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RHU-RPS Long Duration Mars Hard Lander: Meteorology and Seismology Enabled by Radioisotopes (MASER)

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Contents

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1.0 Introduction

Compass Team was tasked by the RPS Project at the NASA Glenn Research Center (GRC) to create an independent concept design for a spacecraft (S/C) utilizing low power radioisotope power (Figure 1.1). A list of candidate missions enabled by milliwatt class radioisotope power was evaluated and thus resulted in the selection of a Mars polar region seismology network of four hard landers. A science rational and science operations plan was developed to establish mission design requirements and S/C subsystem concept designs described herein.

Table 1.1 summarizes the top-level details of each subsystem that was incorporated into the design.

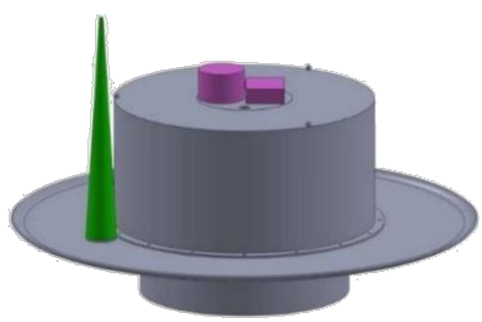


Figure 1.1.—Meteorology and Seismology Enabled by Radioisotopes (MASER) lander.

* Currently retired.

TABLE 1.1.—MISSION AND S/C SUMMARY FOR THE MASER LANDER

Subsystem Area	Details	Total mass with growth (kg)
Top-level system	Purpose: Develop a Design Reference Mission (DRM) to demonstrate how Radioisotope Heater Units (RHU)- radioisotope power system (RPS) systems can enable science missions DRM: MASER are Mars Hard Landers that provide long duration (2 Martian years) science in polar regions Four or more MASER landers (~600 g) encapsulated in an 88 cm aeroshell	19
Mission and operations	Mission: 9 month transit, spun up and separated 1 wk from Mars atmospheric entry (RHU-RPS provide power and heat during the free flight), Aero entry and brought to surface with an 8 m parachute and 6 cm of crushable material, ‘sled’ to handle any horizontal velocity and prevent tipping, deployed ~ 1000 km separation around Phoenix landing site (~65° N)	N/A
Guidance, Navigation, and Control (GN&C)	ALL GN&C is contained in the delivery system Aeroshell and heat system that brakes the MASER into Mars orbit.	N/A
Launch	Launch: TBD – Piggyback or dedicated Mars mission (~70 kg each, 19 kg landed package)	
Science	Science: Deployed seismometer, wind sensor, Temperature and Pressure sensors, optical sensor All run at 100 percent duty cycle except wind and optical sensor (due to high power requirement of the wind sensor > 7 Gb of science data returned for each lander (~ 30 Gb for four landers)	2
Power	Six, Single RHU-RPS (Hi-Z heritage) to provide >220 mW of continuous power – peak power requirements (e.g., 2.5 W communications) supported by ultracapacitors charged by RHU-RPS Alternate RHU-RPS configurations (using multiple RHUs) should be explored	3
Propulsion	No active propulsion system. Aero entry and parachute with crushable material for landing	N/A
Structures and mechanisms	600 g deceleration using parachute (22 m/s vertical impact velocity) and 6 cm thick composite honeycomb, drop down seismometer to ensure contact with surface	7
Communications	Communications: UHF, 8 kbps, two 5 min windows per day through Mars orbiter	0.3
Command and Data Handling (C&DH)	Located in the Aeroshell/Heat Shield module	2
Thermal	internal temperatures kept in -40 to 50 °C (TBR) using the heat from the RHUs and ~2 cm of aerogel	17
Planetary Protection	5 kg bioshield, heat or chemical sterilization as applicable	
Cost	~ $150M for four (RHU-RPS assumed GFE, launch and ops not included), ~$16M recurring for each hard lander	

2.0 Study Background and Assumptions

2.1 Introduction

The purpose of this Compass design session was to develop and enabling DRM for the RHU power systems to drive out requirements for potential further development.

The RHU-RPS Mars Hard Lander Study Goal

Determine utility of different configurations of RHU-RPS for supporting science investigations via power or payload system trades:

- Payload Options
- RHU-Based RPS

  - Number of RHUs
  - Configuration
- Number of RPSs

Investigate payload options to determine the sensitivity of the hard lander system to different payloads and operations concepts

- Option 1: Science Floor
- Option 2: Augmented payload
- Additional instruments
- Increased duty cycle

Trade across RPS size and configuration to determine impacts on mission design.
Focus on one representative hard lander and assume others are identical.
The S/C Carrier or cruise stage is out of the study scope, assuming following characteristics:

- Mass limit per hard lander = 75 kg for total package including entry, descent, and landing (EDL)
  - Could trade mass limit vs. number of hard landers in network
- Mars entry velocity ~ 5.7 km/s (based on previous Pascal small RPS study)

2.1.1 Background

The intent of this study was to identify a mission that might be enabled by very low power (~40 mWe) RPS utilizing the RHU shown in Figure 2.1. The 1 Wth produced by the $^{238}PuO_2$ pellet (2.7 g PuO_2) is converted to ~ 40 mWe by thermoelectric elements. These RHUs have flown in space most notably the Cassini mission and the Huygens Probe on its descent to Titan's surface. These units were also more recently used on Mars Pathfinder Rover and the MER rovers, Spirit and Opportunity.

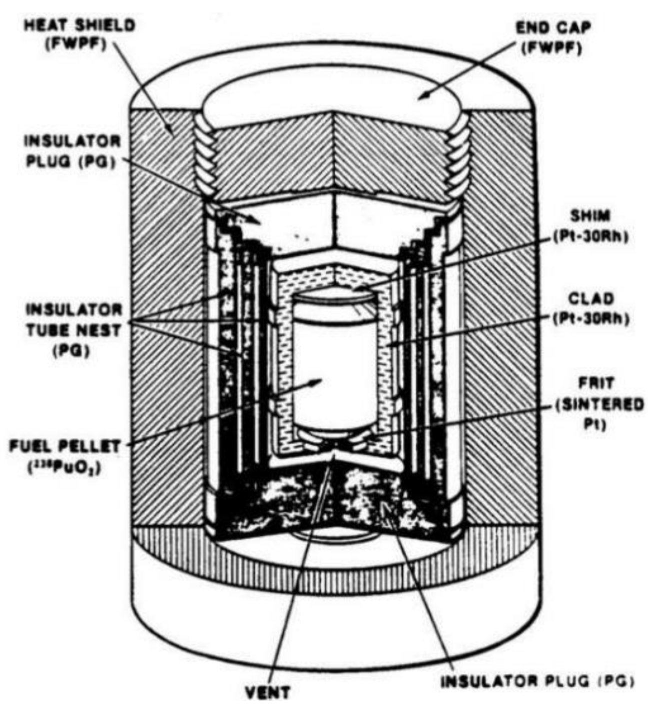


Figure 2.1.—Lightweight Radioisotope Heater Unit.

The RHU capsule is a proven flight component requiring no additional safety testing. Hence this makes it an ideal heat source for a future isotopic power system. The power system concept chosen for this study was the design developed by Hi-Z, Inc., several years ago for a network of Mars landers. While never flown, this design had the most maturity among concepts found in the open literature in that engineering units were built, and high shock load testing showed a unit could survive 1000 to 2000 and possibly higher landing g loads. As science requirements were distilled further, about 240 mWe was needed for the mission with energy storage used for data uplink to a Mars orbit relay.

2.1.2 Report Perspective and Disclaimer

This report captures the study performed by the Compass Team, recognizing that the level of effort and detail found herein will reflect the limited depth of analysis that was possible to achieve during a concept design session. The data generated during the design study and captured within this report is intended as a reference for future work.

2.2 Assumptions and Approach

The assumptions and requirements about the MASER mission and spacecraft design, including those that were known prior to starting the Compass design study session, are shown in Table 2.1. This table gathers the assumptions and requirements and calls out trades that were considered during the design study, and off-the-shelf (OTS) materials that were used wherever possible.

TABLE 2.1.—ASSUMPTIONS AND STUDY REQUIREMENTS

Subsystem area	Assumptions and study requirements	Critical trades
Top-level	Investigate RHU-RPS systems enabling long duration hard landers Figure(s) of Merit (FOMs): Cost, Science returned, 2020-2030 launch window, ~9 month transit, lifetime > 2 Martian years at surface >5 Earth yr (includes assembly, transit, science)	Science suite, landing zones, carrier S/C
System	OTS equipment where possible, TRL 6 cutoff 2018, zero fault tolerant, Mass Growth per ANSI/AIAA R-020A-2006 (add growth to make system level 30 percent)	
Mission and operations, GN&C	~65 kg launch mass, 5.7 km/s, ~ 5 yr life, Spin stabilized descent, parachute final, hard landing <600 g	Descent options, spin stabilized, parachute, crushable structure
Launch vehicle	TBD	
Propulsion	none	Retro thruster
Power	RHU-RPS, >38 mW per RHU-RPS, Landed power average power 230 mW	Four RHU-RPS types, power level, capacitor vs battery, insulation vs vacuum for power system
C&DH, Communications	S/C computer handles all science, flight, S/C operations. Ultra high frequency (UHF) electra-lite class: Data collected over 2 yr mission 7 Gb, ~ two, 5 min uplink day at 8 kb/s	Trade communications data rate (mass, power) vs orbiter passes
Thermal and environment	RHU heat, heaters as needed, thermal insulation	Use of RHU heat, interface with environment (atmosphere and surface)
Science Payloads	Seismometer, Temp/Pressure sensor, optical sensor, accelerometers (only used on descent), wind, <1 kg, avg. power 230 mW, payloads operated on periodically	Magnetometer, camera, neutron-spec, gamma-ray spectrometer
Mechanisms and Structures	Primary: Al-Li, Mechanisms:	Trade Al vs composite for mass savings
Cost	Cost as a piggy-back payload for future Mars missions	
Risk	Identify major Risks	

2.3 Study Summary Requirements

2.3.1 Figures of Merit

- FOMs: RHU Power System (RHUPS) enabling the science mission, cost, mass, data returned, lifetime
- Redundancy: Zero fault tolerant
- Launch year: 2020

2.4 Growth, Contingency, and Margin Policy

2.4.1 Terms and Definitions

Mass — *The measure of the quantity of matter in a body.*

Basic Mass (aka CBE Mass) — *Mass data based on the most recent baseline design. This is the bottoms-up estimate of component mass, as determined by the subsystem leads.*

Note 1: This design assessment includes the estimated, calculated, or measured (actual) mass, and includes an estimate for undefined design details like cables, multi-layer insulation, and adhesives.

Note 2: The mass growth allowances (MGA), and uncertainties are not included in the basic mass.

Note 3: Compass has referred to this as current best estimate (CBE) in past mission designs.

Note 4: During the design study, the Compass Team carries the propellant as line items in the propulsion system in the Master Equipment List (MEL). Therefore, propellant is carried in the basic mass listing, but MGA is ***not*** *applied to the propellant. Margins on propellant are handled differently than they are on dry masses.*

CBE Mass — *See Basic Mass.*

Dry Mass — *The dry mass is the total mass of the system or S/C when no propellant is added.*

Wet Mass — *The wet mass is the total mass of the system, including the dry mass and all the propellant (used, predicted boil-off, residuals, reserves, etc.). It should be noted that in human S/C designs the wet masses would include more than propellant. In these cases, instead of propellant, the design uses Consumables and will include the liquids necessary for human life support.*

Inert Mass — *In simplest terms, the inert mass is what the trajectory analyst plugs into the rocket equation to size the amount of propellant necessary to perform the mission delta-Velocities (ΔVs). Inert mass is the sum of the dry mass, along with any non-used, and therefore trapped, wet materials, such as residuals. When the propellant being modeled has a time variation along the trajectory, such as is the case with a boil-off rate, the inert mass can be a variable function with respect to time.*

Basic Dry Mass	*This is basic mass (aka CBE mass) minus the propellant or wet portion of the mass. Mass data is based on the most recent baseline design. This is the bottoms-up estimate of component mass, as determined by the subsystem leads. This does not include the wet mass (e.g., propellant, pressurant, cryo-fluids boil-off, etc.).*
CBE Dry Mass	*See Basic Dry Mass.*
Mass Growth Allowance (MGA)	*MGA is defined as the predicted change to the basic mass of an item based on an assessment of its design maturity, fabrication status, and any in-scope design changes that may still occur.*
Predicted Mass	*This is the basic mass plus the mass growth allowance for to each line item, as defined by the subsystem engineers.* *Note: When creating the MEL, the Compass Team uses Predicted Mass as a column header and includes the propellant mass as a line item of this section. Again, propellant is carried in the basic mass listing, but MGA is not applied to the propellant. Margins on propellant are handled differently than they are handled on dry masses. Therefore, the predicted mass as listed in the MEL is a wet mass, with no growth applied on the propellant line items.*
Predicted Dry Mass	*This is the predicted mass minus the propellant or wet portion of the mass. The predicted mass is the basic dry mass plus the mass growth allowance as the subsystem engineers apply it to each line item. This does not include the wet mass (e.g., propellant, pressurant, cryo-fluids boil-off, etc.).*
Mass Margin (aka Margin)	*This is the difference between the allowable mass for the space system and its total mass. Compass does not set a Mass Margin; it is arrived at by subtracting the Total mass of the design from the design requirement established at the start of the design study such as Allowable Mass. The goal is to have Margin greater than or equal to zero to arrive at a feasible design case. A negative mass margin would indicate that the design has not yet been closed and cannot be considered feasible. More work would need to be completed.*
System-Level Growth	*The extra allowance carried at the system level needed to reach the 30 percent aggregate MGA applied growth requirement.* *For the Compass design process, an additional growth is carried and applied at the system level to maintain a total growth on the dry mass of 30 percent. This is an internally agreed upon requirement.* *Note 1: For the Compass process, the total growth percentage on the basic dry mass (i.e., not wet) is:* *Total Growth = System Level Growth + MGA*Basic Dry Mass* *Total Growth = 30 percent* Basic Dry Mass*

*Total Mass = 30 percent*Basic Dry Mass + basic dry mass + propellants.*

Note 2: For the Compass process, the system level growth is the difference between the goal of 30 percent and the aggregate of the MGA applied to the Basic Dry Mass.

*MGA Aggregate percent = (Total MGA mass/Total Basic Dry Mass)*100*

*Where Total MGA Mass = Sum of (MGA percent *Basic Mass) of the individual components*

System Level Growth = 30 percent Basic Dry Mass – MGA*Basic Dry Mass = (30 percent – MGA aggregate percent)*Basic Dry Mass*

Note 3: Since CBE is the same as Basic mass for the Compass process, the total percentage on the CBE dry mass is:

*Dry Mass total growth +dry basic mass = 30%*CBE dry mass + CBE dry mass.*

Therefore, dry mass growth is carried as a percentage of dry mass rather than as a requirement for launch vehicle performance, etc. These studies are Pre-Phase A and considered conceptual, so 30 percent is standard Compass operating procedure, unless the customer has other requirements for this total growth on the system.

Total Mass *The summation of basic mass, applied MGA, and the system-level growth.*

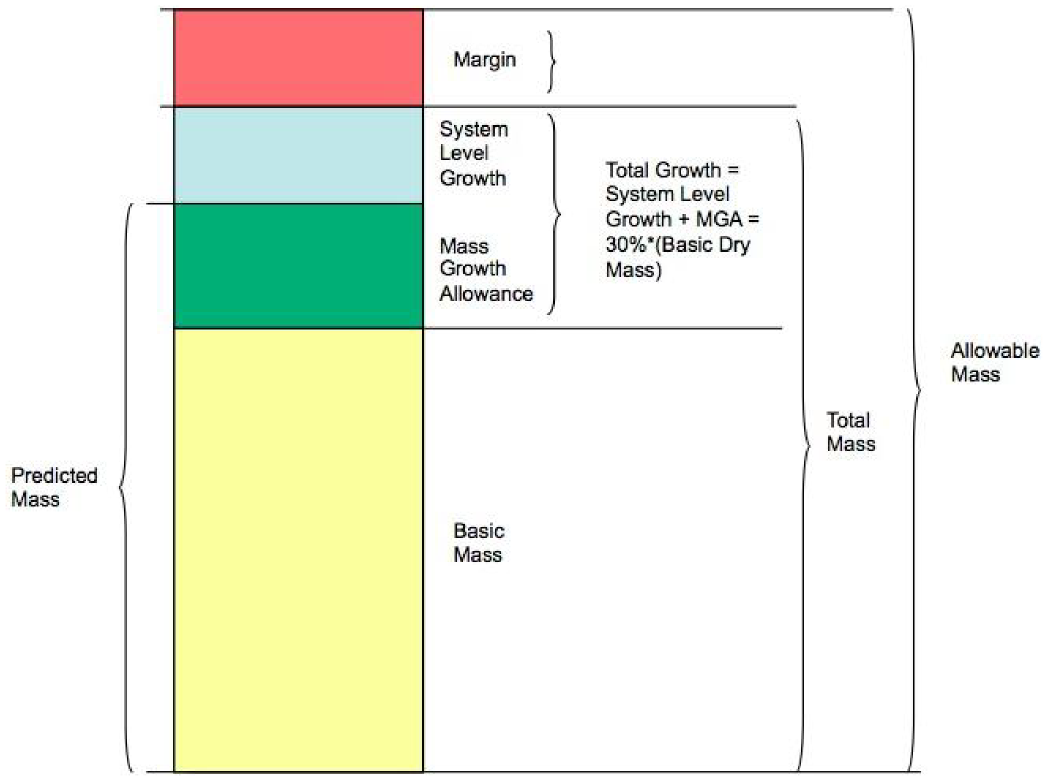
Allowable Mass *The limits against which margins are calculated.*

Note: Derived from or given as a requirement early in the design, the allowable mass is intended to remain constant for its duration.

Table 2.2 expands definitions for the MEL column titles to provide information on the way masses are tracked through the MEL used in the Compass design sessions. These definitions are consistent with those above in Figure 2.2 and in the terms and definitions. This table is an alternate way to present the same information to provide more clarity.

TABLE 2.2.—DEFINITION OF MASSES TRACKED IN THE MEL

CBE mass	MGA growth	Predicted mass	Predicted dry mass
Mass data based on the most recent baseline design (includes propellant)	Predicted change to the basic mass of an item phrased as a percentage of CBE dry mass	The CBE mass plus the MGA	The CBE mass plus the MGA — propellant
CBE dry + propellant	MGA percent * CBE dry = growth	CBE dry + propellant + growth	CBE dry + growth



(Basic = bottoms-up estimate of dry mass)
(MGA = applied per subsystem line item)

Figure 2.2.—Graphical illustration of the definition of basic, predicted, total and allowable mass

2.4.2 Mass Growth

The Compass Team normally uses the AIAA S–120–2006, "Standard Mass Properties Control for Space Systems," (Ref. 1) as the guideline for its mass growth calculations. Table 2.3 shows the percent mass growth of a piece of equipment according to a matrix that is specified down the left-hand column by level of design maturity and across the top by subsystem being assessed.

The Compass Team's standard approach is to accommodate for a total growth of 30 percent or less on the dry mass of the entire system. The percent growth factors shown above are applied to each subsystem before an additional growth is carried at the system level, to ensure an overall growth of 30 percent. Note that for designs requiring propellant, growth in the propellant mass is either carried in the propellant calculation itself or in the ΔV used to calculate the propellant required to fly a mission.

In Table 2.3, a timeline shows how the various mass margins are reduced and consolidated over the mission's life span. The system-integration engineer carries a system-level MGA, called "margin", to reach a total system MGA of 30 percent. This is shown as the mass growth for the allowable mass on the authority to precede line in mission time. After setting the margin of 30 percent in the preliminary design, the rest of the steps shown below are outside the scope of the Compass Team.

TABLE 2.3.—MGA AND DEPLETION SCHEDULE (AIAA S-120-2006) (REF. 1)

Major category	Maturity code	Design maturity (basis for mass determination)	MGA (percent)												
			Electrical/ electronic components			Structure	Brackets, clips, hardware	Battery	Solar array	Thermal control	Mechanisms	Propulsion	Wire harness	Instrumentation	ECLSS, crew systems
			0 to 5 kg	5 to 15 kg	>15 kg										
E	1	Estimated (1) An approximation based on rough sketches, parametric analysis, or undefined requirements; (2) A guess based on experience; (3) A value with unknown basis or pedigree	30	25	20	25	30	25	30	25	25	25	55	55	23
	2	Layout (1) A calculation or approximation based on conceptual designs (equivalent to layout drawings); (2) Major modifications to existing hardware	25	20	15	15	20	15	20	20	15	15	30	30	15
C	3	Prerelease designs (1) Calculations based on a new design after initial sizing but prior to final structural or thermal analysis; (2) Minor modification of existing hardware	20	15	10	10	15	10	10	15	10	10	25	25	10
	4	Released designs (1) Calculations based on a design after final signoff and release for procurement or production; (2) Very minor modification of existing hardware; (3) Catalog value	10	5	5	5	6	5	5	5	5	5	10	10	6
A	5	Existing hardware (1) Actual mass from another program, if hardware will satisfy the requirements of the current program with no changes; (2) Values based on measured masses of qualification hardware	3	3	3	3	3	3	3	2	3	3	5	5	4
	6	Actual mass Measured hardware	No mass growth allowance—Use appropriate measurement uncertainty values												
	7	Customer furnished equipment or specification value	Typically, a "not-to-exceed" value is provided; however, contractor has the option to include MGA if justified												

2.4.3 Power Growth

The Compass Team typically uses a 30 percent growth on the bottoms-up power requirements of the bus subsystems when modeling the amount of required power. The electric propulsion subsystem applies a 5 percent growth to the power requirements needed for the electric thrusters. No additional margin is carried on top of this power growth. The power system assumptions for this study will be show in Table 3.7 in Section 3.2.4.

2.5 Redundancy Assumptions

The S/C is designed to be zero fault tolerant in the design of the subsystems, at least where possible.

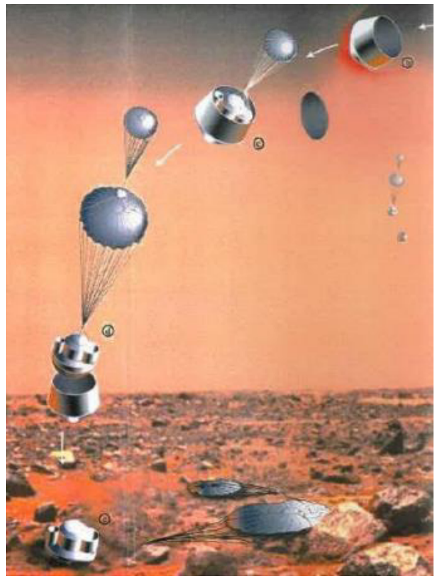


Figure 2.3.—Pascal Hard Lander.

2.6 Mission Description

The Mars Hard Lander Concept Summary is outlined below and represented via the Pascal Hard Lander illustration (Figure 2.3).

- Network of hard landers for a Mars Geophysical and Climate Network
- Long-life seismometry and climate monitoring enabled by RPS, RHU-RPS necessary to fit in EDL aeroshell
- Primary science objectives
    - Characterize the internal structure, thermal state, and meteorology of Mars
- Targeted Measurements
    - Temperature
    - Pressure
    - Seismometry
    - Optical (suspended dust and vapor)
    - Wind

2.6.1 Mission Analysis Assumptions—Baseline Case

The MASER mission assumes that the lander and aerocapture system have been delivered to Mars on some nominal trajectory. The trajectory to Mars is assumed successful and is not modeled explicitly for this design study.

- The mission analysis focused in the EDL maneuvers.
    - From one nominal trajectory, hard land a payload at four specified landing sites
- RPS1 at 75° N 105° W
- RPS2 at 65° N 75° W
- RPS3 at 60° N 110° W
- RPS4 at 65° N 125° W
- Assumptions

- Heat shield
  - 1 m diameter
  - Cd = 1.2
- Main parachute
  - 8.2 m diameter - Sized to ensure impact velocity <= 20 m/s
  - Cd = 0.44 (Assumed similar to Mars Pathfinder)
  - Disk gap band (DGB) with Viking heritage
  - Detailed design with line length, etc., left for future work
- Mars GRAM 2010 atmosphere and wind model
- 50 kg mass entering the atmosphere
- 20 kg laded mass

2.6.2 Mission Details—Baseline Case

Nominal Trajectory:

- Entry parameters assumed similar to Phoenix
  - Velocity = 5.6 km/s
  - Flight Path Angle = –13°
  - Azimuth = 108°
- Landing site at 65° N 100° W - Roughly in the middle of the four landing sites
- Main Parachute Deployment
  - Dynamic Pressure = 400 Pa
  - Altitude = 6.5 km
  - Mach = 1.3
- Max acceleration = 16.2g's
- Landing
  - Ground relative velocity = 22.2 m/s
    - 9.5 m/s horizontal
    - 20.0 m/s vertically
  - Horizontal velocity is largely due to the winds coming from Mars GRAM 2010 for this landing site

2.6.3 Mission Analysis Event Timeline—Baseline Case

Approach:

- Apply ΔV prior to entry to attempt to land the probes at the specified sites
  - 1 wk prior to entry
  - 1 day prior to entry

2.6.4 Mission ΔV Details—Baseline Case: Applying ΔV 1 Week Prior to Entry

- ΔVs applied to land at target sites is between 1.7 and 6.5 m/s (obtained from trial error to hit landing sites, not optimized)
- Mach number at the time of parachute deployment is between 1.15 and 1.56 (in the same range with previous Mars entry vehicles)
- Maximum deceleration experienced is between 16.2 and 17.1g's
- Total time, from entry to descent ranges from 4.5 and 9.1 min

- See Table 2.4 for descent details

2.6.5 Mission ΔV Details—Baseline Case: Applying ΔV 1 Day Prior to Entry

- ΔVs applied to land at target sites is between 11.5 and 55.1 m/s (roughly 5 to 10 times the amount required from applying the ΔV 1 wk prior to entry)
- Unable to find a ΔV that would enable landing at RPS1, starting with the nominal trajectory
    - RPS1 site has the highest latitude
    - It's believed that it requires entering with such a steep flight path angle that the vehicle impacts the surface before the dynamic pressure trigger of 400 Pa is reached to deploy the parachute
    - Could deploy the parachute at a higher dynamic pressure but this is not apples to apples compared to the other trajectories
- See Table 2.5 for descent details

TABLE 2.4.—DESCENT DETAILS 1 WEEK PRIOR TO ENTRY FOR THE SERIES OF LANDING SITES UNDER CONSIDERATION

Parameter	RPS1	RPS2	RPS3	RPS4
Latitude/Longitude (deg)	75° N 105° W	65° N 75° W	60° N 110° W	65° N 125° W
ΔV required (m/s)	1.730	2.025	1.477	6.531
Entry FPA (deg)	–16.5	–9.5	–14.1	–17.1
Max acceleration (g)	16.6	16.2	16.2	17.1
Time, parachute deploy (s)	106	167	119	115
Dynamic pressure, parachute deploy (Pa)	400	400	400	400
Attitude, parachute deploy (km)	3.6	9.6	5.7	3.5
Mach, parachute	1.17	1.56	1.28	1.15
Time, parachute descent (s)	163	382	247	158
Velocity at impact, horizontal (m/s)	14.1	7.6	3.6	7.0
Velocity at impact, vertical (m/s)	19.9	20.1	20.2	20.1
Time, entry to descent (s)	269	549	365	273

TABLE 2.5.—DESCENT DETAILS 1 DAY PRIOR TO ENTRY FOR THE SERIES OF LANDING SITES UNDER CONSIDERATION

Parameter	RPS1	RPS2	RPS3	RPS4
Latitude/Longitude (deg)	75° N 105° W	65° N 75° W	60° N 110° W	65° N 125° W
ΔV required (m/s)		11.500	28.398	55.147
Entry FPA (deg)		–9.4	–13.4	–17.2
Max acceleration (g)		16.2	16.2	17.0
Time, parachute deploy (s)		169.3	123.6	102.9
Dynamic pressure, parachute deploy (Pa)		400	400	400
Attitude, parachute deploy (km)		9.6	6.3	3.5
Mach, parachute		1.56	1.31	1.15
Time, parachute descent (s)		383.7	247	158
Velocity at impact, horizontal (m/s)		7.7	7.0	10.5
Velocity at impact, vertical (m/s)		20.0	20.1	20.0
Time, entry to descent (s)		549	365	273

2.6.6 Concept of Operations (CONOPS)

Table 2.6 outlines the CONOPS for lander assembly, cruise to Mars, EDL, and landed operations. Figure 2.4 and Figure 2.5 illustrate EDL and landed operations, respectively.

