CALTECH SPACE CHALLENGE 2013

ASAPH – 1

Design Proposal for a Human Mission to Phobos

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Executive Summary

Asaph 1 ("the mission") will be the first mission to place a human on Phobos, one of two Martian moons. Asaph 1 will be conducted in two phases over a period of approximately 8 years. Phase 1, beginning in 2026, will act as a precursor. It will deploy one orbiter and two probes that will gather crucial data about the composition and surface characteristics of both moons that will be vital for the next phase of the mission. Phase 2, beginning in April 2033, will be when a crew of three humans and their supplies are launched to Phobos – the furthest from Earth that any human has ever traveled. This mission's prime objective will be to retrieve geological samples from Phobos, and will be enabled by a vehicle stack comprising of a nuclear thermal rocket-based propulsion stage, an International Space Station-derived Deep Space Habitat (DSH), a Space Exploration Vehicle (SEV) for sample retrieval, and the Orion Multi-Purpose Crew Vehicle (MPCV). Together, this stack is called the "Mothership". The Mothership will be assembled in low Earth orbit (LEO) throughout 2032-2033, prior to the commencement of its six month journey to the Martian system. Prior to Trans-Earth Injection from Mars, the SEV will be left in Martian orbit for post-mission science. The crew will transfer geological samples to the DSH prior to leaving the Martian system, and then to the Orion MPCV prior to its safe reentry and splashdown in the Pacific Ocean in July 2034.

The goals of the mission can be divided into two categories: planetary science objectives and biological science objectives. For planetary science, the mission will determine the composition, age, and origin of both Martian moons, which will supplement our understanding of the origins of our solar system; in particular, how small bodies form and how they relate to their host planet and/or other bodies in the solar system. If water ice or other hydrous compounds are discovered, these in-situ resources could become useful raw materials for future space exploration and long-term human habitation in space. Discovery of water could also further our understanding of the origins of life in our solar system. For biological science, the mission will address vital questions regarding human physiology, psychology, and performance during deep space missions. The planned series of scientific investigations aim to uncover fundamental principles underlying human adaptation to the deep space environment, while refining our current understanding of this challenging process. Bioscience research on Asaph 1 will pave the way for an enduring human presence beyond Low Earth Orbit.

This report is organized around the five challenge questions posed by the Caltech Space Challenge committee.

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List of Acronyms

4BMS	4-Bed Molecular Sieve
AES	Air Evaporation System
ANITA	Analyzing Interferometer for Ambient Air
BCBF	Body Centered Body Fixed
CHX	Condensing Heat Exchanger
CM	Command Module
DCS	Decompression Sickness
DE	Deimos Explorer
DSH	Deep Space Habitat
EAWG	Exploration Atmospheres Working Group
ECG	Electrocardiogram
ECLSS	Environmental Control and Life Support System
ELISSA	Environment for Life Support Systems Simulation and Analysis
EMU	Extravehicular Mobility Unit
EVA	Extra Vehicular Activity
EPS	Electrical Power System
FO	Forward Osmosis
GCR	Galactic Cosmic Rays
GLACIER	General Laboratory Active Cryogenic ISS Experiment Refrigerator
GOES	Geostationary Operational Environment Satellites
HZE	High-Z High-Energy Ionizing Radiation
ISPR	International Standard Payload Racks
ISRU	In-situ Resource Utilization
ISS	International Space Station
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
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
MPD	Mars-Phobos-Deimos System
MPLM	Multi-Purpose Logistics Module
MRO	Mars Reconnaissance Orbiter
MS	Mother Ship
MSL	Mars Science Laboratory
NASA	National Aeronautics and Space Administration
NIAC	NASA Innovative Advanced Concepts
NPP	Nuclear Power Plant
NTR	Nuclear Thermal Rocket

PDS	Phobos Deimos Surveyor
PE	Phobos Explorer
PEFC	Polymer Electrolyte Membrane Fuel Cell
PLSS	Portable Life Support System
PPM	Parts Per Million
PRSC	Planetary Retrieval of Subsurface Cores
PV	Photovoltaic
RAD	Radiation Assessment Detector
RFCS	Regenerative Fuel Cell System
RTG	Radioisotope Thermoelectric Generator
SD	Solardynamic
SDO	Solar Dynamics Observatory
SEP	Solar Electric Propulsion
SEV	Space Excursion Vehicle
SFWE	Solid Feed Water Electrolysis
SKG	Strategic Knowledge Gap
SM	Service Module
SOHO	Solar and Heliospheric Observatory
SPE	Solar Particle Event
TCC	Trace Contaminant Control
TCS	Thermal Control System
TEPC	Tissue Equivalent Proportional Counter
THC	Temperature and Humidity Control
TRL	Technology Readiness Level
VPCAR	Vapor Phase Catalytic Ammonia Removal

Introduction

Summarizing some basic facts from Murchie et al. (2009) and Hopkins and Pratt (2011), Mars has two moons, Phobos and Deimos. Phobos is the larger, closer moon, with approximate dimensions of 26.8 × 22.4 × 18.4 km; it orbits Mars rapidly with an orbital period of 7.53 hours, at a mean altitude of ~9376 km. Phobos has a low mean density of 1.88 g/cm^3. Deimos is the smaller, more distant moon, with approximate dimensions of 15 × 12.2 × 10.4 km; it orbits Mars with an orbital period of 30.3 hours, at a mean altitude of ~23,460 km. Deimos has an even lower mean density than Phobos at 1.47 g/cm^3. Phobos has an obvious, large crater at its leading edge, called Stickney that contains a smaller crater, called Limtoc. Radiating outward from Stickney crater is a series of whitish streaks and associated troughs or fractures. Thermal IR data suggest some of this whitish material consists of hydrous phyllosilicates. The rest of Phobos is a distinct reddish color and has the appearance of a powdery, dusty surface, though numerous craters are still evident. Deimos has a more uniform powdery surface, and it is apparent that ancient craters have been almost completely in-filled by thick, fine-grained regolithic material. On Phobos, there is at least one large (thousands of m^3) monolithic block of uncertain origin, but that may have been ejected from the Martian surface.

The origin of the Martian moons is still controversial. Phobos and Deimos both have much in common with carbonaceous chondrites, with spectra, albedo, and density very similar to those of C- or D-type asteroids. Based on their similarity with each other and with the main-belt asteroids, the prevailing hypothesis is that both moons are captured main-belt asteroids. However, both moons have nearly-circular orbits which lie almost exactly in the equatorial plane of Mars; hence a capture origin requires a mechanism for circularizing the initially highly-eccentric orbit and adjusting its inclination into the equatorial plane. It is unclear how this could be achieved, as the current Mars atmosphere is apparently too thin to capture a Phobos-sized object by atmospheric braking. Such capture might have occurred if the original body was a binary asteroid that separated under Martian tidal forces. Other competing hypotheses for the formation of the moons are: (1) They are remnant debris left over from the Martian accretionary process, (2) They are second-generation Solar System objects that coalesced in orbit after Mars formed, rather than forming concurrently out of the same birth cloud as Mars, (3) They are two of many small bodies that were ejected from the Martian surface by collision from a large bolide, (4) They are volcanic ejecta from Martian volcanoes, such as Olympus Mons, (5) They are captured cometary nuclei.

Because of the larger size, the reddish and whitish colors on the surface that indicate different rock types, the streaks and troughs that radiate from Stickney crater, the suggestion of a thinner powdery regolith cover, and the presence of at least one large monolith, we are convinced that Phobos offers the greatest scientific returns for a manned mission to a Martian moon and therefore should be the focus of such a mission. Nevertheless, a concurrent study of Deimos' composition and structure via remote and/or robotic experimentation will provide vital information about the differences between the two moons and may shed additional light on the formation of the moons.

Because of the scope and duration of a mission to a Martian moon, technologies critical for future Martian missions, including safe habitation in deep space, nuclear propulsion, and tele-robotic operation, will be demonstrated by this mission.

I) Science and Technology Objectives (Question 1)

Mission Statement

Asaph-1 is a mission designed to land humans on a Martian moon, either Phobos or Deimos, and return them along with a sample, safely to the Earth; with a launch date no later than January 1, 2041.

Primary Science Goals

- To understand the long-term effects of deep space exploration on human physiology and psychology
- To understand the origin of our solar system and its evolution by determining the composition and origin of the moons, and understanding their similarity with asteroids, if any
- To determine the presence of water and the distribution of hydrous compounds on Phobos and Deimos

Secondary Science Goals

- To perform mass-density modeling of both moons (minimum resolution of 100 meters in all three dimensions, to a sub-surface depth of at least five kilometers)
- To capture visible imagery of both moons from orbit (minimum resolution of one meter pixels)
- To generate three-dimensional topographic maps (digital elevation models) of at least 80% of Phobos and Deimos (minimum hundred-meter resolution)
- To investigate the chemical composition of the regolith on both Phobos and Deimos
- To assess potential for in-situ resource utilization (ISRU) by future missions (especially for water and metals)
- To quantitatively assess flux of radiation and high energy particles in the Mars-Phobos-Deimos (MPD) system
- To measure the magnetic fields at both Phobos and Deimos

Minimum Mission Success Criteria

- Complete a safe two-way trip for astronauts to Phobos
- Land a minimum of one astronaut on the surface of Phobos
- Have astronaut(s) perform extra-vehicular activity (EVA) and retrieve one geological sample from Phobos

Full Mission Success Criteria

- Impact the surface of both moons at 4 different locations on each moon using penetrators to determine depth and competence of surface regolith (precursor mission)
- Place landers on both moons to carry on ‘precursor science’ to assist in planning future human exploration of the MPD system (precursor mission)
- Station science instruments to remotely (and autonomously) gather environmental data on Phobos and Deimos
- Obtain multiple rock, dust and core samples from at least two geologically diverse locations on Phobos
- Deploy outreach science payloads on the surface of Phobos

Technological Requirements for Mission Success

- A safe habitat needs to be designed for astronauts to survive for about 500 days in deep space. This includes radiation shielding, smart utilization of resources and enough space to provide comfortable living conditions for astronauts.
- Efficient propulsion systems that provide a reasonable level of thrust at high I_{sp} are required to take the crew and supplies to the Martian system and back in a reasonable time period.
- Multiple launches are required to transport all the required modules to the Martian system.
- The ability to safely abort the mission at various stages needs to be designed.
- A precursor mission is required to characterize the surface of Phobos in order to make any improvements in design necessary to make the mission a success.

Strategic Knowledge Gaps

A large portion of our mission utilizes technology that is already in existence. However, there exist a few knowledge gaps that we would like to address prior to the human mission being undertaken.

- The surface properties of Phobos and Deimos such as regolith thickness and strength are not known. It is important that these two properties are characterized before a human landing.
- Advanced propulsion concepts such as Nuclear Thermal Propulsion (NTR) need to be developed and tested in flight before the mission.
- The deep space vehicles need to be tested for survival in deep space conditions prior to usage by astronauts.
- Custom fairings need to be developed in order to accommodate high volume payloads on launch vehicles.
- Methods for faster turnaround times for launch vehicles need to be developed in order to allow for more launches in a short period of time. This allows for faster assembly of deep space cargo in LEO before a long mission is undertaken.
- Better operational coordination between different launch sites around the world needs to be researched in order to allow for launches from different sites over a short duration.
- On-orbit assembly on a large scale needs to be perfected through research and orbital testing.

In the event of a full mission success, we will have obtained high resolution images of Phobos and Deimos and will have learned the chemical composition of the surface of Phobos and Deimos, based on results from the remote mission. The experiments will also provide information about the depth and surface strength of the regolith at various locations on both moons, differences in chemical composition between the whitish streaks and reddish background areas on Phobos as well as refinements of the orbital characteristics of Phobos and Deimos. Further, we will be able to characterize the radiation environment in the Martian system. All of these properties will be monitored over a longtime span, providing us with trends and variations that will supplement our understanding of the MPD system.

The moons of Mars are a great option for human exploration prior to the exploration of Mars because they provide a test-bed for many essential technologies that are required for a mission to Mars, while negating the need to address complex issues such as Martian atmospheric entry of very large payloads and the prevention of forward contamination. The moons are a good place to investigate the potential for ISRU, which is an essential component for long-duration missions and possible colonization of Mars. Aside from these advantages, the moons also offer a great way 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 planets. Many of these studies are best conducted with a hands-on approach rather than from a distance. A human mission enables us to perform in-situ studies and also to bring back samples so they can be analyzed on Earth with all the resources present, without the many constraints placed on us by deep space. Therefore, we feel that it is essential for humans to visit the moons of Mars to further the frontiers of our knowledge about the solar system.

II) Reasons for the Proposed Work (Question 2)

In addition to studying the nature of the MPD system, the undertaking of this mission will significantly advance the development of technology for and experience with long-duration human spaceflight beyond Low Earth Orbit (LEO). These technologies, insights, and scientific data collected are key necessities for the future of human missions to Mars, as well as future exploration of the solar system and, eventually, interstellar space. In addition to the human implications, there are fundamental science questions concerning solar system origins that can be addressed that can guide and constrain future robotic missions.

Knowledge Needed Before a Human Mission to Mars

To meet the mission objectives, a robotic precursor mission is required. The rationale for this is to understand and determine the area(s) where the astronauts will be performing EVAs on the main mission. The precursor mission will provide important data to be used to optimize the design of certain components, such as the anchoring system used by the Space Excursion Vehicle (SEV) and the core-drilling system deployed by the astronauts, prior to embarking on the main mission. The precursor mission will begin transmitting data to Earth approximately 6 years before the manned mission embarks to allow adequate time for modifications to the engineering design to be made.

Science motivation

There are numerous science questions the answers to which may be illuminated by a mission of this scope. Some of the fundamental science questions include:

Planetary Science

(1) What are the composition(s) and age(s) of the Martian moons?

Currently, we have only very limited visible imagery and IR spectroscopic data from the moons that indicate they may be comprised of at least two compositions--reddish material (of possible Martian origin) and whitish material (of possible bedrock origin). In addition, low reflectivity and possibly the presence of hydrous compounds have been observed, and there is some indication that the moons could be C-type carbonaceous chondrites (Murchie et al., 2009).

(2) What is the origin of the Martian moons? Do they share the same origin?

Because of their small size, low reflectivity, and general proximity to the asteroid belt, both moons are hypothesized to be captured main-belt asteroids. However, this hypothesis remains tenuous until the compositions and ages of the moons are better constrained. Other hypotheses for their origin(s) include: remnant material left over from planetary accretion, impact ejecta from the Martian surface, eruptive ejecta from one or more Martian volcanoes, captured comets.

(3) What does the origin of the moons suggest about the planetary history of Mars?

An accretionary or ejecta origin for the moons suggests a very different planetary history for Mars than that of captured asteroids. Determining the ages of the moons will help to constrain early Martian history.

(4) Is there water in any form on either of the moons?

Preliminary IR spectroscopy indicates the presence of hydrous compounds (hydroxylated minerals) in the whitish material on Phobos. In the interest of ISRU and future human exploration and habitation in the solar system, additional data must be acquired to support this initial finding.

(5) Are there any chemical compounds on the moons that could indicate the presence of life?

This is as yet unknown. It has been stated that Phobos and Deimos are currently incapable of supporting life, but the possibility remains that they were once able to support life. The samples returned from Phobos will be analyzed for organic compounds and isotopic compositions to determine the possibility that life once existed on the moon.

(6) What are the surface characteristics of the Martian moons, especially with regard to landing humans on the surface?

Craters on both moons are partially to completely filled with what appears to be powdery, fine-grained regolith. Much more surface interaction data is required in order to make informed decisions about landing a spacecraft.

Biological Science

(1) 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 factors on humans.

(2) What is the exact form and degree of radiation exposure during the mission profile?

Data obtained from the Mars Science Laboratory (MSL) cruise phase, the Asaph 1 precursor, and the Asaph 1 manned mission detectors will better define radiation expectations in anticipation for a human mission to the surface of Mars.

III) The Asaph Mission, How? (Question 3)

Overview of mission plan

The Asaph mission is divided into two phases: the precursor phase, aimed to provide data on the topography and composition of Phobos and Deimos to assist in planning and technology development for the manned mission; and the primary phase, in which the crew will travel to the Martian system to conduct their exploration and sample retrieval tasks.

In order to provide sufficient time to study the Martian moons and determine an operational schedule, complete with confirmation of landing sites and specific equipment required during operation (such as anchors specific to the moon composition), the precursor mission will be sent several years in advance of crew deployment, as discussed in the following sections.

The overarching mission architecture has been developed in order to leave a legacy component in active operation within the Martian system, acting to both increase the cost-specific benefit of the mission and to provide infrastructure for future expeditions to Mars and beyond.

Phase 1, Precursor Mission

Mission Summary

Phase I, the precursor mission, will be a robotic survey mission that sets up a remote-sensing communications relay orbiter around Mars and gathers data on both of the moons. Phase I will be launched from Earth in 2026 in a Falcon 9, based on trades described in further detail in the Precursor Trajectory and Propulsion Section. It will spiral to Mars using solar-electric propulsion over the course of 2 years. Upon reaching Mars in 2028, science payloads will deploy as described in Phobos and Deimos Operations. The landers will continue to take science data over the course of several years, until their power supplies run out; while the orbiter will make remote sensing observations before, during, and probably after the primary mission.

Modules

The precursor mission will contain the following modules:

1. Phobos-Deimos Surveyor (PDS). This orbiter contains a remote-sensing science payload and a communications relay system.
2. Phobos Explorer (PE). This includes a Phobos lander and an impactor package.
3. Deimos Explorer (DE). This package is identical to the PE except it will target Deimos, or, in the event that the PE is unsuccessful, it will be a back-up to target Phobos.

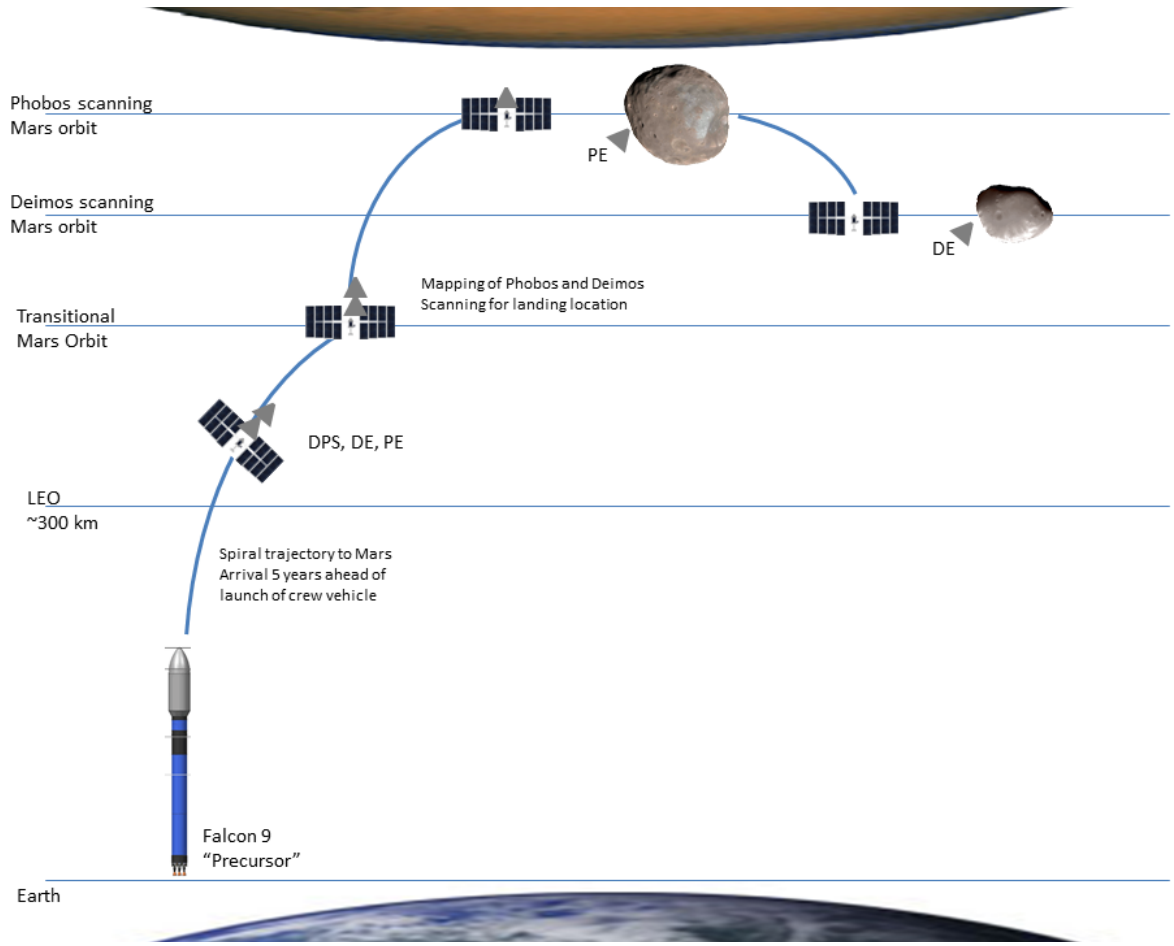


Figure 1: Bat chart of the precursor mission

The PDS will have some 7 remote-sensing instruments that will characterize Phobos and Deimos, as detailed in the Science section. The PDS will act as a critical communication relay system, relaying data and commands between the landers and Earth. It will remain in place for the primary mission, supporting communications with the astronauts at that time.

