their development and integration into the rover. In the case of the sample container, which must be transferable to a sample return vehicle at a later date, the project expects to assume responsibility for its development.

The total resources available to the lander and its payload are described in Sections 3.3 and 3.4 of this document. The following sections describe the resources allocated to the rover payload only.

Rover Payload Resource Summary

Mass = 15 kg. This mass allocation includes all science instruments, science electronics, and associated payload equipment (e.g. sampling devices, payload deployment, etc.). This mass allocation is based on a total rover and rover payload allocation of 45 kg, and the strawman rover point design mass of 30kg.

Power = 15 watt-hours per sol. This is an estimate and will depend on the actual way the rover is operated.

Computer = 32 bit processor, 15Mhz clock speed, 0.5 Gigabits data storage.

I/O Devices = Can be accommodated modularly, and will be negotiated with the PI.

Data Interfaces = RS 422

Volume = 33 Liters (33,000 cc) as described in Section 3.5.1.

3.5.1 Strawman Configuration

The baseline point design configuration is shown in Figure 3.5.1.a (in the stowed configuration during cruise and entry), and Figure 3.5.1.b (in the deployed position). This configuration is based on the work which has been and will be done in NASA's Telerobotics and Rover Technology Program through the end of FY98.

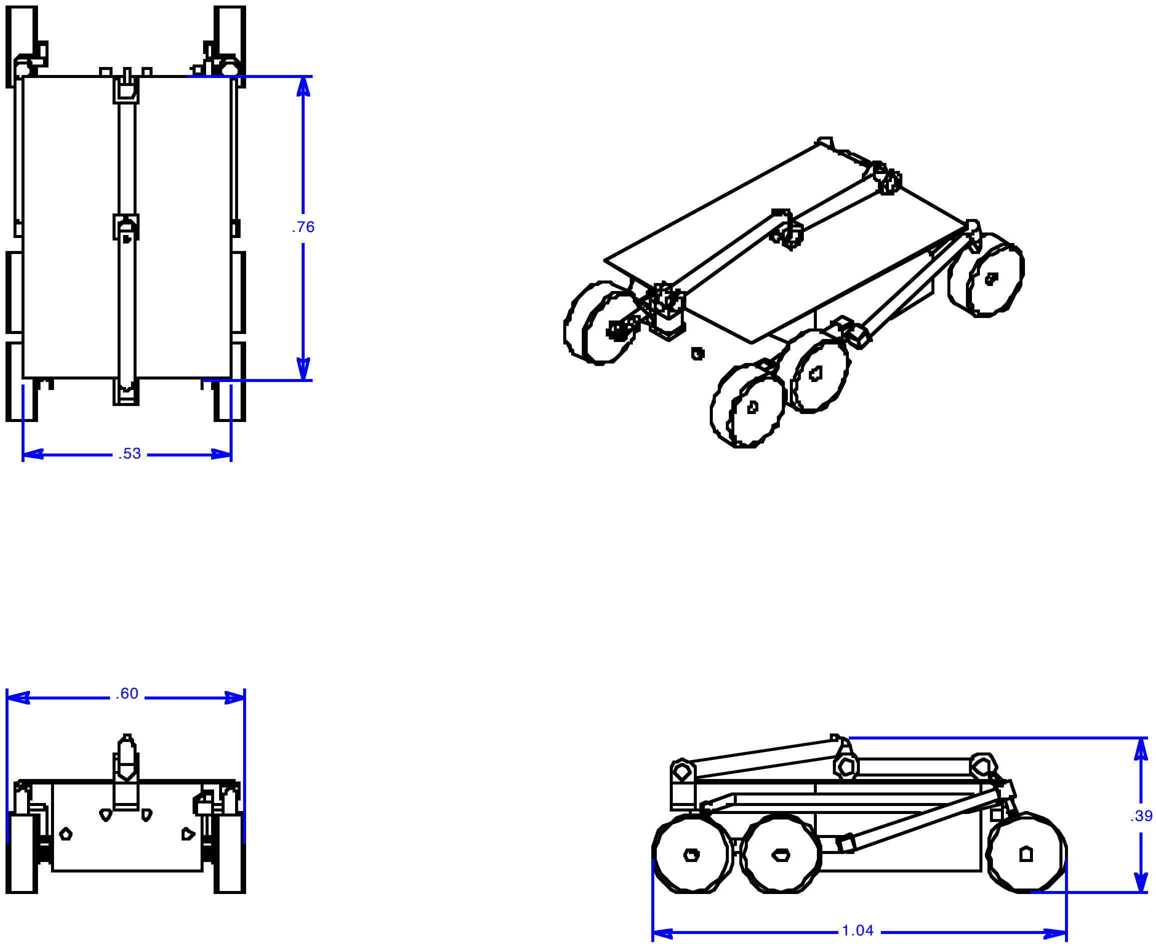


Figure 3.5.1.a - Rover in Stowed Configuration

This configuration shows a six-wheeled design wherein the two rear wheels will have Ackerman steering for relatively agile turning. All six wheels are individually driven.

The mast shown provides for stereo black and white cameras for navigation and two front and two rear cameras 35 to 40 cm above the ground for hazard detection . The solar array is top-mounted; its size is limited by the volume available for rover stowage on the lander. This configuration will accommodate up to 28 liters (28,000 cc) of payload mounted externally at the front of the rover 22 cm high X 37 cm wide X 35 cm long (direction of travel), and 5 liters (5000 cc) mounted internal to the WEB (warm electronics box). Internal volume may be increased by a corresponding reduction in external payload volume. The WEB is a low mass, composite, “sheet & spar” construction that is not well suited for the design of generic load paths or interface points, therefore, the actual mechanical interface between the payload and the WEB will be established after instrument suite selection. Proposers should assume that external payload will interface to the front wall of the WEB (no load path through the solar cell substrate), and internal payload will interface to the floor of the WEB.

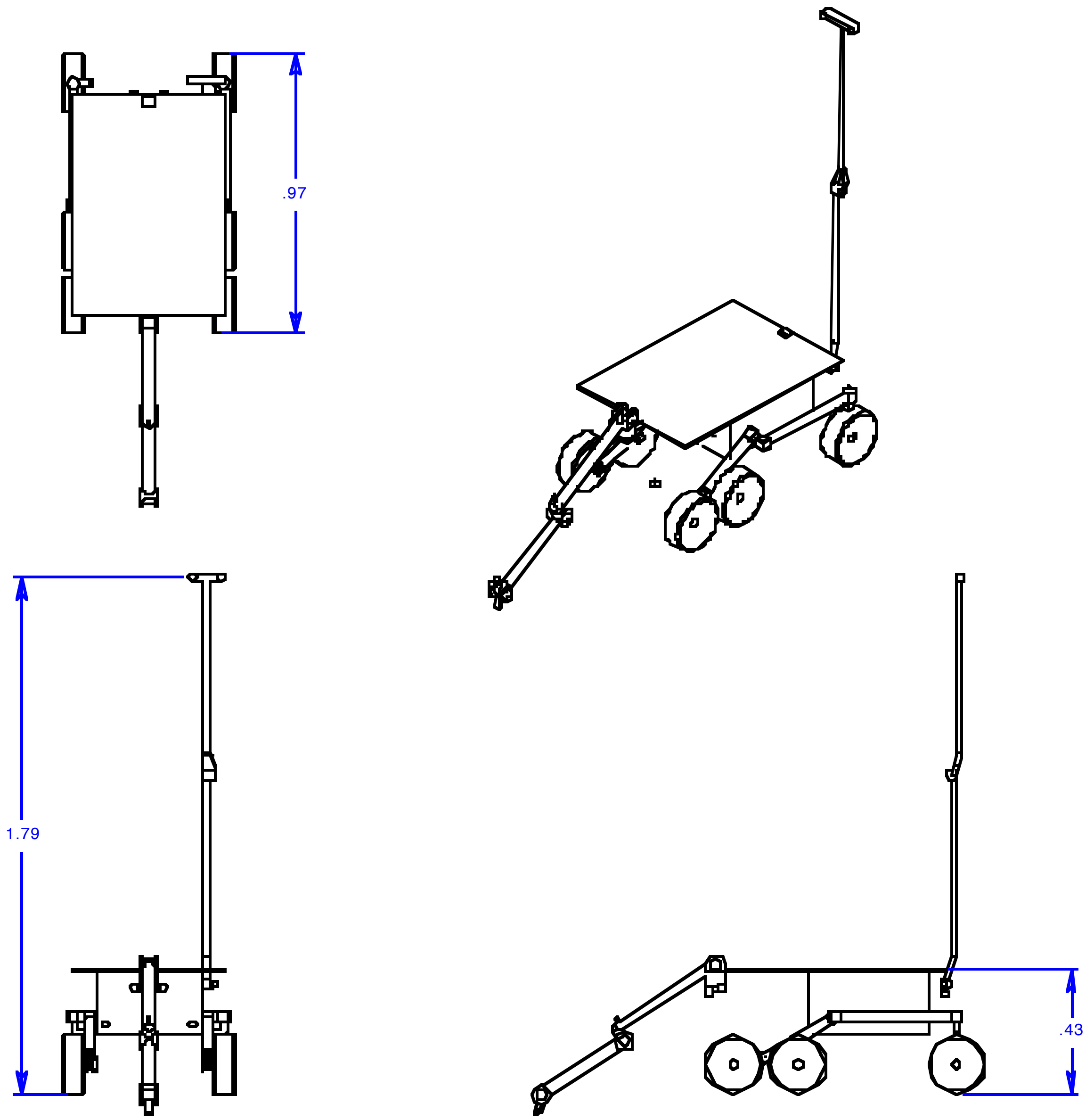


Figure 3.5.1.b - Rover in Deployed Configuration

3.5.2 Resources Available to the Payload

a) Mobility Capability

The rover is capable of moving at a rate of 6 cm per second on level ground. This capability is determined by the computational speed as well as the mobility design and the power available. The actual mobility capability per sol or during mission lifetime will depend on the design that the PI chooses for the mission. That is, the distance traveled per sol will depend heavily on the number of stops that the rover makes to investigate or sample areas of interest, on the roughness of the terrain, and on whether the rover is directed toward a target at some distance, or directed to wander to explore a specific area.

The rover is capable of moving through VL1 terrain as describe in Section 3.8. More rugged terrain will slow the progress and perhaps even make it impossible for the rover to traverse through the terrain.

b) Power

The power subsystem consists of the following components:

1) Solar Array: The solar array provides power to run daytime operations, and to charge the battery. The array is currently baselined as a 0.36 m^2 panel with gallium arsenide solar cells. This array area assumed that 10% is lost due to camera boom and potential manipulator arm attachments points, clearance, and stowage. Panel performance is assumed to be degraded an additional 15% as a result of shadowing from the camera boom and manipulator arm. There are several additional solar array options that can be negotiated with the PI that may provide better performance, at possible additional cost.

2) Battery: A rechargeable lithium chemistry has been baselined for the rover battery. The battery has been sized at approximately 10 Ahrs with an operating voltage of 11 to 16 volts. The battery is required for load sharing during traverse and science activities during the day. It is also required for communications during the 4 AM and 4 PM telecom passes.

3) Power Electronics: The power electronics include battery chargers, battery discharge regulator, and power conditioning. The battery charge will either be controlled autonomously by the battery charger, or by utilizing the CPU in a low clock speed, low power mode. The battery discharge regulator maintains the battery voltage below the solar array voltage, but close enough to the array maximum power point to avoid significant solar power loss during power sharing conditions. Power conditioning hardware provides the appropriate voltages to the rover electronics. Table 3.5.2.a provides a summary of voltages in the initial baseline rover design. Power interfaces and additional voltages will be negotiated with the PI after selection.

Table 3.5.2.a Rover Voltages

	Voltage
Unregulated Core Bus	
nominal operations	16.0 - 22.0 v
off nominal operations[1]	11.0 - 22.0 v
Control and Navigation	5.0 v
Telecom	28.0 v

[1]. Off nominal operations allow the battery discharge regulator to be bypassed in the event of a failure, or to improve efficiency during operations when only the battery is providing power (such as during night time telecom passes).

c) Energy

The graph below shows the energy available from the baseline solar array design at a range of martian latitudes. It provides estimates over the course of a 12 month Mars mission, assuming that the lander arrives at an aerocentric longitude of approximately 315 degrees. The estimates do not account for any effects from dust deposition on the arrays.

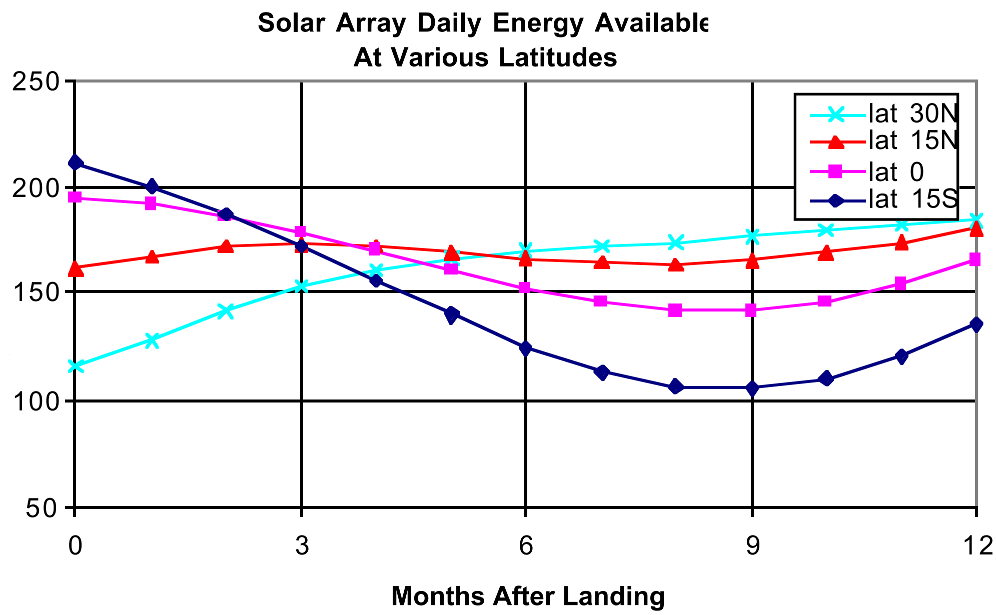


Figure 3.5.2.c

These solar array predictions bound the energy available for distance traverse and local operations as described in Section 3.5.4. Table 3.5.2.b provides the projected energy required for daily traverses of various times. An example of a typical energy budget for local operations is provided in Table 3.5.2.c. Table 3.5.2.d gives current estimates for the power usage of the rover engineering subsystems.

Table 3.5.2.b Estimates of Rover Daily Energy Requirements For Various Traverse Times

Drive Time (hours)[1]	**1.0**	**1.3**	**1.7**	**2**
Housekeeping Computer, I/O, SSM, night timer	56.9	66.6	76.4	86.1
Traverse Drive motors, gyro, accels	20.5	27.3	34.2	41.0
Navigation Deploy motors, stereo imager, sun sensor	4.1	5.4	6.8	8.1
Communications Receiver, transmitter	16.3	16.3	16.3	16.3
Heating	5.0	5.0	5.0	5.0
Instruments[2]	15.0	15.0	15.0	15.0
Battery Charge/Discharge	27.3	27.5	27.8	28.1
Total Whrs	**145.0**	**163.2**	**181.4**	**199.7**

1. 1.0 hours of driving is approximately equivalent to 50 meters.
2. Instrument energy usage of 15 Watt-hours is a placeholder. Actual value is TBD, will be determined by landing site and other operational requirements .

Table 3.5.2.c Estimates of Rover Daily Energy Requirements During Local Operations

Event	Energy Required (Watt-hours)
Housekeeping Computer, I/O, SSM, night timer	56.9
Traverse Drive motors, gyro, accels	4.1
Navigation Deploy motors, stereo imager, sun sensor	4.9
Communications Receiver, transmitter	16.3
Heating	5.0
Sampling and Caching[1]	25.0
Instruments[2]	15.0
Battery Charge/Discharge	22.3
Total Whrs	**149.4**

1. Assumes approximately 1 hour of coring for nominal sample.
2. Instrument energy usage of 15 Watt-hours is a placeholder. Actual value is TBD, will be determined by landing site and other operational requirements

Table 3.5.2.d Rover Power Requirements

Component	Power (watts)	Notes
Computer	6.0	Scaleable w/ clock speed
I/O	2.0	
Solid State Memory	0.6	
Clock	0.2	
Drive/steering motors	3.0	Per motor
Lasers	0.5	
Accels	1.5	
Mast motors	3.0	
Stereo imager	3.0	
Sun sensor	0.5	
Lasers	1.0	
Receiver	5.0	
Transmitter	55.0	

d) Computational resources

Rover Computer

The computational environment of the rover will be a 32-bit rad-hard processor supporting clock rates of at least 15 MHz (with several variable rates with power consumption roughly proportional to clock rate). Memory of 0.5 gigabits will be provided by a solid state recorder (SSR). The SSR may exhibit single event upsets due to radiation but not large-scale permanent degradation. It is anticipated that some degree of error correcting codes will be employed to maintain the data in the SSR at the desired level of integrity between data acquisition and downlink in consultation with the selected PI It is anticipated that the processor system will have a power consumption of <5 W at 10MHz and that the SSR will have a power consumption of 0.6 W. All computing resources (except the SSR) will be fully occupied during terrain navigation; if the selected PI wishes to stop the vehicle for other activities essentially all the computing functions can be made available for science data gathering. Compression and formatting of downlink data is expected to consume essentially all available computing resources (limited by the power budget) so as to maximize the utilization of the downlink data stream.