TABLE 2.6.—CONOPS FOR ASSEMBLY, CRUISE, EDL, AND LANDED OPERATIONS MISSION PHASES

Assembly to Cruise CONOPS	Cruise CONOPS	EDL CONOPS (Figure 2.4)	Landed Operations CONOPS (Figure 2.5)

• Lander items manufactured in clean facility Class 1000 facility
  ◦ Items on lander that can handle >115°C are bagged and heated individually - RHURPS should be able survive the dry heat microbial reduction protocols.
• Items on lander (known at this time are the energy storage capacitors) that cannot handle >115 °C are bagged and treated with vapor H_2O_2 VHP
  ◦ Capacitors (max temperature 84 °C)—Assume during capacitor manufacturing the internal materials accumulates less than 300 spores/m^2
• Assemble components behind clean wall at PHSF
  ◦ Install RHU-RPS into lander Install lander into aeroshell
  ◦ Fans to cool RHU-RPS if needed
• Install bioshield
• Treat bioshield interior with VHP
• Install bioshield/aeroshell/lander to carrier
  ◦ Cooling needed for above steps—no, 6 W from RHUs very small impact
  ◦ Install harness for power/housekeeping to/from carrier S/C
  ◦ Carrier S/C integrated to launch vehicle - Cooling on pad—probably nothing additional based on the RHUs
• Launch (possible aeroshell integration lander on side—5g) - Jettison the backshell portion of bioshield after launch (heat shield side stays with carrier S/C)
	• Cruise to Mars (~ 9 months)
	  ◦ Power/housekeeping from carrier S/C
	• Separation (~ 1 wk before entry)
	  ◦ Spun ~10 rpm
	• Solo Cruise (~1 wk)
	  ◦ Tones from lander health back to carrier (5 min, twice a day representative) assumes back shell penetration or RF Transparent
	  ◦ Aeroshell CPU powered by batteries
	  ◦ Heat for aeroshell/chute components
	  ◦ Six RHUs from power systems provide approximately enough heat for the system in deep space assuming 10 layers of MLI on aeroshell/backshell
		• Entry angle/speed
		• Acceleration activated drogue chute
		• Separation of backshell
		• Deployment of main parachute (8 m) signaled dynamic pressure or Acceleration
		  ◦ Lanyard for greater chute separation from lander (avoid falling on lander)
		• Separation of aeroshell
		  ◦ All pyros/chute/ processor/ sensors/ batteries on aeroshell so lander not required to run descent systems
		• Acceleration science on descent (5 to 10 min)
		• Impact at ~22 m/s, crushable honeycomb doughnut
		  ◦ 4 to 14 m/s handled by skid plate
		• Chute separated using accelerometer
		• Side motion handled by sled (ratio center of gravity (CG) to radius of sled <1 per past landers to minimize tipping)
		• Deploy seismometer
		  ◦ Free fall from storage tube down to surface
		    ▪ Timer accelerometer activated? Commanded from orbiter?
		  ◦ Protected from 'weather' by crushed honeycomb doughnut
		    ▪ Wind impacts minimized
		  ◦ Isolated from lander by all but power/data lines
		  ◦ No dedicated heating from lander
		• Self-checkout—Up/downlink from orbiter (5 min twice a day)
			• Four landers, 500 km separation distance
			• 1 wk of data storage (~20 Mb)
			• P/T always recorded
			• Pre-Dawns
			  ◦ Turn on optical every hour (detect twilight and frost)
			  ◦ Seismometer
			  ◦ Always on
			  ◦ Take a day's worth, extract data based on corresponding wind data to 'reduce' seismic data returned
			  ◦ Wind Sensor - 8 percent duty cycle
			• Option—Wind sensor periodically (~8 percent on)
			  ◦ If wind, then turn off seismometer
			    ▪ Leave on wind sensor till wind falls off
			    ▪ Then turn back on seismometer
			  ◦ Or take wind data and determine proper frequency—impact on seismometer
			• Upload data during pass
			  ◦ MRO representative—2 downlinks per day—>5 min
			  ◦ Minimal data reduction (event detection only)—but utilize 3 times lossless compression
			  ◦ Communications system turns on ~10 min before planned pass
			    ▪ Orbiter transmits to begin data uplink - Instructs lander to re-upload any data that was in error from lander memory
			    ▪ Lander housekeeping and science uplinked
			    ▪ Orbiter instructs lander when to turn on for next pass

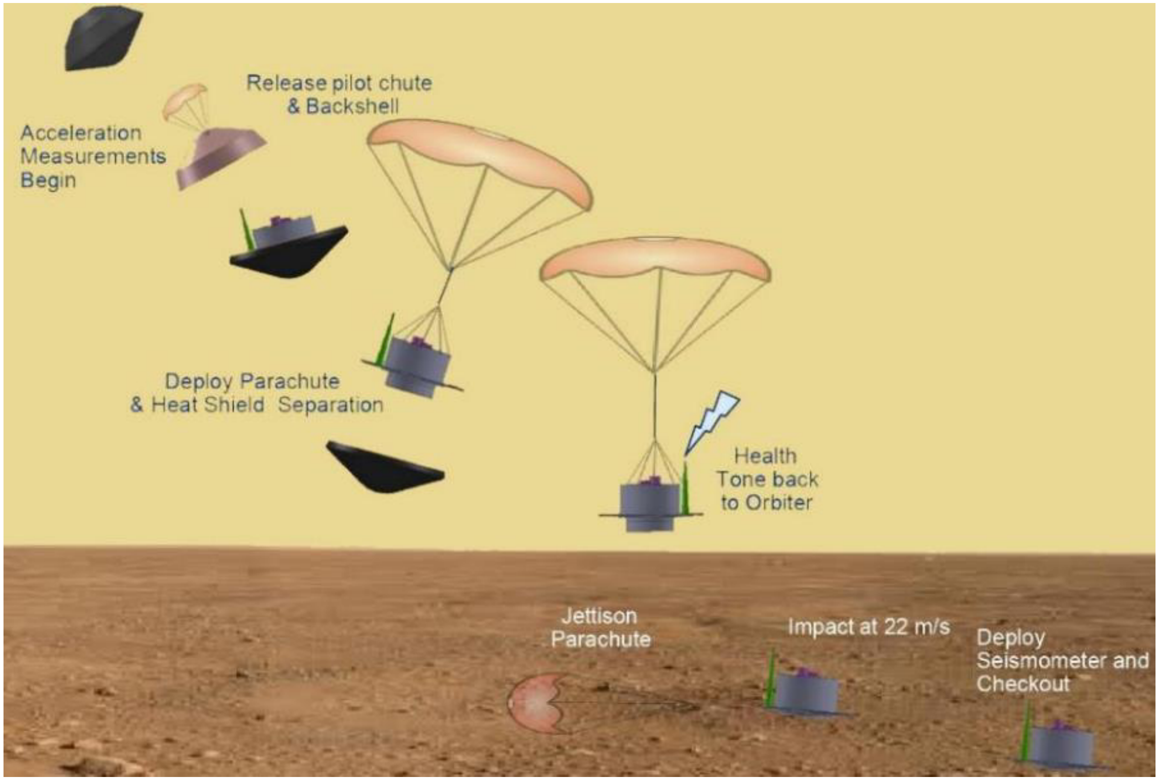


Figure 2.4.—Notional MASER EDL

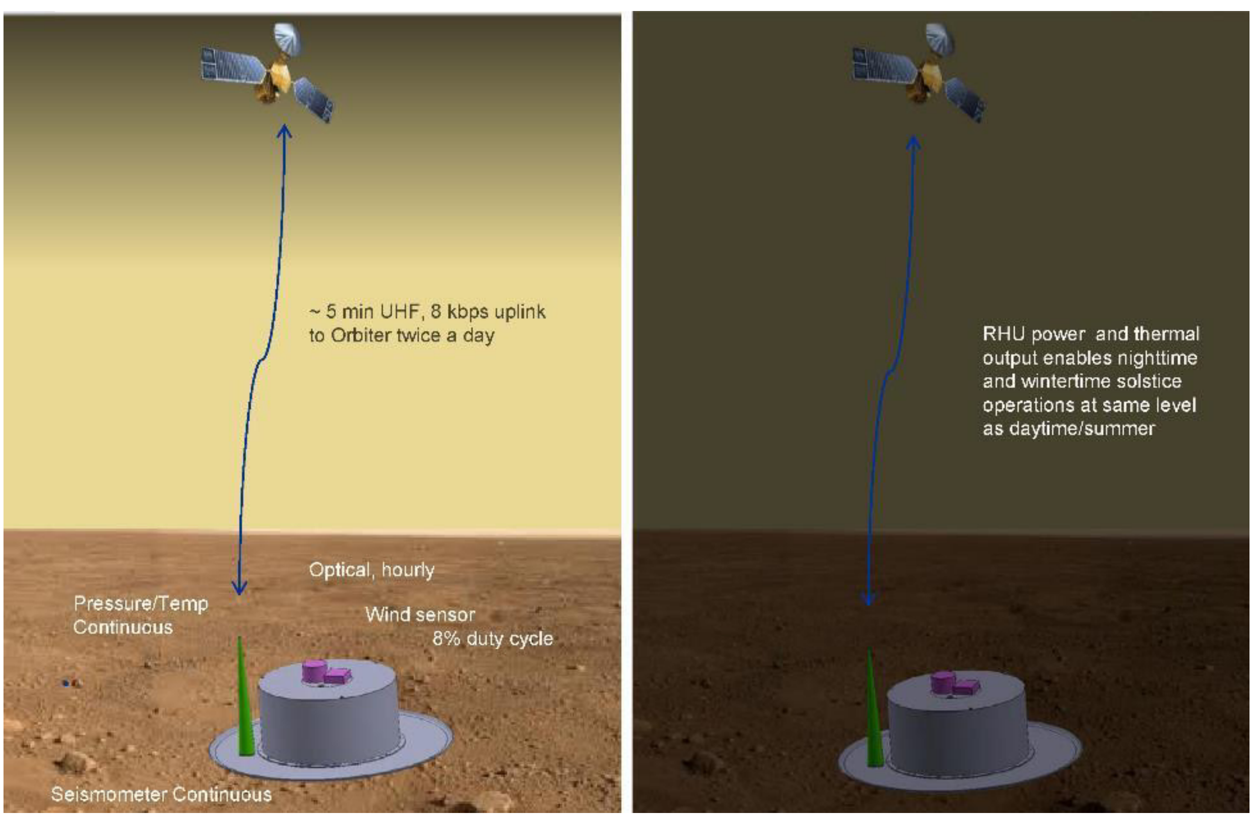


Figure 2.5.—Landed operations.

2.7 Science Rationale

According to “Vision and Voyages for Planetary Science in the Decade 2013-2022” (National Academies Press, 2011) (Ref. 2):

> *Probing the interior is best done through a network of geophysical stations, and such a network has not yet been implemented at Mars.*
>
> *Major progress in understanding Mars’s interior requires obtaining key geophysical data through a network. Seismic data will enhance understanding of the Martian interior structure, including present lithosphere/crust structure and thickness, the current seismic and volcanic activity, the depth of crustal magnetization, the basal structures of the crust under large topographic highs (e.g., Tharsis and Elysium) and lows (e.g., Hellas Basin), and place boundary conditions on models of the early thermal profiles, heat flows and geologic evolution.*

(Note: Insight mission, selected in 2012, will perform single-station measurements at Elysium) MASER’s proposed proximity to Tharsis is therefore highly complementary)

2.8 Launch Packaging Trade Space

The goal in designing the layout of the Mars Hard Lander was to allow it to fit within the smallest aeroshell possible (less than 1-m in diameter), thus allowing multiple landers to fit within the same launch. The required packaged volume of the parachute; required height and area of the crushable landing pad on the lander; the lander’s antenna height; and the height of the RHUs; all drove the sizing of the aeroshell. Further details on these components and their design requirements can be found in their respective sections of this report. Figure 2.6 shows the Mars Hard Lander and parachute canister packaged inside the aeroshell.

A 30° heat shield with a diameter of 88 cm was chosen to allow the CG of the lander to be at or below the separation plane between the backshell and heat shield thus maximizing the stability upon entry and descent. A circular deck was added to the lander that extends outwards to the inside wall of the heat shield. This deck not only provides a good attachment point between the lander and backshell, but also provides a wider landing base to prevent the lander from tipping over upon landing or while on the Martian surface.

A biconic backshell was sized to allow the parachute canister to fit within while maintaining clearance between it and the lander components. This sizing also allowed the lander antenna to fit within the backshell without the need for deployment during any phase of the mission. It should be noted that no deployment of any lander components is required during any phases of the mission except for the seismometer that will be dropped to the Martian surface from inside the lander once the lander is safely on the surface. Figure 2.7 shows the overall dimension of the aeroshell for this mission. Additional isometric views of the lander within the aeroshell can be seen in Figure 2.8.

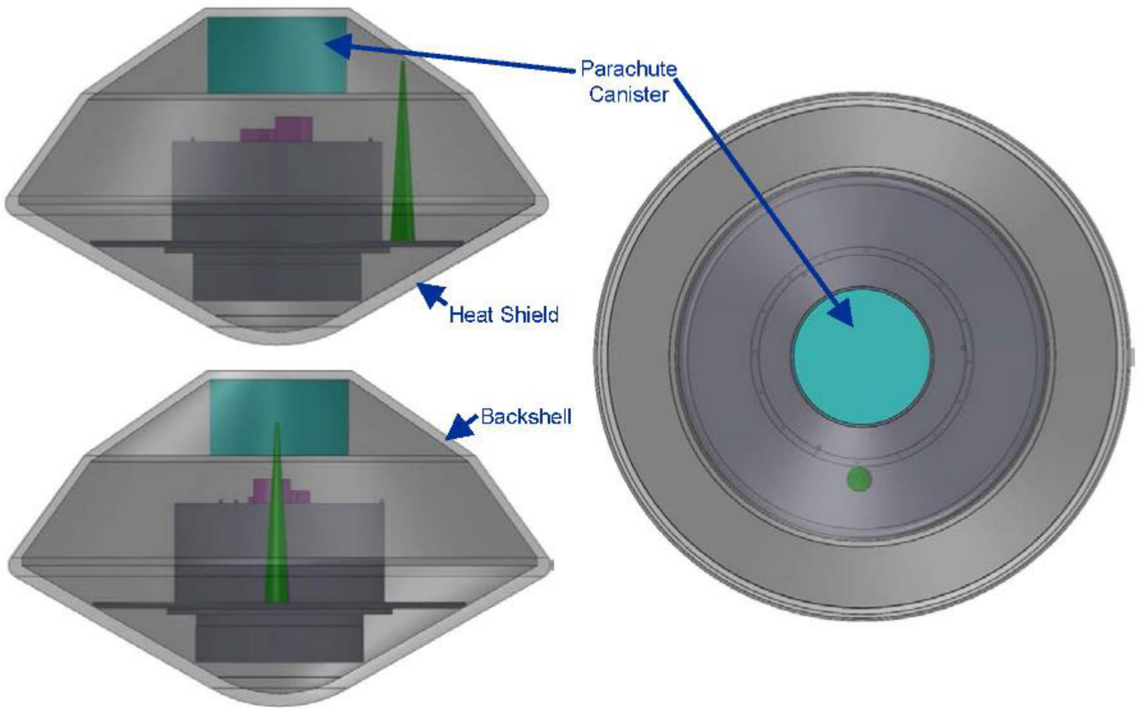


Figure 2.6.—Mars Hard Lander Aeroshell Packaging.

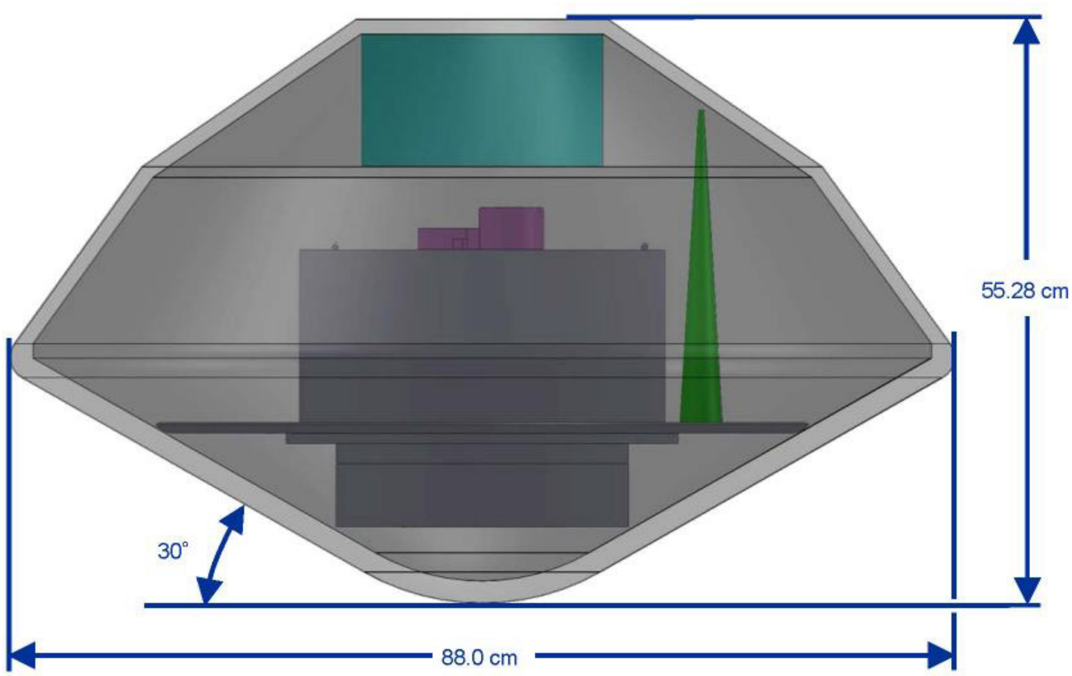


Figure 2.7.—Mars Hard Lander Aeroshell Dimensions.

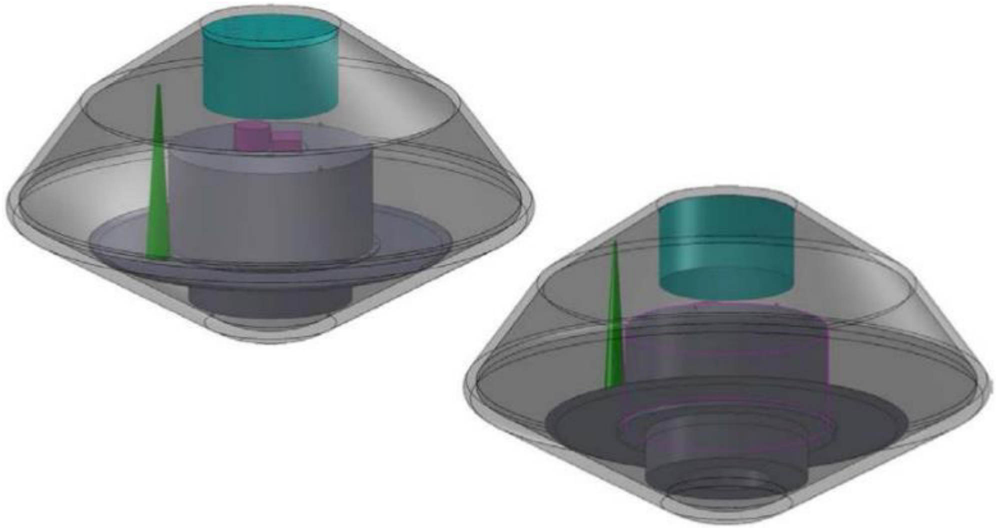


Figure 2.8.—Isometric Views of the Mars Hard Lander within the aeroshell.

3.0 Baseline Design

3.1 System-Level Summary

The system block diagram that captures the theory behind the MASER design is shown in Figure 3.1.

3.2 Top-Level Design Details

3.2.1 Master Equipment List (MEL)

The MASER is composed of three elements and all three are required to fit inside of the same physical volume and to fit inside a total mass allocation as a requirement for this analysis. Therefore, the MEL lists three major elements in terms of the major subsystems within them. The MASER is listed as work breakdown structure (WBS) Element 06. The Hard Lander itself is listed in the MEL as WBS element 06.1. The aeroshell/heat shield, is listed as WBS element 06.2. The carrier interface is listed as WBS element 06.3.

Table 3.1 shows the MEL listing of the Mars hard lander, aeroshell/heat shield and carrier interface as the elements MASER, performed by the Compass Team and documented in this study.

3.2.2 SER Architecture Details

Table 3.2 gathers the top level architecture details of the MASER design. The MASER Packaged inside the EDL Package and had to fit within the performance minus the performance reserve.

3.2.3 S/C Total Mass Summary

The MEL shown in Table 3.1 captures the bottoms-up estimation of CBE and growth percentage of the MASER that the subsystem designers calculated for each line subsystem. The MELs shown in Table 3.3, Table 3.4, and Table 3.5 provide the system summary of the hard lander, aeroshell/heat shield, and carrier interface respectively. To meet the total required mass growth of 30 percent, an allocation is necessary for growth on basic dry mass at the system level, in addition to the growth calculated on each individual subsystem. This additional system-level mass is counted as part of the inert mass to be flown along the required trajectory. Therefore, the additional system-level growth mass impacts the total propellant required for the mission design.

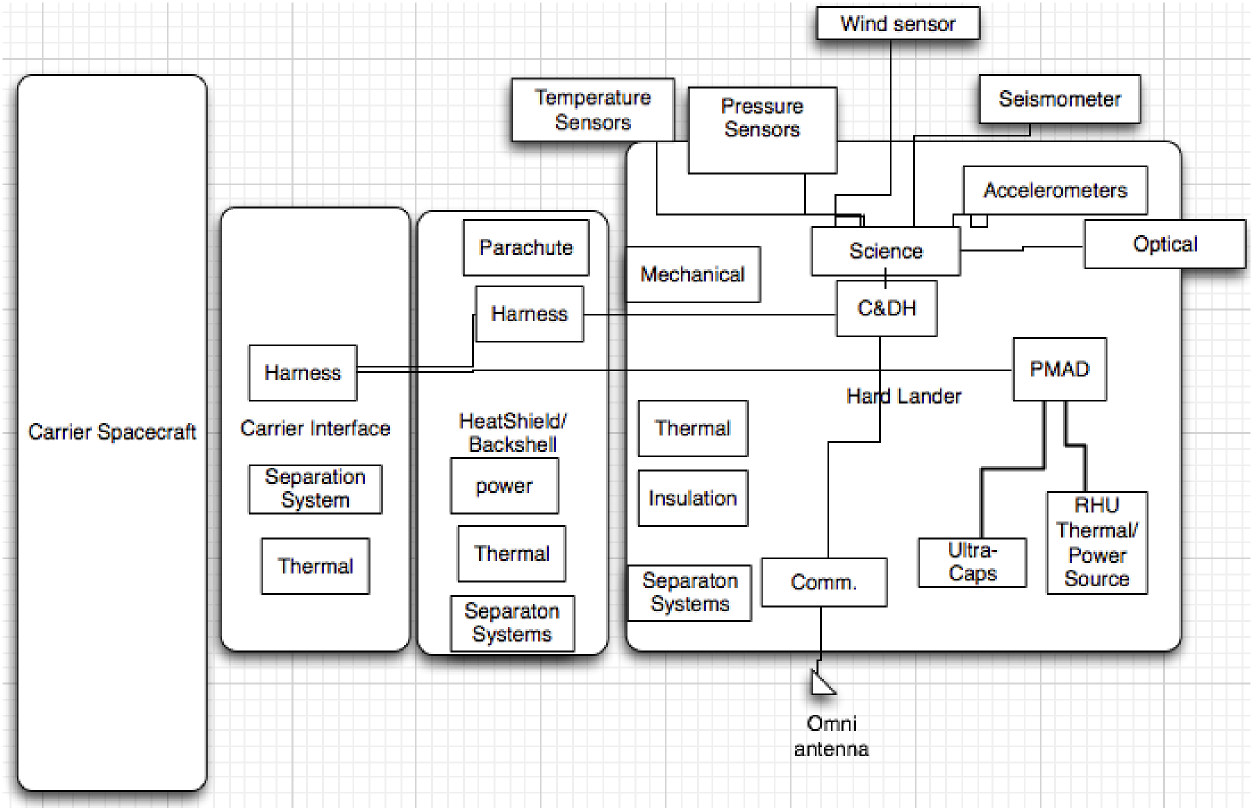


Figure 3.1.—MASER block diagram.

TABLE 3.1.—MASTER MEL WBS FORMAT—BASELINE CASE 1

WBS no.	Description Case #1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Basic mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
06	Mars Hard Lander	50.10	19.6	9.82	59.92
06.1	Hard Lander	14.72	20.7	3.04	17.76
06.1.1	Science	1.47	30.0	0.44	1.91
06.1.3	Command & Data Handling	0.30	0.0	0.00	0.30
06.1.4	Communications and Tracking	0.17	31.2	0.05	0.22
06.1.5	Electrical Power Subsystem	2.65	27.6	0.73	3.38
06.1.6	Thermal Control (Non-Propellant)	0.11	15.0	0.02	0.12
06.1.11	Structures and Mechanisms	10.03	18.0	1.80	11.83
06.2	Aeroshell/Heat Shield	34.83	19.2	6.68	41.51
06.2.1	Science	0.00	0.0	0.00	0.00
06.2.2	Attitude Determination and Control	11.78	30.0	3.53	15.32
06.2.3	Command & Data Handling	2.00	0.0	0.00	2.00
06.2.4	Communications and Tracking	0.00	0.0	0.00	0.00
06.2.5	Electrical Power Subsystem	0.30	0.0	0.00	0.30
06.2.6	Thermal Control (Non-Propellant)	14.72	15.0	2.21	16.93
06.2.11	Structures and Mechanisms	6.02	15.6	0.94	6.96
06.3	Carrier Interface	0.55	18.0	0.10	0.65
06.3.11	Structures and Mechanisms	0.55	18.0	0.10	0.65

TABLE 3.2.—CASE 1 STACK ARCHITECTURE

Architecture details (ELV performance)	
Launch vehicle	EDL Package
Entry V_∞	3.70 km/s
ELV performance (pre-margin)	75 kg
ELV Margin (%)	10%
ELV performance (post-margin)	68 kg
ELV custom adaptor (stays with ELV)	0 kg
ELV performance (post-adaptor)	68 kg
MASER S/C total wet mass	65 kg
Available ELV margin	3 kg
Available ELV margin (%)	4%

TABLE 3.3.—HARD LANDER MEL – BASELINE CASE 1

Spacecraft MEL Rack-up (Mass)-Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013					
WBS	Main Subsystems	Basic Mass (kg)	Growth (kg)	Predicted Mass (kg)	Aggregate Growth (%)
06	Mars Hard Lander	50.1	9.8	60	--------
06.1	Hard Lander	14.7	3.0	18	21
06.1.1	Science	1.5	0.4	2	30
06.1.2	Attitude Determination and Control	0.0	0.0	0	TBD
06.1.3	Command & Data Handling	0.3	0.0	0	0
06.1.4	Communications and Tracking	0.2	0.1	0	31
06.1.5	Electrical Power Subsystem	2.7	0.7	3	28
06.1.6	Thermal Control (Non-Propellant)	0.1	0.0	0	15
06.1.7	Propulsion (Chemical Hardware) - not used	0.0	0.0	0	TBD
06.1.8	Propellant (Chemical) - not used	0.0		0	TBD
06.1.9	Propulsion (EP Hardware) - not used	0.0	0.0	0	TBD
06.1.10	Propellant (EP) - not used	0.0		0	TBD
06.1.11	Structures and Mechanisms	10.0	1.8	12	18
	Element 1 consumables (if used)	0	----	0	--------
	Estimated Spacecraft Dry Mass (no prop, consumables)	15	3	18	21
	Estimated Spacecraft Wet Mass	15	3	18	--------
System Level Growth Calculations Hard Lander					Total Growth
	Dry Mass Desired System Level Growth	13	4	17	30
	Additional Growth (carried at system level)		1		7
	Total Wet Mass with Growth	15	4	19	--------

TABLE 3.4.—AEROSHELL/HEAT SHIELD MEL – BASELINE CASE 1

Spacecraft MEL Rack-up (Mass)-Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013					
WBS	Main Subsystems	Basic Mass (kg)	Growth (kg)	Predicted Mass (kg)	Aggregate Growth (%)
06.2	Aeroshell/Heat Shield	34.8	6.7	41	19
06.2.1	Science	0.0	0.0	0	TBD
06.2.2	Attitude Determination and Control	11.8	3.5	15	30
06.2.3	Command & Data Handling	2.0	0.0	2	0
06.2.4	Communications and Tracking	0.0	0.0	0	TBD
06.2.5	Electrical Power Subsystem	0.3	0.0	0	0
06.2.6	Thermal Control (Non-Propellant)	14.7	2.2	17	15
06.2.7	Propulsion (Chemical Hardware) - not used	0.0	0.0	0	TBD
06.2.8	Propellant (Chemical) - not used	0.0	----	0	TBD
06.2.9	Propulsion (EP Hardware) - not used	0.0	0.0	0	TBD
06.2.10	Propellant (EP) - not used	0.0	----	0	TBD
06.2.11	Structures and Mechanisms	6.0	0.9	7	16
	Element 2 consumables (if used)	0.0	----	0	--------
	Estimated Spacecraft Dry Mass	35	7	41	19
	Estimated Spacecraft Wet Mass	35	7	41	--------
System Level Growth Calculations Aeroshell/Heat Shield					Total Growth
	Dry Mass Desired System Level Growth	35	10	45	30
	Additional Growth (carried at system level)	----	4	---	11
	Total Wet Mass with Growth	35	10	45	--------

TABLE 3.5.—CARRIER INTERFACE MEL – BASELINE CASE 1

Spacecraft MEL Rack-up (Mass)-Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013					
WBS	Main Subsystems	Basic Mass (kg)	Growth (kg)	Predicted Mass (kg)	Aggregate Growth (%)
06.3	Carrier Interface	0.6	0.1	0.6	18
06.3.1	Science	0.0	0.0	0.0	TBD
06.3.2	Attitude Determination and Control	0.0	0.0	0.0	TBD
06.3.3	Command & Data Handling	0.0	0.0	0.0	TBD
06.3.4	Communications and Tracking	0.0	0.0	0.0	TBD
06.3.5	Electrical Power Subsystem	0.0	0.0	0.0	TBD
06.3.6	Thermal Control (Non-Propellant)	0.0	0.0	0.0	TBD
06.3.7	Propulsion (Chemical Hardware) - not used	0.0	0.0	0.0	TBD
06.3.8	Propellant (Chemical) - not used	0.0	----	0.0	TBD
06.3.9	Propulsion (Aux. Hardware) - not used	0.0	0.0	0.0	TBD
06.3.10	Propellant (Aux.) - not used	0.0	----	0.0	TBD
06.3.11	Structures and Mechanisms	0.6	0.1	0.6	18
	Element 3 consumables (if used)	0.0	----	0.0	----
	Estimated Spacecraft Dry Mass	1	0	1	18
	Estimated Spacecraft Wet Mass	1	0	1	
System Level Growth Calculations Carrier Interface					Total Growth
	Dry Mass Desired System Level Growth	1	0	1	30
	Additional Growth (carried at system level)	----	0		12
	Total Wet Mass with Growth	1	0	1	

3.2.4 Total Mass and System Level Margin Calculations

Table 3.6 lists the mass margin calculations for MASER. The Compass Team uses the mass calculations as outlined in Section 2.4.

