A Design Proposal for Asaph-1: A Human Mission to Phobos

Jason B. Price[1] and Melissa M. Tanner[2]
California Institute of Technology, Pasadena, California, 91125, USA

Siddharth Krishnamoorthy[3]
Stanford University, Stanford, California, 94305, USA

Ashley R. Chadwick[4]
University of Adelaide, Adelaide, South Australia, 5005, Australia

Andrew R. Dahir[5], Abhijeet Kumar[6], and Ashley A. Williams[6]
University of Colorado, Boulder, Colorado, 80309, USA

Chantz P. Thomas[7] and Osazonamen J. Igbinosun[8]
University of Washington, Seattle, Washington, 98195, USA

Paul Nizenkov[9], Karin M. Schlottke[10], and Emil Nathanson[9]
Universität Stuttgart, Stuttgart, Baden-Württemberg, 70569, Germany

Sherrie A. Hall[11] and Sydney Do[12]
Massachusetts Institute of Technology, Cambridge, Massachusetts, 02139, USA

Jan Kolmas[13]
Yale University, New Haven, Connecticut, 06520, USA

Bakari N. Hassan[14]*
University of Michigan, Ann Arbor, Michigan, 48109, USA

The Asaph-1 mission was designed as part of the 2013 Caltech Space Challenge, a program that selected graduate and undergraduate students from 11 different countries to develop plans for a viable crewed mission to one of the two Martian moons, Phobos or Deimos. The program divided 32 participants into two teams of 16 for an intensive five-day workshop, with assistance from JPL and other NASA agency staff in Pasadena, California, USA. Asaph-1 was chosen as the favored mission proposal by senior members of NASA, JPL, SpaceX, Lockheed Martin, and the California Institute of Technology.

The goals of the Asaph-1 mission are to initiate human travel to the Martian system and to

[1] PhD. Candidate, Division of Geological and Planetary Sciences, MC 100-23
[2] PhD. Candidate, Mechanical and Civil Engineering, MC 138-78
[3] PhD. Candidate, Department of Aeronautics and Astronautics
[4] Postgraduate student, School of Mechanical Engineering
[5] PhD. Candidate, Department of Aerospace Engineering Sciences
[6] Graduate Student, Department of Aerospace Engineering Sciences
[7] Graduate Research Assistant, Department of Chemistry, Box 351700
[8] Graduate Research Assistant, William E. Boeing Department of Aeronautics and Astronautics, Box 352400
[9] PhD. Candidate, Institute of Space Systems
[10] Graduate Research Assistant, Institute for Aerospace Thermodynamics (ITLR)
[11] PhD. Candidate, Department of Aeronautics and Astronautics, Building 37-107
[12] PhD. Candidate, Department of Aeronautics and Astronautics, Building 33-409
[13] Bachelors of Science, 2014, Mechanical Engineering
[14] Bachelors of Science, 2013, Aerospace Engineering
*currently: MIT Lincoln Laboratory, Lexington, Massachusetts, 02420

gather scientific data from its two moons. Information on the composition, age and origin of the two moons will help to illuminate current theories concerning the formation of our solar system: in particular, how small bodies form and their relationship to their host planet and other celestial bodies. An investigation into the presence of natural resources on these moons may reveal potential for in-situ resource utilization, assisting in long-term habitation within the Martian system and the opportunity to increase the reach of human exploration within the solar system. With a transit time in excess of 400 days, this mission provides a unique opportunity to study the effects of deep space habitation on humans, which would undoubtedly inform and refine current procedures that seek to extend human presence well beyond Low Earth Orbit.

Asaph-1 is an opposition-class mission comprising three phases. First, a precursor mission will acquire useful remotely-sensed data from the moons, such as topography, chemical composition, and surface density characteristics, and will establish key infrastructure, especially a new communications satellite which will support all three stages of the mission. The primary mission will send humans to the surface of Phobos and return them to Earth along with geologic samples collected from Phobos and remotely-sensed data procured from Deimos. Finally, a legacy operation will continue after the crew departs, with equipment remaining on the surfaces of the two moons and in areocentric orbit, which will supply ongoing scientific data and will support future exploration of the Martian system.

In order to achieve such ambitious objectives, the Asaph-1 mission architecture utilizes currently available technology in an innovative configuration. In addition, a global perspective, encompassing national and international cooperation as well as a combination of government and commercial involvement, is requisite to provide the necessary resources to realize the mission within its ideal operational window, which opens in 2025. The precursor mission will be launched in 2026, and the crewed transit to the Martian system will begin in April, 2033. Launch vehicle modifications, advanced closed-loop life support systems, and new extra vehicular activity procedures are among the innovations featured in Asaph-1, by which futuristic goals meld with present technologies.

Nomenclature

ΔV	=	Delta-v, a measure of propellant use
I_{sp}	=	Specific impulse
CM	=	Crew Module
CMG	=	Control moment gyroscope
DCS	=	Decompression sickness
DE	=	Deimos Explorer
DSH	=	Deep space habitat
DSN	=	Deep Space Network
DTE	=	Direct-to-Earth
ECG	=	Electrocardiogram
ECLSS	=	Environmental control and life support system
EMU	=	Extravehicular mobility unit
EPS	=	Electrical power system
ESA	=	European Space Agency
EVA	=	Extra-vehicular activity
GCR	=	Galactic cosmic radiation
GLACIER	=	General Laboratory Active Cryogenic ISS Experiment Refrigerator
GOES	=	Geostationary Operational Environment Satellites
HEO	=	Highly elliptical orbit
HLV	=	Heavy-lift vehicle
HZE	=	High-Z high-energy ionizing radiation
ISPR	=	International standard payload racks
ISRU	=	In-situ resource utilization
ISS	=	International Space Station

JAXA	=	Japan Aerospace Exploration Agency
JPL	=	Jet Propulsion Laboratory
LEO	=	Low Earth Orbit
MELFI	=	Minus eighty-degree laboratory freezer for ISS
MIT	=	Massachusetts Institute of Technology
MMU	=	Manned Maneuvering Unit
MOI	=	Mars Orbit Insertion
MPCV	=	Multi-Purpose Crew Vehicle
MRO	=	Mars Reconnaissance Orbiter
MS	=	Mothership
MSL	=	Mars Science Laboratory
NASA	=	National Aeronautics and Space Administration
NEEMO	=	NASA Extreme Environment Mission Operations
NSSDC	=	National Space Science Data Center
NTP	=	Nuclear thermal propulsion
PDS	=	Phobos Deimos Surveyor
PE	=	Phobos Explorer
PLSS	=	Portable Life Support System
PROP1	=	Propulsion System One
PROP2	=	Propulsion System Two
PRSC	=	Planetary Retrieval of Subsurface Cores
REID	=	Radiation exposure induced death
SDO	=	Solar Dynamics Observatory
SEP	=	Solar-electric propulsion
SEV	=	Space Excursion Vehicle
SKG	=	Strategic knowledge gap
SLS	=	Space Launch Systems
SM	=	Service Module
SOHO	=	Solar and Heliospheric Observatory
SPE	=	Solar particle event
STK	=	*Systems Took Kit* (modeling software from Analytical Graphics, Inc.)
TCS	=	Thermal control system
TEPC	=	Tissue equivalent proportional counter
TRL	=	Technology Readiness Level
UHF	=	Ultra-high frequency
VPHPS	=	Vapor-phase hydrogen peroxide sterilization

I. Introduction

THE moons of Mars are an excellent option for human exploration prior to the exploration of Mars. The moons provide a test-bed for many essential technologies that are required for a manned mission to Mars, while removing some of the complex issues that also must be addressed, such as Martian atmospheric entry of very large payloads and the prevention of forward contamination. Further, the moons are a good place to investigate the potential for in-situ resource utilization (ISRU), which is an essential element for long-duration missions and possible colonization of Mars. Aside from these advantages, the moons also offer the unique opportunity to study asteroid-like small bodies in the solar system without having to undertake the risk of going into the asteroid belt itself. The study of small bodies will help in answering important questions about the formation of the solar system and the presence of life on other planets. A human mission to these moons will enable the performance of in-situ studies and also the return of samples to Earth, which can be analyzed with all the resources we have at hand without the constraints introduced by deep space operations.

Due to its larger size and interesting surface morphology, including the presence of numerous craters and at least one large monolith, we believe that exploring Phobos offers the greatest scientific returns for a given cost. Nevertheless, a concurrent study of Deimos' composition and structure via remote and/or robotic experimentation will provide vital information about the differences between the moons and may shed additional light on the formation of the moons.

Motivated by these points, the Asaph-1 mission (named for Asaph Hall, the discoverer of the Martian moons in 1877), a manned mission to Phobos, was proposed by this 16-member "Team Voyager" as part of the Caltech Space Challenge held March 25-29, 2013, at the California Institute of Technology, Pasadena, California, USA. The mandate from the senior scientists, engineers, and organizers to the students was to design a manned mission to one of the Martian moons with a launch date no later than January 1, 2041. During the workshop, Team Voyager divided into subsets of student-experts to address such considerations as science objectives, remote-sensing instrumentation, trajectory, propulsion, communications, habitation design, human health, sample return, biologic contamination, and risk. The group arrived at a consensus on key design items by, first, discussing their merits with scientists and engineers from JPL, NASA, Lockheed Martin, and SpaceX; second, voting on them as a group; and third, affixing them to our "wall of truth*." Once a design consideration reached the wall of truth, it became a permanent part of the mission plan. This paper is a summary of the results of our detailed mission plan[1], including: (1) scientific motivation for the mission, (2) a summary of the mission architecture, (3) first-order details of the mission, such as trajectory design, propulsion systems and habitat design, and (4) a brief discussion of the long term impact of such a mission. Owing to the condensed, intense nature of the workshop some contingencies and peculiarities of the mission, such as abort trajectories, alternative lower-ΔV trajectories, and multiple re-entry scenarios (i.e., aerobraking and aerocapture maneuvers), could not be evaluated.

II. Scientific Motivation for the Mission

The Asaph-1 mission is motivated by scientific discovery and demonstration of novel technologies, including those needed to support the extended duration of humans in space. Several outstanding physical and biological science questions that may be answered by the mission include:

(1) *What are the compositions, ages, and origins of Phobos and Deimos?*

Phobos is the larger, closer moon with approximate dimensions of 26.8 x 22.4 x 18.4 km. Deimos is the smaller, more distant moon with approximate dimensions of 15 x 12.2 x 10.4 km[2,3]. To date, only a limited amount of visible imagery and infrared spectroscopic data has been acquired to determine the compositions of either of the moons, which, at least at the surface, consist of phyllosilicates (serpentine and/or kaolinite) with lesser feldspars or feldspathoids[3,8]. The ages and origins of the moons are unknown. Both moons are very similar in composition to C- and D-type asteroids, which leads to the hypothesis that they are captured asteroids[4]. However, they both have nearly circular and equatorial orbits around Mars, which would necessitate an explanation for the circularization and adjustment of the inclination of their orbits after capture. Additional hypotheses for their origin(s) are: (1) They are remnant debris left over from the Martian accretionary process[5], (2) They are second generation solar system objects that coalesced in orbit after Mars formed[6], (3) They are two of many small bodies that were ejected from the Martian surface by collision with a large bolide[7], (4) They are captured cometary nuclei[2]. Measuring radiometric ages on the moons[9,10] will help to constrain the formational history of the moons and, by extension, Mars itself.

(2) *Are there any compounds--particularly water, hydrocarbons, or metals--on Phobos or Deimos that could be used for human habitation in space (e.g., to establish a station on one of the moons)?*

Current data suggest that there is no free water (ice) on the surface of the moons. To date, all 'water' observed by spectroscopy occurs in hydroxyl groups bound within phyllosilicate minerals. A temperature on the order of ~500°C is required to dehydroxylate phyllosilicates, thereby liberating free water[11,12]. Such a process may represent an engineering challenge but does not preclude the use of phyllosilicates as a source of water. The estimated densities of Phobos and Deimos are 1.87 and 1.54 g/cm^3, respectively. A back-of-the-envelope average of seven common phyllosilicate minerals yields a density of ~2.61 g/cm^3. Because the bulk density of the moons is significantly less than the average density of common phyllosilicate minerals, there must be a significant amount of lower density material present within the moons, i.e., various ices or potentially clathrate-like combinations of light hydrocarbons and water. If clathrates were to be found, they could prove useful for human habitation and transportation. The low bulk density of the moons argues against the presence of significant quantities of metals.

(3) *Are there any compounds that may indicate the presence of life?*

As yet, the answer to this question is unknown. Based on the criteria discussed by Clark et al.[13] for small bodies within 2 A.U. of the Sun, it is likely that the Martian moons are sterile. Nevertheless, samples returned from the moons should be carefully shielded from organic contaminants, as these samples may yield important data to help answer this question.

(4) *What are the surface characteristics of the Martian moons, especially with regard to landing a spacecraft*

*Regards to Nigel Angold for this idea.

there?

Images captured by MRO suggest that craters on both Phobos and Deimos are partly to completely filled with what appears to be powdery, fine-grained regolith. The craters of Deimos appear to be more filled with powder than those on Phobos. It is critical to know the depth and nature of the powdery regolith in order to make informed decisions about landing a spacecraft on either of the moons.

(5) *What physiological and psychological anomalies can be characterized using scans and samples from our crew during their incursion into deep space?*

Pre-, mid-, and post-mission analyses of crew health indicators will clarify the effects of radiation exposure, extended mission stress, and other, possibly unforeseen, factors on humans.

(6) *What will be the profile of radiation exposure encountered during the mission?*

Radiation data from the Mars Science Laboratory (MSL) cruise phase and the Asaph-1 precursor mission will better define the quantity and intensity of radiation that the crew must endure during the Asaph-1 manned mission to Phobos and, eventually, during the first human mission to the surface of Mars.