Input/Output

The input/output (I/O) architecture of the processor will support at least 21 motors, 7 active pixel sensor cameras, a 3 axis accelerometer, 4 laser spot projectors for hazard sensing, power switching for various devices, and miscellaneous digital and (relatively high performance) analog I/O. Rover engineering subsystems require support for 11 motors, leaving 10 currently available for the payload. Additional motors, cameras, etc. can be accommodated if required by the PI. The 7 navigation cameras are monochrome wideband Silicon array detectors organized as fixed front-mounted and rear-mounted stereo pairs, and a mast-mounted stereo pair, and a sun sensor. The fixed stereo pairs will have a spatial resolution of approximately 3 mrad/pixel; the mast mounted (pointable) pair will have a resolution of approximately 1 mrad/pixel. Cameras of the type used on the mast have been integrated with miniaturized filter wheels and near-IR spectrometer (1-2.5 micron) under the NASA Telerobotics technology development program as described in [Wilcox]; these functions could be added but are not needed for engineering purposes.[1]

Software

Software for the rover will be primarily in C/C++ following the models of the Sojourner rover and the Rocky-7 research rover. It is anticipated that the rover on-board computer will perform the functions of command queuing (i.e. it will transmit instrument commands to instruments as ASCII strings at appropriate points in the execution sequence as determined by Earth operators) and it will buffer instrument data and perform packetization and other data formatting functions necessary for downlink to Earth. The specific interface protocols, data storage formats, volumes, and periods will be negotiated with the selected PI The payload must meet the software development schedule shown in Section 5.

Data Interfaces

It is assumed that the standard interface to science instruments will be via serial RS 232/422/485-style interfaces using a standard device driver for the serial port. Additional parallel, analog, or other interfaces with instruments will be negotiated with the selected PI

e) Thermal Control

The baseline rover point design employs a “warm electronics box” or WEB to maintain electronics (including instrument electronics) between plus and minus 40 degrees Centigrade, and batteries between plus and minus 30 degrees Centigrade. The WEB will utilize up to 5W of heat from five radio-isotopic heater units (RHU’s). The rover thermal design is passive, i.e. it requires little or no electric heater power other than perhaps short spot heating of components. External equipment must be qualified to operate in accordance with Section 3.8 of this PIP.

The WEB structure uses opacified silica aerogel insulation incorporated within its walls. The WEB is heated each sol by a combination of RHU’s, electronic and waste heat. Thermal straps will be used as necessary to distribute the heat.

f) Telecommunications

The baseline rover point design employs a UHF transponder to communicate with any orbiters that are still functional during the 2001 rover mission. These may include one or more of the Mars Surveyor 2001 orbiter or the 1998 orbiter. Each orbiter is assumed to provide two 6-minute passes per day for the rover communications, with a 8 kbps command link and a 128 kbps telemetry data link . The engineering data from the rover is assumed to require 1 to 5 % of the telemetry data link capacity per pass, with the remainder being available for science data.

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[1] Wilcox, Brian et al, "Nanorovers for Planetary Exploration", proc. AIAA Forum on Advances in Space Robotics, Madison, WI, May 1996.

g) Lifetime

The rover's operational lifetime is a function of its location on the planet (see Figure 3.5.2.2 power profile). It is expected that the rover can survive and operate for up to one Earth year at the most southerly latitude.

3.5.3 Activity Time Lines

A strawman rover activity time line has been developed to provide context for the proposers.

A description of rover activities over the duration of a Martian day is presented. The narrative provided here is to be taken as an approximation of the actual mission time-line that will only emerge as the rover design matures and specific science objectives are defined. Nevertheless, this description provides the following:

- A representative time-line of actions taken by the rover, and events both internal and external to the rover.
- A measure of the expected engineering related (i.e. non-science) consumption of rover resources such as time, power, energy, communication, and sensing.
- An indication of operational dependencies between rover activities.

3.5.4 Operational Modes

Due to the limited availability of solar energy, the rover will not be actively involved in exploration related activities through most of the Martian day. While the actual "hibernation" time varies with the season, activity may be restricted, for example, to between 1030 hrs to 1330 hrs (Mars local time). Daily operations consist of one of the following:

1. Distance Traverse. The rover proceeds to a distant target as specified by a heading vector and distance.
2. Local Operations. The rover performs a number of science operations such as image, core, dig, etc., all within close proximity to each other (within 5-15 meters of the starting point).
3. Battery Charge. The rover remains stationary and re-charges batteries in anticipation of a busy schedule the next day, or as a result of large energy drain from the previous day's activities. These battery charge days could be more frequent during the winter season, during periods of heavy dust loading in the atmosphere, or after dust accumulation on the solar panels.

Distance traverse refers to the means by which the rover moves beyond this local region of operations, and represents translation of the rover by 100's of meters. The local operations region of operations is defined by the panaromic view returned from the rover mast cameras and represents a view extending out a few 10's of meters from the point where the panaromic images were taken.

a) Communication Operations

Communications are needed for both distance traverse and local operations. The rover communication related operations are common to both modes of operations

All rover communication is via relay orbiters in polar orbit around Mars. If the rover location is equatorial, then about 2 communication opportunities are possible, each lasting about 10 minutes. At higher latitudes the number of relay opportunities increases. For the selected mission design, the communication passes are at approximately 0400 hrs and 1600 hrs i.e. early in the morning, and late in the evening. During both these opportunities, the rover is essentially quiescent with only those subsystems operational that are needed to support the communications. Each pass has about 6 minutes of full duplex transmission with commands to the rover transmitted at 8 kbps and data from the rover sent at 128 kbps. For the current mission design, a maximum of 40 Mbits per sol can be transmitted from the rover via the relay orbiter to the DSN receiving station on Earth. This limited availability reflects performance limitations on the communication equipment on the relay orbiter, Earth/Mars distances, as well DSN operational and resource constraints .

Communications consists of receiving commands defining the day's activities, and transmitting the data accumulated by the rover. A maximum of about 2.8 Mbits of command data can be sent to the rover during each 6 minute relay orbiter pass. The size of the commands devoted to robotic operations is expected to be fairly small since the rover supports a high-level command dictionary of operations that it can perform, removing the need to specify detailed actuator trajectories or sensor operations. About 0.1 Mbit of the command capability is reserved for other engineering related functions on the rover, leaving the rest of the capacity to science related functions.

As the lander based experiments and operations come to an end approximately 100 days after landing, all of data link resources could be devoted to the rover.

The possibility exists that the data returned to Earth from the evening pass can be used to refine the selection of data that is to be returned from a subsequent pass the next day. Such a capability could be used to send to Earth summary data (e.g. "thumbnail" images) in one communication, to be followed by a limited but selective set of "high resolution" images in a subsequent transmission. Care must be taken to ensure that this buffering does not cause the on-board data storage (approx. 0.5 Gbit) to be exceeded. It should also be noted that the overall turn-around time to perform this kind of data selection would probably exclude the possibility of having the data in an evening transmission be used to select the data for the following morning's transmission.

b) Distance Traverse

Long distance traversal is expected to be performed by providing the rover with a heading direction and a total distance of traverse. The rover has the capability of maintaining this heading during the traverse while avoiding the obstacles that it encounters along the way. The direction and length of such a traverse is usually determined by orbital imagery or descent imagery from the lander.

During such a traverse, the rover has only an approximate knowledge of its position with respect to its starting point. Furthermore, this knowledge degrades with time, as a result of cumulative odometry errors resulting from wheel slippage, as well as errors in sun-sensing used to maintain heading. The error in on-board knowledge of the rover position could be as much as 200 meters for a 1000 meter traverse.

The rover position can however be estimated more accurately using other techniques. An earth-based processing of successive, overlapping navigation images returned from the rover during the traverse could, in theory, propagate the rover position to an accuracy of a few meters. However, the extent to which such off-board analysis is performed is a function of the overall mission requirements. The important point is that rover itself does not depend on a high accuracy estimate of its position in order to perform the long distance traverse.

One point to be noted is that the heading/distance mode of operation will allow the rover to make progress only if the obstacle field is not too dense. In terrain similar to Viking Lander Site 2, the rover will expend much of its driving time avoiding obstacles and may not make much radial progress away from the starting point. There is currently no global path planning and execution capability planned for the rover that could reliably make progress in very difficult terrain.

The time-line of activities for a distance traverse could then consists of:

1. Rover wakes up in the morning and processes the command specifying the details of the required distance traverse: heading direction, distance, frequency of forward looking navigation images; frequency and nature (filters, resolution, compression etc.) of science images.

2. Rover orients itself to heading direction and commences traverse, avoiding all local obstacles it encounters. Actual drive time would be approximately 1/3 of total elapsed time.

3. At command specified frequency, the rover stops and proceeds to take requested navigation and science images.

4. Rover achieves required distance of traverse and stops.

5. If the rover is at its final traverse destination, or if it has been instructed to do so, it proceeds to take a panaromic images suitable for planning a science operation at that destination.

6. Rover proceeds to ready all the data acquired for the evening relay to the orbiter. This could involve compression of the data to fit the constraints of the communication link.

7. Rover goes to sleep with only the communications receiver active to receive the "wake-up" call from the orbiter.

8. Rover wakes up at the late evening orbiter pass and transmits the requested data, and then goes back to sleep.

   Rover wakes up early in the next morning for the next orbiter pass, transmits the requested data, and then goes back to sleep to await daylight.

c) Local Operations

Local operations are where the bulk of the science operations are performed by the rover. They are confined to the region viewed in the panoramic image sent back to Earth in the previous communication cycles. With this panaromic image, mission scientists and planners select science targets and operations, provide nominal "via points" for the rover to traverse in order to achieve the desired science operation points, and sufficient description of the target in order for the rover to be able to confirm the achievement of the science goals.

The panaromic images needed to support these operations are a major consumer of the total communication bandwidth resources available to the rover. While the total size of the panaromic image is driven by the nature of the panaromic image (resolution, spectral bands, compression) needed to support the selection of science targets, a minimum size is needed to perform the engineering functions of path via-point specification and target characterization (size, local rock shapes and constellations).

Representative data size numbers for a panoramic image that satisfies these engineering requirements are based upon mission plans for the Mars Pathfinder mission and indicate that an adequate, full 360 degree, 2-tier, stereoscopic, black/white panaromic image, is possible with 22 Mbits after lossy compression. A similar data set size to achieve the objectives of science target selection can be bounded by other Pathfinder data that indicate a need for 1.11 Gbits for a complete panorama in 13 geology filters. Lossless compression of 1.5:1 reduces this size but clearly the total data volume dwarfs the daily data return capability of the mission. Thus lossy compression is needed for the science content of the panorama and must be addressed when planning the science objectives.

Another option is to initially return only a partial panaromic image that is adequate to select a science target within a subset of the full 360 degree, two tier panaroma. While the rover is commanded to perform science operations based upon this limited set, the rest of the panaromic image can be returned. Depending on the data storage capabilities and needs, the subsequent transmissions could be of the originally obtained (but only partially transmitted) panoramic data set, or it could be of a freshly acquired set of images. While the latter is obviously feasible, it should be noted that the mast location for the successive partial panaromas may not coincide.

In addition to the engineering requirements on the panaromic image, a portion of the data set is also to be reserved for engineering health and system monitoring data. This is expected to take only 0.5 Mbits.

The time-line of activities for a local operations could then consist of:

1. Rover wakes up in the morning and processes the command specifying the details of the science task operations to be performed at the site: via-points, target locations, target descriptions, number of samples, instrument deployment/activation, operations priority, time-outs, etc.

2. The rover then services any overnight data from science instruments that have been active through the night .

3. The rover proceeds to the highest priority science target and performs the operations, using sensor feedback where possible to confirm successful operations. Auxiliary imaging data is taken as needed to provided "documentation" of the science operations. In the event of an unsuccessful operation (either as a result of a detected failure or a time-out), or upon successful completion of the science task, the rover proceeds to the next science task.

4. The rover monitors its on-board resources and when these reach a critical threshold, the local operations are suspended for the day. Any scheduled overnight science experiments are continued or activated.

5. Rover proceeds to ready all the data acquired for the evening relay to the orbiter. This could involve compression of the data to fit the constraints of the communication link.

6. Rover goes to sleep with only the communications receiver active to receive the "wake-up" call from the orbiter.

7. Rover wakes up at the late evening orbiter pass and transmits the requested data, and then goes back to sleep.

8. Rover wakes up early in the next morning for the next orbiter pass and transmits the requested data, and then goes back to sleep to await daylight.

3.5.5 Payload Accommodation Issues to be Addressed in the Proposal

The proposal should identify the impact of the investigations on the rover systems in the following areas:

- Position or orientation of the rover with respect to the object being sampled. Does the rover need to be positioned or oriented in any special way?
- Control requirements for the sampling and/or instrument deployment system being proposed.
- Vibration amplitude and frequency while operating the payload..
- Is the chassis of the rover required to exert force during sampling or instrument deployment? If so, how much force, in what direction, with what degree of accuracy?

3.6 Planetary Protection

NASA's planetary protection policy, as established by NPD 8020.7E and consistent with the Committee on Space Research (COSPAR) objectives, is that planetary protection measures be undertaken in order to control contamination of other planets by terrestrial microorganisms and organic constituents during planetary missions and protect Earth from potential hazards in a returned sample. Under the existing policy, missions to the surface of Mars, such as the Mars 2001 Rover and Lander, must exercise extreme care to avoid such contamination of the planet's surface.

The current NASA requirements are being rewritten as a NASA Policy Guideline 8020.12B. For purposes of the AO, it is assumed that the lander spacecraft will be a Category IV-A mission and the rover will be a Category IV-B mission (because of the planned Earth return of the collected sample). See "Planetary Protection Provisions for Robotic Extraterrestrial Missions" document NPG 8020.12B for category details.
All lander and rover instruments proposed in response to this AO must be compatible with the Planetary Protection measures required for the mission. The final arbiter of what constitutes life detection is NASA's planetary protection officer.

3.6.1 Lander

It is assumed that the lander will be a Category IV-A mission as was Mars Pathfinder and Mars Surveyor 1998. Therefore, it is anticipated that maximum bioburden requirement will be similar to the Mars Pathfinder. The procedures in place for the Mars Pathfinder approach should achieve a total exposed surface bioburden of $3x10^5$ aerobic spores and 300 spores/m^2 average on exposed surfaces at launch. The implementation model planned for achieving the yet-to-be-specified burden requirement is as follows:

a. All components of any lander module and/or lander subsystem, including science payload, will be precision cleaned with isopropyl or ethanol alcohol and bagged prior to their use at the next level of assembly.
b. The modules and/or subsystems will be assembled in a clean room environment, Class 100,000 or cleaner, and with the most stringent personnel garmenting and control requirements.
c. A statistically valid number of bioassay samples will be taken from the interior of each module and/or subsystem.
d. The module and/or subsystem exterior surfaces will be cleaned by wiping, bioassayed, and bagged at the completion of that assembly level (after any testing is complete).
e. The lander (without aeroshell subsystem) will be assembled in a class 100,000 or cleaner clean room environment. During any testing at this level of assembly, the interior of all modules that are not enclosed must be protected from contamination (e.g. temporary covers). After any such testing is completed a bioassay will be taken. Only if there is contamination will the lander be disassembled and then only to the requisite level. The disassembled portions will be recleaned and bioassayed, all in a Class 100,000 environment.
f. Following testing and reassembly, if required, the lander instruments are then installed. Then a final statistically valid number of bioassays of exterior exposed surfaces (at this level of assembly) will be taken just prior to the next step.

g. The lander will be encapsulated into the aeroshell subsystem (previously cleaned and bioassayed). After this assembly and any testing are completed, the exterior surfaces of the aeroshell subsystem will be cleaned by wiping and a statistically valid number of bioassays will be taken. This configuration will be considered "the lander in its biobarrier." No further disassembly should be planned, and as a contingency, this step would be repeated, possibly after a repeat of the previous step.
h. The biobarrier will be "sealed" in a cover and will use GN_2 purge to protect its exterior surfaces from contamination during transportation to the launch site and launch vehicle integration.

3.6.2 Rover

The MSP 2001 rover will collect samples of martian surface materials for possible return to Earth laboratories in the 2005 opportunity. Upon return, they will be intensively investigated for evidence of the existence of past or present life. Therefore, this mission is considered to carry life detection instrumentation and has been Classified IV-B under NASA Planetary Protection requirements. For the purpose of this classification, a life detection experiment is defined as one whose life detection scientific objectives may be compromised by terrestrial biological contamination.

Recent reports of the Space Studies Board (SSB) of the National Research Council recommended that landers carrying instrumentation for in situ investigation of extant Martian life should be subject to at least Viking-level sterilization. This implies that the goals of the Viking process are still required, but that the specific implementation can be different. Specific methods of sterilization are to be determined. Essentially all external interfaces must achieve Viking level sterilization. During Phase A/B, the project will work with the NASA Planetary Protection Officer to define the specific procedures to be used.

As a point of reference for consideration of instrument design, the Viking sterilization process is described here. Viking sterilization was conducted in a two step process:

1) The Viking Lander was assembled in a Class 100,000 clean room, cleaned of organics, and assayed by NASA standard microbiological procedures (NHB 5340.1B) to demonstrated an average aerobic bacterial spore burden of < 300 spores per square meter and quantitatively and qualitatively assess the vegetative single cell organisms..

2) Following cleaning and assay, the Viking Lander was subjected to a terminal dry heat sterilization cycle of approximately 30 hours at a temperature of 111.7 degrees C and a specified humidity of 1.3 mg/L. Dry heat remains the method of choice for spacecraft. Its effectiveness and specifications for use have been demonstrated and are well documented.

Integrated instrument suites capable of compliance with the dry heat sterilization requirement will be considered to be in total compliance with the previous Viking level requirements. The limitation this places on instrumentation options is recognized, however, and thus integrated instrument suites may be proposed which are capable of meeting the first cleaning step of the procedure for the electronics portion of the proposed instrument suite. The current rover design intends to accommodate that portion of the instrument suite's electronics which can be isolated from the external environment within the warm environment box (WEB). The content of the WEB will be sealed and isolated from the Martian environment except for electrical and optical access. Pending approval by the NASA Planetary Protection Officer it is anticipated that the contents of this box will not be subjected to dry heat sterilization, but rather to an as yet to be determined surface sterilization method.

Any portion of the experimental device outside the WEB must be capable of enduring dry heat sterilization unless it can be demonstrated by the investigator that its placement on the rover will not risk compromising the sterility of the samples.

While proposers are invited to propose methods for cleaning and sterilization of the external portion of the instrument suite, it however, should be recognized that any alternative methods of sterilization will require documentation of effectiveness provided by the proposer.

Following final cleaning, sterilization and assay by the project, the instrument suite will not be made available for further planned access. The investigator should specify the latest planned access to the hardware and can expect to be held to this commitment. Late access to the rover will not be permitted Proposers should address issues such as shelf life of reagents and sensor elements, and any post-integration calibration needs for the proposed instrument suite It is recognized that last minute emergency access to the instrument suite may be required, and proposers should address this issue in their proposals.

