

Design Study for a Mars Geyser Hopper

Geoffrey A. Landis¹, Steven J. Oleson², and Melissa McGuire³
NASA Glenn Research Center, Cleveland, Ohio, 44135

The Mars Geyser Hopper is a design reference missions (DRMs) for a Discovery-class spacecraft using Advanced Stirling Radioisotope Generator (ASRG) power source. The Geyser Hopper is a mission concept that will investigate the springtime carbon-dioxide geysers found in regions around the south pole of Mars. The Geyser Hopper design uses Phoenix heritage systems and approach, but uses a single ASRG as the power source, rather than twin solar arrays, and is designed to last over a one-year stay on the South Pole. The spacecraft will land at a target landing area near the south pole of Mars, and have the ability to "hop" after a summertime landing to reposition itself close to a geyser site, and wait through the winter until the first sunlight of spring to witness first-hand the geyser phenomenon.

I. Introduction

A. The Design Reference Mission Process

The NASA "Discovery" class mission is a series of small, cost-capped NASA missions that are led by scientists, and designed to accomplish specific, well-defined science objectives. Since the first Discovery missions, the Near Earth Asteroid Rendezvous / Shoemaker probe in 1995 and the Mars Pathfinder in 1996, Discovery has been a highly successful program of innovative, science-driven spacecraft that have investigated targets as diverse as Mars, the moon, Mercury, comets, asteroids, and extrasolar planet detection.

The NASA Discovery Mission site states¹:

NASA's Discovery Program gives scientists the opportunity to dig deep into their imaginations and find innovative ways to unlock the mysteries of the solar system. It represents a breakthrough in the way NASA explores space, with lower-cost, highly focused planetary science investigations designed to enhance our understanding of the solar system.

NASA announced that missions to be proposed in response to the twelfth NASA Discovery mission solicitation in 2010 would be allowed to use the Advanced Stirling Radioisotope Generator (ASRG) power source^{2,3}, which would be provided as government-furnished equipment for the mission. The purpose of this was twofold: first, to use a Discovery mission as a demonstration flight of the ASRG, a power system design which has the possibility of producing a significantly higher amount of power from a smaller amount of radioisotope than early radioisotope thermoelectric generator (RTG) power systems; and secondly, to enable the low-cost Discovery class missions to access to a wide variety of new targets, including targets in the outer solar system, for which use of a radioisotope power system instead of solar arrays would be enabling.

B. Mars Geyser Design Study

The Mars Geyser Hopper is a design reference mission (DRM) for a Discovery-class spacecraft using Advanced Stirling Radioisotope Generator (ASRG) power source. The purpose of the Concurrent Multidisciplinary Preliminary Assessment of Space Systems (COMPASS) team DRM study is to provide design requirements for the integration of the ASRG into a mission and understand issues involved in use of the ASRG in the context of a mission design, and to explore the impacts of the use of an ASRG on a mission and the impacts of mission requirements on the ASRG⁴.

After evaluation of the options, the COMPASS team elected to pursue a design of a Mars Geyser Hopper mission. This mission would land a spacecraft, which was capable of hopping at least twice from its landed location on the Martian South Pole. The science on-board would be capable of studying the geysers that appear in the region of the South Pole of Mars in the Martian spring when the temperatures raise, over a stay of nearly one Martian year on the South Pole.

¹ Senior Research Engineer, Power and In-Space Propulsion Division, MS 301-3, AIAA Associate Fellow.

² Lead Engineer, Systems Analysis Branch, MS 105-3, AIAA Senior Member.

³ Senior Engineer, Systems Analysis Branch, MS 105-3, AIAA Member.

The mission constraints for this design were that the mission is to meet the Discovery-mission life-cycle cost cap of no more than \$425M (in FY-2010 dollars), not including the launch cost, and have a launch date no later than December 31, 2016. In order to reduce the cost and minimize risk, the spacecraft concept was based on a previous spacecraft design, the Mars “Phoenix” lander⁵, which has a demonstrated flight heritage. We make use of the fact that the Phoenix lander design incorporates soft landing capability and incorporates a restartable rocket propulsion system⁶, suitable to be repurposed for our requirements.

II. Mission Concept

A. Mars Geysers

Images taken by the Mars Global Surveyor spacecraft in the region of the seasonal south polar cap of Mars showed interesting dendritic features, called “dark dune spots” or “spiders.” These features occur on a region of Mars near the south polar, but not on the permanent polar cap, where the ground is covered with the seasonal polar cap of carbon-dioxide ice during the winter, but clear of ice during the summer.

The origin of these features was originally mysterious; however, analysis of many thousands of these features shows that the features are apparently the trace left on the ground of carbon-dioxide geysers^{7,8}. During the polar winter, the regions in which these geysers form is glazed with a layer of carbon dioxide ice, which can be one meter or more in thickness. The carbon dioxide ice is nearly transparent to sunlight, and hence when spring arrives in the southern hemisphere, and sunlight again illuminates the surface, the ice allows the incident solar energy to be transmitted the surface below. The resultant heating of the surface below the ice causes the carbon dioxide to sublime from the bottom surface of the ice, where it is trapped by the overlying ice sheet, building up pressure. When sufficient pressure builds up, the ice sheet bursts, resulting in a brief “geyser” as the carbon dioxide trapped below the ice is released. Carbon dioxide flows beneath the ice to the rupture location at speeds of up to 160 km/hr. This flow will entrain soil, basaltic sand, and other debris, which is blown into the atmosphere along with the escaping gas. This plume of debris (primarily dark basaltic sand) ejected into the air falls back onto the surface creates the dark fan pattern (figure 1), while erosion due to this flow carves radial patterns (“spiders”) in the surface (figure 2). Repeated imaging of the same area shows that the debris spots form in a period of weeks or less⁹.

This polar geyser phenomenon is unique to Mars. While seasonal polar ice caps form on Earth as well as Mars, polar deposits in the Earth’s high latitude regions are composed of water ice, which into liquid upon being heated, rather than sublimating into a gas, and hence does not build up pressure. As a phenomenon that is unseen elsewhere in the solar system, the Martian polar geysers are well worthy of further investigation.

Yet investigation of these features will be extremely challenging. The typical scale of one of the spider features is about 500 meters, requiring a high accuracy in placement to investigate the debris pattern and channel.

The geyser phenomenon occurs following an extended period of complete darkness, and the geysers themselves occur at the beginning of polar spring, when temperatures are in the range of -150°C, and the sun angle is only a few degrees above the horizon. The extreme environment, low sun angles during the geyser occurrence, and the fact that it would be desirable to emplace the probe well before the occurrence of the geysers, during a period of no sunlight, makes this a difficult environment for the use of solar arrays as the primary power source.

Thus, this is an attractive mission for use of the radioisotope Stirling power source.

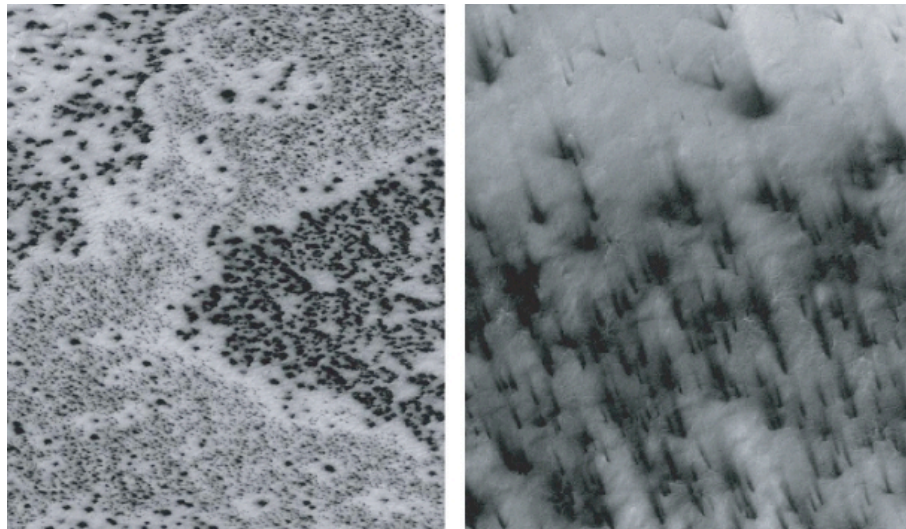


Figure 1: Orbital view of geyser regions in the south polar area of Mars, showing dark spots (left) and debris fans (right) of material ejected from the geysers onto the light polar cap. Image credit: NASA/JPL/Malin Space Science Systems.