III. Mission Architecture

This section details the mission architecture intended for the Asaph-1 mission, including: the benefits of implementing a precursor mission for such a program, the over-arching mission structure, a general timeline to achieve the scientific and operational goals, and other important engineering considerations, such as technological considerations and strategic knowledge gaps.

A. Phase One: Precursor Mission, Motivation and Benefits

Just as the Surveyor program evaluated landing sites for the Apollo missions, a robotic precursor mission to Phobos and Deimos will reduce the risks involved in a manned mission by surveying potential landing sites and demonstrating technological feasibility. Phase One of the mission consists of an orbiting, remote-sensing Phobos-Deimos Surveyor (PDS), an impactor-lander Phobos Explorer (PE) and an identical impactor-lander Deimos Explorer (DE). The PDS-PE-DE system (Fig. 1) will launch from Earth in 2026 in a Falcon 9 and will use solar-electric propulsion to spiral out to Mars slowly over the course of two years. Upon reaching Mars in 2028, the PDS system will survey Phobos, its primary objective, and then Deimos and will deploy the PE and DE packages near their respective landing points. Having completed those missions, the PDS will remain in orbit around Mars to act as a communications relay for the Phase Two manned mission.

At Mars, the PDS-PE-DE system will enter an areocentric orbit below Phobos with an inclination of 20°. This orbit will cause the PDS-PE-DE system to gradually overtake Phobos, giving surveillance coverage of both the north and south pole regions. Then the surveyor will transition to an orbit above Phobos, which will allow for the mapping of over 80% of the moon's surface. From this higher vantage point, the PDS will release the PE, which contains an impactor experiment and lander. The impactor package will release four penetrometers to strike widely-spaced sites on each moon (Fig. 2). Using the results from the impactor experiment and the orbiting surveyor, the PE lander will reconnoiter the site most suitable for the landing of the manned mission, with sites

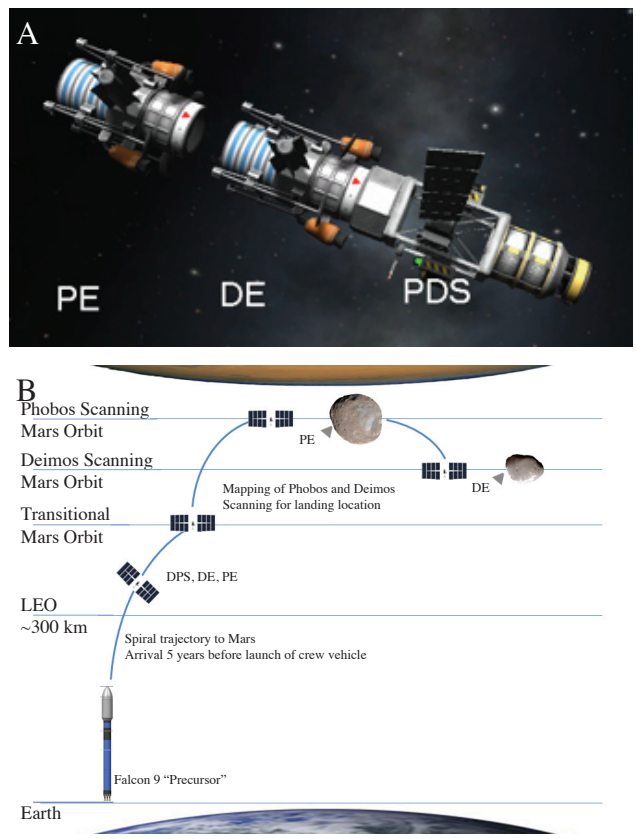


Figure 1. Elements of the robotic precursor (Phase One) mission. A) Initial exploratory craft will consist of a Phobos-Deimos Surveyor (PDS), which will carry the Phobos Explorer (PE) and Deimos Explorer (DE) packages (modified from the Kerbal Space Program, https://kerbalspaceprogram.com). B) Bat chart showing the Asaph-1 precursor mission plan, highlighting the robotic survey and communications relay setup.

'A' and 'B' being the priority sites.

The PDS system will then enter a higher-altitude Mars orbit, just below Deimos, and will release the DE. Again, this orbit will be slightly inclined from the ecliptic. The PDS will slowly move from below Deimos to trailing it, and then to a higher altitude orbit, thus mapping up to 50% of the moon's surface. As on Phobos, the DE will release four penetrometers, which will be viewed from this higher altitude. Immediate results from the impacts will determine the landing site for the DE lander. Once the impact experiments have been performed, the PDS will move back down to an areocentric orbit slightly below Phobos, thereby maintaining sufficient communications with both landers. Because both moons are tidally locked to Mars, all of the impactor sites on Deimos, and all but site 'B', within Stickney crater on Phobos, have full view of the Martian surface at all times[3]. Although the explorer packages will necessarily be highly autonomous, this will allow windows for the explorers to communicate information to the orbiting PDS system.

Although the movements to raise and lower the PDS system do complicate the precursor mission plan, the movement is required in order to survey both moons with the understanding that Phobos is the principal target of interest. Because Phobos is the priority, if the PE were to fail to initialize or if it yielded unsatisfactory results, the DE could be substituted for the faulty PE. If this were to be the case, the secondary raise to Deimos' orbit would be obviated.

Both landers will collect scientific data over the course of several years, until their power supplies run out. Meanwhile, the orbiting PDS will make remote sensing observations before, during, and probably after the Phase Two manned mission, and will also act as a key communications relay during Phase Two activities.

B. Phase Two: Primary Mission Overview

Phase Two, the manned mission, is planned to be an operation with a human crew in which surface operations, including sample collecting, will be conducted on a Martian moon, nominally Phobos, depending on favorable results from the Precursor mission. The crew is anticipated to return to Earth with geological samples and other data collected at the surface. A human crew was chosen to carry out sample collection and operations of this mission, as opposed to a teleoperated robotic system, because human astronauts on the ground are uniquely suited to make rapid decisions about geologic sample collection, and they possess a situational awareness necessary to meet mission goals at Phobos.

Phase Two is achieved using a sequence of launches from Earth to LEO, where the modules will rendezvous to form the mothership (MS). Once the assembly is complete, the MS will use an impulsive propulsion maneuver

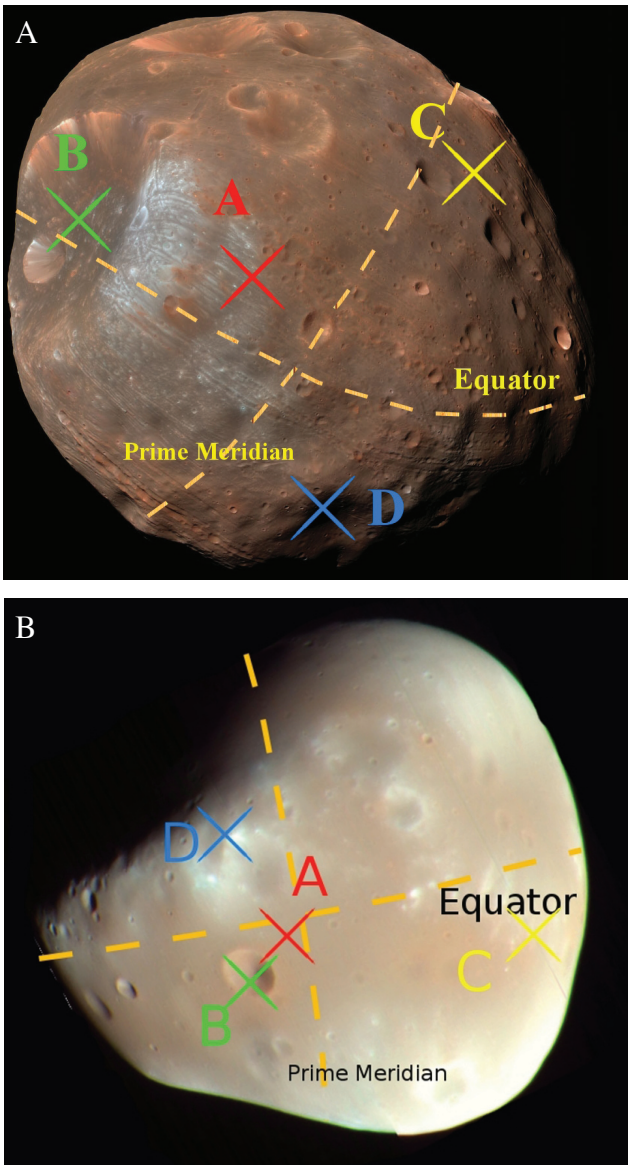


Figure 2. A) Penetrometer impact sites on Phobos. First priority Site A, on the Stickney highlands, has a mixture of red and white material, possibly dust and bedrock. Site B would allow investigation of the central uplift of Stickney crater. Sites C and D in the northern and southern hemispheres, respectively, appear to have deep fine-grained regolithic cover. B) Penetrometer impact sites on Deimos, all of which appear to contain thick regolithic material; such thickness may hinder a conventional landing on Deimos. From a geologic standpoint, the bright material along a mostly-buried crater at site 'D' is of greatest interest. The other three sites, ('A', 'B', 'C') were chosen to compare the depths of the powdery regolith in flat ('A', 'C') and cratered ('B') areas. Imagery from NASA/JPL/Univ. of Arizona HiRISE project (image PSP_007769_9010, Phobos; image ESP_012065_9000, Deimos).

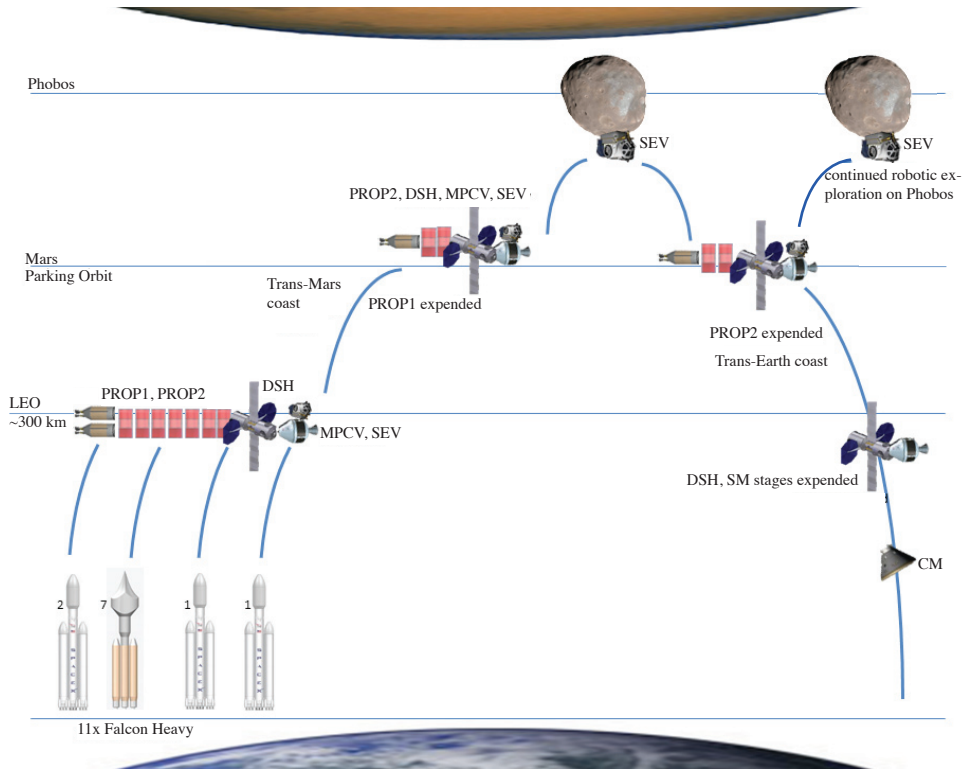


Figure 3. Bat chart showing principal components of the Phase Two (manned) mission.

to reach the Martian system within six months. At Mars, the MS will enter a parking orbit for approximately one month. During this period, the Space Exploration Vehicle (SEV) will approach the surface of Phobos to perform scientific activities. After returning the crew to the MS, the SEV will return again to the surface of Phobos as a probe to continue autonomous scientific operations over the course of several years. The rest of the MS will depart from Mars using another impulsive propulsion maneuver to return to Earth. A bat chart of the mission (Fig. 3) is provided for easier visualization of the primary mission.

The primary mission will utilize the following modules: (1) Propulsion Systems 1 and 2 (PROP1 and PROP2), which contains a nuclear thermal propulsion (NTP) system including liquid hydrogen tanks, (2) SEV, a vehicle that will bring astronauts from the MS to Phobos proximity and back, (3) Deep Space Habitat (DSH), a module that provides additional habitable volume for the crew. (4) Multi-purpose Command Vehicle (MPCV) with Orion Crew Module (CM), a vehicle that serves as the habitable volume shuttle from Earth to the MS, and will be used for the reentry of the crew.

C. Primary Mission Timeline and Considerations

The primary mission has a nominal duration of 465 days, including a 185-day-outbound transfer, a 30-day stay at Mars and a 250-day-inbound transfer. The crew will leave Earth's orbit in April, 2033, arrive at Mars during October of the same year, and return back to Earth in July, 2034.

The determination of the key dates and trajectories for the mission is based on multiple factors. The first trade-off is between undertaking a short-stay (opposition class) mission versus a long-stay (conjunction-class) mission. Considering crew safety issues due to radiation exposure in deep space and taking into account that a longer round-trip duration will lead to a higher probability of contingencies, we prefer the opposition-class mission concept.

The total ΔV from LEO to Mars, as a function of round-trip time and departure date during the ideal launch window, is plotted in Figure 4A. The investigated departure dates are a result of the time needed to develop the required technologies (leading to a highly optimistic early departure in 2020), with a launch date no later than January 1, 2041, as defined in the mission statement.