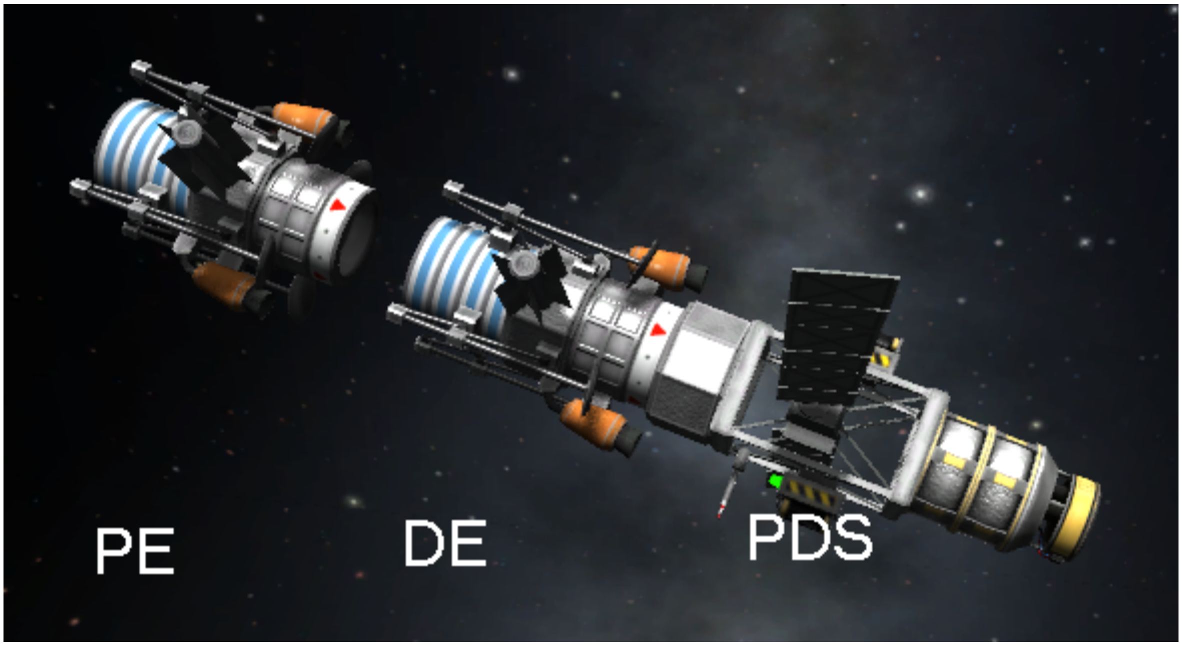


Figure 2: Artist's impression of the precursor mission: Deimos Explorer (DE), Phobos Explorer (PE), Phobos-Deimos Surveyor (PDS) satellite. PDS is equipped with solar-electric propulsion.

Motivation

The goals of the precursor mission, in order of importance, are as follows:

1. Determine if humans can safely land on Phobos during the primary mission, or on Deimos in the event that Phobos is not feasible.
2. Set up a communications relay system that will facilitate the primary mission.
3. Acquire important surface data regarding its nature and composition to plan for a landing of the primary mission.
4. 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 through four impactor experiments at preselected sites, in-situ sampling and analysis conducted remotely using an immobile lander, and a combination of remote observations from the Surveyor satellite (Table 1). In-situ sampling sites will be determined based on the findings from initial remote sensing surveys conducted by the Surveyor satellite.

Science

Impactor sites were selected in order to target sites of geologic interest, sites where a future mission might land, and other widely-spaced sites to learn more about the distribution of the surface characteristics. On Phobos, site 'A' is a priority site because it is located on the highlands east of Stickney crater ("Stickney highlands") where there is also red (possibly Martian) and white (possibly bedrock) material. If the surface characteristics at this site are favorable, this would be the top-choice landing site during the principal part of the mission. Site 'B' is located on the central uplift in the middle of Stickney crater to determine if this could be a good site for a manned lander. Sites 'C' and 'D' are located in both the northern and southern hemisphere to impact flat “powdery” sites that appear to have deep, fine-grained regolithic cover.

On Deimos, the fine-grained regolithic cover appears to be thicker than on Phobos, and it is unclear if a conventional landing is possible there. From a geologic standpoint, the bright material along a mostly-buried crater at site ‘D’ is probably 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.

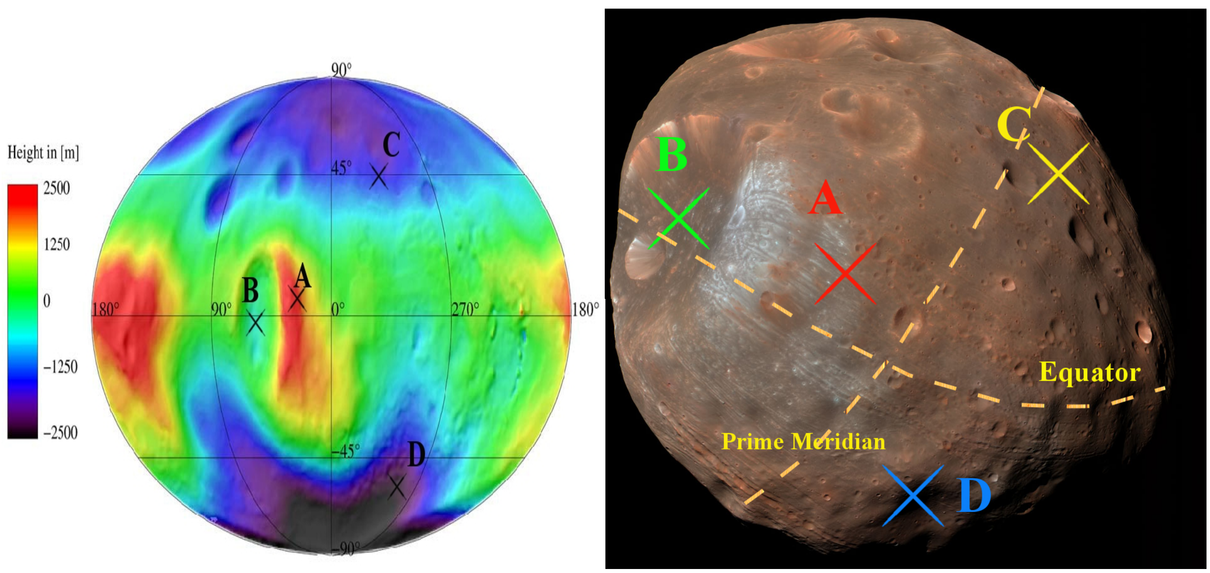


Figure 3: (left) Topographic map of Phobos with elevation indicated by a color ramp. Four sites (A-D) to deploy impactors are shown. Scale is ~0.24 km per 1° longitude at the equator. The point at 0°E,0°N is the nadir. Image modified after Wählisch et al. (2010). (right) Color image of Phobos showing the same points as in the left image. Imagery from the NASA/JPL/University of Arizona HiRISE project (image PSP_007769_9010; http://hirise.lpl.arizona.edu/phobos.php).

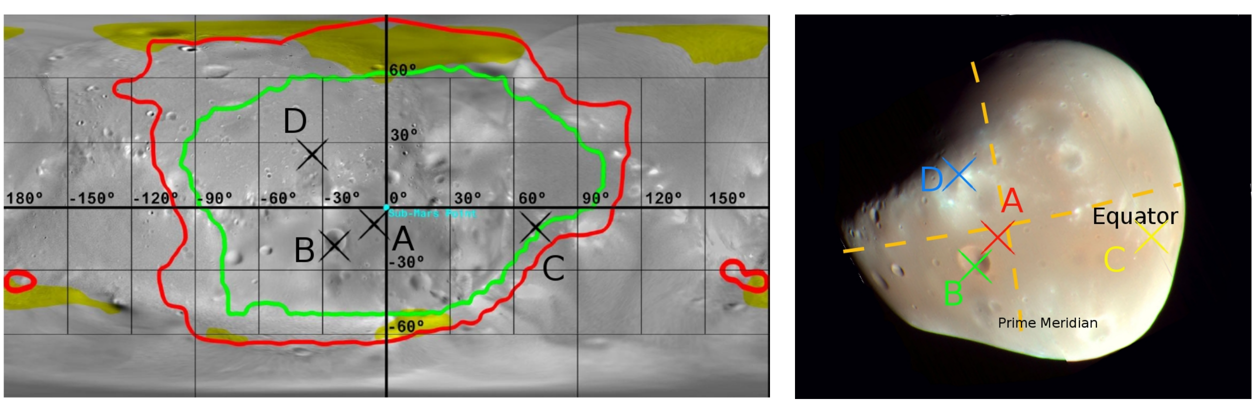


Figure 4: (left) Grayscale image of Deimos with four sites (A-D) to deploy impactors (modified after Hopkins and Pratt, 2011). Scale is ~0.13 km per 1° longitude at the equator. The point at 0°E,0°N is the

nadir. Green line represents extent of moon in full-view of Mars; red line is extent of moon with only a partial-view of Mars. (right) Color image of Deimos showing the same points as in the left image. Imagery from the NASA/JPL/University of Arizona HiRISE project (image ESP_012065_9000; http://hirise.lpl.arizona.edu/deimos.php).

The surface characteristics revealed by the four impactor tests will be a strong driver in determining how and where the lander is deployed, with the priority sites being site 'A' on Phobos and site 'D' on Deimos. If these sites prove not to be amenable for a lander, the other sites will be considered.

To reduce complexity, DE and PE will be identical lander and impactor packages (Table 2). 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, as described by Ulamec and Biele (2006). Philae has a mass of 98 kg, with approximately 27 kg of science payload. The science payload on this precursor lander is estimated to be about 40 kg, based on the instruments listed in the Science section and the desired addition of a Phoenix-like sampling arm. Thus, the lander must be scaled up from Philae to approximately 145 kg. This is an acceptable mass, since studies show similar asteroid landers can be scaled up to 150 kg (Ulamec and Biele, 2006). Unlike Philae, which is solar powered, the lander will use a Radioisotope Thermoelectric Generator (RTG) in order to obtain science results from sometimes-shadowed landing sites.

The Philae lander uses harpoons to anchor to the asteroid surface. While adequate for a rocky, solid surface, this landing strategy may fail in the powdery regolith we expect to encounter on both Phobos and Deimos. Other anchoring options should be investigated, such as those described in further detail in the Main Mission Landing and Anchoring section. In addition, these landers will require their own propulsion system, unlike the ballistically-launched Philae. Such a propulsion system can be used to lower the craft to the body surface more gently, thereby preventing damage to the craft or instrumentation. Finally, in the absence of any sampling or scooping activities, the extremely low gravity of the moons will eventually pull a lander back to its surface.

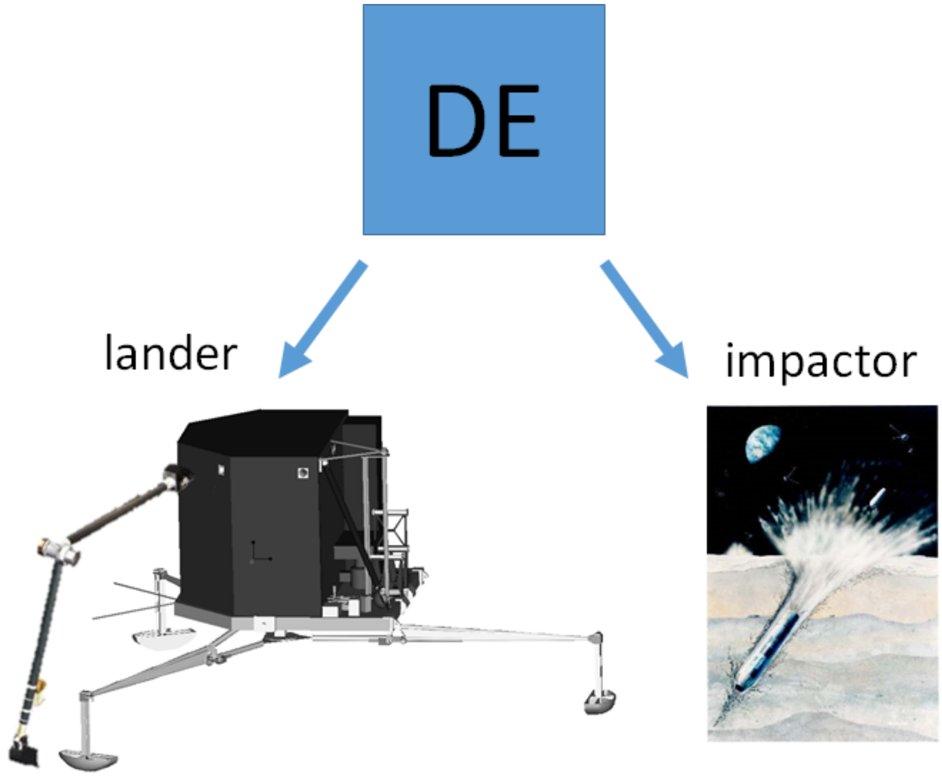


Figure 5: The Deimos Explorer (DE) separates into an impactor (activates first) and an immobile lander (activates second).

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 1) the safety of the astronauts and 2) 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. Determination of the nature of the regolith (uppermost, loose soil) allows assessment of the mechanical and chemical properties of the surface. Identifying the strength and porosity of the surface provides critical information with regard to docking or anchoring maneuvers during the manned component of the mission. Additionally, studies of the flux of interstellar material and radiation levels will help develop shielding techniques.

We have designed three unique science instrument suites (Surveyor, Explorer, and Expedition) to achieve the aforementioned science objectives during the mission timeline. Tables 1-3 detail the science objectives each suite will complete along with instrument details.

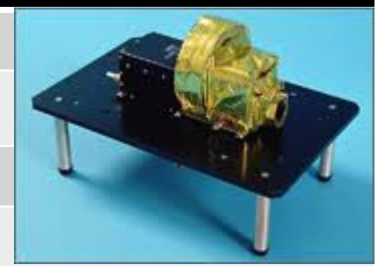
The Surveyor suite is comprised of a number of heritage spectrometers and cameras. Each instrument is configured to investigate regolith properties remotely from orbit. Other instruments in the Surveyor suite will provide valuable data on the topography of Phobos and Deimos, the flux of interplanetary material to the surface of Phobos and Deimos, and the strength of the magnetic fields on the moons. These data will provide critical details about the moons and Mars. Details of the capabilities of each instrument are listed in Table 1.

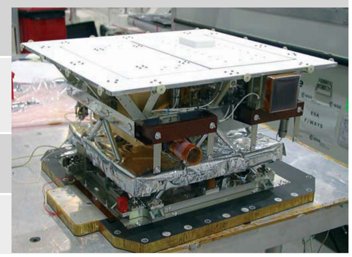
The Explorer Suite is modeled after instruments from past NASA and JAXA missions. Table 2 details the instruments and the science objectives that will be completed. The penetrometer device within the impactor package is modeled after the piezoelectric sensing element used in the Huygens probe. It was uniquely calibrated to withstand cryogenic temperatures and future development will allow impact velocities of 300 m/s. The voltammetry, spectroscopy, diffraction and fluorescence instruments (Wet Chemistry Lab, LIBS, and CheMin, respectively) did 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, the micrometeoroid detector used previously on a number of NASA missions to estimate the flux of interstellar material has been modified for a lander spacecraft. Once positioned on the surface of Phobos or Deimos, micrometeoroid detector panels will deploy along the sides of the lander.

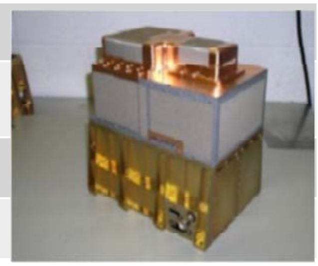
Astronauts will manually deploy the Expedition suite of science instruments during EVA sorties (see the 'Surface and Science Mission Operations' section for more details). Each instrument is listed in Table 3. The PRSC (Planetary Retrieval of Subsurface Cores) and ChipSat instruments are will be developed to interface with the environment of Phobos. These experiments have never been deployed before and are an innovative approach to meet a principal science objective of sample return; they also promote the use of smaller-scale science experiments. On the other hand, seismic and radiation studies will utilize heritage instruments.

Lastly, the team considered setting up a system to return a sample from Deimos by robotic means. However, our primary mission SEV will never enter an orbit from which we could reasonably collect an orbiting Deimos sample; in addition to this complexity, it would cost too much time and delta-V on the part of the SEV to retrieve the sample. Nevertheless, because of the intrinsic scientific value of a sample from Deimos, additional designs should be explored that would make a small, automated sample-return craft that would be self-propelled and would seek the orbit of the SEV.

Table 1: Summary of science objectives and instruments for 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	

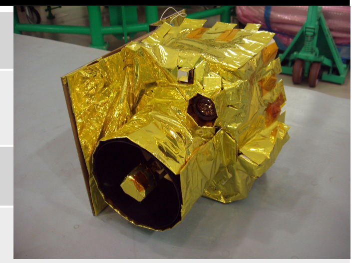
Objective: Remote analysis of surface mineral composition	
Technique: High-resolution imaging spectroscopy	
Instrument Precedent: Visible Infrared Thermal Imaging Spectrometer (Rosetta)	
Priority: High	
Requirements: Mass: 23kg, Power 32W ; Spectral Range: 0.025-5.0µm	

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	

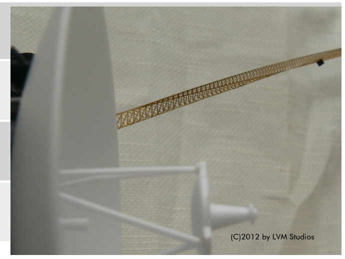
Objective: Remote analysis of surface mineral composition	
Technique: Spectroscopy	
Instrument Precedent: VIS/NIR Spectrometer (Dawn)	
Priority: High	
Requirements: Mass: 15kg, Power 15W; Spectral Range: 1-3µm Resolution	

Table 1: Summary of science objectives and instruments for Surveyor Suite (cont.)

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

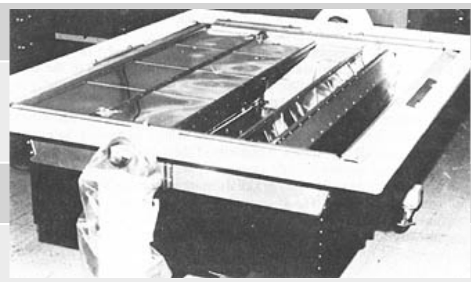
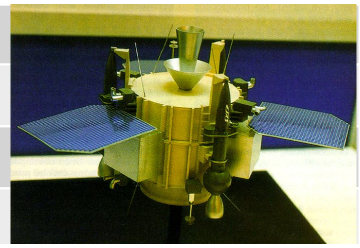
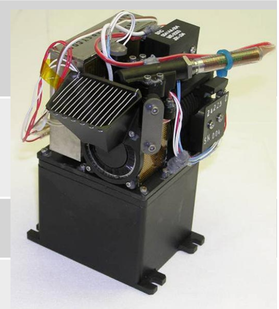


Table 2: Summary of science objectives and instruments for Explorer Suite

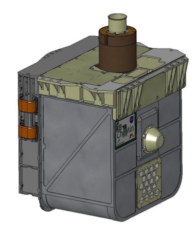
Objective: Determine near-surface thermal and structural properties
Technique: Piezoelectric sensing
Instrument Precedent: Impactor Package, (Lunar-A, Hayabusa)
Priority: High
Requirements: Mass: 180kg, Power 4W



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

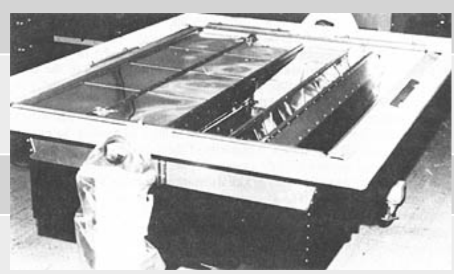
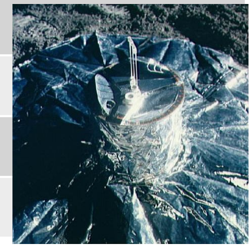


Table 3: Summary of science objectives and instruments for Expedition Suite

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

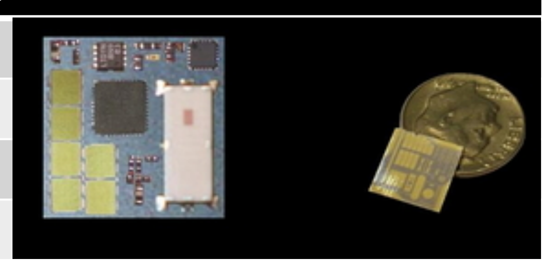


Objective: Laser ranging of Phobos altitude
Technique: Specular Reflection
Instrument Precedent: Laser Ranging Retroreflector (Earth Resources Spacecraft-ESR)
Priority: Low
Requirements: Mass: 3kg, Power N/A

Objective: Collect robust samples of subsurface
Technique: Core Drilling
Instrument Precedent: Planetary retrieval of subsurface coreS [PRSC] (Under Development--*Not Pictured*)
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



Trajectory and Propulsion

The trajectory used for mapping was carefully selected, due to the need to map as much of the surfaces of the moons as possible. For this reason, orbits were chosen that will allow the Phobos-Deimos Surveyor to map over 80% of Phobos' surface, and 50% of Deimos' surface. The PDS will begin in an orbit at an altitude slightly lower than Phobos with an inclination of 20 degrees. This orbit will cause the PDS to overtake Phobos, while gaining coverage of the north and south pole regions. Once Phobos has been overtaken, the PDS will raise its orbit to an altitude above Phobos. This vantage point will allow the PDS to map the opposite side of Phobos. Much care must be taken when raising the PDS orbit in an effort to avoid a Phobos collision with the PDS. If timed carefully, the PDS can successfully perform an orbit raise to view the zenith-pointing side of Phobos. The same maneuver will be performed to map Deimos. A simple diagram of these maneuvers is shown below(Figure 6).