3.2.5 Power Equipment List (PEL)

Table 3.7 shows the assumptions about the power requirements in all the modes of operation. The power requirements from the bottoms-up analysis on the MASER listed in Table 3.7 are used by the power system designers (described in Section 5.5) to size the power system components.

TABLE 3.6.—CASE 1 MASS GROWTH CALCULATIONS

Mars_Hard_Lander Summary Mass Calculations	Basic Mass (kg)	Growth (kg)	Predicted Mass (kg)	Aggregate Growth (%)
Mars_Hard_Lander Total Dry Mass	50	10	60	---
Mars_Hard_Lander Total Wet Mass	50	10	60	16
Dry Mass Desired System Level Growth	49	15	63	30
Additional Growth (carried at system level)	---	5	------	14
Total Useable Propellant	0	---	0	---
Total Trapped Propellants, Margin, pressurant	0	---	0	---
Total Inert Mass with Growth	50	15	64.61	---
Mars_Hard_Lander Total Wet Mass with system level growth	50	15	65	---

TABLE 3.7.—MASER CASE 1 PEL

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Power Mode 1 (W)	Power Mode 2 (W)	Power Mode 3 (W)	Power Mode 4 (W)	Power Mode 5 (W)	Power Mode 6 (W)	Power Mode 7 (W)
	Power Mode Name	Launch	Cruise	Separation	Descent	Science Day	Science Night	Communications
	Power Mode duration	20 min	9 months	1 Week	10 min	12 hr	12 hr	10 min 2x/day
06	Mars Hard Lander							
06.1	Hard Lander	0.05	0.05	0.09	2.56	0.12	0.12	2.62
06.1.1	Science	0	0	0	0.01	0.071	0.071	0.071
06.1.2	Attitude Determination and Control	0.000	0.000	0.000	0.000	0.000	0.000	0.000
06.1.3	Command & Data Handling	0.050	0.050	0.050	0.050	0.050	0.050	0.050
06.1.4	Communications and Tracking	0.000	0.000	0.035	2.500	0.000	0.000	2.500
06.1.5	Electrical Power Subsystem	0.000	0.000	0.000	0.000	0.000	0.000	0.000
06.1.6	Thermal Control (Non-Propellant)	0.000	0.000	0.000	0.000	0.000	0.000	0.000
06.1.11	Structures and Mechanisms	0.000	0.000	0.000	0.000	0.000	0.000	0.000
06.2	Total	0.050	0.050	0.085	2.560	0.121	0.121	2.621
	Aeroshell/Heat Shield							
06.2.11	Structures and Mechanisms	0.000	0.000	0.000	2.802	0.000	0.000	0.000
	Total	0.000	0.000	0.000	2.802	0.000	0.000	0.000
	Power, System Total	0.0500	0.0500	0.0850	5.3625	0.1210	0.1210	2.6210
	30% growth	0.0167	0.0167	0.0283	1.7873	0.0403	0.0403	0.8736
		0.0667	0.0667	0.1133	7.1498	0.1614	0.1614	3.4946

3.3 Concept Drawing and Description

The configuration of the Mars Hard Lander was driven by the requirement to fit it within the smallest aeroshell possible, allowing multiple landers to be on the same launch. This led to a lander design that has a maximum diameter of 60.87 cm and a maximum height of 39.49 cm pre-landing (35.39 cm after landing). Overall dimensions of the Mars Hard Lander are shown in Figure 3.2. Details on the launch packaging and resultant aeroshell design are discussed in Section 2.8 (Launch Packaging Trade Space) earlier in this document.

The primary structures that comprise the Mars Hard Lander include a crushable donut shaped landing pad, a structural deck, bus structure, and an external shell. The donut shaped landing pad is an Al honeycomb structure that has an external diameter of 27.25 cm and an internal diameter of 15.00 cm and is 6.00 cm thick. These dimensions provided the proper surface area and thickness to allow the lander to be slowed to 600 g upon landing, while providing an internal area large enough to allow the seismometer located within the center of the lander to be deployed/dropped directly to the surface after landing. The landing deck has an external diameter of 60.87 cm and a hole in the middle that is 11.87 cm in diameter that allows the seismometer to be dropped/deployed to the surface after landing. The deck also varies in thickness with the thickest portion in the middle moving outward to the thinnest portion. The thickest portion of the deck has a diameter equal to a circle that encompasses all the components within the bus structure, on top of the bus structure, and the power components located around the bus structure. This thickness was sized to eliminate any bending of the deck from the loads put on it by all the bus and power components during the 600 g landing. The bus structure itself is a cylinder with internal decks for mounting those components contained within. The next thickest portion extends out from the thickest portion to a diameter equal to the flange diameter on the bottom of the external shell. This external shell is mounted directly to the top of the deck and provides a protective cover for the aerogel insulation surround the power components as well as providing the interface to the parachute used during descent. The thickness of the deck in this area was sized to eliminate any bending of the deck from the shock forces placed on the lander when the parachute is deployed as well as from the forces of the outer shell and insulation upon the 600 g landing. The deck is significantly thinner from the outer shell diameter out to its maximum diameter of 60.87 cm. This section provides mounting for the antenna, acts as a “skid-plate” for landing to help prevent the lander from tipping over due to any horizontal velocity present upon landing, and provides a large footprint diameter ensuring that the lander will not fall over while on the surface. A section view of the lander can be seen in Figure 3.3 showing all the structural elements discussed. A detailed discussion on the structural design can be found in Section 5.7.

Those components located externally on the Mars Hard Lander include the UHF antenna, the wind sensor, the temperature sensor, the optical monitor, and the outer shell containing the three parachute attachment points. While the outer aerogel insulation is contained within the outer shell and is technically an external component, it will be mentioned here as it does enclose all the other internal components on the lander. All these external components can be seen in Figure 3.4.

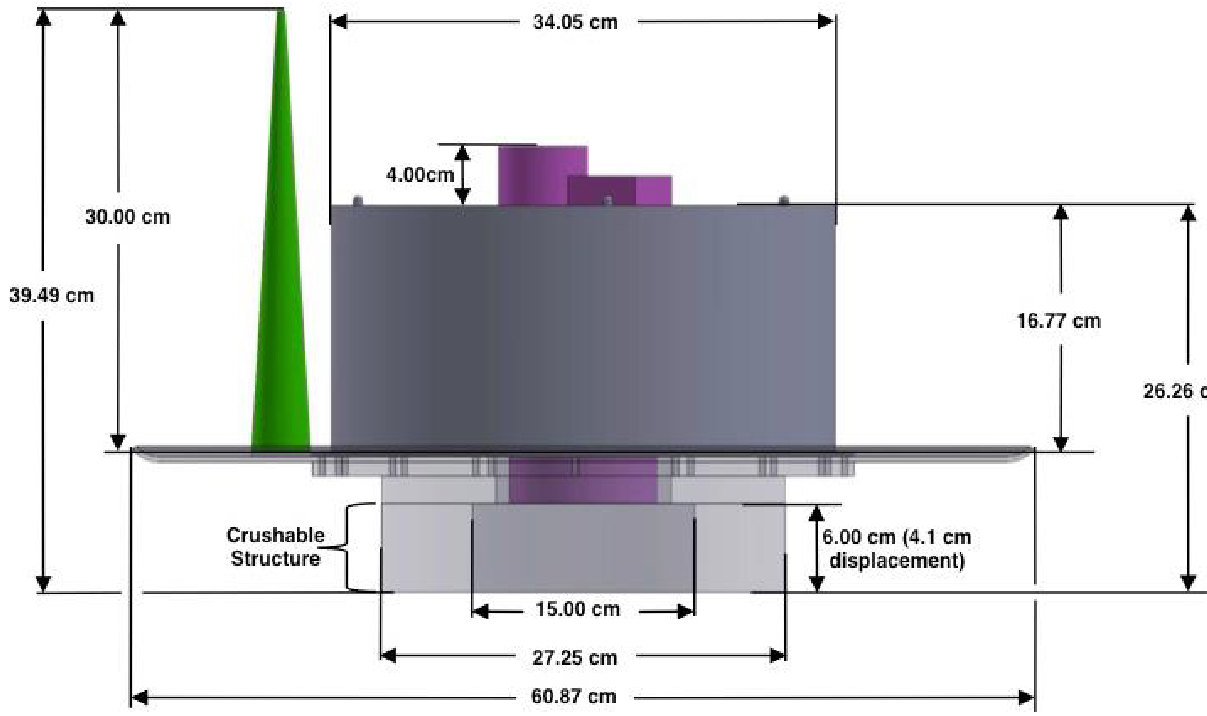


Figure 3.2.—Mars Hard Lander Overall Dimensions.

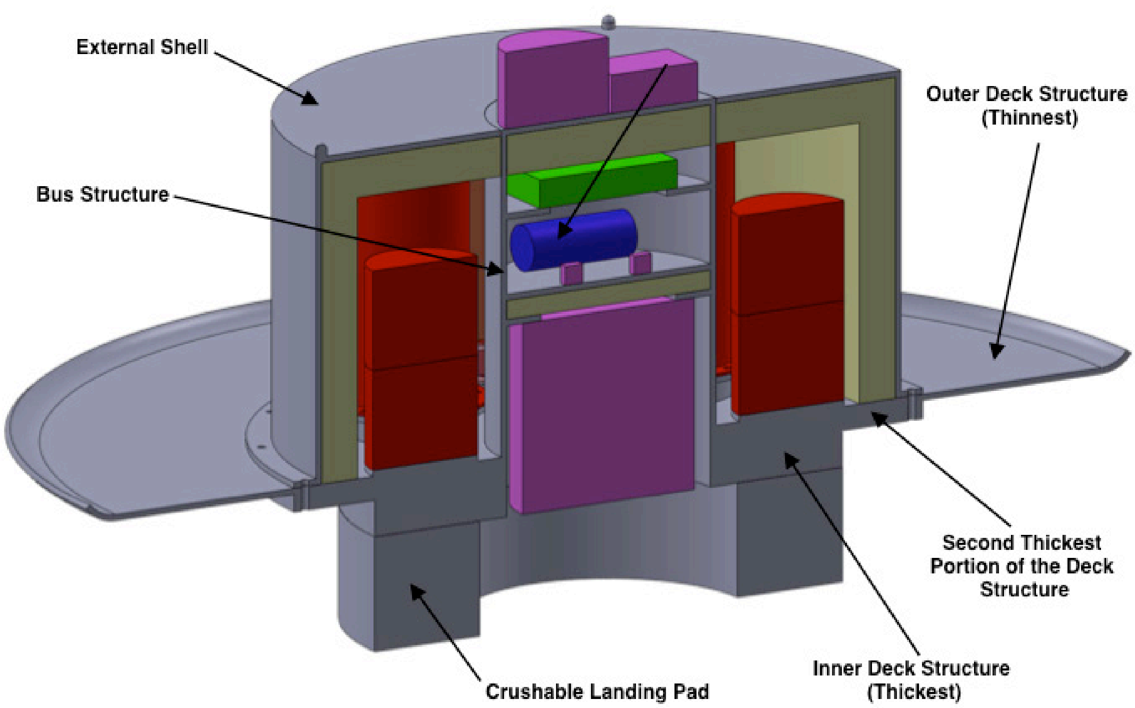


Figure 3.3.—Structural elements of the Mars Hard Lander.

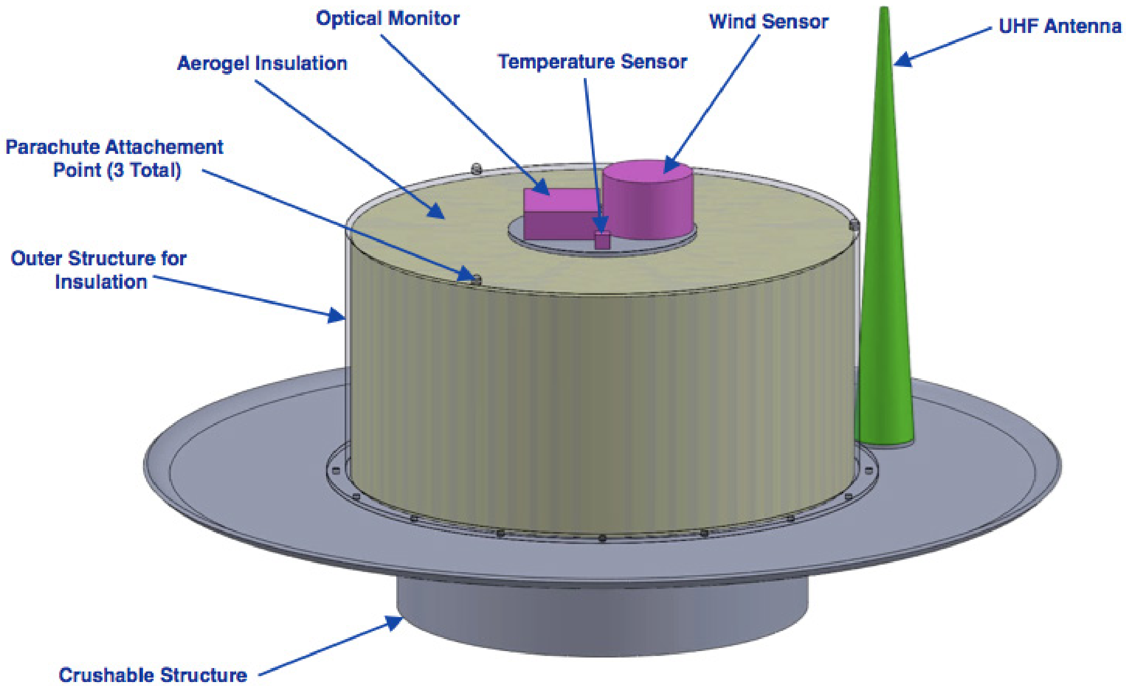


Figure 3.4.—Mars Hard Lander External Components.

The UHF antenna is mounted directly to the deck and pointed upwards in a direction perpendicular from the deck. This allows a clear view of the Martian sky while on the surface. Potential interference from the outer shell needs to be examined further. Raising the antenna up off the deck to eliminate any potential interference would not allow the lander to fit in the current sized aeroshell and deploying the antenna after the backshell is jettisoned would add risk to the mission. A different antenna design could also eliminate any potential interference from the outer shell. The current location of the antenna and antenna design was deemed sufficient for the scope of this study.

The wind sensor, temperature sensor, and optical monitor are all part of the suite of science instruments on the Mars Hard Lander. All three of these components are mounted to the top of the cylindrical bus structure to ensure maximum exposure to the winds, atmospheric temperature, and provide an unobstructed view of the Martian sky.

The outer shell structure is mounted directly to the deck structure and serves several purposes. It provides a "mount" for the aerogel insulation that covers the power components while protecting the aerogel insulation from the elements. It also provides the three attachment points for the parachute and carries the shock load from the parachute upon deployment.

The electrical power system components are comprised of six HiZ RHUs and four Ultra Capacitors. These components are all shown in Figure 3.5. All six of the RHUs are mounted to raise flanges located on the deck. The four Ultra Capacitors are stacked on top of one another in two pairs and mounted directly to the deck structure. All ten of these components are located around the circumference of the cylindrical bus structure allowing them to provide heat to all the electronics contained within the bus.

Those components that are located inside the cylindrical bus structure include the UHF transceiver; the memory module and computer for the C&DH system; and a suite of science instruments including a pressure sensor, temperature sensor, impact and entry accelerometers, and a seismometer. All these components, except for the seismometer have a layer of aerogel insulation above and below their location to help maintain the proper working temperatures throughout the duration of the mission. These internal components are shown in Figure 3.6.

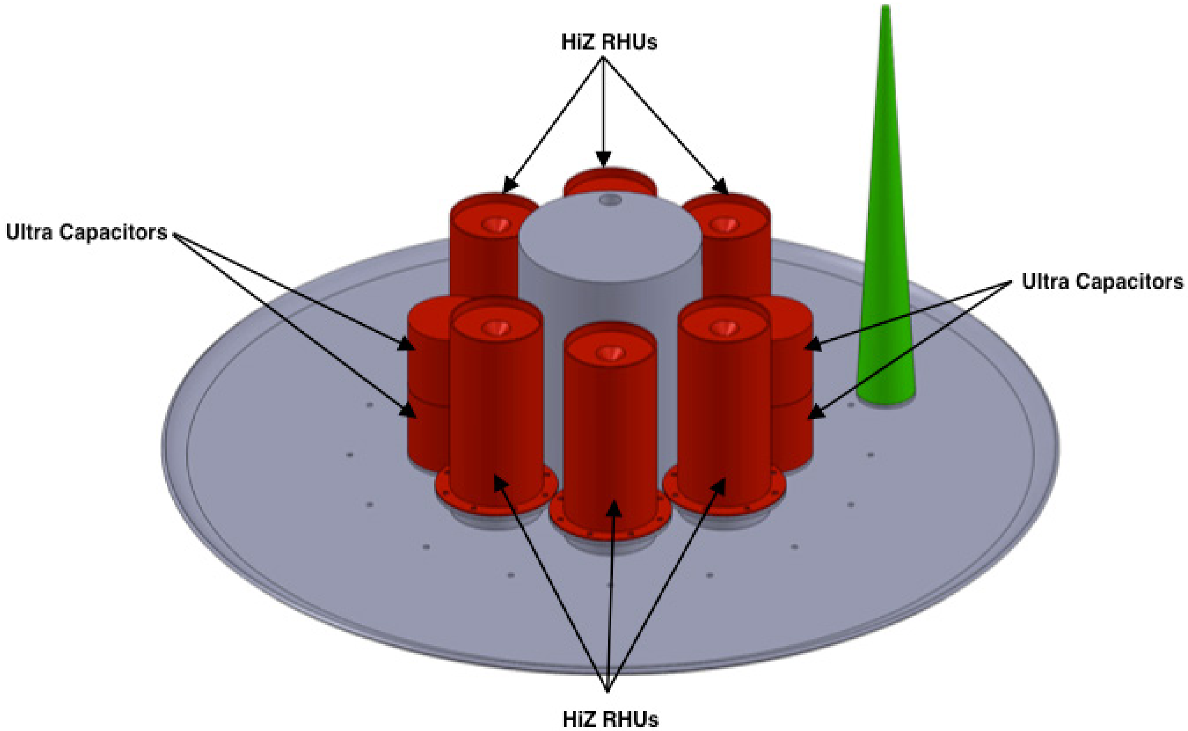


Figure 3.5.—Mars Hard Lander Electrical Power Components.

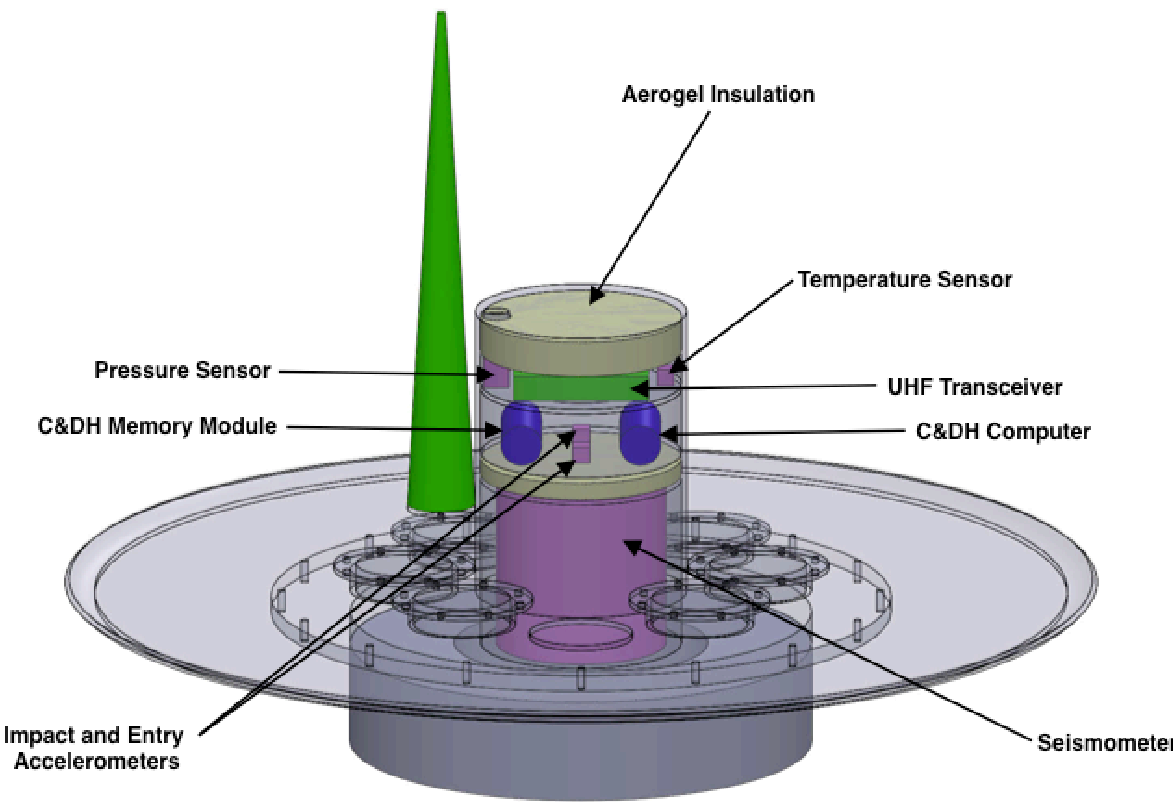


Figure 3.6.—Mars Hard Lander Electrical Power Components.

Inside the top of the cylindrical bus structure is a layer of aerogel insulation. Below this upper layer of insulation is a deck structure to which the UHF transceiver, temperature sensor, and pressure sensor are mounted. A hole is cut through the insulation and top of the bus structure leading down to the pressure sensor. This allows the sensor itself to be in a thermally controlled environment yet allowing the sensor to measure the pressure of the Martian atmosphere. The temperature sensor is used for measuring the temperature within the bus itself.

Below this upper deck is another deck that is used to mount the impact and entry accelerometers and the memory module and computer of the C&DH system. Both accelerometers are located as close to the centerline of the bus as possible to obtain more accurate data during entry and the hard landing. The computer and memory module are mounted to the deck on either side of the accelerometers to balance the mass of these two components. Below this deck is another layer of aerogel insulation which when combined with the top layer as well as the insulation within the outer shell, thermally isolates these electronics from the outside Martian atmosphere.

The seismometer, while not contained within the thermally isolated portion of the bus, is located within the cylindrical structure of the bus, and mounted to a deck located directly below the lower layer of insulation. This location was selected because it shortened the overall height of the lander, but still allowing the seismometer to be deployed to the Martian surface after being released and falling through the hole within the deck structure and crushable landing pad.

Figure 3.7 shows a transparent view of the Mars Hard Lander. Additional images of the lander design can be found in Appendix C.

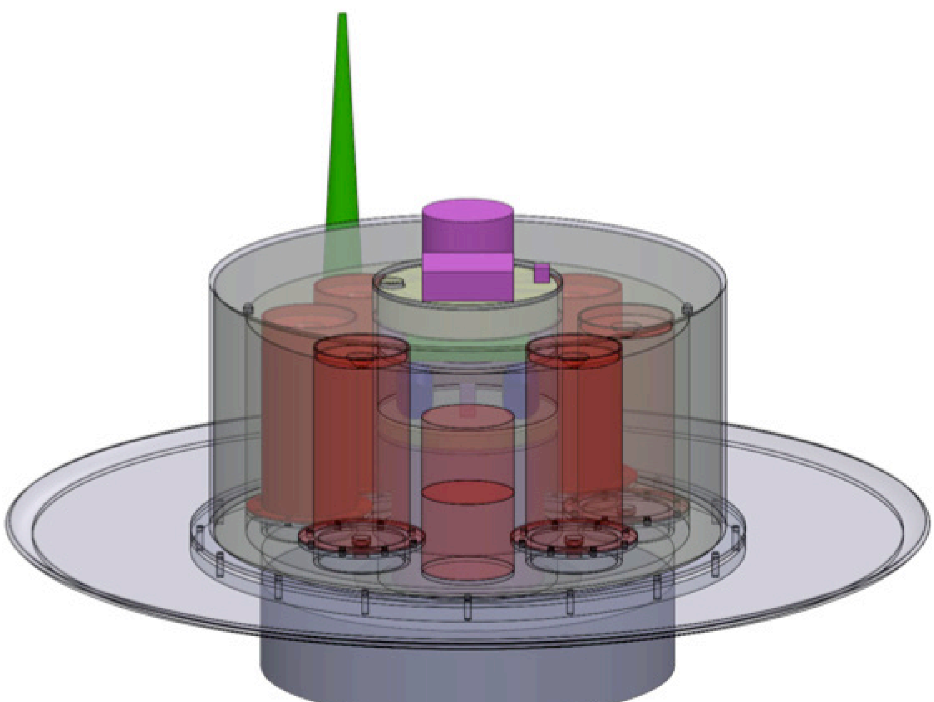


Figure 3.7.—Transparent view of the Mars Hard Lander.

4.0 Lessons Learned and Areas For Future Study

4.1 Lessons Learned

- Even at ¼ W of power RHU-RPS systems can enable hard landers that house long duration sensors in challenging environments
    - Power/heat enables night-time operations
    - Power/heat enables polar winter operations
    - Power/heat simplifies in space free flight (no solar arrays/batteries needed after carrier separation 1 wk before entry)
- The heat from the RHU-RPS, combined with capacitor systems and low temperature tolerable electronics (–40 °C) are as important as the power output
- Due the insulation required the RHU-RPS power system dominates the interior volume of the lander
- Baseline design used six, single RHU-RPS; different building blocks of two, four, or six RHUs RPS would decrease the number of RPS needed and would save some volume but may not allow for as even heating of internal systems

4.2 Forward Work

- Apply RHU-RPS to other missions to determine mission specific requirements
    - Titan, lunar, asteroid, comet, Moon, nanosat?
- Determine best RHU-RPS building block/design
    - Evaluate a multiple RHU-RPS design for MASER to determine potential packing efficiency and heat rejection requirement mods.

5.0 Subsystem Breakdown

5.1 Science Package

- Accelerometers on only during entry/descent
- Optical monitor has very low duty cycle, negligible energy requirement
- Pressure sensor always on (for stability) 1 mW
- *Seismometer*—Allocation of 49 mW for continuous operation without force-feedback
- *Anemometer*—20 mW allows 8 percent duty cycle of Beagle-2 type thermal anemometer

The science instrumentation suite is detailed in Table 5.1.

5.1.1 Science Requirements

While the Decadal Survey scientific priority for 2013-2022 is stated to be sample return, the present study was to specifically consider science enabled by small RPS. This is particularly suited to long duration missions of low power (for short durations, primary batteries may be competitive) in regions where solar power may be unavailable for extended periods. Seismology and meteorology at Martian high latitudes is therefore a good fit.

5.1.2 Science Assumptions

The principal instruments are a seismometer and a meteorology package, see Table 5.1. Other possible payloads were considered (e.g., camera, magnetometer, mineralogy experiments), but were considered of lower scientific priority, too resource-demanding, or incompatible with a small hard-landed vehicle. The meteorology package includes an accelerometer for entry measurements, pressure and temperature sensors, an optical monitor for dust/water vapor/cloud measurements and a wind sensor.

The seismometer is not required to be an elaborate and delicate broadband instrument, such as that to be flown on the InSight mission. The 'added value' of a seismometer network is in the identification of source locations and propagation speeds and thus a large, long-period seismometer is not justified. Short-period devices can be small, simple, and robust: an end-member is an entirely passive geophone (the Ranger moon landers were to have geophone-type instruments, which could tolerate 2000 g), although modern signal conditioning techniques allow higher sensitivity. It would be desired to have the instrument operating continuously and thus instrument power is a key consideration: force-feedback instruments presently qualified for space have powers that are too high. Advanced conditioning on geophone-type instruments (e.g., Lennartz LE-1DV) yields quite good (3 nm/s) noise levels, yet with a (non-space-rated) power draw of 36 mW: we therefore allocate 50 mW to the measurement.

The principal meteorological measurement would be a pressure sensor. Time series pressure data yields insight on the annual CO_2 frost cycle, atmospheric waves, the passage of frontal systems, and the presence of dust devils. The combination of time series from a regional network allows the propagation of waves and systems to be observed by cross-correlation and the influence of terrain on local weather to be exposed. Compact and accurate pressure sensors have flown on Pathfinder (Ref. 3). This sensor has a minimal power draw and would operate continuously for maximum measurement stability. Semiconductor or thermocouple temperature sensors would be installed at a few locations on the vehicle. Although it is recognized that some lander thermal perturbations will be inevitable, azimuth diversity ensures at least on sensor will be upwind. A deployable mast would obviate such issues but is challenging to accommodate on a hard-landing vehicle.

TABLE 5.1.—SCIENCE INSTRUMENTATION SUITE

Instrument	Measurement/Rationale	Basis	Mass (kg)	Dimensions/ Configuration/Mounting
Pressure / Temperature	Seasonal pressure cycle, atmospheric tides, cyclonic systems, dust devils. MEMS diaphragm pressure sensor or ion current gauge	Phoenix, Mars-96	0.07	Internal sensor, enclosure must be vented. Stable temperature essential. 1.5- by 2- by 2-cm / 1- by 1- by 1-cm
Seismometer	Seismic monitoring (short period seismic signals only). MEMS micro-seismometer or Ranger/Lunar-A geophone type.	Lunar-A, Ranger, Insight	0.5	Forebody (for minimal wind effects and maximum seismic coupling). 10- by 10-cm diameter
Optical Monitor	Set of windowed up-looking photodiodes/filters to measure UV/near-IR light levels for water vapor, cloud, dust loading	Beagle / Mars-96/ MSL	0.1	Top side, sky view 2- by 6- by 5-cm
Accelerometer Package	MEMS. Atmosphere profile during entry/descent. Surface mechanical properties; post-impact tilt	DS-2	0.05	Entry/Tilt accel near CG. Impact accel in forebody 1 cm^3 each.
Wind	Hot film anemometer. Seasonal, synoptic and diurnal weather systems, dust devils, and gusts.	Beagle/ MSL	0.15	Top side, minimal azimuthal obstruction 4- by 6-cm diameter

An optical monitor comprises simply a set of photodiodes with wavelength filters and collimating masks to measure the direct and scattered solar beam, at a set of wavelengths to discriminate dust and ice, and (via differential absorption in a water band, e.g., at ~900 nm) the column abundance of water vapor. Such monitors were developed for Netlander and Mars 96, and flew on Beagle 2 and MSL. Brief measurements (e.g., a few seconds once/hour) would be made.