Concerning radiation exposure, it is most favorable to perform a deep space mission during solar maximum[3]. The first solar maximum within the shown departure dates will peak around 2022 (solar cycle 25); the following solar cycle

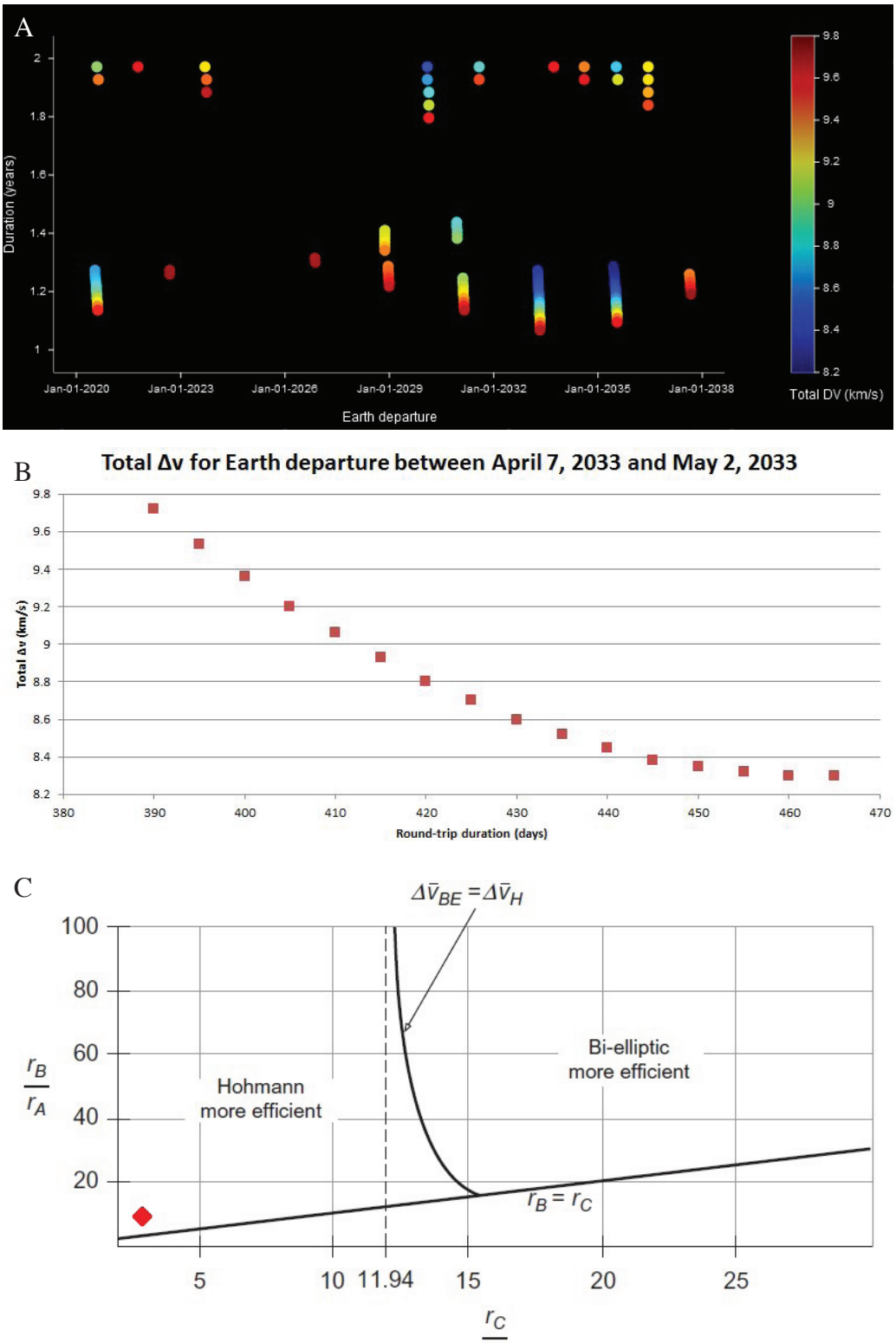


Figure 4. A) Total ΔV from LEO as a function of the Earth departure date and round-trip duration (Ames Research Center Mission Design Center Trajectory Browser, http://trajbrowser.arc.nasa.gov/traj_browser.php). B) Total ΔV requirements for Earth departure between April 7, 2033 and May 2, 2033. C) Comparison of bi-elliptic and Hohmann transfers (rA: starting orbit, rB: bi-elliptic apoapsis, rC: final orbit[16]).

26 peaks between 2033 and 2035. Solar cycle 25 is predicted to be one of the weakest in centuries. Additionally, there are only a few possible launch dates in 2022 for an opposition-class mission. For these reasons, April, 2033 is selected for further investigation.

The total ΔV has been calculated from LEO as a function of round-trip duration (Fig. 4B). It shows that a shorter round-trip duration automatically leads to an increase in the total ΔV required. It should be noted that the smallest ΔV, (i.e., longest round-trip duration) corresponds to the earliest departure date (April 7, 2033). With later departure dates, the round-trip duration decreases while ΔV increases. As a result, April 7, 2033 is determined to be the optimal departure date. This date selection allows for a launch slip of up to 25 days. Choosing this trajectory, there is calculated to be a constant line of sight from the spacecraft to Earth while in transit to and from Mars. Such a line of sight will be highly beneficial for flight control communications and crew safety.

Trajectories were calculated using a robust Lambert solver[14], with ephemerides from JPL[15]. At Mars, a bi-elliptic transfer is chosen to transport the crew safely from the mothership to Phobos. In theory, a Hohmann transfer would be more efficient to do this, where the red diamond marks the used transfer's position on the graph (Fig. 4C). However, as Mars is only just beginning to capture the spacecraft when starting this transfer, the actual ΔV required to do a Hohmann transfer is a factor of ten larger than the ΔV using the bi-elliptic transfer.

D. Technological Requirements and Strategic Knowledge Gaps

It is important to understand the technology required for the accomplishment of the mission. The technologies employed in the mission are currently at various readiness levels. Development time is taken into account in the mission architecture, and some of them will be discussed in detail in later sections of this report. A few of the key technological requirements are as follows: (1) A safe habitat needs to be designed for astronauts to survive for about 500 days in deep space. This includes radiation shielding, smart resource utilization, and comfortable living space for the astronauts. (2) Efficient propulsion systems that provide reasonable thrust at high I_{sp} are required to transport the crew and supplies. (3) Multiple, carefully-timed launches are required to transport all the modules to the Martian system. (4) The capability to abort the mission safely at various stages needs to be assessed.

In addition, there are certain important strategic knowledge gaps (SKG) that need to be retired before the undertaking of the manned phase of the mission: (1) The surface properties of Phobos and Deimos, such as regolith thickness and strength, are completely unknown and must be characterized before humans can be sent to either moon, (2) Deep space vehicles need to be tested for survival in deep space conditions prior to usage by astronauts, (3) Custom fairings need to be developed in order to accommodate high volume payloads on launch vehicles, (4) Methods for faster turnaround times for launch vehicles need to be developed in order to facilitate more launches in shorter periods of time, which allows for faster assembly of deep space cargo in LEO, (5) On-orbit assembly on a large scale needs to be perfected through research and testing, and (6) Improved thermal protection must be developed to protect spacecraft from the heat generated by reentry velocities in the range of 14-16 km/s.

IV. Details of the Phase Two Primary Mission

In this section we present pertinent details about specific aspects of the primary mission, including: (1) trajectory, (2) propulsion and vehicle selection, (3) habitation design and considerations for human success in deep space, (4) surface mission operations that will realize the scientific goals of the mission, (5) systems engineering, (6) planetary protection, (7) risk matrices for the mission and program as a whole, and (8) anticipated costs and partnerships.

A. Trajectory

The proposed trajectory (Fig. 5) is designed for an opposition-class mission with a round-trip duration of 465 days. Neither PROP1 nor PROP2 can be assembled and launched as a whole from Earth. Therefore, the launch campaign for the unmanned modules will start approximately five months prior to crew departure. Both PROP1 and PROP2 will each be brought into LEO through multiple launches over a period of several weeks. The design choice for LEO is further explained in Section IV.B.: Launch Vehicle Selection and Propulsion. After both modules have established a stable orbit of 300 km and are successfully assembled, the DSH will be launched to the same position and docked to the PROP1-PROP2 assembly. Only then, the crew, along with SEV, CM, and Service Module (SM), will be launched on April 7, 2033. The crew will enter LEO to rendezvous with the PROP1-PROP2-DSH assembly. Altogether, these six major components (PROP1-PROP2-DSH-SEV-CM-SM) comprise the MS, which will depart LEO later in April, 2033 for arrival at Mars in October, 2033. The MS will remain there for 30 days before beginning its return to Earth in November, 2033 with a planned Earth arrival in July, 2034. Figure 5A provides an overview of the heliocentric trajectories and the respective dates.

Upon successful assembly of the MS, PROP1 and PROP 2 will provide a ΔV of 3.5 km/s in order to achieve a C3 energy of 6.15 km^2/s^2. This C3 will place the spacecraft on a hyperbolic trajectory for arrival at Mars on October 10, 2033. The Earth escape trajectory will have an outgoing asymptote right ascension of 272°, a declination of -23°, and a velocity azimuth at the periapsis of 90° in the Earth inertial reference frame.

At Mars, the MS will burn with a ΔV of 2.2 km/s to achieve a Mars Orbit Insertion (MOI) and enter a 250 x 33,813 km parking orbit around Mars (orbital period of 1 sol) with an inclination of 34°. The eccentricity of this orbit (white dashed line in Figs. 5B, C) will aid in the transfer to Phobos' orbit (dark blue line in Figs. 5B, C) by losing much of the velocity from the approach. What follows is the phasing period, which could require a minimum of twelve hours to a maximum of 14 days. Phasing ends once two criteria are fulfilled: (1) MS and Phobos have a phase difference of 180°, and (2) the first condition is met when the MS is located at the parking orbit apoapsis (Point 2 in Figs. 5B, C).

As soon as these two criteria are met, the crew will board the SEV and depart for Phobos rendezvous. The desired orbit will be reached through a bi-elliptic Hohmann transfer with apse rotation requiring a ΔV of 0.4 km/s, which will change the SEV orbit inclination to 8° and raise the periapsis to 9377 km (light blue line in Figs. 5B, C). The periapsis will then match the radius of the Phobian orbit. After a 15-hour transfer, the SEV will perform a ΔV of -0.7 km/s to place the crew in a circular Phobos trailing orbit with an inclination of 1° (Point 3 in Figs. 5B, C). The SEV will trail Phobos for a minimal duration of 14 days. This duration can be increased if the initial MS-Phobos orbit phasing requires less than 14 days to complete. The SEV will visit several sites on the Phobian surface, which are described in Section IV. F: Science Mission and Surface Operations.

Upon completion of all surface operations, the MS will exit the highly eccentric parking orbit and enter the Phobos

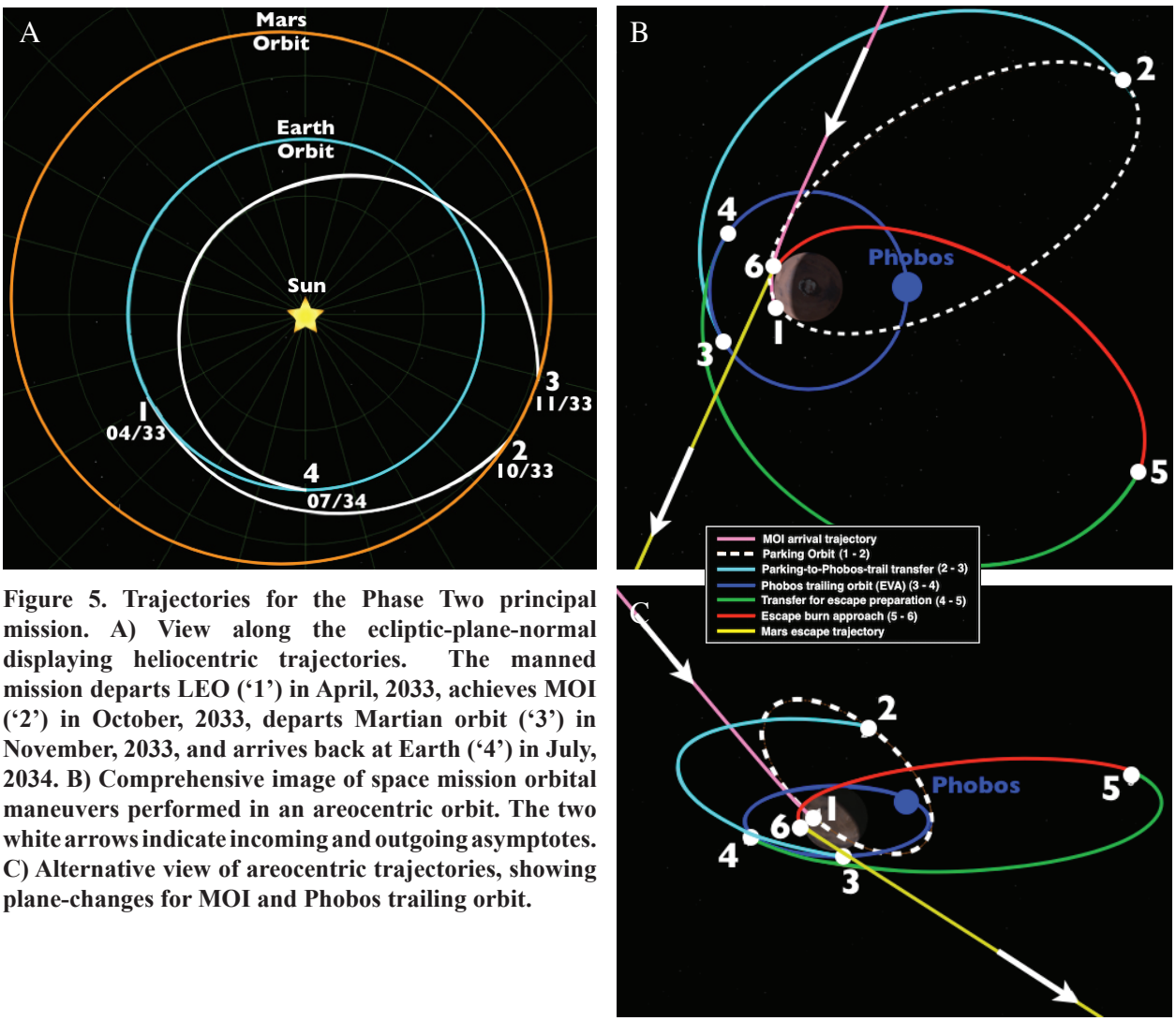


Figure 5. Trajectories for the Phase Two principal mission. A) View along the ecliptic-plane-normal displaying heliocentric trajectories. The manned mission departs LEO ('1') in April, 2033, achieves MOI ('2') in October, 2033, departs Martian orbit ('3') in November, 2033, and arrives back at Earth ('4') in July, 2034. B) Comprehensive image of space mission orbital maneuvers performed in an areocentric orbit. The two white arrows indicate incoming and outgoing asymptotes. C) Alternative view of areocentric trajectories, showing plane-changes for MOI and Phobos trailing orbit.

trailing orbit of the SEV, requiring a total ΔV of 1.1 km/s. Docking of MS and SEV will occur on November 6, 2033. After the crew transfers from the SEV back to the MS, the SEV will return to the Phobian surface. It will use the same anchoring system previously used during EVA activities to attach itself to Phobos. The crew will continue to collect scientific data from Phobos using tele-robotic systems during their return to Earth.