Pending approval by the NASA Planetary Protection Officer it is anticipated that proposers may propose four options for placement of instrument suites that will meet the Planetary Protection Requirements with one of the following:.

Option 1) The instrument suite may be mounted external to the WEB, provided it is capable of undergoing the Viking level cleaning and dry heat sterilization procedures outlined above.

Option 2) The instrument suite may be mounted entirely internal to the WEB and exempted from the terminal dry heat sterilization step, provided it is not in direct physical contact with the martian environment. One example of such an mounting would be one in which all electronic elements are internal to the WEB and a fiber optic cable acts as a view port to the martian environment.

Option 3) The instrument suite may be mounted entirely external to the WEB provided the proposer provides a containment box, which meets or exceeds the level of containment achieved by the rover WEB and provided the external surface is sterilized.

Option 4) The instrument suite mounting may be a hybrid of the above three options. One example of mounting would be one in which heat sterilizable detectors (option 1) were linked, via project supplied optical or electronic connectors crossing the WEB wall, to electronics housed within the WEB that are expected to be exempted from heat sterilization.

3.7 Lander Environments

The preliminary environment descriptions which follow are subject to modification as the design matures. The values quoted are current best estimates which will allow the proposer to determine the scope of the effort required to design for safety, quality and proper function. Environments are presented by mission phase and are not repeated unless unique or maximum limits are established in the mission phase.

The following environment descriptions represent the maximum envelope expected to be encountered from ground transportation on Earth to operations at Mars. In addition to the actual flight environments, system level thermal vacuum, shock, acoustic, and EMI/EMC tests of the integrated payload and spacecraft will be performed at levels somewhat higher than the maximum expected flight environments. Any of these maximum test and environment levels may be design drivers for a proposed payload. The proposer is responsible for recognizing which environments are applicable and developing a plan for a payload package which will, at a minimum, be safe to handle and transport and will not interfere with the operation of the launch vehicle, spacecraft or other instrument or payload systems.

Integrated spacecraft thermal vacuum, shock, acoustic and EMI/EMC testing, which includes payloads, is performed at protoflight levels. Protoflight levels are usually somewhat higher than what the actual expected environments are. Protoflight implies that system-level qualification for flight is being performed on the actual flight spacecraft - as opposed to a dedicated test article.

The proposer should design with protoflight test levels in mind. For example, if the minimum nighttime temperature at the proposed Mars landing site was expected to be -100° C, with ±10°C uncertainty, the low temperature protoflight test level would be set at -100° - 10° = -110°C. The proposer could choose to add an additional payload or component qualification margin beyond this. Development test levels are left to the discretion of the proposer; these should be included in the proposal.

Table 3.7.a summarizes the mission phases, their associated environments and their location in this document.

Table 3.7.a Summary of Environments

Mission Phase	*Environment*	*Document Sections*
Pre-Launch	a. Shipping and Transportation Vibration b. Shipping and Transportation Shock c. Thermal/Humidity d. Explosive Atmosphere e. Electromagnetic Compatibility	3.7.1
Launch	a. Acoustics b. LV Steady State & Transients c. Sinusoidal Vibration d. Random Vibration e. Shock f. Launch Pressure	3.7.2
Cruise	a. Cruise Thermal/Vacuum b. Charged Particles/Radiation c. Micrometeoroids	3.7.3
Mars Aeroentry, Descent and Landing	a. Deceleration and Landing Shock	3.7.4
Mars Surface Operations	a. Thermal Radiation b. Thermal c. Shock d. Charged Particles/Radiation e. Contamination by Descent Engines	3.7.5

3.7.1 Pre-Launch

It is the responsibility of the selected Principal Investigators proposer to deliver their payload package and required support equipment, including sensor simulators if required, to the assembly, test and launch operations (ATLO) site [Denver]. Any subsequent shipment will be carried out by the spacecraft contractor, and will use the following environment guidelines. Proposers should factor these environments into their design as appropriate.

a. Shipping and Transportation Vibration

Shipping containers should be designed such that the contents are subjected to a vibration environment less severe than that of the launch vehicle (Section 3.7.2).

b. Shipping and Transportation Shock

Flight assemblies in shipping containers should be subjected to shock environments less severe than those experienced due to launch and pyroshock.

c. Thermal/Humidity

The controlled thermal environment for ground handling, transportation and storage subsequent to payload delivery is shown below.

Control Parameter	Low Limit	High Limit
Temperature	+5°C	+45°C
Temperature Change	-5°C/hr	+5°C/hr
Pressure	$7 \times 10^{+4}$ N/m^2 (520 torr)	$1 \times 10^{+5}$ N/m^2 (760 torr)
Relative Humidity	≥ 30% (Could be as low as 0% during shipping or storage)	≤ 70%

The pre-launch checkout ends 4 days prior to launch, however, no external thermal control will be available during this time (except for fairing air conditioning) until the vehicle is powered up in the launch sequence.

d. Explosive Atmosphere

All hardware shall be designed to operate without igniting an explosive atmosphere existing within the pressure, temperature and auto ignition ranges stated below.

Physical Characteristics	Range
Pressure	$1.3 \times 10^{+4}$ to $10.6 \times 10^{+4}$ Pa (100 to 800 torr)
Temperature	5° C to 45° C
Auto-Ignition Temperature	350° C to 750° C
Chemical Constituents	Hydrogen (fuel) and air (oxidizer) combined in any potentially explosive mixture ratio

e. Electromagnetic Compatibility

The Proposer's flight payload, electronics, and electrical ground support equipment (if any) must be compatible with MIL-STD-461C Part 3 Conducted and Radiated (Emissions and Suscepibility) . The Following Electromagnetic Compatibility values are applicable for proposers with respect to their payload. These values reflect ground and spacecraft generated disturbances throughout all mission phases:

Physical Characteristics	Values
Conducted Emissions Spikes, DC Power	MIL-STD-461 Part 3 CE07
Conducted Emissions Antenna Conducted	MIL-STD-461C Part 3 method CE06
Conducted Emissions Power Quality: Ripple and Noise	CE01 and CE03 shall be measured on the Proposer's equipment and shall comply with the limits in figure 3.7.1-1.
Radiated Emissions Electric Fields Narrowband	Unintentional radiated narrowband electric field levels produced shall not exceed the levels specified in Figure 3.7.1-2
Radiated Susceptibility	The Proposer's equipment shall operate without degradation of performance when subjected to the electric field strengths shown in Figure 3.7.1-3.
Conducted Susceptibility Rejection of Undesired Signals	MIL-STD-461C Part 3 method CS04
Conducted Susceptibility Switching Transient	MIL-STD-461C Part 3 method CS06. The switching transient shall not exceed 60 volts line to line or 30 volts line to chassis.

Conducted Susceptibility Narrowband Ripple (Unregulated Bus)	MIL-STD-461C methods CS01 and CS02 as tailored in Figure 3.7.1-4.
Surge Voltage	for a minimum of 5 minutes per polarity, as indicated in figure 3.7.1-5.

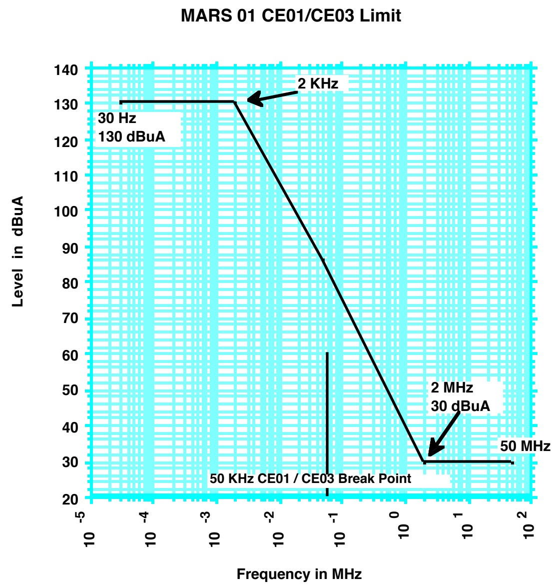


Figure 3.7.1.a, CE01 / CE03 Limits Narrowband

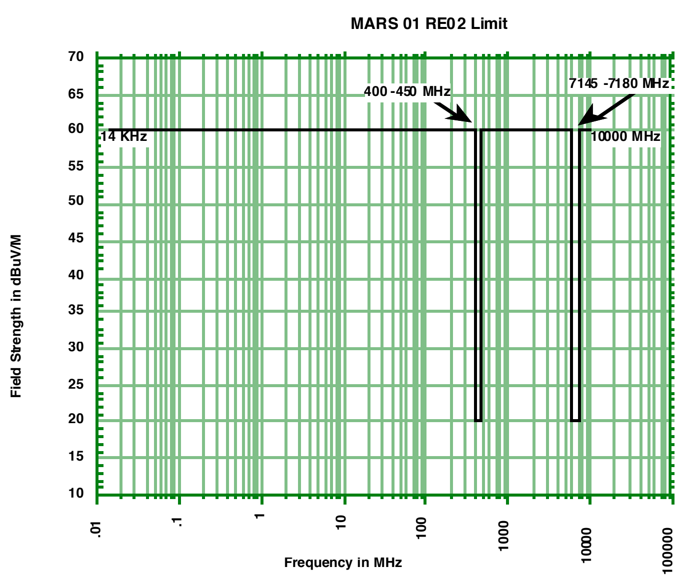


Figure 3.7.1.b, Limit for Radiated Emissions Electric Fields, Narrowband

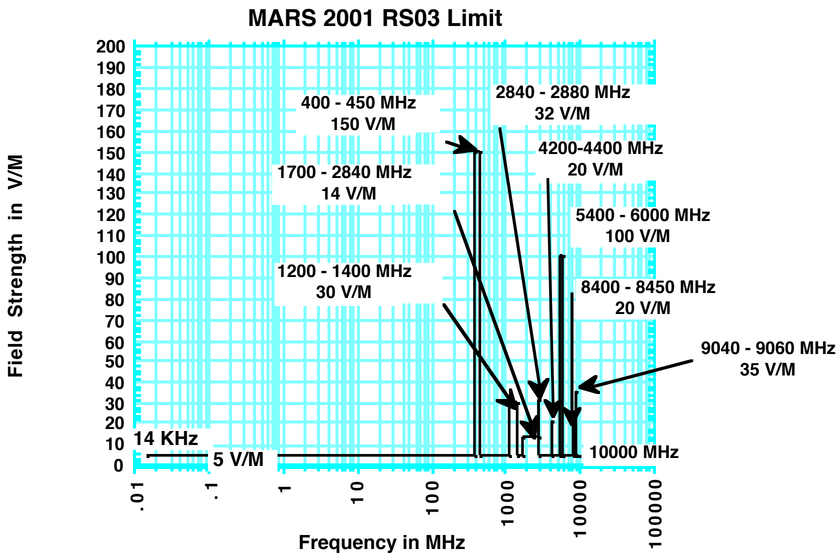


Figure 3.7.1.c, Radiated Susceptibility Limits

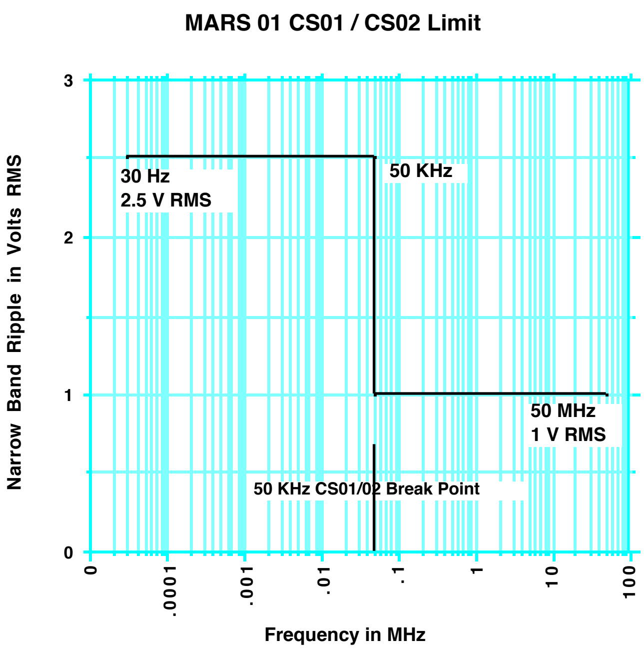


Figure 3.7.1.d, CS01 / CS02 Narrowband Injected Ripple

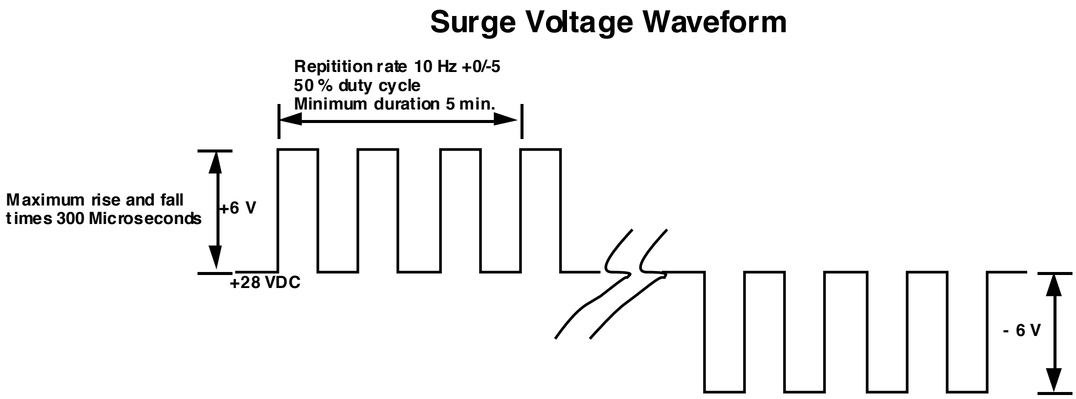


Figure 3.7.1.e Surge Voltage Waveform

3.7.2 Launch

The Med-Lite launch vehicle launch environments are based on the performance requirements specified in the Delta II 7425 Data Book.

a. Acoustics

Acoustics is a component level design requirement for high surface area components. The maximum acoustic environment in the launch vehicle fairing occurs during liftoff and transonic flight, the flight duration at this maximum level is 10 seconds. The protoflight test will last for 60 seconds. The overall sound pressure level (OASPL) for qualification will be 138.4 dB. For protoflight testing add 4 dB to get 142.4 dB OASPL. The extent to which the aeroshell muffles (or amplifies) acoustic energy has not been determined and should not be considered at this time. The pullback of blankets to accommodate the 2.65 m aeroshell may add 3db to the spectrum. Figure 3.7.2a shows the protoflight levels.

Acoustic Spectrum for Delta 7425

One-Third Octave Center Band (Hz)	Max Sound Press
31.5	123.4
40	126
50	130.5
63	129.6
80	130.7
100	131.3
125	131.8
160	131.9
200	132.5
250	132.6
315	132.6
400	131.7
500	130
630	127.5
800	124.5
1000	120.5
1250	118
1600	115.5
2000	113
2500	111.8
3150	110.5
4000	108.5
5000	108
6300	107
8000	106.5
10000	106.5

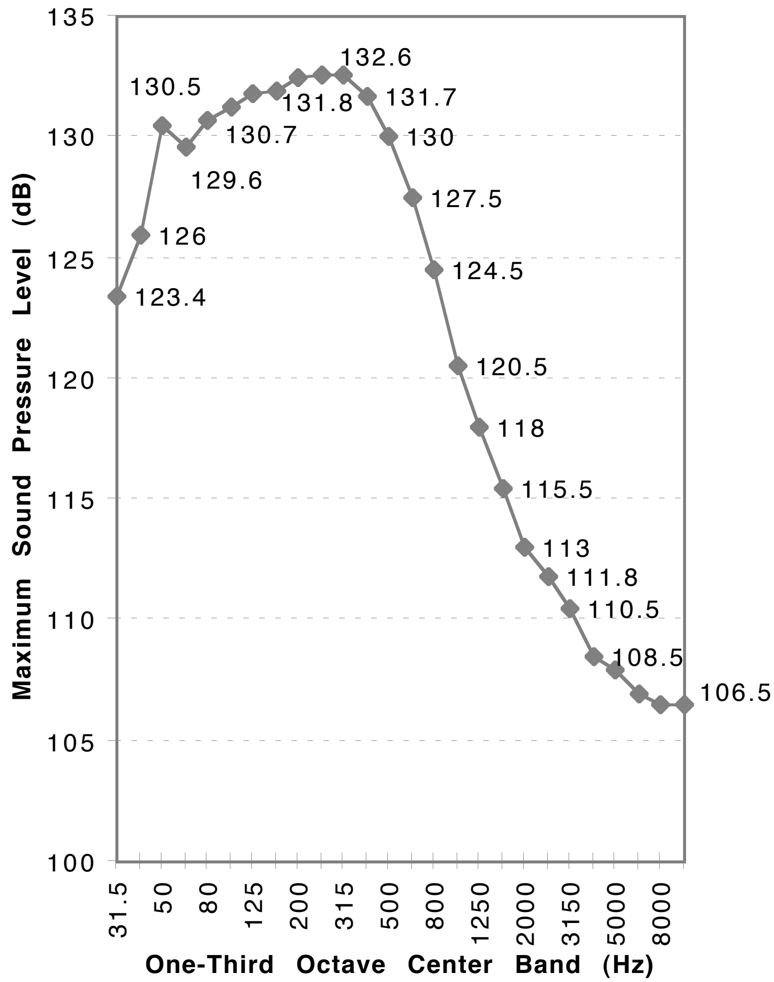


Figure 3.7.2.a Acoustic Environment

b. LV Steady State & Transients

The maximum quasi-steady state loads that will be induced by the launch vehicle are shown below.

Axis[3]	Liftoff/Transonic	Stage III Flight
Lateral[4]	±4.3(7)	±0.2
Thrust	+1.3±2.1 (3) (1)	+9.7[5,6]

(1) Loads are applicable at the spacecraft center of gravity.
(2) Limit load factors should be multiplied by a 1.25 factor to obtain ultimate loads.
(3) Plus indicates compression load and minus indicates extension load.
(4) Lateral load factor to provide correct bending moment at the spacecraft to PAF interface.
(5) 3-sigma steady state acceleration for a 637 kg (1405 lb) spacecraft.
(6) Third stage spin-induced loads should be added to these values. [70±7 rpm spin rate]
(7) Based on results of initial coupled loads analysis. May change subject to subsequent coupled loads analyses.