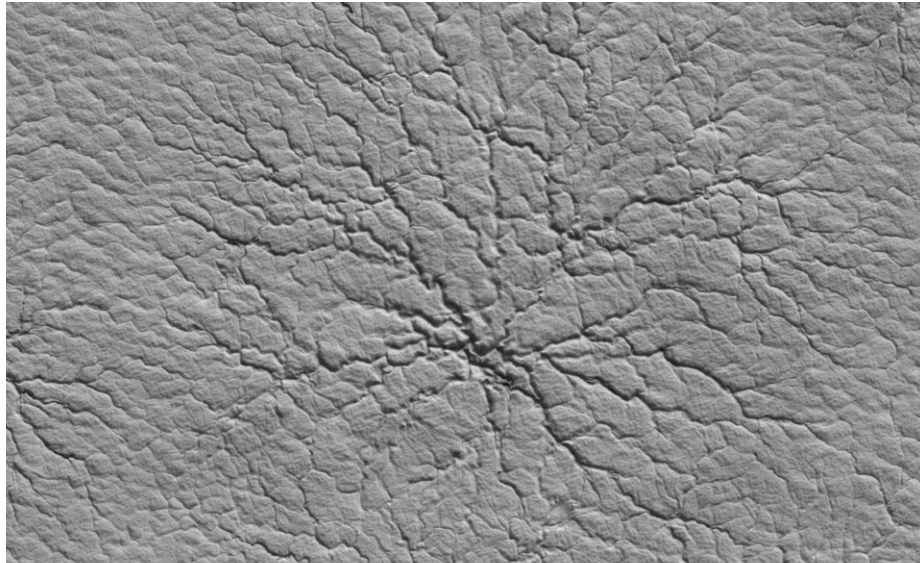


Figure 2: A close-up view of a geysers-carved “spider” pattern at Martian latitude 86.4°S, as viewed by the Mars Reconnaissance Orbiter “HiRise” camera, showing channels carved into the surface after the seasonal ice has retreated. This feature is about 500 meters across. [Image from Hansen and McEwan, Reference 9] http://www.uahirise.org/images/2007/details/cut/PSP_005579_0935_cut_b.jpg

B. Spacecraft

The hopper mission concept is shown in figure 3. The Geyser Hopper design builds on the Mars Phoenix Lander as a heritage system^{5,10}, but uses a single ASRG^{2,3} as the power source, rather than the twin solar arrays used by Phoenix. The ASRG is currently undergoing space qualification and lifetime testing at NASA Glenn¹¹. The spacecraft is shown in Figure 4 in its landed configuration. Figure 5 shows the lander stowed into the aeroshell with the interplanetary cruise stage attached.

The science instruments include stereo cameras to view the geysers events and a robotic arm (from Phoenix⁵) to dig beneath the soil surface and gather soil samples for chemical analysis on the Hopper. A light detection and ranging instrument (LIDAR), a landing camera and a thermal spectrometer (for remote geological analysis as well as weather sensing) are included. The target-landing site is on the South Pole, a region where geysers exist over a stretch of several hundred kilometers with densities of at least a geysers every 1 to 2 km.

Design details of the spacecraft, along with a discussion of the mission criteria and the DRM design process, are found in reference [4].

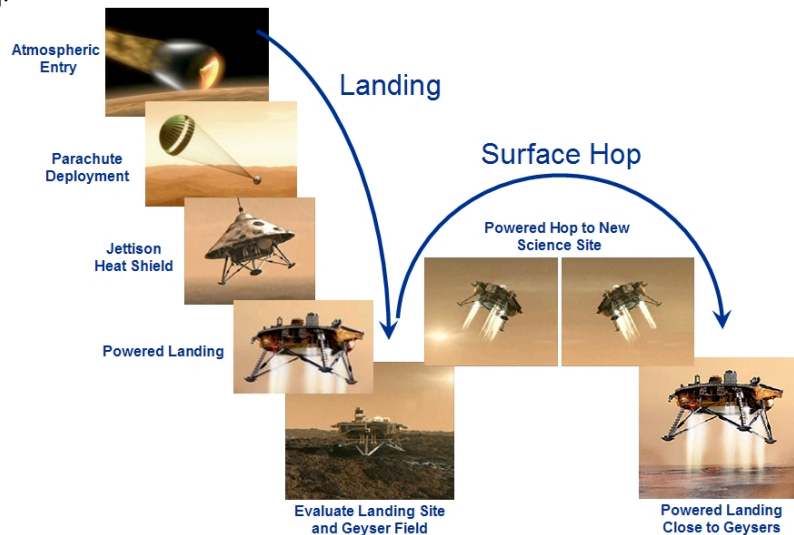


Figure 3: Mission concept summary.

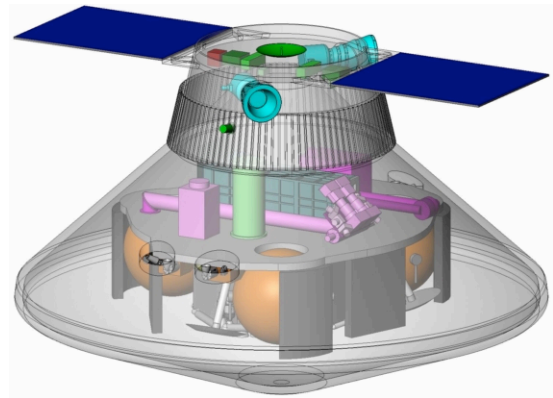
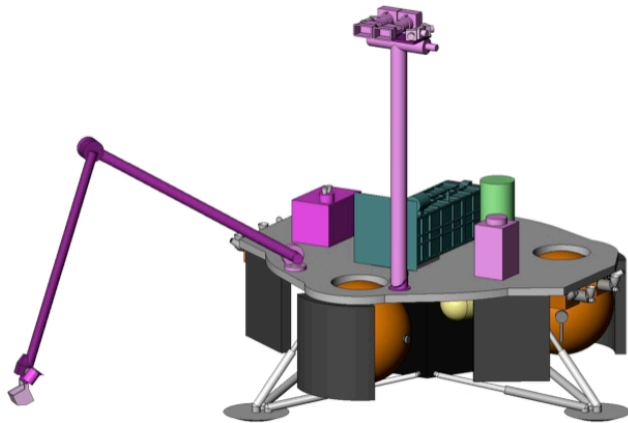


Figure 4 (left): Mars Geyser hopper shown as deployed on Martian surface, with sampling arm extended.

Figure 5 (right): Mars Geyser hopper shown inside the entry aeroshell (bottom) with the interplanetary cruise stage (top) attached (bioshield and insulation not shown).

III. Mission

A. Mission Overview

The Mars Geyser Hopper is a mission concept that will make a detailed investigation of one of the thousands of springtime geysers around the south pole of Mars.

The mission concept is:

1. The spacecraft will enter the atmosphere, and make a rocket-powered soft landing in a region of the South Pole where geysers are known to form. This landing will take place during the polar summer, when the surface is free of ice. The predicted landing ellipse is 20- by 50-km around the nominal landing site, and hence the landing will be targeted to a region, and not to a specific geyser location.

2. During the first landed science phase, the lander will conduct science operations to characterize the landing site, to understand the surface geology of the area during the ice-free summer period. Images taken during descent, as well as surface images, orbital images, and radio-signal tracking, will identify the landing location precisely. From this location, orbital images will be used to identify a nearby location where the signature of a geyser can be found.