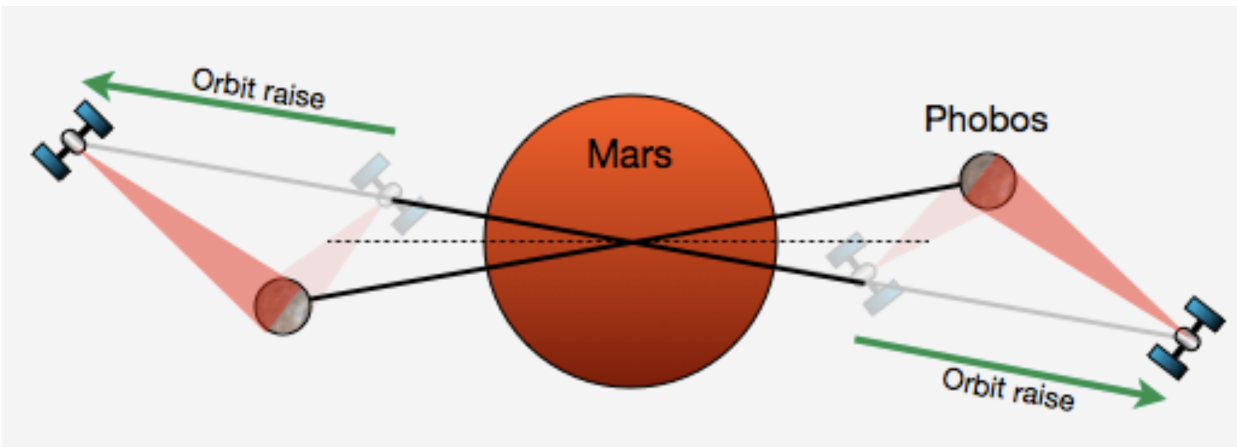


Figure 6: The two orbits used to image over 80% of the surface of Phobos are displayed. The first orbit (grayed) has an altitude of 9000 km, while the second has an altitude of 9700 km.

For the propulsion of the precursor mission, electric propulsion is used to spiral the spacecraft into Mars orbit. The required velocity change is calculated by

$$\Delta v = \sqrt{\mu}\left(\frac{1}{\sqrt{r_0}} - \frac{1}{\sqrt{r_1}}\right),$$

where μ is the gravitational constant of the central body (in our case the Sun), r_0 is the radius of the starting orbit and r_1 of the destination orbit respectively. Though the total velocity change is larger compared to a Hohmann transfer with a short-duration impulsive burn, propellant mass can be reduced due to the high specific impulse of electric propulsion. The required propellant mass is then calculated with the Tsiolkovsky equation

$$m_{pr} = m_0\left(e^{\frac{\Delta v}{v}} - 1\right).$$

The flight time for the transition to Mars was assumed to two years with an array-to-jet efficiency $\eta = 60\%$ for the thruster. The mass of the power source (solar) was approximated to 30 kg/kW. The required mass flow is then calculated by:

$$\dot{m} = \frac{m_{pr}}{t}$$

where t equals the flight duration. Now, the required power is calculated by:

$$P_{req} = \frac{\dot{m}}{2\eta v_e^2}$$

$v_e = g_0 I_{sp}$ is the exit velocity of the burned fuel. Finally, the total mass of the SEP system is approximated by:

$$m_{SEP} = P_{req} \cdot m_{mf,array} + m_{mf,tank} \cdot m_{pr}$$

where $m_{mf,array}$ is equal to the inert mass fraction of the solar array per kW power. The assumptions are summarized in Table 4.

Table 4: Solar Electric Propulsion LEO - Mars assumptions

Flight duration	2 years
Specific impulse, I_{sp}	2000s
Array-to-jet efficiency, η	60%
Inert mass fraction array, $m_{mf,array}$	30 kg/kW
Xenon tank mass, $m_{mf,tank}$	15% of fuel mass

The total mass required in LEO is then in the range of a medium lift-launcher like Falcon 9, Ariane 5 ECA and Delta IV. The total mass in LEO includes the solar electric propulsion and the power source. Falcon 9 is selected due to low-cost and an appropriate margin of around 40% (Table 5).

Table 5: Launch selection precursor mission

Launch mass to LEO	approx. 7 t
Launcher, Performance	Falcon 9, approx. 10t
Margin	approx. 28%
Launcher cost	approx. 60 $M

Communication

The PDS will have communication facilities on the UHF band and the Ka-band to relay data from experiments on board Deimos and Phobos Explorers from the surfaces of Phobos and Deimos. The Phobos and Deimos Explorers also include UHF radios with low gain antennas for communication with PDS. This gives the capability to establish a simple (albeit intermittent) Martian system communication network. The PDS goes into a high orbit around Mars to provide relay facilities when the astronauts arrive in the Martian system.

Phobos and Deimos Operations

The PDS, DE, and PE package will enter an orbit around Mars, near Phobos but at a slight inclination and slightly lower altitude than Phobos, such that the orbiting PDS passes by a different portion of Phobos with each orbit. There, the PE will be dropped off. The PE will separate into the lander and impactor packages. The impactor module will release four rocket-propelled penetrators, equipped with penetrometers, which impact Phobos and are remotely sensed by the orbiting PDS package. As this occurs, the orbiting PDS system slowly transits from a lower altitude than Phobos to one slightly higher than Phobos, observing the impactors in several different locations. Based on what is learned from the impactor tests, the landing site is selected by an Earth-based team. Then, the lander, which contains a wet chemistry lab (with air collector attachment to analyze airborne dust), LIBS, CheMin, and micrometeoroid flux detector, lands in the most appropriate location.

The PDS and DE system then enters a higher-altitude Mars orbit, just below Deimos, where the Deimos Explorer is dropped off. Similarly, this orbit will be slightly inclined from the ecliptic. Like on Phobos, the four impactors, each with a penetrometer, will determine the landing site for the lander. The PDS slowly moves from below Deimos, to trailing it, to a higher altitude orbit, thus covering a large area for communication purposes. One the impact experiments have been performed and remotely sensed by the PDS, the PDS system moves back down to an orbit slightly lower than Phobos, around Mars, thereby maintaining sufficient communications with both landers.

Attitude Determination and Control

In the proposed mission, the attitudinal states of all physical stages are described by three angular variables along X, Y and Z axes. The coordinate frame is always Body-Centered-Body-Fixed (BCBF). A combination of a Star Tracker and a Sun Position Sensor will be used to accurately determine state awareness. Rationale behind the selection was:

1. Non-dependence on moving parts
2. Extremely low mass and volume
3. Starfield view availability for a large fraction of orbits
4. Availability of line-of-sight with Sun during rare Solar saturation

The difference between the desired and measured attitude states is fed into an Attitude Control System which, in turn, physically corrects the attitude. Several strategies can be employed to achieve this. It was decided that all small stages be equipped with Pulsed Plasma Thrusters to achieve this. Rationale behind the selection was:

1. Light weight
2. Small volume
3. Precision attitude control

Primary Mission, Phase 2

Mission Summary

Phase 2, the primary mission, will be a human mission that lands humans on a Martian moon (nominally Phobos) and returns them, along with a sample, safely to Earth. Phase 2 will be launched in several stages from Earth into a LEO, where the different modules will rendezvous to form the mothership (MS).

The MS will use an impulsive propulsion maneuver to reach Mars within six months and stay there for one month. At Mars, the MS will enter a parking orbit and the SEV will approach the surface of Phobos to perform scientific activities as described in a later section. After returning the crew to the mothership, the SEV will return to the surface of Phobos to continue to acquire scientific data over the course of several years. The rest of the MS will leave Mars using another impulsive propulsion maneuver to return to Earth.

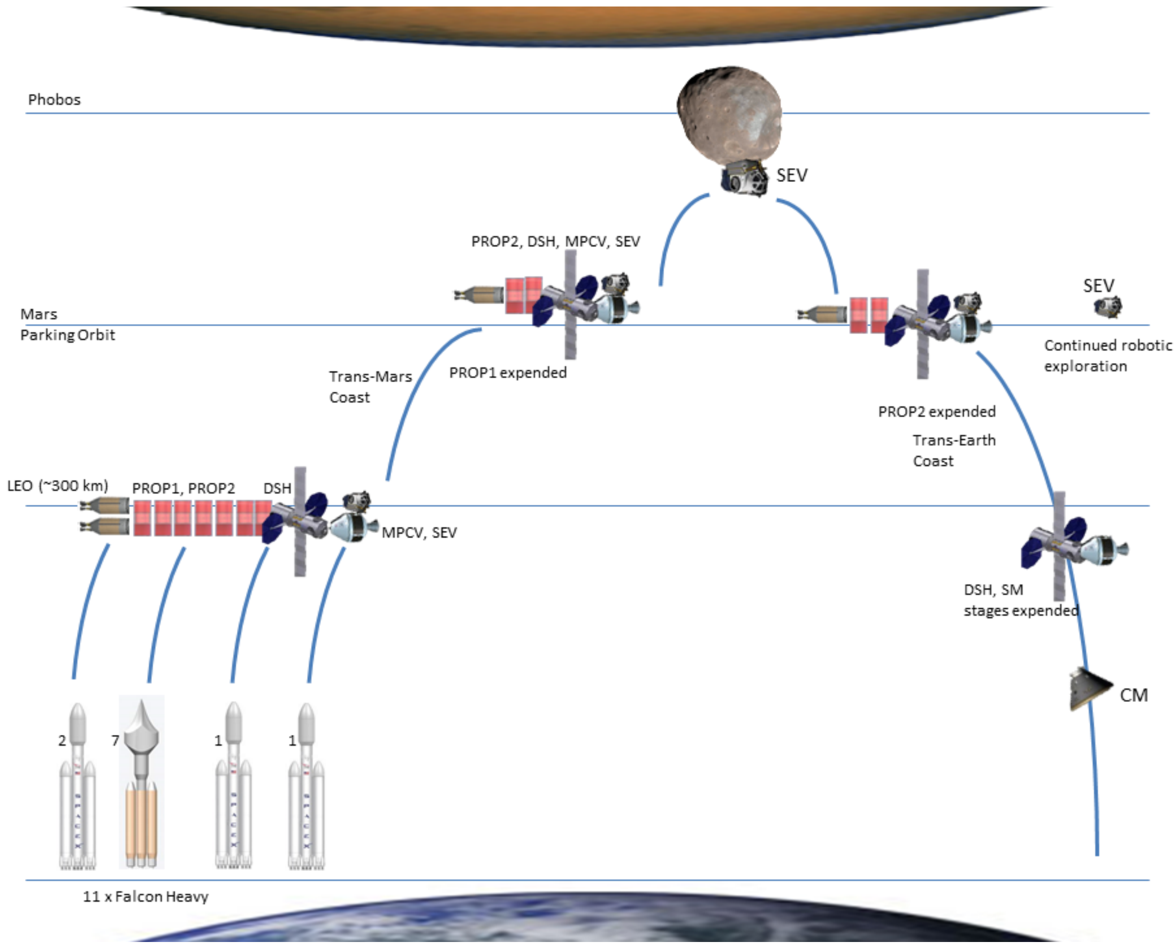


Figure 7: Bat chart of the main mission

Modules

The main mission will contain the following modules:

1. Propulsion Systems 1 and 2 (PROP1 and PROP2). This module contains the nuclear thermal propulsion system including the LH2 tanks.
2. Space Exploration Vehicle (SEV). This vehicle will bring Astronauts from the mothership to the Phobian orbit and back.
3. Deep Space Habitat (DSH). This module provides additional habitable volume for the crew.
4. Orion Command Module (CM).This vehicle is where the crew initially operates during launch, and will be used for the reentry of the crew.

5. Orion Service Module (SM). This vehicle provides propulsion for the CM to dock with the PDS, SEV and DSH after it is launched into LEO.

The CM and SM combine to form the Multi-purpose Crew Vehicle (MPCV).

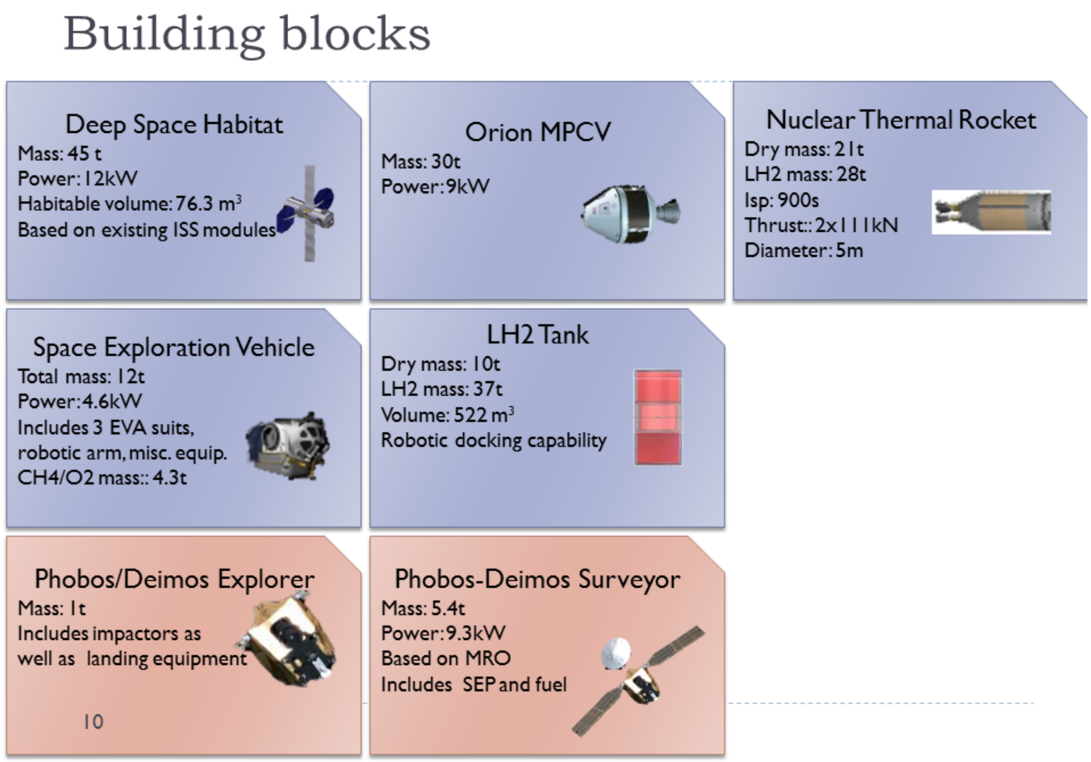


Figure 8: Building blocks of the Asaph precursor and main mission

Trajectory Design

The planned trajectory is designed for an opposition class mission with a round-trip duration of 465 days. Eighteen weeks prior to crew departure, the Propulsion Systems 1 and 2 (PROP1 and PROP2) and the Deep Space Habitat (DSH) will be launched in segments. Since it cannot be assembled as a whole on Earth, PROP1 and PROP2 will be launched first and assembled in LEO. Once the DSH has been launched to LEO and docked to PROP1 and PROP2, human spaceflight will commence. The crew will then be launched with the Space Exploration Vehicle (SEV), Command Module (CM), and Service Module (SM). The crew will enter LEO to rendezvous with PROP1 and PROP2 and DSH, as the final piece of the mothership. Upon successful docking, the mothership will depart Earth in April 2033 for arrival at Mars in October 2033. The mothership will then burn to achieve a Mars orbit insertion (MOI), where it will remain for 30 days before returning to the Earth in November 2033 for Earth arrival in July 2034. The heliocentric round-trip trajectory is displayed in Figure 9.

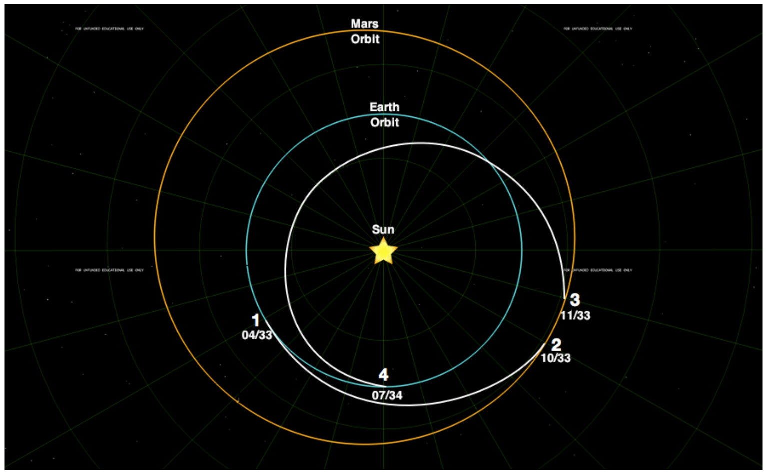


Figure 9: View along the ecliptic plane normal displaying heliocentric trajectories. The Earth-Mars trajectory connects points 1 and 2. The Mars-Earth trajectory connects points 3 and 4.

Once PROP1, PROP2, and the DSH have reached a 300 km low Earth orbit, the crew will be launched with the SEV, CM, and SM on April 7, 2033. Upon successful docking, PROP1 and PROP2 will provide a delta-v of 3.5 km/s 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, displayed in Figure 10.

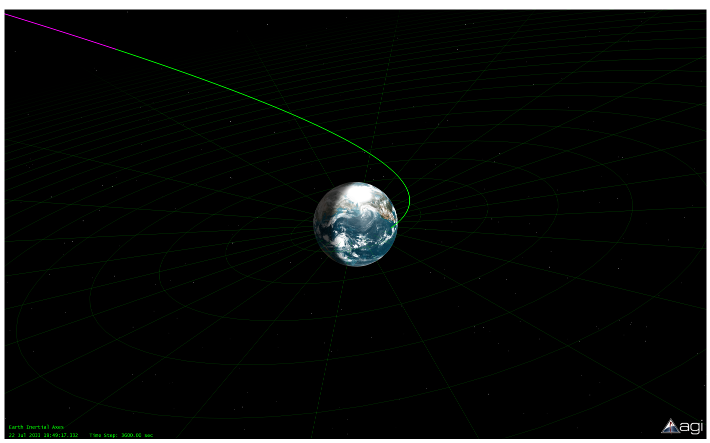


Figure 10: Earth escape trajectory with an outgoing asymptote right ascension of 272°, declination of -23°, and velocity azimuth at periapsis of 90° in the Earth inertial reference frame.

The mothership will then burn to achieve a Mars orbit insertion (MOI) delta-v of 2.2 km/s to enter a 250 x 33,813 km parking orbit around Mars (orbital period of 1 sol) with an inclination of 34°. Once the mothership is in the parking orbit, the crew will remain onboard until the mothership and Phobos have the correct phase for rendezvous. This phasing period could require a minimum duration of 12 hours to a maximum of 14 days. The crew must wait until two criteria are fulfilled. First, the mothership and Phobos must have a phase difference of 180°. The second condition is that the first condition must be met when the mothership is located at the parking orbit apoapsis. When the two conditions are met, the crew will board the SEV and depart for Phobos rendezvous. A bi-elliptic Hohmann transfer with apse rotation achieving a delta-v of 0.4 km/s will change the SEV's orbit inclination to 8° and raise the periapsis to 9377 km. The periapsis will then match the radius of the Phobian orbit. After a 15-hour transfer, the SEV will perform a delta-v of -0.7 km/s to place the crew in a circular orbit with 1°

inclination for Phobos trailing. The SEV will trail Phobos for a minimal duration of 14 days. This duration may increase if the initial mothership-Phobos parking orbit phasing requires less than 14 days. The SEV will visit several sites on the Phobian surface. When complete, the mothership will exit the parking orbit to enter the Phobos orbit for docking on November 6, requiring a total delta-v of 1.1 km/s. A detailed description of the SEV activities is provided in a previous section. The Areocentric operations orbits are displayed in Figures 11 and 12. Table 6 shows delta-v budgets for the mothership and the SEV. Figure 13 is an artist's view of the mothership arriving at Mars.

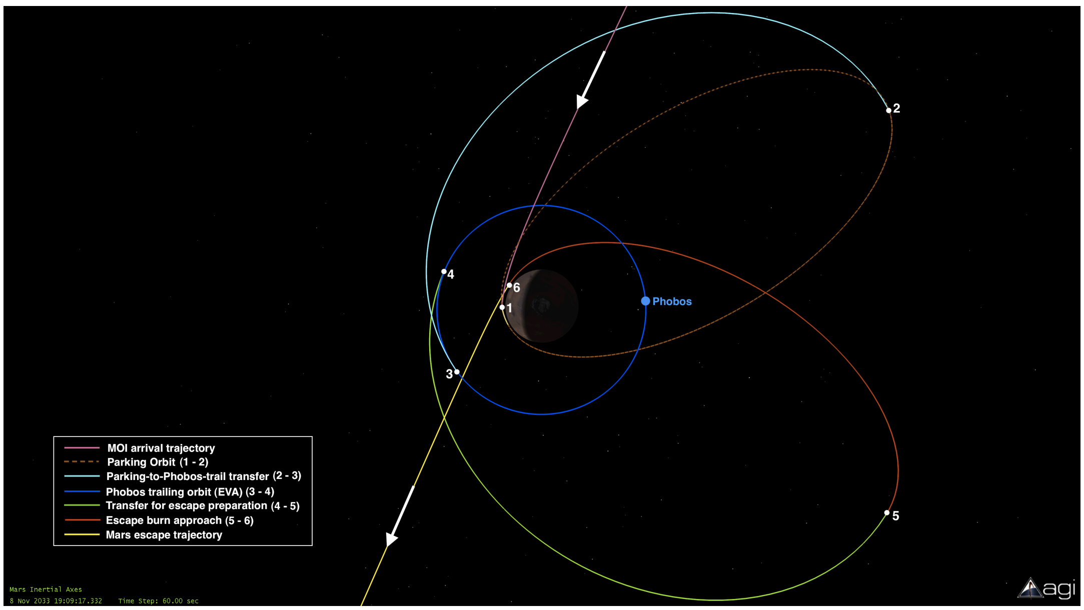


Figure 11: A comprehensive photo of space mission orbital maneuvers performed in an Areocentric orbit. The two white arrows indicate incoming and outgoing asymptotes.