Spacecraft Design CG Limit-Load Factors, Including 1.4 Uncertainty Factor (g)[1,2]

Component loads and the corresponding secondary and tertiary support structure design loads are determined from the worst case launch (and/or EDL) environments.

c. Component Loads

Initial design limit load factors for components may be developed from combination of the spacecraft CG limit load factors and random vibration environments, or taken from the mass acceleration curve (MAC) shown in Figure 3.7.2.b. As noted in the figure, the MAC load factor (based on the mass of the component) is applied in the worst case direction simultaneously with a 2.8 G load in the liftoff direction (typically spacecraft Z axis).

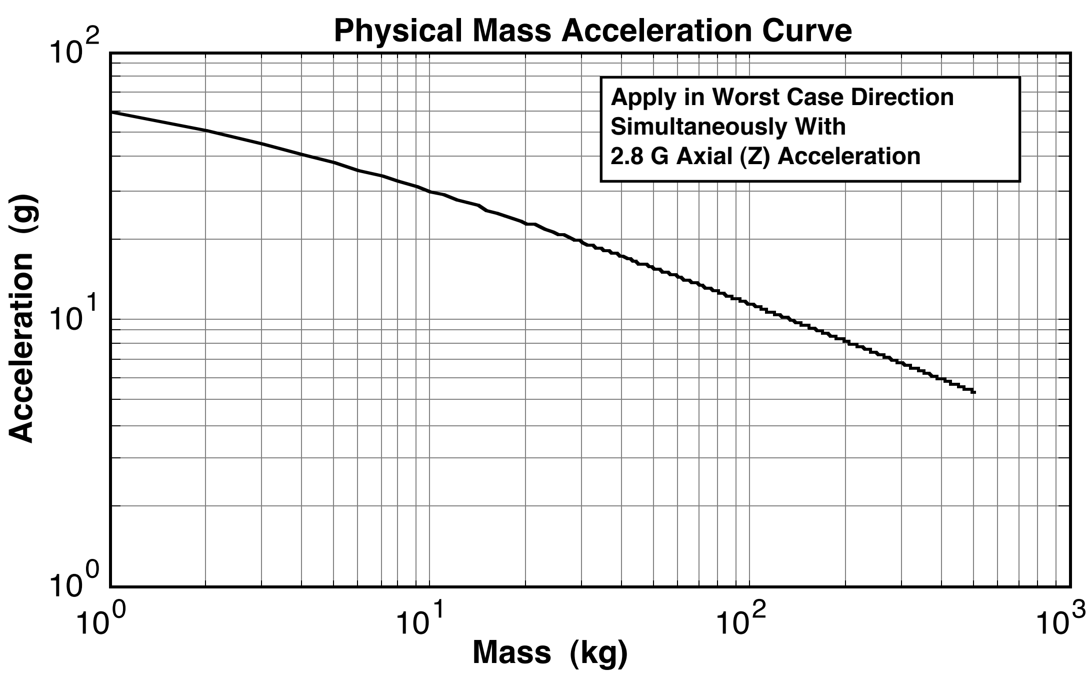


Figure 3.7.2.b Component Mass Acceleration Curve

c. Sinusoidal Vibration Levels

High energy sine survey and dwell testing are currently not planned at the system level on the protoflight vehicle. Acoustic testing and a modal survey will be performed.

d. Random Vibration

Random vibration levels (derived from acoustic and launch vehicle transient vibration spectrums) that will be experienced at the payload science deck during protoflight testing are shown in Figure 3.7.2.c.

Lander Top deck

Frequency	**Protoflight**	**Acceptance**
(Hz)	(g2/Hz)	(g2/Hz)
20	0.045	0.018
50	0.280	0.111
800	0.280	0.111
2000	0.045	0.018
Grms	18.7	11.8

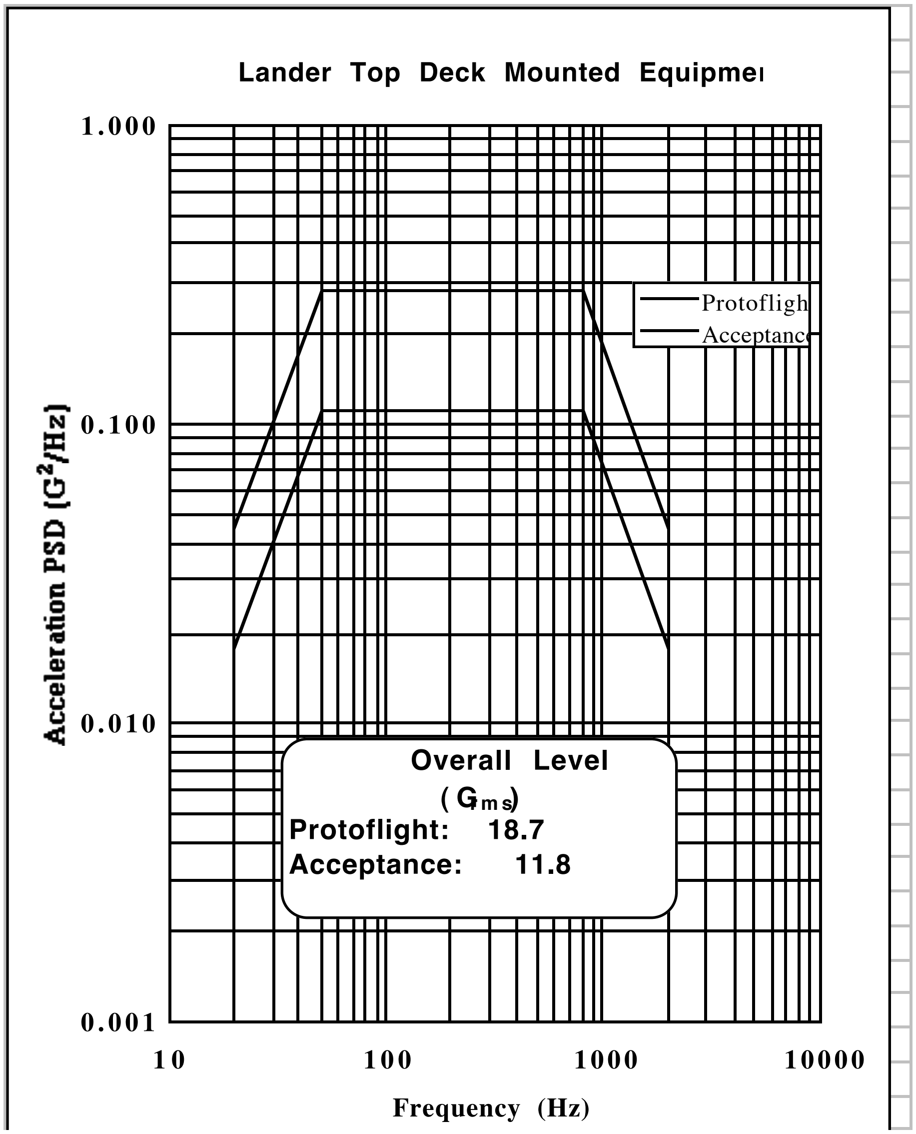


Figure 3.7.2.c Payload Protoflight Random Vibration Environment

e. Shock

Pyroshock testing is planned for the integrated spacecraft/payload configuration during the assembly, test and launch operations (ATLO) phase and will be performed by the spacecraft contractor. The graph in Figure 3.7.2.d shows an estimated worst case pyro shock environment at the payload science deck.

Frequency Hz	Protoflight g	Acceptance g
100	80	50
2000	2100	1300
10000	2100	1300

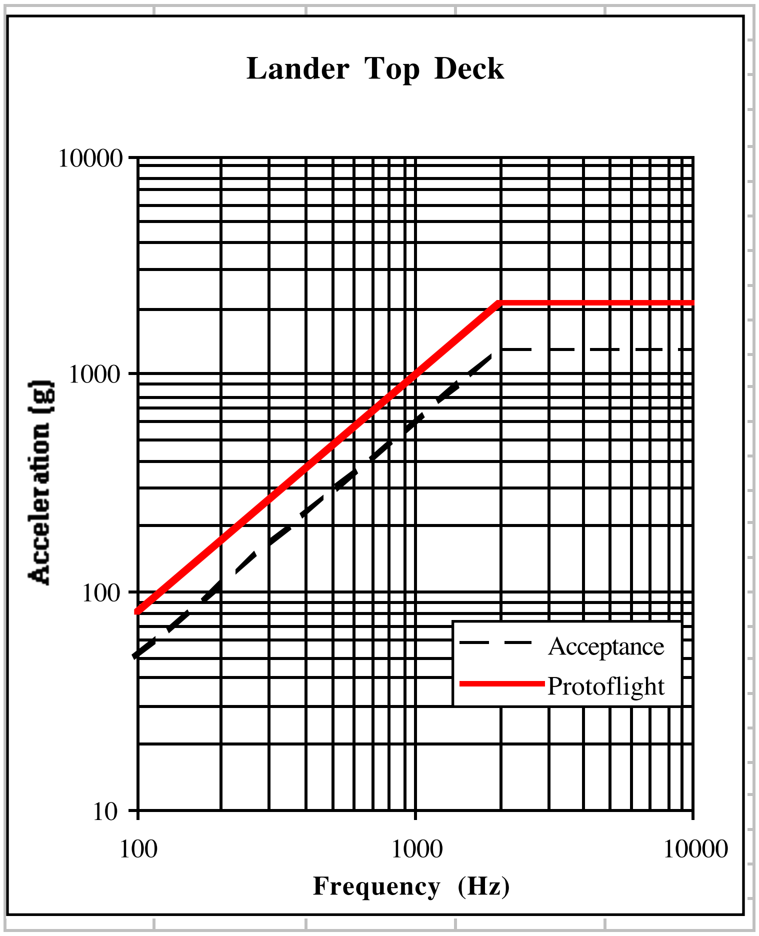


Figure 3.7.2.d Pyroshock Environment

f. Launch Thermal

The Launch phase thermal conditions are expected to be encompassed by the pre-launch vehicle environments. During the launch phase, the Delta Fairing Temperature ranges from 40° to 60° C. The heat flux at fairing separation (Free Molecular Heating) drops below 1135 w/m^2.

g. Launch Pressure

The launch pressure requirements are applicable to the spacecraft for venting purposes. Spacecraft components shall be compatible with a launch pressure decay of at least 45 ± 5 torr/sec for at least 10 seconds. Spacecraft components shall be compatible with an Earth orbit pressure of < 1 x 10^{-2} N/m^2 (0.75 x 10^{-4} Torr). This rate profile may occur anywhere from a one atmosphere condition (i.e. near liftoff) to a 10% atmosphere condition depending on the launch vehicle trajectory. During launch thru spacecraft ejection, the design pressure within the aeroshell is expected to decrease from 760 torr to 10^{-13} torr.

3.7.3 Cruise

a. Cruise Thermal/Vacuum

The design pressure for the mission will decrease from 1.0 x 10^{+5} Pa (760 torr) to 1.3 x 10^{-11} N/m^2 (10^{-14} Torr) in space.

The expected temperatures extremes inside the aeroshell during cruise for payloads mounted on the top deck are -135° C to -70°C. The integrated spacecraft and lander thermal vacuum testing on the protoflight spacecraft, which includes the flight science payload package, will be performed to temperatures of -145°C to +50°C inside the spacecraft thermal enclosure. This represents the normal extremes which will be experienced throughout the mission including surface operations (inside the latitude and time-of-season constraints of the lander).

The backshell should be assumed to have a surface emissivity greater than 0.8. The deck, which represents the mounting surface, and thus both a radiative and a conductive interface, should be assumed to have an effective surface emissivity less than 0.1.

b. Charged Particles/Radiation

Table 3.7.3.a describes the heavy-ion peak flux vs. linear energy transfer (LET) through aluminum shielding at l AU for the 95th percentile expected flare.

Table 3.7.3.a Flare Integral Flux Environment

LET Aluminum Shielding (particles/cm^2-day)

Peak integral heavy ion flux at 1 AU for a Single 99th Percentile Flare; behind aluminum shielding (spherical geometry). Note that no radiation design margin (RDM) has been included in any of the environments.

LET (MeV-cm^2/mg)	Flux (cm^{-2} day^{-1}) behind Al Spherical Shield Thickness (mils)					
	Zero mils	60 mils	100 mils	250 mils	500 mils	1000 mils
1.00	1.352e+08	1.565e+06	2.01e+05	4.71e+04	4.40e+03	2.57e+02
1.26	1.323e+08	8.589e+05	1.00e+05	2.17e+04	1.96e+03	1.13e+02
1.58	1.302e+08	2.682e+05	1.89e+04	1.68e+03	5.64e+01	8.10e-01
2.00	1.289e+08	2.031e+05	1.38e+04	1.14e+03	3.79e+01	5.40e-01
2.51	1.136e+08	1.457e+05	9.66e+03	7.46e+02	2.43e+01	3.28e-01
3.13	1.109e+08	8.966e+04	5.70e+03	4.25e+02	1.32e+01	1.69e-01
3.98	9.149e+07	6.002e+04	3.73e+03	2.75e+02	8.46e+00	1.08e-01
5.01	7.947e+07	5.597e+04	2.03e+03	1.36e+02	3.78e+00	4.11e-02
6.31	7.087e+07	1.750e+04	9.73e+02	5.82e+01	1.51e+00	1.53e-02
7.94	3.319e+07	7.067e+03	3.43e+02	1.36e+01	2.29e-01	1.66e-03
10.00	2.537e+07	3.976e+03	1.76e+02	6.09e+00	9.42e-02	4.97e-04
12.59	1.783e+07	2.008e+03	7.85e+01	2.04e+00	2.09e-02	8.13e-05
15.85	9.761e+06	5.856e+02	2.08e+01	4.49e-01	3.04e-03	7.34e-06
19.95	8.908e+06	3.079e+02	1.04e+01	2.17e-01	1.32e-03	2.82e-06
25.12	5.550e+06	1.057e+02	3.57e+00	7.38e-02	4.38e-04	9.04e-07
31.62	1.149e+03	2.518e-02	7.44e-04	7.91e-06	3.90e-08	6.02e-11
39.81	1.291e+02	1.923e-03	4.08e-05	3.00e-07	8.48e-10	6.42e-13
50.12	5.055e+01	6.429e-04	1.17e-05	6.74e-08	1.37e-10	5.14e-14
63.10	2.036e+01	1.372e-04	2.21e-06	1.11e-08	1.86e-11	6.92e-15
79.43	1.130e+00	5.403e-06	6.57e-08	1.66e-10	1.67e-13	2.15e-17
100.00	4.332e-02	1.433e-07	1.58e-09	2.51e-12	1.75e-15	7.95e-20

110.00	1.000e-20	1.000e-20	1.00e-20	1.00e-20	1.00e-20	1.00e-20

Table 3.7.3.b shows the average annual omnidirectional fluence of protons during cruise phase (assumed one year cruise, 1 AU data scaled to 1.37 AU average distance during cruise).

Table 3.7.3.b. Dose vs Depth Data for the Mars Lander Cruise Phase

		Total Ionizing Dose (rads - si) (RDM=1)
Thickness (mils Aluminum)	g/cm^2	Lander Cruise Phase
1.458E+00	1.000E-02	8.506E+04
1.836E+00	1.259E-02	6.911E+04
2.311E+00	1.585E-02	5.635E+04
2.909E+00	1.995E-02	4.726E+04
3.663E+00	2.512E-02	3.897E+04
4.611E+00	3.162E-02	3.312E+04
5.805E+00	3.981E-02	2.770E+04
7.308E+00	5.012E-02	2.201E+04
9.201E+00	6.310E-02	1.786E+04
1.000E+01	6.858E-02	1.579E+04
1.158E+01	7.943E-02	1.356E+04
1.458E+01	1.000E-01	1.100E+04
1.836E+01	1.259E-01	8.931E+03
2.001E+01	1.372E-01	8.028E+03
2.311E+01	1.585E-01	7.018E+03
2.909E+01	1.995E-01	5.742E+03
2.999E+01	2.057E-01	5.300E+03
3.663E+01	2.512E-01	4.312E+03
4.000E+01	2.743E-01	3.892E+03
4.611E+01	3.162E-01	3.456E+03
5.000E+01	3.429E-01	3.057E+03
5.805E+01	3.981E-01	2.658E+03
6.000E+01	4.115E-01	2.456E+03
7.001E+01	4.801E-01	2.185E+03
7.308E+01	5.012E-01	2.015E+03
7.999E+01	5.486E-01	1.824E+03
9.000E+01	6.172E-01	1.653E+03
9.200E+01	6.310E-01	1.552E+03
1.000E+02	6.858E-01	1.425E+03
1.158E+02	7.943E-01	1.255E+03
1.200E+02	8.230E-01	1.148E+03
1.400E+02	9.601E-01	1.015E+03
1.458E+02	1.000E+00	9.410E+02
1.600E+02	1.097E+00	8.613E+02
1.799E+02	1.234E+00	7.709E+02
1.836E+02	1.259E+00	7.177E+02
2.001E+02	1.372E+00	6.592E+02
2.200E+02	1.509E+00	6.167E+02
2.311E+02	1.585E+00	5.848E+02
2.400E+02	1.646E+00	5.476E+02
2.600E+02	1.783E+00	5.077E+02

2.800E+02	1.920E+00	4.620E+02
2.909E+02	1.995E+00	4.375E+02
2.999E+02	2.057E+00	4.147E+02
3.663E+02	2.512E+00	3.594E+02
4.000E+02	2.743E+00	3.169E+02
4.611E+02	3.162E+00	2.754E+02
5.001E+02	3.430E+00	2.467E+02
5.805E+02	3.981E+00	2.132E+02
7.308E+02	5.012E+00	1.691E+02
9.200E+02	6.310E+00	1.318E+02
1.158E+03	7.943E+00	1.042E+02
1.458E+03	1.000E+01	7.921E+01

Table 3.7.3.c. Integral Solar Proton Fluence for the Mars Lander Cruise Phase

	Integral Fluence (protons/cm^2) (RDM=1)
Energy (MeV)	Cruise Phase
1.000E+00	8.740E+10
4.000E+00	2.926E+10
1.000E+01	1.130E+10
3.000E+01	3.096E+09
6.000E+01	1.391E+09
1.000E+02	7.727E+08
1.500E+02	4.828E+08
2.000E+02	3.464E+08
3.000E+02	2.169E+08
5.000E+02	1.204E+08
1.000E+03	5.383E+07
2.000E+03	2.425E+07
3.000E+03	5.329E-01

Notes:
1. Derived from JPL solar flare model using 95% confidence level.
2. Cruise to Mars was modeled by scaling data from 1 AU to 1.37 AU average distance ($1/r^2 = 0.533$).
3. Fluence is external to any shielding.
4. Yearly proton fluence accumulated on the Martian surface is negligible.