After two preparatory burns requiring a total ΔV of 0.7 km/s (Points 4 and 5 in Figs. 5B, C), a ΔV burn of 3.7 km/s will send the MS on a hyperbolic return trajectory on November 8, 2033. Arrival at Earth will be on July 16, 2034 with an Earth-relative velocity 16.2 km/s. An additional burn or aerobraking maneuver will reduce reentry velocity to approximately 14 km/s. A summary of the proposed trajectory with ΔV requirements can be found in Table 1.

Table 1. ΔV summary for MS and SEV mission operations

Description	ΔV (km/s)
Place MS on hyperbolic trajectory	3.5
Mars orbit insertion (MOI)	2.2
SEV burn at apoapsis when Phobos-HEV phase difference is 180° with plane change of 11.6° from ecliptic to 1.1° with respect to Mars' equatorial plane	0.4
SEV Phobos trailing orbit insertion for astronaut EVA	0.7
MS departure from parking orbit	0.4
Phobos trailing orbit insertion for MS	0.7
Phobos trailing orbit exit when EVA is complete	0.5
Burn at apoapsis to prepare for escape trajectory	0.2
ΔV for Mars sphere of influence escape for return to Earth	3.7
Total ΔV requirement for SEV	1.1
Total ΔV requirement for MS	11.2

B. Launch Vehicle Selection and Propulsion

1. Technology Selection

For the sizing of the propulsion system and subsequent selection of the required launch program, the mission is divided into several cargo launches and a separate crew launch. The propulsion subsystem is also responsible for decelerating the manned capsule to 14 km/s before re-entry. Table 2 gives an overview of the assumptions for the considered propulsive elements.

A trade-off is made on the staging orbit for the cargo and crew on the way to Mars. Both a highly elliptical orbit (HEO) and LEO are considered. Moreover, cryogenic and NTP systems are investigated. The options of staging in HEO included using solar electric propulsion (SEP) for the cargo deployment. On one hand, the lower velocity change required to reach Mars from HEO reduces the amount of propellant for the impulsive burn of the outbound stage and thus total mass. On the other hand, the mission complexity increases and development of additional elements is required. Especially, the mass savings for the NTP system are in a range of just a few percent, making the use of the SEP system questionable and may not justify the development of an additional module against the cost of one heavy lift launch. Table 3 shows the comparison of the four considered options with the assembly in LEO and a cryogenic propulsion stage as the baseline.

In conclusion, NTP to accelerate the cargo and crew from LEO has been selected as the desired mission architecture. While it has been considered to launch cargo directly into HEO, the reduced launcher performance (similar to the performance to geostationary transfer orbit) limits the mass of the individual modules and launcher availability.

Table 2. Overview of assumptions for propulsive elements

	Cryogenic	Nuclear Thermal	Solar Electric
Specific Impulse	465 s	900 s	2000 s
Structure Ratio	23%	27%	30 kg/kW
Burn duration	impulsive	implusive	1 year

Table 3. Comparison of considered propulsion options

	Mass Percentage (%)	Additional Power (MW)
Cryo from LEO	100	-
Cryo from HEO	68	1.8
NTP from LEO	**37**	**-**
NTP from HEO	35	1.0

2. Technology Justification

As described in Table 2, the high I_{sp} of an NTP system compared to existing cryogenic propulsion systems has the potential to reduce significantly the total mass in LEO, thereby driving down the number of launches and overall cost of the mission. In order to determine if the NTP option is available, the Technology Readiness

Level (TRL) and safety concerns related to radioactive material are taken into consideration. The United States developed and tested the technology in the 1960s in the ROVER and NERVA programs, and NTP is currently at TRL 6.[17,18] The Soviet Union was also developing a solid-core nuclear thermal engine[19]. In this report, it is assumed that the NTP will have been successfully tested on a mission such as one to a near-Earth asteroid. It has been suggested multiple times in the literature that NTP is an essential technology for trans-lunar missions.[20,21]

Disposal of equipment containing radioactive devices calls for special consideration. Failures in missions that use radioactive equipment, such as a radioisotope thermoelectric generator, have demonstrated different modes of deorbit scenarios. In the COSMOS 954 accident, the RTG disintegrated high in the atmosphere and spread radioactivity over a wide area.[22] In contrast, the RTG of the Apollo 13 lunar module entered the atmosphere at 11 km/s and survived reentry without breaking containment. In the case of NIMBUS B-1, which suffered a failure before reaching orbit, the RTG landed in the ocean without breaking containment and was later retrieved and reused.[23] However, there are important differences between RTG units and a NTP nuclear core. The nuclear core will be larger, more exposed, and will likely contain uranium carbide rather than plutonium. Extrapolating from the Apollo 13 accident, controlled reentry of the NTP and water landing is possible without breaking containment. The NTP core would have to be designed to survive reentry in case of launch failure before reaching orbit. However, considering negative public opinion on deorbiting nuclear material, it has been decided that in order to consider NTP as an option, the used stages have to stay in orbit with no chance of intersecting a planet or another celestial body.

The Earth-bound NTP stage will separate after performing a trans-Earth burn. A small fraction of the initial fuel mass will be left in the stage to perform two maneuvers. First, an out-of-plane burn to increase the inclination of the orbit in the Sun reference frame will ensure that the disposed NTP and Earth orbits never cross. Second, a retrograde burn at perihelion will ensure that the stage will not cross Mars orbit. While it would be fuel-intensive and uneconomical to incinerate the NTP in the sun, the described set of maneuvers will put the NTP in a safe orbit of no interest to other spacecraft, thus minimizing the risk of collision. Moreover, when technology to refurbish the nuclear core is available, a mission can be designed to rendezvous with the disposed NTP and recover the core.

Special attention needs to be given to risk management because of the presence of a radioactive payload. At a minimum, the following precautions should be put in place: (1) the nuclear core will be placed in a reinforced casing that will prevent containment breach in case of re-entry, (2) an automatic system will shut down the engine if the amount of radiation reaches a critical level, (3) every effort will be made to ensure that, once in space, the NTP stage does not collide with either Earth or Mars, and (4) the nuclear propulsion system will not be engaged until the the propulsive maneuver in space (i.e., no nuclear activity in Earth's atmosphere).

3. Detailed Design

The mass of the NTP module is approximated using the same assumptions as the NASA Human Spaceflight Architecture Team*. The propulsion system comprises two different modules. The first module consists of the engine and nuclear core as well as some propellant. The second module is a tank carrying the bulk of the liquid hydrogen.

The first stage consists of two engine cores generating a total thrust of 444 kN, which results in a thrust-to-weight ratio of 0.09. It is favorable to achieve a ratio of 0.1 for an impulsive burn, though in the case of starting from a circular orbit (LEO), it is not as critical as launching from an elliptical orbit. The burn duration is 82 min. The second stage (return trip) generates a thrust of 222 kN, resulting in ratio of 0.12 with a burn duration of 45 min. The elliptical orbit at Mars requires the increased ratio.

The main propellant for the NTP is liquid hydrogen with a very low density of 70.85 kg/m³. To be able to exploit the full launch mass capacity, modifications to the fairing diameter, as well as length, are required. A simple increase in the diameter has significant implications for launcher performance. In order to increase the payload volume while still meeting the structural and control requirements, a shroud optimized for aerodynamics is proposed (Fig. 6). The optimized configuration allows for a near-doubling of the payload volume while still achieving the same launcher performance.[24]

The mass increase of the fairing due to the additional structure is approximated to be 36% of the standard payload fairing design. The standard Atlas-V HLV payload fairing has a mass of 4,400 kg, which results in an increase of 1,600 kg. This increase is subtracted from the launcher performance. Analogously, the Falcon Heavy fairing and performance is adapted. Finally, one has to take into account the additional cost for the development of the new shroud.

*Mazanek, D., "Considerations for designing a human mission to the Martian moons," lecture given at the Caltech Space Challenge, March 26, 2013.

4. Launcher Selection and Launch Campaign

The mass of an individual propulsion module and tank is determined by the maximum launch capacity of common heavy-lift launch vehicles (mostly in development). Four different heavy-lift launchers are considered, Falcon Heavy with a performance of 53t, Space Launch System (SLS) Crew/Cargo I with 70t, SLS Cargo II with 120t and Atlas-V HLV with 29.4t to LEO.

The estimation of cost is given as a range due to the uncertainty in development and operational cost of the vehicle. Table 4 compares the total amount of launches and gives a cost range. Due to the prospective lower cost, the Falcon Heavy option is the preferred choice. However, the increased number of launches compared to the SLS option poses the risk of an increased probability of failure. Through multiple available launch sites/pads (e.g., Cape Canaveral Air Force Base, Florida, United States, and the ESA Spaceport, Kourou, French Guiana), the reaction time in case of failure can be reduced. The launch campaign must begin 30 weeks before the crewed launch if a turn-around time of three weeks is assumed. If it is only possible to achieve four launches of the Falcon Heavy per year[25], a combination of Falcon Heavy and Atlas-V HLV launches may be used to reduce the duration of the launch campaign at the expense of having two additional launches with increased associated costs.

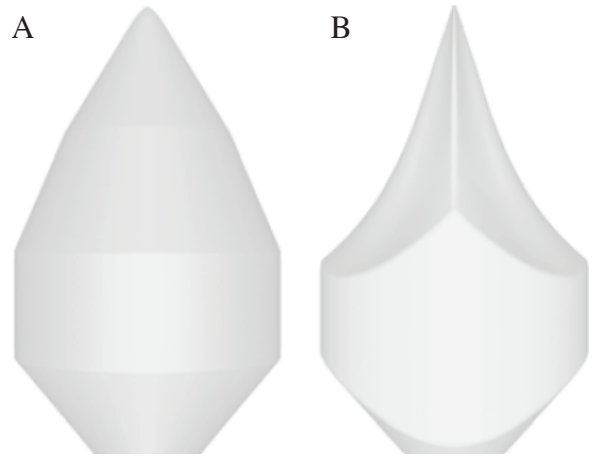


Figure 6. Initial (A) and optimized (B) fairings (modified after Ref. 24).

5. Staging Alternatives and Pre-deployment

There are a few options which would enable the use of SEP by lowering the power requirements of the launch. One scenario would be to lift the cargo either one after another and reuse the SEP module; a second would be to launch multiple SEP modules at the same time. The latter would reduce the maximum power required per module, although it would also add complexity to mission planning and, in the first case, it would increase the time needed for advanced planning.

The option of pre-deploying the return-trip propulsion modules at Mars before the crew arrives has been rejected due to safety reasons and abort capability. An alternative option would be to pre-deploy propulsion modules on the trajectory of the crewed vehicle. Similar to a checkpoint system, it would allow an abort at every point of the trajectory and would reduce the mass that is launched to Mars with high thrust impulsive propulsion. The propulsion modules would spiral with electric propulsion to the required orbit in advance and then meet up with the manned vehicle on its way to Mars. This architecture has increased mission complexity and requires thorough planning; it is not considered further here.

Table 4. Cost comparison of the various launcher options considered

Vehicle	Capacity (t)	No. of Launches	Total Cost (US$B)
Falcon Heavy	53	11	0.88-1.38
SLS Crew/ Cargo I/Cargo II	70/120	6	2.58-12.62
Atlas-V HLV	29.4	17	1.62-1.88
Atlas-V/Falcon combo	29.4/53	13	1.44-1.72

C. Habitation Elements

In developing the habitation elements for the mission, the following general systems architecture guidelines were followed to maximize system and operational reliability and flexibility, and, ultimately, the safety of the crew: (1) Leverage systems that are currently in use or development to minimize development cost and risk, (2) Maximize commonality across all mission elements to increase system robustness, lowering the number of spares required, and decreasing the costs of system development and manufacturing, (3) Maximize multifunctionality and synergies among systems, yielding increased functionality for less mass, (4) Account for crew safety during all mission modes, and (5) Implement lessons learned from past programs. Based on these guidelines, the following architectural choices were made.