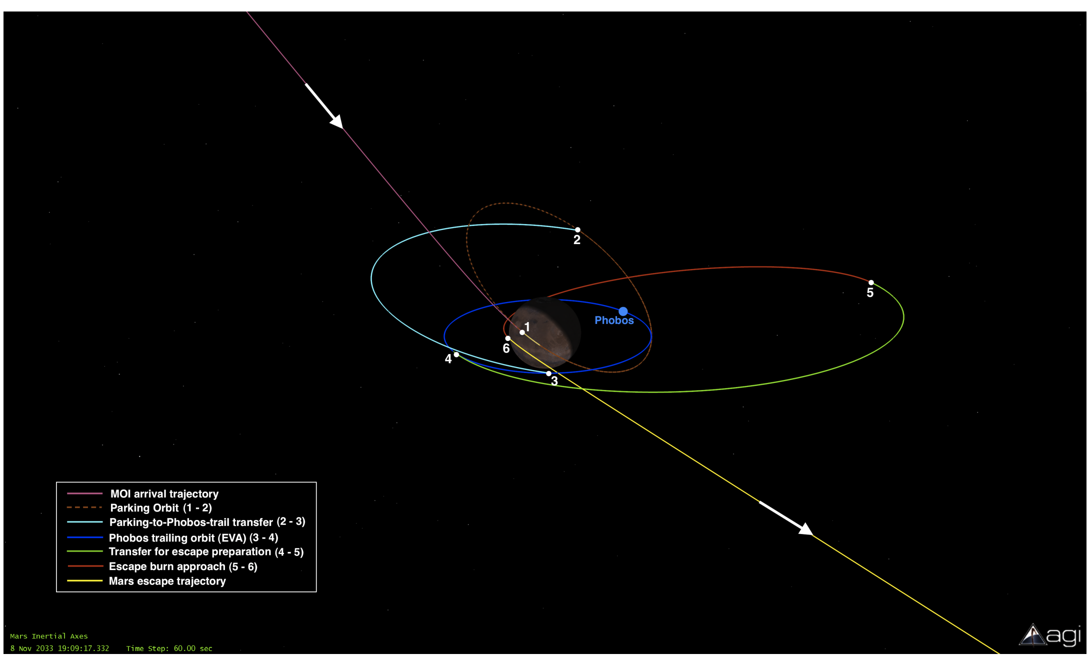


Figure 12: Alternative view of Areocentric trajectories, showing plane-changes for MOI and Phobos trailing orbit.

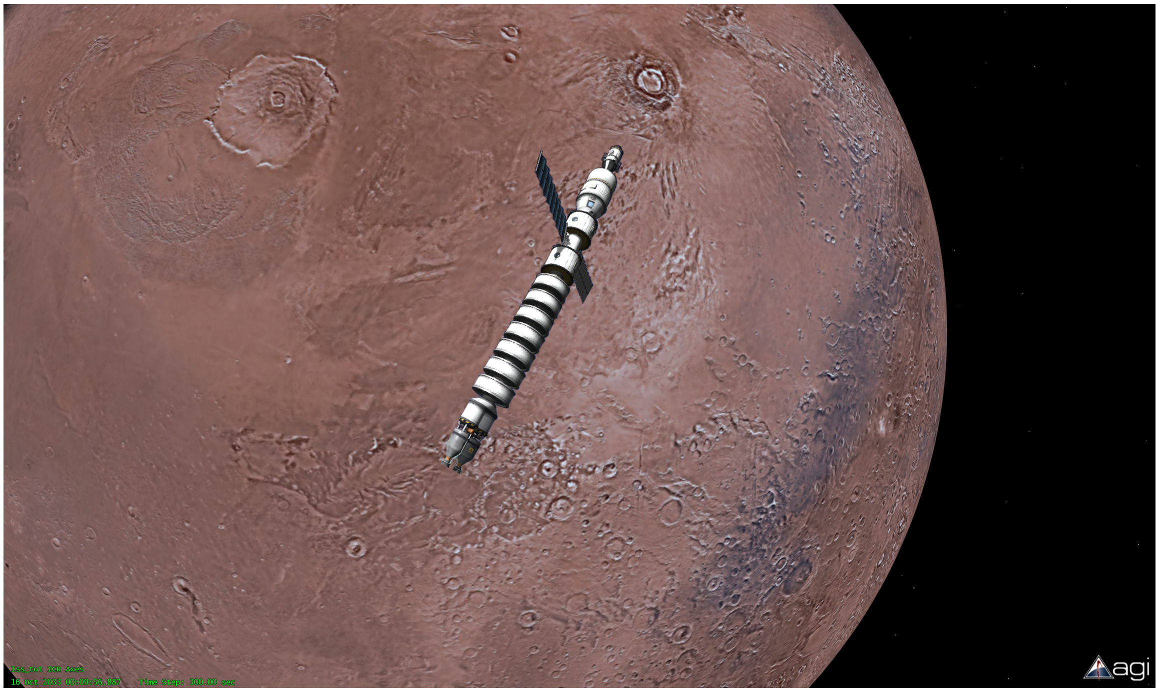


Figure 13: An artist's interpretation of the mothership arriving at Mars orbit.

Description	delta-v
Place mothership on hyperbolic trajectory	3.5 km/s
Mars orbit insertion (MOI)	2.2 km/s
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 km/s
SEV Phobos trailing orbit insertion for astronaut EVA	0.7 km/s
Mothership departure from parking orbit	0.4 km/s
Phobos trailing orbit insertion for mothership	0.7 km/s
Phobos trailing orbit exit when EVA is complete	0.5 km/s
Burn at apoapsis to prepare for escape trajectory	0.2 km/s
Delta-v for Mars sphere of influence escape for return to Earth	3.7 km/s
Total delta-v requirement for SEV	**1.1 km/s**
Total delta-v requirement for mothership	**11.2 km/s**

Table 6: Delta-V summary for mothership and SEV mission operations

After the crew transfers from the SEV back to the mothership, the SEV will return to the Phobian surface. It will use the anchoring system it used during EVA activities to attach itself to Phobos. From the mothership, the crew will collect further scientific data from Phobos using tele-robotic systems on their way back to Earth. A delta-v burn of 3.7 km/s on November 8, 2033 will send the mothership on a hyperbolic return trajectory to arrive at Earth on July 16, 2034 with a reentry speed of 16.2 km/s.

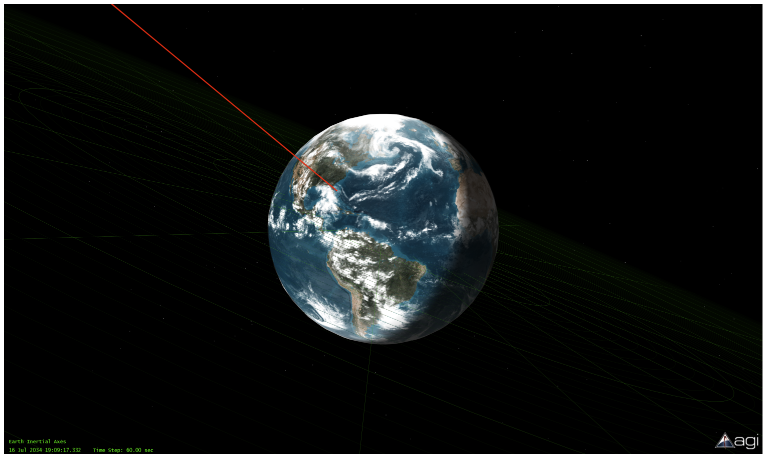


Figure 14: Earth atmosphere re-entry trajectory

The nominal duration of the mission is 465 days, with a 185-day outbound transfer, a 30-day stay at Mars and a 250-day inbound transfer.

The decision to choose the proposed mission key dates and trajectories were based on multiple factors. The first trade-off was between undertaking a short-stay (opposition class) mission or 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, the opposition-class mission concept was chosen.

The total delta-v from LEO to Mars as a function of round-trip time and departure date is shown in Figure 15. 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) and the fact that the launch date must not be later than January 1, 2041.

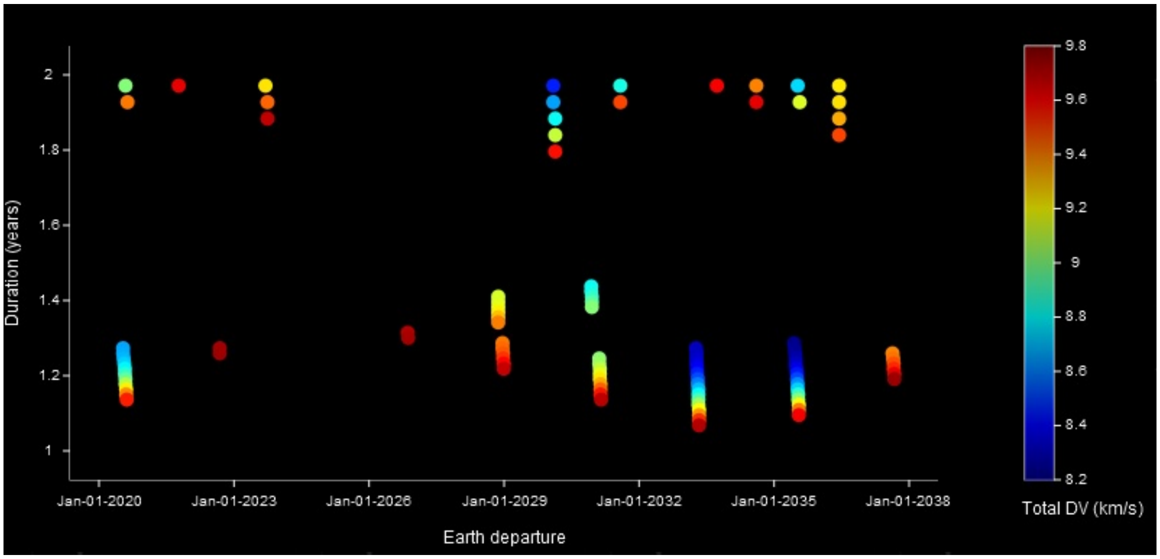


Figure 15: Total delta-v from LEO as a function of the Earth departure date and round-trip duration (Ames, 2013).

Concerning radiation exposure, it is most favorable to perform a deep space mission during solar maximum. The first solar maximum within the shown departure dates will peak around 2022 (solar cycle 25); the following cycle peaks between 2033 and 2035 (solar cycle 26). Solar cycle 25 is predicted to be the weakest one in centuries. Additionally, there are only a few possible launch dates in 2022 for an opposition-class mission. This is why April 2033 was chosen for further investigation.

Figure 16 shows the total delta-v from LEO as a function of round-trip duration. It shows that a shorter round-trip duration automatically leads to an increase in the total Δv required. It has to be noted that the lowest delta-v, i.e. longest round-trip duration, corresponds to the earliest departure date (April 7, 2033) and with later departure dates the round-trip duration decreases while delta-v increases. Following from this, April 7, 2033 was determined as the nominal departure date, as this would allow for a launch slip of up to 25 days. Choosing this trajectory, there will be constant line of sight from the spacecraft to Earth while in transit to and from Mars. This will be beneficial for flight control communications and crew safety.

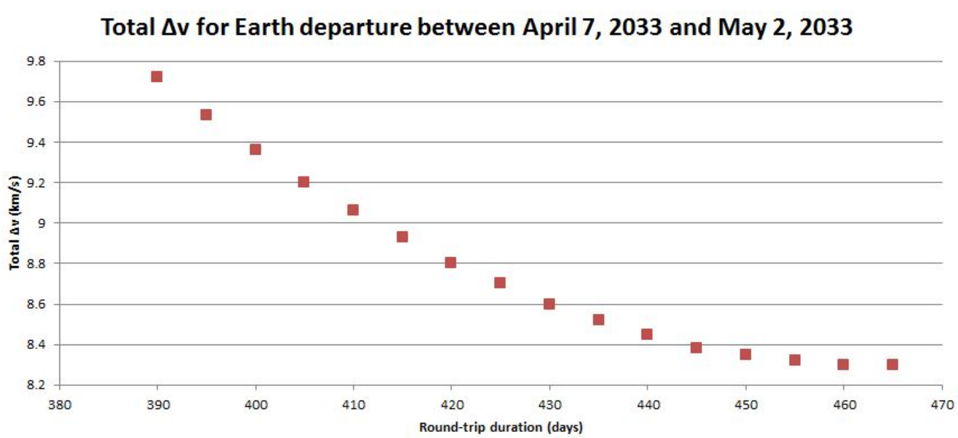


Figure 16: Total Δv requirements for Earth departure between April 7, 2033 and May 2, 2033.

The nominal trajectories were calculated using a robust lambert solver (Oldenhuis, 2010), with ephemerides from JPL (Jet Propulsion Laboratory, 2012). At Mars, a bi-elliptic transfer was chosen to safely transport the SEV from the mothership to Phobos. In theory, a Hohmann transfer would be more efficient to do this (Figure 17), where the red square marks the used transfer's position on the graph. However, as the spacecraft is just being captured by Mars when starting this transfer, the actual delta-v required to do a Hohmann transfer is ten times larger than the delta-v using the bi-elliptic transfer.

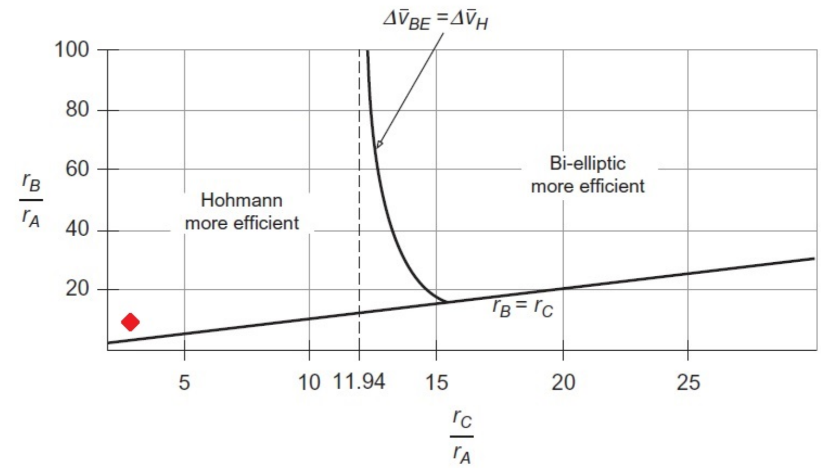


Figure 17: Orbits for which the bi-elliptic transfer is either less efficient or more efficient than the Hohmann transfer (r_A referring to the starting orbit, r_B referring to the bi-elliptic apoapsis, r_C referring to the final orbit; Curtis, 2010).

Propulsion and Launch Vehicle Selection

Technology selection

For the sizing of the propulsion system and subsequent selection of the required launch program, several assumptions and simplifications have been made. The mission was split up into the precursor mission and the human exploration mission. Additionally, the latter was divided into several cargo launches and a separate crew launch. Table 7 gives an overview of the considered propulsive elements for an impulsive burn.

Table 7: Overview of propulsive elements properties

	Cryogenic Stage	Nuclear Thermal Stage	CH4 Stage (SEV Propulsion Module)	Service Module Propulsion
Specific Impulse	465s	900s	355s	328s
Structure Ratio	23%	27%	15%	(Part of Service Module)

Table 8: Solar Electric Propulsion LEO - HEO assumptions

Flight duration	1 year
Specific impulse	2000 s
Array-to-jet efficiency	60%
Inert mass array	30 kg/kW

Xenon tank mass	5% of fuel mass

The first trade-off was between the choice of moving the whole spacecraft or just a space exploration vehicle to the proximity of Phobos to do the extra-vehicular activities. A reduction of 36% of total mass in low Earth orbit (LEO) can be achieved, if a separate vehicle is used and the mothership stays in Mars parking orbit. The propellant required to maneuver in Phobos proximity reduces drastically consequently reducing total mission mass.

Another trade-off was made on the staging orbit for the cargo and crew on the way to Mars. A highly elliptical orbit (HEO) and a low Earth orbit (LEO) were considered. Moreover, cryogenic and nuclear thermal propulsion were investigated. The options of staging in HEO included using solar electric propulsion (with similar assumptions as for the precursor mission) for the cargo which can be launched in advance. The assumptions for the SEP system required to lift the cargo mass from LEO to HEO are summarized in Table 8.

The lower velocity change required to reach Mars from HEO reduces the amount of propellant for the impulsive burn of the first stage. The mass is reduced by 32% and 6% (comparing NTP from HEO to NTP from LEO system) for the cryogenic and nuclear thermal propulsion, respectively. Though the total mass of propellant needed in LEO is lower, the mission complexity increases due to the required development of the additional solar electric propulsion system and high-output power system. Especially, the mass savings for the NTP system are in a range of 6%, making the use of the SEP system questionable and might not justify the development and qualification of an additional module against the cost of one heavy lift launch.

The mass of the fourth option can now be compared depending on the mass of the vehicles, including crew, service modules, deep space habitat, and space exploration vehicle. Table 9 shows the comparison of the four options. Every configuration has a 10% margin on the propellant to account for trajectory correction maneuvers.

Table 9: Comparison of cryogenic and nuclear thermal propulsion from LEO and HEO

	Cryo from LEO	Cryo from HEO	NTP from LEO	NTP from HEO
Power required	-	1.8 MW	-	1 MW
Mass percentage	100%	68%	37%	35%

In conclusion, nuclear thermal propulsion to accelerate the cargo and crew from low Earth orbit was chosen as the desired mission architecture. While it was 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. In the following section the safety issues and concerns of nuclear thermal propulsion are discussed in more detail.

Technology justification

TECHNOLOGY READINESS

As described in Table 7, the high specific impulse of Nuclear Thermal Propulsion (NTP) compared to existing cryogenic propulsion systems has the potential to significantly reduce the total mass in LEO, therefore driving down the number of launches and overall cost of the mission. Moreover, the NTP uses a single propellant (LH2) as opposed to a propellant-oxidizer mixture of cryogenic propulsion, which simplifies the design of the NTP engine. The United States have ground tested the technology in the 1960s in the ROVER and NERVA programs and is currently at TRL 6 (*Robbins, 1991; Mankins, 1995*). The Soviet Union was also developing a solid-core nuclear thermal engine (*Zakirov, 2007*). In order to determine if the NTP option is available, we need to take into consideration the Technology Readiness Level and safety concerns related to radioactive material.

The NTP option is using its engines for interplanetary travel, which is an essential part of the mission. Moreover, presence of crew calls for safer, tested technology. Therefore, the NTP technology needs to be at TRL 9 when the vehicle is built. 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 translunar missions (*Cohenn 1989 , Gunn, n.d.*).

SAFETY CONSIDERATIONS

DISPOSAL OF USED STAGES

Cryogenic propulsion stages are typically left in Earth orbit or deorbited in Earth atmosphere when they are no longer needed. However, disposal of equipment containing radioactive devices calls for special consideration. Failures in missions that use radioactive equipment like RTG, 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 (*Weiss, n.d.*). The RTG of the Apollo 13 lunar module entered the atmosphere at 11 km/s and survived reentry without breaking containment. Another example is NIMBUS B-1, which suffered a failure before reaching orbit, and the RTG landed in the ocean without breaking containment and was later retrieved and reused (*Furlong, 1991*).

There will be differences between RTG units and the NTP nuclear core. The core will be larger, more exposed and 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 has 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 design described in this report uses one NTP stage. It will separate after performing a trans-Earth burn. In order to prevent the NTP stage from reaching Earth orbit, two maneuvers will be performed. A small fraction of the initial fuel mass will be left in the stage to perform those maneuvers. First will come an out-of-plane burn after leaving the Mars sphere of influence to increase the inclination of the orbit in the Sun reference frame by 1-2 degrees, which will ensure the disposed NTP and Earth orbits never cross. Second will come a retrograde burn at perihelion that will ensure the stage will not cross Mars orbit.

RISK MANAGEMENT

Special attention needs to be given to risk management because of the presence of a radioactive payload. For that reason, the following precautions will be put into place: The nuclear core will be placed in a casing that will prevent containment breach in case of reentry. Second, the nuclear core will be kept passive during the flight and final disposal orbit, and only be turned on during the burn. If the radiation levels reach critical level, an automatic system will shut down the engine. Risk management scenarios are described in an earlier table.

Detailed design

ENGINE

The mass of the nuclear thermal propulsion module is approximated using assumptions in Table 10 (Mazanek, 2013). The propulsion system comprises two different modules. The first module consists of the engine and nuclear core as well as some propellant. The remaining mass on the chosen launcher performance is filled with propellant. The second module is a tank carrying the bulk of the liquid hydrogen. Now, the thrust of the stages is compared with the respective total mass to calculate the thrust-to-weight ratio.

Table 10: Nuclear Thermal Propulsion module assumptions

Thrust per engine core	222 000 N
Solid engine core mass	14,000 kg
Stage 1 Thrust-to-weight	0.09
Stage 2 Thrust-to-weight	0.12

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. The elliptical orbit at Mars requires the increased ratio. The duration of the burn is 45 min.

FAIRING

The main propellant for the nuclear thermal propulsion is liquid hydrogen with a very low density of 70.85 kg/m^3. 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 the drag, structural and control requirements.

In order to increase the payload volume while still meeting the structural and control requirements, an aerodynamic shroud is proposed. The initial configuration, with the usable payload in red, can be seen in Figure 18 (Ochinero, 2009). In the paper by Ochinero (Ochinero, 2009), the initial configuration is optimized to achieve the same launcher performance while almost doubling the payload volume.