Wind speed and direction is in some ways the most important measurement, in that as well as the scientific information it provides, it contributes to the (presently very small) inventory of surface wind speed measurements that are of importance in EDL and surface operations for future missions. Additionally, the wind data will provide a quality flag for seismometer data which will (despite the direct emplacement of the seismometer on the ground and the shielding afforded by the lander body) increase the seismic noise background via wind loads on the ground and lander. Seismic signals may also be correlated (as on the moon) with solar heating of the ground, and with passing pressure systems.

Wind sensing in the thin Mars atmosphere is challenging - mechanical wind sensors are difficult to use (especially given the landing scenario), and ultrasonic methods require computationally demanding cross-correlation which drives the required power. Ion anemometers and thermal (hot-film or hot-wire, as flown on Pathfinder (Ref. 3)) anemometers, and optical wind/dust sensors are all well-suited mechanically to this application, and relatively low-power, but this still means a draw of 0.25 to 0.5 W which is prohibitive. Some entirely passive possibilities exist to detect high winds (e.g., a whisker driving a flexible piezoelectric element) but are not likely to be quantitatively accurate. Another possibility that would require modest mechanical development might be a telltale indicator like that on Phoenix, but with an optical or magnetic sensing system rather than an imaging-based one. Pending such a development, the Compass Team considers that the wind sensing objective can be adequately met by a Beagle-2 type wind sensor operated at a modest duty cycle (presently ~8 percent but could be scaled up after the on-Mars power budget is better understood and margins can be released.

5.1.3 Science Design and MEL

Seismometer details

- Force-feedback systems (e.g., Insight, Mars-96, Lunar-A) may draw 100 to 150 mW. MEMS microseismometer quoted at 100 mW. So could operate these systems at 50 percent+ duty cycle
- But would really like 100 percent duty cycle to allow for post-hoc examination of times during which event is detected at other stations (regardless of wind veto, etc.)
- Simple geophone is passive, you pay only for the amplifier (assuming lander CPU handles digitization, etc.)
- Lennartz LE-1DV MkII single-axis seismometer (1.1 kg) achieves 3 nm/s noise at 1 Hz with 36 mW. Suggest allocating ~50 mW for continuous operation of a space-rated version.
- Faced with the challenge of a 50-mW allocation, the instrument community could likely respond with a 3-axis solution with better sensitivity/long period performance than this, but this type of instrument meets the main science goal.

Wind sensor details

- Beagle 2 wind sensor = 250 mW (Pathfinder anemometer of similar concept—thermal anemometry—drew 380 mW) It is fundamentally harder to do much better with this technique at Mars.
- Mechanical anemometers perform poorly in the thin Mars atmosphere. In addition, there are impact shock/uncaging issues.

- Ultrasonic anemometry is tough in thin CO_2 but can do well with cross-correlation (but this has a CPU power demand).
- Ion anemometer on Mars-96 demanded 150 mW
- Optical dust monitor/anemometer by Merrison demands 0.4 W
- Some other techniques (passive acoustic?) could be examined, but have a low TRL
- Recommend use of the Beagle 2 device with 8 percent duty cycle. Period to be determined after a study of wind speed persistence and warm-up time of instrument

The full science payload, summarized in the MEL for the MASER in Table 5.2, consists of a series of wind, temperature, and pressure sensors as well as seismometers, and other monitoring devices.

Science Phase and Uplink

- Science phase duration has minimum of 1 Mars yr and baseline of 2 Mars yr
  - Need at least 1 yr to observe seasonal changes; 2 yr is better.
- The study baseline for duty cycles is shown in Table 5.3
- Total data with margin before compression: 5.6 Mbit/d
- Data will be uplinked to relay vehicle on cadence as a function of bandwidth available during each relay.

Science Data and Telecom Rate

- >7 Gb returned in 2 yr from each Hard Lander
- A group of four MASER landers would provide almost 30 Gb of seismic and weather data from Mars polar regions

TABLE 5.2.—SCIENCE BASELINE MEL

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	----	------	50.10	19.6	9.82	59.92
06.1	Hard Lander	----	------	14.72	20.7	3.04	17.76
06.1.1	Science	----	------	1.47	30.0	0.44	1.91
06.1.1.a	Science Sensor package	----	------	1.47	30.0	0.44	1.91
06.1.1.a.a	Wind Sensor	1.0	0.15	0.15	30.0	0.05	0.20
06.1.1.a.b	Temperature Sensors	2.0	0.01	0.02	30.0	0.01	0.03
06.1.1.a.c	Pressure Sensors	1.0	0.1	0.05	30.0	0.02	0.07
06.1.1.a.d	Seismometer	1.0	1.1	1.10	30.0	0.33	1.43
06.1.1.a.e	Optical Monitor	1.0	0.1	0.10	30.0	0.03	0.13
06.1.1.a.f	Entry Accelerometer	1.0	0.0	0.04	30.0	0.01	0.05
06.1.1.a.g	Impact Accelerometer	1.0	0.0	0.01	30.0	0.00	0.01

TABLE 5.3.—MASER SCIENCE INSTRUMENTS

	Power (mW)	Data Rate (bps)	Duty Cycle (%)
Pressure/Temperature	1	20	100
Seismometer	50	20	100
Optical Monitor	20	1.6	8.33
Wind	250	24	8

5.2 Communications

Communication is effected with a UHF link using Mars orbiting S/C as store-and-forward relays. A transceiver-on-a-chip would transmit with 0.5 W RF output power through a short antenna with –3 dB of gain, achieving up to 8 kbps to a range of 1500 km.

5.2.1 Communications Requirements

Since there are no sophisticated tasks such as attitude determination, only data acquisition, compression and transmission, the control computer can be a simplistic single board computer based on a low power processor such as the ATmega168V or the TI Wolverine microcontroller 430FR59 series. The processor power consumption is only 0.33 to 0.45 mW, and the board power consumption including memory is 49 mW. Requirements are noted in Table 5.4.

5.2.2 Communications Assumptions and Constraints

The Communications System assumptions were as follows:

- Maximum data rate 8 kbps
- Maximum distance 1500 km
- Minimum elevation angle 10° of horizon
- Maximum acceleration 600g
- Lifetime 2 Mars yr

The Communications System constraints were as follows:

- 2.5 W DC power
- Minimize mass
- 64 cm^2 area
- Communicate to MRO at UHF

TABLE 5.4.—SCIENCE DATA AND TELECOM REQUIREMENTS

	Data (bps)	Compression/ selection	Duty cycle	Mbit/d	Mbit/d Martian d
Pressure/temperature	20	1	1	1.73	1.78
Seismometer	20	3	1	0.58	0.59
Optical	1.6	1	0.08333	0.01	0.01
Wind	24	3	0.08	0.06	0.06
			Total (Mbit/d)	2.38	2.44
Margin	0.5				
Passes (d)	2				
Pass duration (min)	5				
Data rate (kbps)	8				
Telecom capability (Mbit/d)	4.8				

5.2.3 Communications Design and MEL

The Communications MEL is shown in Table 5.5 and cross-section of the Communications System is shown in Figure 5.1.

- Use 400 MHz transceiver on a chip
- Use an electric dipole antenna that is foreshortened
- Transceiver on top of the S/C
- Single string design
- Antenna fixed - Gain –3 dB
- RF output power is 0.5 W

5.2.4 Communications Trades

No trades were performed.

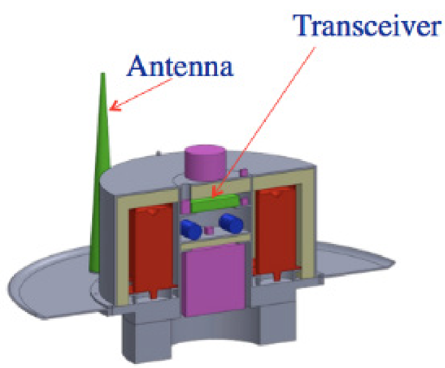


Figure 5.1.—Communications System.

TABLE 5.5.—COMMUNICATIONS CASE 1 MEL

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	--	------	50.10	19.6	9.82	59.92
06.1	Hard Lander	--	------	14.72	20.7	3.04	17.76
06.1.1	Science	--	------	1.47	30.0	0.44	1.91
06.1.4	Communications and Tracking	--	------	0.17	31.2	0.05	0.22
06.1.4.a	UHF System	--	------	0.17	31.2	0.05	0.22
06.1.4.a.a	Transceiver	1	0.05	0.05	30.0	0.02	0.07
06.1.4.a.b	Antenna	1	0.06	0.06	30.0	0.02	0.08
06.1.4.a.c	Harness	1	0.01	0.01	50.0	0.01	0.02
06.1.4.a.d	Box around the transceiver	1	0.05	0.05	30.0	0.01	0.06

5.2.5 Communications Analytical Methods

Mars Reconnaissance Orbiter (MRO) was used to determine what a reasonable uplink exposure there might be for the four lander's data return. Table 5.6 shows the visibility of MRO for each lander.

The daily data return requirement is supported by a single, relay orbiter, in this case, MRO was assessed.

The power system ultracapacitors was sized to have sufficient energy allow two consecutive uplinks, disregarding any recharge.

Figure 5.2 shows a graphic visualization of MRO with respect to the four landers. Figure 5.3 shows the communications system link budget.

Mars Hard lander circuit schematic is shown in Figure 5.4.

5.2.6 Communications Risk Inputs

The main communications risk is the failure of the electrical and mechanical at the end of the mission due to the 600 g during penetration. The risk is medium; however, the consequence is high because there is a loss-of-mission possibility. There is no mitigation of the risk as it has been decided the communication system is single string.

TABLE 5.6.—MRO VISIBILITY FOR 2021

Visibility in Minutes (min 10° elevation above horizon)					
Lander	Latitude	Min	Max	Mean	Min number passes/d > 5 min
1	75	0.9	10.5	8.1	9
2	65	0.7	10.4	7.8	4
3	60	0.4	10.3	7.8	3
4	65	1.2	10.4	7.9	4

Note: MRO does not "see" each lander on every pass. For all the passes for a given lander that MRO does "see" that lander the minimum time that the lander is visible is in the "Min" column.

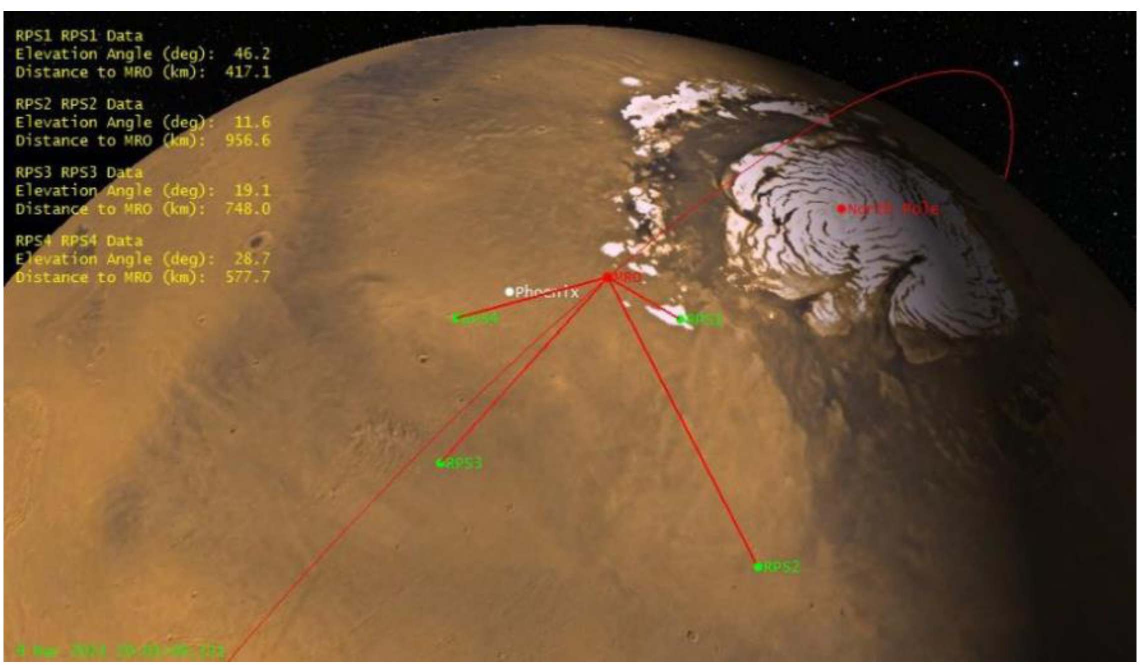


Figure 5.2.—The four landers of MRO.

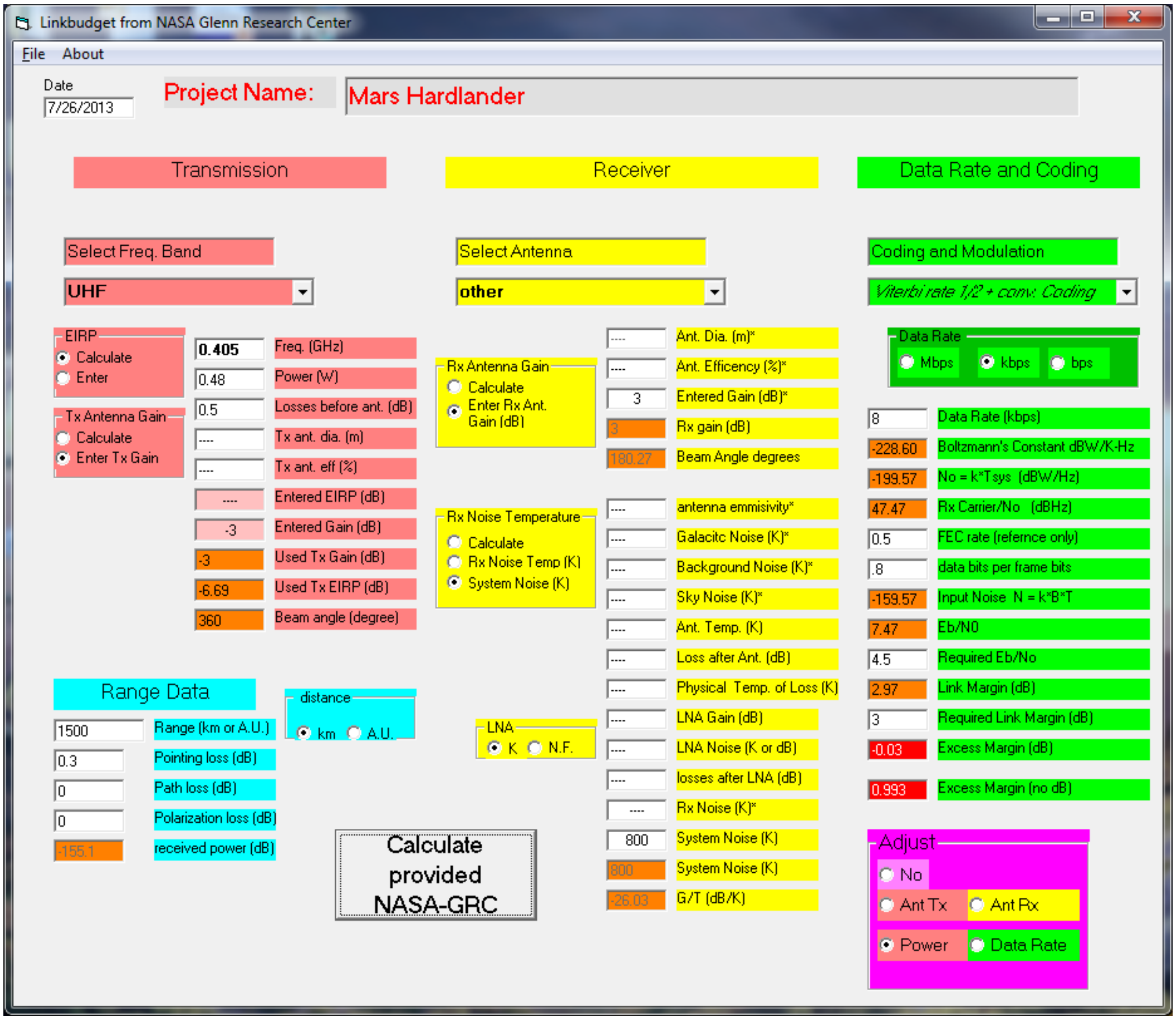


Figure 5.3.—Communications System 1/2 W RF, –3 dBi antenna.

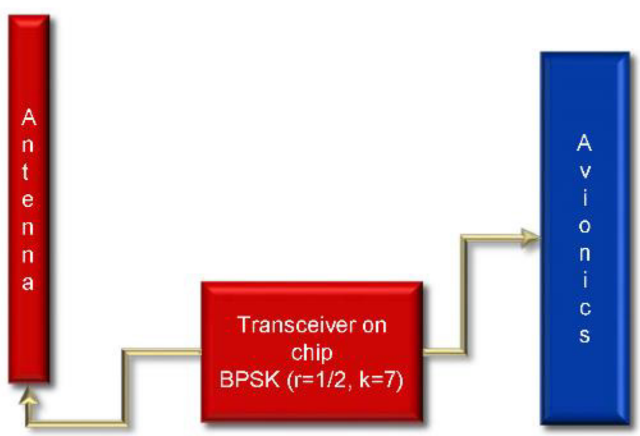


Figure 5.4.—Communications System circuit schematic.

5.2.7 Communications Recommendation

N/A

5.3 Command and Data Handling (C&DH)

The objective of this design is to provide a design and sizing of the C&DH system for the Mars hard lander weather and seismometer station. This will include identifying the components needed for the system for both the lander and aeroshell. Determining the component masses, volumes, and power consumption.

5.3.1 C&DH Requirements

The CD&H System provides computer control and data storage for the lander and computer control for aeroshell including pyro activation and system monitoring. The electronics operating temperature range shall remain between 233 to 323 K (–40 to 50 °C). The design requirements for the C&DH system are as follows:

- Avionics components and parts shall be Class S, per MIL–STD–883B, and be screened and commercially-screened.
- Particle radiation tolerance level of shall be 65 krad or higher.
- The technology level shall be based on what is likely available in 2015, since any advancements made before Phase A design are impossible to predict.
- Avionics shall be zero fault tolerant.
- Data storage unit shall be capable of handling at least 5 GB.

5.3.2 C&DH Assumptions

The following design assumptions are based on the mission requirements:

- Implemented with rad-tolerant microcontrollers, field-programmable gate arrays (FPGAs) when appropriate, and data storage using solid-state random-access memory (RAM) and flash memory.
- Avionics spare circuitry for fault tolerance are implemented as cold spares to minimize power consumption.
- Hardware design heritage are based on previous S/C and lessons learned.
- Data storage technology will continue to follow Moore's Law and, by mid-decade, will store dramatically larger volumes of data.
- By mid-decade, advances in semi-automatic code generation will help guarantee very capable, secure, and reliable operating system execution.
- Some breakthroughs in semi-autonomous, semi-intelligent software code design may occur during the life of the S/C design.

5.3.3 C&DH Design and MEL

The C&DH system consists of PowerPC-class processor boards configured to provide for single fault tolerance. Each processor board includes an FPGA-embedded PowerPC-class main processor capable of supporting C&DH functions, a 5-plus GB solid-state memory card, as well as communications and payload interface cards. The primary processor is capable of autonomous failover to a redundant cold spare unit if a fault is detected.

Flight computers will use a real-time operating system such as VxWorks or Green Hills Integrity. However, this estimate and implied development cost should be tempered with the understanding that recent developments in autocode technologies that generate known good instruction loads will become a design standard.

The following list is comprised of the main avionics components and their quantities, as input to the MEL shown in Table 5.7:

- Main computers (one main computer and one redundant cold spare)
- Data acquisition units contain redundant paths
- Solid-state memory
- Instrumentation (Lander)
    - Maximum of 48 sensors, mass of 6 oz. each, power requirement of 50 mW each
    - Sensor estimate based on a preliminary assumption of number of channels for input and output

To minimize power consumption the system was based on utilizing just the basic components needed to complete the mission. Ultra-low power consumption components were utilized where possible.

Lander: The CD&H components for the lander include the control computer, data storage memory, and wiring harness.

Aeroshell: The CD&H components for the aeroshell include the control computer, pyro activation card, sensor data collection card, and wiring harness.

TABLE 5.7.—C&DH CASE 1 MEL

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	--	-----	50.10	19.6	9.82	59.92
06.1	Hard Lander	--	-----	14.72	20.7	3.04	17.76
06.1.1	Science	--	-----	1.47	30.0	0.44	1.91
06.1.3	Command & Data Handling	--	-----	0.30	0.0	0.00	0.30
06.1.3.a	C&DH Hardware	--	-----	0.30	0.0	0.00	0.30
06.1.3.a.b	Command and Telemetry Computer	1	0.15	0.15	0.0	0.00	0.15
06.1.3.a.e	4 GB memory Module	1	0.05	0.05	0.0	0.00	0.05
06.1.3.a.f	Command and Control Harness (data)	1	0.10	0.10	0.0	0.00	0.10
06.2	Aeroshell/Heat Shield	--	-----	34.83	19.2	6.68	41.51
06.2.3	Command & Data Handling	--	-----	2.00	0.0	0.00	2.00
06.2.3.a	C&DH Hardware	--	-----	1.50	0.0	0.00	1.50
06.2.3.a.a	FPGA IP CPU rad hard LEON3 - Main	1	0.50	0.50	0.0	0.00	0.50
06.2.3.a.d	Command and Control Harness (data)	1	0.50	0.50	0.0	0.00	0.50
06.2.3.a.g	Pyro Card	1	0.50	0.50	0.0	0.00	0.50
06.2.3.b	Instrumentation & Wiring	--	-----	0.50	0.0	0.00	0.50
06.2.3.b.a	Sensor Card	1	0.50	0.50	0.0	0.00	0.50

5.3.3.1 Flight Computers and Software

The flight computers and software shall have the following capabilities:

- Load, initialization, executive functions, and utilities
- Flight computer redundancy management
- Data acquisition and control
- Command and telemetry processing
- Health monitoring and management
- Power management, control, and distribution
- Event sequence management
- Fault detection, diagnostics, and recovery

5.3.3.2 Data Acquisition System

The main purpose of the data acquisition system is collecting and distributing non-flight-critical sensor data from the instrumentation throughout the mission and storing it on mass memory via high-speed data buses.

5.3.4 C&DH Analytical Methods

As a matter of common practice, the design of a new S/C's C&DH system is often based on one that is proven effective on another S/C, and that requires minor or no modifications for the mission currently under development. The MASER C&DH system is based on previous S/C, such as Dawn, New Horizons, and Extrasolar Planet Observation (EPOXI).

5.3.5 C&DH Risk Inputs

The risk identified for C&DH is failure of the control computer, memory or data acquisition card.

***Risk Statement*:** The failure of one or more of the components from the C&DH system can occur due to shock damage during launch or landing, electrostatic discharge, low power quality during operation, exceeding the thermal limitations or excessive radiation during the mission.

***Context*:** The control computer and other electronics are utilized to operate the aeroshell after separation from the main S/C and the lander while on the surface of Mars. The system is responsible for all data collection and storage.

***Approach: Research / Accept / Watch / Mitigate*:** A loss of one or more of the C&DH system components will have a significant impact on the ability to complete the mission. To reduce the potential for a C&DH failure the components can be qualified prior to launch to insure they can withstand the space and launch environments. Redundant components or connections can be utilized to enable backup operation in the event of a failure. Also insure that the correct operating environment is maintained through the mission.

5.3.6 C&DH Recommendation

The following are the recommendations of the C&DH subsystem lead:

- The MASER Landers must have sufficient electromagnetic/radio frequency interference and particle shielding, due to its long-term space orbital.
- It must also be ground-bonded and surge-protected to resist on-pad lightning damage.

5.4 Guidance, Navigation and Control (GN&C)

5.4.1 GN&C Requirements

The GN&C subsystem was required to design an ED&L profile for the landers such that they would contact the surface with a relative velocity of no more than 20 m/s. Given the allowable displacement of the crush pad designed by the Structures subsystem, this velocity corresponded to a maximum allowable acceleration experienced on impact.

5.4.2 GN&C Assumptions

It was assumed that the drag coefficient of the main parachute would be similar to that of the parachute on Mars Pathfinder (Ref. 3), which was ~ 0.44. It was also assumed to be a Disk Gap Band (DGB) parachute with Viking heritage. Also, it was assumed that the parachute would be sized solely to achieve an effective drag. In other words, the detailed design of the parachute, including line length etc. would be left for future work.

5.4.3 GN&C Design and MEL

Table 5.8 shows a breakdown of the GN&C MEL for the MASER study. It consists solely of the parachute system that was sized to meet the requirement of landing with a ground relative velocity less than 20 m/s.

TABLE 5.8.—GN&C BASELINE MEL

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	----	-------	50.10	19.6	9.82	59.92
06.1	Hard Lander	----	-------	14.72	20.7	3.04	17.76
06.2	Aeroshell/Heat Shield	----	-------	34.83	19.2	6.68	41.51
06.2.2	Attitude Determination and Control	----	-------	11.78	30.0	3.53	15.32
06.2.2.a	Guidance, Navigation, & Control	----	-------	11.78	30.0	3.53	15.32
06.2.2.a.b	Parachute System	1.00	11.78	11.78	30.0	3.53	15.32
06.3	Carrier Interface	----	-------	0.55	18.0	0.10	0.65

The size and mass of the parachute system was estimated as shown here.

An estimate of the terminal velocity, V_T can be expressed as:

$$V_T{}^2 = \frac{\beta g}{\rho}$$

where β is the ballistic coefficient, g is the acceleration due to gravity and ρ is the atmospheric density. The ballistic coefficient, β, is calculated as follows:

$$\beta = 2M/SC_d$$

where M represents mass, C_d is the drag coefficient and S is the reference area. Plugging the expression for β in the above equation for the terminal velocity and then solving for the reference area yields:

$$S = \frac{2Mg}{V_{T^2} C_d \rho}$$

Assuming that the landed mass of the vehicle is 20 kg, the acceleration due to gravity at the surface of Mars is 3.71 m/s^2, a terminal velocity of 20 m/s, drag coefficient of 0.44 and a value of 1.56×10^{-2} kg/m^3 for the atmospheric density at the surface of Mars yields a required area of 52.81 m^2, or a diameter of 8.2 m.

With the required parachute diameter now known, the mass of the entire parachute system was scaled using data from the Huygens parachute system. The following is an estimate of the Huygens parachute system:

- Mass of Huygens parachute system = 12.1 kg
  - Mortar
  - 2.59 m diameter pilot chute
  - 8.31 m diameter main chute
  - 3.03 m diameter stabilizer chute

The mass of the MASER parachute system was estimated by simply scaling the mass of the Huygens parachute system by the diameter of the main parachute. This yielded an estimate of the MASER parachute system to be 11.8 kg.

- Mass of MASER parachute system = 11.8 kg
  - Mortar
  - 2.59 m diameter pilot chute
  - 8.2 m diameter main chute
    - Sized to ensure impact velocity is <= 20 m/s
    - Released upon impact with the ground

5.4.4 GN&C Trades

No trades were performed on the GN&C subsystem for this design study.

5.4.5 GN&C Risk Inputs

EDL Risk: The landing accuracy of the probes will be reduced by, among other things, the navigation accuracy of the carrier vehicle prior to probe deployment, uncertainty in the atmosphere model, aerodynamic properties of the heat shield and parachute, etc.

Risk Statement: While it was shown that a lander could be placed at each of the four landing sites with a few meters per second of ΔV applied 1 wk prior to atmosphere entry, the actual accuracy will be reduced by the navigational accuracy of the carrier vehicle prior to probe deployment, uncertainty in the atmosphere characteristics, aerodynamic properties of the heat shield and parachute, etc.

Mitigation Approach: Run Monte Carlo analysis dispersing the above-mentioned uncertainties to assess the actual landing accuracy that may be achieved with the probes

Likelihood:	5
Consequences	
Cost:	3
Schedule:	1
Performance:	4
Safety:	1

5.4.6 GN&C Recommendation

It is recommended to run Monte Carlo analysis, dispersing atmospheric and aerodynamic uncertainties to assess the actual landing accuracy that may be achieved by applying a ΔV 1 wk prior to atmosphere entry.

5.5 Electrical Power System

Radioisotope Power Systems (RPS) provides the S/C with electrical power when sunlight intensity or availability is low thereby making the use of solar arrays impractical. Traditional RPS systems have consisted of RTGs, which have provided many NASA lunar, Mars, and outer planet missions with highly reliable power. Current RPS systems use as their heat source building block the general-purpose heat source (GPHS) whose heat is then converted into electrical power. Each GPHS module provides about 250 W of thermal power when launched. In addition to the GPHS the 1 W RHU was developed to provide localized heating on S/C. Studies and engineering analysis has been performed using RHU's coupled with thermoelectrics to create very low power RTG's. One such design (HiZ RHU RPS) converted the 1 W of RHU power into 40 mW of DC power. This study considered an extremely low power derivative design (RHU-RTG) and integrates it onto the MASER S/C body.