1. Deep Space Habitat (DSH)

ISS-derived habitat structures were chosen as a baseline architecture for the DSH (Fig. 7), with modifications most notably made in the radiation protection to protect the crew for a long duration mission. Using modified ISS modules for the habitat is advantageous as the development work will be minimal, the system reliability has been demonstrated, ISS hardware is already flight-qualified, and ISS infrastructure such as payload racks and MPCV integration can be easily incorporated. The habitable volume is 76.3 m^3 [26], which is about 25% greater than the optimal recommended habitable volume for a crew of three for a mission duration of this length, according to the Celentano curve[53]. This habitat will be configured for both on- and off-duty use.

The primary Environmental Control and Life Support System (ECLSS) in the DSH is a closed-loop system similar to what is used on the ISS to minimize consumable mass. The secondary ECLSS is a passive system, known as Water Walls, that filters waste products through a series of forward osmosis treatment bags[27]. Including both of these systems in the DSH design provides redundancy and increased radiation protection. The primary ECLSS design was validated for our crew size and mission duration using the software tool *Environment for Life-Support Systems Simulation and Analysis* developed at the Institute for Space Systems (Institut für Raumfahrtsysteme) at the University of Stuttgart, Germany.

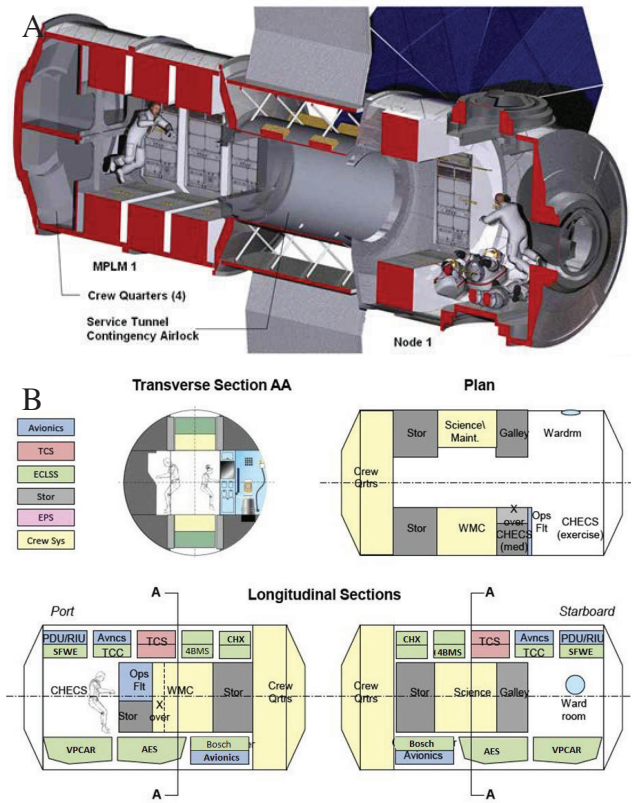


Figure 7. Conceptual illustrations of the DSH[26]. A) MPLM and node internal structure. B) Internal layout of MPLM, airlock and node.

2. Space Exploration Vehicle (SEV)

The SEV (Fig. 8) is a pressurized "roving vehicle" currently being developed at NASA Johnson Space Center capable of short duration missions. It facilitates flexible exploration by the astronaut in both the intravehicular and extravehicular environments through the use of robotics and spacewalks, respectively. Moreover, the use of suitports in the vehicle enables the rapid transition of crew members between intravehicular and extravehicular activities when required. The SEV has a pressurized volume of 54 m^3 and is capable of sustaining a two person crew for a maximum duration of 30 days. Due to the short mission duration for this vehicle, an open-loop ECLSS system architecture has been chosen to ensure high reliability, reduced complexity, and commonality between the vehicle and the Portable Life Support System (PLSS) of the spacesuits.

3. Extravehicular Mobility Unit (EMU)

The NASA-ILD Dover Mark III Spacesuit will be used for exploration outside of the SEV. This spacesuit has been baselined by NASA as the next generation spacesuit design, and has been designed to interface with the suitports onboard the SEV. The PLSS, which interfaces with the Mark III suit, will provide life support for the astronauts during extravehicular operations.

4. Orion Multipurpose Crew Vehicle (MPCV)

The Orion MPCV has been chosen as the baseline Earth reentry vehicle. This vehicle has been under extensive development by Lockheed Martin to support future NASA exploration missions, and has been designed with safety during all mission phases as its primary objective.

5. Spacecraft Atmospheres

Spacecraft atmospheres were chosen based on those suggested by the NASA Exploration Atmospheres Working

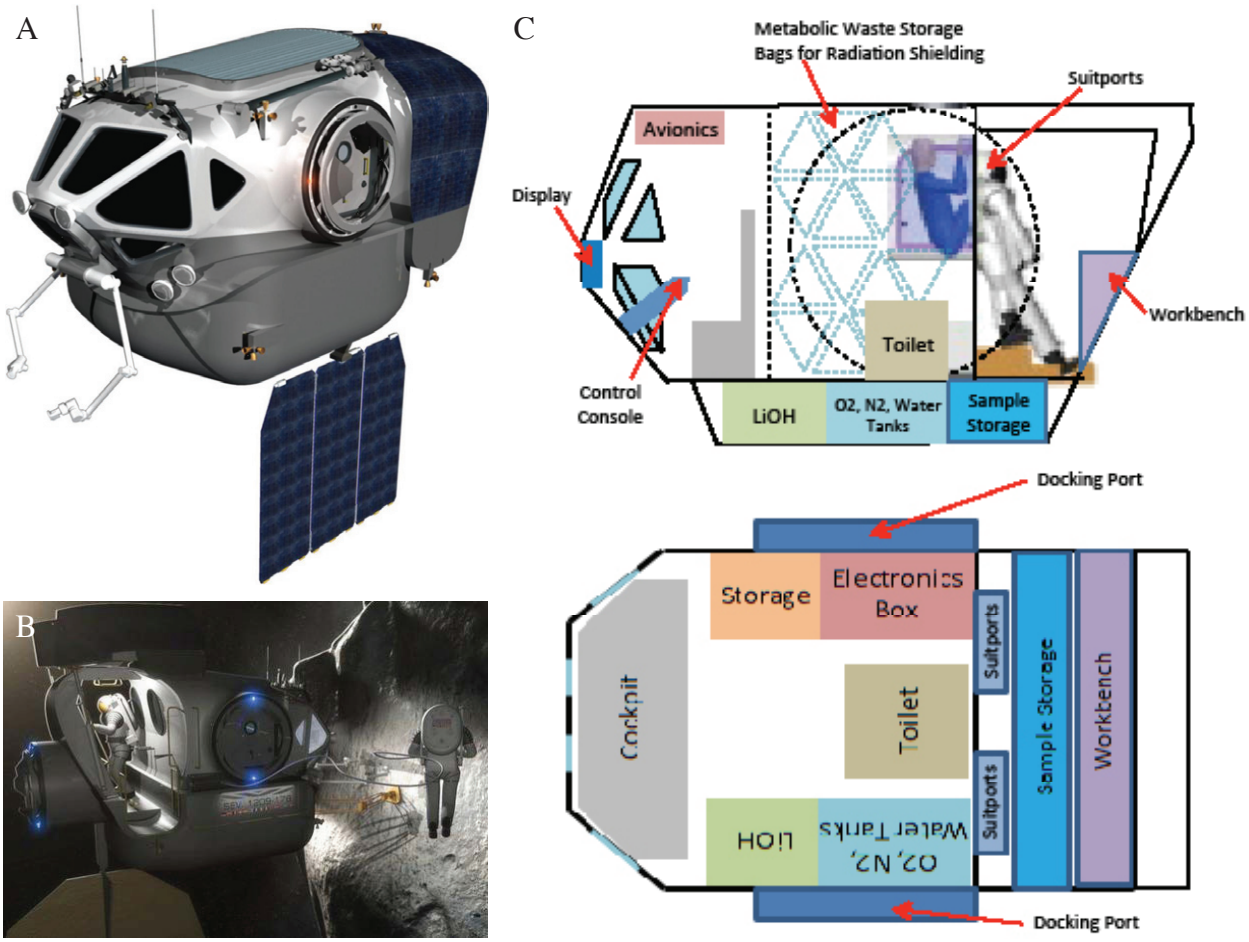


Figure 8. Illustrative views of the Space Exploration Vehicle. A) Front (http://www.oceaneering.com/wp-content/uploads/2011/08/OSS-SEV-space.jpg), B) Side view in operation, C) Conceptual layout.

Group[28] to ensure atmospheric capability between spacecraft elements while ensuring that pre-breath time for the required EVA frequency is properly accounted for. Table 5 lists the atmospheres selected for each habitation element to be used in the mission. It should be noted that nitrogen was chosen as the diluent gas in each atmospheric composition. The design presented here for the EMU requires no pre-breathe time.

D. Human Factors

1. Crew Selection

Three individuals will be selected for participation in the mission. This crew complement facilitates simultaneous science and support operations at both Phobos and Deimos, while ensuring continual presence in each habitable element. ECLSS and other mass considerations dictate the smallest crew possible to meet our mission goals. Psychological research indicates that a three-person crew is capable of establishing stable group dynamics[29]. Additionally, the three-person crew model is the only design that has demonstrated success in an exploration class mission environment. The crew selection board will employ select-out procedures to obtain a pool of candidates with adequate intellectual and physical competencies. All potential crew members must possess medical clearance similar to that permitted by the Federal Aviation Administration first-class examination (analogous to space shuttle pilot clearance). Select-in procedures including detailed psychological examinations will then be used to refine the applicant pool into a set of primary and alternate crew members.

Crew selection will not demonstrate gender selectivity. However, our process will favor individuals between 40 and 55 years old. Additional basic research and continued statistical analysis of emerging ISS data will be required to establish optimal age-based selectivity. Estimated remaining life years help dictate the risk of cancer incidence in radia-

tion-exposed astronauts, favoring older crew candidates. However, bone mineral density declines linearly after entry into mid-life, which dictates selectivity favoring younger astronauts. Additionally, bone density in women leaves the initial linear trend at menopause, leading to a temporary increase in bone loss rate that stabilizes to a more moderate rate of loss after a few years. Although women have lower radiation tolerances, as defined by NASA's 3% radiation exposure induced death (REID) standard, we do not have a selectivity bias against one gender due to the low likelihood of exceeding tolerance given a nominal mission profile and the stipulation that applicants will volunteer for the astronaut corps with a full understanding of the unique dangers that each gender may face.

Table 5. Atmospheres selected for each habitation

Habitation Element	Atmospheric Pressure and Composition
DSH	101.3 kPa (14.7 psi), 21% O_2 nominally, 70.3 kPa (10.2psi), 26.5% O_2 during pressurization with the SEV
SEV	70.3 kPa (10.2 psi), 26.5% O_2
EMU	57 kPa (8.3 psi), 100% O_2 (Mark III suit)
MPCV	101.3 kPa (14.7 psi), 21% O_2 nominally, 70.3 kPa (10.2 psi), 26.5% O_2 during depressurization prior to EVA from the vehicle

2. Crew Training

The crew will need extensive training in geology, vehicle maintenance, and medical event management. The two crew members participating in the SEV expedition to Phobos and associated EVA should receive the equivalent of six weeks of dedicated geology training (240 hours) including tutorials and terrestrial sampling expeditions. Vehicle maintenance training similar to that received by ISS crew members should be a key element in the preparation of at least one SEV crew member and the DSH-only crew member. All crew members should be trained in basic first aid and must be proficient in the use of the medical kits. Two crew members need to be trained in advanced medical techniques so that incapacitation of the medical expert will leave one person to provide advanced non-surgical care and emergency surgery. One SEV crew member and the crew member who remains in the DSH should be trained in rudimentary surgical skills and must be able to perform simple operations.

Three crews will be trained simultaneously. In the event of a health contingency, issues arising within three months of launch will result in a substitution of individual crew members. Prior to the three month cut-off, whole crew substitution will be the primary option.

3. Crew Health Care

a. Medical care

Medical equipment and supplies consist of a standard ISS medical kit scaled up from 460 kg to 1000 kg of equipment, including a high-resolution ultrasound imager and expanded surgical supply kit. Medical consumables will also resemble those used in the ISS Health Maintenance System, expanded from 260 kg to 500 kg of pharmaceuticals and other consumable supplies. This provides a total medical supply kit for the mission with a mass of 1500 kg and an approximate volume of 6.5 m^3, a size that fits comfortably into the larger mission design.

b. Psychological considerations

Long-term spaceflight produces extreme psychological stress, which, if ignored, can result in serious degradation of mental health that puts the mission and crew at risk but, if recognized in advance, can be mitigated. Major sources of psychological stress include isolation, interpersonal conflict, physical deterioration, separation from family, and lack of privacy. To improve the psychological well-being of the astronauts, it is important to provide them with nutritious food, communication with family, entertainment, and exercise throughout the duration of the mission. Typically, astronauts are provided with a variety of dehydrated food for their meals. To supplement their nutritional intake, small plants that serve as a food source may be included in the mission. Psychological benefits may be gained by both maintaining plants and harvesting them to obtain fresh food. When not conducting on-board science experiments, the crew members will be able to spend leisure time much as they would on Earth, reading books, listening to music, and emailing with friends and family.

c. Countermeasures and mitigation strategies

As much as possible, deleterious effects of space travel will be minimized through various countermeasures and mitigations strategies (Table 6). Spinning the habitat to create an artificial gravitational force is unreasonable due to the size of the spacecraft. There is a level of risk accepted in astronauts developing long-term adverse side effects due to the microgravity deconditioning. Once the mission is complete, the crew will have access to a full range of medical facilities to regain pre-flight levels of health and fitness.