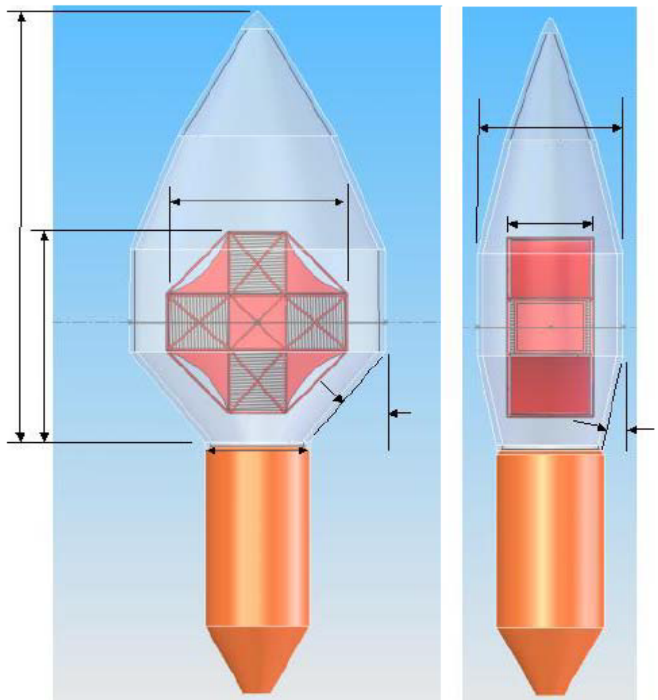


Figure 18: Initial payload fairing design

The mass increase of the fairing due to the additional structure is approximated to 36% from the standard payload fairing design. For the Atlas V HLV the standard 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. The volume increase for the Falcon Heavy fairing is approximately 100%. One has to take into account the increased cost for the development of the new shroud.

Launcher selection and launch campaign

The mass of an individual propulsion module (PM) and tank is determined by the maximum launch capacity of common heavy-lift launch vehicles (mostly in development). Thus, it determines the number of propulsion modules and launches. Four different heavy-lift launchers were 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, respectively.

Table 11: Launch manifest for Falcon Heavy option

Launch	Launcher	Payload	Mass [t]	Margin	Cost
1-9	Falcon Heavy	2xPM/7xTank	50/47	6%/11%	approx. 80$M-125$M per launch
10	Falcon Heavy	DSH	47	11%	approx. 80$M-125$M per launch
11	Falcon Heavy (manned)	CM+SM+SEV (+CH4 Stage)	41	23%	approx. 80$M-125$M per launch

Table 12: Launch manifest for SLS option

Launch	Launcher	Payload	Mass [t]	Margin	Cost
1/2	SLS Cargo II [120t]	2xPM	117	3%	approx. 500$M-2500$M per launch
3/4	SLS Cargo II [120t]	2xTank	97.5	19%	approx. 500$M-2500$M per launch
5	SLS Cargo I [70t]	DSH	47	33%	approx. 500$M-2500$M per launch
6	SLS Crew [70t]	CM+SM+SEV (+CH4 Stage)	41	42%	approx. 500$M-2500$M per launch

Table 13: Launch manifest for Atlas V HLV option

Launch	Launcher	Payload	Mass [t]	Margin	Cost
1-13	Atlas V HLV	2xPM/11xTank	28.4/29.2	3%/1%	approx. 125$M-135$M per launch
14/15	Atlas V HLV	DSH	47	20%	approx. 125$M-135$M per launch
16/17	Atlas V HLV	CM+SM+SEV (+CH4 Stage)	41	31%	approx. 125$M-135$M per launch

Table 14: Launch manifest for Atlas V HLV/Falcon Heavy option

Launch	Launcher	Payload	Mass [t]	Margin	Cost
1-4	Falcon Heavy	2xPM/DSH/CM+SM+SEV (+CH4 Stage)	28.4/29.2	3%/1%	approx. 80$M-125$M per launch
5-13	Atlas V HLV	Tank	29.2	1%	approx. 125$M-135$M per launch

The estimation of cost is given as a range due to the uncertainty in development and operational cost of the vehicles currently in development. Table 11-14 present the launch manifest for the different options whereas Table 15 compares the total amount of launches and cost range.

Table 15: Comparison of Falcon Heavy, SLS, Atlas V HLV and Atlas/Falcon launch options

	Falcon Heavy	SLS Crew/Cargo I+II	Atlas V HLV	Atlas V HLV/Falcon Heavy

Number of launches	11	6	17	13
Launch cost [$M]	880-1,375	2,580-12,625	1,625-1,880	1,445-1,715

Due to the lower cost, the Falcon Heavy option is the preferred choice. However, the increased amount of launches compared to the SLS option poses an increased failure probability. Through multiple available launch sites/pads the duration can be reduced and the reaction time in case of failure increased. The launch campaign has to start 30 weeks before the crewed launch if a turn-around time of 3 weeks is assumed.

If only a total amount of four launches of Falcon Heavy per year are possible (SpaceX, 2012), a combination of Falcon Heavy and Atlas V HLV launches is used to reduce the duration of the launch campaign. Moreover, an Atlas V HLV option only was considered as well since Atlas V HLV does not require any changes to the payload fairing.

Notes and further investigation

There are a few options, which would enable the use of solar electric propulsion by lowering the power requirements. One could lift the cargo either one after another and reuse the SEP module or at the same time with multiple SEP modules. This would reduce the maximum power required per module though add complexity to mission planning and in the first case increase the time needed in advance.

The option of pre-deploying the return trip propulsion modules at Mars before the crew arrives was discarded 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 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 requires thorough planning and has increased mission complexity.

Another country with medium to heavy lift launch capability is Russia, though the lack of information on costs and fairing sizes prevented further analysis.

Habitation Elements

General Habitat Design Approach

In developing the habitation elements for the mission, the following general systems architecture guidelines were followed to maximize system and operational reliability, flexibility, and ultimately crew safety:

- Leverage systems that are currently in use, or development - this is based on minimizing development cost and risk
- Maximize commonality across all mission elements - this increases the system robustness by creating options for scavenging parts. Furthermore, it lowers the number of spares required, as well as system manufacturing and development costs.

- Maximize multifunctionality and synergies among systems - this leads to more function for less mass
- Account for crew safety during all mission modes
- Implement lessons learned from past programs

Based on these guidelines, the following architectural choices were made:

- Employ an International Space Station-derived Deep Space Habitat as the baseline design for the Deep Space Habitat (DSH)
- Employ the Multi-Mission Space Exploration Vehicle currently under development at NASA Johnson Space Center, as the baseline for the Space Exploration Vehicle (SEV)
- Employ the NASA-ILC Dover Mark III Spacesuit for the Extravehicular Mobility Unit - 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
- Employ the Orion Multi-Purpose Crew Vehicle (MPCV) as the baseline 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
- Choose spacecraft atmospheres based on those suggested by the NASA Exploration Atmospheres Working Group (EAWG) (NASA EAWG 2006) to ensure atmospheric capability between spacecraft elements while ensuring that pre-breath time for the required EVA frequency is properly accounted for
- Employ the “water-walls” concept, currently being developed under a NIAC Proposal (Cohen 2012) to exploit consumables and functions for their radiation protection mass potential

The following sections further expand on these top-level architectural choices and describe, in detail, the specific systems that were sized for this mission.

Water Walls for Life Support and Radiation Shielding

The general life support system concept is based on the water wall concept currently being developed under a NIAC proposal [Cohen and Flynn, GLEX 2012]. It consists of lining the shell of a habitable volume with a series of polyethylene bags to simultaneously perform life support functions, while providing radiation shielding. Currently, the technology is rated for TRL 3. One space flight experiment has been performed to demonstrate the basic technology. This general concept has been baselined for the Mars 1 mission proposed by Dennis Tito, where metabolic waste produced by the crew is stored in the walls of the spacecraft for added radiation protection. Further experiments on a larger scale for a longer duration are necessary before the system can be implemented. Further information on the system can be found in the Water Walls System section found later in the report.

Habitat Atmosphere Selection

One of the key driving architectural decisions related to habitation systems is the choice of atmospheric pressure and composition. This decision was considered to be so important that an agency-wide working group was formed during the early stages of NASA’s now-cancelled Constellation Program to develop guidelines for use by the wider community (NASA EAWG 2006). Specifically, this decision is important because it primarily dictates:

- crew health over the duration of the mission
- the frequency of extravehicular activity (EVA) able to be facilitated from a habitable volume - this is because spacesuits typically operate at approximately 29.6kPa (4.3psi) and 100% O2. The composition and pressure of the atmosphere dictates the amount of pre-breathing time required to avoid decompression sickness (a.k.a. “the bends”). If not properly selected, the pre-breathing time can be so long as to limit the number of EVAs that can be performed
- the sizing of the Environmental Control and Life Support System (ECLSS), including the size of the required tanks, leakage rate, and overall Air Revitalization strategy
- the structural rigidity of the habitat - a higher pressure requires more structure to maintain it, and hence more mass. This, in turn, impacts propulsion and attitude control systems, among others.
- fire safety within the habitat - more oxygen increases the risk of fire within a habitat

Based on the NASA EAWG group’s analysis of these considerations, the following atmospheres have been selected for the various habitation elements employed over the course of the mission.

Table 16: Atmosphere Selection for Habitation Elements

Habitation Element	**Atmospheric Pressure and Composition**[1]
Deep Space Habitat (DSH)	101.3kPa (14.7psi), 21% O2 nominally, 70.3kPa (10.2psi), 26.5% O2 during pressurization with the SEV
Space Exploration Vehicle (SEV)	70.3kPa (10.2psi), 26.5% O2
Extravehicular Mobility Unit (EMU)[2]	57kPa (8.3psi), 100% O2 (Mark III suit)
Orion Multi-Purpose Crew Vehicle (MPCV)	101.3kPa (14.7psi), 21% O2 nominally, 70.3kPa (10.2psi), 26.5% O2 during depressurization prior to EVA from the vehicle

[1]In all of these atmospheric compositions, nitrogen is chosen as the diluent gas

[2]The selection of this suit means that no pre-breathe time is required. Additionally, in the case that the portable life support system (PLSS) of the current International Space Station (ISS) EMU is employed, a prebreath time of only 40 minutes is required

Deep Space Habitat (DSH)

DSH HABITAT DESIGN

As mentioned earlier, ISS derived habitat structures were chosen as a baseline architecture for the deep space habitat, 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 specifications for the DSH can be found in Table 17 (Smitherman, 2012).

Table 17: DSH Specifications

Property	Value
Pressurized Volume	185 m^3
Habitable Volume	76.3 m^3
Average TRL	8.1
TRL 9/Heritage	65%
Habitat Length	13 m
Habitat Diameter	4.5 m
Wet Mass	46.3T

The four crew, 60 day mission habitat architecture was selected for its appropriate habitable volume for a three-person crew. The Celantano Curve, based on heuristics, establishes an optimal habitable volume per person for long duration missions as 20 m^3, as seen in Figure 19 [Larson, 1999].For a three-person crew, this equates to 60 m^3, which is less than the available habitable volume offered by the DSH.

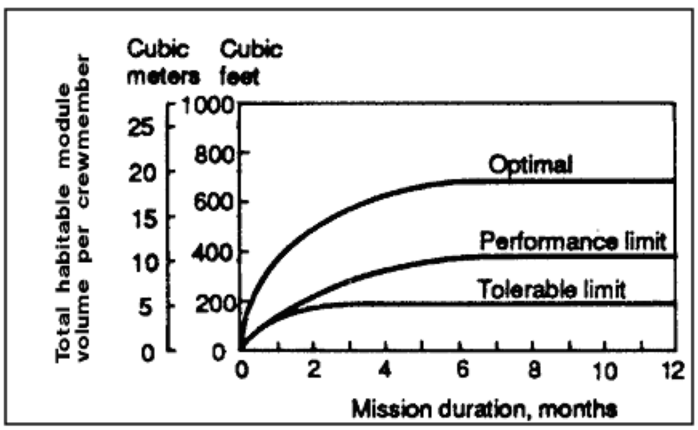


Figure 19: Celantano Curve for Habitable Volume (Larson, 1999)

The DSH consists of the following ISS components:

- Utility Tunnel / Airlock
- Node 1
- Multi-Purpose Logistics Module (MPLM)

The MPCV and SEV both dock to the Node, with the MPCV docking at the end and the SEV docking on the side of the Node. The propulsion stage attaches to the other end of the MPLM. Figure 20 shows the docking configuration. Due to the differences in atmosphere composition and pressure between the DSH and SEV, pumps will be used to decompress the DSH to equalize the pressure with that of the SEV prior to crew egress. The habitable volume of the DSH is shown in Figure 21 with crew quarters placed at the end of the MPLM. Each crew member is allocated $4m^3$ of personal space in this section of the spacecraft. Figure 22 shows a more detailed internal volume. Subsystems such as ECLSS are located in between the ceiling/floor and external structure while payload racks and storage are located in accessible locations. An internal layout of main subsystems within the habitat is shown in Figure 22 [Smitherman, 2012]. Radiation shielding within the DSH consists of aluminum and water, as discussed in sections covering structures and the Water Walls.

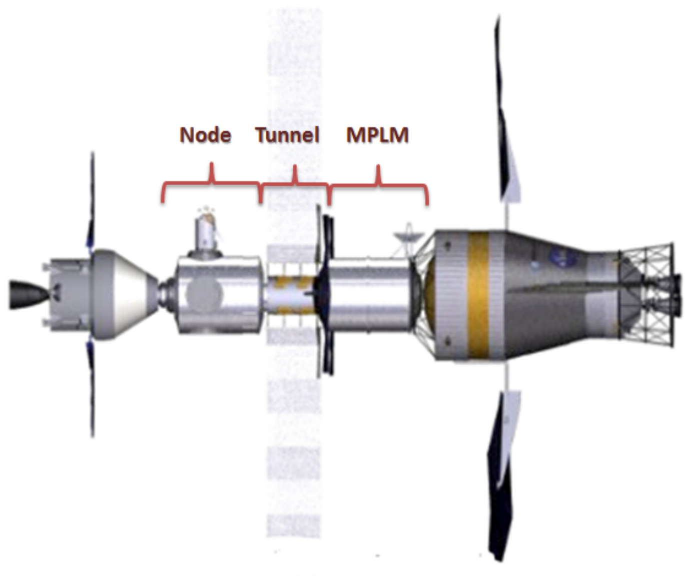


Figure 20: Deep Space Habitat Configuration (Smitherman, 2012)

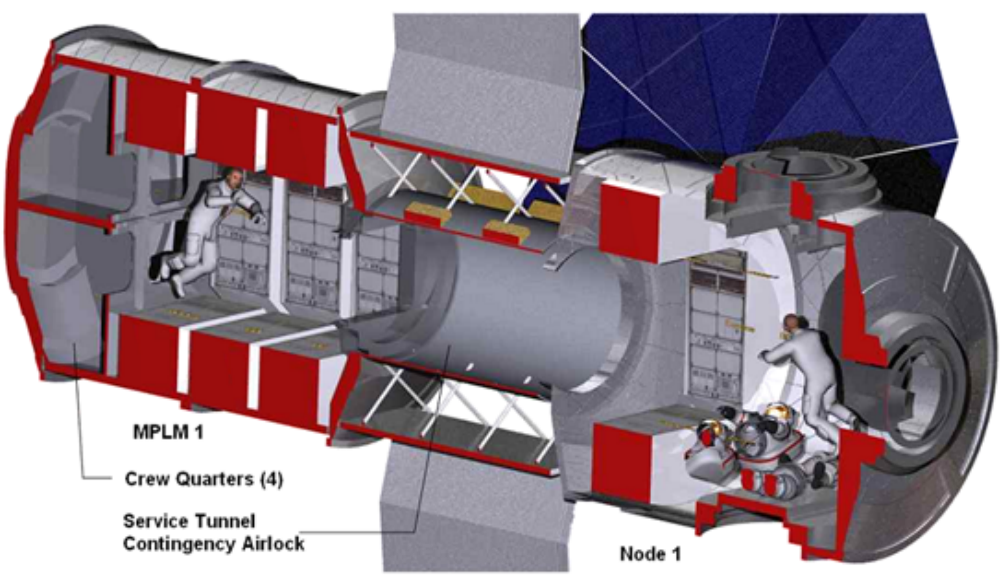


Figure 21: Deep Space Habitat MPLM and Node Internal Structure (Smitherman, 2012)

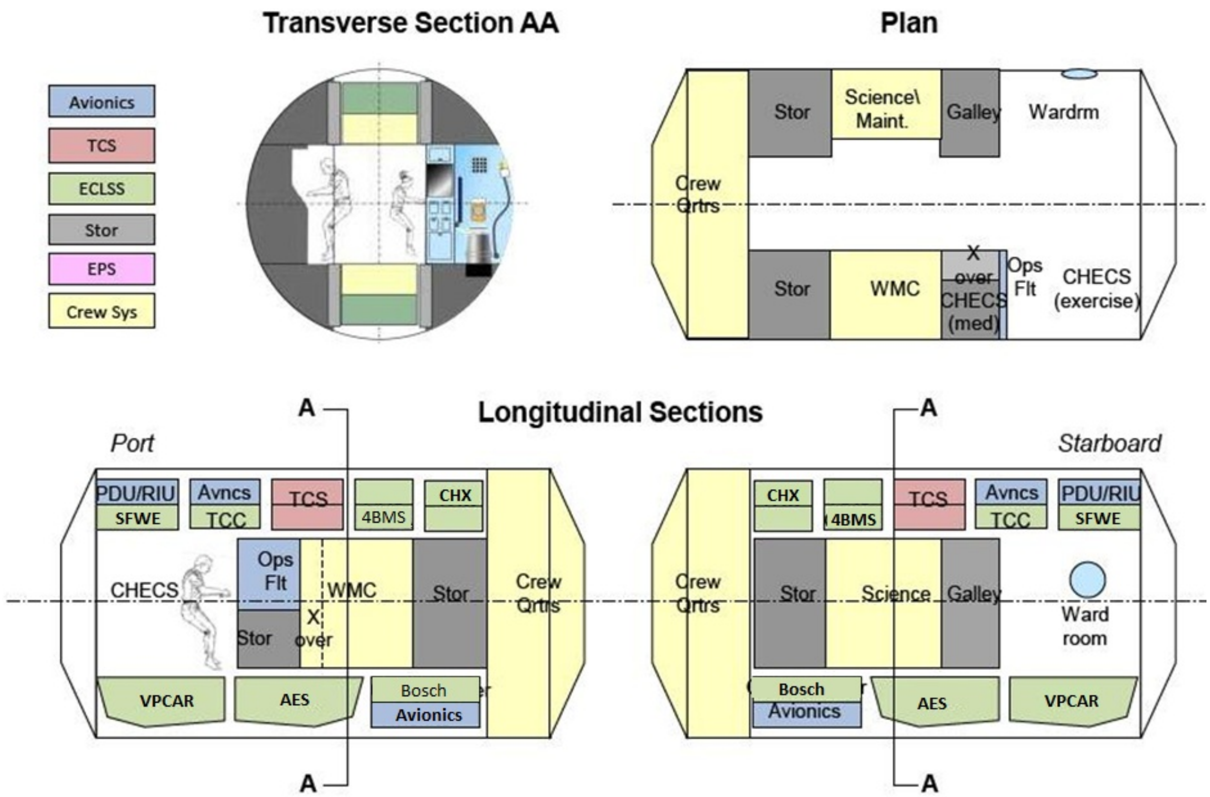


Figure 22: DSH Internal Layout of MPLM, Airlock and Node (Smitherman 2012)

DEEP SPACE HABITAT ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM

The DSH ECLSS consists of two different systems. The primary ECLSS is a closed-loop system similar to what is used on the ISS. The secondary ECLSS is a passive system, known as Water Walls, that filters waste products through a series of forward osmosis treatment bags. Including both of these systems in the DSH design provides redundancy, increased radiation protection and validation of the Water Walls technology. Each system will address the four main components of the DSH ECLSS:

1. Atmosphere Control
2. Water Recycling and Provision
3. Food Provision
4. Waste Management

ECLSS design is supported by a software tool called “Environment for Life-Support Systems Simulation and Analysis” (ELISSA). It was developed at the Institute of Space Systems (Institut für Raumfahrtsysteme, IRS) at the University of Stuttgart, Germany. ELISSA is part of a software tool collection, which was developed at the IRS for the Space Station Design Workshop and was provided by courtesy of the IRS. It allows the design analysis and validation of primarily life support systems and also studies about parametric optimization or failure mode effects.
The software is implemented using the commercially available software LabVIEW®, which provides a graphical programming language. Convenient simulation features and easy customization and extension

are provided through a powerful, intuitive graphical user interface. Subsystem component libraries have been predefined for the ECLSS as well as for the power supply and attitude/orbit control subsystems. Simulations can be interactively controlled, which allows analysis of dynamic problems, or real-time operator training.

DSH PRIMARY ECLSS

The DSH ECLSS is designed as a closed-loop system to minimize consumable mass. This is an important design choice due to the long mission duration and the fact that resupply missions to the Martian system are considered as being unfeasible. Thus an open loop system would have mass values orders of magnitude higher than a regenerative closed-loop system.

Figure 23 shows a flow diagram of the Primary ECLSS incorporating technologies categorized as atmosphere, waste, water and food. Inputs and outputs from the crew cabin as well as flows between the technologies are shown through the arrows. Technologies are represented as rectangles while storage tanks are represented as circles.