5.5.1 Power and Energy Requirements

Table 5.9 shows the power and energy consumption for the MASER lander. There is a wide range of power requirements for the lander with power consumption dominated by the transmitter (Figure 5.5). However, when we look at energy requirements (over the 1 Earth day + 40 min or Martian day) we see a more balanced profile (Figure 5.6). Knowing the extremely low power output of the RHU RPS the strategy for the lander was to allow continuous low power measurements and housekeeping functions while storing energy for the high power loads which would be operated periodically. Specifically, the pressure, temperature sensors and seismometer would operate continuously while the wind sensor, optical monitor and communication system would operate periodically. This is necessary to provide context to the seismometer data with and without wind disturbance.

Table 5.10 shows the number of cycles per day for all the power loads.

TABLE 5.9.—POWER AND ENERGY CONSUMPTION

Load	Basic power (mW)	Power with margin (mW)	Duty cycle (%)	Total energy spent (mW-hr)
Continuous power for electronics	50	65	100	1560
Pressure/temperature	2	2.6	100	62.4
Seismometer	50	65	100	1560
Wind sensor	250	325	8	650
Optical monitor	20	26	8	52
Transmitter	2500	3250	1	1083.33333
Self-discharge of capacitor	15	15	100	360

TABLE 5.10.—NUMBER OF CYCLES PER DAY FOR ALL THE POWER LOADS

Component	Number of cycles per day
Continuous power for electronics	1
Pressure/temperature	1
Seismometer	1
Wind sensor	12
Optical monitor	3
Transmitter	2

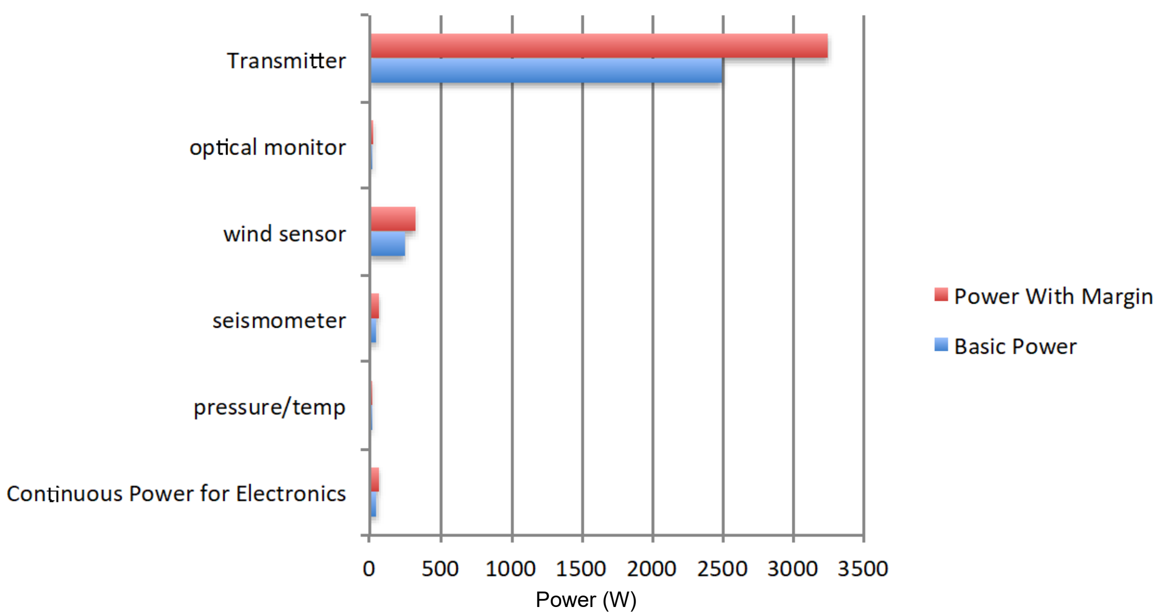


Figure 5.5.—Power requirements per science instrument over 1 day.

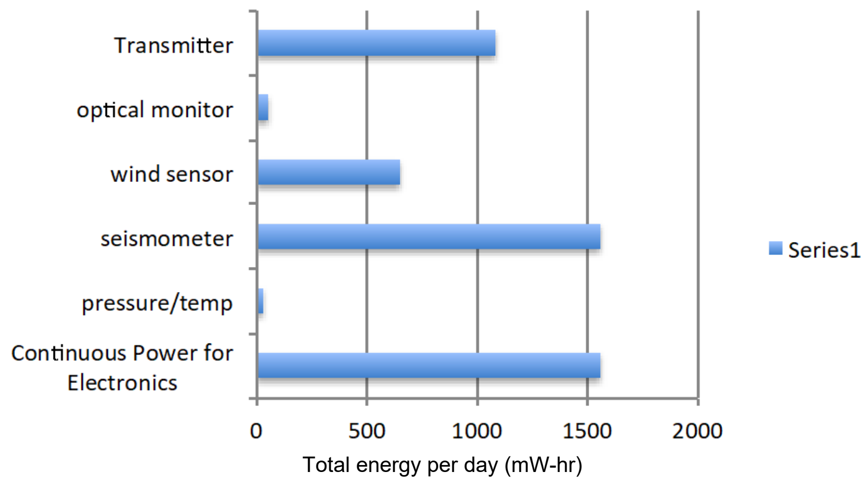


Figure 5.6.—Total energy requirement per science instrument over 1 Martian day.

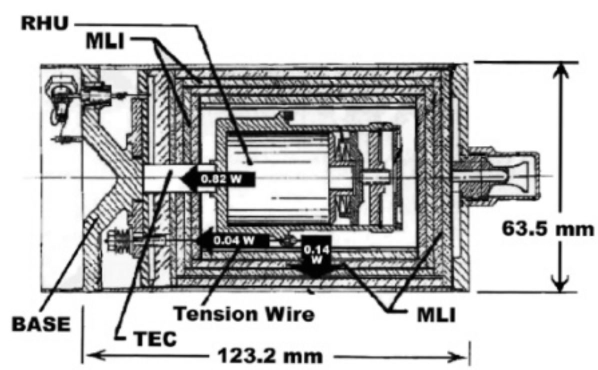


Figure 5.7.—Hi-Z 40 mWe RPS using RHU heat source and BiTe Thermoelectric Conversion.

5.5.2 Power Assumptions

The MASER system is based upon combining a RHU with thermoelectric energy conversion. The RHU is an encapsulated 1 W ^{238}Pu capsule which provides the heat to drive the thermoelectrics. It is flight qualified and has flown on many NASA missions. The baseline design which these power systems are built around are the HiZ RHU RPS which was a detailed design study and testing program performed in the 1990s (Figure 5.7). The HiZ design produces 40 mW of 5 V electrical power with a 250 °C hot side and 50 °C cold side junction temperatures. Each HiZ has a mass of 0.33 kg, is cylindrical in shape with a 73.5 mm diameter and 123.2 mm height. Because the HiZ was designed for a deep space environment an analysis was performed at JPL to estimate how the higher sink temperatures and low-pressure CO_2 environment would affect performance. JPL estimated that 40 mW would still be produced and regardless of the sink temperatures at the latitudes of interest and the HiZ would not change power levels over the day/night or yearly cycles. Heat rejection from the cold end of the thermoelectrics is to a baseplate. These baseplates are thermally integrated into the S/C design to ensure other electronic components are kept within their required temperatures. It is assumed that the HiZ RHU RPS degrade at the same rate as GPHS RTG which is 1.6 percent per year and provide 38 mW at the end of mission (5 yr).

5.5.3 Power Design and MEL

The power system design consists of 6 HiZ RHU RPS systems attached in a parallel string to the power system bus. Energy storage is provided by Maxwell Ultracapacitors. These were selected because of their wide operating temperature range and relatively lower energy storage requirements. For this S/C the BCAP0650 was selected with the maximum voltage of the capacitors of 2.7 V. Two of these were placed in series to match the 5 V bus and then a redundant string was added to increase reliability. Each BCAP0650 weighs 160 grams and has a 650 °F capacity. Temperature range is from –40 to 65 °C. This is far lower than typical Li ion batteries with a low temperature of around –10 °C. This is important because it provides greater flexibility to the thermal designer when using the relatively low amount of heat provided by the HiZ RHU RPS. One concern with these devices is that they cannot exceed the 65 °C due to the evaporation of the electrolyte. This prevented heat sterilization of the Ultracapacitors and required a separate vaporous H_2O_2 sterilization which is discussed in the planetary protection section. Maxwell Ultracapacitor. Figure 5.8 shows the power system architecture layout. Self-discharge from the Ultracapacitors proved to be an important load to track. The worst case leakage current for each of the caps used is 1.5 mA. Each Capacitor is running at 2.5 V (stable since we have a great deal of excess capacity). Total leakage power is 15 mW (4 X 3.75 mW for each). Total energy lost by each at worst case consumption over the day is 90 or 360 mW-hr for all the capacitors. Of all the energy consumed it comes in fifth most important.

Figure 5.9 and Figure 5.10 show both energy stored and bus voltage for a 2 day cycle at EOM. Bus voltage varies about 0.2 V while the energy storage system uses about 0.6 W-hr of energy during each day/night cycle. The sawtooth lines show the periodic high energy drains from both the wind measurements, optical sensors. The large drop each 12 hr shows the transmitter operating.

All the components of the power subsystem and their masses are shown in Table 5.11.

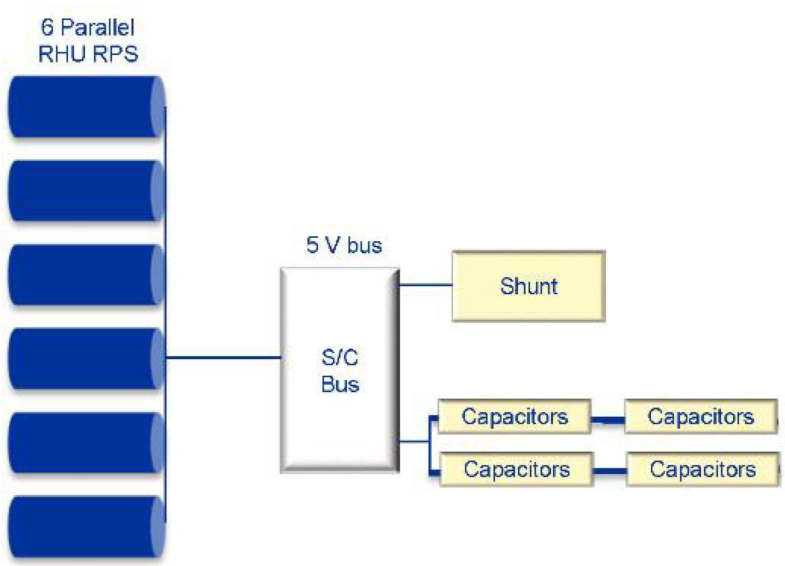


Figure 5.8.—Power System Architecture.

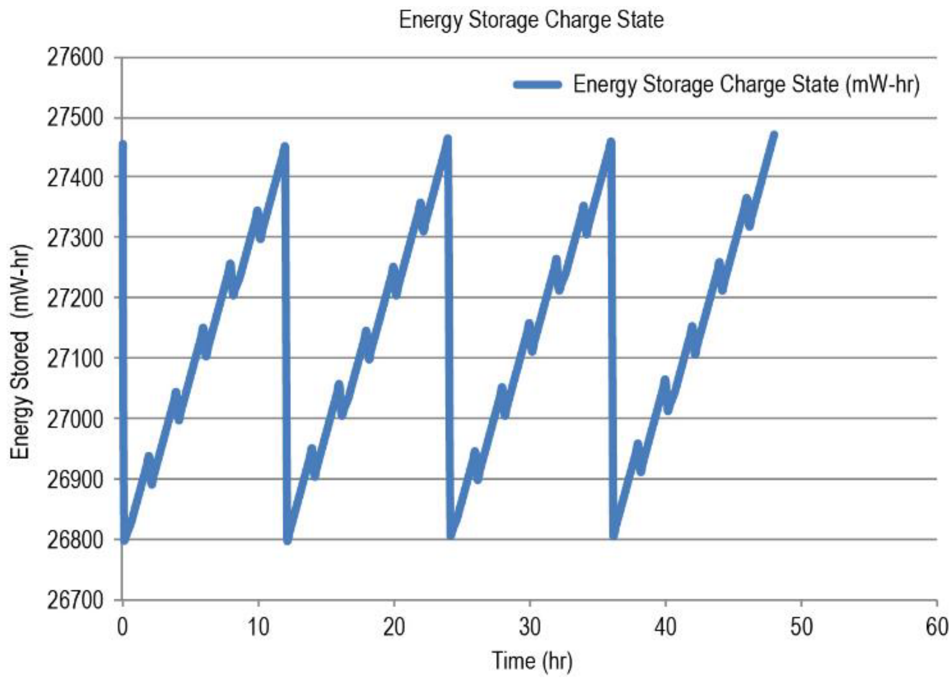


Figure 5.9.—Energy storage versus time.

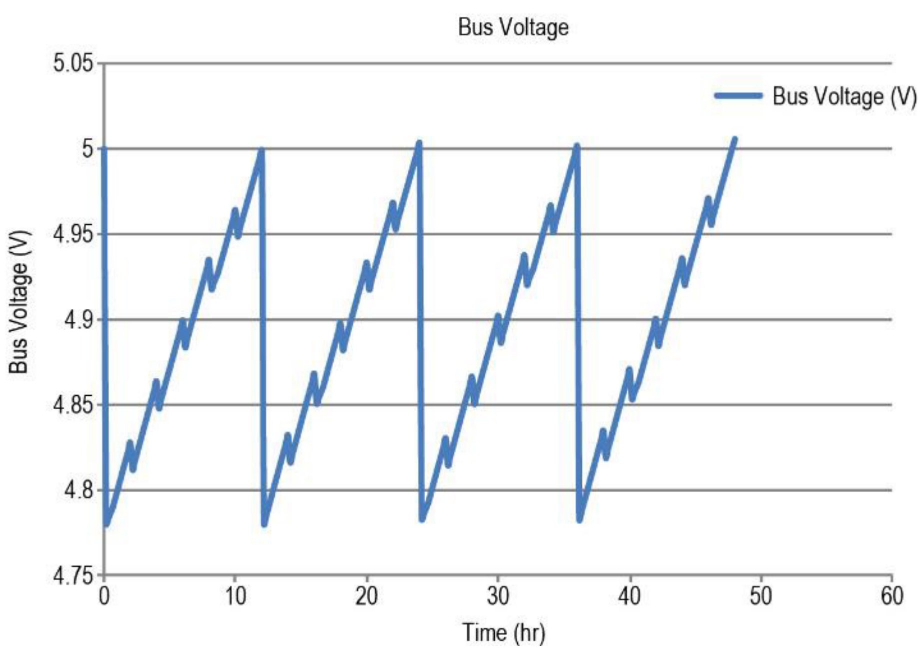


Figure 5.10.—Bus voltage versus time.

TABLE 5.11.—ELECTRICAL POWER SYSTEM CASE 1 MEL

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	--	-----	50.10	19.6	9.82	59.92
06.1	Hard Lander	--	-----	14.72	20.7	3.04	17.76
06.1.5	Electrical Power Subsystem	--	-----	2.65	27.6	0.73	3.38
06.1.5.a	Power Conversion	--	-----	2.62	27.3	0.72	3.34
06.1.5.a.a	HiZ RHU RPS	6	0.33	1.98	20.0	0.40	2.38
06.1.5.a.b	Ultracapacitor	4	0.16	0.64	50.0	0.32	0.96
06.1.5.b	PMAD	--	-----	0.03	50.0	0.02	0.05
06.1.5.b.e	Power and Data Wiring	1	0.03	0.03	50.0	0.02	0.05
06.2	Aeroshell/Heat Shield	--	-----	34.83	19.2	6.68	41.51
06.2.5	Electrical Power Subsystem	--	-----	0.30	0.0	0.00	0.30
06.2.5.a	Power Conversion	--	-----	0.30	0.0	0.00	0.30
06.2.5.a.a	Secondary Descent Battery	1	0.30	0.30	0.0	0.00	0.30

5.5.3.1 Technology Maturity

Solar arrays are at TRL-6.

5.5.4 Power Trades

After completing the baseline power system, several trades were run to understand how variations in available power would impact the science (Table 5.12). As was noted earlier the pressure and temperature measurements as well as the optical monitor required a very small fraction of the available power. If we consider the use of four RHU-RPS (Table 5.13) rather than the baseline six (Table 5.14) even with removing of the wind sensor to remove noise the system does not provide enough energy to transmit the data back. Using eight (Table 5.15) (rather than six) allows us to increase the wind sensor operational time up to 30 percent from the nominal 8 percent. This should provide some increase in seismometry data fidelity. Increasing the number of RHU-RPS to 12 (Table 5.16) increases the wind monitor up time to 70 percent.

Observations

- Pressure/temperature very little power
- Transmitter effect of lower data rate unclear (time to acquire/transmit)
- Optical monitor small fraction of energy
- Pressure/temperature with optical only should dramatically reduce power
- Four RHU-RPS rather than six
    - Reduces Science gathered.
    - Removal of Seismometer closes case
    - Remaining: Weather Station: Pressure, temp, wind, optical
    - Won’t close with seismometer only
- 2 X 4 RHU RTG’s producing 8 by 40 mW BOM
    - Allows wind sensor to increase monitoring to 30 percent (up from 8 percent) of the time
- 3 X 4 RHU RTG’s producing 12 by 40 mW BOM
    - Increases Wind sensor monitoring up to 70 percent

TABLE 5.12.—POWER SYSTEM TRADES

Power System Trades	Details
Baseline	
Option 1	Four RHU RTG
Option 2	Eight RHU RTG
Option 3	12 RHU RTG

TABLE 5.13.—FOUR RHU RTG

Load	Basic power (mW)	Power with margin (mW)	Duty cycle (%)	Total energy spent (mW-hr)
Continuous power for electronics	50	65	100	1560
Pressure/temperature	2	2.6	100	62.4
Seismometer	0	0	100	0
Wind sensor	250	325	8.3	650
Optical monitor	20	26	8.3	52
Transmitter	2500	3250	1.4	1083
			Energy used	3757.0
			Energy produced	3648.0

TABLE 5.14.—POWER TRADES BASELINE

Load	Basic power (mW)	Power with margin (mW)	Duty cycle (%)	Total energy spent (mW-hr)
Continuous power for electronics	50	65	100	1560
Pressure/temperature	2	2.6	100	62.4
Seismometer	50	65	100	1560
Wind sensor	250	325	8.3	650
Optical monitor	20	26	8.3	52
Transmitter	2500	3250	1.4	1083.33333
			Energy used	5476.9
			Energy produced	5472.0

TABLE 5.15.—EIGHT RHU RTG

Load	Basic power (mW)	Power with margin (mW)	Duty cycle (%)	Total energy spent (mW-hr)
Continuous power for electronics	50	65	100	1560
Pressure/temperature	2	2.6	100	62.4
Seismometer	50	65	100	1560
Wind sensor	250	325	30	2340
Optical monitor	20	26	8.3	52
Transmitter	2500	3250	1.4	1083
			Energy used	7340.2
			Energy produced	7296.0

TABLE 5.16.—TWELVE RHU RTG

Load	Basic power (mW)	Power with margin (mW)	Duty cycle (%)	Total energy spent (mW-hr)
Continuous power for electronics	50	65	100	1560
Pressure/temperature	2	2.6	100	62.4
Seismometer	50	65	100	1560
Wind sensor	250	325	70	5460
Optical monitor	20	26	8.3	52
Transmitter	2500	3250	1.4	1083
			Energy used	10,780.0
			Energy produced	10,944.0

5.5.5 Power Risk Inputs

The following are the power risks:

- RHU RTG
- RHU RTG Development
- Environmental Impact on Power Generation
- High G survival

5.6 Propulsion System

There was no Propulsion system designed in this study.

5.7 Structures and Mechanisms

5.7.1 Structures and Mechanisms Requirements

The Mars Hard Lander structures must contain the necessary hardware for research instrumentation, communications, and power while fitting within the confines of the aeroshell and backshell assembly. The structural components must be able to withstand applied loads from the launch vehicle, operational maneuvers, and landing. In addition, the structures must provide minimum deflections, sufficient stiffness, and vibration damping. The maximum axial load of 600 g is anticipated upon landing on the planet surface. Other parts of the flight may impose a 25 g axial load during atmospheric entry and a potential 5 g lateral load from the launch vehicle. The goal of the design is to minimize mass of the components that comprise the structure of the S/C bus and must also fit within the physical confines of the launch vehicle.

The mechanisms are required to function for single events or continuously throughout the mission, depending on the types of mechanisms. Separation mechanisms must release the drogue parachute, the aeroshell, the main parachute, the backshell, the main parachute tethers, and finally, the science seismometer.

5.7.2 Structures and Mechanisms Assumptions

The main bus consists of a thrust tube and a disk platform which is assumed to provide the optimum architecture for housing the necessary operational hardware. The bus components are made of Al, 2090-T3, from the Metallic Materials Properties Development and Standardization (MMPDS) (Federal Aviation Administration, 2012) (Ref. 4) and G-10 glass fiber reinforced epoxy as noted in the Materials Engineering Materials Selector 1988 (Ref. 5). The crush pad is of Hexcel HexWeb Nonmetallic

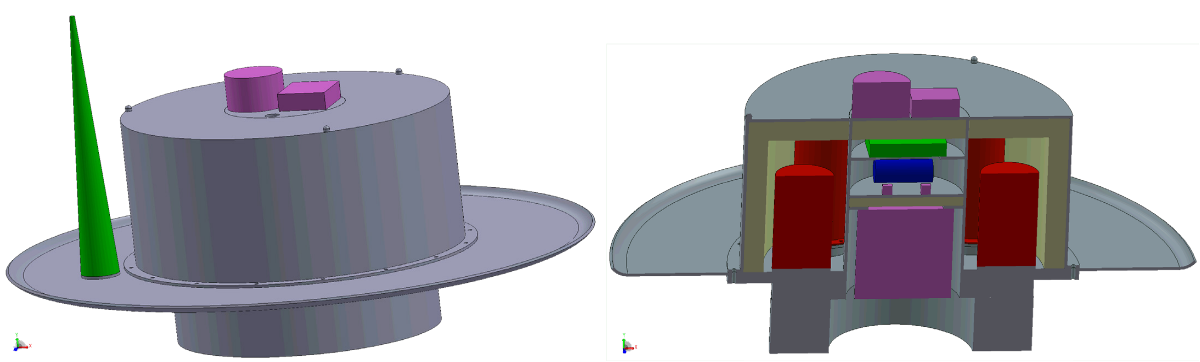
Flex-Core honeycomb. Figure 5.11 illustrates views of the main bus. Joining of components is by welding, bonding, and threaded fasteners. The analysis performed in this study assumed a maximum axial load from launch of 600 g.

The lander requires the use of a few mechanisms as it goes through its trajectory. There is an actuated release for a drogue parachute. A pyrotechnic release mechanism is used to separate S/C from the aeroshell. That is followed by an actuated release of the main parachute. During the parachute decent a pyrotechnic release mechanism is used to separate the backshell. Pyrotechnics are used to sever the tethers of the main parachute. Lastly, actuated release mechanisms are used to drop the science seismometer to the planet surface.

5.7.3 Structures and Mechanisms Design and MEL

The lander bus consists of a 2090-T3 Al thrust tube on top of a low thermal conductivity G-10 glass fiber reinforced polymer composite disk. A low thermal conductivity Hexcel HexWeb Nonmetallic Flex-Core honeycomb crushable material is mounted to the bottom of the lander to disperse energy upon landing on the Martian surface. The various components are mounted to Al decks within the thrust tube. RHUs are mounted to the disk outside of the thrust tube. The mounted hardware includes components for communications and tracking; C&DH; GN&C; electrical power; thermal management, and science. Figure 5.12 illustrates the lander mounted within the aeroshell/backshell assembly. The aeroshell and backshell are transparent in the figure for illustration purposes. Figure 5.13 shows a cut section of the Lander with equipment boxes being visible.

The 2090-T3 Al has a yield strength of 434 MPa and an ultimate strength of 517 MPa per the MMPDS(Ref. 4). Safety factors are 1.25 on the yield strength and 1.4 on the ultimate strength per the NASA standard, NASA-STD-5001 (1996) (Ref. 6), for a protoflight design. The resulting allowable stress is 347 MPa limited by the yield stress. The glass/epoxy composite G-10 has a tensile strength of 241 MPa per the Materials Engineering Materials Selector 1988. NASA-STD-5001 (Ref. 6)provides a safety factor of 2.0 for a protoflight composite with discontinuities. The resulting allowable stress for the G-10 composite is 121 MPa. The material and bus architecture provides a technology readiness level (TRL) of six as per Mankins (1995) (Ref. 7).



(a) (b)

Figure 5.11.—(a) A view of the Mars Hard Lander. (b) A sectional view of the bus and mounted hardware

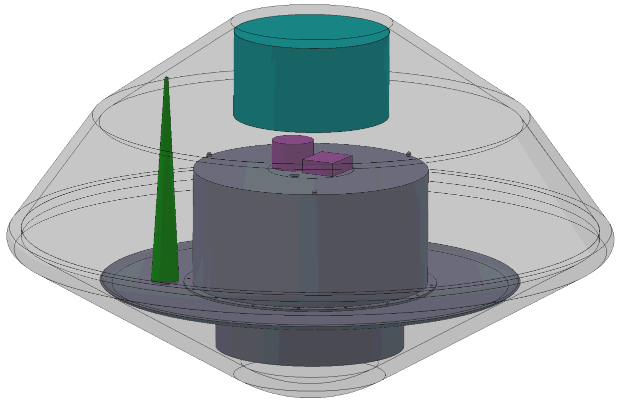


Figure 5.12.—The Lander mounted in the Aeroshell/Backshell assembly. The aeroshell and backshell are transparent for illustration purposes. The top cylinder is the parachute container.

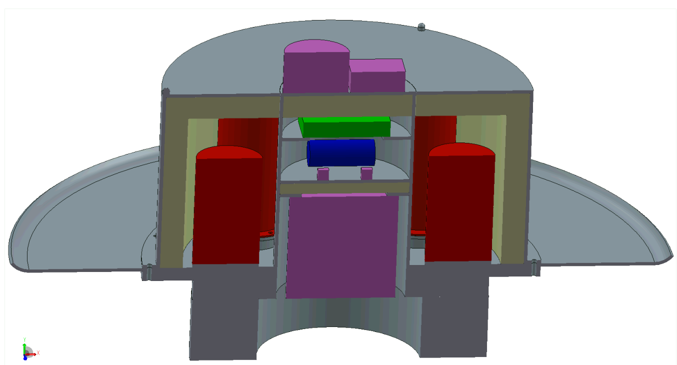


Figure 5.13.—A cross sectional view of the Lander and its mounted hardware.

TiNi Aerospace E500 Ejector Release Mechanisms are specified for holding and releasing the aero shell and backshell. Three E500 mechanisms are specified per component. A TiNi Aerospace Frangibolt FC2 is specified for the main parachute release. Lastly, two TiNi Aerospace P5-403 pin pullers are specified for retaining and releasing the science seismometer.

Table 5.17 shows the expanded MEL for the structures subsystem on the Hard Lander. Table 5.18 shows the expanded MEL for the structures subsystem on the Aeroshell/Heat Shield. Table 5.19 shows the expanded MEL for the structures subsystem on the Carrier Interface. This MEL breaks down the structures line elements to the lowest WBS.

TABLE 5.17.—MASER STRUCTURES MEL—HARD LANDER

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	---	------	50.10	19.6	9.82	59.92
06.1	Hard Lander	---	------	14.72	20.7	3.04	17.76
06.1.11	Structures and Mechanisms	---	------	10.03	18.0	1.80	11.83
06.1.11.a	Structures	---	------	9.73	18.0	1.75	11.48
06.1.11.a.a	*Primary Structures*	---	------	9.13	18.0	1.64	10.77
06.1.11.a.a.a	Main Body	1	6.18	6.18	18.0	1.11	7.29
06.1.11.a.a.d	Outer Cover	1	2.95	2.95	18.0	0.53	3.48
06.1.11.a.b	*Secondary Structures*	---		0.60	18.0	0.11	0.71
06.1.11.a.b.d	crush pad	1	0.10	0.10	18.0	0.02	0.11
06.1.11.a.b.e	8-32x1/2 bolt assembly RHU	48	0.00	0.16	18.0	0.03	0.19
06.1.11.a.b.f	8-32x1 bolt assembly Top Hat	16	0.00	0.07	18.0	0.01	0.08
06.1.11.a.b.g	Truss, S/C to aeroshell	3	0.09	0.27	18.0	0.05	0.32
06.1.11.b	Mechanisms	---	------	0.30	16.9	0.05	0.35
06.1.11.b.b	*Science Payload*	---	------	0.02	2.0	0.00	0.02
06.1.11.b.b.a	TiNi Aerospace P5-403 pin puller	2	0.01	0.02	2.0	0.00	0.02
06.1.11.b.f	*Installations*	---	------	0.28	18.0	0.05	0.33
06.1.11.b.f.d	C&DH Installation	1	------	0.16	18.0	0.03	0.19
06.1.11.b.f.e	C&T Installation	1	------	0.01	18.0	0.00	0.01
06.1.11.b.f.f	Electrical Power Installation	1	------	0.11	18.0	0.02	0.13
06.1.11.b.f.g	Thermal Control Installation	1	0.00	0.00	18.0	0.00	0.01

TABLE 5.18.—MASER STRUCTURES MEL—AEROSHELL/HEAT SHIELD

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	--	------	50.10	19.6	9.82	59.92
06.1	Hard Lander	--	------	14.72	20.7	3.04	17.76
06.2.11	Structures and Mechanisms	--	------	6.02	15.6	0.94	6.96
06.2.11.a	Structures	--	------	5.00	18.0	0.90	5.90
06.2.11.a.a	*Primary Structures*	--	------	0.00	0	0.00	0.00
06.2.11.a.b	*Secondary Structures*	--	------	5.00	18.0	0.90	5.90
06.2.11.a.b.f	Bio Shield	1	5.00	5.00	18.0	0.90	5.90
06.2.11.b	Mechanisms	--	------	1.02	3.6	0.04	1.06
06.2.11.b.e	*Adaptors and Separation*	--	------	1.02	3.6	0.04	1.06
06.2.11.b.e.a	Backshell TiNi Aerospace Ejector Release Mech E500	3	0.10	0.30	2.0	0.01	0.31
06.2.11.b.e.b	Aeroshell TiNi Aerospace Ejector Release Mech E500	3	0.10	0.30	2.0	0.01	0.31
06.2.11.b.e.c	Pyrotechnic fasteners & springs, back shell	3	0.10	0.30	2.0	0.01	0.31
06.2.11.b.e.d	Drogue chute release	1	0.10	0.10	18.0	0.02	0.12
06.2.11.b.e.e	Main chute release TiNi Aerospace Frangibolt FC2	1	0.02	0.02	2.0	0.00	0.02
06.3	Carrier Interface	--	------	0.55	18.0	0.10	0.65
06.3.11	Structures and Mechanisms	--	------	0.55	18.0	0.10	0.65

TABLE 5.19.—MASER STRUCTURES MEL—CARRIER INTERFACE

WBS Number	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	--	------	50.10	19.6	9.82	59.92
06.1	Hard Lander	--	------	14.72	20.7	3.04	17.76
06.1.11	Structures and Mechanisms	--	------	10.03	18.0	1.80	11.83
06.2	Aeroshell/Heat Shield	--	------	34.83	19.2	6.68	41.51
06.2.11	Structures and Mechanisms	--	------	6.02	15.6	0.94	6.96
06.3	Carrier Interface	--	------	0.55	18.0	0.10	0.65
06.3.11	Structures and Mechanisms	--	------	0.55	18.0	0.10	0.65
06.3.11.a	Structures	--	------	0.30	18.0	0.05	0.35
06.3.11.a.a	*Primary Structures*	--	------	0.30	18.0	0.05	0.35
06.3.11.a.a.a	Mount hardware	1	0.30	0.30	18.0	0.05	0.35
06.3.11.b	Mechanisms	--	------	0.25	18.0	0.05	0.30
06.3.11.b.e	*Adaptors and Separation*	--	------	0.25	18.0	0.05	0.30
06.3.11.b.e.e	Harness	1	0.25	0.25	18.0	0.05	0.30

5.7.4 Structures and Mechanisms Trades

The initial assumption was to use Al throughout the structure. Thermal conductivity was found to be excessive creating a need for a lower conductivity structural material in the area near the planet surface. The glass fiber reinforced polymer matrix composite, G-10, was chosen as the material for the disk base of the Lander structure.