Total ionizing dose (TID) radiation environment: The total dose (99th percentile) behind 2.54 mm (0.100 inch) of aluminum shielding is 1.43 krads (Si) per year of cruise with an RDM of 1.

In addition to the naturally occurring dose, the rover is expected to have 5 radioisotope heating units (RHU's) that could impact the radiation monitor. The nominal mission profile does not permit operation of any instruments until after rover deployment, so this dose would be while the instrument was not operating.

3.7.4 Mars Aeroentry, Descent and Landing

a. Deceleration and Shock

Maximum entry steady state deceleration loads will be less than 25 G. Note that these loads are opposite in direction to the maximum launch loads. Maximum parachute snatch loads are expected to be below 15 G. Maximum surface touchdown loads are expected to be less than 12 G. Maximum induced rates at touchdown are expected to be below 250 rad/s^2.

The aeroentry and landing shock environments (parachute tractor rocket firing and leg deployment) are expected to be encompassed by the launch environments described in Section 3.7.2.

3.7.5 Mars Surface Operations

The environment of Mars will be only briefly described in this document,. An extensive description of the Mars atmosphere, including pressure, temperature, density, winds and dust storm effects as a function of longitude, latitude, altitude, time-of-season and time-of-day, can be found in “Mars Global Reference Atmosphere Model (MarsGram) Release #2”, by C.G. Justus of the Georgia Institute of Technology and Bonnie F. James of the Marshall Space Flight Center, a technical report dated 3-1-93. It is important to note that Mars’ 6 to 10 mBar atmosphere must be taken into account in instrument design, to avoid potential corona discharge. A fully referenced description of Mars surface properties, including chemical and physical properties, thermal inertia, dielectric constants, soil temperature, charge particle radiation and solar insolation, can be found in NASA Technical Memorandum TM 100740, “Environment of Mars, 1988”, October 1988 and TM 108513 “A Revised Thermosphere for the Mars Global Reference Atmospheric Model (Mars-GRAM version 3.5). A collection of information specific to sand and dust on Mars is contained in NASA Conference Publication 10074, “Sand and Dust on Mars”, February 1991. Other payload design considerations regarding the atmosphere include convection heat transfer and wind loading.

a. Thermal Radiation

The thermal radiation environment at Mars is shown in Table 3.7.5.a. A dust optical depth factor (extinction factor) of 0.2 applies for a nominal day. On a clear day the optical depth approaches 0. In a dust storm, the optical depth may be >>0.2.

Table 3.7.5.a Surface Thermal Radiation Environment

THERMAL RADIATION	Perihelion	Aphelion
Direct Solar:	710.0 W/m^2	490.0 W/m^2
Albedo	0.33 W/m^2	0.25 W/m^2

b. Thermal

Atmospheric Temperatures on Mars may range from -123° C to 0° C. Payloads mounted on the top deck may expect to follow temperature profiles of the environment. The lander enters “survival mode” where the lower limit of the lander thermal range cannot be controlled. Before the lander enters survival mode the payload systems will be powered down as required in order to conduct basic lander functions. Once the lander enters survival mode, no power is available to the payload.

The total day and nighttime power and energy budgeted to the payload is described in Section 3.3. Heater power required on the surface for thermal control of payload equipment outside the lander thermal enclosure, shall be taken from the payload power budget.

Thermal control of payload equipment outside of the thermal enclosure is the responsibility of the proposer and must be factored into the mass and power budgets of the payload. This would include thermistors, heaters, thermostats, insulation, coatings, etc.

c. Shock

The surface shock environment is expected to be encompassed by the launch vehicle environment described in Section 3.7.2. All shock events such as propulsion system deactivation and solar array deployment will take place prior to activation of the surface payload.

d. Charged Particles/Radiation

The Mars surface charged particle/radiation environment is encompassed by the Cruise environment tables presented in Section 3.7.3. As an approximation, these values may be multiplied by 0.444 to account for Mars surface operations at 1.5 AU.

Total ionizing dose (TID) radiation environment: The total dose (99th percentile) behind 2.54 mm (0.100 inch) of aluminum shielding is 7 rad (Si) per Earth year of surface operations.

e. Descent Pressurization Rate

During descent, the ambient external pressure will increase from interplanetary hard vacuum (< 10-11 N/m2 or 10-14 torr) to the mars surface pressure of less than 1300 N/m2 (10 torr). Maximum repressurization rate is < 130 N/m2 sec (1 torr/sec).

f. Contamination by Monopropellant Engines

The lander will employ a hydrazine monopropellant propulsion system to effect a soft landing. The engines will be active until touchdown. Products of decomposition of the hydrazine monopropellant, shown in Figure 3.7.5.a, will be exhausted by the engines and will impact and contaminate the landing site. The hydrogen and nitrogen will remain gaseous and will dissipate rapidly depending on the wind conditions. Most of the ammonia will be carried away by the wind as well, but a portion will be deposited on and entrained in the soil. A summary of the bulk composition of the hydrazine propellant prior to burning, and the net mass of contaminants released during final descent is shown in Table 3.7.5.b.

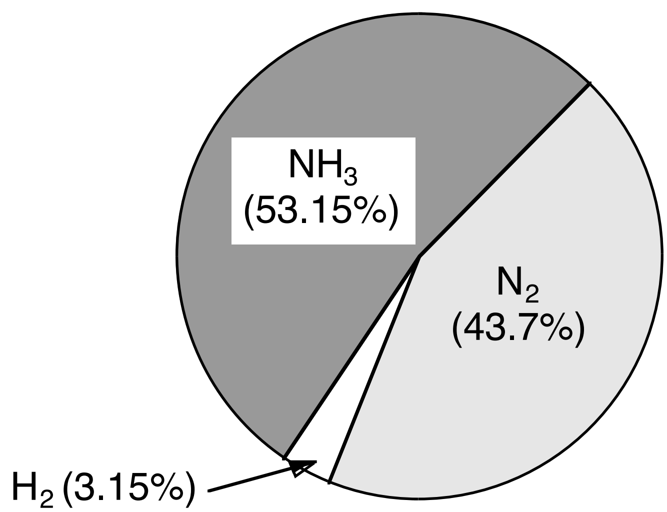


Figure 3.7.5.a Products of Decomposition of Hydrazine

The results of Viking lander engine tests conducted at White Sands are presented below, where the amount of ammonia deposited by the engines is shown both as a function of soil depth directly below the engine (Figure 3.7.5.b) and radial distance from the engine centerline (Figure 3.7.5.c).

This represents the best available experimental data on surface contamination and is considered representative of a '98 Mars Surveyor lander. Two important conclusions can be drawn from these. First, the surface level concentration of ammonia will probably be in excess of 100 ppm for a large radial distance (~30 m) from the lander. Second, for a soil porosity comparable to lunar soil (the test reference), a concentration of less than 100 ppm can be achieved at a soil depth of 1 cm, even directly below the engine.

Table 3.7.5.b Propellant Chemical Species

Chemical Species	Pre-Burn In Monopropellant**	Contaminants Released * Final Descent, kg
Major & Minor Constituents		
• Hydrazine (N_2H_4)	99.82%	0.20
• Ammonia (NH_3)	0.06%	0.0035
• Nitrogen Gas (N_2)	0%	0.0038
• Hydrogen Gas (H_2)	0%	0.0004
Trace & Ultratrace Constituents		
• Aniline	7 ppm	0.0000051
• Water (H_2O)	0.4%	0.00003
• Helium Gas (He)	<0.02	<0.0007
• Other Organics	0 ppm	0.0000000
• Nonvolatiles	2 ppm	0.0000015
• Particulates	0 mg/liter	0.0000000
• Chloride	0.2 ppm	0.0000001
• Iron	1 ppm	0.0000007
• Carbon Dioxide (CO_2)	17.3 ppm	0.0000126
• Silicon	0.01 ppm	0.0000000

*Only NH_3 Was Detected in Soils During Viking Contamination Tests
**Basis: Drum EPL-5024A (Analyzed 3/29/94 by Rocket Research)

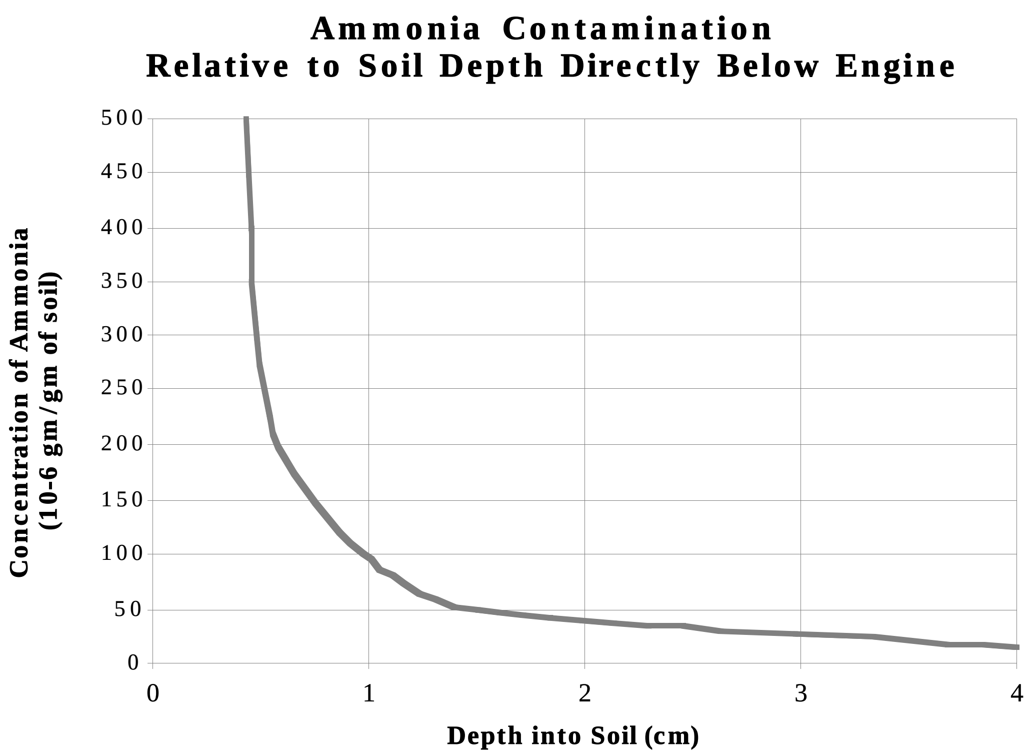


Figure 3.7.5.b Soil Contamination by Depth

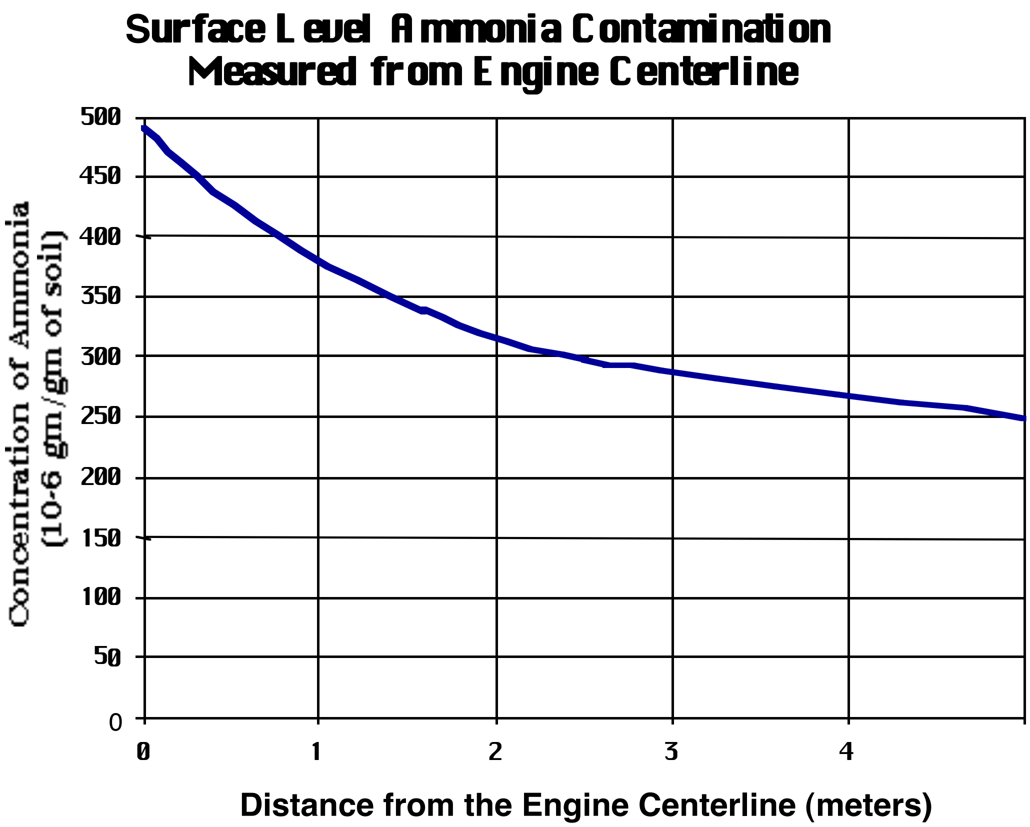


Figure 3.7.5.c Soil Contamination by Radial Distance

In addition to frozen ammonia, ammonia vapor may react with the CO_2 atmosphere to form solid ammonia carbonates which may be precipitated onto the surface. The extent to which this may occur has not been characterized. Trace amounts (<10 ppm) of other impurities may be deposited on the surface as well, including hydrogen cyanide and aniline (see Table 3.7.5.a), and propulsion system materials from the tanks, valves and engines. If the proposer's payload is sensitive to any of these contaminants, a plan should be included to mitigate their effects.

g. Cruise Heat Rejection System

The cruise heat rejection system contains ammonia and may freeze and rupture sometime after landing, releasing approximately 50 grams of ammonia (mostly gaseous).

3.8 Payload Integration

This section describes the payload integration process. All delivery dates are contained in Section 5.2.

A Payload Integration Working Group (PIWG) will be established soon after payload selection. Payload interface control documents (ICDs) will 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.).

After receipt of payload hardware in the dedicated payload staging area at the spacecraft contractor clean room facility, each instrument will be subjected to wipe sampling for microbiological assays to determine if bioburden levels meet requirements. If the payload component does not pass, limited cleaning will be performed by the accompanying instrument engineer. After planetary protection requirements have been passed, the instrument will be functionally checked using the instrument ground support equipment (GSE). Each instrument is stored in the bio-controlled area and installed in the lander at the appropriate entry point. Facility support at the spacecraft contractor's facility required for standalone and integrated operations shall be specified in the ICD.

Subsequent to science payload delivery, the payload supplier will be responsible for all aspects of the standalone payload verification prior to spacecraft installation. All instrument-unique test equipment, purge equipment and sensor simulators will be the responsibility of the payload supplier. Following installation onto the spacecraft by the spacecraft contractor, the instruments will be subjected to a series of system tests (as shown in Figure 3.8.a). These system tests include acoustic, thermal vacuum, EMI/EMC and pyro shock. In addition to system environmental tests, integrated system tests (ISTs) will be conducted. The IST will verify end-to-end mission critical events and functions utilizing flight sequences. To provide assurance for mission success, payload functions with the spacecraft will be verified. On-line remote support of these tests is in general acceptable. However, there will be mandatory inspections of instruments and spacecraft configurations planned throughout the ATLO flow. PI representatives must support these activities at the spacecraft contractor's facility and at the launch site. Payload suppliers shall provide sensor stimulators, as required, to facilitate stimulation and response verification of payload equipment and the spacecraft (end-to-end mission function).

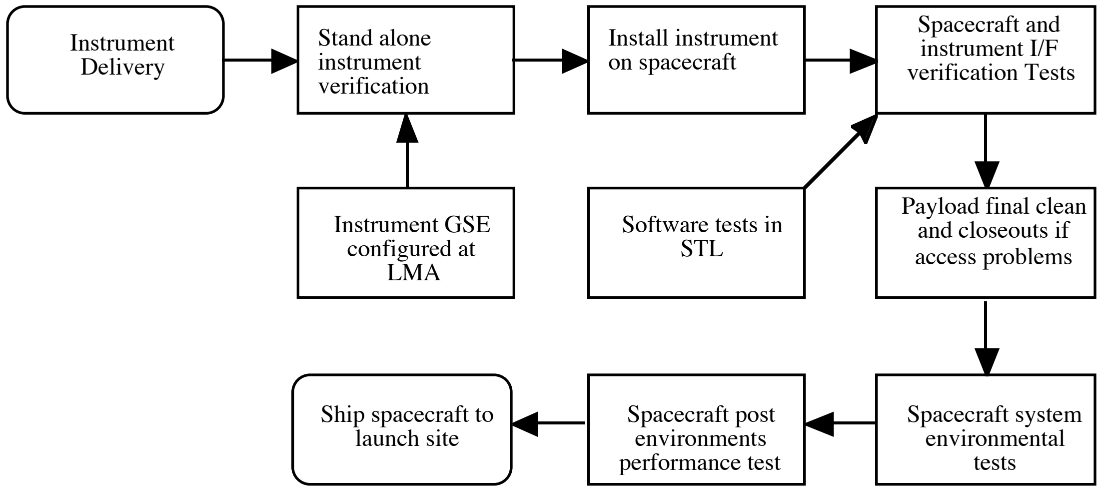


Figure 3.8.a Integrated System Tests

After installation, each component will be functionally tested, with an Instrument Engineer present, under lander control. Sterile dry nitrogen purge will be provided for instruments that require it. Trouble-shooting will be accomplished under documented control, including failure investigation, verification, and correction validation testing.