3. The spacecraft will stow its science instruments and re-ignite the engines for a first hop of a distance of up to 2 km. This hop is designed to place the lander in a location where it can directly probe the geyser region, examining the surface at a spot where a geyser had been. The flight also will move the spacecraft closer to the region for the winter-over site.

4. During the second landed science phase, the lander will conduct science operations to characterize a geyser site during the summer period. Images taken during the flight will identify the landing location to within meter accuracy, allowing very precise targeting of the next hop.

5. The spacecraft will stow its science instruments and re-ignite the engines for a second hop, a distance of ~100 meters. This hop will place the lander onto the winter-over site, a spot chosen to be a relatively high elevation where the geyser can get a good view of the surroundings, close to but not located on the site of a known geyser, and outside the fall-out pattern of the expected debris plume.

6. The spacecraft will characterize the local area during the remaining sunlight, and then go into “winter over” mode. Waste heat from the ASRG will insure that the lander itself will remain ice-free during the winter. The lander will continue to transmit engineering status data and meteorological reports during the winter, but will not conduct major science operations.

7. On the arrival of polar spring, the lander will observe the geyser phenomenon from the location selected for

optimum viewing.

8. Following the geyser-season observations, the prime mission is completed. Extended mission operations, if desired, would continue the observation through a full Martian year and into the second Martian summer.

B. Hop Details

A key element of the spacecraft is that it will have the ability to hop after a summertime landing to reposition itself close to a geyser site, and wait through the winter until the first sunlight of spring to witness first-hand the geyser phenomenon. The ability to “hop” after a propulsive soft landing was done once before, on the Surveyor-VI lander on the moon in 1967. The 300-kg landed mass of the Surveyor-VI is about 20% less than the mass of the lander assumed here.

The hop is done using the same engines that are used to accomplish the soft landing, similar to those used on the Phoenix lander⁶. Since the engines are pulsed hydrazine monopropellant engines, restart capability is already incorporated into the design. The spacecraft must be modified to ensure that the engines are preheated before the flight, and the thermal system is designed so that the hydrazine propellant is kept liquid at the approximately -75 °C ambient temperature of the polar summer. (However, note that the engines and fuel are not intended to operate after the Martian winter, where ambient temperature falls to -150°C or below; they will be used only during the much less stressful summer.)

The vehicle is assumed to be capable of performing up to two hops, the first hop to bring the Lander to the region of interest for characterization of the geyser fields during the summer, and a second smaller hop to take the Lander to the final “winter over” site, where it is in view of, but not directly over, a geyser site. In addition a “fine tune” hop to optimize the landing site may be done if fuel margins permit.

To obtain an estimate of how much propellant would be consumed for the two hops, each hop was modeled in the Mission Analysis and Simulation Tool in Fortran (MASTIF) program. The hops were modeled in 3 Degrees of Freedom (DOF) with open loop control. The burn times were adjusted until the required distance for each hop was achieved. The following is a description of how the hops were modeled:

- Two second vertical rise
- Thrust at a 35° angle relative to local vertical
- Ballistic coast
- Orient thrust vector to cancel horizontal velocity
- Vertical descent for soft landing
- A constant atmospheric density of 0.017314 kg/m³ was used in the analysis of the hops.

For each hop, an amount of propellant equal to the expected usage plus reserves is determined. Table 1 shows the results for the two hops that were modeled. These distances are the science minimum values. Since the hop propellant is stored in the same tank that was used for landing, the vehicle will also have available any unused fuel from the landing. The landing propellant requirements incorporate 30% reserves, and thus the actual hop distance, assuming a nominal landing, will be considerably greater.

At the landing, and after each hop, the engineering team will assess the amount of propellant left in the tanks to determine the allowable hop distance; thus, the propellant reserves from each preceding engine firing can be added to the allowable use, and thus if the mission is nominal, and does not require digging into the reserves, longer distances can be planned on later flights. Alternately, if fuel margin is high but longer distances are not required, a short final hop can be added into the plan, in order to “fine tune” the winter-over site to the optimal location.

	Hop 1	Hop 2
Distance, km	2.0	0.1
ΔV (m/sec)	248	60
Mass _{initial} , kg	500	452
Mass _{final} , kg	452	440
Propellant consumed, kg	48	12

Table 1—Propellant consumed for each hop

C. Spacecraft mission design criteria and science instrumentation

This mission is subject to some challenging requirements on the ASRG system, including planetary protection, dusty environments and use in a planetary-entry aeroshells.

The science instruments include stereo cameras to view the geyser events and an arm (from Phoenix⁵) to gather soil samples for analysis on the Hopper. A LIDAR, landing camera and thermal imager (for remote weather sensing) is included. The target-landing site is on the South Pole where fields of geysers exist across an extent of hundreds of

km with densities of at least a geyser every 1 to 2 km. Since the landing ellipse is assumed to be 20 by 50 km (assumed to be identical to the MSL error ellipse¹²) the hopper is equipped with an additional 80 kg of hydrazine propellant. This allows the landing engines to be reused after landing, to hop at least 2 km to be close to a geyser site for imaging in the spring. The hopping system reuses the Phoenix landing propulsion system¹⁰ and counts on orbital imaging spacecraft (which also provide for UHF relay of telemetry and science data) to accurately locate a geyser location within range of a rocket-propelled hop. Even though the hop will be made before the end of summer, sufficient heat (taken from the ASRG waste heat) will be needed to keep the hydrazine tanks and fuel lines warm due to the south pole's low ambient temperature.

Table 2 summarizes the main factors of the spacecraft design.

Subsystem area	Details	Total mass with growth
Top level system	Investigate some of the thousands of springtime geysers around the South Pole of Mars over a 1 Mars year stay on the South Pole. This mission will provide an example of an ASRG mission.	1092 kg (including 207 kg growth)
Mission, operations, guidance, navigation & control (GN&C)	Design based on Mars Phoenix, highly symmetric for minimal solar torques, three-axis stabilized ballistic entry at Mars, Sun sensors, Star Trackers (on cruise deck), IMU, Landing radar (on lander)	8 kg (cruise deck) + 17 kg (hopper)
Launch	Launch March 2016, (date set by Mars launch window and requirement to land during Mars southern summer)	
Launch Vehicle	Atlas V 401 direct to declination of 47.5° and $C_3 = 13 \text{ km}^2/\text{s}^2$	
Science	Geyser science: cameras, LIDAR, MastCAM, Landing science: Descent imager; microscopic imager; meteorology package, chemical analysis package, robotic trenching tool, drill.	34 kg
Power	Fixed solar array mounted to cruise deck to provide 150 W, One ASRG on Hopper for 133 W, Li-ion battery on Hopper.	6 kg (cruise deck), 126 kg (hopper)
Propulsion	Hopping propulsion based on Phoenix landing system. Integrated hydrazine monopropellant blow-down system; 15 Aerojet MR-107N thrusters with Isp = 230 sec for landing and hopping. RCS is four pairs of Aerojet MR-103D thrusters at 215 sec Isp, and one Aerojet MR-102 thruster at 220 sec Isp.	62 kg (dry) + 191 kg (propellant)
Structures and mechanisms	Lander: composite sandwich structure deck. Hexagonal tubular space frame, 2090-T3 Al allow.	166 kg (cruise deck) + 178 kg (aeroshell) + 58 kg (Hopper)
Communications	X-band direct to Earth (DTE) on cruise deck for transit comm., UHF antenna on Hopper to communicate through TBD Mars orbiting relay sat.	16 kg (cruise deck) + 7 kg (hopper)
Command & data handling (C&DH)	All located on Lander/hopper. Two Flight computers, two data acquisition units, 100 GB data storage.	29 kg
Thermal	RHU provides primary heat; aerogel foam, radiator with louvers, cold plates with heat pipe connections to radiator. Radiators sized for thermal on cruise deck and then for thermal on Hopper inside the entry aeroshell.	77 kg (Cruise deck) + 35 kg (aeroshell) + 18 kg (hopper)
Cost	Discovery Cost Cap: \$425M (does not include launcher, ASRGs are GFE)	\$350M

Table 2: Mission and Spacecraft Summary

Power requirement estimates show that a single ASRG is sufficient (assuming 30% growth on power requirements) given the same large Li-ion battery pack (from Phoenix) to provide peak powers for periodic high-power propulsion and science events. This single ASRG is placed on the top deck of the hopper with the majority of cooling provided by pumped loops which take the roughly 350W of thermal waste heat to keep the hopping propulsion system warm as well as the rest of the spacecraft. These loops terminate in vertically hung (to reduce dust degradation) radiators around the edges of the spacecraft. These radiators will also ensure that the hopper spacecraft will not be covered in the 1m or so depth of CO₂ ice from the atmosphere that 'snows' out of the atmosphere during the winter months.