Passive and active release mechanisms for the science seismometer were evaluated. Initially, a passive system which released the seismometer upon landing was suggested. Upon evaluation it was determined that the probability of a lateral velocity and potential damage to the deployed seismometer was excessive. Another option was a hot wire support with an active system. It was determined that the high acceleration upon landing was excessive for a practical wire support/release system. An active pin release system was found to provide the greatest integrity for the harsh landing and the greatest potential reliability.

5.7.5 Structures and Mechanisms Analytical Methods

Preliminary structural analysis and modeling was performed using given launch and landing loads and the dimensions of the proposed S/C bus structure. Analytical methods utilizing a spreadsheet tool were employed to analyze the bus. A maximum axial load of approximately 600 g is anticipated on the S/C upon landing on the planet surface.

A simple analysis was performed on the thrust tube structure of the Lander bus. It was assumed that 15 kg is supported by the thrust tube and a 600 g acceleration is applied due to the landing. The resulting axial stress is 72 MPa. With an allowable stress of 347 MPa the resulting margin is 3.8.

The composite disk was analyzed as an annular disk using equations for stress from Young and Budynas (2002) (Ref. 8). Using the allowable stress of 121 MPa a minimum thickness was determined. The disk is assumed to support 15 kg with a 600 g acceleration. The resulting minimum thickness is 3.18 cm which has the disk loaded at the allowable stress.

The honeycomb crush pad was sized to decelerate the lander from the approach velocity of 22 m/s to zero upon landing, while limiting the maximum acceleration to 600 g. Assuming a constant deceleration, the necessary crush height for the honeycomb is 4.1 cm. Assuming that 70 percent of the honeycomb crushes, leaving the rest of the material packed solidly results in a minimum needed overall height of

6.0 cm. The cross sectional area was calculated to hold the force of the supported mass of 15 kg at the maximum acceleration of 600 g such that the stress in the crush pad is held at the Hexcel HexWeb Nonmetallic Flex Core crush strength for HexWeb HRP/F35-3.5. The crush strength was reported as 2.2 MPa. The honeycomb shape is a hollow cylinder which leaves a passage for the seismometer. A 20 cm inside diameter was assumed to provide the necessary clearance for the seismometer. The resulting necessary outer diameter is 30.1 cm to provide a cross sectional area of 0.040 m^2 and a mass of 0.10 kg.

An initial assumption was to have a wire supporting the seismometer such that the load during landing breaks the wire for a passive deployment. A wire diameter range of 1.3 to 1.9 mm of AISI 316 stainless steel would provide a break load below the maximum load due to the 600g acceleration but sufficiently high to prevent fracture at lower loads encountered during launch and the rest of the trajectory. Due to concerns of a lateral velocity which would drag the sensor on the Martian surface it was decided to have an active release system.

An additional installation mass was added for each subsystem in the mechanisms section of the structures subsystem. These installations were modeled using 4 percent of the CBE dry mass of each of the subsystems. The 4 percent magnitude for an initial estimate compares well with values reported by Heineman (1994) (Ref. 9) for various manned systems. This is to account for attachments, bolts, screws, and other mechanisms necessary to attach the subsystem elements to the bus structure, and not book kept in the individual subsystems. An 18 percent growth margin was applied to the resulting installation mass. These margins are placed onto the subsystem elements prior to the additional margin that was added to reach the 30 percent MGA required on the dry mass elements.

5.7.6 Structures and Mechanisms Risk Inputs

Risks for the structures includes excessive g loads, a potential impact with a foreign object during flight, a harsh landing, insufficient stiffness in the bus may cause too much deformation, vibrations, or fracture of sections of the support structure which may affect the performance of the S/C and its instrumentation. Insufficient damping in the structure may cause issues with long-term fatigue. Consequences include lower performance from mounted hardware to loss of mission. The likelihood is 3 with consequences as follows

Cost: 3
Schedule:.................................4
Performance:............................4
Safety: 1

As a mitigation step the structure is to be designed to NASA standards to withstand expected g loads, a given impact, and to have sufficient stiffness and damping to minimize issues with vibrations. Ground transport and mission trajectories are to be planned to minimize the probability of excessive loads and impact with foreign objects or too much approach velocity for landing.

Mechanism risks may be encountered with poor installation, excessive g loads or impact from a foreign object which may cause too much deformation, vibrations, binding, or fracture of components. The likelihood ranking is 3. Consequences include an inability to separate from components resulting in lower performance to loss of mission or inability to deploy necessary hardware also resulting in lower performance or failed mission. The likelihood is three where rankings for consequences with mechanisms are

Cost: 4
Schedule:.................................4
Performance:............................4
Safety: 1

To mitigate risks with mechanisms the devices are to be designed to NASA standards to withstand expected g loads, a given impact, and to have sufficient stiffness and damping to minimize issues with vibrations. Installation instructions per manufacturer specifications and/or NASA standards are to be followed. Ground assembly and transport and mission trajectories are to be planned to minimize the probability of excessive g loads and impact with foreign objects.

5.7.7 Structures and Mechanisms Recommendation

For a complete design, a finite element analysis (FEA) should be conducted to provide a high fidelity model of the structure. The FEA results would determine stresses, displacements, modal frequencies for vibrations, and the structural response due to static forces and forced vibrations from various sources. The FEA results would aid in keeping natural frequencies away from the operating frequencies of the mounted hardware and help determine the damping requirements.

- More advanced material systems and architectures may be applied for greater mass reduction. Greater use of fiber reinforced composite, orthogrid, and/or isogrid panels may be utilized.
- Greater use of fiber reinforced polymer matrix composites

5.8 Thermal and Environmental Control

The thermal system for the long duration Mars hard lander weather and seismometer station (MASER) was modeled and designed to meet the desired operating temperature on the surface of Mars for the duration of the mission. The thermal model provided an estimate of the heat loss to the environment and the corresponding operating temperature of the lander electronics and RHU power modules. The goal of the design was to provide a passive thermal system that would remain within the operating temperature requirements throughout the mission. The overall thermal system for the MASER vehicle consists of the following elements, some of which are illustrated in Figure 5.14.

- Thermal paint
- Aerogel Insulation
- Thermal monitoring system (thermocouples, data acquisition)
- Aeroshell and heat shield

5.8.1 Thermal Requirements and Operating Environment

The thermal requirements for the mission were to provide a means of passively maintaining the S/C interior component temperature for the S/C electronics and RHU modules both during transit and while on the surface of Mars. The worst-case hot and cold Mars temperatures were used to determine the internal components operating temperatures while on the surface. The 1.5 AU space environment was used to determine the operating temperature of the components while approaching Mars prior to entry. Once released from the carrier S/C the aeroshell will need to operate for up to 14 day without any active thermal control prior to entry into the Mars atmosphere. The main inputs to the thermal system sizing are listed in Table 5.20.

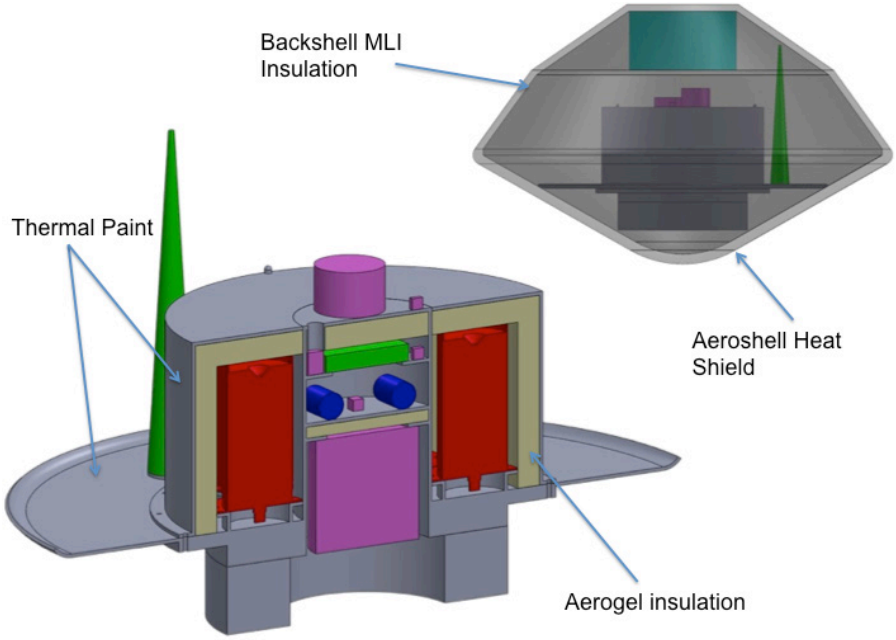


Figure 5.14.—Thermal System Components.

TABLE 5.20.—INPUTS TO THE THERMAL SYSTEM DESIGN

Requirement	Value
Lander enclosed components dimensions	Length 0.15 m diameter 0.15 m (cylindrical in shape)
Number of RHUs on the lander	Six at 1 W thermal power each
Aeroshell Dimensions	Diameter 0.88 m (based on the Stardust Aeroshell Geometry)
Operating temperature range	233 to 323 K (–40 to 50 °C)
Aeroshell Insulation	The interior of the backshell was wrapped with 10 layers of MLI
Environment	Surface operation at the Mars Phoenix lander location (65° N latitude)
Lander Insulation	Aerogel insulation wrapped around the outer surface of the RHU and electronics package and in between each of the individual RHU modules

To determine the operating conditions on the Mars surface data from the two northern most landers (Viking 2 and Phoenix) were used. Viking 2 operated for 1281 Martian days at a latitude of 48° N and the Phoenix lander operated for 157 Martian days, from later spring to late summer, at 68° N latitude, as shown in Figure 5.15. The objective was for the lander to operate near the pole throughout the Martian year. However, since the Phoenix lander only operated during the summer its operating temperature was utilized as the worst-case warm condition and the Viking 2 lander winter operating temperature was utilized as the worst-case cold condition.

The Phoenix lander temperature data is shown in Figure 5.16. The lander begins to operate just before the beginning of the summer season in the Northern hemisphere of Mars. From the figure the temperature gradually increases and reaches a peak of 195 K (–78 °C) after approximately 35 days of operation. From this time on to the end of the mission temperatures gradually decrease. The total mission length of 157 day does not cover the winter season.

The Viking 2 lander operated for a much longer period that extended over several Martian years. Therefore, there is winter seasonal data available from the Viking 2 lander as shown in Figure 5.17. This figure shows a minimum operating temperature of 153 K (–120 °C).

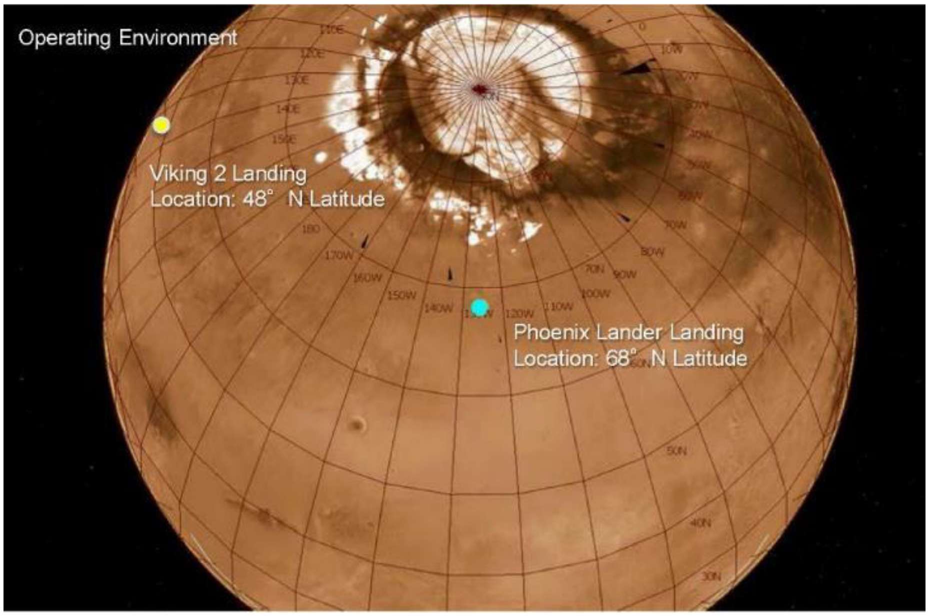


Figure 5.15.—Previous Northern Most Mars Lander Locations.

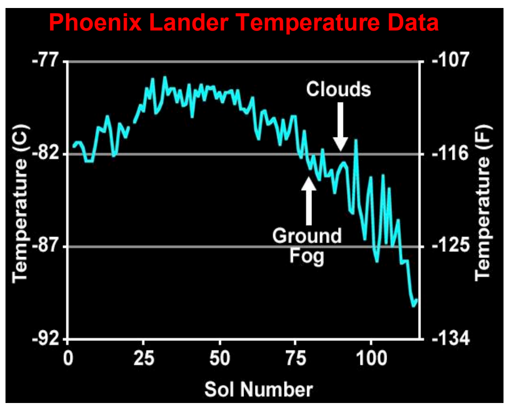


Figure 5.16.—Phoenix Lander atmospheric temperature data.

5.8.2 Aeroshell Thermal Control

The aeroshell consists of a heat shield and back shell. The heat shield needs to be able to withstand the aerodynamic heating that will be encountered during entry into the Mars atmosphere. The heat load will depend on the entry angle and speed.

The heat shield for Mars entry was scaled from the Stardust and Genesis Earth entry vehicles. All Mars entry vehicles had lower entry velocities then that of Stardust (~ 11 km/s) as shown in Figure 5.18. Therefore, the stardust or Genesis heat shield should be more than sufficient for Mars entry. In these heat shield designs PICA is used as the ablative material.

The heat shield and backshell geometry were scaled from the Stardust aeroshell design, shown in Figure 5.18. Based on these previous missions, the PICA thickness utilized was 5.82 cm.

To maintain the MASER lander within its operating temperature limits during transit 10 layers of MLI were used to insulate the backshell and inner surface of the aeroshell. With this insulation in place the worst case hot internal temperature will occur at 1 AU at an operating temperature of 290 K and the worst case cold internal temperature will occur at 1.5 AU at an operating temperature of 258 K. The Thermal environmental are illustrated in Figure 5.19.

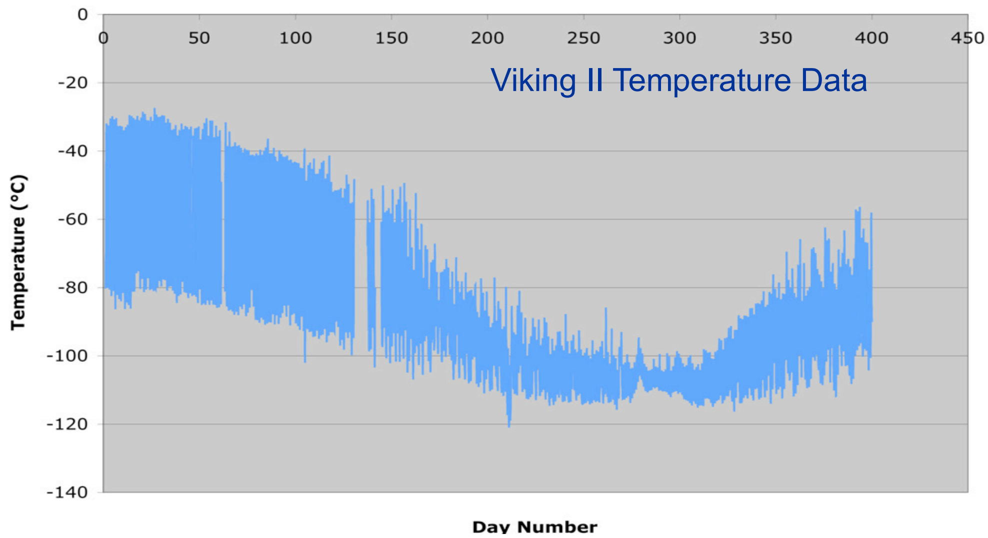
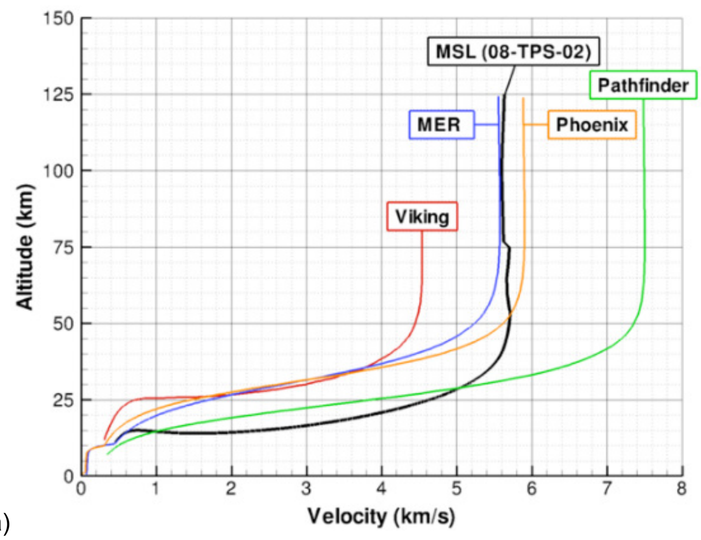
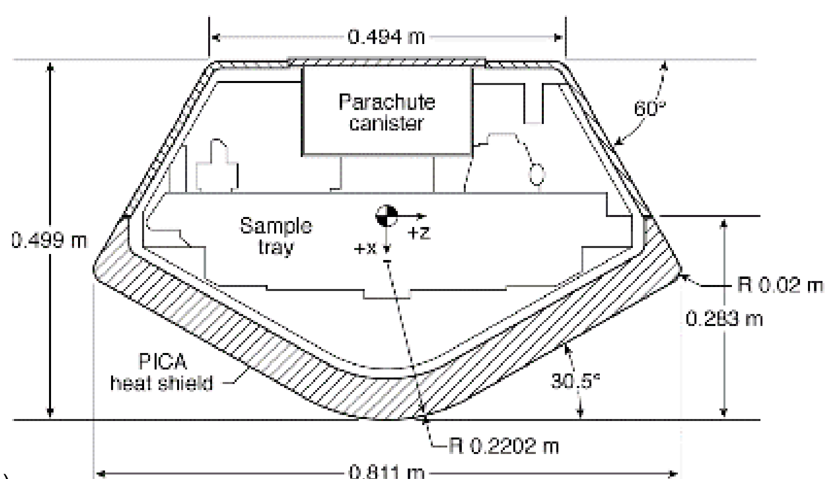


Figure 5.17.—Viking II atmospheric temperature data.



(a)



(b)

Figure 5.18.—(a) Atmospheric Entry Velocity Profiles for Various Mars Missions and (b) Stardust Aeroshell Geometry.

To minimize heat transfer into and out of the S/C during transit it is insulated using MLI. MLI is constructed of several layers of metalized material with a nonconductive spacer between the layers. The metalized material has a low absorptivity that resists radiative heat transfer between the layers. The insulation can be molded to conform to the interior of the aeroshell.

The MLI was used to line the complete interior of the aeroshell and backshell. Passthroughs, used to allow wiring and components to go through the insulation, were also accounted for in the insulation sizing and heat loss analysis. The amount of MLI is optimized for the thermal environment to maintain the RUHs and electronics within their desired operating temperature range.

The insulation modeling was based on radiative heat transfer analysis from the S/C interior through the MLI to space. The specifications for insulation sizing are shown below in Table 5.21.

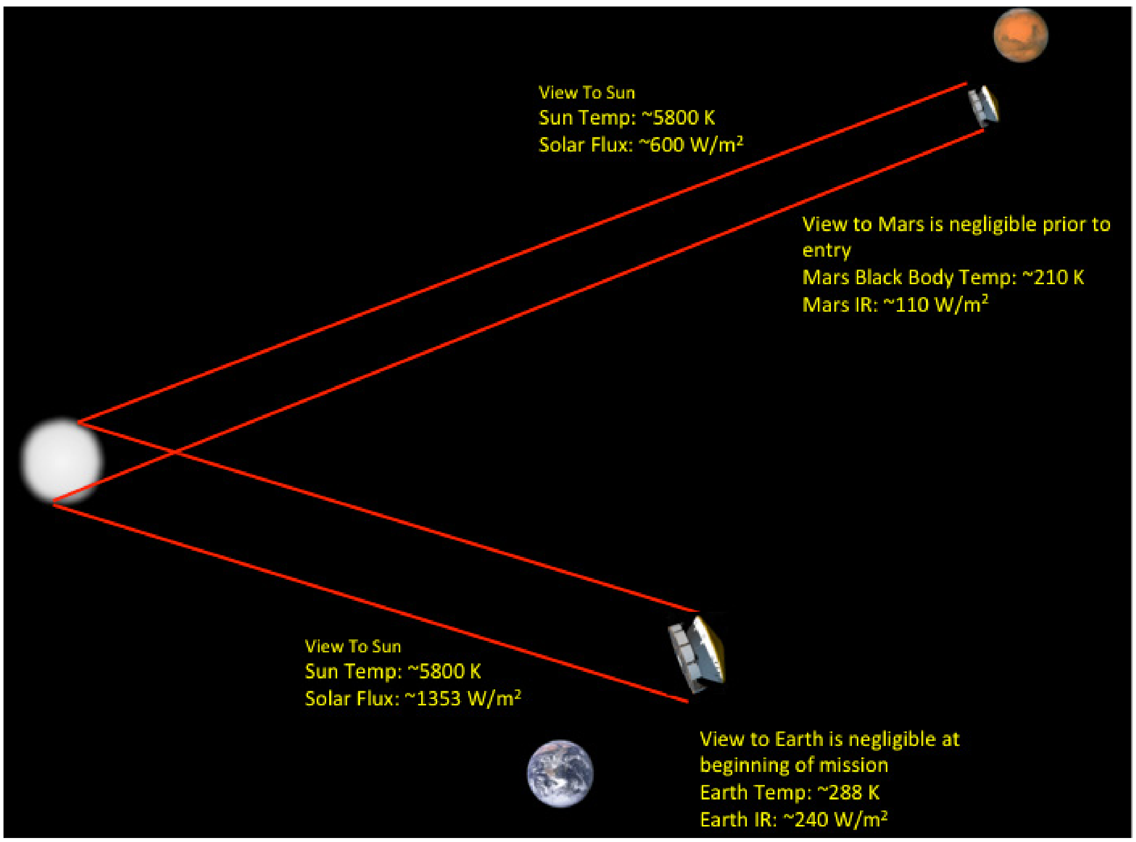


Figure 5.19.—Thermal Conditions in Transit to Mars.

TABLE 5.21.—THERMAL SYSTEM INSULATION SPECIFICATIONS

Variable	Value
MLI Emissivity (ε_i)	0.07
MLI Material	Al
MLI Material Density (ρ_i)	2,770 kg/m^3
Internal S/C Temperature (T_i)	300 K
MLI Layer Thickness (t_i)	0.025 mm
Number of insulation Layers (n_i)	10
MLI Layer Spacing (d_i)	0.5 mm
S/C Inner Wall Surface Emissivity	0.98
S/C Outer Wall Surface Emissivity	0.93

5.8.3 Lander Thermal Control

The thermal system for the lander was passive and utilized aerogel insulation to minimize heat loss to the surroundings and maintain the internal component temperatures within the required range of 233 to 323 K.

Aerogel is an open-cell insulation, which is lightweight with a very low thermal conductivity. The cells within the aerogel limit natural convection of the atmospheric gas within the insulation enabling the thermal conductivity of the insulation to approach that of the atmospheric gas. Aerogel is comprised of silica gel and is available in several different densities.

Since the insulation does not have strong mechanical properties an outer shell encases the insulation to prevent it from being crushed or damaged which would reduce its insulation properties. The aerogel insulation surrounds all internal components and extends 3 cm beyond the RHUs and top of the enclosure as illustrated in Figure 5.20.

The variables used in the baseline thermal analysis and results are listed in Table 5.22. An off-design analysis was also performed. This analysis looked at the operating temperature of the interior for the different RHU power levels (four, six, eight, and 12 RHUs) and over a range of insulation thicknesses. The maximum temperature for each power level occurred during the summer, maximum atmosphere temperature operating conditions and the minimum temperature for each power level occurred during the winter, minimum atmosphere temperature operating conditions. The results for this analysis are shown in Figure 5.21. The shaded area between the maximum and minimum temperature curves represents the operating temperature of the internal components throughout the year for a given insulation thickness. For example with six RHUs providing 6 W of thermal power (represented by the green curves and shaded area on the graph) and 3 cm of insulation on the outside of the enclosure the temperature will vary from approximately 252 to 295 K throughout the year.

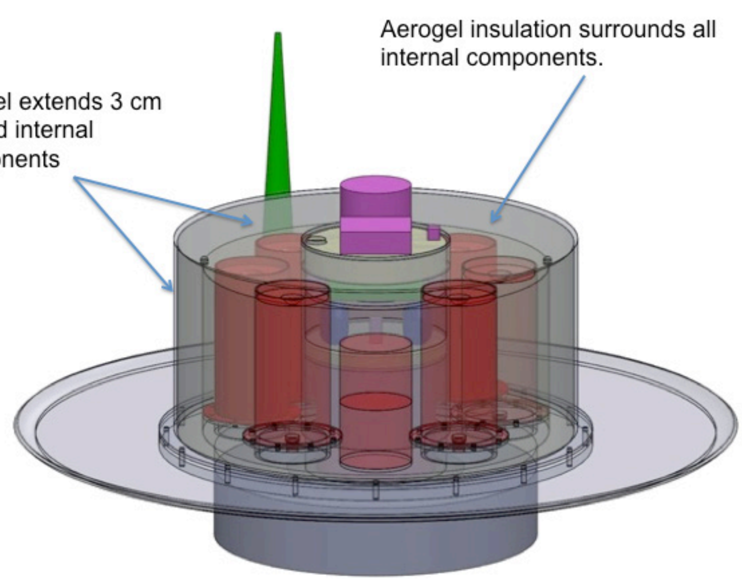


Figure 5.20.—Lander Aerogel Insulation.