Throughout ATLO the PIs will be responsible for the analysis of all payload data gathered during system test activities, and for providing any technical support personnel deemed necessary. The payload may be powered on to support mission sequence 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, IST and 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 the spacecraft contractor’s facility. The payload supplier shall provide any and all equipment required to support payload integration and launch site operations. Figure 3.8.b shows the sequence and flow of launch site activities.

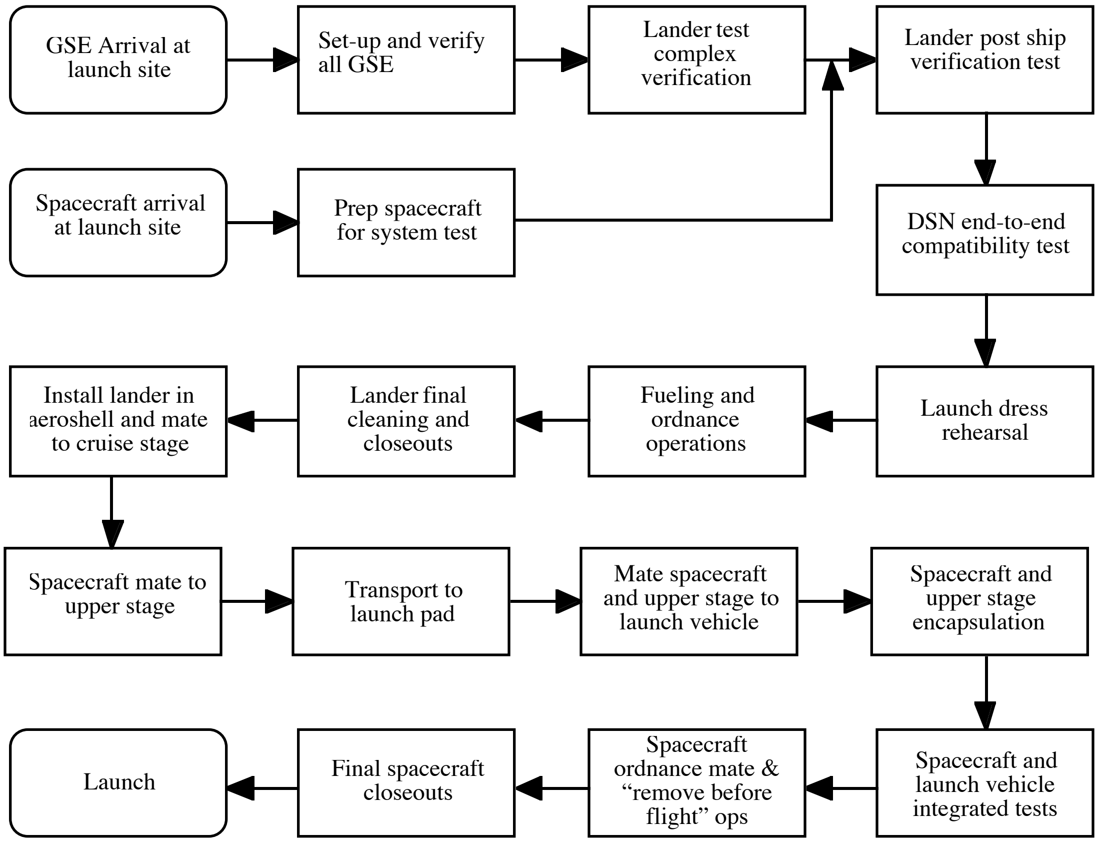


Figure 3.8.b Launch Site Sequence and Flow

4.0 Ground Data System/Mission Operations

4.1 Ground Data System (GDS)

4.1.1 Description

The Mars Surveyor Ground Data System utilizes the full resources of JPL's institutional capabilities, including the Telecommunications and Missions Operations Directorate, which consists of the Deep Space Network and Multi-mission Ground System Office (MGSO); the Mars Surveyor Operations Project (MSOP); the Planetary Data System (PDS); and project specific resources. The Mars Surveyor mission will utilize uplink and downlink capabilities of the Advanced Multi-Mission Operations System (AMMOS) for sending and receiving spacecraft data.

Adaptations of subsets of these system capabilities are phased to support spacecraft subsystem and system development. Figure 4.1.1.a provides a general description of the ground data system configuration for Mars'01 development and launch operations.

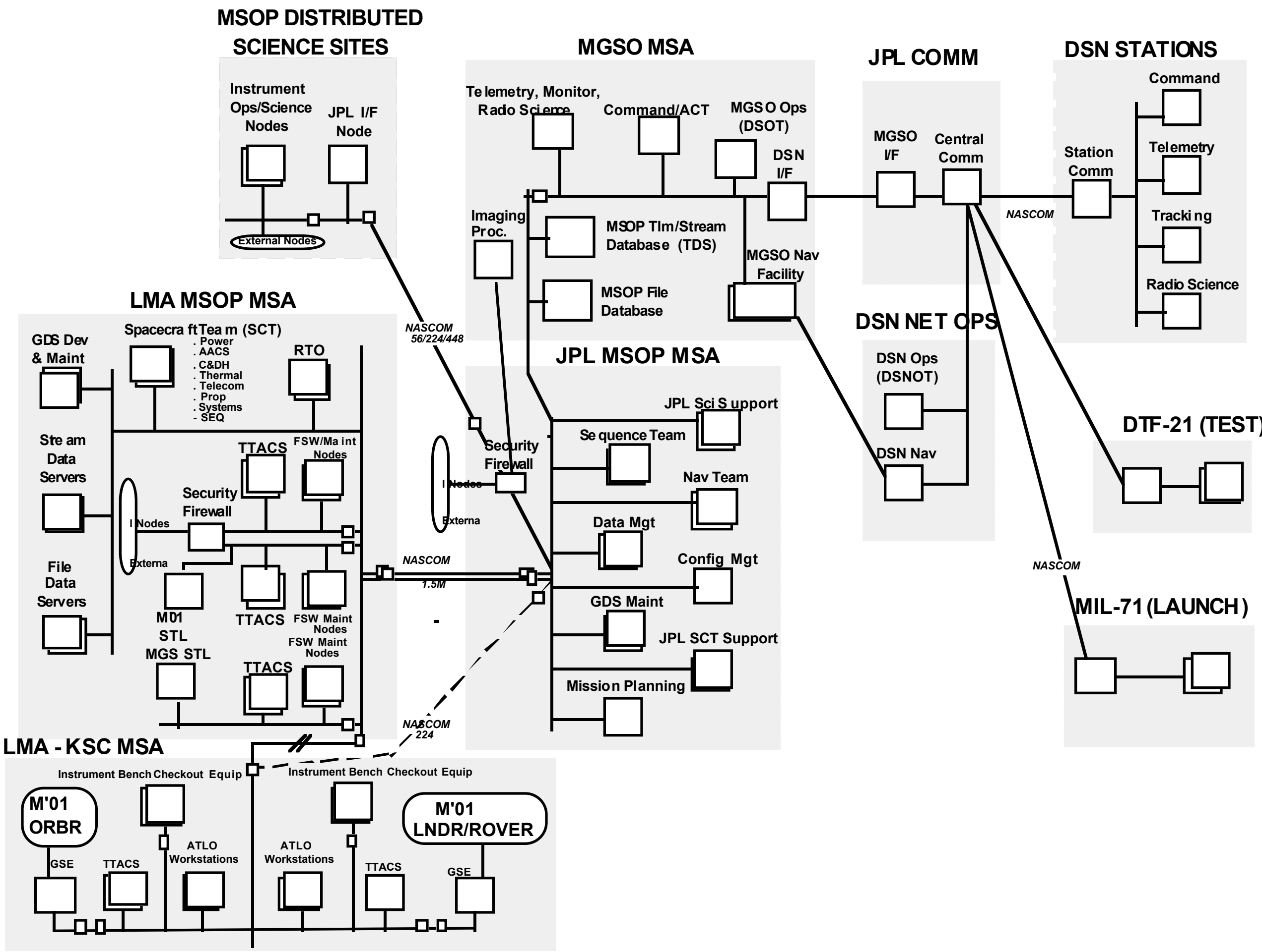


Figure 4.1.1.a Ground Data System Configuration (at MSP'01 Launch)

The Telemetry Data Flow Diagram is shown in Figure 4.1.1.b. Spacecraft data acquisition is provided by the Deep Space Network (DSN), which also provides basic decoding and frame synchronization of telemetry data. Navigation data (doppler and ranging) is also supplied by the

DSN. The Ground Communications Interface Facility (GIF) provides the interface between the DSN and AMMOS.

The Telemetry Input Subsystem (TIS) performs initial processing on telemetry frames and/or packets. Telemetry processing includes frame synchronization, decoding for error correction (Golay, Reed/Solomon), synchronous and asynchronous extraction, depacketization, decommutation, and channelization. Data from the TIS is sent to the Telemetry Distribution Subsystem (TDS), and also broadcast on multiple broadcast channels for real-time display on the Data Monitor and Display (DMD) nodes.

The TDS provides a common user interface to both the real-time and non-realtime data. Near real-time data extends from a few minutes to a few days. Longer retrieval periods are obtained via the CDB which can hold thirty to forty days of mission data on-line, and older data in an off-line archive.

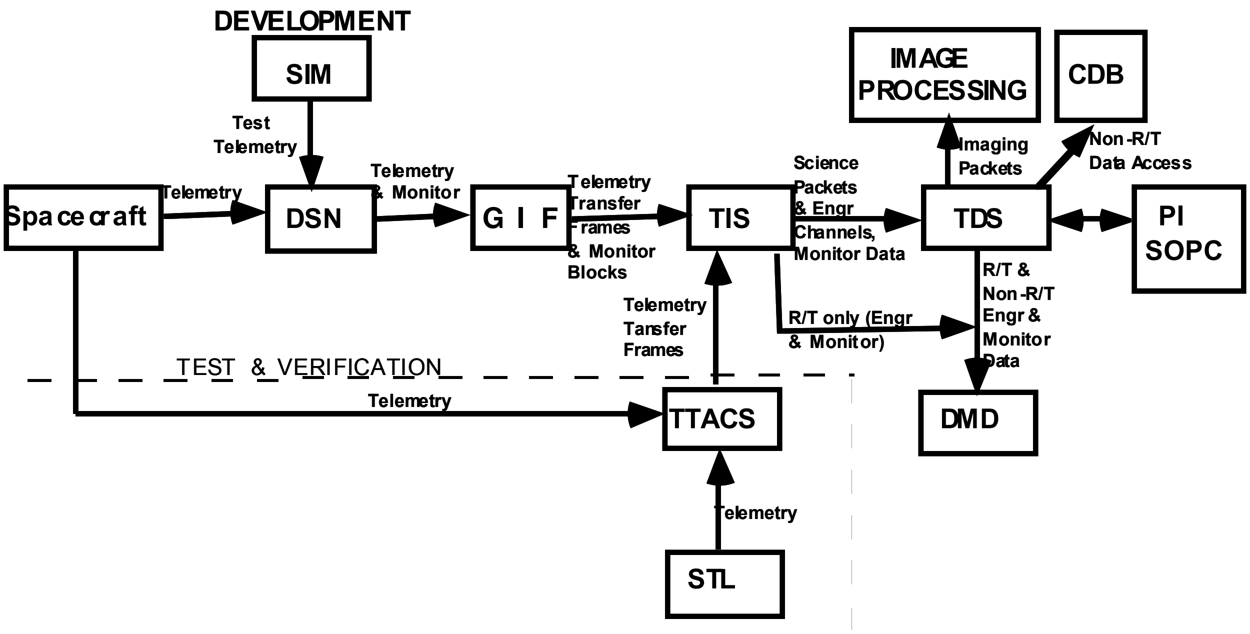


Figure 4.1.1.b Telemetry Dataflow

The CDB provides data management services to other subsystems. The CDB loads, archives, and catalogs data using relational database technology and provides a file service for data products. The data may then be retrieved by query for later analysis.

Imaging packets will require further processing, along with ancillary data. The selection and budget for processing image science data are the responsibility of the PI.

Data Monitor and Display (DMD) performs channel processing that includes channel derivation, conversion, alarm checking, and display generation. Display types currently include plots, matrices, and lists. A display template language is also provided that allows the user to define the content of display windows.

During development, the Test Telemetry and Command System (TTACS) is used to communicate with the spacecraft (for commands and telemetry). The Spacecraft Test Laboratory (STL) is used for test and verification. All commands, sequences and flight software are tested on the STL to ensure expected operability prior to actual testing on the spacecraft. During flight, the STL is used to validate unique sequences and updates to flight software.

Command Data Flow in shown is Figure 4.1.1.c. The CDB is used as the repository for all files which flow from one team to another. Once all the necessary input files are generated, the Sequence (SEQ) subsystem provides manual and autonomous tools for planning, generating, and checking spacecraft and payload command sequences. The Command (CMD) subsystem translates spacecraft and payload command messages into a command bit stream, transfers the command file to the DSN and provides operations control for the transmission of these commands to the spacecraft. During Assembly Test and Launch Operations (ATLO) the Test Telemetry and Command System (TTACS) is used to interface with the spacecraft.

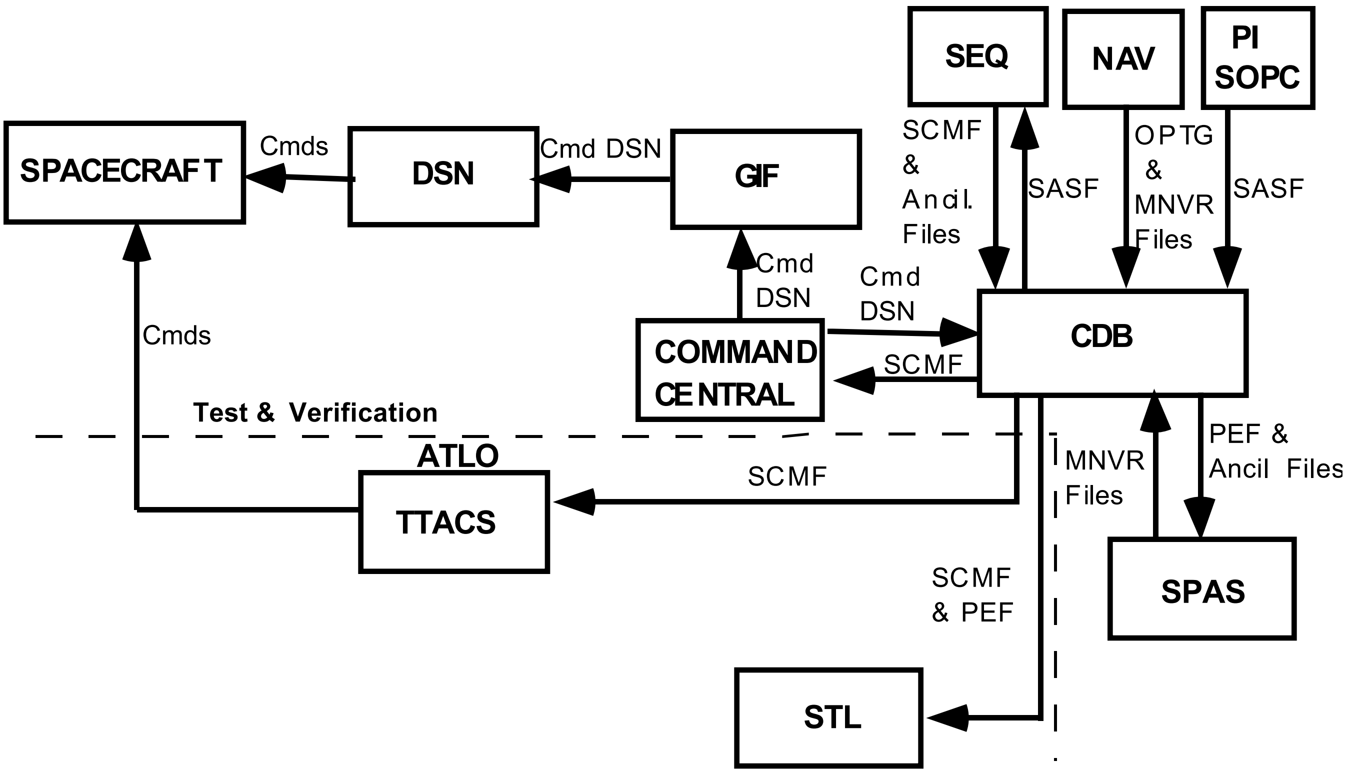


Figure 4.1.1.c Command Dataflow

4.1.2 Downlink Ground Data System Constraints

To minimize GDS development operational costs, proposers are encouraged to follow the downlink telemetry constraints below:

a. If science data, engineering, or instrument health packets using a channelization scheme (1) do not use internal channelization, (2) follow the data typing rules below, and (3) only switch channels based on packet id, spacecraft clock value, or flag values (i.e., Memory Read-Out (MRO) on).

b. The standard MRO is a bus code of 8 bits, address of 32 bits, and values are an even number list of 8 bit bytes.

c. The standard data types are : 1-4 bit flags, 1-32 bit unsigned integers or collection of bits, 2-32 bit 2 complement signed integers, 32 or 64 bit IEEE floating point numbers, or 1-12 bytes of 8 bit ASCII characters.

d. Byte alignment and packing rules have the most significant byte first in a word, and the most significant word first in a long word.

e. The most significant bit in a field is bit 0.

4.1.3 SOPC Technology Utilization

A Science Operations and Planning Computer (SOPC) will be provided to each science investigation. The hardware component is configured to have a minimum 22 MIPS performance with 64 MB of RAM, 2 GB of high-speed disk capacity and high resolution color graphics.

The primary function of the SOPC includes command planning, retrieval of data packets and their associated ancillary data from the project database, display of engineering data, and science data processing. All operations software necessary to submit commands and retrieve data will be installed, tested and maintained by the GDS.

The SOPC may be used to support instrument development and then later connected to the MSOP GDS within the mission operations facility at JPL. Engineering display and processing software, and data retrieval software are provided by JPL. Science data processing operations may also use these dedicated SOPCs.