Perhaps the most challenging part of the geyser hopper design is planetary protection. Since the hopper has two plutonium General-Purpose Heat Source units and lands on the South Pole, which has subsurface water (considered a "special region" in the planetary protection protocol), a risk may exist for a non-sterilized spacecraft to introduce life to Mars in the event of a failure. Several approaches to spacecraft sterilization were explored but it was decided that the entire hopper and aeroshell would need to be sterilized. Past experience with Viking provided for heating the entire aeroshell/lander combination (with two RTGs installed (but cooled) inside of a bioshield, which is not jettisoned until after launch. Due to heat limits on certain portions of the ASRG, in addition to other factors an approach to utilize vapor hydrogen peroxide (VHP) to sterilize the entire hopper/aeroshell and place it in a bioshield was created.

The launch mass of the hopper design fits easily on an Atlas 401 launcher. Given the launch margin it may be possible to launch a second spacecraft (*i.e.*, a secondary or “ride-along” spacecraft, funded by another program) below the Geyser Hopper. Initial cost estimates of the Geyser Hopper show a cost estimate of \$350M with 25% margin for everything except launch services, the ASRG and the planetary protection system. This is well under the Discovery mission cost cap.

IV. Mission Profile

A. Interplanetary Cruise

The Earth-to-Mars 2016 mission window was assumed. The Earth-Mars trajectory was optimized such that Earth Departure C3 as well as Mars arrival excess speed were minimized.

This chemical trajectory mission was optimized using Copernicus, a generalized trajectory design and optimization program. The Earth to Mars trajectory is modeled such that the spacecraft is injected by the Atlas booster directly into the Earth to Mars transit orbit on March 1, 2016, and arrives at Mars Entry Interface (EI) 260 days later, October 8, 2016. Entry Interface was defined such that the spacecraft would not exceed acceleration loads and aeroshell heating loads.

Earth Departure parameters:

- $C3 = 13 \text{ km}^2/\text{s}^2$
- $\text{Dec} = 47.5^\circ$

Launch performance was extrapolated from the NASA ELV Performance Estimation Curves to a C3 of $13 \text{ km}^2/\text{s}^2$ at the required declination. The Atlas V (401) prediction for a DLA of 1255 kg is conservative (due to the southerly launch azimuth) but the Mars Geyser Hopper total wet mass (1092 kg) still fits with margin of 163 kg (which translates to a 15% ELV margin).

A directed entry into the South Pole was examined in the design of the aero-entry maneuver at Mars. A Southern entry was selected instead of a Northern Entry because it allows for shorter range and steeper flight path angle at entry. Shorter ranges at steeper flight path angles are desirable in order to reduce the size of the landing zone error ellipse at the South Pole.

It was necessary to add a trajectory correction maneuver (TCM) of 61 m/s just prior to reaching entry interface in order to achieve an entry velocity of 6.25 km/s. Allowing a steeper flight path angle reduces the magnitude of this correction maneuver but also increases acceleration loads on the vehicle. This burn was modeled impulsively within Copernicus.

After interplanetary cruise, the spacecraft reaches Mars atmospheric entry interface at an altitude of 125 km above the planet’s surface. This altitude represents what is historically considered the start of the Mars Entry Descent and Landing (EDL) trajectory.

B. Entry, Descent and Landing (EDL)

After performing a burn to slow down the entry velocity, the aerobraking shell performs the atmospheric entry maneuver and slows down the descent of the Lander/Hopper.

The EDL section of the spacecraft trajectory allows the spacecraft to decelerate from orbital velocities to a state that allows the Geyser Hopper’s engines to land safely on the Martian surface. At atmospheric interface, the vehicle has an entry velocity of 6.25 km/s at a flight path angle of approximately -11° relative to the local horizontal. With these entry conditions, a MATLAB tool designed for basic EDL analysis was used to develop a trajectory for the Geyser Hopper EDL. This tool provides information for the hypersonic and supersonic regions of flight.

Mars entry parameters:

- Altitude = 125 km
- Entry velocity = 6.25 km/s
- Flight path angle = -11.0°
- Azimuth = 0.0°
- Downrange from Pole = 1090 km

Based on aerodynamics of the Phoenix entry body¹³, a constant coefficient of drag (CD) of 1.6 is assumed for the hypersonic and supersonic regions of flight. In reality, CD changes with Mach number but the approximation of constant CD was necessary due to the fidelity of the tool used. Comparisons have been completed using the Phoenix mass and entry conditions to determine the accuracy of this simplified simulation.

Three important parameters are determined from the entry trajectory: peak deceleration of 5.6 times (Earth) g; a peak heat rate $61.3 \text{ W}/\text{cm}^2$; and a total heat load of $4,886 \text{ J}/\text{cm}^2$. The peak deceleration is less than the peak deceleration of nine-Earth g seen by the Mars Phoenix¹³, allowing the currently designed EDL structures to be used.

However, both the peak heat rate and the total heat load are higher than those on the Phoenix trajectory, which had a peak heat rate of approximately 47 W/cm² and heat load of 2428 J/cm². The differences in these parameters are predominantly from the higher entry velocity of the hopper mission of 6.25 km/s compared to the 5.59 km/s Phoenix entry velocity. This velocity also necessitates the use of a less negative flight path angle to control peak deceleration loads. However, the lower flight path angle, and higher heating rates result in a longer trip through the atmosphere and a higher total heat load. A predicted landing ellipse of at least 20- by 50-km around the nominal landing site would be expected if the entry is identical to the error ellipse of the Phoenix EDL systems. However, the decrease made to entry flight path angle may increase landing ellipse length.

The large increase in peak heat rate is not expected to be a design issue since SLA-561V, the ablative material used for the Phoenix aeroshell¹³, has been used at heat rates upwards of 100 W/cm² in past missions landed on Mars. The higher total heat load will necessitate a thicker layer of ablative heat shield material.

For Mars entry, checks are performed to ensure that the trajectory passes through the acceptable region of parachute deployment conditions for the Disk-Gap-Band parachute currently used for all Martian EDL systems.

Following entry, the heat shield is jettisoned and the parachute deployment further slows down the Lander/Hopper on its descent. The parachute and backshell are then jettisoned, and the main propulsion system performs a powered landing with total ΔV of 248 m/s.

The mission timeline from launch through the beginning of landed operations is shown in Table 3:

Operations Time Line: launch, cruise, and EDL (For the timeline, E = Entry, and L = Landing)

<ul style="list-style-type: none"> ▪ Launch—March 1, 2016 (Phase E begins) ▪ Cruise <ul style="list-style-type: none"> – Perform any necessary midcourse Trajectory correction burn(s) – Entry burn; orient for aero descent ▪ EDL operations October 8, 2016 <ul style="list-style-type: none"> – Final EDL parameter update—E-3 hr; Entry state initialization—E-10 min – Cruise stage separation—E-7 min – Entry Turn Starts—E-6.5 min; Turn completed by E-5 min – Entry—E-0 s – Peak heating – Parachute deployment—E+221 s – Heat shield jettison—E+236 – Leg deployments—E+246 s – Radar activated—L-143 s – Lander Separation—L-42 s – Gravity turn start—L-39 s – Constant velocity start—L-18 s – Touchdown—L-0 s – Dust Settling—L+ 15 min – Reconfigure as required; deploy antenna and establish communications; transmit health status – Landed mission begins
--

Table 3: Mission timeline (launch through start of landed operations)

C. Science Operations Timeline

Once landed, the science instruments onboard evaluate the landing site, and prepare for the powered hopping maneuver to a new science-determined landing site. There is sufficient propellant onboard for a second, smaller hop maneuver in order to allow for science readings to be taken in a second location on the Martian South Pole once the first location has been investigated.