TABLE 5.22.—SURFACE LANDER HEAT TRANSFER VARIABLES

Variable	Value
Aerogel Thermal Conductivity	0.016 W/mK
Aerogel Density	20 kg/m^3
Prandlt Number	0.802
Raleigh Number	3310
Nusselt Number	5.94
Convective Coefficient	1.44 W/m^{2}K
Design Point Insulation Thickness	3 cm
Design Point Max/Min Operating Temp	295/253 K

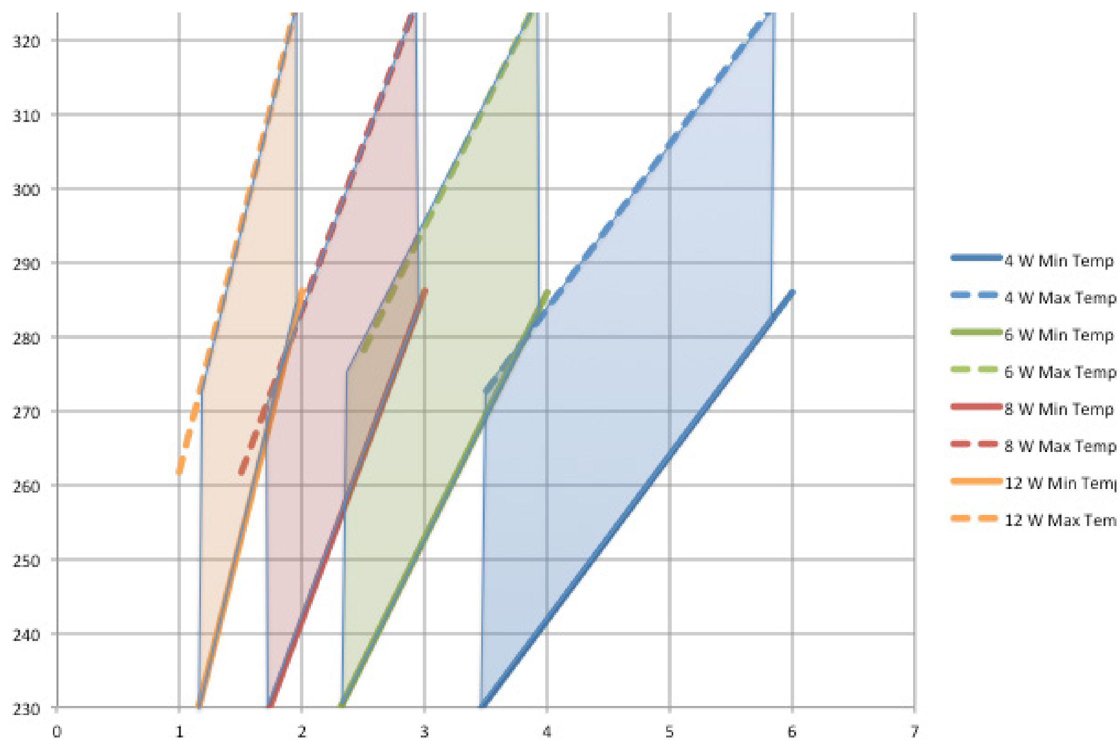


Figure 5.21.—Internal Temperature vs. Insulation Thickness for Various Power Levels.

5.8.4 Thermal System Mass Breakdown

Table 5.23 lists the MEL items for the thermal system in the MASER three element system.

TABLE 5.23.—MASER AEROSHELL/BACKSHELL AND LANDER THERMAL MEL

WBS no.	Description Case 1 Mars Hard Lander CD-2013-95, Aug. 2, 2013	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	Mars Hard Lander	---	------	50.10	19.6	9.82	59.92
06.1	Hard Lander	---	------	14.72	20.7	3.04	17.76
06.1.11	Structures and Mechanisms	---	------	10.03	18.0	1.80	11.83
06.1.11.a	Structures	---	------	9.73	18.0	1.75	11.48
06.1.11.a.a	*Primary Structures*	---	------	9.13	18.0	1.64	10.77
06.1.11.a.a.a	Main Body	1	6.18	6.18	18.0	1.11	7.29
06.1.11.a.a.d	Outer Cover	1	2.95	2.95	18.0	0.53	3.48
06.1.11.a.b	*Secondary Structures*		------	0.60	18.0	0.11	0.71
06.1.11.a.b.d	crush pad	1	0.10	0.10	18.0	0.02	0.11
06.1.11.a.b.e	8-32x1/2 bolt assembly RHU	48	0.00	0.16	18.0	0.03	0.19
06.1.11.a.b.f	8-32x1 bolt assembly Top Hat	16	0.00	0.07	18.0	0.01	0.08
06.1.11.a.b.g	Miscellaneous 06.1.11.a.b.g	3	0.09	0.27	18.0	0.05	0.32
06.1.11.b	Mechanisms	---	------	0.30	16.9	0.05	0.35
06.1.11.b.b	*Science Payload*	---	------	0.02	2.0	0.00	0.02
06.1.11.b.b.a	TiNi Aerospace P5-403 pin puller	2	0.01	0.02	2.0	0.00	0.02
06.1.11.b.f	*Installations*		------	0.28	18.0	0.05	0.33
06.1.11.b.f.d	C&DH Installation	1	0.16	0.16	18.0	0.03	0.19
06.1.11.b.f.e	C&T Installation	1	0.01	0.01	18.0	0.00	0.01
06.1.11.b.f.f	Electrical Power Installation	1	0.11	0.11	18.0	0.02	0.13
06.1.11.b.f.g	Thermal Control Installation	1	0.00	0.00	18.0	0.00	0.01
06.2	Aeroshell/Heat Shield	---	------	34.83	19.2	6.68	41.51
06.2.11	Structures and Mechanisms	---	------	6.02	15.6	0.94	6.96
06.2.11.a	Structures	---	------	5.00	18.0	0.90	5.90
06.2.11.a.a	*Primary Structures*	---	------	0.00	0	0.00	0.00
06.2.11.a.b	*Secondary Structures*	---	------	5.00	18.0	0.90	5.90
06.2.11.b	Mechanisms	---	------	1.02	3.6	0.04	1.06
06.2.11.b.e	*Adaptors and Separation*	---	------	1.02	3.57	0.04	1.06
06.3	Carrier Interface	---	------	0.55	18.0	0.10	0.65
06.3.11	Structures and Mechanisms	---	------	0.55	18.0	0.10	0.65
06.3.11.a	Structures	---	------	0.30	18.0	0.05	0.35
06.3.11.a.a	*Primary Structures*	---	------	0.30	18.0	0.05	0.35
06.3.11.a.a.a	Miscellaneous 06.3.11.a.a.a	1	0.30	0.30	18.0	0.05	0.35
06.3.11.b	Mechanisms	---	------	0.25	18.0	0.05	0.30
06.3.11.b.e	*Adaptors and Separation*	---	------	0.25	18.0	0.05	0.30
06.3.11.b.e.e	Miscellaneous 06.3.11.b.e.e	1	0.25	0.25	18.0	0.05	0.30

5.9 Radiation Exposure

Operation of the S/C will originate in near Earth and transition through the radiation belts out to deep space. The exposure time through Earth's radiation the belts would be minimal and not require additional radiation shielding. The only other significant source of radiation will be from cosmic rays and solar particle events encountered throughout the mission. Based on the operational life of the mission standard 50 krad electronics should be sufficient to withstand the expected radiation environment.

5.9.1 Thermal Risk Inputs

Although the thermal system is passive there are still a potential for the MLI or aerogel to fail.

Damage to the MLI in the aeroshell or the aerogel insulation in the lander would cause the insulating capability is significantly reduced.

The insulation is utilized to maintain the interior temperature of the S/C and lander. If does not function properly it could lead to the failure of the electronics, component or propulsion system and jeopardize all or part of the mission.

The insulation is a critical part of the thermal control system. Although loss of the insulation is extremely unlikely, any degradation in its performance can jeopardize the mission and vehicle. The mitigation approach is to inspect the insulation installation prior to launch to insure it will not come loose or be dislodged during launch. Debris impacts or the hard landing can tear or remove chucks of insulation. To mitigate any insulation degradation, designing the system to operate above the minimum electronics temperature would provide some margin within the design to accommodate minor damage to the insulation.

5.10 Planetary Protection

5.10.1 Planetary Protection Requirements

PP Category IVc: Missions investigating Martian special regions (Figure 5.22).

- Implementation Requirements: Impact avoidance and contamination control, including cleanroom assembly, microbial reduction, trajectory biasing, organics archiving.
- Planning and Documentation Requirements: Mission Certification, Planetary Protection Plan, Planetary Protection Implementation Plan, Pre-Launch Planetary Protection Report, Post-Launch Planetary Protection Report, (Planetary Protection Extended Mission Report), End-of-Mission Report.
- Required Reviews: Project Planetary Protection Planning Review, Pre-Ship Planetary Protection Review, Pre-Launch Planetary Protection Review, and Flight Readiness Review.
- Mars 'Special Region' Specific Requirements:
  - No more than an average of 300 spores per square meter of exposed external and internal S/C surfaces.
  - The entire landed system shall be restricted to a surface biological burden level of 30 spores.

5.10.2 Planetary Protection Assumptions

The lander was designed with planetary protection in mind, specifically the provisions laid out in NPR 8020.12D (Ref. 10) and Hand, 2009 (Ref. 11).

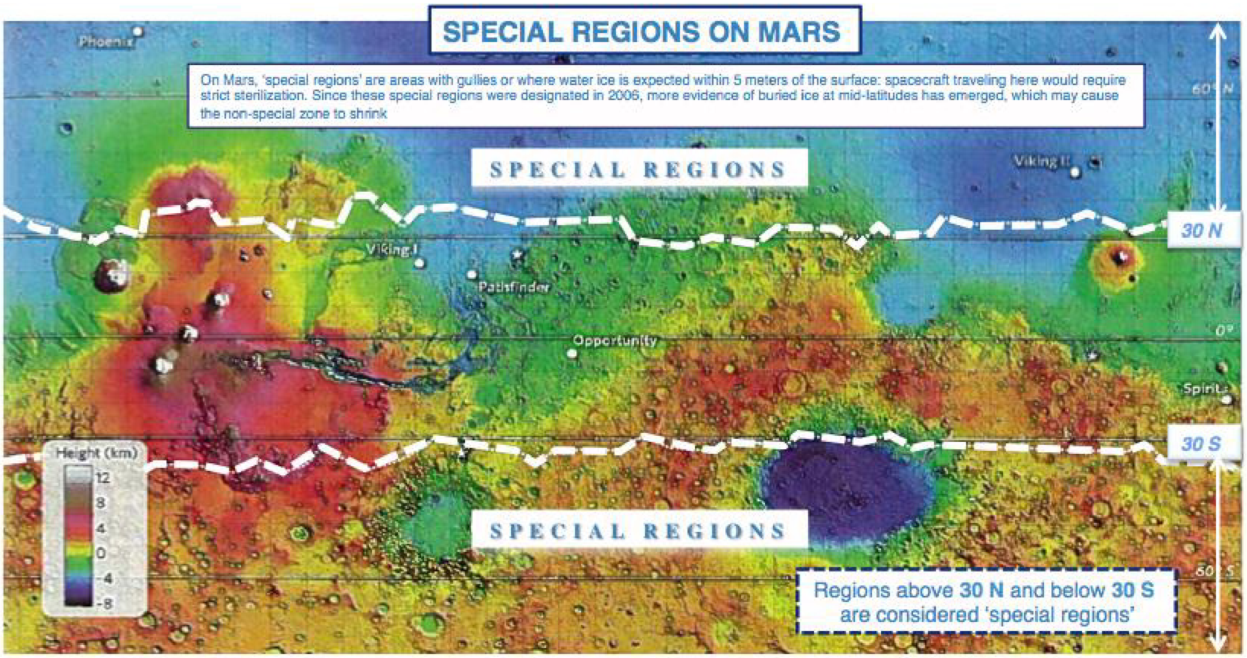


Figure 5.22.—Special Regions of scientific interest on Mars.

5.10.3 Planetary Protection Microbial Reduction Plan

- S/C design will include components compatible with desired microbial reduction treatment (dry heat or other approved process).
- Components compatible with heat will be submitted to nominal Dry Heat Microbial Reduction (DHMR) cycles, i.e., 35 to 50 hr at temperatures of 111 to 125 °C (surviving fraction of hardy organisms is 1×10^{-4}).
- Items on the landed system that are not compatible with DHMR will be treated with Vaporous Hydrogen Peroxide (VHP).
- After the microbial reduction process, a microbial barrier or biobarrier will be necessary to protect an S/C or associated component(s) against microbial recontamination.
- The treated components will be aseptically assembled at the launch site and contained in the lander bioshield.

The surfaces interior to the bioshield will then be submitted a final VHP treatment.

Microbial reduction calculations shall be supported by data from reproducible laboratory tests or by suitable technical references for parameters not specified in the Parameter Specification Sheet in NPR 8020.12D (Ref. 10).

5.10.4 Planetary Protection Analytical Methods

- RHS Mission Bioshield = 5 kg
- Mass estimated scaled from Phoenix arm biobarrier.
- 1.75 kg for 235- by 9- by 9-cm (surface area 0.8622 m^2)

Phoenix biobarrier material, Tedlar, a trademarked polyvinylflouride material with commercial uses ranging from durable surfaces of airline cabin furnishings to backing sheets for photovoltaic panels. The biobarrier film is supported by a skeleton of spring-loaded, Al-tube ribs to maintain its shape. On the Martian surface, the springs retract the ribs and the film, allowing the arm to deploy.

Bioshield will be structurally similar to the Viking bioshield, see Figure 5.23, Figure 5.24 and Figure 5.25.

Figure 5.23.—Viking Lander 1 in Aeroshell (1975).

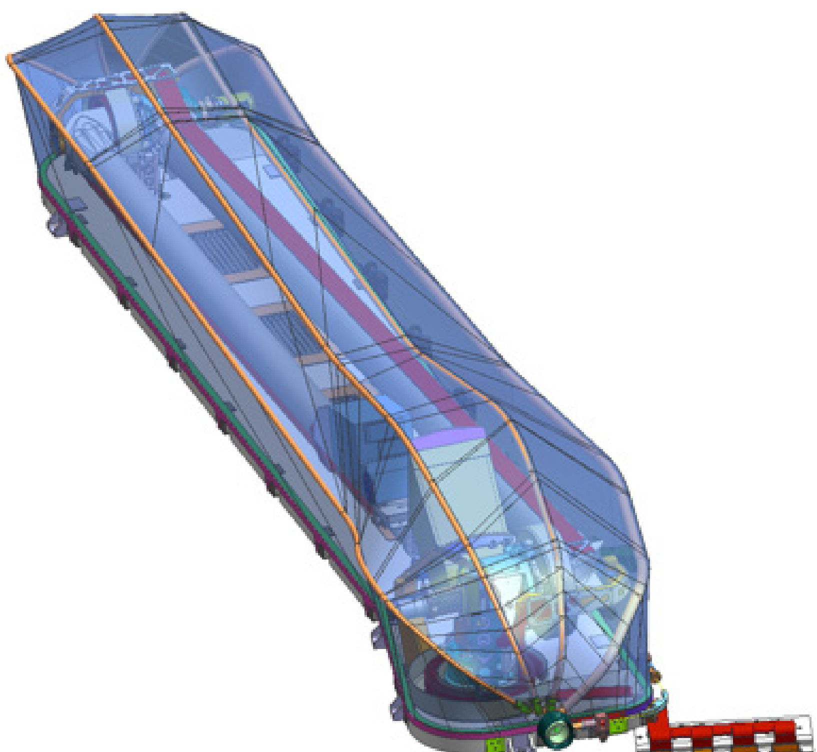


Figure 5.24.—Viking Bioshield.

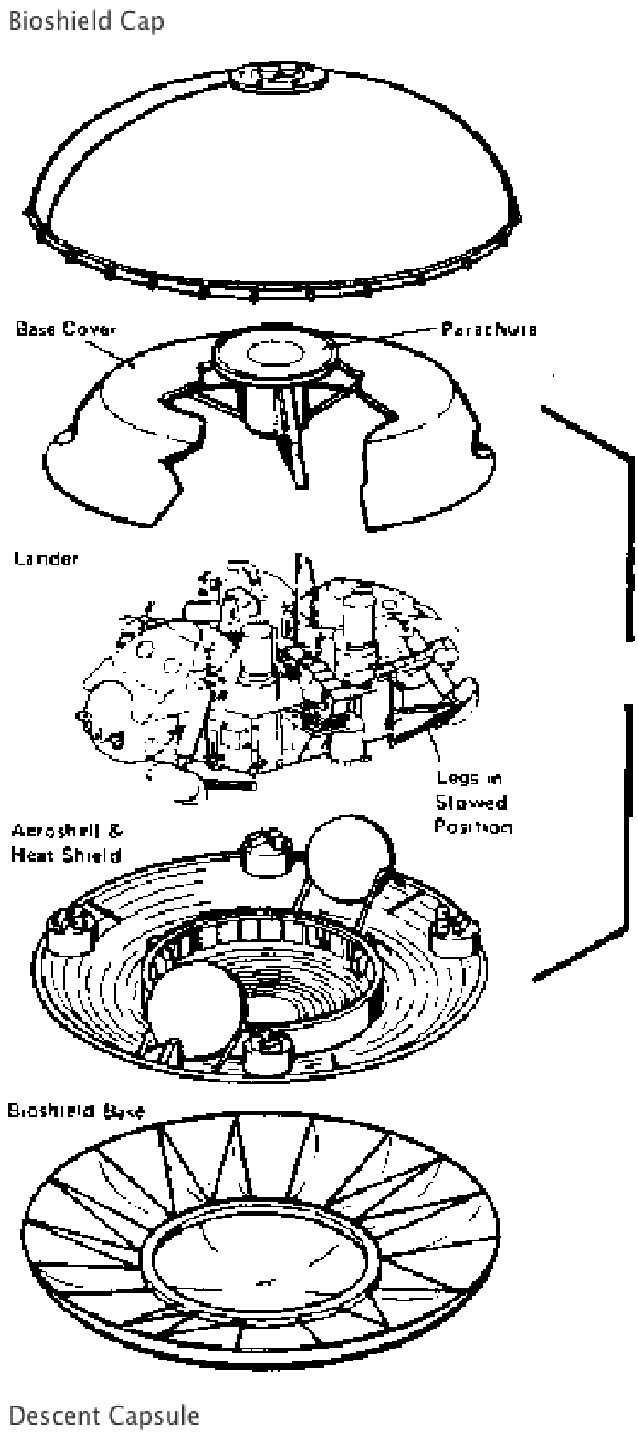


Figure 5.25.—Viking shown in Bioshield cap and aeroshell cover.

5.11 ATLO Power System Installation Integrated with Planetary Protection

5.11.1 Background

Each lander power system consists of six RHU-RPS (1.0 Wth each) and four ultracapacitors. Assembly of the lander, power system integration, sterilization process and bioshield installation is baselined to occur in the KSC Payload Hazardous Processing Facility (PHSF).

RTGs are typically installed at the launch pad complex as the case with MSL (MMRTG, ~110 We, 2000 Wth), PNH and Cassini missions (GPHS RTG, ~290 We, 4500 Wth). Radiological health physics safety is certainly the prime factor for this customary process due to the amounts of radioisotope inventory associated with the larger RTGs. They are physically larger and heavier than RHUs and have a significant heat management need and thus require special installation and cooling equipment and are installed via special door(s) cut into the launch vehicle fairing. The physical size of an RHU—C size battery—creates a concern hazard for installing them on the S/C once atop a launch vehicle. Therefore, RHUs are typically integrated with the S/C at the PHSF just before transport to the launch site. It is anticipated that the size of the RHU-RPS will follow the installation protocols established for RHUs adopted on some previous missions (e.g., Pathfinder (Ref. 3)). Also, thermal assessments indicate no additional cooling is required beyond that provided by the standard launch vehicle services.

5.11.2 Installation Assumptions

The S/C design dictates that the RHU-RPS units be installed within an insulation cap. Therefore, terminal Category IVc sterilization, baselined for this mission must be done with the RHU-RPS and ultracapacitors integrated within the lander. DHMR of the entire lander (Viking "gold" standard) would most likely be the sterilization option of choice; however, the ultracapacitors cannot tolerate temperatures above 65 °C, which is well below temperatures seen with DHMR (see Planetary Protection, Sec. 5.10). Therefore, a two-step sterilization process is envisioned using DHMR coupled with a final VHP to accommodate the capacitor installation.

Thus, the Team's baseline approach for lander sterilization is to utilize DHMR for all hardware except the ultracapacitors and also perform a final VHP sterilization for the assembled lander's external surfaces. A bioshield will be utilized to surround each lander similar to that used for the Phoenix arm.

5.11.3 Installation CONOPS

Figure 5.26 illustrates the installation of the lander power system. All six RHU-RPS are permanently mounted and electrically connected to the lander bus. The four ultracapacitors are mounted and electrically connected to perform the full up power system checkout. Following check out, the capacitors are then removed so that the lander can be DHMR. The capacitors will later be sterilized during the subsequent VHP process.

All lander hardware except for the capacitors receives DHMR sterilizing (Figure 5.27) all lander internal components, insulation dome, parachutes, cables, etc. that would land on the surface of Mars. A temporary biobarrior is used to prevent hardware recontamination during subsequent ATLO assembly steps.

Lander hardware is placed in the VHP facility as shown in Figure 5.28. Required tools and fasteners for final S/C assembly are placed inside the room for sterilization. The room is purged with argon and filled with VHP where the hardware is left exposed for approximately 2 to 3 hr, sterilizing external surfaces of the temporary biobarriers and the four ultracapacitors. The VHP is vented and again purged with argon. A sterilization assay swab will be taken to verify the lander bioburden. The facility would be fitted with an access panel and gloves for final assembly of the lander.

Installation of the ultracapacitors first requires removal of the insulation dome. The four ultracapacitors are mounted and electrically connected. The insulation dome is then mounted to the lander skid plate followed by installing the back shell to the heat shield. The flight bioshield is mounted to the back shell to maintain its sterilization integrity. The heat shield will get hot enough during Mars atmospheric entry to self-sterilize. These steps are followed for each of the other three landers.

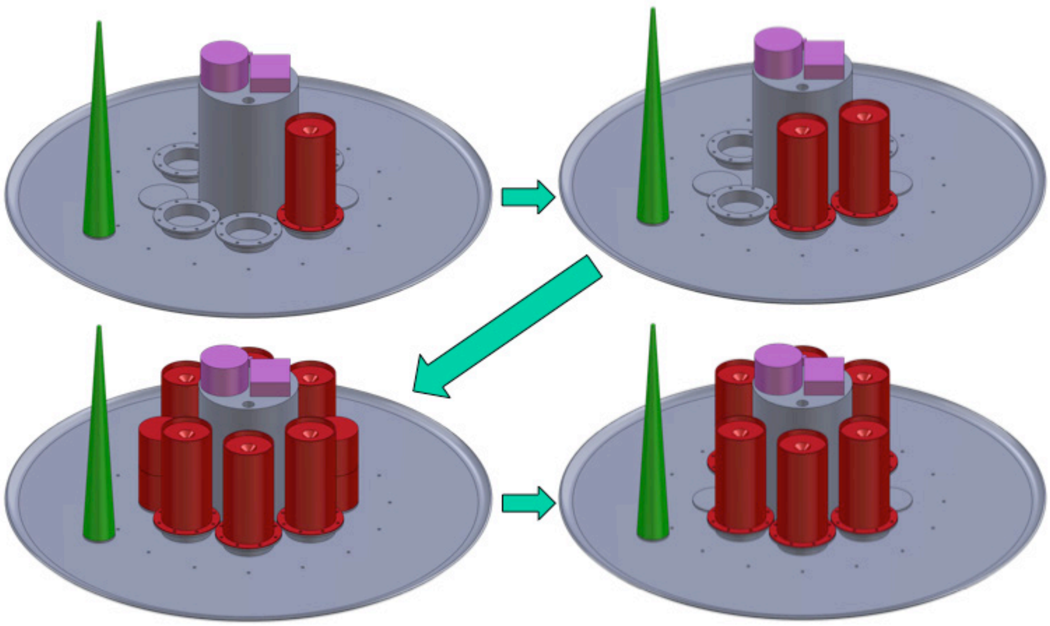


Figure 5.26.—RHU-RPS Installation Sequence.

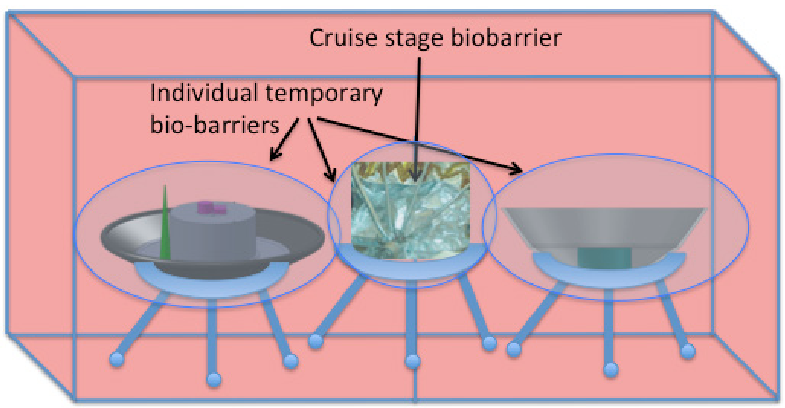


Figure 5.27.—Lander Hardware in the DHMR Facility.

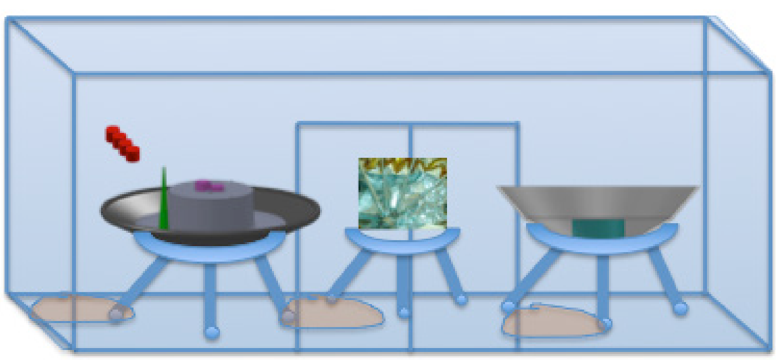


Figure 5.28.—VHP Facility.

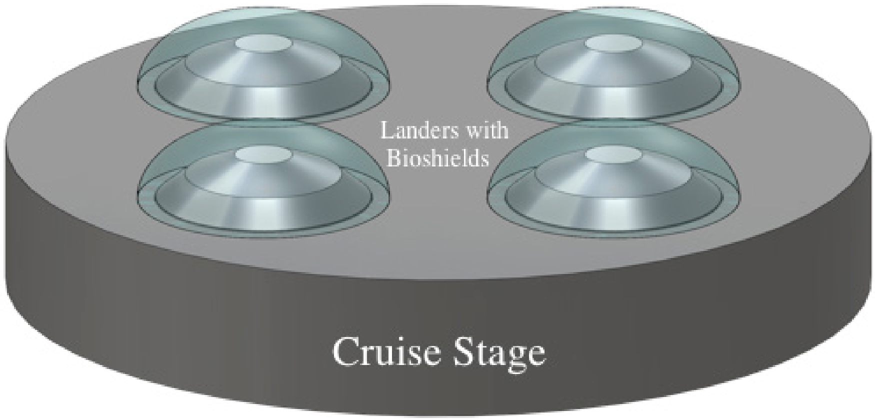


Figure 5.29.—Cruise Stage Flight Configuration.

Other options exist where the chamber for DHMR and VHP could be the same facility and the interim biobarriers would not be needed since the hardware would not have to be removed from a sterile environment.

Figure 5.29 shows each of the four landers within their bioshields. As the S/C approaches Mars entry, the bioshields will be retracted to allow jettison of each lander, as was similarly done on the Viking mission.

This plan has been formulated with knowledge and guidance from NPR 8020.12D, April 2011 (Ref. 10). Neither the NASA Planetary Protection Officer, nor anyone from the office, has reviewed this plan. The team has developed a conservative approach and was reflected in the cost estimations. Further study would hopefully reveal a simpler more streamlined process.

6.0 Cost and Risk

6.1 Cost

Please note that the cost estimates presented in this section should be considered rough order of magnitude (ROM) costs for the S/C that is early in its design phase.

To estimate the cost of the Mars Hard Lander Study S/C design, the MEL generated by the Compass team is linked to an Excel-based cost model. Costs are estimated at the subsystem and component levels using mostly mass-based, parametric relationships developed with historical cost data. Quantitative risk analysis is performed on these costs using Monte Carlo simulation based on mass and CER uncertainties. The pertinent cost modeling assumptions apply for these designs:

- The S/C is assumed to be developed using a proto-flight development approach for all subsystems.
- No ground spares are included.
- The science payload is estimated with the NASA Instrument Cost Model (NICM) using the following parameters to best approximate the expected operating environment:
    - Instrument Type: In-situ
    - Location: Arm/Mast
    - Flagship: No
    - The individual Instruments are grouped/estimated as follows:
        - *Wind Sensor*—Mass: 0.1 kg, Power 1 W, Total Cost: $500K

- *Pressure and Temperature Sensors*—Mass: 0.1 kg, Power 1 W, Total Cost: $500K
    - *Seismometer*—Mass: 1.4 kg, Power 1 W, Total Cost: $1,700K
    - *Optical Monitor*—Mass: 0.1 kg, Power 1 W, Total Cost: $500K
    - *Entry/Impact Accelerometers*—Mass: 0.1 kg, Power 1 W, Total Cost: $500K
    - Note: The minimum accepted inputs for NICM are mass of 0.1 kg and power of 1 W
  - The science instrument costs are split 70/30 between development and flight hardware costs, respectively.
- The technology development costs for the RPS system are assumed to be covered by the RPS office and are not included in the following estimates:
- The flight hardware costs for the RPS system is assumed to be government furnished equipment (GFE) and, therefore, no cost for the flight hardware is included in the following estimates.
- This parametric modeling approach assumes that all components are at TRL-6; therefore, this section does not include any technology development costs necessary to bring any technology up to this level.
- To account for planetary protection, the following assumptions apply:
  - The bioshield is estimated based on the bioprotection system used for the robotic arm on the Mars Phoenix mission.
  - The cost of the bioshield is included within the aeroshell/heat shield subsystem.
  - Planetary Protection requirements are also addressed within the systems integration element as an additional SE&I element cost (actual costs shown in the cost details), also based on Phoenix.
- To try to capture some of the cost uncertainly associated with Planetary Protection, the cost estimate is generated for several scenarios/cases. The baseline cost estimate includes planetary protection based on the Phoenix arm. The cost of PP is then calculated by removing all PP costs and subtracting this estimate from this baseline case. To generate range of PP costs, a high-end for PP is generated by applying a complexity factor of 2.0 to both the bioshield and the SE&I for PP CERs.
- Standard planetary systems integration wraps are used to determine costs for Integration, Assembly and Check-out (IACO), Systems Test Operations (STO), Ground Support Equipment Hardware (GSE), Systems Integration and Test (SE&I), Program Management (PM) and Launch and Orbital Operations Support (LOOS).
- The cost estimate represents the ‘most likely’ point estimate based on the cost risk simulation results and roughly equates to the 35th percentile on a pseudo-lognormal distribution
- The cost of propellant is not included in these estimates.
- Costs are in this section are all in fiscal year 2013 (FY13) $M.