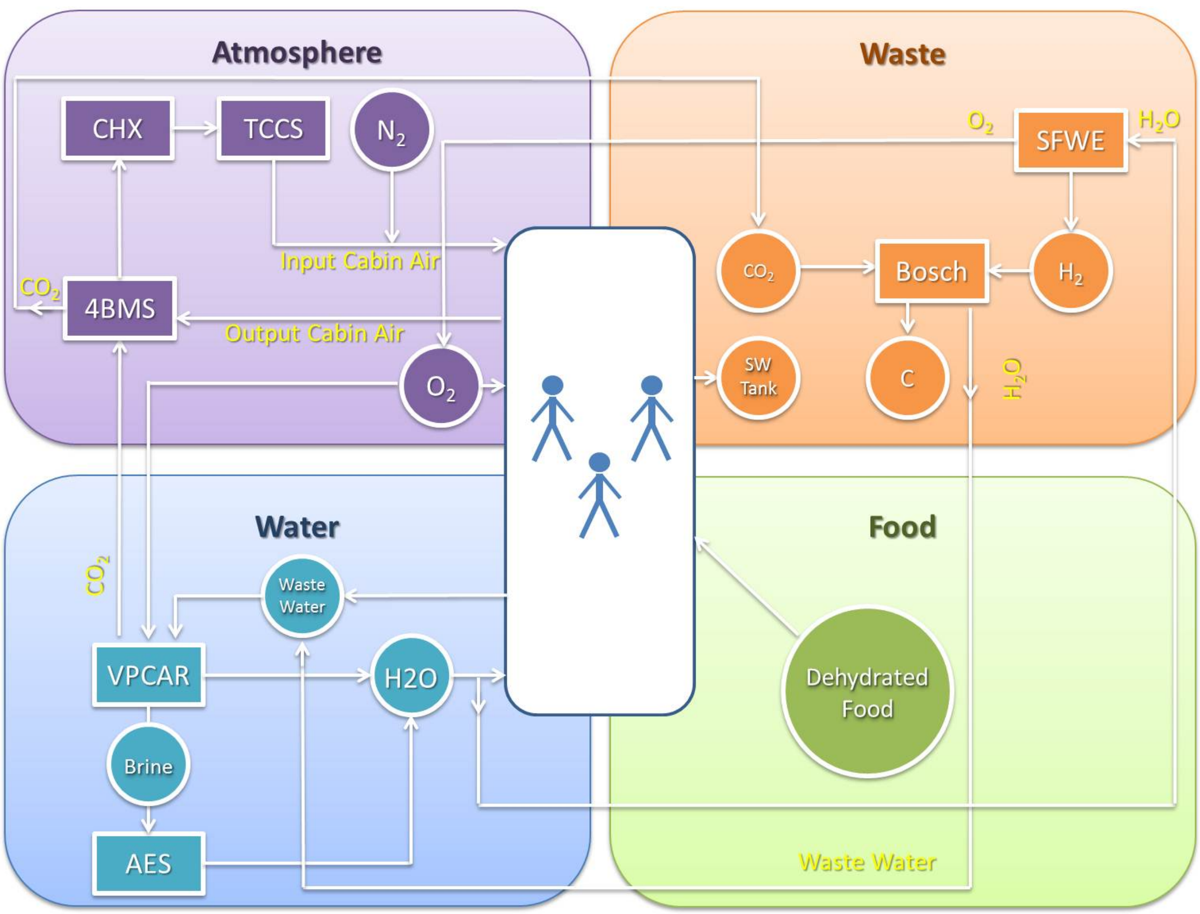


Figure 23: DSH Primary ECLSS Flow Diagram

Specific technologies along with their functions, mass values and design choice rationales are described in Table 18. This table also includes other dry mass components such as clothing, storage tanks, food packaging, fire suppression and air monitoring system. The ECLSS design was based on the current ISS system with new technologies substituted for carbon dioxide reduction, waste water management, air monitoring and solid waste management.

Table 18: DSH Primary ECLSS Dry Elements

Function	Hardware	Mass (kg)	Rationale
Carbon Dioxide Removal	4-Bed Molecular Sieve (4BMS)	120	Standard ISS CO_2 removal system; regenerable (Eckart 96)
Oxygen Generation	Static Feed Water Electrolysis (SFWE)	100	Standard ISS system (Wydeven 88)
Temperature and Humidity Control (THC)	Condensing Heat Exchanger (CHX)	100	Standard ISS THC system (estimate)
Trace Contaminant Control	Trace Contaminant Control (TCC)	100	Standard ISS TCC system (Eckart 96)
Carbon Dioxide Reduction	Bosch Reactor	102	More advanced technology that produces carbon instead of methane as byproduct [Eckart 96]
Waste Water Treatment, Urine Pretreatment	Vapor Phase Catalytic Ammonia Removal (VPCAR)	340	Processes all waste water in one system (Yeh 99)
Brine Treatment	Air Evaporation System (AES)	178	Necessary for processing VPCAR output (Yeh 99)
Food Packaging	15% Food Mass	139	(Hanford 2004)
Clothing	-	443	(Hanford 2004)
ECLSS Storage Tanks	High pressure storage tanks for gases	4357	(estimate)
Air Monitoring System	Analyzing Interferometer for Ambient Air 2 (ANITA 2)	27	Advanced, lightweight air monitoring system (Stuffler 2012)
Fire Suppression	Water droplet fire extinguisher (16kg per unit)	48	3 units, each 16kg deployed throughout the DSH (Carriere. 2012)
Solid Waste Management	Heat Melt Compactor	49	Advanced system that compresses waste to reduce volume and increase radiation shielding (Pace

Total Dry Mass	**-**	**6103**	-

Of note from this table, is the implementation of the water droplet fire extinguisher that is currently being developed by ADA Technologies under a NASA research contract. Instead of using CO_2, which can be toxic to the crew at high concentrations, this fire extinguisher uses water as its suppressant. In addition, heat melt compactor technology currently being developed in collaboration with NASA Ames Research Center is employed. This device compacts solid waste elements into “pucks”, which can then be used as radiation shielding. Figure 24 shows an example of a puck produced by an experimental version of this device.

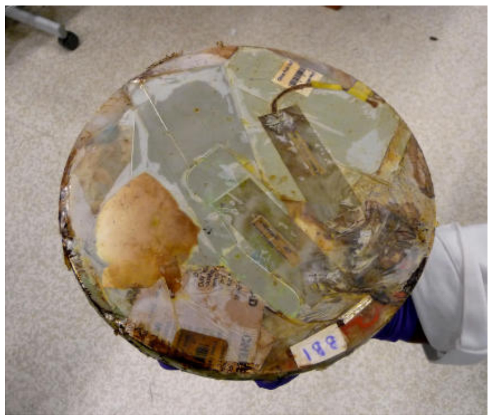


Figure 24: Example of a waste “puck” produced by an experimental Heat Melt Compactor

Table 19 includes wet mass consumables such as food, oxygen and water. These values will be supplied to the ECLSS system initially and used by the system throughout the mission duration. Initial values were based on crew member requirements such as amount of food and oxygen per crew member per day based on values from the literature. Values were increased by appropriate amounts to accommodate ECLSS system requirements.

Table 19: DSH Primary ECLSS Consumable Elements

ECLSS Consumable	Mass (kg)	Comments/Rationale
Dehydrated food	925 (0.62 kg/person/day)	(Hanford 2004)
O_2	150	Estimated from ELISSA simulation to provide crew with 2.8 kg of oxygen per day and maintain cabin atmosphere
Potable H_2O	500	Estimated from ELISSA simulation to provide crew with 2.8 kg of oxygen per day and maintain ECLSS systems
Hygiene H_2O	0	This water will come from the Water Walls system; estimates for hygiene H2O are 0.4 kg/p/d (Hanford 2004)

H_2	15	Estimated from ELISSA simulation to maintain ECLSS systems and cabin atmosphere
N_2	290	Estimated from ELISSA simulation to maintain ECLSS systems and cabin atmosphere
Total Wet Mass	**1880**	

ELISSA was used to validate the Primary ECLSS model closure ability and consumable mass estimates. Results from the simulation are shown in the following graphs.

Figure 25 shows ELISSA simulation results. Plotted are the gas masses (tank level) over time. Nitrogen is depleted over time as it is used in the atmosphere composition and pressure maintenance. Oxygen and hydrogen remain constant due to the closed-loop nature of the system. Carbon dioxide is removed after reaching an estimated opening tank level and retains its cycling pattern throughout the mission duration.

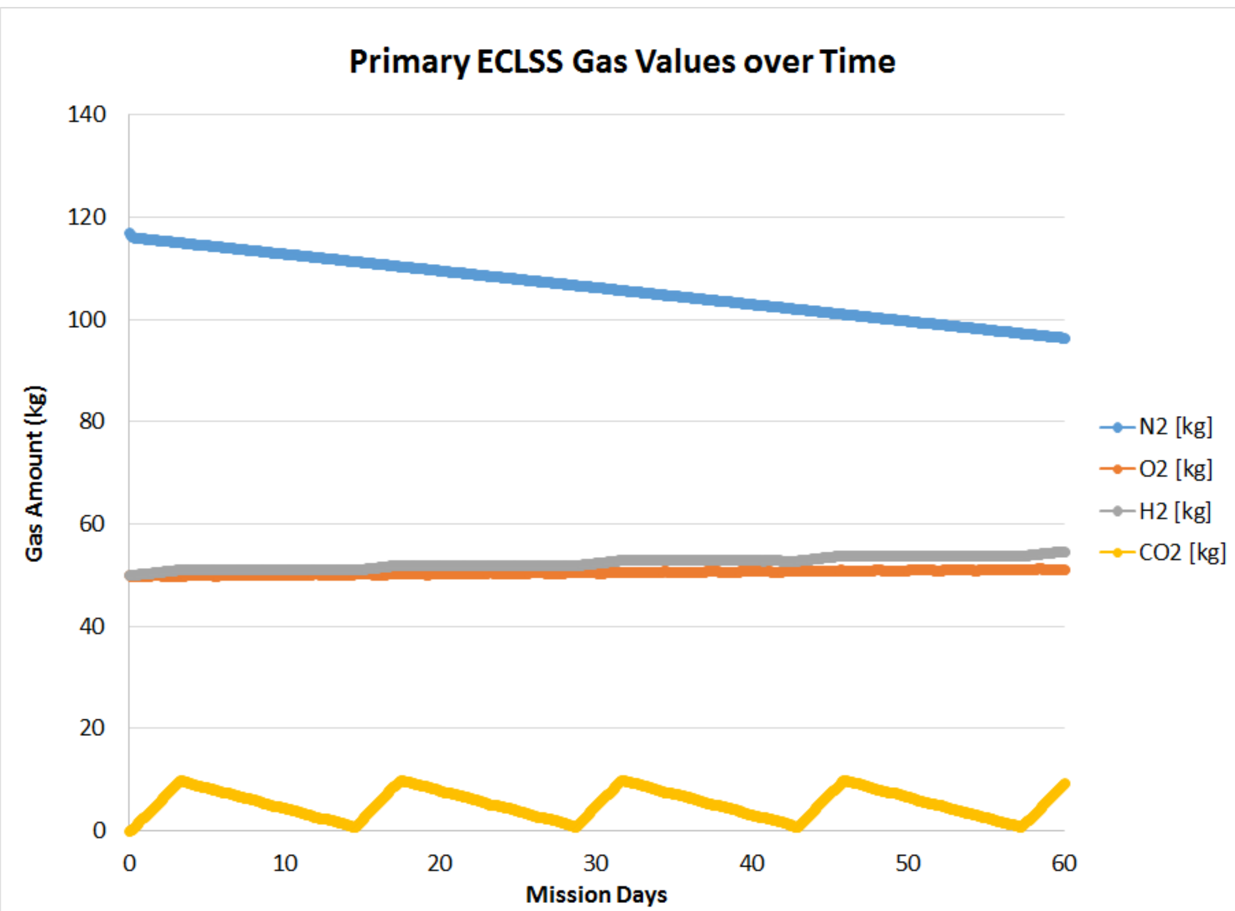


Figure 25: Atmospheric Management System Constituent Amounts in Cabin over Time

Figure 26 shows the water management system component values over time. Water is cycled but always remains well above necessary levels for consumption and ECLSS function. Waste water level also cycles between minimum and maximum values based on predefined tank capacity and system capabilities. Urine and brine operate in a similar fashion. Overall, the Primary ECLSS functioned nominally throughout the simulation.

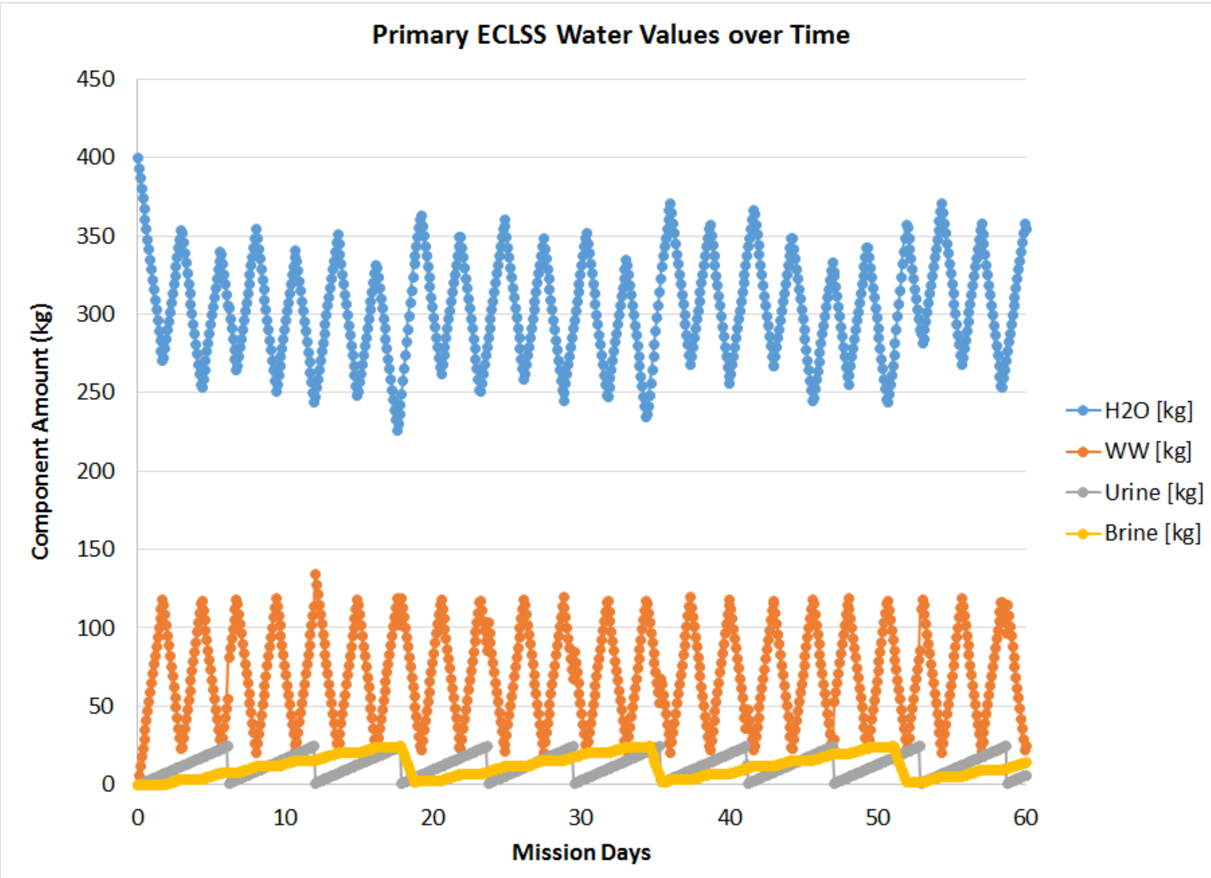


Figure 26: Water Management System Constituent Amounts in Cabin Over Time

WATER WALLS SYSTEM

As previously stated, the Water Walls System incorporates both highly passive ECLSS functions along with nutrient production and radiation protection for the crew. Specifically, the system addresses thermal, radiation, water, solids and air treatment (Flynn, et al., 2001). The basic process consists of a system of polyethylene bags in series utilizing forward osmosis and algae growth to filter wastes while allowing water to process through the series of Forward Osmosis (FO) bags. The Water Walls System offers a reliable ELCSS for long duration spaceflight due to the largely passive system design along with the added benefits of nutrient supplementation and radiation protection. It is assumed that the technology will be fully developed with a high TRL by the time of the mission.

Forward Osmosis (FO) bags are designated for specific waste filtration functions, and these bags are placed in series according to their function. Figure 27 shows a model of the Water Walls System organization. Once an FO bag has reached waste holding capacity, the bag ceases filtration operations and serves as a radiation shield with its organic waste inside. Aside from waste processing, the FO bags can serve as condensing heat exchangers for humidity control by using osmotic potentials to condense water vapor. FO bags filled with algae are also incorporated into the system to reduce the CO_2 and supply oxygen (Cohen and Flynn, GLEX 2012).

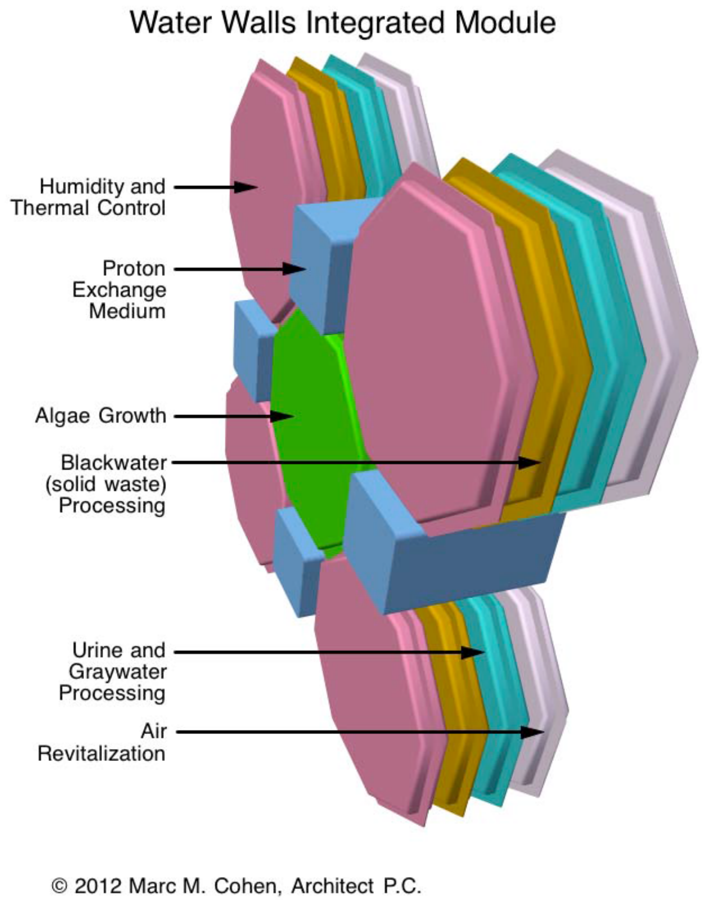


Figure 27: Water Walls FO Bag System Structure [Cohen and Flynn, GLEX 2012]

Streams are processed according to the waste being removed. Figure 28 shows a representation of the mass flows and processes occurring in the system. Forward osmosis pushes the water through the system of bags while leaving wastes (graywater, blackwater, and CO_2) behind in the bags. The bags can be replaced if necessary. Ground experiments on the system have shown a combined water recovery ratio of 99%. A flight test has been conducted on the urine and solid wastes processes in microgravity, but further testing and design modifications are necessary to ensure a high loop closure and water reclamation rate[Cohen and Flynn, GLEX 2012].

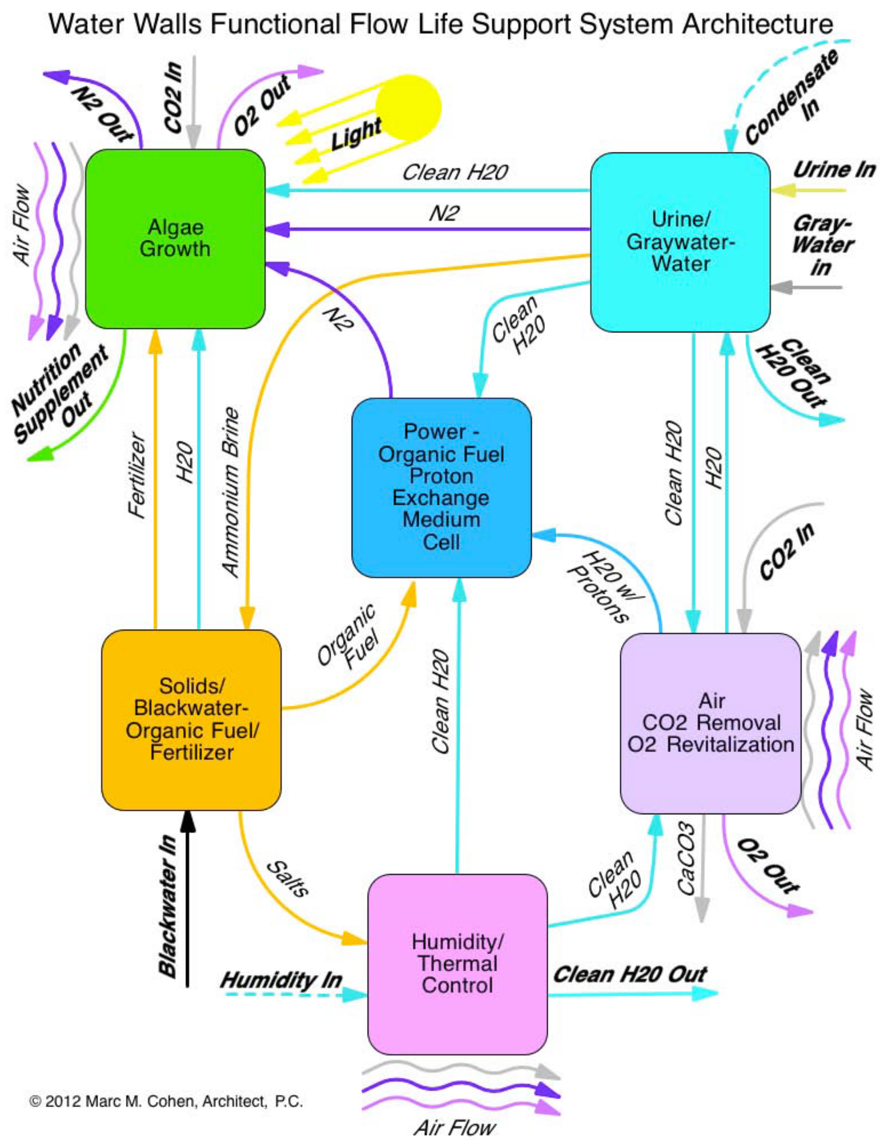


Figure 28: Water Walls Mass Flow Diagram [Cohen and Flynn, GLEX 2012]

Space Exploration Vehicle (SEV)

HABITAT DESIGN

The Space Exploration Vehicle (SEV) is a pressurized "roving vehicle" currently being developed at NASA Johnson Space Center that makes short duration sortie missions to perform exploration science. 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. Figure 29 shows the baseline SEV as developed at NASA while Figure 30 presents its notional system layout, as designed specifically for this mission. Moreover, Table 20 summarizes the key characteristics of the SEV.