The primary mission duration, starting from launch, is 30 months, comprising 8 months of interplanetary cruise followed by a primary mission of 22 months (one Mars year) on the surface. The timeline of science operations is shown in table 4.

Immediately after landing, there are four Science Operations Phases: Initial Landing Site Operations terminated by the “hop” to the Summer site; Summer site Operations, including an optional “Summer site Fine-Tuning Hop” and terminated by a “hop” to the Winter-Over Site; the Winter-Over Site Operations, and the springtime Geyser Operations phase. Science Concept of Operations for each phase is described below.

Science Operations Time Line

<ul style="list-style-type: none">▪ Characterize initial site— October 8, 2016—107 day duration<ul style="list-style-type: none">– Phase-1 science; characterize landing site and select site for summer science operations▪ Hop 1—summer science site—January 23, 2017—60 day duration<ul style="list-style-type: none">– Hop 1 maneuver– Reconfigure as required; deploy antenna and establish communications; transmit health status– Observe local region 1; relay and archive data– Process data; choose next “fine tuned” science site (if necessary and fuel permitting)▪ Hop 1a—“fine tune” hop, if necessary and fuel permitting—February 8, 2017<ul style="list-style-type: none">– Perform Hop maneuver– Reconfigure as required; deploy antenna and establish communications; transmit health status– Observe local region 1a; relay and archive data– Process data; choose final science site▪ Final Hop—winter-over site—March 25, 2017—40 day duration<ul style="list-style-type: none">– Perform hop maneuver– Reconfigure as required; deploy antenna and establish communications; transmit health status– Observe local winter-over site; relay and archive data– Process data▪ South Pole Winter observations—May 6, 2017, 376 day duration<ul style="list-style-type: none">– Perform Meteorology and climate science– Relay data▪ Detection and Science Observations of Geysers—May 14, 2018—90 day duration<ul style="list-style-type: none">– Await the Spring thaw– Detect geysers– Observe and acquire and relay data– Primary mission completed—August 10, 2018– Send final data archive to Planetary Data System (PDS): <i>following end of mission</i>▪ Extended mission, (optional)—August 11, 2018—unknown duration

Table 4: Science Mission timeline

1 Landing—Initial Landing Site Operations

Once the vehicle lands, it performs self-diagnostics, relaying all health data to the Science Center. Vehicle systems are initialized, and articulation of landing arm is tested, then the arm is stowed into “hop” configuration to prepare for the first hop to the summer science site.

The spacecraft is oriented using on-board Inertial Measurement Unit (IMU) and GN&C, combined with imaging data from both the downward-looking camera and site images using the multi-spectral camera. The vehicle's initial landing site position is independently confirmed with orbiter images. Imaging (and all data relaying) is coordinated with Mars Orbiter operations team. Direct to Earth communication will be maintained, but this will be a lower data rate than the orbital UHF link.

For the science operations at the initial landing site, the landing site chemistry and geology is surveyed using the multispectral camera and the mini-Thermal Emission Spectrometer (TES), a scanning infrared spectrometer system previously flown on the MER mission. This will be the first landing ever on the pole of another planet, and the local geology of the region is expected to quite different from that of any earlier landing sites on Mars. Characterization of the initial landing site, to understand polar processes and geology on Mars, is a major scientific objective.

Meteorology and climate studies at this site will also be a primary scientific objective. Data from this site will fill in our knowledge of Martian atmospheric processes. Atmospheric measurements will be monitored with the mini-TES, allowing remote measurement of surface temperatures as well as altitude profiling of atmospheric temperature¹⁴. Barometric pressure is monitored, and wind speed is measured using the LIDAR. Atmospheric Argon composition is analyzed with the Alpha Particle X-Ray Spectrometer (APXS) instrument, allowing diagnostic measurement of the atmospheric mixing during the condensation of carbon dioxide at the pole.

Since the actual geyser locations and associated debris fans cover only a small fraction of the landing error ellipse, it is unlikely that the initial landing site will be at a location to characterize the landform at a geyser location. The hop to the summer site will be selected for this purpose. (Alternately, if the initial landing happens to be at a

location to characterize a geyser trace, the summer site could be selected to characterize a site that does not show the traces of geyser activity.)

The summer site is selected during this time, based on a trade-off of between distance (closer is better), low risk from obstacles (boulders, hills), and proximity to promising geyser fields. The summer site must also have good accessibility to a good candidate winter site. Planning includes a comparison of line-of-site images with images taken during descent and also existing images from orbiters. An optimal trajectory is chosen, and then the “hop” maneuver to the summer site is executed.

2. Summer Site Hop and operations

Once again, on landing the first task is to orient again, to know exactly where the landing site is, based on surface images from the site, as well as orbital images if available. The science team will decide if this is good summer site, or if there is a desire for a fine-tune hop. Is this best spot for summer science? How good is the science traded against risk and fuel and so on? If the observations reveal a superior site at a short distance away, a fine-tune hop is requested, the engineering team will evaluate the vehicle health as well as fuel margins and, if they are determined to be adequate, release any fuel margin reserved for this hop.

Science operations:

- Take surface images of landing site.
- Unstow the arm, dig, do chemical analysis of samples. Relay data; analyze, validate and archive data.
- Do optical spectroscopy with camera and infrared spectroscopy with mini-TES
- Do atmospheric science.

Based on the information from the surface as well as orbital information, the team will review winter site selection, with new information about fuel margin, and local images from summer site—also incorporating the considerations of safety; proximity to, but not on top of a geyser. The site preference will be for high ground, to get better line-of-view, allowing the spacecraft to see more geysers. With the final selection of site, the spacecraft will do the last hop.

<ul style="list-style-type: none"> • Phase 1: Initial landing site science (107 days) – Primary Science Goal: Science of polar region landforms. Characterize initial landing site with multispectral camera and mini-TES infrared spectrometry – Secondary Science: Atmospheric science – Engineering goal • Locate landing site in orbital images; select next landing site, prepare for first hop ▪ Phase 2: Summer science (60 d) – Science goal: Characterize geomorphology and geochemistry of geyser site during summer (nongeyser) season – Secondary Science: Polar stratigraphy (measure depth and layers of ice); atmospheric science – Engineering goal • Locate landing site and select winter-over (final) landing site, prepare for second ‘adjustment’ hop – Optional engineering goal • Decide if a “fine tuning” hop is needed to optimize landing site, and if so, prepare for and execute second hop ▪ Phase 3: Characterize final site (40 d) – Science goal: Characterize the winter-over landing site before sun is gone. – Secondary Science: Atmospheric science – Engineering goal” • Preparations for winter ▪ Phase 4: Winter-over science (1 yr) – Science goal: Atmospheric science (meteorology and climate science) during Martian winter – Engineering goal • Survival ▪ Phase 5: Geyser science (90 d) – Science goal: Observe geysers – Secondary Science: Atmospheric science – Engineering goal: • Maintain spacecraft health and data downlink

Table 5: Science observation goals for each phase of the landed mission.

3. Winter-Over Site

At the final site, we use local and orbital images to determine our precise location again, then do local observations; and characterize the site—photography, spectroscopy—before the sun sets below the local horizon for the long winter darkness. Chemical will be done analysis, if time permits, but this is secondary science, since the lander is not on a geyser field area. The spacecraft then prepares to winter as necessary.

As winter arrives, we transition to winter mode operations, where the lander will continuously do meteorological observations and relay data home to archive all winter. During this period the visibility of the Earth will be poor, and the data relayed through an orbiting spacecraft. Waste heat from the isotope power source will prevent ice accumulation on the spacecraft.