Table 6. Physiological systems with corresponding risks and mitigation strategies[30]

System	Risk	Mitigation Strategy	Countermeasure development
Musculoskeletal	Debilitation, degradation, bone fracture	Gravity loading countermeasure skinsuit developed at MIT[31]. Standard NASA exercise protocol involving 1.5 hours of aerobic and resistive exercise per day will be undertaken by crew members during the mission. The equipment required to meet this requirement includes an Advanced Resistive Exercise Device, Cycle Ergometer with vibration isolation and stabilization system, and Treadmill Two.	Study hormones impacting serum ion concentrations, pharmacological inhibition of bone loss
Cardiovascular	Cardiovascular degradation, cardiac events, heart disease	Cardiovascular exercise to keep cardiac muscle strong	ECG studies to search for coherent etiology
Vestibular	Motion sickness	Crew training prior to mission to aid in vestibular adaptability, motion sickness medication	---
Immunological	Immunosuppression	Manage crew stress and sleep deprivation; provide adequate nutrition	Proteomics for the study of cytokine/chemokine flux impact
Nutrition	Inadequate nutrition	Provide adequate nutrition to meet crew metabolic and health demands	Investigate Vitamin D and other nutrient levels
Ophthalmological	Optic nerve/retina damage	Observation	Papilledema development and mitigation must be investigated to prevent damage to the retina[32]
Auditory	Hearing loss due to noise levels of cabin systems	Implement NASA standards for allowable noise levels from all systems and payloads	---
Neurological	Neural damage due to radiation	Observation	Evaluation of optimal strategies for psychological stability
Psychological	Debilitation, impaired function	Mission preparation, provide family contact, ensure routine, maintain workload, include experiment with live plants, exercise, maximize habitat volume.	Evaluation of optimal strategies for psychological stability
Systemic	Radiation damage	Spacecraft architecture and exposure monitoring	Investigate use of antioxidants
	Decompression sickness (DCS)	Decompression procedures that follow NASA standard pre-breath protocols to prevent DCS prior to EVA	---
	Trauma/Illness	Medical kits, crew health care system, surgical tools	---

4. Radiation

a. Monitoring

Tissue Equivalent Proportional Counters (TEPCs)--currently in use on the ISS--measure radiation doses for complex radiation fields and should be deployed in several locations in the DSH and SEV to measure radiation levels during transit, exploration, and EVA. Radiation levels throughout the DSH can be actively evaluated using portable TEPCs, allowing the crew to move to the most highly protected region of the vehicle during a solar particle event (SPE). An instrument similar to the Radiation Assessment Detector on MSL (also soon to be deployed on the ISS) will be deployed by the science team on the exterior of the DSH to record charged particle and neutron incidence for scientific use. SPE monitoring will be conducted using the existing network of solar observatories (i.e., SDO, SOHO, GOES) and any future expansion.

b. Mitigation

The mission architecture provides for 20 g/cm^2 of uniform radiation shielding in the DSH. This degree of shielding is referenced in NASA documentation as the convergent design option for human missions based on an SPE mitigation/mass trade[33]. Radiation shields that incorporate low atomic mass materials are capable of suppressing damaging secondary radiation in the form of neutrons that are ejected during particle transit through the aluminum hull of a module.

c. Exposure estimates

NASA has calculated the safe number of days that a person can travel in space when their vehicle is designed as mentioned above. These values are based on the need to prevent astronauts from exceeding an increased risk of 3% for REID (at the 95% confidence level). Failure to mitigate the effects of GCR and SPEs could lead to acute health effects including radiation sickness leading to incapacitation or death. Long-term risks include carcinogenesis, neural tissue damage, stem cell disturbances, and cataracts.

5. Biological Science Payload

In-flight monitoring of a panel of proteins and ions in blood, urine, and saliva at weekly intervals will allow for customized changes in physiological degradation mitigation efforts for each crew member. One of the DSH's two international standard payload racks (ISPR) would contain NASA-provided hardware intended for biological analysis. The second ISPR will be used to accommodate experiments selected by peer review from a pool of academic and industrial proposals.

To transport biological samples back to Earth, a modified freezer based on the Minus Eighty Degree Laboratory Freezer for ISS (MELFI) has been considered (Fig. 9). In this modified design (four storage units each divided into four subsections) will employ a modified cooling mechanism based on the General Laboratory Active Cryogenic ISS

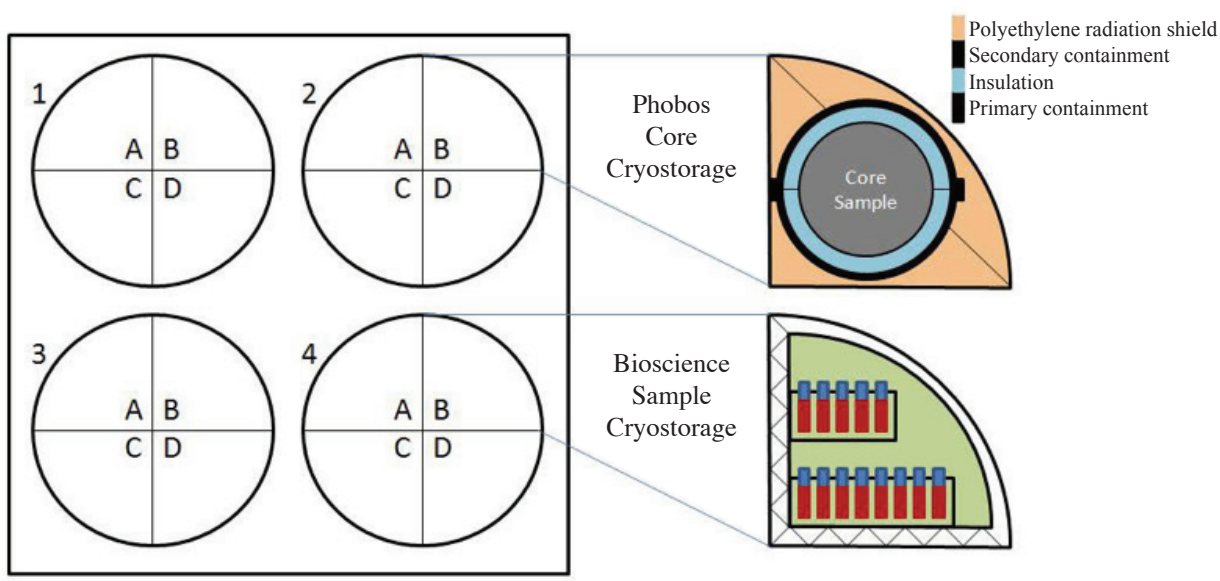


Figure 9. Schematic diagram of the sample storage locker configuration on the DSH modified after the MELFI design.

Experiment Refrigerator (GLACIER) to achieve a storage temperature of -100°C. Geological samples will be stored in units 1 and 2, while bioscience samples can be stored in units 3 and 4.

E. Science Mission and Surface Operations

1. Precursor Operations

The goals of the precursor mission, in order of importance, are as follows: (1) to determine if humans can safely land on Phobos during the primary mission, or on Deimos in the event that Phobos is not feasible, (2) to establish a communications relay system that will facilitate the primary mission, (3) to gain important information regarding the nature and composition of the primary landing site to plan for a landing of the primary mission, and (4) to acquire remote sensing data on both of the moons to be used to understand their composition.

In order to carry out the primary mission of landing humans on one of the Martian moons, we must characterize the structure and surface properties of Phobos and Deimos. This will be achieved on both moons using four impactor experiments at preselected sites, in-situ sampling and analysis conducted remotely using an immobile lander with an extendable arm, and a combination of remote observations from the PDS. In-situ sampling sites will be determined based on the findings from initial remote sensing surveys conducted by the PDS and from the impactor experiments.

Impactor sites were selected in order to target sites of geologic interest, sites where future missions might land, and other widely-spaced sites to learn more about the distribution of the surface characteristics. The surface characteristics revealed by the four impactor tests will be a strong driver in determining how and where the lander is deployed and how and where the manned mission will dock and operate.

To reduce complexity, DE and PE will be identical lander and impactor packages. The impactor package will be modeled after the one planned for the Japanese Lunar-A mission, but with four penetrometers instead of two. The Deimos and Phobos landers could be modeled after the Philae lander used in the Rosetta mission[34].

2. Science Instrumentation

The primary instrument objective is to assess the surface environment to optimize human interactions with the surface environment of Phobos. In order to do this it is important to execute a comprehensive study of the planetary bodies to ensure the safety of the astronauts and the completion of mission objectives. The instrument suites have been designed to investigate the nature of the surface and subsurface of the Martian moons. This is a useful investigation for several reasons: (1) Determination of the nature of the regolith (uppermost, loose 'soil') allows assessment of the mechanical and chemical properties of the surface, (2) Identifying the strength and porosity of the surface provides critical information to help plan docking and anchoring maneuvers during the manned component of the mission, and (3) Studies of the flux of interstellar material and radiation levels will help to develop shielding techniques.

Three unique science instrument suites (Surveyor, Explorer, and Expedition; Table 7) have been designed to achieve the aforementioned science objectives during the mission.

The Surveyor suite is comprised of a number of heritage spectrometers and cameras, configured to investigate regolith properties remotely from orbit. Instruments in the Surveyor suite will provide valuable data on the topography of Phobos and Deimos, the flux of interplanetary material crossing the orbital plane of Phobos and Deimos, surface mineral composition, volatile abundance (such as water and CO_2), and the strength of the magnetic fields on the moons.

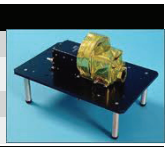
The Explorer suite is modeled after instruments from past NASA and JAXA missions[35-42]. The penetrometer device contained within the impactor package is modeled after the piezoelectric sensing element used in the Huygens probe. It was uniquely calibrated to withstand cryogenic temperatures, and future development will allow impact velocities of 300 m/s. The voltammetry, spectroscopy, and x-ray diffraction/fluorescence instruments (Wet Chemistry Lab, LIBS, and CheMin, respectively) do not require modification and are replicas of the original instruments. The Phobos and Deimos landers will employ robotic arms built on 360° swivels to deliver multiple regolith samples to the experiment chamber. Lastly, an additional micrometeoroid detector (modified for a lander spacecraft) will be deployed in the Explorer suite to assess impact rates and material deposition. This instrument will provide details about the nature of the landing environment in which astronauts will execute future EVAs. Once positioned on the surface of Phobos or Deimos, micrometeoroid detector panels will deploy along the sides of the lander.

Astronauts will manually deploy the Expedition suite of science instruments during EVA sorties (see Section IV.E.3. Manned Phobos Operations for more details). Seismic and radiation studies will be undertaken using heritage instruments. The PRSC (Planetary Retrieval of Subsurface Cores) will be based on core drilling that was done on the moon during the Apollo 15-17 missions but will have a somewhat larger core diameter for increased sample

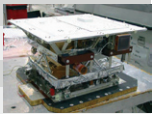
Table 7. Detailed Instrument suites[35-42].

Surveyor Suite

Objective: Global topographic and colorimetric mapping of surface features
- Technique: Optical Imaging
- Instrument Precedent: Asteroid Multi-band Imaging Camera (Hayabusa)
- Priority: High
- Requirements: Mass: 6kg, Power 16W



Objective: Remote analysis of surface mineral composition
- Technique: High-resolution imaging spectroscopy
- Instrument Precedent: Visible Infrared Thermal Imaging Spectrometer (Rosetta)
- Priority: High
- Requirements: Mass: 23kg, Power 32W ; Spectral Range: 0.025-5.0µm



Objective: Remote analysis of surface hydrogen/water and elemental analysis of surface composition
- Technique: Energetic Particle detection
- Instrument Precedent: Gamma Ray and Neutron Detector (Dawn)
- Priority: High
- Requirements: Mass: 14kg, Power 4.5W; Spectral Range: 1-3µm Resolution

Objective: Remote analysis of surface mineral composition
- Technique: Spectroscopy
- Instrument Precedent: VIS/NIR Spectrometer (Dawn)
- Priority: High
- Requirements: Mass: 15kg, Power 15W; Spectral Range: 1-3µm Resolution

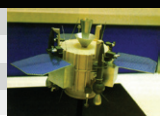
Objective: Global mapping of surface topography
- Technique: Laser altimetry
- Instrument Precedent: Light Detection and Ranging--LIDAR (Hayabusa)
- Priority: High
- Requirements: Mass: 2kg, Power 7W

Objective: Determine strength of magnetic field
- Technique: Magnetometry
- Instrument Precedent: Magnetometer (Voyager 1 and 2)
- Priority: High
- Requirements: Mass: 6kg, Power 2W

Objective: Micrometeoroid Detection of the Martian system
- Technique: Spallation
- Instrument Precedent: Micrometeoroid Detector (Multiple Missions--Explorer, Apollo, Pioneer, etc)
- Priority: High
- Requirements: Mass: 2kg, Power 10W

Explorer Suite

Objective: Determine near-surface thermal and structural properties
- Technique: Piezoelectric sensing
- Instrument Precedent: Impactor Package, (Lunar-A, Hayabusa)
- Priority: High
- Requirements: Mass: 180kg, Power 4W



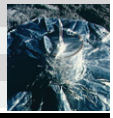
Objective: In-situ analysis of soil chemistry
- Technique: Cyclic Voltammetry, Anodic Stripping Voltammetry, Laser-induced breakdown spectroscopy (LIBS)
- Instrument Precedent: Wet chemistry Lab (Mars Phoenix Lander), LIBS (Mars Science Laboratory-MSL)
- Priority: High
- Requirements: Mass: 3kg, Power 13W