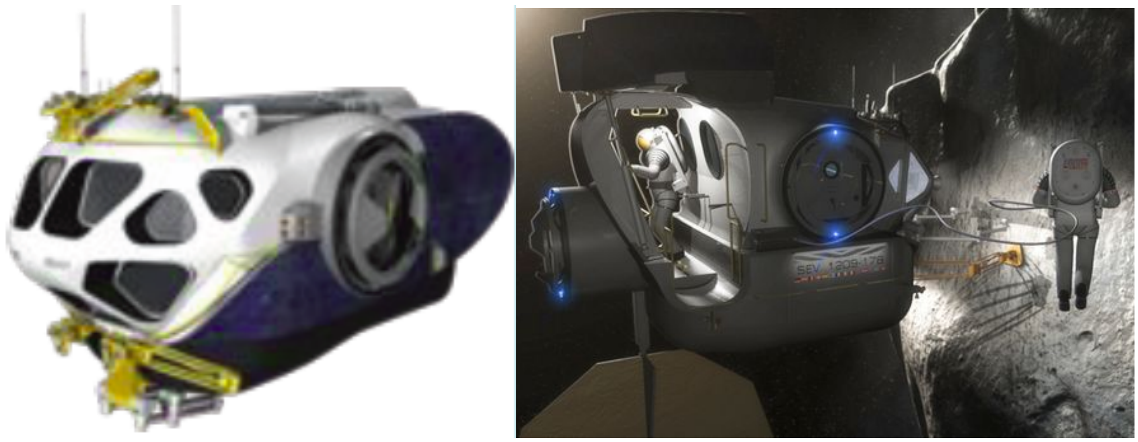


Figure 29: Space Exploration Vehicle - Front View (Left), Side View (Right)

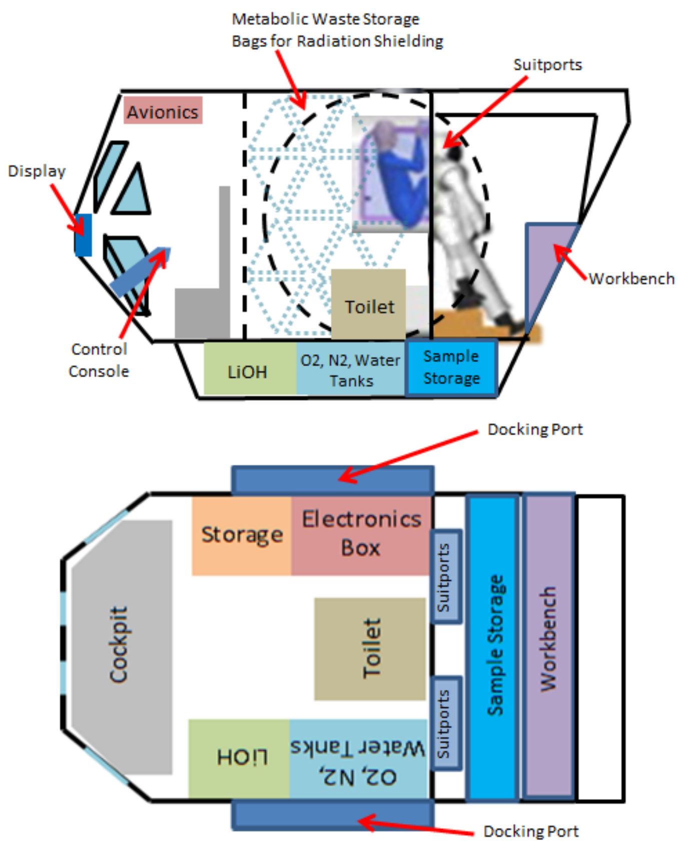


Figure 30: SEV Notional Plan View, Side View (above), Top View (below). Note the configuration by which a crewmember enters a spacesuit via a suitport in the top figure

Table 20: Summary of SEV Key Properties

Property	**Value**
Pressurized Volume	54 m^3
Dimensions	4.5m x 4m x 3m
Wet Mass	7T

As can be seen in Figure 30, the developed SEV employs the water-wall concept to perform both ECLSS and radiation shielding functions. For this particular implementation, the polyethylene water bags are used only for waste water and respiration product storage. This, along with the rest of the ECLSS architecture developed for this vehicle and this mission, are described in the following section.

LIFE SUPPORT

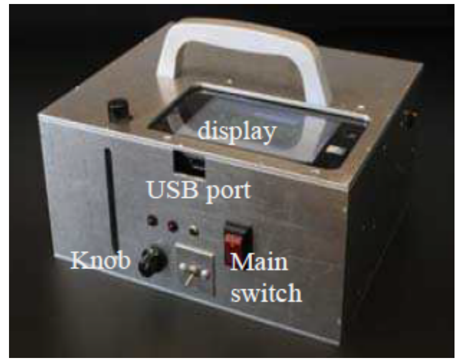
Based on the mission objectives discussed earlier, the SEV is required to sustain a two person crew for a maximum duration of 30 days. Because this duration is relatively short, an open loop architecture has been chosen for the SEV life support system. This ensures high reliability due to the need for less moving parts. Moreover, selecting an open loop architecture allows for commonality to be exploited with the Portable Life Support System (PLSS) of the space suits. Key examples of this include:

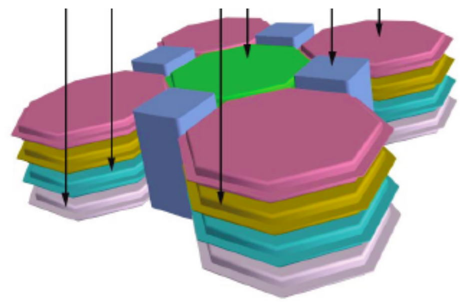
- Sharing the same water required for intravehicular activity (IVA) in the SEV, and extravehicular activity (EVA) in the same tank
- Sharing the same oxygen required for intravehicular activity (IVA) in the SEV, and extravehicular activity (EVA) in the same tank
- Employing the same Lithium Hydroxide (LiOH) CO_2 removing technology in both systems such that the same cartridges can be used in both the SEV and the EMU
- Employing the same activated carbon bed technology for trace contaminant control in both systems such that the same cartridges can be used in both the SEV and the EMU
- Using the same fan model for the air circulation circuits in both the EMU PLSS and the SEV - this minimizes the amount of spares required, and opens opportunities for parts scavenging in the event of a component level failure(Stambaugh 2012)

The following table summarizes these, as well as the other technologies selected to perform the various functions required for the SEV life support system. Please note that in this design, the atmosphere requirements were sized for three people rather than two, the amount of consumables was sized for three EMU PLSS's rather than two, and the amount of EVA consumables was sized for a three person crew, rather than a two person crew. This decision was based on current NASA guidelines for SEV sizing (Stambaugh 2012)

Table 21: Technology Choices and Consumable Masses for the SEV Life Support System

Function	**Technology Choice**	**Mass (kg)**	**Rationale**
Food Provisions	Food	63.9	Includes water in food and packaging mass
Food Provisions for EVA	Food bars	5.75	Sized for ten EVAs
Clothing	Shirts	20.58	Sized for two crewmembers over 30 days
Clothing after-life strategy	Use as radiation shielding	-	This was deemed to be a more efficient strategy than implementing laundry on any part of the mission
O2 Providing	Compressed O2 kept in Tanks Stored O2 O2 Leakage	34 92.4 0.4	Sized to meet IVA and EVA requirements of the crew, based on a scaling law provided in (Hanford 2004)
N2 Providing	Compressed N2 kept in Tanks Stored N2 N2 Leakage	178 104.7 0.015	Based on a scaling law provided in (Hanford 2004)
Potable Water Providing	Water Tank Stored H2O	197 328	Sized to meet IVA and EVA requirements of the crew, based on a scaling law of 60% structural efficiency
CO2 Removal	LiOH canisters	285	Accounts for the LiOH needs of crews during both IVA and EVA
Temperature and Humidity Control	Condensing Heat Exchanger	100	Standard component used on both ISS and the Space Shuttle Orbiter
Trace Contaminant Control	Expendable Activated Charcoal Beds	30	These are common components across the SEV and PLSS systems. They can hence be interchanged
Contaminant Monitoring	Portable Gas Analyzer - micro-Gas Chromatograph/Flame	4	This is currently under development but can be potentially used for both ECLSS and science purposes. See discussion

<table>
<tr><td></td><td>Ionization Detector (micro-GC/FID)</td><td></td><td>after this table


(Bae 2004)</td></tr>
<tr><td>Liquid and Solid Waste Management</td><td>“Water-Wall” - Polyethylene bags lined along the habitat wall</td><td>50</td><td>
Intended for dual-use as both a waste storage mechanism, and a radiation shield (Cohen 2012)</td></tr>
<tr><td>Fire Suppression</td><td>Water droplet fire extinguisher (16kg per unit)</td><td>16</td><td>The same as that used in the DSH

(Carriere, 2012)</td></tr>
<tr><td>TOTAL ECLSS WET MASS</td><td></td><td>1510</td><td></td></tr>
</table>

Of note from this table, is the use of the Portable Gas Analyzer - micro-Gas Chromatograph/Flame Ionization Detector (micro-GC/FID) currently under development. The small size of this unit allows it to be transported between the DSH and the SEV, as is required. In addition, its potential use in detecting organic compounds in the analysis of scientific samples has also been identified. This potential for multi-functionality and portability make this unit very attractive for the SEV life support system.

In addition, a strategy consisting of using all waste materials for radiation shielding has been employed. This can be seen via the end-of-life strategies for all clothing and metabolic waste. This strategy was selected based on the guideline of striving to maximize multifunctionality of selected components.

Orion Multi-Purpose Crew Vehicle (MPCV) & Extravehicular Mobility Unit (EMU)

Because we have chosen the baseline Orion MPCV and EMU Mark III model for the basic system architecture, a detailed discussion on their life support systems has been omitted. Instead, we present the characteristics of these systems that are pertinent to the overall mission. This is summarized in the figures and tables below.

Figure 31: NASA Mark III EMU (with PLSS designed to interface with suitport)

Table 22: NASA Mark III EMU Primary Characteristics

Component	**Mass (kg)**	**Comments**
Spacesuit Assembly (SSA)	54.8	Includes the helmet, visor, gloves, and all components of the spacesuit and pressure garment assembly
Portable Life Support System (PLSS)	87.5	Includes the common CO2 and trace contaminant control cartridges employed in the SEV ECLS system, as well as batteries to operate the PLSS unit
TOTAL PER SPACESUIT	**142.3**	
TOTAL FOR 3 SPACESUITS[1]	**426.9**	

[1]As previously mentioned, the sizing of the SEV ECLSS system has been sized for three spacesuits, even though there are only two crew members. The intent of the third unit is to be a flight spare, that can be used as a single unit, or cannibalized on orbit if the need arises.

Figure 32: Orion Multi-Purpose Crew Vehicle

Table 23: Orion Multi-Purpose Crew Vehicle Primary Characteristics

Property	**Value**
Pressurized Volume	19.56 m^3
Dimensions	Cone of height ~3m and diameter 5m
Inert Mass (Command Module + Service Module)	9.8T + 4.5T = 14.3T

Biological Science Payload

Payload Manifest:

- International Standard Payload Rack (2)
- In-flight Bioscience Instrument Suite
- Use of two modified-MELFI pods
- Use of medical suite for sample-taking

The In-flight Bioscience Instrument Suite will be required to achieve the Mars surface mission-facilitating objectives of this mission. Monitoring a panel of proteins and ions in blood, urine, and saliva at weekly or monthly intervals will allow for crucial in-flight changes in physiological degradation mitigation efforts in a way that is customized for each crew member. Additionally, in-flight data-taking and transmission to earth would allow for reduced mass and refrigeration requirements in the re-entry capsule. One of the DSH's two International Standard Payload Racks (ISPR) would contain NASA-sponsored project hardware intended to analyze the concentration of a range of proteins and ions. A protein analyzer could employ mass spectrometry to measure levels of large non-protein molecules and small proteins. Ion detection could be conducted through the development of space-rated clinical chemistry analyzers. Vibration isolation may be necessary during equipment operation. While the technology needed to achieve these goals has not been demonstrated in space to date, much of the equipment is of sufficiently low mass to enable microgravity testing at its current TRL. The demand for reliable, portable, and high-resolution medical equipment will drive development in tandem with concerted NASA development efforts.

The second ISPR will be used to accommodate experiments selected by peer review from a pool of academic and industrial proposals. These experiments will be conducted in ISPR lockers that must be able to export return samples to a standardized canister for temperature-controlled return to Earth using volume-saving GLACIER technology. Externally-solicited experiments will allow scientists to ask cutting-edge questions about the effects of microgravity and deep space radiation exposure on biological materials, chemical systems, physical phenomena, and demonstration technologies. Microbiome analysis is an example of a potential investigation-of-merit for the mission. Experiments not requiring dedicated payload mass include psychological investigations composed only of computerized surveys and medical studies including the systematic review of medical events, medical telemetry data, and other health-related data sources including ultrasound imaging.

Bioscience operations will continue past the mission's return to Earth. The impact of dust exposure on model organisms, including inhalation-induced lung damage, must be evaluated. If technology permits, the crew could also undergo whole genome sequencing of sufficient read depth, potentially with a variety of tissues and even single cell targets, to facilitate comparison with pre-mission genome sequences for the purpose of radiation exposure indexing.

Radiation Monitoring

Tissue Equivalent Proportional Counters (TEPCs), which are currently in use on the ISS, measure radiation doses and dose equivalents for complex radiation fields and should be deployed in several locations in the DSH and in/on the SEV to measure radiation levels during transit, exploration, and EVA. Radiation levels throughout the DSH can be 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.

SPE monitoring will be conducted using the existing network of solar observatories (i.e., SDO, SOHO, GOES) and any future expansion.

RADIATION 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 [Cucinotta, 2012]. Radiation shields that incorporate low atomic mass materials are capable of suppressing damaging secondary radiation in the form of neutrons that is formed during particle transit through the aluminum hull of a module.

RADIATION EXPOSURE ESTIMATES:

NASA has calculated the safe number of days that a person can travel in space when their vehicle includes 20 g/cm^2 of shielding (See figure below). These values are based on the need to prevent astronauts from exceeding an increased risk of 3% for Radiation Exposure Induced Death (at 95% confidence).

Age at Exposure	NASA 2012 US Average	NASA 2012 Never Smokers
MALES		
35	306 (357)	395 (458)
45	344 (397)	456 (526)
55	367 (460)	500 (615)
FEMALES		
35	144 (187)	276 (325)
45	187 (232)	319 (394)
55	227 (282)	383 (472)

Table 24: Safe days in space for males and females of various age groups comparing the US average to the values for never smokers. Values in parentheses account for the presence of a radiation storm shelter. From Turner, R. "Radiation Risks and Challenges Associated with Human Missions to Phobos/Deimos." CSC: March 2013.

Failure to mitigate the effects of GCR and SPEs could lead to acute health effects including radiation sickness with incapacitation or death. Long-term risks include carcinogenesis, neural tissue damage, stem cell disturbances (bone marrow, etc.), and cataracts.

The biological impact of particles of high mass and energy (HZEs) is yet to be fully analyzed. The mission's bioscience payload racks will facilitate a unique perspective on HZE-tissue interactions.

GCR-SPECIFIC EXPOSURE

GCR Exposure without shielding (EVA): 0.959 mSv/day

GCR Exposure with 20g/cm^2 shielding (DSH, SEV): 0.795 mSv/day

GCR - Estimated Mission Total (461 days shielded, 4 days EVA): 370.3 mSv

Permissible Exposure Limits at Solar Maximum:

Age (years)	30	40	50	60
Male, Never-Smoker	780 mSv	880 mSv	1000 mSv	1170 mSv
Female, Never-Smoker	600 mSv	700 mSv	820 mSv	980 mSv

Table 25: Career-length Permissible Exposure Limits for <3% REID. From Turner, R. "Radiation Risks and Challenges Associated with Human Missions to Phobos/Deimos." CSC: March 2013.

DSH

The maximum proton flux incident on the habitat occurs on the return voyage from the Martian system, at a Helio radius of approximately 1 AU. At this distance the habitat requires an additional 9 g/cm^2 in addition to its baseline 11 g/cm^2 water and polyethylene environmental protection layer (Smitherman, 2012). This requires additional shielding material to meet the 20 g/cm^2 mandate, which will be achieved by applying a single layer of Water-Wall bags on the interior surface area in order to achieve parallel objectives with ECLSS. This protection ensures the entire crew cabin is shielded from both SPE and GCR, with crew activities to remain undisturbed in case of an event. The large mass (11,400 kg) is deemed justified as it also forms part of the water supply and waste management system for the duration of the mission.

SEV

Due to the difference in orbital location between Mars and Earth, the flux of protons incident on the SEV whilst conducting operations in the Martian system may be scaled proportional to $1/R^{3.3}$ (Lario et al, 2006). This leads to a required shielding thickness of 5 g/cm^2, provided by Water Wall bags for the same dual-purpose reasoning as the DSH. In contrast to the DSH, the SEV will utilize the moon body as protection on the underside of the vehicle and Water Wall bags on the roof, port and starboard walls. The protection will cover approximately half of the vehicle to create a habitable zone whilst minimizing vehicle weight.

Human Factors

The goals of the Asaph 1 Mission provide human spaceflight experts with a remarkable series of engineering and science challenges. The ensuing paragraphs demonstrate a comprehensive plan for astronaut health and safety, using systems that have demonstrated high degrees of technological readiness. The development of new countermeasures is also discussed.

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 [Kanas and Manzey, 2008]. 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 an FAA first-class examination (analogous to space shuttle pilot clearance). Select-in procedures including detailed psychological examinations, facilitated by psychiatrists and PhD-level psychologists, will then be used to refine the applicant pool into a set of primary and alternate crew members with personalities that indicate a high likelihood for enduring harmonious interaction, as well as for adaptability, effective stress management, and mission preservation-promoting qualities.

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. Remaining life years help dictate the risk of carcinogenesis in radiation-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. The proportion of women having experienced menopause at age 40 is 1% and rises to 10% at age 46. This indicates that 9.1% of women that are pre-menopausal at age 40 will enter menopause by age 46. This physiological consideration may be of concern during crew selection. Although women have lower radiation tolerances, as defined by NASA's 3% 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 their gender may face.

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. The primary crew medical officer could be a physician-astronaut or other experienced care provider, but mission medical requirements can be met via medical training of any astronaut.

Crew Health Care

MEDICAL CARE

A mission to Phobos demands a comprehensive medical care system to ensure optimal astronaut health within reasonable mass limits. This habitat subsystem must be able to support a wide variety of procedures, including undertakings as significant as major surgery. Certain parameters must also be monitored continuously to ensure health during extended deep space travel, like optic nerve/retinal changes, renal ion excretion, and vestibular/sensory alterations.

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.

PSYCHOLOGICAL CONSIDERATIONS

Long-term spaceflight produces extreme psychological stress which, if ignored, can result in serious degradation of mental health that put the mission and the crew at risk. For example, interpersonal conflicts between crewmembers have been demonstrated to be mission-terminating events. Mission stress is generated by a number of factors that may be mitigated with adequate foresight. Major sources of psychological stress include:

- Isolation in an enclosed space
- Tense relationships between crewmembers
- Physical deterioration from long-term presence in microgravity
- Prolonged separation from family
- Lack of privacy from other crewmembers

Countermeasures

MITIGATION OF PHYSIOLOGICAL EFFECTS FROM SPACEFLIGHT

Table 26: Physiological systems with corresponding risks and mitigation strategies [Buckley 2006]

System	Risk	Mitigation Strategy
Musculoskeletal	Debilitation, degradation, bone fracture	Gravity Loading Countermeasure Skinsuit developed at MIT [Waldie 2010]. 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 2.
Cardiovascular	Cardiovascular degradation, cardiac events, heart disease	Cardiovascular exercise to keep cardiac muscle strong
Vestibular	Motion sickness	Crew training prior to mission to aid in vestibular adaptability, motion sickness medication
Immune System	Immunosuppression	Manage crew stress, sleep deprivation, and provide adequate nutrition
Nutrition	Inadequate nutrition	Provide adequate nutrition to meet crew metabolic and health demands
Depressurization	Decompression sickness (DCS)	Decompression procedures that follow NASA standard pre-breath protocols to prevent DCS prior to EVA
Hearing	Hearing loss due to noise levels of cabin	Implement NASA standards for allowable noise levels

	systems	from all systems and payloads
Trauma/Illness	Impaired crew	Medical kits, crew health care system, surgical tools

COUNTERMEASURE DEVELOPMENT

Significant opportunities for microgravity effect and countermeasure research exist during the twenty years leading up to the mission. Year-long ISS expeditions, which will begin in 2015, will significantly increase the amount of data available for addressing astronaut health during long-duration space missions. ISS-based research will also allow for the development of countermeasures that are currently at low TRL stages. NASA-defined areas-of-weakness in our space medical knowledge and potential compensatory investigations are as follows:

Musculoskeletal/Renal: Study hormones impacting serum ion concentrations, traditional pharmacological inhibition of bone loss, and use of a Sclerostin antagonist (antibody) for bone density loading

Immunological: Proteomics for the study of cytokine/chemokine flux impact

Ophthalmological: Papilledema development and mitigation must be investigated to prevent retinal damage in the eye [Mader 2011].