4.1.4 GDS Processing

a. The GDS performs the Level 0 processing (defined as a raw packet) which includes the following: transfer frame synchronization, Reed-Solomon decoding, and packet extraction for all packet types. The packets are available via interactive or batch-mode queries shortly after the receipt of the transfer frames.
b. Raw engineering packets and Level 0 science data packets are archived by the GDS. The archive will be to CD-ROM Media.
c. The GDS performs the Level 1 processing for engineering packets only.
d. All packets conform to the CCSD Standard Formatted Data Unit (SFDU).
e. Science instrument health data is included in science packets.

4.1.5 GDS Products

The ground system shall supply the following support:

a. Science data acquisition by the lander/rover, stored on-line on a computer system at JPL in packet form, along with ancillary data relating to the conditions under which the data were acquired (examples include selected engineering data, solar elevation angle, etc.). Ancillary data is accessed via catalogue system.
b. Authorized remote user electronic access to Project Databases.
c. NASA level zero data archive to payload investigations. This shall include:
    1) The best available original data, obtained by merging all versions of science data received at JPL from all sources (multiple spacecraft replay, multiple DSN station replays, etc.).
    2) Ancillary data collected by the project.
    3) Additional information of a textural nature describing the data records, to be supplied by the Principal Investigator after data receipt at JPL for incorporation onto the CD-ROM data set.
d. Archival of reduced data products, or “Level 1” data to the PDS in accordance with the Science Data Management Plan is the responsibility of the PI.

4.2 Mission Operations

Mission operations for the Mars’01 lander/rover will be performed by the Mars Surveyor Operations Project (MSOP). These operations are distributed between Lockheed Martin, JPL and the Principal Scientists, but managed by JPL. The MSOP has an annual operations budget of $20 million to fly all spacecraft under the Mars Surveyor program. Strategies, processes and procedures developed to fly earlier Mars Surveyor spacecraft will be modified/enhanced as necessary to meet the needs of the M'01 mission.

4.2.1 Pre-Launch Activities

Prior to launch the PI(s) will be involved in the development of nominal and mission critical sequences which involve the lander instruments, the rover, and the rover instrument suite (such as the first-day landed activities), that need to be tested in ATLO. The building of spacecraft commands and blocks require PI 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 PI(s) will use their dedicated SOPC for subsystem GDS testing which involves the payload. The PI(s) 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.

4.2.2 Cruise Activities

All activities planned for the mission are documented in the project Mission Plan. During Cruise, payload health checks and calibrations will be carried out. Cruise stored sequences are nominally 3-5 weeks in duration. Non-stored payload command capability shall be available. Command requests for certain class instruments which do not affect spacecraft resources can be processed and transmitted totally autonomously.

4.2.3 Entry Descent and Landing

Command sequences for the entry, descent and landing will be stored on the lander/rover prior to launch and updated in late cruise, as required. The first sol will also be pre-programmed for deployments, assessing the health of the lander and rover, determining the environment, and, if permissible, performing minimal science to ensure some mission success.

4.2.4 Surface Operations

The landed mission depends almost entirely on the requirements of the selected payload. Operations for the lander and rover will occur in parallel while they are both alive (the nominal mission for the lander is 100 days, the rover mission is approximately one earth year). The daily telemetry data budget is split between the lander and the rover, with 40 Mbits per day allocated to the rover and 10 Mbits be day allocated to the lander. Since surface operations in an unknown environment are not predictable prior to arrival, the operations environment shall accommodate near real-time commanding.

Utilizing the project supplied SOPC at their home institutions, the PIs will send command requests and access data in non-realtime electronically from the project data base.

The rover flight spare will be available for testing of sequences and anomaly investigation. The 2001 Mars Rover Activity Time-Lines in section 3.5.3 describes the potential day to day activities of the rover.

5.0 Payload Management/Deliverables - LANDER INSTRUMENTS

5.1 PI Responsibilities

The instrument principal investigator (PI) is responsible for instrument design and development, fabrication, test, and calibration, and delivery of flight hardware, software, and associated support equipment, within project schedule and payload resources. The PI is responsible for planning and operational support of instrument operation, data analysis and overall conduct of the investigation.

A lander contractor payload integration engineer will represent the payload at the lander contractor's site as a participant in the contractor's integrated product teams (IPTs) and to negotiate interfaces with the instrument. The specific responsibilities of the PI(s) 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 of the investigation 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. Participate in PSG meetings and associated working groups. PSG meetings may be held in conjunction with reviews at the S/C contractor's facility.
9. Conduct payload reviews as required by section 5.2.1.
10. 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.
11. Support payload integration and system test procedure development and maintenance. Support instrument and GSE integration, and lander system testing at LMA and KSC. 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 LMA and KSC, Thermal Vacuum Test, compatibility/EMI, and on-pad power-on/final payload close-out.
12. Support definition of mission database contents, including but not limited to: flight rules and constraints, sequences, payload telemetry, and commands.
13. Support integrated mission data/sequence development and flight software integration, using the spacecraft test laboratory (STL).
14. Support planning and executing mission operations, including end to end test support.

15. Ensure that the reduction, analysis, reporting, and archival of the results of the investigation meet with the highest scientific standards, and completeness consistent with budgetary and other recognized constraints.
16. Preparing, certifying and releasing data products (to PDS) according to the Science Data Management Plan.

5.2 Deliverables

As described in the following sections, during Phase B, C & D, meeting both schedule and cost, the PI(s) 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. Support development and maintenance of ICDs.
4. Provide monthly technical progress reports (TPRs) and financial management reports (FMRs).
5. Deliver flight hardware (including thermal blankets if required) to the lander contractor which meets planetary protection requirements, with suitable shipping containers and any covers required. Also, deliver connectors to the lander contractor 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.c., and a payload interface simulator to the lander contractor. 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, and launch operations.
8. Provide an instrument end item data package (EIDP), as described in section 5.2.4.e.
9. If the S/C central computer is utilized for sequencing or data processing, deliver the necessary flight s/w to be resident in the lander computer (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 (see Table 5.2.a).

Figure 5.2.a is an overview of the schedule for lander deliverables. Project milestones are indicated. Table 5.2.a contains a detailed list of deliverables and associated dates.

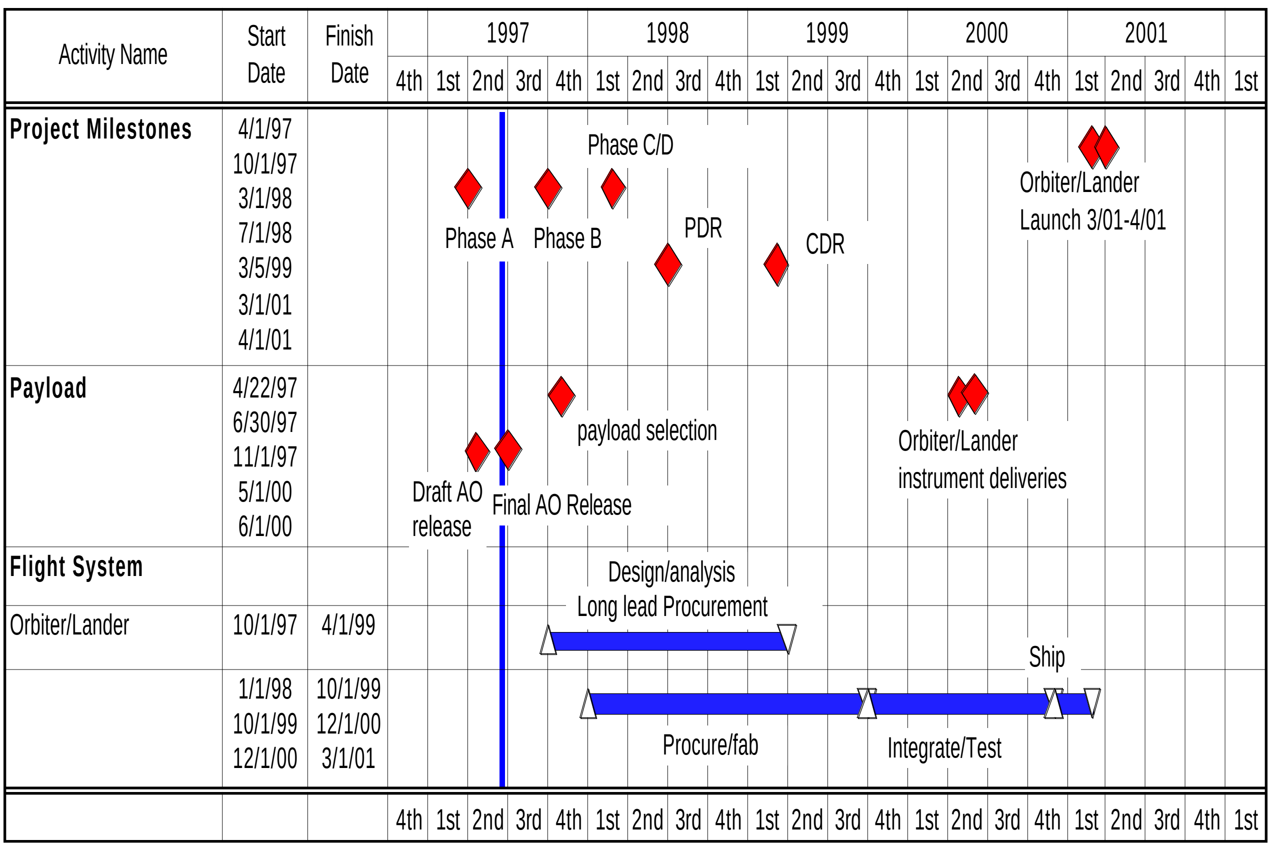


Figure 5.2.a MSP' 01 Program Master Schedule

Table 5.2.a MSP'01 Lander Payload Delivery Schedule

EVENT	DESCRIPTION	EVENT DATE OR DUE DATE
Payload Selection	....	01 Nov 97
P/L PIRDR	Preliminary Interface Requirements and Design Review....................	01 Aug 98
P/L FIRDR	Final Interface Requirements and Design Review...........................	15 Apr 99
P/L Hardware Delivery		01 June 00
Interface Connectors	for harness side of interface	01 Mar 99
Payload Fit Check Templates	....	01 Oct 99
Payload Interface Sim	....	01 Aug 99
Initial Test Software	....	06 Aug 99
P/L H/W Flight Units & GSE		01 June 00
P/L Software Delivery		01 Aug 00
• Flight Rules and Constraints	Definition of Instrument Operating Constraints and Requirements	*Start Effort: 7/1/99 *Prel. System Test & Prelaunch Ops: 11/1/99 *Launch & Cruise: 12/1/99 *EDL: 12/1/99 *Landed: 2/1/00
• Command and Telemetry Data	Definition of Instrument Commands and Operations Modes; Definition of Instrument Telemetry Parameters	*Start Effort: 7/1/99 *Prelim. Cmd List: 11/1/99 *Prelim. Tlm List: 12/1/99 *Final Cmd List: 1/2/00 *Final Tlm List: 2/1/00
• Telemetry Calibration Data	Definition of Instrument Telemetry Calibration Curves, Algorithms and Tolerances	*Start Effort: 7/1/99 *Prelim. Data: 1/1/00 *Final Data: 1/1/01
• Flight Sequences	Definition of Instrument Sequences for Use in System Test to Include All Instrument Operations Modes	*Start Effort: 7/1/99 *Prel. Systems Test and Prelaunch Ops: 11/1/99 *Launch and Cruise: 12/1/99 *EDL: 12/1/99 *Landed Sequences: 2/1/00
• Seq. Validation Support	Support for S/C and Instrument Validation	4/1/00 - 8/1/00
• Updates to Flight Databases	Definition of Updates to Flight Sequences, Command and Telemetry Data, Telemetry Calibration Data, and Flight Rules and Constraints Based on System Test	10/1/00 - 2/1/01
Documentation		
MOA/Contract	....	01 Feb 98
FRD/Safety	....	01 Feb 98
GDS/MOS Req. (Preliminary)	....	15 Dec 98
Thermal Models	....	@ PIRDR
GDS/MOS Req. (Final)	....	15 Oct 99
ICD's (Preliminary)	....	15 Jul 98
ICD's (Final)Start Config Control	....	01 Mar 99
Initial S/W Req. ICD	....	08 Sep 98
S/W ICD Update	....	02 Sep 99
Final S/W ICD	....	15 Jan 00
Final S/W Baseline Delivery	....	01 Jun 00
P/L Handling Req. List (Prelimin)	....	01 Sept 99
P/L Handling Req. List (Final)	....	01 Aug 00
Unit History Log Books	....	@ IDR
End Item Data Package	....	@ IDR

5.2.1 Reviews

The payload PI(s) (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 PI or PI representative.

The PI will conduct the instrument Preliminary Interface Requirements and Design Review (PIRDR). The PIRDR 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 PI will conduct the instrument Final Interface Requirements and Design Review (FIRDR). The FIRDR 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 PI will conduct an Instrument Delivery Review (IDR). This review is held just prior to instrument delivery to LMA, 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.

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.

Tables 5.2.2.a, give requirements for hardware deliverables to support the instrument interface to the spacecraft C&DH subsytem.

Table 5.2.2.a Interface Deliverables

Item	Respon-sibility	Description	Due Date
RS422 Serial Interface Definition	S/C	Defines the electrical interface functionality, protocols, and drivers/receivers characteristics of the RS422 serial interface	8/15/99
Payload Interface Simulator	PI	The payload interface simulator will be interfaced to the development C&DH module to verify the transfer of data/commands to/from the C&DH processor using the flight electrical interface and transfer bit rates/protocols.	8/15/98
Initial Flight S/W	PI	Provide the initial FSW load to support STL (Required if the central computer is used)	11/1/98
Final S/W Baseline	PI	Provide the final FSW load to support STL (Required if the central computer is used)	01/1/00

5.2.3 Software

Interface software delivery dates are tabulated in Table 5.2.2.a. Note, a through c of this section apply only if the PI plans to use the resources of the S/C central computer.

a. Software Documentation - Software/Computer Systems ICD

Preliminary definition of operational timeline requirements and related resource demands (characterized by peak and typical parameters) will be documented in a software-specific ICD for:

1. volatile and non-volatile memory
2. process activation frequency and duty cycle
3. storage demands with storage durations
4. I/O requirements for all classes - backplane bandwidth, data bus bandwidth, command/telemetry bandwidth - including best available information on any protocol standards or unique data transfer methods are due by preliminary ICD delivery.

Updated information for all items in the first delivery, with projections of final commitments for all resource demands, plus protocol specifications for all transactions using the spacecraft C&DH, including characteristics of timing where it is relevant to correct operations of the science payload/mission is due by the Initial S/W ICD date (See Table 5.2.a).

The committed baseline for all elements of the Software/Computer System Section of the ICD is the third delivery, due by the Final S/W Requirements ICD date (See Table 5.2.a). Note that this delivery supports the evaluation cycle for certification of the STL in the first quarter of calendar 2000.

b. Software Documentation - Other

Requirements, design, build, test, and evaluation information that provides insight into the software implementation should be provided as they become available, in accordance with the PI's normal development plan.

c. STL Operations: Required Evaluation Procedures

For evaluation activities identified from development integration planning to be performed in the STL, procedures will be required and will be subject to approval. The fidelity of the procedure and level of approval will correspond to the potential risks involved in the procedure. Generally, as the STL is primarily a simulation and Engineering Development Unit (EDU) environment, the risk is minimal, requiring approval from only the cognizant personnel for the item under evaluation and STL operations. Circumstances which may require further approvals include:

1. use of flight hardware in the configuration
2. requirements for special interfaces - either hardware or software - that may require STL setup and verification
3. exclusive operations or continuous operations that produce resource conflicts not reconcilable among other parties

d. Software Source Materials

The mission load (all executable spacecraft and payload flight software and data) will be generated as an integrated load image, including initial/nominal values for all updatable mission data/system files. To develop the mission load, source code for compilation, and materials for binding, and data/file load will be provided in a timely fashion to support software development integration in the STL, assembly and integration tests during science payload integration, and mission readiness tests at the launch site.

5.2.4 Documentation

a. Contract/MOA

Shortly after payload selection, the project will enter into an agreement with each PI 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 project and the PI (and PI organization). The agreement will take the form of a contract for non-government entities, and a memorandum of agreement (MOA) for government entities

b. FRD / Safety

The PI 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.

c. ICDs

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.

d. 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.

e. End Item Data Package (EIDP)

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).

6.0 Payload Management/Deliverables - ROVER PAYLOAD

6.1 Rover Payload PI Responsibilities

The principal investigator (PI) is responsible for the design, development, fabrication, test, calibration, and delivery of flight hardware, software, and associated support equipment for the rover payload, defined as a suite of instruments and a sample acquisition system, within project schedule and allocated resources. The PI is responsible for the planning and operational support of rover payload operations, data analysis and overall conduct of the investigation.

The specific responsibilities of the PI include but are not limited to:

1. Develop an internal management plan.
2. Ensure that the design, development, fabrication, and testing the payload, and any deployment/mobility devices (if applicable), are appropriate to the objectives of the investigation and meet the environmental and interface constraints.
3. Manage rover payload design 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 rover payload development meets the approved schedules and cost plans.
6. Be the primary point of contact with the Project for the purpose of establishing functional requirements (FRs), mechanical and electrical interface control documents (MICD & EICD), schedules and transfer of funds.
7. Ensure that the instruments are properly calibrated.
8. Participate in PSG meetings and associated working groups.
9. Conduct rover payload reviews as required by section 6.2.1.
10. Participate in software working group (SWG) meetings, as required by the proposed science mission use of rover computational resources and services to resolve requirements and interface issues, and resolve resource allocations and operational timelines.
11. Support rover payload integration, rover system test procedure development and maintenance, instrument and GSE integration, system testing at JPL, and pre-lander integration/delivery acceptance testing at KSC.

    Remote support of tests is in general acceptable, however, on-site field engineering support is required for mandatory inspections and the following rover tests at JPL: initial instrument power turn-On, rover payload functional electrical test, 1st rover functional test, thermal vacuum test, and compatibility/EMI; and for any delivery acceptance testing of the rover system at KSC.
12. Support definition of mission database contents, including but not limited to: flight rules and constraints, sequences, rover payload telemetry, and commands.
13. Support integrated mission data/sequence development and flight software integration.
14. Plan and execute mission operations, including end to end test support.
15. Ensure that the reduction, analysis, reporting, and archival of the results of the investigation meet with the highest scientific standards, and completeness consistent with budgetary and other recognized constraints.