Objective: In-situ analysis of mineralogy of soil mineralogy
- Technique: X-Ray Diffraction/Fluorescence
- Instrument Precedent: CheMin (Mars Science Laboratory-MSL)
- Priority: High
- Requirements: Mass: 10kg, Power 46W

Objective: Micrometeoroid Detection at the surface of Phobos/Deimos
- Technique: Spallation
- Instrument Precedent: Micrometeoroid Detector (Multiple Missions--Explorer, Apollo, Pioneer, etc)
- Priority: High
- Requirements: Mass: 2kg, Power 10W

Expedition Suite

Objective: Detection of tidal forces/seismic activities
- Technique: Seismometry
- Instrument Precedent: Passive Seismic Experiment (Apollo 16)
- Priority: High
- Requirements: Mass: 10kg, Power 8W

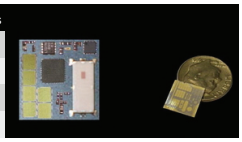


Objective: Laser ranging of Phobos altitude
- Technique: Specular Reflection
- Instrument Precedent: Laser Ranging Retroreflector (Earth Resources Spacecraft-ESR)
- Priority: Low
- Requirements: Mass: 3kg, Power N/A

Objective: Collect robust samples of subsurface
- Technique: Core Drilling
- Instrument Precedent: Planetary retrieval of subsurface coreS [PRSC]
- Priority: High
- Requirements: Mass: 50kg, Power 1kW

Objective: Collect accurate radiation data across a wide energy regime and particle families
- Technique: Energetic particle analysis
- Instrument Precedent: Radiation Assessment Detector (Mars Science Lander-MSL)
- Priority: High
- Requirements: Mass: 2kg, Power 4W

Objective: Perform a variety of 'sensors-on-chip' type experiments
- Technique: Energetic particle analysis
- Instrument Precedent: Chipsats (Under development)
- Priority: Low
- Requirements: Mass: 5kg, Power N/A



return. The ChipSat instruments will be developed as a public outreach effort to achieve the aims of independent science groups from around the world. Together, the employment of these instruments represents an innovative approach to meet a principal science objective of both on-the-ground data collection and sample return; they also promote the use of scientific equipment that is smaller in scale and lighter in weight.

3. *Manned Phobos Operations*

Although one could argue that many of the mission's scientific goals could be achieved through robotic means, there is a decided advantage to having humans "on the ground" to collect samples and to deploy instruments. Humans are versatile in that they can evaluate samples for quality and quantity in real time and can troubleshoot instruments on the fly. Up to this point in time, no robotic mission has returned extraterrestrial samples to Earth, and those samples that have been returned via manned EVAs have reaped great rewards for the scientific community.

With these points in mind, the primary mission is designed to have two human crew members collecting samples directly on Phobos' surface. The largest challenge to realizing this task is the near-zero gravity of Phobos. To operate on the surface, the crewed SEV will perform a rendezvous and docking procedure with the moon at each of the two

pre-selected landing sites, where the vehicle will be anchored to the surface. Whether a conventional harpoon or drill-type anchor[43-46] will be used, as opposed to an unconventional method such as microspines or netting[47,48], will depend on site surface material characteristics (grain size, depth, density, cohesion). This composition information will be provided by data collected during the precursor mission.

Once on the surface, the astronauts will have two modes available for EVA operations, depending on surface conditions. The first mode consists of one astronaut collecting samples and placing instruments with their feet fixed to the end of a robotic arm that extends from the SEV. In this configuration, one crew member must remain inside the vehicle in order to operate the robotic arm. This type of EVA has been shown to provide the most mobility of the methods investigated during the NEEMO under-water simulation program for manipulation of equipment in a microgravity environment[49]. The robotic arm configuration would be especially advantageous if the surface regolith proves to be so thick and fine-grained that conventional maneuvering is unfeasible. The second mode, appropriate for sand- to boulder-sized regolith, anchors the astronauts to the Phobian surface using a tether and a scaled-down version of the regolith anchor used by the SEV. A secondary safety tether is connected to the SEV. For safety purposes, this second mode nominally involves only one crew member on EVA at a time, while the other performs monitoring and non-EVA activities inside the SEV. For both modes, use of an MMU-type device may aid astronaut maneuvers on the surface.

Surface activities include collecting geologic samples and placing seismometers and retro-reflectors, as well as other experiments, such as ChipSat deployment. A full schedule of surface operations for the two SEV crew members over the two-week mobilization on Phobos is shown in Table 8. The Phobian surface exploration segment of the mission is designed for a nominal length of 14 days, as constrained by the planned mission trajectory. Using state-of-the-art portable life support technology and relevant suit design, the safe duration for a single EVA has been estimated to be approximately four hours. Considering both rest periods and the constraints of the PLSS, the four-hour-EVA duration will allow for a maximum total of ten EVAs. Including contingency PLSS operational supply and potential for unanticipated required surface activity, a target number of eight EVAs has been planned, with four EVAs at each of the two predetermined landing sites.

Surface samples will include rock and regolith scoop samples, as well as drilled core samples. Drill cores are planned to be 40-50 mm diameter x 3 m in length. The drill will be an electrically powered percussive hammer system, with a similar foot treadle contingency design for core removal as used on the Apollo 15-17 missions. Core samples will remain in their sleeves for direct placement into storage on the SEV. Thin samples of top-layer, fine-grained regolith will also be collected using adhesive pads. This will allow specific study of the regolith immediately exposed to the space environment. Loose geologic samples in collection bags and core samples will be stored in an exterior containment unit that will be placed in the SEV airlock using the robotic arm. The samples will be stored at appropriate cryogenic temperatures once on the SEV. Upon returning to the DSH, drill core will be stored in the modified MELFI freezer along with biologic samples (Fig. 9) to bring back to Earth.

According to the surface operations schedule, on days three and ten passive seismic arrays will be placed on the

Table 8. Concept of operations and EVA schedule at Phobian surface

Day	Task	Notes
1	Secure and EVA prep	
2	EVA, site 1, astronaut 1	Collect contingency sample, surface samples, place retro-reflectors
3	EVA, site 1, astronaut 2	Surface samples, passive seismometer placement
4	Rest day	
5	EVA, site 1, astronaut 1	Core samples
6	EVA, site 1, astronaut 2	Core samples, public outreach deployment
7	Move to site 2	
8	EVA prep	
9	EVA, site 2, astronaut 1	Contingency sample, surface samples
10	EVA, site 2, astronaut 2	Surface samples, passive seismometer placement
11	Rest day	
12	EVA, site 2, astronaut 1	Core samples
13	EVA, site 2, astronaut 2	Core samples, public outreach deployment

Phobian surface to record seismic waves generated by internal strain in the moon. At each of the two landing sites, the seismometers must be placed with a spacing of 10s to 100s of meters apart and the exact location of each instrument recorded. EVA mode two would be preferable for this experiment, as it allows astronauts greater reach, and seismometers may be placed farther apart, which in turn, allows for deeper imaging into the crust of the moon. Inclusion of an active-source seismic array remains under consideration.

On day two, one or more retro-reflector(s) will be placed on the Phobian surface. Retro-reflectors are mirrors that are used to reflect an electromagnetic signal back to its source. This instrument, once placed on Phobos, will have future application when a signal can be directed to the moon from rovers or stations on the Martian surface in order to determine the orbital distance of Phobos. The reflector measurements are regularly made over a long period of time (i.e., decades) to determine deviations in orbit. This method has yielded excellent results using our own moon[50], and we anticipate that it will answer such scientific questions as the rate at which Phobos is encroaching on Mars[51].

After the full two week period on the surface, the SEV crew will return to the DSH and send the SEV (remotely) on a return path to Phobos, where it will re-anchor to the surface to prevent risk of drift-off and to comply with the requirements of planetary protection. The crew will be able to operate the SEV robotic arm remotely from the habitat, in order to continue to perform surface experiments after completion of the main mission and to demonstrate feasibility of performing telerobotic operations from orbit.

F. Systems Engineering

1. Structures

In order to determine the size of the propulsion system, hence the launch configuration, a detailed mass and energy budget has been developed for every unit. The total value for mass lifted off the Earth is about 514 metric tons. With a majority of the vehicle modules available for direct purchase, structural modifications are largely related to structural integration. Our proposed mission will require large-scale manufacturing in general and the generation of modified fairings. The high number of launches required to place individual mission components into orbit will drive a robust capability to assemble components from Earth-based ground stations via telerobotic operations. Although many of the pre-built structures such as the DSH and MPCV are equipped with their own solar arrays, other modules such as the SEV present the opportunity to utilize collapsible-parts technology.

2. Attitude Determination and Control

A trade study was conducted for attitude sensors with the single point requirement of reliability and redundancy in consideration. Choices included magnetometers, earth sensors (for initial phase only), sun sensors, star sensors and gyroscopes. It was decided that both precursor and primary missions be equipped with a combination of a Star Tracker and a Sun Position Sensor. Rationale behind the selection was: (1) Non-dependence on moving parts, (2) Extremely light on mass and volume budget, (3) Starfield view availability for a large fraction of orbits, (4) Availability of line-of-sight with sun during rare Solar saturation. Clementine Star Tracker cameras [NSSDC ID: 1994-004A-07] are the preliminary first choice. During times of sun saturation, the system will fall back on the sun position sensor which derives its heritage from the proven GOES-15 [NSSDC ID: 2010-008A]. It is believed with a high level of confidence that the above stated two-line system will be reliable. A similar trade study was performed for attitude control system. The choices included thrusters, spin stabilization (only for certain modules at a time), momentum wheels, control moment gyros, solar sails (for cruise phase only), gravity gradient (for near-Earth phase only) and magnetic torquers. It was decided that the SEV and DSH be equipped with a combination of a Control Moment Gyroscope (CMG) and monopropellant hydrazine thrusters and that all stages of Phase One and the smaller stages of Phase Two be equipped with just monopropellant hydrazine thrusters. The rationale behind this selection was: (1) Non-dependence on magnetic fields or gravity gradients, (2) The demonstrated nature of technologies involved, (3) Sufficient capability for relatively fast maneuvers, and (4) Sufficiently high level of achievable precision. Since SEV and DSH are the only very massive stages, they require special attention. It is being proposed that 3 single-gimbal CMGs be used on SEV and DSH. In addition there should also be a backup system of monopropellant thrusters using Hydrazine. This will provide a three-axis control with built-in contingency while maintaining simplicity and reliability.

3. Communications

The communication system for our mission will be similar in principle to that of MSL but with added redundancies. The following communication links are required: video link, voice communication link with Earth, telemetry

link, crew voice communication link. A direct-to-earth (DTE) X-band radio link and a high-volume, parallel UHF link via a Mars orbiter similar to MRO[52] will be used. On Earth, the Deep Space Network (DSN) will be used to establish communication, either directly with the crew vehicle or through a Mars orbiter. The DTE link will be used to transmit Multiple Frequency-Shift Keying tones containing software updates and commands. The Phobos-Deimos Surveyor (PDS) from the precursor mission will serve as an additional communication relay to Earth for the spacecraft while in the Martian system. The PDS will have communication facilities on-board for the UHF and Ka-bands. The UHF band will be primarily used for communication between the two spacecraft and between the spacecraft and PDS; the Ka-band link will be used by the PDS to communicate with Earth through a high-gain antenna to provide fast data transfer rates. On Earth, we will use 34 m and 70 m facilities on the DSN. Pairs of Electra Lite transponders will be used to communicate over the UHF band. The DTE data rate will be between 500-32,000 bits per second and will be 100-250 megabits per second via the UHF link. This data rate is sufficient for the fast transfer of low quality video. Communication through multiple antennas over the two bands provides good redundancy for the mission in case of failure. There are two independent systems on board the DSH and SEV, either of which may be used in the event of a failure of one of them.

The transfer orbit has been designed to ensure that solar occultation never occurs during the cruise phase. Once the spacecraft reaches the Martian system, however, there will be Earth occultations. STK simulations of the orbits of the PDS, CM, and SEV showed that the primary landing site on Phobos would be visible from the CM and PDS twice a day each, with an average encounter length of 3 hours each. Additionally, the primary site will be visible to the Earth, via the DSN, twice a day with an average encounter length of 4 hours each. The MS in a parking orbit would have twice-a-day communication windows with the DSN, with a maximum duration of 8 hours each. A small relay station will be set up on the edge of the Stickney Crater prior to EVA excursion into Site B in order to maintain contact with the PDS. The dates of the EVA need to be carefully planned so as to ensure no occultation of the PDS by Mars. The exact times of occultation could not be evaluated during the course of this review owing to a variability in EVA dates.

Occultations can be reduced and blackout times shortened by sending more communications satellites into high-inclination Martian orbits that provide a good view of Phobos and Earth. If NASA's Mars Exploration Program continues to send MRO-type missions into orbit around Mars, or if ESA's 2016 ExoMars Orbiter succeeds and continues to run for over a decade, we propose to utilize communications resources from these missions to improve communications coverage for the Asaph-1 mission. In any case, the spacecraft will be approximately 390 million km away from Earth at its farthest point, implying a maximum propagation delay of 22 minutes. Given such a lag in communications, the astronauts must be trained to operate autonomously.

4. Electrical Power Systems

As a first step, a trade analysis for various EPS systems was performed for the electrical power converters and the electrical power storages for various stages and modules. First, possible electrical power converter technologies are identified. Among others the following power converters exist: (1) photovoltaic, (2) solardynamic, (3) nuclear power plant, and (4) radioisotope thermoelectric generator. At the end of the trade study, we chose to utilize solar panels. The design is based on the Ultraflex solar panels of the MPCV. These panels are superior to conventional solar panels because they rely on a simple hinge and spring mechanism and are more lightweight and flexible resulting in easier deployment. They are equipped with GaInP/GaAs/Ge triple-junction solar cells. Since the DSH and the SEV must be independent, both are equipped with solar arrays. They are dimensioned for the minimum solar flux available, which is about 590 W/m² at Mars. The power demand of the DSH and the SEV are 12.6 kW and 4.6 kW, respectively. Considering the efficiencies and an estimated solar cell life time of three years, the solar panel areas sum up to 134 m² and 50 m², respectively, including a 15% overage in design. Using these values, the masses of the panels become 163 kg and 63 kg, respectively. The required amount of stored energy is calculated to 302 kWh and 110 kWh for the DSH and SEV, respectively.