4. Geyser Observations

After the winter, the science observations of the geyser field begin again at the approach of the spring thaw. The expected area of geyser activity is continuously scanned with wide-angle camera. Automated geyser detection on-board the spacecraft will scan the environment; although the routine imagery will be buffered on the spacecraft, images will not be relayed to Earth until the spacecraft detects a geyser. This triggers high-speed, hi-resolution imagery, including LIDAR characterization of particle motion and infrared spectroscopy. All data is stored for relay to Earth. Simultaneously, the science instruments will do chemical analysis of any fallout particles spewed onto the surface of the lander. Geysers erupt at a rate of about one a day during peak springtime season. If more than one is detected simultaneously, the spacecraft algorithm will focus on the nearest or “best”.

The lander will continue this primary geyser science for a period of about 90 days. Tens of geyser observations are expected over the spring/summer season.

5. Primary mission ends.

Science Mission observations end at end of geyser season, and the data will be calibrated, validated, and archived at the Planetary Data System. This may optionally be followed by an extended mission to observe the polar region into a second Martian autumn.

Table 5 shows the science goals for each location.

V. Spacecraft and Margins

1. Spacecraft Cruise Configuration

The basic Geyser Hopper configuration was based on the Mars Phoenix Lander⁵ (which was itself based on the Mars-98 Surveyor Lander/Mars Polar Lander design), modified for mission-specific design differences. The use of a heritage design with previous flight experience reduces the risk. The launch and cruise configuration consists of the Mars Geyser Hopper stowed within the entry aeroshell with a cruise deck mounted on the outside of the aeroshell. The aeroshell design is identical in size to that used for the Mars Phoenix Lander, while the cruise deck structure is modified to account for the difference in the launch vehicles between the two missions, since the cruise deck provides the structure to mate directly to the payload adaptor.

The overall dimensions of the Mars Geyser Hopper in its stowed and cruise configurations are shown in Figure 6. The radiators required to reject the heat of the ASRG contained within the aeroshell and the heat of the avionics of the Lander/Hopper dominate the surface of the cruise deck at 1.43 m^2 , while the aeroshell diameter is only 2.65 m . Given these dimensions, the entire stack can easily fit within the 3.65 m envelope of the Atlas V 4-m LPF

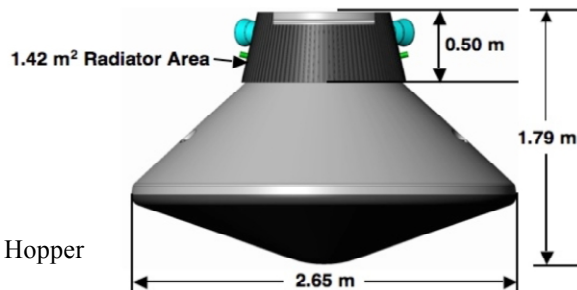
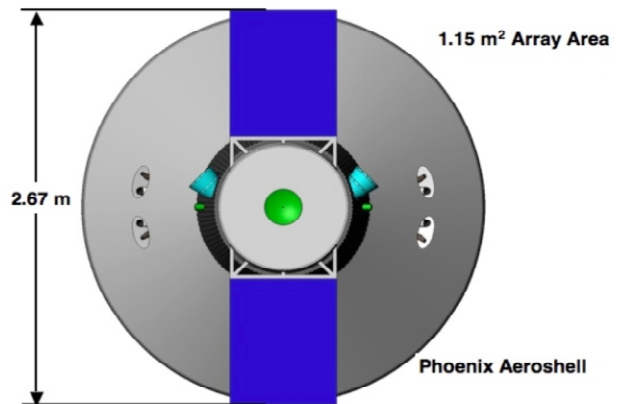


Figure 6: Dimensioned drawing of Mars Geyser Hopper interplanetary cruise stage and aeroshell.

(large payload fairing). The solar arrays are folded for launch. Figure 6 also shows the cutouts that are required on the backshell of the aeroshell to allow the attitude control and mid-course correction thrusters, which are integrated with the Lander/Hopper, to be fired during the cruise and descent as needed prior to the backshell being jettisoned.

Primary functions of the cruise deck include providing structure for mating the aeroshell to the Payload Adaptor, providing 1.15 m² of solar array area to produce the additional power required during cruise, providing the required communications during the cruise phase, providing the star trackers and Sun sensors the required view for guidance, navigation and control during the cruise stage, providing a location for the secondary battery used prior to starting up the ASRG, and providing the required 1.43 m² of radiator area for rejecting the waste heat from the ASRG and solar array and other components contained on the Lander/Hopper while enclosed in the aeroshell.

2. Spacecraft Landed Configuration

Figure 7 shows the spacecraft in its landed configuration. The deck stands 0.66 meters above the surface, while the stereo imaging system and science instrument mast extends to a maximum of 2 meters above the surface. The robotic digging arm (shown in the figure in an unstowed configuration) has the capability of extending out as far as 2.5 meters from the hopper's deck, and sufficient flexibility to examine the terrain under the lander, and inspect much of the bottom surface.

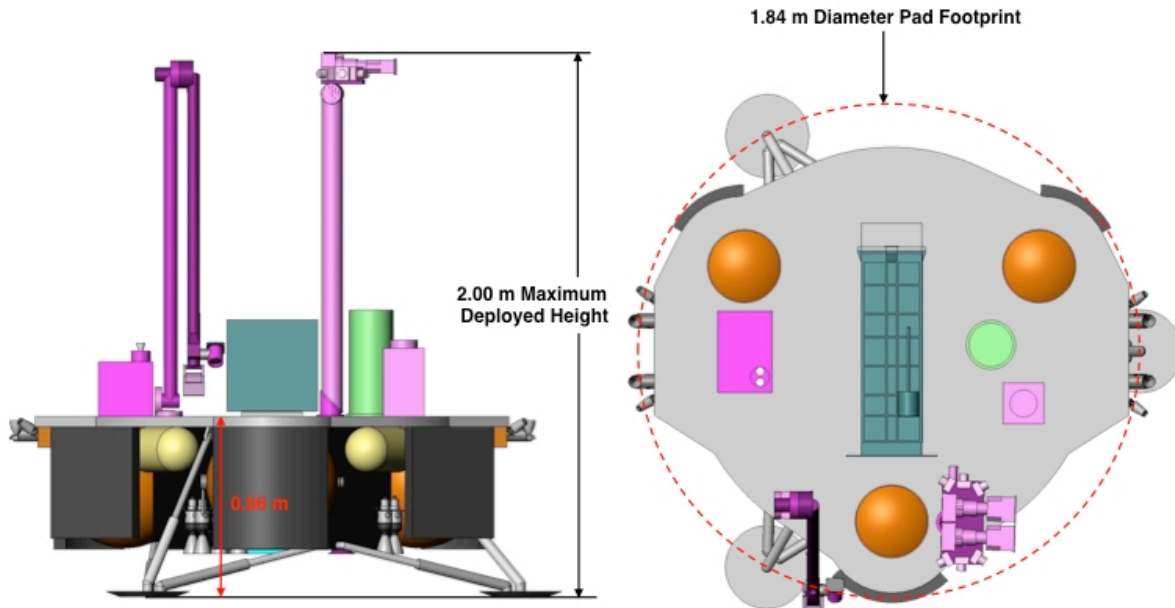


Figure 7: Height and footprint diameter of the Mars Geyser Hopper while deployed on the surface (insulation covering the ASRG and propellant tank not shown for clarity)

2. Mass and Power

1. Mass Growth and Margin

A significant part of the design exercise was to incorporate realistic growth, contingency, and margin in all systems. These used AIAA Standard AIAA S-120-2006, Standard Mass Properties Control for Space Systems¹⁵, in which the percent mass growth matrix is specified by level of design maturity as a function of the specific subsystem. Mass Growth Allowance (MGA) is defined as the predicted change to the basic mass of an item based on an assessment of the design maturity and fabrication status of the item, and an estimate of the in-scope design changes that may still occur.