16. Prepare, certify and release data products (to PDS) according to the Science Data Management Plan.

6.2 Deliverables

As described in the following sections, during Phase B, C &D, meeting both schedule and cost, the PI(s) shall:

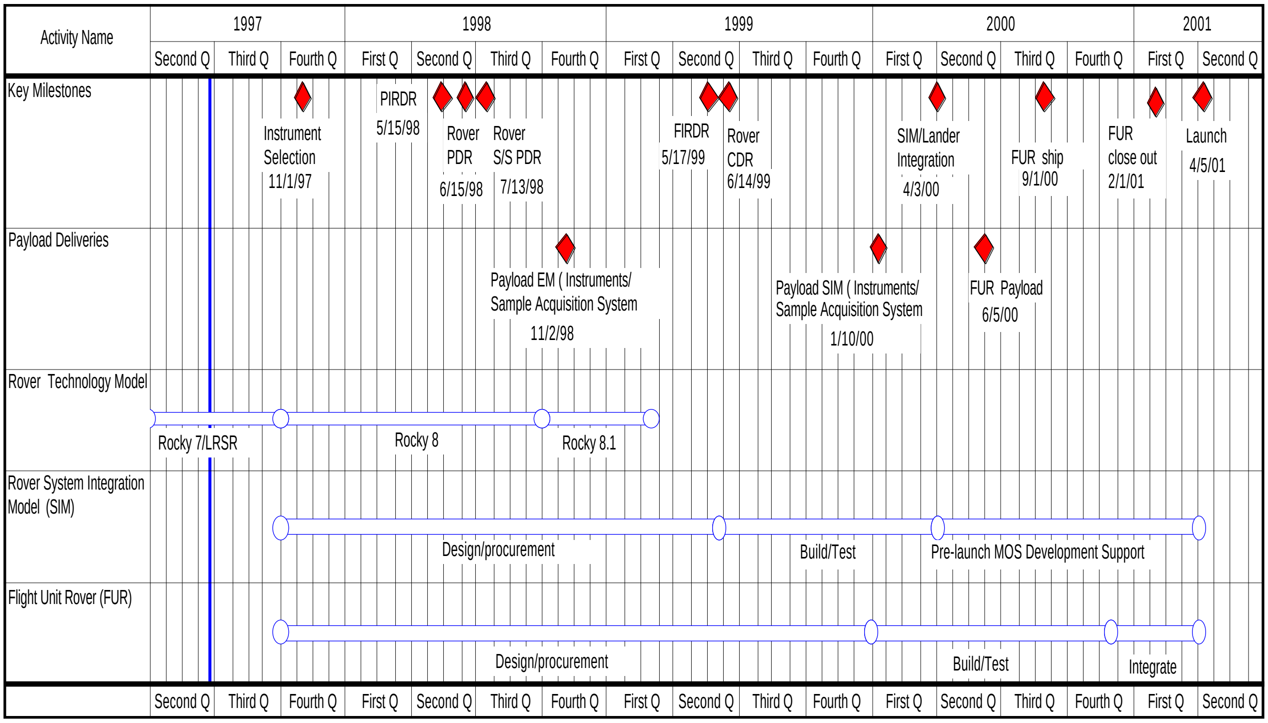
1. Shortly after selection, sign a memorandum of agreement (MOA) or contract (as applicable) with the project, documenting resource allocations.
2. Provide and maintain required documentation (see Section 6.2.4).
3. Support development and maintenance of ICDs.
4. Provide monthly technical progress reports (TPRs) and financial management reports (FMRs).
5. Deliver a flight unit rover payload, consisting of a suite of flight instruments and a flight sample acquisition system, to JPL which meets planetary protection requirements, with suitable shipping containers and any covers required.
6. Deliver a rover payload system integration model (SIM). The payload SIM, shall include the suite of flight instruments and the sample acquisition system. It shall be flight quality except it is not required to be submitted to full environmental testing prior to delivery.
7. Deliver a form fit and function rover payload engineering model (EM), which includes the suite of flight instruments and the sample acquisition system. The EM design shall incorporate the elements of a mass simulator and a fit check template.
8. Deliver a a single node analytical thermal model for each assembly of the rover payload.
9. Provide any necessary instrument-unique ground support equipment (GSE) to support the EM, the SIM, and the flight unit rover payload for rover development, standalone testing, integration at JPL, the lander contractor and KSC, and launch operations.
10. Provide a rover payload end item data package (EIDP), as described in section 6.2.4.e.
11. If the rover central computer is utilized for sequencing or data processing, deliver the necessary flight s/w to be resident in the rover computer. (see Section 6.2.3).
12. Provide timely information to establish and maintain controlled baselines for software interfaces, shared computational resources, mission data, and mission operations timelines and sequences (see Table 6.2.a).

Figure 5.2.a in section 5.0 provides an overview of the master project schedule for MSP' 01. Table 6.2.a contains a detailed list of rover payload deliverables and associated dates. Figure 6.2.a contains a rover development schedule.

Table 6.2.a MSP' 01 Rover Payload Delivery Schedule

EVENT	DESCRIPTION	EVENT DATE OR DUE DATE
Rover payload Selection Rover payload PIRDR Rover payload FIRDR	.... Preliminary Interface Requirements and Design Review.......................... Final Interface Requirements and Design Review................................	01 Nov 97 15 May 98 15 Mar 99
Hardware/Software Deliveries		Also see table 6.2.2.a
Flight Unit Rover Payload - H/W & S/W Payload EM Payload SIM	.... includes any necessary GSE includes any necessary GSE includes any necessary GSE	 05 June 00 02 Nov 98 10 Jan 00
Deliveries for Operations		
• Flight Rules and Constraints	Definition of Instrument Operating Constraints and Requirements	*Start Effort: 7/1/99 *Prel. System Test & Prelaunch Ops: 11/1/99 *Landed: 2/1/00
• Command and Telemetry Data	Definition of Instrument Commands and Operations Modes; Definition of Instrument Telemetry Parameters	*Start Effort: 7/1/99 *Prelim. Cmd List: 11/1/99 *Prelim. Tlm List: 12/1/99 *Final Cmd List: 1/2/00 *Final Tlm List: 2/1/00
• Telemetry Calibration Data	Definition of Instrument Telemetry Calibration Curves, Algorithms and Tolerances	*Start Effort: 7/1/99 *Prelim. Data: 1/1/00 *Final Data: 1/1/01
• Flight Sequences	Definition of Instrument Sequences for Use in System Test to Include All Instrument Operations Modes	*Start Effort: 7/1/99 *Prel. Systems Test and Prelaunch Ops: 11/1/99 *Rover Sequences: 2/1/00
• Seq. Validation Support	Support for S/C and Instrument Validation	4/1/00 - 8/1/00
• Updates to Flight Databases	Definition of Updates to Flight Sequences, Command and Telemetry Data, Telemetry Calibration Data, and Flight Rules and Constraints Based on Rover System Test	10/1/00 - 2/1/01
Documentation		
MOA/Contract	....	01 Feb 98
FRD/Safety (Combined)	....	01 Feb 98
GDS/MOS Req. (Preliminary)	....	15 Dec 98
Thermal Models	....	@ PIRDR
GDS/MOS Req. (Final)	....	15 Oct 99
MICD/EICD (Preliminary)	....	@ PIRDR 98
MICD/EICD (Final) Config Contr	....	14 Mar 99
Prelim. S/W Req. Document	....	@ PIRDR
S/W Req. Document Update	....	02 Sep 99
Final S/W Req. Document	....	15 Jan 00
Instr. Handling Req. List (Prelim.)	....	01 Sept 99
Instr. Handling Req. List (Final)	....	01 April 00
Unit History Log Books	....	@ IDR
End Item Data Package	....	@ IDR

Figure 6.2.a MSP'01 Rover Development Schedule



6.2.1 Reviews

The rover payload PI (or designate) will be expected to attend/host rover design reviews, ground system reviews, and occasional informal reviews as scheduled by the Rover Manager.

The PI will conduct the rover payload Preliminary Interface Requirements and Design Review (PIRDR). The PIRDR is scheduled as early as possible after the completion of the Functional Requirements Document (FRD), Electrical Interface Control Document (EICD), and the Preliminary Mechanical Interface Control Document (MICD). Topics include: discussion of the FRD, and description of interfaces.

Likewise, the PI will conduct the rover payload Final Interface Requirements and Design Review (FIRDR). The FIRDR follows the rover CDR, at the completion of the rover payload detailed design and the final EICD and MICD. 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 PI will conduct the rover payload Delivery Review (IDR). This review is held just prior to flight unit rover payload delivery to JPL, and topics include how well the instrument complies with the Functional Requirements and the ICDs, the results of environmental testing, and the completeness of the EIDP.

6.2.2 Rover Central Computer Interface Definition & S/W Delivery

Tables 6.2.2.a, give requirements for hardware and software deliverables to support the instrument interface checkout to the rover computer subsytem:

Table 6.2.2.a Rover Central Computer Interface Deliverables

Item	Respon-sibility	Description	Due Date
RS422 Serial Interface Definition	S/C	Defines the electrical interface functionality, protocols, and drivers/receivers characteristics of the RS422 serial interface	6/15/98
Prelim. I/F S/W	PI	Provide the preliminary Interface S/W load to support Rover System Integration Model (SIM) testing. (Required if the central computer is used)	10/1/98
Final I/F S/W	PI	Provide the final Interface S/W load to support Rover SIM (Required if the central computer is used)	12/1/99

6.2.3 Interface Software

Software delivery dates are tabulated in Section 6.2.2. Note this section applies only if the PI plans to use the resources of the rover central computer.

a. Software Documentation

Initial definition of operational timeline requirements and related resource demands (characterized by peak and typical parameters) will be documented in a software-specific section of the Preliminary S/W Requirements Document (See Table 6.2.a) for:

1. Volatile and non-volatile memory
2. Process activation frequency and duty cycle
3. Storage demands with storage durations
4. I/O requirements for all classes - data bus bandwidth, command/telemetry bandwidth - including best available information on any protocol standards or unique data transfer methods are due by first preliminary S/W requirements document delivery.

Updated information for all items in the first delivery, with projections of final commitments for all resource demands, plus protocol specifications for all transactions using the rover computer, including behavioral characteristics of timing where it is relevant to correct operations of the science rover payload/mission is due in the preliminary S/W requirements document delivery .

The committed baseline for all elements of the software/computer system section is the third delivery, due by the final S/W requirements document date (See Table 6.2.a). Note that this delivery supports the evaluation cycle for certification of the System Integration Model (SIM) in the forth quarter of 1999

b. Software Documentation - Other

Requirements, design, build, test, and evaluation information that provides insight into the software implementation should be provided as they become available, in accordance with the PI's normal development plan.

c. Rover SIM Operations: Required Evaluation Procedures

For evaluation activities identified from development integration planning to be performed in the rover SIM, test procedures are required and are subject to approval. The fidelity of the procedure and level of approval corresponds to the potential risks involved in the procedure. Generally, as the rover SIM is primarily a simulation and Engineering Development Unit (EDU) environment, the risk is minimal, requiring approval from only the cognizant personnel for the item under evaluation and rover SIM operations. Circumstances which may require further approvals include:

1. Use of flight hardware in the configuration
2. Requirements for special interfaces - either hardware or software - that may require rover SIM setup and verification
3. Exclusive operations or continuous operations that produce resource conflicts not reconcilable among other parties.

d. Software Source Materials

The mission load (all executable rover and rover payload flight software and data) is generated as an integrated load image, including initial/nominal values for all updatable mission data/system files. To develop the mission load, source code for compilation, and materials for binding, and data/file load shall be provided in a timely fashion to support software development integration in the rover SIM, assembly and integration tests during science rover payload integration, and mission readiness tests at the launch site.

6.2.4 Documentation

a. MOA - Prepared by JPL

Shortly after payload selection, the project will enter into an agreement with the PI for the implementation of their selected proposal. The agreement will document the rover payload resource allocation (mass, power, volume and fiscal resources) and schedule between the project and rover payload PI (and PI organization). The agreement will take the form of a contract for non-government entities, and a memorandum of agreement (MOA) for government entities.

b. FRD / Safety - Prepared by the PI

The PI will be responsible for writing a functional requirements document (FRD) for the rover payload, and supplying the necessary payload safety information for the development of the range safety plan and the payload safety reviews at the launch site.

c. Mechanical and Electrical ICDs (MICD AND EICD) - Prepared by JPL

The MICD AND EICD identify all rover payload interfaces, including but not limited to, the payload envelopes, 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.

d. Payload Handling Requirements and Unit History Logbook - PI supplied

A rover 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.

e. End Item Data Package (EIDP) - PI supplied

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).

Appendix A - Acronyms

τ	(tau) atmospheric opacity
ΔV	Delta Velocity (change)
A/C	Aerocapture
Ah	Amphour
AMMOS	Advanced Multi-Mission Operations System
AO	Announcement of Opportunity
ATLO	Assembly, Test and Launch Operations
ATP	Authority to Proceed
AU	astronomical unit
bps	bits per second
C	(degrees) Centigrade
C&DH	Command and Data Handling (subsystem)
CCSDS	Consultative Committee on Space Data Standards
CDB	Central Data Base
CD-ROM	Compact Disc - read only memory
CDU	Command Detector Unit
CG	Center of Gravity
cm	centimeter
CMD	the GDS Command subsystem
COSPAR	Committee on Space Research
CPL	Capillary Pumped Loop
CPV	Common Pressure Vessel (battery)
D	diameter
D/L	Downlink
dB	decibel
deg	degrees
Dia	diameter
DIS	Data Interface Simulator
DMA	Dexterous Manipulator Arm
DMD	Data Monitor and Display
DN	Data Number
DSN	Deep Space Network
DTE	Direct-to-Earth
EDL	Entry Descent and Landing
EDR	Experiment Data Record
EDU	Engineering Development Unit
EEPROM	Electronically Erasable Program Read-Only Memory
EICD	Electrical ICD
EIDP	End Item Data Package
EIP	Experiment Implementation Plan
EMC	Electromagnetic Compatibility
EM	Engineering Model
EMI	Electromagnetic Interference
EU	Engineering Unit
FDCR	Final Design and Cost Review
FET	Functional Electrical Test
FIRDR	Final Interface Design and Requirements Review
FMR	Financial Management Review
FOV	Field of View
FRD	Functional Requirements Document
FUR	Flight Unit Rover
G	acceleration of one Earth gravity
g	acceleration of one Earth gravity
Gb	Gigabits
GB	Gigabytes
GDS	Ground Data System
GHz	gigahertz
GIF	Ground Support Equipment
GRS	Gamma-ray Spectrometer
GSE	Ground Support Equipment
He	Helium
HGA	High Gain Antenna
hr	hour
Hz	Hertz (cycles per second)
ICD	Interface Control Document
IDCR	Initial Design and Cost Review
IDDD	Instrument Design and Description Document
IPT	Integrated Product Team
IPTO	Initial Power Turn-on Test
IST	Integrated System Test
JPL	Jet Propulsion Laboratory
kbps	kilobits per second
km	kilometer
krads	kilorads (radiation)
L/D	Launch Date
LET	Linear Energy Transfer
LGA	Low Gain Antenna
LHP	Loop Heat Pipe
LMA	Lockheed Martin Astronautics

L_S	Time of season in degrees (0° = Northern Spring)
LV	Launch Vehicle
M	mass
m	meter
m/s	meters per second
max	maximum
Mb	Megabits
MB	MegaBytes
mBar	millibar (pressure)
mg	milligram
MICD	Mechanical ICD
MGA	Medium Gain Antenna
MGS	Mars Global Surveyor
MGSO	Multi-Mission Ground System Office
MHz	megahertz
min	minimum
min	minute
MIP	Mars In-Situ Propellant Production
MIPS	Millions of Instructions per Second
MIPS	Multi-Mission Image Processing System
MLI	Multi-Layer Insulation
mm	millimeter
MMI	Mineralogical/morphological Investigation
MOA	Memorandum of Agreement
MSA	Mission Support Area
MSOP	Mars Surveyor Operations Project
MSP	Mars Surveyor Program
N	Newton
N	North
NASA	National Aeronautics and Space Administration
Nav	navigation
Nom	nominal
NTE	not to exceed
nsec	nanosecond
OASPL	overall sound pressure level
OSA	Operational Science Analysis
Pa	Pascal (N/m^2)
P/L	Payload
PI	Principal Investigator
PIP	Proposal Information Package
PIRDR	Preliminary Interface Requirements and Design Review
PIWG	Payload Integration Working Group
pk	peak
ppm	parts per million
PSG	Project Science Steering Group
rad	radian (angle)
rad	unit of radiation
RCS	Reaction Control Subsystem
RDM	Radiation Design Margin
RFP	Request for Proposals
RHU	Radio-iosotopic Heater Unit
RPC	Remote Power Controller
RISC	Reduced Instruction Set
rms	root mean square
RSACS	Rock Sample Acquisition and Cache System
s	second
S	South
S/C	Spacecraft
SEQ	Sequence
SFOC	Space Flight Operations Center
SIM	System Integration Model (rover and rover payload), also, the GDS Simulation system
sol	One Mars Day (24.6 Earth hours)
SOPC	Science Operations Planning Computer
SPAS	Spacecraft Performance Analysis Software
SPE	Sun-Probe-Earth (angle)
SPV	Single Pressure Vessel (battery)
SSB	Space Studies Board
SSPA	Solid State Power Amplifier
SSR	Solid State Recorder
STL	Spacecraft Test Laboratory
SWG	Software Working Group
Tau	(τ)atmospheric opacity
TBD	To Be Determined
TBR	To Be Reviewed
TCM	Trajectory Correction Maneuver
TCS	Thermal Control System
TDS	Telemetry Distribution System
TID	Total Integrated Dose (radiation)
TIS	Telemetry Input System
TMU	Telemetry Modulation Unit
TPR	Technical Progress Report
TPS	Thermal Protection System
TTACS	Test Telemetry and Command System

UHF	Ultra High Frequency
V	velocity
VCHP	Variable Conductance Heat Pipe
Vdc	volts of direct current
V_H	horizontal velocity
V_V	vertical velocity
W	watt
WEB	Warm Electronics Box
Whr	watt hour
wk	week