The Compass team requested a cost estimate for a single S/C design as well as a lifecycle cost estimate for the mission using four S/C flight articles. The design, development, testing and engineering (DDT&E) represents the non-recurring costs associated with the first S/C while the flight hardware represents the recurring cost for a single S/C. The most-likely cost risk simulation results for the Compass S/C DDT&E only (including system integration wraps and prime contractor fee) are shown in Table 6.1 in FY13 $M.

TABLE 6.1.—COMPASS SUBSYSTEM LEVEL COST BREAKDOWN—BASELINE CASE

WBS	Description	DDT&E	FH	Total
06.1.1	Science	3.4	1.2	4.6
06.1.3	Command & Data Handling	2.0	1.3	3.2
06.1.4	Communications and Tracking	1.0	0.6	1.7
06.1.5	Electrical Power Subsystem[a]	0.7	0.1	0.8
06.1.3	Thermal Control (Non-Propellant)	0.2	0.1	0.3
06.1.11	Structures and Mechanisms	1.0	0.4	1.4
06.2	Aeroshell/Heat Shield	11.9	5.0	16.9
	Subtotal	20.2	8.8	29.0
	IACO	1.3	0.4	1.7
	STO	1.2	1.2	-----
	GSE Hardware	2.4	2.4	-----
	SE&I	6.2	2.1	8.3
	SE&I: Planetary Protection	3.6	2.4	6.0
	PM	4.5	0.6	5.1
	LOOS	2.0	2.0	-----
	Spacecraft Total	41.4	14.2	55.6
	Prime Contractor Fee (10%)	4.1	1.4	5.6
	Total Project with Fee included	45.5	15.6	61.2

[a]Does not include any development or flight cost for RPS related power system

The total development cost range for this project (excluding any RPS related costs) is on the order of \$45M to \$54M with Planetary Protection estimated to be on the order of \$9M to \$17M. The flight unit cost range for this project (excluding any RPS related costs) is on the order of \$16M to \$21M for a single flight unit with Planetary Protection estimated to be on the order of \$5 to \$10M (for each flight article). In total, the development and flight unit cost range for a single flight article (excluding any RPS related costs) is on the order of \$61< to \$75M with Planetary Protection estimated to be on the order of \$14M to \$27M. The impact of the PP on the total mission, including additional lander flight articles, is detailed in the next section.

Table 6.3 shows a partial lifecycle cost estimate in FY13\$M but does not include any costs for mission operations or launch costs. NASA insight/oversight for the mission is estimated as 12 percent of the prime contractor cost less fee. Phase A costs are estimated at 5 percent of the development cost less fee. The development cost (excluding any RPS related costs) comes directly from Table 6.1 and Table 6.2. The flight unit cost includes four flight articles at the two different cost estimates but as noted earlier, excludes any cost associated with the GPHS. For this initial estimate, no learning is assumed and therefore each flight unit is estimated at the same unit cost. LV costs are not included in this partial lifecycle cost estimate; the mission is assumed to be a ride share at no cost to the mission. Any additional costs to account for NEPA and NLSA compliance as part of additional launch service fees associated with launching nuclear material are also excluded from this cost section; these costs could exceed \$20M. The Mission Operations and Ground Data Systems (GDS) costs are also excluded in this initial cost estimate. Finally, reserves are calculated at 30 percent of the identified cost categories for Phase A-D (less any fee).

TABLE 6.2.—COMPASS SUBSYSTEM LEVEL COST BREAKDOWN— HIGH-END PP CASE

WBS	Description	DDT&E	FH	Total
06.1.1	Science	3.5	1.2	4.7
06.1.3	Command & Data Handling	2.0	1.3	3.2
06.1.4	Communications and Tracking	1.0	0.7	1.7
06.1.5	Electrical Power Subsystem[a]	0.7	0.1	0.8
06.1.3	Thermal Control (Non-Propellant)	0.2	0.1	0.3
06.1.11	Structures and Mechanisms	1.0	0.4	1.4
06.2	Aeroshell/Heat Shield	14.1	6.5	20.6
	Subtotal	22.4	10.3	32.8
	IACO	1.5	0.4	1.9
	STO	1.3	1.3	-----
	GSE Hardware	2.7	2.7	-----
	SE&I	6.7	2.4	9.2
	SE&I: Planetary Protection	7.3	4.8	12.1
	PM	4.8	0.7	5.5
	LOOS	2.3	2.3	-----
	Spacecraft Total	49.1	18.6	67.8
	Prime Contractor Fee (10%)	4.9	1.9	6.8
	Total Project with Fee included	54.1	20.5	74.6

[a]Does not include any development or flight cost for RPS related power system

TABLE 6.3.—PARTIAL LIFECYCLE COST RANGE FOR THE MARS HARD LANDER MISSION (LOW END)

Launch of Four Systems	Low-end	High-end	Notes
NASA insight/oversight	7	8	12 percent of prime contractor costs (less fee) Phase A
Phase A	2	2	5% of Development Cost (less fee)
Prime Contractor Development	46	54	Prime Contractor Costs Plus Fee (10%)
Flight Hardware	63	82	Estimated lander cost for four flight articles (including fee)
Mission Ops	0	0	TBD
UFE	32	40	30% Unallocated Future Exp (less fee)
Total	149	187	

All costs in FY13$M
Mission Operations costs are currently not included
Any Launch Costs are excluded

Overall, the initial ROM costs show that the mission is on the order of $149 to $187M with an estimated $40 to $70M being directly attributed to Planetary Protection costs. The life cycle cost estimate for this mission will increase as any of the following additional mission parameters are included: mission operations costs, any relay satellite costs, NEPA compliance costs for launch services, any launch costs, and any other additional mission related expenses.

6.2 Integration, Assembly and Checkout (IACO)

The IACO element contains all labor and material required to physically integrate (assemble) the various subsystems into a total system. Final assembly, including attachment, and the design and manufacture of installation hardware, final factory acceptance operations, packaging/crating, and shipment are included. IACO charged to DDT&E represents those costs incurred for the integration, assembly, and checkout of major test articles. IACO charged to the flight unit includes those same functions applied to the actual flight unit.

This item excludes the engineering effort required to establish the integration, assembly, and checkout procedures necessary for this effort. These engineering efforts are covered under systems engineering and integration.

6.3 System Test Operations (STO)

The STO element includes development testing and the test effort and test materials required for qualification and physical integration of all test and qualification units. Also included is the design and fabrication of test fixtures.

Specifically included are tests on all STH to determine operational characteristics and compatibility with the overall system and its intended operational parameters. Such tests include operational tests, design verification tests, and reliability tests. Also included are the tests on systems and integrated systems to verify acceptability for required mission performance. These tests are conducted on hardware that has been produced, inspected, and assembled by established methods meeting all final design requirements. Further, system compatibility tests are included, as well as, functions associated with test planning and scheduling, data reduction, and report preparation.

6.4 Ground Support Equipment (GSE)

Functional elements associated with GSE include the labor and materials required to design, develop, manufacture, procure, assemble, test, checkout, and deliver the equipment necessary for system level final assembly and checkout. Specifically, the equipment utilized for integrated and/or electrical checkout, handling and protection, transportation, and calibration, and items such as component conversion kits, work stands, equipment racks, trailers, staging cryogenic equipment, and many other miscellaneous types of equipment are included.

Specifically excluded is the equipment designed to support only the mission operational phase.

6.5 Systems Engineering and Integration (SE&I)

The functions included in the SE&I element encompass: (1) the system engineering effort to transform an operational need into a description of system requirements and/or a preferred system configuration; (2) the logistics engineering effort to define, optimize, and integrate logistics support considerations to ensure the development and production of a supportable and cost effective system; and (3) the planning, monitoring, measuring, evaluating, and directing of the overall technical program. Specific functions include those for control and direction of engineering activities, cost/performance trade-offs, engineering change support and planning studies, technology utilization, and the engineering required for safety, reliability, and quality control and assurance. Also included is the effort for system optimization, configuration requirements analyses, and the submittal and maintenance of Interface Control Documents (ICDs).

Excluded from the SE&I element are those functions which are identifiable to subsystem SE&I.

6.6 Program Management (PM)

Elements included in the PM function consist of the effort and material required for the fundamental management direction and decision-making to ensure that a product is developed, produced, and delivered.

Specifically included are direct charges for program administration, planning and control, scheduling and budgeting, contracts administration, and the management functions associated with engineering, manufacturing, support, quality assurance, configuration and project control, and documentation.

The PM element sums all the effort required for planning, organizing, directing, coordinating, and controlling the project to help ensure that overall objectives are accomplished. This element also includes the effort required to coordinate, gather, and disseminate information.

Excluded from the PM element are those functions commonly charged to subsystem level activities.

6.7 Launch and Orbital Operations Support (LOOS)

This category includes the effort associated with pre-launch planning, launch and ascent, and initial on-orbit operations. The pre-launch activities include bus and payload preparation, as well as interface activities with the launch vehicle.

The launch and ascent period includes final assembly, checkout, and fueling, lift-off, telemetry, pre-launch TT&C, recovery operations, and post-processing of lift-off data. Final on-orbit support includes maintenance of the ADCS operation, attitude and orbit control, support of on-orbit testing, routine monitoring and fault detection of space vehicle subsystem functions, and support of anomaly investigation and correction. This period ends when the newly deployed satellite is turned over to the operational user, typically after a period of 30 days.

7.0 Trade Space Iterations

Only one design was performed in this study.

Appendix A.—Acronyms and Abbreviations

ΔV	delta velocity
6DOF	six degrees-of-freedom (Monte Carlo simulation)
AA	Associate Administrator
ACS	Attitude Control System
AD&C	Attitude, Determination & Control
AF	U.S. Air Force
AFRL	U.S. Air Force Research Lab
AIAA	American Institute for Aeronautics and Astronautics
ANSI	American National Standards Institute
ANSYS	Analysis System
AO	Announcement of Opportunity
AOS	acquisition of signal
APL	Applied Physics Laboratory
APU	auxiliary power unit
APXS	Alpha Particle X-ray Spectrometer
ARC	NASA Ames Research Center
ARD	Architecture Requirements Document
ASMO	American Student Moon Orbiter
ASRG	Advanced Stirling Radioisotope Generators
ATCS	Active Thermal Control System
AWG	American Wire Gauge
BAE	British Aerospace
BDM	Boost Deceleration Motors
BER	bit error rate
BOL	beginning of life
BOM	beginning of mission
C&DH	Command and Data Handling
C&T	command and telemetry
C&TN	Communications & Tracking Network
C/CAM	Collision/Contamination Avoidance Maneuver
CAD	computer aided design
CaLV	cargo launch vehicle
CAM	collision avoidance maneuver
CARD	Constellation Architecture Requirements Document
CAT	Cryogenic Analysis Tool
CBE	current best estimate
CCB	Common Core Booster
CCF	common cause factor
CER	cost estimating relationships
CEV	crew exploration vehicle
CFE	customer furnished equipment
CG	center of gravity
CHAMP	CHAllenging Our Minisatellite Payload
CLV	crew launch vehicle
CM	crew module
CMG	control moment gyro
COM	center of mass
Comm	communications
CONOPS	Concept of Operations
COPV	composite overwrapped pressure vessel
COTS	commercial off the shelf
COTS	NASA Commercial Orbital Transportation Services
CTC	command and telemetry computers
CTN	Constellation Tracking Network
CxP	Constellation Exploration Program
CY	calendar year
DDD	Design Definition Document
DDT&E	design, development, test, and evaluation
DMR	design for minimum risk
DOD	depth of discharge
DOE	Department of Energy
DOF	degree(s) of freedom
DPAF	dual payload attach fitting

DRL	Germany Aerospace Center
DSN	Deep Space Network
DTE	direct to Earth
Eb/N0	energy per bit to noise power spectral density ratio
ECLS	Environmental Control and Life Support
ECLSS	Environmental Control and Life Support System
EDL	entry, descent, and landing
EDS	Earth departure stage
EELV	evolved expendable launch vehicle
EGA	Earth gravity assist
EIRP	equivalent isotropic radiated power
ELMO	electric mission optimizer
ELV	expendable launch vehicle
EMA	electromechanical actuators
EMI	electromagnetic interference
EOL	end of life
EOM	end of mission
EP	electric propulsion
EPOXI	Extrasolar Planet Observation
ERV	Earth Return Vehicle
ESA	European Space Agency
ESAS	Exploration Systems Architecture Study
ESM	Encapsulated Service Module
ESMO	European Student Moon Orbiter
ESPA	EELV secondary payload adaptor
EVA	extra-vehicular activity
EVR	extra-vehicular robotics
FC	flight computers
FEA	finite element analysis
FEM	finite element model
FOM	figure(s) of merit
FPGA	Field Programmable Gate Array
FPR	flight performance reserve
FRAM	flight releasable attach mechanism
FRT	full rated thrust
FTE	full time equivalent
FY	fiscal year
GaAs	gallium arsenide
GLIDE	GLobal Integrated Design Environment
GN&C	Guidance, Navigation and Control
GPHS	general purpose heat source
GPS	Global Positioning System
GRC	NASA Glenn Research Center
GS	Ground Systems
GSFC	NASA Goddard Space Flight Center
GTO	geostationary transfer orbit
GTO	geosynchronous transfer orbit
hab	habitat
HAPS	Hydrazine Auxiliary Propellant System
HF	high frequency
HGA	high gain antenna
HiVHAC	High Voltage Hall Accelerator
HQ	NASA Headquarters
ICD	Interface Control Document
ICU	Instrument Control Unit
IDAC3	Integrated Deign Analysis Cycle 3
IEM	integrated electronics module
IIE	Innovative Interstellar Explorer
ILN	International Lunar Network
IMDC	I M Design Center
IMU	inertial measurement unit
IOS	Internetwork Operating System
IP	internet protocol
IRD	Interface Requirements Document
I_{sp}	specific impulse
ISPT	In-Space Propulsion Technologies
ISRU	in situ resource utilization
ISS	International Space Station
IV&V	independent verification and validation
IVT	interface verification test

JPL	NASA Jet Propulsion Laboratory
KBO	Kuiper belt objects
KSC	NASA Kennedy Space Center
LAN	local area network
LAT	Lunar Architecture Team
LAT2	Lunar Architecture Team 2
LCC	launch commit criteria
LCC	life cycle costs
LCCD	line charge coupling device
LCT	Lunar Communications Terminal
LEMS	Lander Environmental Monitoring Station
LEO	low Earth orbit
LGA	low gain antenna
LIDAR	Laser Detection and Ranging
Li-SO_2	lithium sulfur dioxide
LL	Lunar Lander
LLO	low lunar orbit
LM	Lockheed Martin
LMO	low Mars orbit
LNA	low-noise amplifier
LOC	loss of crew
LOI	lunar orbit insertion
LOM	loss of mission
LORRI	Long Range Reconnaissance Imager
LPRP	Lunar Precursor and Robotic Program
LRO	Lunar Reconnaissance Orbiter
LRS	Lunar Relay Station
LSAM	Lunar Surface Access Module
LSC	linear shaped charge
LSP	Launch Service Program
LSTO	Launch Service Task Order
MAC	media access control
MAT	Mars Architecture Team
MAV	Mars Ascent Vehicle
MECO	main engine cutoff
MEL	Master Equipment List
MESSENGER	MErcury Surface, Space ENvironment, GEochemistry, and Ranging
MGA	mass growth allowance
MIT	Minimum Impulse Thruster
MLI	multilayer insulation
MMH/NTO	monomethyl hydrazine and nitrogen tetroxide bipropellant system
MMOD	micrometeoroid and orbital debris
MO	Mission of Opportunity
MPS	main propulsion system
MPU	makeup power unit
MPU	mobile power unit
MS	mission systems
MSFC	NASA Marshall Spaceflight Center
MSL	Mars Science Laboratory
MSR	Mars Sample Return (mission)
N/A	not applicable
N_2	nitrogen
NAC	narrow angle camera
NASA	National Aeronautics and Space Administration
Nav	navigation
NEAR	Near Earth Asteroid Rendezvous
NEXT	NASA Evolutionary Xenon Thruster
NG SIRU	Northrop Grumman Scalable Inertial Reference Unit
NGIMS	Neutral Gas and Ion Mass Spectrometer
NH	New Horizon
NIMS	Near Infrared Mapping Spectrometer
NIST	National Institute of Standards and Technology
NLS	NASA Launch Services
OEP	Office of Educational Programs
OML	outer mold line

OMS	orbital maneuvering system
OS	operating system
OSS	Office of Space Science
OTH	over the horizon
OTS	off-the-shelf
PAF	payload attach fitting
PEL	Power Equipment List
PICA	Phenolic Impregnated Carbon Ablator
PLUTO	PLanetary Underground Tool
PMAD	power management and distribution
PMC	polymer matrix composites
PMS	Propellant Management System
PN	pseudo-noise
PNP	probability of no penetration
PoD	point of departure
PPO	Planetary Protection Officer
PSD	Planetary Science Division
PU	power processing unit
RAD	radiation dosimetry
RAID 5	Redundant Array of Independent Disks
RAM	random access memory
RBI	regulator/bus protection
RC	reaction control
RCS	Reaction Control System
REMS	Rover Environmental Measurement System
REP	radioisotope electric propulsion
RF	radio frequency
RFI	radio frequency interference
RHU	radioisotope heater unit
ROLAND	Rosetta Lander-Magnetometer
ROMAP	Rosetta Lander Magnetometer and Plasma Monitor
RPS	Radioisotope Power System
RSEN	Reduced State Encounter Navigation
RSLP	U.S. Air Force Rocket System Launch Program
RTG	radioisotope thermoelectric generators
RTOS	Real-Time Operating System
S/C	spacecraft
SA	solar array
SADA	solar array drive assembly
SBIR	Small Business Innovative Research
SCA	spring cartridge assemblies
SDI	serial digital interface
SDO	serial data output
SEAKR	SEAKR Engineering, Inc.
SEP	solar electric propulsion
SEPTOP	Solar Electric Propulsion Trajectory Optimization Program
SEU	single event upset
SGI	square grid interface
SLOC	source lines of code
SLV	space launch vehicle
SM	service module
SMA	semimajor axis
SMD	NASA’s Science Mission Directorate
SN	signal-to-noise
SNOE	Student Nitric Oxide Explorer
SOAP	Satellite Orbit Analysis Program
SOFI	spray-on foam insulation
SPACE	System Power Analysis for Capability Evaluation
SPL	secondary payloads
SPM	lander plasma monitor
SPU	solar power unit
SRB	solid rocket boosters
SRC	sample return capsule
SRD	System Requirements Document
SRM	solid rocket motor

SSETI	Student Space Exploration and Technology Initiative
SSRD	split spool retention device
SSTO	single-stage-to-orbit
STP	Space Test Program
SUA	systems uncertainty analysis
TBD	to be determined
TBR	to be resolved
TCS	Thermal Control System
TDRSS	Tracking and Data Relay Satellite System
THEMIS	Thermal Emission Imaging System
TLI	trans-lunar injection
TMI	trans-Mars injection
TOF	time of flight
TPA	turbine pump assemblies
TPV	thermo-photovoltaic
TRL	technology readiness level
TSTO	two-stage-to-orbit
TT&C	telemetry, tracking and command
TVC	thrust vector control
TWTA	traveling wave tube amplifier
UHF	ultra high frequency
ULA	United Launch Alliance
UPC	unpressurized cargo
US	upper stage
USO	ultra-stable oscillator
UVS	ultraviolet sensor
WAC	wide angle camera
WAN	wide area network
WBS	work breakdown structure
WGA	weight growth allowance
WGS	weight growth schedule
WLAN	wireless local area network
WSB	weak stability boundary

Appendix B.—Study Participants

MASER Design Session			
Subsystem	**Name**	**Affiliation**	**Contact email**
Design Customer POC/PI	June Zakrajsek	GRC	june.f.zakrajsek@nasa.gov
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Appendix C.—Additional Design Images

Figure C.1 to Figure C.3 provide additional views of the Mars Hard Lander components, both external and internal.

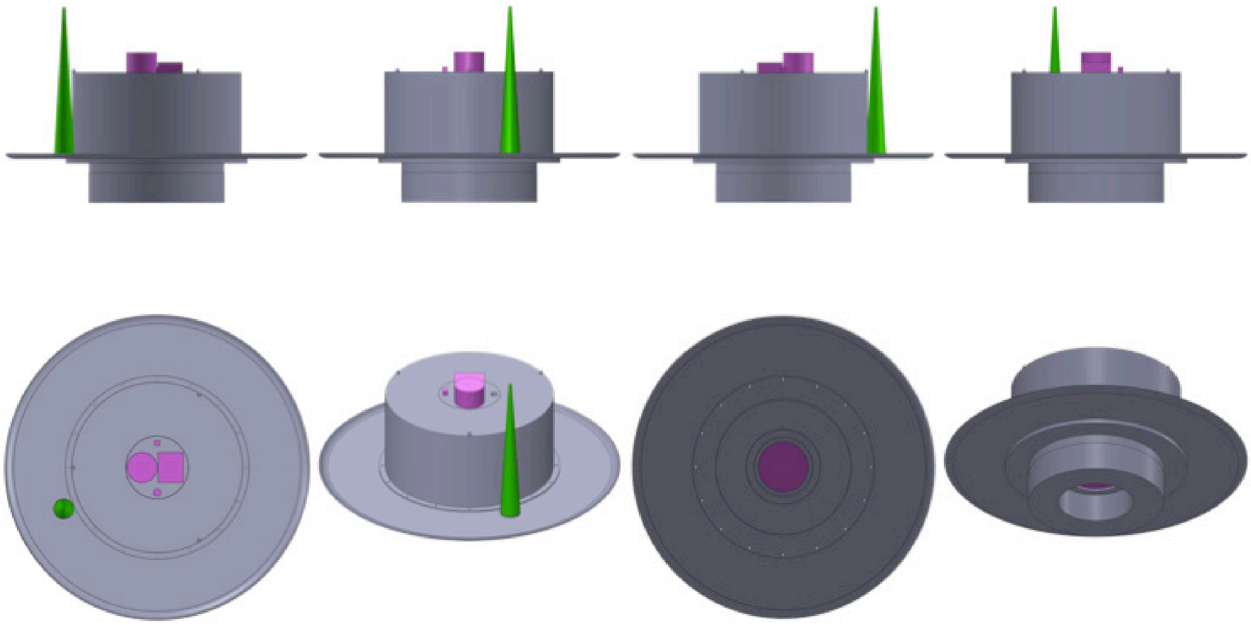

Figure C.1.—Various views of the Mars Hard Lander design.

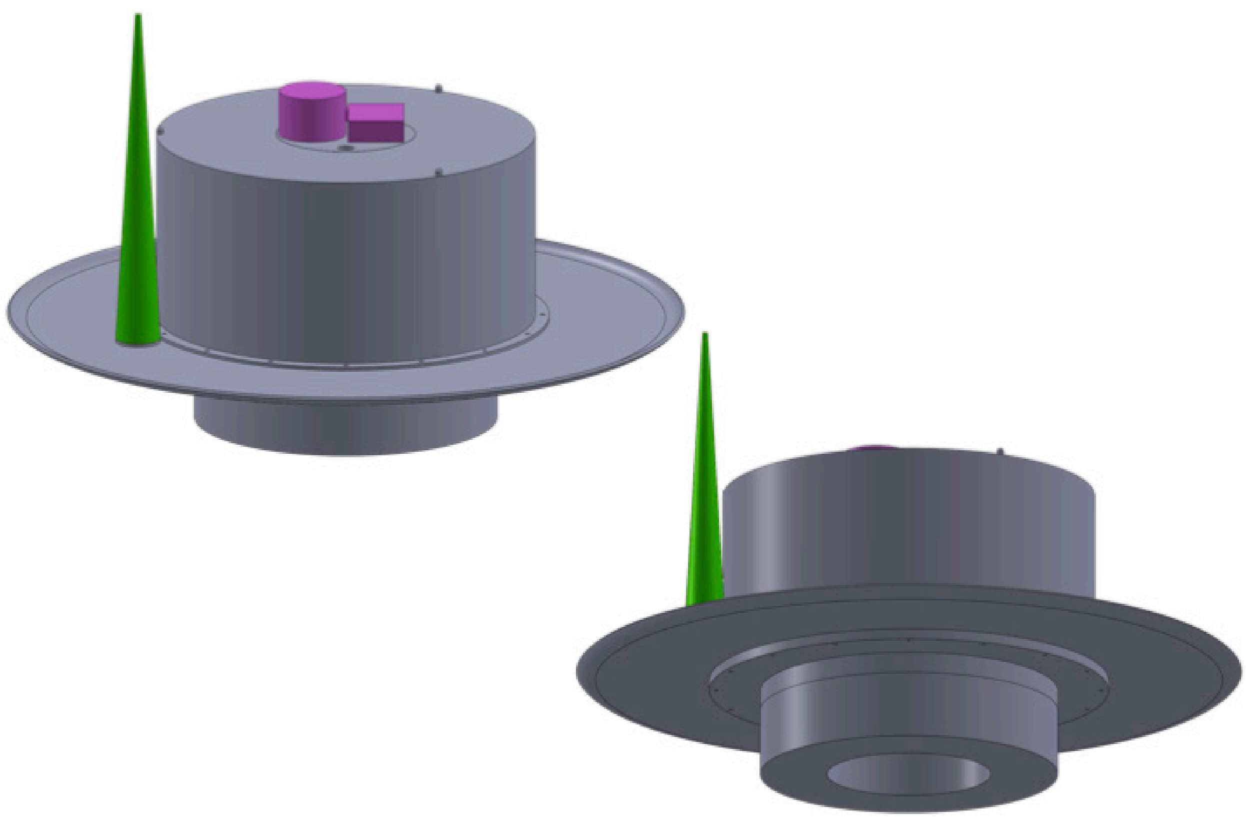

Figure C.2.—Isometric views of the Mars Hard Lander design.

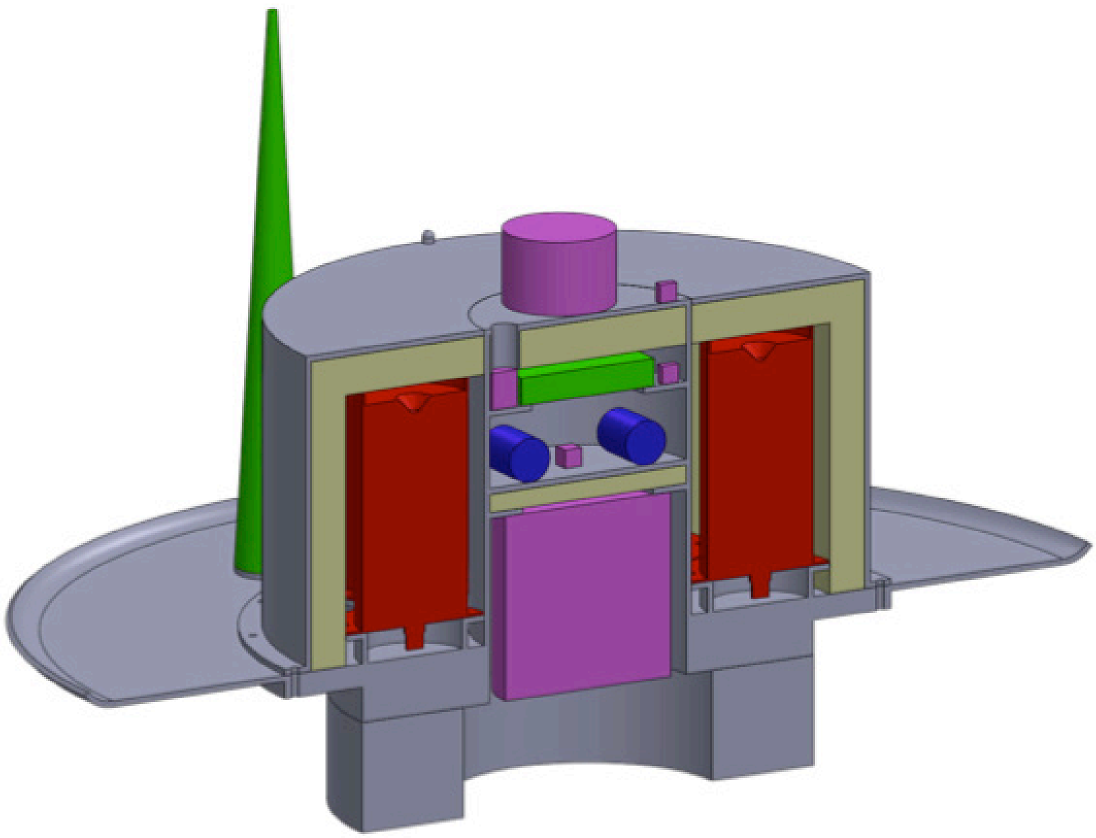

Figure C.3.—Cutaway view of the Mars Hard Lander design.

Appendix D.—Compass Internal Details

D.1 Compass Description

The Compass team is a collaborative engineering team whose primary purpose is to perform integrated-vehicle systems analysis and provide trades and designs for both Exploration and Space Science Missions.

D.2 GLIDE Study Share

GLobal Integrated Design Environment (GLIDE) is a data collaboration tool that enables secure transfer of data between a virtually unlimited number of sites from anywhere in the world. GLIDE is the primary tool used by the Compass design team to pass data real between subsystem leads in real-time.

While GLIDE 2 was being tested during this design session, the old shares are being used to store the data and the MELs. The data on the share can be found here:

https://glidesharename/2013/Mars_Hard_Lander

D.2.1 GLIDE Architecture

For this study, the Compass Team is testing the GLIDE 2 application and server. The architecture and database information will be referencing the GLIDE 2 server.

Architecture: **Mars_Hard_Lander**

D.2.2 GLIDE Study Container

Table D.1 lists the study container and descriptions of the cases run with the GLIDE-specific data necessary for the Compass Team members to conduct the study.

TABLE D.1.—STUDY CONTAINER AND DESCRIPTIONS

Study name	Description	Study container
Case no.1	Design of a lander to be delivered to Mars, via Aeroshell EDL system.	MHL_Case1

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