The percent growth factors are applied to each subsystem, after which the total system growth of the design is calculated. The COMPASS design team standard operation is to design to a total growth of 30% or less. An additional growth is carried at the system level in order to add up to a total system growth of a maximal 30% limit on the dry mass of the system. Note that for designs requiring propellant, growth in propellant is either carried in the propellant calculation itself or in the ΔV used to calculate the propellant required to fly a mission, but not both. Since propellant (ΔV) growth is accounted separately, the convention for referencing basic mass and growth is applied to the basic dry mass, not to the total mass.

2. Master Equipment List (MEL)

Table 5 lists the top level of the Master Equipment List (wet mass) of the design with all the subsystem line elements hidden such that only the top-level masses are shown. The total growth on the dry mass of the spacecraft is then rolled up to find a total growth mass and growth percentage. The MEL incorporates the Current Best Estimate (CBE) mass of all of the items into totals and calculates a total CBE mass, a total spacecraft mass and a total growth mass.

Spacecraft Master Equipment List Rack-up (Mass) - RPS_DRM1					COMPASS S/C Design	
WBS	Main Subsystems	Basic Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)	
06	RPS DRM1 Spacecraft	884	142	1026		
06.1	Cruise Deck	231	41	272		
06.1.1	Attitude Determination and Control	6	1	8		20%
06.1.2	Command and Data Handling	0	0	0		TBD
06.1.3	Communications and Tracking	12	4	16		30%
06.1.4	Electrical Power Subsystem	5	1	6		25%
06.1.5	Thermal Control (Non-Propellant)	67	10	77		15%
06.1.6	Propulsion and Propellant Management	0	0	0		TBD
06.1.7	Propellant (Chemical)	0		0		TBD
06.1.8	Structures and Mechanisms	141	25	166		18%
06.2	AeroCapture Shell	181	32	213		
06.2.1	Thermal Control (Non-Propellant)	31	5	35		15%
06.2.2	Structures and Mechanisms	150	28	178		18%
06.3	RPS DRM 1 Hopper Main Spacecraft	473	68	541		
06.3.1	Science Payload	29	5	34		17%
06.3.2	Attitude Determination and Control	13	3	17		24%
06.3.3	Command and Data Handling	24	6	29		24%
06.3.4	Communications and Tracking	5	2	7		30%
06.3.5	Electrical Power Subsystem	95	32	126		33%
06.3.6	Thermal Control (Non-Propellant)	16	2	18		15%
06.3.7	Propulsion and Propellant Management	53	9	62		17%
06.3.8	Propellant	191		191		
06.3.9	Structures and Mechanisms	50	9	58		17%
	Estimated Spacecraft Dry Mass	694	142	836		20%
	Estimated Spacecraft Wet Mass	884	142	1026		
System Level Growth Calculations						Total Growth
	Dry Mass Desired System Level Growth	694	208	902		30%
	Additional Growth (carried at system level)		66			10%
	Total Wet Mass with Growth	884	208	1092		

Table 5: Master Equipment List, incorporating current best estimate (CBE), growth and margins

3. Power

The single ASRG located inside the aeroshell on the Lander vehicle is capable of producing 133 W of power³ for the surface operations. A battery is available onboard for use during Entry/Descent/Landing (EDL) as well as during the hops when there is a short duration requirement for additional power. During the cruise mode of the mission, when there will be continuous power requirements for the electronics systems (avionics, GN&C) as well as regular communications (at least one hour out of every day), a solar array on the cruise stage will supply approximately 150 W. This array will then be jettisoned with the cruise deck during EDL at Mars. The single 133-W ASRG on the Hopper is enough to handle the surface operations and science (123 W) and still have 10 W available as a trickle charge to the battery. This additional battery is included on the Hopper/Lander to supply the spike in power for the thrusters during EDL breaking for soft landing, and during the two hops.

The COMPASS team typically uses a 30% total growth on the bottoms-up power requirements in modeling the power system.

4. Thermal Waste Heat

Thermal waste heat generated by the components of the spacecraft (in all three elements: aeroshell, cruise deck and hopper/lander) was also calculated and used by the thermal subsystem designer in sizing the thermal subsystem elements. The thermal system has been sized to radiate the power in the three different stages based on the power

requirements to the vehicle when those stages are in use. For example, the cruise deck is operational only during Power Mode 1. During this mode, a radiator on the surface of the cruise stage is sized to radiate the waste heat generated by the ASRG as well as the electronics on the hopper and the cruise deck. Similarly, the radiators on the Hopper itself are sized to radiate the waste heat from the ASRG itself as well as the electronics.

The waste heat from the ASRG has a secondary use; it will be used to keep the hopper vehicle itself from being buried under the seasonal ice during the polar winter. Thermal calculations show that the radiated heat from the lander should be sufficient to keep the lander, and a portion of the surrounding terrain, warm enough to remain free of ice.

3. Communications

In the landing site at or near the South Pole of Mars, the vehicle can only see the Earth periodically. During the Martian winter there is no contact with Earth for $\sim 1/2$ of a Martian year. In the spring and fall of the Martian year there may be periodic viewing of the Earth each Martian day. During the summer of the Martian year the Earth will be in view continuously during the Martian day.

These long periods of no contact with Earth demands that, if Direct to Earth (DTE) communications is used, the Lander must have the ability to store critical information for transmission to Earth at a later time. This leads to an estimate the amount of critical data to be stored. Fortunately, the long time periods with minimal Earth communication contact correspond to time periods of minimal data activity. A total of 98.2 MB of storage is required for the winter observations; the 1 GB of storage assumed will be more than adequate for this time period. As the Earth comes up over the Martian horizon and the Martian spring initially approaches, the Earth is only visible for a few minutes or hours a day. It is assumed that the stored data will be downloaded over the days that have a total of 10 hr available for communication to Earth. The required data rate is 22 kbps. Over the course of the mission, the distance between the Earth and Mars varies from 0.37 to 2.7 A.U. These assumptions lead to the link calculation using some reasonable assumption on antenna size, frequency, and power. The link budget analysis leads to an acceptable transmit power and a reasonable size antenna only when the Earth and Mars are closest to each other.

During the main part of the mission, it is required that the lander observe the eruption of the geysers and then sending back science data and high-quality images. To assess the data needs, some assumptions must be made on the frequency and the duration of the eruptions. The desired image quality is HDTV quality, 30 frames/second, 1280 by 720 pixels, at 24 bits per pixel, for an assumed 30 second duration of eruption. (Lower quality images can hence be taken at a longer duration; *e.g.*, 60 seconds of imagery at 12 bits per pixel).

The data rate required for clips at HDTV resolution exceeds the available bandwidth for the direct-to-Earth transmission; there is no combination of antenna size, transmit power, or Mars to Earth distance that gives a reasonable power and antenna size to support even the lowest data rate of compressed HDTV. Even if new video codex will deliver HDTV in as little as 6 Mbps, the amount of power required to transmit directly to Earth is prohibitive.

Thus, the mission design assumes, in addition to the DTE communications link, use of a UHF orbital relay of telemetry to download the science data. Use of a UHF relay was the communications strategy used for the Mars Phoenix mission as well^{16,17}. Mars orbiters are typically placed into near-polar sun-synchronous orbits, and hence the availability of the line of sight to an orbiter from the south polar landing site is excellent, with multiple overflights from each orbiter every day. Spacecraft that are currently in orbit around Mars (*e.g.*, MRO¹⁷, Mars Express) incorporate the capability to relay data from surface landers. However, since the mission arrival date in 2016 is beyond the currently extended mission lifetime of existing Mars orbiters, this assumes either that additional extensions to their missions will continue to maintain these orbiters in operation as relays, or assumes future orbiters will have relay capability available.

4. Launch Date

The launch date was set by the combined requirements for a direct injection trajectory to access the south pole of Mars, the requirement for landing during the polar spring or summer, when the landing site is ice free and in sunlight, and the requirement of the Discovery DRM process that the mission must be launched before December 2016. Combining these requirements with the orbital mechanics and desire for minimizing both the launch energy (C3) and the arrival energy at Mars resulted in a launch window centered around March 1, 2016.