5. Thermal Control System

The requirements of the Thermal Control System (TCS) are driven by the crew as well as by the systems onboard the spacecraft. Thermal loads have to be transported, distributed, and, eventually, radiated into space. The maximum thermal load is assumed to be at LEO. Thus, the thermal loads are calculated at LEO for those resulting from direct solar radiation, as well as Earth albedo radiation and Earth infrared radiation. In addition, the TCS has to be able to radiate thermal loads from inside the modules. The latter is assumed to equal the electrical power demand of the modules. Based on a few assumptions and the prescribed parameters, the radiator areas must be 81 m² and 26 m² for the DSH and SEV, respectively. Since the required radiator areas are less than half of the module surface areas, all

radiators can be mounted directly onto the outer surfaces of the concerned modules. In order to transfer the heat loads from inside the modules to the radiators outside, liquid cooling loops are installed. The specific radiator system mass is estimated to be 7 kg/m², including peripherals such as pumps, liquids, and valves. The radiator system masses can then be calculated to be 564 kg and 185 kg for the DSH and SEV, respectively.

G. Planetary Protection

Phobos is classified as a Class II (restricted) object, reflecting the NASA Office of Planetary Protection's position that the moon may harbor life from Mars, but does not warrant concern regarding contamination by our mission. A lack of accessible liquid water and extensive radiation exposure at the moon's surface indicate that forward contamination by Earth organisms is not feasible. As such, SEV disposal at Phobos is an acceptable mission outcome if the vehicle is anchored to the surface of the moon.

Phobos's environment does not exclude the endurance of Martian organisms within the accreted material that coats the moon's surface, which means that sample return should be undertaken with highly-elevated precautions. Transporting potentially-pathogenic organisms to Earth, in a process known as back contamination, is a risk which must be mitigated in order to avoid endangering the planet's population. While many potential mitigation technologies have not been demonstrated in space, Vapor Phase Hydrogen Peroxide Sterilization (VPHPS) is a JPL-tested mechanism that can break the chain of back contamination. Upon returning to the DSH with the SEV, sealed metallic sample containers will be processed using VPHPS in a 1.0 m^3 sample transfer airlock. Sterilized containers will be brought into the DSH and immediately transferred and sealed into the freezer assembly. Freezer sample housings capable of return through Earth's atmosphere and impact at the surface must be developed to deal with crew-loss-level contingencies. The exterior adaptors for the DSH airlocks or portions of the incoming SEV may also need to be plasma sterilized by DSH-mounted equipment in the final iteration of the Phobos return planetary protection plan. The procedure proposed above would minimize the possibility of transmission of pathogenic Martian microbes to Earth.

Despite precautions, crew members may be exposed to Phobian material. Extended isolation and monitoring during the return mission would help to partially evaluate their disease status. However, microbial colonization can occur without symptoms, allowing for later pathogen transmission. Hence, extended isolation and monitoring upon return to Earth will also be necessary. Isolation procedures used by the Apollo missions did not afford adequate protection to the human population, meaning that a new method for continuous crew isolation upon return will be required. Continuous isolation could be accomplished by hoisting the Orion capsule into a portable containment unit before the crew opens the hatch. The crew would then be transported to a more comfortable isolation and microgravity rehabilitation facility.

Sealed sample housings from the freezer assembly must be transported from the landing site to a containment and research facility similar to the Lunar Receiving Laboratory and Lunar Sample Laboratory Facility used for Apollo samples. These facilities will need to demonstrate Biosafety Level 4 biological containment capabilities.

H. Risk Assessment

There are considerable risks involved in a mission of this magnitude. In an attempt to categorize the level of risk involved in each element of the mission and program, we constructed matrices for mission and program risks that both identify threats and offer possible mitigation strategies to reduce those threats (Table 9, Appendix 1). Multiple precursor scenarios have been considered to ensure the best chance of gaining usable data to help guide planning of the manned mission. In addition, several scenarios are considered for a new deep space mission. Contamination with Phobian material is considered to be the most severe mission risk, due to the classification of back contamination. Cancellation or failure of the precursor mission is considered to be the most severe program risk.

I. Estimated Cost and Partnerships

A first order cost estimate was made using the high-level Advanced Mission Cost Model. This model determines mission costs based on mass and type of mission or vehicle being designed. Mission vehicles have been separated into the precursor, DSH, SEV, and Orion systems, with individual costs estimated for each. These were defined as a lower-difficulty planetary lander, two very-high-difficulty crewed habitats, and a low-difficulty crewed reentry vehicle, respectively. A first order estimate for the cost of the entire mission is approximately US$30 billion (February, 2014 dollars)[53].

Due to the scale of such an undertaking, this mission is intended to utilize collaboration between international agencies, as well as large-scale government and commercial cooperation. Primary responsibility will be maintained by NASA, including the development of NTP technology and overall mission administration, while the extensive utiliza-

Table 9. Risk Matrices
(Explanations in Appendix 1)

Mission Risk

	Insignificant *(Nuisance)*	***Minor*** *(Minor Contingency)*	***Moderate*** *(Major Contingency)*	***Major*** *(Crew Safety Risk)*	***Severe*** *(Crew Fatality)*
Almost Certain	A27				
Likely		A20, A22			
Possible		A25, A26	A2, A23, A24, A7*		A31
Unlikely		A4	A1, A3, A9, A11, A29	A14, A15, A16, A32, A33	A12, A34
Rare	A28	A6, A18, A19	A5, A17		A8, A10, A13, A30

Program Risk

	Insignificant *(Changes in plans)*	***Minor*** *(Minor delay/ Descope)*	***Moderate*** *(Minor delay/ Redesign)*	***Major*** *(Major delay/ Redesign)*	***Severe*** *(Mission cancellation)*
Almost Certain				B1, B2	
Likely				B3	
Possible				B4, B5	B6
Unlikely			B7		B8
Rare					

tion of SpaceX resources would position them as the primary commercial counterpart. External government agencies ESA and JAXA may assist with the development of scientific tools and specialist vehicles such as the SEV, with the responsibility assumed to be distributed as 50% NASA, 25% ESA and 25% JAXA. Collaboration with each of these partners is expected to decrease the total mission preparation timeline through concurrent development. Additionally, the involvement with agencies other than those in the US allows a broader resource pool when considering operational infrastructure such as communication ground stations and launch facilities.

V. Conclusion

Various aspects of a manned mission to Phobos were studied in detail and presented in this paper. The analysis successfully demonstrated the feasibility of such a mission in about a 25-year time frame. The undertaking of such a mission has a far-reaching impact on our understanding of the solar system, as well as the development of new technologies for use on Earth and in space. The Asaph-1 mission would be the farthest man has ever travelled into the solar system. Given the number and nature of modules left behind in the Martian system after the crew has departed, this mission has been developed to maintain a presence and a measure of operational capability for years to come. This capacity may be utilized in the form of further observation from the PDS, or further exploration using the SEV module in a similar fashion to current assets such as the MSL. The ability to monitor another planetary system such as Mars after the crew has departed may also provide insight into manned mission impacts in different environments that may assist in future mission planning. The operating conditions of the Asaph-1 mission are similar to that of missions to Near Earth Objects; hence the results from this program may also assist in the development of parallel programs under NASA's current agenda. This mission provides an excellent opportunity to begin searching for outpost locations for ISRU. As is clearly evident from the mission architecture, a majority of the launches are assigned to transporting the necessary fuel from Earth to its destination. If bodies such as Phobos and Deimos prove to contain resources such as water or hydrocarbons which may be feasibly extracted, the opportunity to establish fuel outposts would provide a strong basis for exploration farther out into the solar system.

Finally, the gains from a mission of this scale extend beyond science and technology. The grand scale of this mission both mandates and also facilitates collaboration between the space-faring nations of the world. A small fraction of the payload space may also be reserved for student experiments (e.g., ChipSats) which will provide students an

opportunity to become directly involved in the mission. Modern communication networks will afford the opportunity to broadcast mission events to a much wider audience than did the Apollo missions. The advancement in scientific knowledge and space technology, coupled with the growth in the field of STEM education, makes this mission a serious, lucrative, and momentous prospect for the future.

Appendix 1

Explanation for Mission Risk Matrix

A1	Precursor Mission fails to reach Phobos	New precursor mission launched before crew launch window closes.
A2	PE fails to reach and/or to study Phobos	DE could be repurposed to study Phobos.
A3	DE and PE fail to reach and/or to study Phobos	Precursor EVA studies partially compromised, PDS able to perform some preliminary studies of Phobos. Manned mission capability for successful EVA to be assessed.
A4	PDS fails to study Phobos or Deimos	Precursor EVA studies and science studies partially compromised, PE and DE still able to perform some preliminary studies. Manned mission capability for successful EVA will be assessed.
A5	PDS, DE, PE fail to reach and study Phobos	Precursor EVA studies completely compromised. Mission and EVA goals will be reassessed based on future Mars missions unrelated to initial precursor mission.
A6	Communications for main mission compromised	PDS in place to add communication capabilities for Crew mission with Earth, communication redundancy in place using other Mars orbiters. High TRL for communications.
A7	Initial Launch catastrophically fails with Nuclear propulsion cargo	Requires launch site with sufficient safety radius. Active monitoring of radioactivity at launch site. *Tentative place in matrix: requires further testing because it is a new, emerging technology.
A8	Initial Launch catastrophic failure with crew aboard	Orion abort capability in place.
A9	Failure or loss of cargo during launch in upper atmosphere	Requires launch site with sufficient safety radius.
A10	During launch contamination or loss of crew's atmosphere	Orion abort capability in place.
A11	Staging failure in orbit	Send rescue mission depending on severity of situation. Nuclear propulsion stages and cargo options will be assessed accordingly.
A12	Return propulsion capabilities compromised	Rescue mission will be assessed based on severity of failure and the remaining supplies available.
A13	MPCV reentry compromised	Difficult reentry rescue capabilities.
A14	Minor fire on MS or SEV	Fire extinguisher available on both MS and SEV.
A15	Life support in habitat compromised	High TRL, but redundancies will be in place.
A16	Life support in SEV compromised	EVA capabilities on Phobos compromised.
A17	Disease because of food or water contamination	High TRL but will be monitored.
A18	Crew psychology compromised	Careful pre-screening of crew, large habitable volume, communication with Earth through outreach programs.
A19	Crew physiological status compromised	Follow ISS-protocol, depending on severity of compromise.
A20	SPE in deep space in MS	Proximity to sun is limited, radiation shielding in place.
A21	SPE while in SEV	Shielding in place.
A22	SPE during EVA	SPE warning system in place. EVA activity rescheduled to avoid exposure.
A23	Micrometeorite impact of MS in deep space	Unscheduled EVA options.
A24	Micrometeorite impact of SEV in deep space	Scheduled and unscheduled EVA options.
A25	Minor failure of external mechanisms of MS	Unscheduled EVA options.
A26	Minor failure of external mechanisms of SEV while separate from MS	Scheduled and unscheduled EVA options.
A27	Minor failure of internal mechanisms	Tools and spares included in provisions.
A28	Ka-band system failure on DSH during cruise	Use Ka-band system on SEV for cruise, evaluate options for communication during Phobos visit.

Appendix 1 (cont.)

A29	UHF system failure on either DSH or SEV during Phobos visit	EVA capabilities on Phobos compromised. Re-orient spacecraft to establish Ka-band communication.
A30	Crew illness (Infectious, fatal or non-fatal) during return to Earth with Phobian exposure implications	Planetary protection dictates no return for crew and total loss.
A31	Overt exposure to Phobian material	In-transit isolation and monitoring, with potential for prohibited return.
A32	One or more wrong interplanetary burns that result in an incorrect trajectory that misses either Mars or Earth	Propellant reserves for minor trajectory corrections.
A33	Poor prediction of atmospheric conditions or other anomalies during reentry	Provide an abort option into an Earth-captured orbit and have a rescue mission launch ready during the reentry maneuver.
A34	Failure of heat shield upon reentry	Inspect heat shield prior to reentry using cameras and circuitry, make repairs as necessary.

Explanation for Program Risk Matrix

B1	Political delay of precursor mission	Precursor EVA studies completely compromised. Mission and EVA goals will be reassessed based on future Mars missions unrelated to initial precursor mission.
B2	Political delay of main mission	If opportunity for 2033 launch window is lost, backup for 2035 could be used; beyond that mission cancellation may be unavoidable.
B3	Cost growth	Use commercially available solutions where available. External audits and reviews.
B4	Mass growth	Use commercially available solutions where available.
B5	Phobos found to be unsuitable for EVA	Deimos option could be evaluated since precursor mission would have data for that. Propulsion to arrive at Deimos would be available since it is in an easier orbit to reach, and there would be adequate time to adjust propellant budget.
B6	Precursor cancellation	Precursor studies completely compromised. Mission and EVA goals will be reassessed based on future Mars missions unrelated to intital precursor mission.
B7	Precursor partial failure	Precursor studies and science studies partially compromised. If PE, DE, or DPS still able to perform some preliminary studies, manned mission capability for successful EVA will be assessed.
B8	Precursor complete failure	Precursor studies completely compromised. Mission and EVA goals will be reassessed based on future Mars missions unrelated to initial precursor mission.

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