Orbital mechanics dictates that low-energy trajectories from Earth to Mars occur with a periodicity of approximately 780 days, the synodic period of Mars. The launch opportunities in 2018 and 2020 also allow landing during the southern polar summer, albeit with a somewhat shorter period of landed operations before the beginning of southern hemisphere autumn. However, while the flexibility of choosing a type-I trajectory or the longer type-II

trajectory allows some choice of arrival time and asymptote, not all Earth-Mars launch opportunities will allow arrival at the southern polar regions during the spring or summer.

Nevertheless, the spacecraft design will also have uses for Mars exploration launch windows that do not allow summer landing at the pole. Although the polar geyser mission was the particular mission concept analyzed, the hopper design is not limited to polar exploration, and could be used to explore many other locations on Mars. The ability to make multiple rocket-powered hops from an initial landing site to a science region of interest would give the ability to traverse far more rugged terrain than any previous missions, and could enable exploration of a large number of regions previously inaccessible to landers and rovers.^{18,19}

VI. Conclusions

A study for a design reference mission for a Discovery-class spacecraft, the “Mars Geyser Hopper,” was done, with the objective to clarify the issues involved in the integration of an Advanced Stirling Radioisotope Generator (ASRG) power source onto a mission with a planetary surface mission.

The Geyser Hopper mission would land a spacecraft capable of making at least two “hops,” of distance 2 km and 100 meters, after landing on the Martian South Pole. The lander would characterize the landforms at the geyser region of Mars, and then, after a 20-month winter-over stay on the South Pole, the science instrument would be capable of studying one of the most fascinating phenomena in the polar regions of Mars, the geysers that appear near the South Pole in the Martian spring when the temperatures raise. The use of the ASRG power source is an enabling technology for this mission, allowing an adequate power supply despite low sun angles in the polar regions at the times of interest, and a period of no sunlight over the nearly year-long polar winter. In addition, the provision of waste heat rejected by the thermal radiators on the power system allows the spacecraft to avoid being frozen under the seasonal polar ice.

In addition to being the first spacecraft to reach the pole of another planet, emulating on Mars the accomplishments of Amundsen and Scott on Earth, and conducting science operations to investigate and understand the polar landforms, the mission will investigate the south polar carbon-dioxide geysers, a phenomenon unique to Mars. High-definition color movies of the geysers in action will be one of the most dynamic planetary mission results ever, with an unrivaled public engagement factor.

The estimated cost of the mission is \$350M, not including the launch, ASRG, and planetary protection costs, which meets the constraints of the Discovery-mission life-cycle cost cap of no more than \$425M not including the GFE or launch cost. The estimated launch in March of 2016, with landing in October 2016, meets the Discovery requirement of a 2016 launch.

The hopper concept analyzed could also be used for exploration missions other than the polar geyser observation mission discussed here. The ability to make multiple rocket-powered hops from an initial landing location to a science region of interest would be valuable across a large range of terrain on Mars, as well as elsewhere in the solar system, and would demonstrate a new form of rover with the ability to traverse far more rugged terrain than any previous missions, a mission concept that would be applicable to exploration of many planets and moons.

Acknowledgments

We would like to acknowledge the contributions of the Concurrent Multidisciplinary Preliminary Assessment of Space Systems (COMPASS) team, most particularly Les Balkanyi, Laura Burke, Anthony Colozza, Jim Cockrell, Jon Drexler, Ian Dux, James Fittje, John Gyekenyesi, Peter Kascak, Terri McKay, Mike Martini, Tom Packard, Thomas Parkey, Carl Sandifer, Paul Schmitz, Anita Tenteris, Glenn L. Williams, Joe Warner, and Jeff Woytach.

References

1. NASA Discovery Missions; <http://discovery.nasa.gov/>
2. Sutliff, T. J., and Dudzinski, L. A., "NASA Radioisotope Power System Program – Technology and Flight Systems," paper AIAA 2009-4575, 7th International Energy Conversion Engineering Conference, Denver CO, August 2-5 2009.
3. Chan, J., Wood, J. G., and Schreiber, J. G., *Development of Advanced Stirling Radioisotope Generator for Space Exploration*, NASA/TM—2007-214806; *Space Technology and Applications International Forum (STAIF)*, Albuquerque, NM, February 11-15, 2007.
4. NASA Glenn COMPASS team (S. R. Oleson, team lead), *COMPASS Final Report: Radioisotope Power Systems, Design Reference Mission 1: Mars Geyser Hopper*, CD-2009-37, NASA Glenn Research Center (2009).
5. Goldstein, B., and Shotwell, R., “Phoenix – The First Mars Scout Mission (A Mid-Term Report),” paper IAC-06-A3.3.02, *International Astronautical Congress 2006*, Valencia, Spain, October 2 - 6, 2006.
6. McAllister, J. G., and Parish, C., “Phoenix Landing Propulsion System Performance,” paper AIAA 2009-5264, 45th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Denver, CO, August 2-5 2009.

7. Piqueux, S., Byrne, S., and Richardson, M. I., "Sublimation of Mars's Southern Seasonal CO₂ Ice Cap and the Formation of Spiders," *J. Geophysical Res.*, Vol. 108, No. E8, pp. 5084-5093, doi:10.1029/2002JE002007, 2003.
8. Kieffer, H. H., Christensen, P. R., and Titus, T. N., "CO₂ Jets Formed by Sublimation Beneath Translucent Slab Ice in Mars' Seasonal South Polar Cap," *Nature*, Vol. 442, pp. 793-796, Aug. 17, 2006.
9. Hansen, C. J., and McEwen, A., "Spring at the South Pole of Mars," *American Geophysical Union meeting 2007*, December 2007.
10. Prince, J. L., Desai, P. N., Queen, E. M., and Grover, M. R., "Entry, Descent, and Landing Operations Analysis for the Mars Phoenix Lander," *J. Spacecraft and Rockets*, Vol. 48, No. 5 pp. 778-783 (2011); doi: 10.2514/1.46563 1.
11. Wilson, S. D., and Oriti, S., "Advanced Stirling Converter Testing at NASA Glenn Research Center," paper AIAA 2010-6694, *8th Annual International Energy Conversion Engineering Conf.*, Nashville TN, July 25-28 2010.
12. NASA Jet Propulsion Laboratory (JPL), *MSL Landing Site Selection: User's Guide to Engineering Constraints, Version 4.5* (August 2007); http://webgis.wr.usgs.gov/msl/docs/MSL_Eng_User_Guide_v4.5.1.pdf
13. McDaniel, R. D., Wright, M. J., and Songer, J. T., "Aeroheating Predictions for Phoenix Entry Vehicle," paper AIAA 2008-1279, *46th AIAA Aerospace Sciences Meeting and Exhibit*, Reno, NV, January 7 - 10, 2008.
14. Smith, M., *et al.*, "One Martian Year of Atmospheric Observations Using MER Mini-TES," *Journal of Geophysical Research - Planets*, Vol. 111, No. E12, E12S13 (Dec. 2006).
15. AIAA S-120-2006, *AIAA Standard Mass Properties Control for Space Systems*; ANSI/AIAA R-020A-1999, *Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles*.
16. Lewicki, C. A., Krajewski, J. A., Ilott, P. A., and Dates J. A., "Phoenix Mars Scout UHF Relay-Only Operations," paper AIAA 2006-5980, *9th International Conference on Space Operations*, Rome, Italy, June 19-23, 2006.
17. Gladden, R. E., "Mars Reconnaissance Orbiter: The History of Supporting the Phoenix Lander," paper AIAA 2009-6750, *AIAA Space 2009 Conference & Exposition*, Pasadena, CA, September 14-17, 2009.
18. Landis, G. A., and Linne, D., "A Mars Rocket Vehicle With In-situ Propellant Production," *AIAA Journal of Spacecraft and Rockets*, Vol. 38, No. 5, Sept.-Oct. 2001, 730-735.
19. Linne, D., Barsi, S., Sjaun En Wa, W., and Landis, G., "Mars Surface Mobility Leading to Sustainable Exploration," *50th AIAA Aerospace Sciences Conference*, Nashville, TN, January 9-12, 2011.