Neurological: Evaluation of potential central or peripheral nervous system degradation

Behavioral: Evaluation of optimal strategies for psychological stability

Radiation Damage Mitigation: Investigate use of antioxidants, etc.

Nutritional: Investigate Vitamin D and other nutrient levels, etc.

Cardiac Arrhythmia: ECG studies to search for coherent etiology

More thoroughly addressing the health effects of long-duration spaceflight using pre-mission ISS-based research will allow mid-mission bioscience to focus on the unique problems of Mars system expeditions, including elevated radiation exposure.

PSYCHOLOGICAL COUNTERMEASURES

Crew training and mission preparation allows crewmembers to get to know each other before spaceflight, to develop as a team, and to build trust. These are key elements to carrying out a successful mission, and they have a strong history of improving crew relationships and preventing interpersonal conflicts in long duration or particularly stressful spaceflight missions. Throughout the mission, even time-delayed communication opportunities with family on Earth can help preserve crew morale. Maintaining a strong routine and work schedule during extended transit to and from the Mars system is crucial in preventing situational stresses. To provide this routine, crewmembers will carry out scientific experiments in transit. These include both biological experiments (self-monitoring and biological sample collection) to understand the effects of deep space travel on humans and extended microgravity experiments solicited from and specified by universities/students for public outreach. At least one of these experiments will be reserved specifically to involve live vegetation, as the presence of plant life has been proven to increase astronaut morale and to reduce stress. The presence of multiple experiments also provides variation in work type which will minimize monotony for the crew. A rigorous exercise regime will also be a part of the astronaut's daily routine. Apart from preventing physical degradation of crewmembers, exercise provides an excellent outlet for stress relief. Large habitat size also helps to maintain crew happiness.

Surface and Science Mission Operations

SEV Landing, Anchoring, and Departure

Owing to their extremely low gravity and possible captured-body origin, Phobos and Deimos can be treated like asteroids for the purposes of landing, anchoring, and moving about the surface. Unlike most asteroids, however, Phobos may be covered by a meter to several 10s of meters of powdery regolith (Castillo-Rogez, 2013). The science team has attempted to select science targets in locations showing probable bedrock, both to improve mobility and to investigate the interior composition of the moon. However, until the precursor mission's landers and orbiter return detailed data, the exact surface properties of the two moons will remain unknown. Therefore, several options for anchoring are presented, with the qualifier that an anchoring strategy will be selected, and a system developed, in the interim before SEV launch. Small oxygen/methane reaction control system thrusters, such as those developed by Dynetics will be used for maneuvering the lander down to the surface. Since the thrusters must be pointed away from the surface for this operation, no significant dust scouring is anticipated.

It may be possible to "dock" a lander with the surface, through the continuous use of thrusters for stationkeeping. While continuous fuel use is undesirable, it remains a viable backup option in case other anchoring methods fail, or in case the surface properties cannot be determined before landing. This will be aided by Phobos' small but present gravity. With gravity ranging from 0.0019–0.0084 m/s^2, it will take approximately 16 to 32 seconds for an object to fall 1 meter, unassisted. If an object is not accelerated past the escape velocity of Phobos, it will eventually come to rest on the surface of the moon.

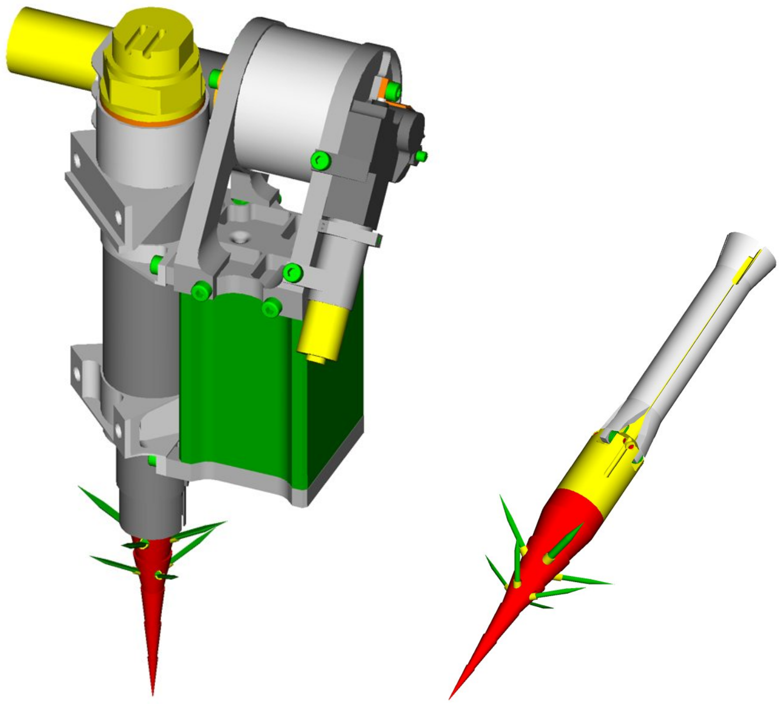


Figure 33: CAD drawing of representative anchoring harpoon for a lander, from Rosetta's Philae lander.

Figure 34: Photograph of earth anchor used for sand and loose soil. Something similar could be used to anchor in regolith.

A more plausible use of thrusters is to provide counteracting forces to hold the lander in place as it anchors to the surface. This could be done with harpoons, as described by Basso et al.(2006), Buet (2013), and Ross (2001), or as Rosetta's Philae lander plans to do (Glassmeier et al., 2007). Similarly, the rover could drill or auger into the surface (Basso et al., 2006; Ross, 2001; Buet, 2013). These mechanisms and variations on them are by far the most common proposed anchoring strategies; others have suggested using large-area augers or screw-plates, pitons, welded tie-downs, fluked anchors, or contra-rotating screws to burrow (Ross, 2001). The most common anchoring mechanisms are currently being studied by NEEMO 16 (NEEMO Fact Sheet, 2012). While an auger or screw could presumably be removed by rotating the mechanism backwards, harpoons are more permanent and would need to be released with pyro kinetics. Still, for a limited number of landings, harpoons and augers are the most attractive strategies.

There are several alternate, low-TRL anchoring strategies being proposed and studied. For an excellent summary on robotic mobility in microgravity, which relates to anchoring, see the summary paper by Tunstel and Palmer (Tunstel and Palmer, 2010). These alternate strategies include:

1. Velcro-like microspines, as developed by Aaron Parness at JPL. Not only have the microspines been thoroughly prototyped in several generations of climbing robot, but recent iterations are also detachable, rendering this anchoring strategy highly desirable. (Parness and Frost, 2012; Parness, 2011).

2. Adhesive feet, such as those in development by Northrop Grumman for the Automated Walking Inspection and Maintenance Robot (AWIMR) project.

3. Tie the spacecraft down with a rope passing around the entire moon (Ross, 2001).
4. Attach a net and moving around along its cables (Prado).

While the latter suggestions seem implausible, or simply far from ready, the detachable microspines show promise in the event that rocky surfaces are encountered.

One or more of these anchoring systems can be selected based on input from the precursor mission. To move to another landing site, the SEV departs the surface, moves around Phobos, reattaches itself, and then departs again. In order to prevent unnecessary dust plumes, the SEV can depart without using any thrusters. Instead, it bends its robotic arm underneath it and uses it to push off the surface from directly beneath the SEV's center of mass. Holding a 13-ton SEV on the ~0.006 m/s^2 gravity surface requires only 78 N, a relatively small force for which the arm should already be designed to withstand.

EVA Operations with SEV on Phobos

The purpose of the EVA portion of the mission is "to put boots on the ground" in order to investigate the geology in person and to select important geological samples. This EVA portion of the mission is designed for a Phobian surface exploration time to last for nominally 14 days. This two-week period is constrained by the planned trajectories for the Mothership assembly and the separation of the SEV from the Mars orbit. Safe duration of an EVA with state-of-the art technology is about 4 hours. Allowing for rest periods, and anticipated constraints of the PLSS, this allows a maximum of ten EVAs. To include contingency PLSS operational supply for unanticipated surface activity, a target number of eight EVAs was chosen – four at each of the two preselected Phobos landing sites. The timeline for these eight EVA operations spread over the 14-day mission duration is outlined in Table 27. Upon return to the DSH, the SEV will be returned to Phobos and re-anchor to the surface to prevent drift-off risk and accommodate planetary protection requirements. The crew on the DSH will be able to operate the SEV robotic arm system remotely to preform experiments and demonstrate feasibility of preforming telerobotic operations from orbit.

Table 27: Concept of operations and EVA schedule at Phobian surface

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

Day 14	Contingency – rest day, post mission activities	

The two astronauts in the SEV are on rotation for EVAs, so that one remains in the SEV both to monitor the PLSS status for and to maintain constant communication with the astronaut on EVA. Suit-ports on the SEV allow the SEV crewmember easy access to quickly assist the EVA astronaut if the need arises. Having crewmembers on rotation rather than having a dedicated SEV pilot allows adequate rest periods for astronauts between EVAs. The contingency rest period (day 14) allows some flexibility in the overall EVA schedule, such that the mission can accommodate unforeseen schedule changes.

Upon reaching each landing site, the SEV will land and anchor to the surface. The characteristics of the site's surface material (grain size, depth, density, cohesion) determined by the precursor mission will determine the type of anchoring mechanism selected and installed on the SEV. These surface characteristics will also determine the EVA mode used to carry out operations at each site. There are two modes described in detail below. Each includes use of a Manned Maneuvering Unit (MMU). Regardless of use for mobility during normal operations, the MMU may be used for emergencies in which the Astronaut is disconnected from the SEV.

EVA MODE 1 – RIGID RESTRAINT TO SEV

Outside the SEV, the astronaut's feet are fixed to the end of a robotic arm on the SEV. This provides high mobility and function for the astronaut when handling complex surface equipment or tasks, while limiting interaction with a thick (1m or greater) surface layer of dust. Astronauts remain tethered to the SEV for safety. This system is similar to an EVA method explored during the NEEMO underwater asteroid-analogue missions, and was found to provide the most mobility for manipulation of equipment in a microgravity environment (Chappell, 2013).

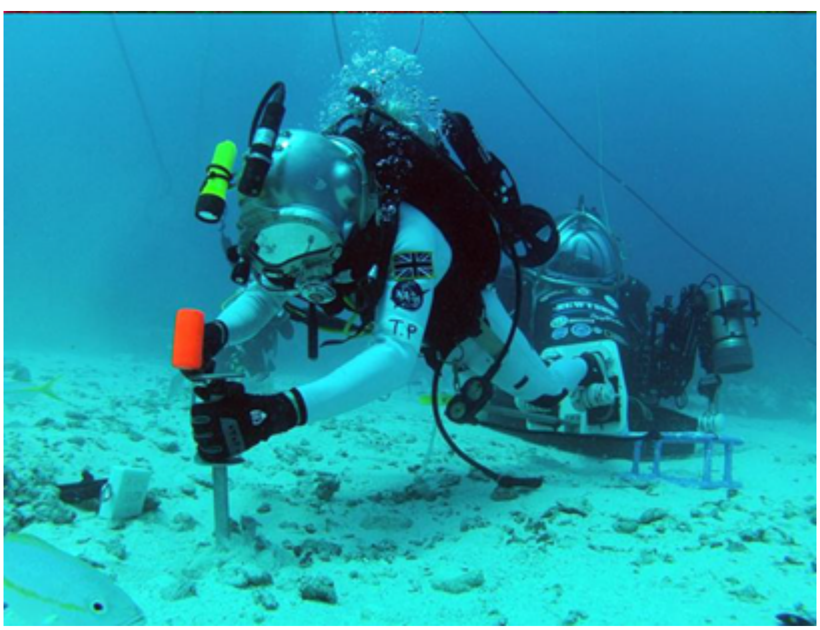


Figure 35: NEEMO feet-fixed EVA method

EVA MODE 2 – SINGLE POINT ANCHORS WITH SEV TETHER

Astronauts on EVA are anchored to the surface by an individual anchor for stability. These anchors are the same design as the anchors used for scientific equipment being left behind for post-mission studies. Astronauts remain tethered to the spacecraft and the anchor during mission activities. This mode is designed for rockier or gravelly surfaces. Thus, the crewmember may navigate to a point on the terrain on the tether alone using MMU, hands, and feet, then anchor themselves to remain stationary while carrying out EVA objectives. Because a tether can be far longer than the robotic arm, a tether allows exploration of a larger area in a single EVA than the rigid restraint mode and is the preferred method for boots-on-the-ground science.

Figure 36: Concept drawing of astronaut tethered to SEV using MMU

SAMPLE COLLECTION AND CORE DRILLING

Surface sample collection involves standard surface rock collection and gathering scoops of surface regolith. A total of 30 rock samples with an approximate volume of 1.5 L per sample will be collected from a total of two sites (~15 at each site). Forty sample bags will be included in the SEV for rock sampling. For return-to-Earth weight considerations, we have used an approximate density of serpentine, ~2.7 g/cm^3, for the rock samples. Because the surface of Phobos is a micro-gravity environment, and for reasons of possible damage and injury caused by shrapnel, breaking rocks with a hammer will be discouraged. A total of 12 soil samples (~6 at each site) will be collected in 0.5 L bottles using a standard-sized (~10-15 cm length) gardening trowel. Eighteen soil bottles will be provided in the SEV. In addition to the rock and soil samples, thin-layer (<1 cm thick) samples of surface dust will be collected using thin sticky-pad collection devices laid on an undisturbed surface.

A secondary goal to collecting surface samples on Phobos is to drill two core holes that are ~3 m deep with a vertical orientation with the PRSC instrument (Table 3). Core diameter will be in the range of 40-50 mm. The drill will be a percussive hammer drill with diamond cutters, a hollow drill stem, and a sample recovery sleeve inside the barrel (Figure 37). It will have an electric motor with an approximate power requirement of 1100 W. Drilling will be done dry. A problem encountered on the Apollo 15 mission with the use of a small core drill was in recovering the core. A foot treadle was incorporated into the design, and core was successfully recovered on Apollo 16 and 17. A similar contingency design should be considered for this mission. Drill samples will remain in their sleeves and will be placed in cryogenic storage for return to Earth. An additional thickness of high-density polyethylene will envelope the core to act as shielding against radiation. Drill selection and small design optimization will be dependent on the results of the precursor mission, after which drill-site characteristics will be known.

Figure 37: Examples of prototype portable drills to be used in space. (left) Example of the drill used on Apollo 15, 16, and 17 missions (http://www.hq.nasa.gov/alsj/a15/a15carrier1.jpg); (right) Commercial "backpacker drill" sold by Shaw Tool Company used for mineral prospecting in remote locations. This gasoline-powered drill would need to be retrofitted with an electric motor. (http://www.backpackdrill.com/images/home003.JPG; image care of Shaw Tool Co.)

SAMPLE STORAGE OPERATIONS

Each surface sample collected on Phobos will be placed immediately into a sample collection bag, sealed, and labeled. These surface samples will be stored in an exterior containment unit fixed to the SEV suit-port porch. During rendezvous with the mothership, this sample container will be brought into the SEV airlock with the robotic arm, allowing crew to transport it directly to the sample cryo-containment lockers on the DSH as soon as possible after docking. This configuration maintains sample preservation and containment while minimizing total SEV mass. Core samples require a more restrictive containment environment and will be stored on-board the SEV in sample cryo-containment lockers. Once docked with the mothership, these lockers will be transferred from their cooling unit on the SEV to the unit on the DSH. Measures will be taken to prevent any cross-contamination between Phobos sample material and the DSH environment (see section on Planetary Protection). All samples will be moved to the Multi-Purpose Crew Vehicle for reentry to Earth.

FREEZER ASSEMBLY

A freezer based on the Minus Eighty Degree Laboratory Freezer for ISS (MELFI) architecture (four storage units each divided into four subsections) will employ a modified cooling mechanism based on the General Laboratory Active Cryogenic ISS 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 (Fig. 38)

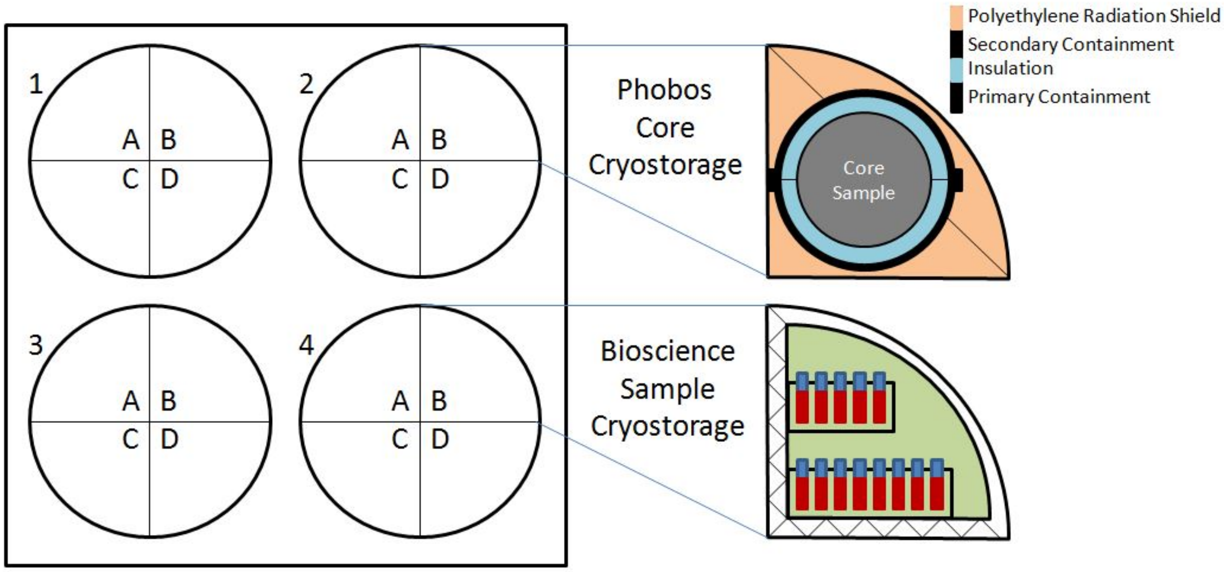


Figure 38: Schematic diagram of the sample storage locker configuration on the SEV. A similar locker configuration will be on the DSH for the return transit to Earth.

SEISMOMETER EXPERIMENTS

As indicated on days 3 and 10 of the surface operations schedule, passive seismometers (Table 3) will be placed on the surface to investigate the internal stresses of Phobos. These devices must be placed at specific distances from each other at each landing site, with the exact location of each instrument known. Because of these constraints, the final configuration of these experiments is dependent upon the data from the precursor mission and the EVA mode being used at each location. EVA mode 2 is preferable for this experiment, as it allows the individual seismometers to be placed farther apart. Depending on the time and tethering constraints, an active-source seismometer (vibroseis-type) should be considered to perform an active seismic survey.

RETRO-REFLECTOR

A retro-reflector is essentially a mirror than is used to reflect an electromagnetic signal back to its source (Table 3). The reason to place this mirror on Phobos is for future use when a signal can be directed from the Martian surface up to Phobos to determine the orbital distance of Phobos. This signaling may be achieved by either rovers or humans. The reflector measurement is regularly made over a long period of time (decades, centuries) to determine deviations in orbit. For Phobos, in particular, it will help determine the rate at which it is encroaching on Mars (Bills et al., 2005).

Structures

Mass Budget

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 values for mass, volume and power are based on existing systems (ISS ECLSS, MRO), design studies (DSH, PROP) and calculations for aspects specific to the Asaph mission (propellant mass, EPS).

	Unit	Mass (t)	Power (kW)
Phase 1	Precursor		
Precursor	DE	1	0.1
	PE	1	0.1
	DPS	2.18	2
	SEP	0.56	7.1
	Propellant	1.75	0
	Public Outreach Payload	1	0
	Total Precursor Dry Mass (incl. margins)	5.74	**9.3**
	Total Precursor Wet Mass (incl. margins)	**7.49**	
Phase 2	Cargo		
DSH	Structure	11.75	0
	ECLSS	5.4	3.8
	EPS	1.02	0
	TCS	0.56	0.29
	Consumables	2.33	0
	Comm and Avionics Subsystem	0.1	0.2
	Attitude Determination & Control System (ADCS)	0.7	1
	Medical Suite	1.5	1
	Exercise Equipment	0.54	0.55
	Science Equipment	3.6	2.88
	Cryocontainment Locker (part of Sci. Equip.)	0	0
	Environmental Protection (Baseline)	3.99	0
	Environmental Protection (Rmin = 1 AU)	11.44	0
	Crew Systems	0.78	0
	Sample Sterilization Airlock	0.75	0.6
	Non-propellant fluids	0.23	0
	Total DSH Wet Mass (incl. 15% margins)	**44.78**	**10.32**
SEV	Structure	1.77	0
	Radiation Shielding	1.15	0
	ECLSS	1.92	0.25
	EPS of SEV	0.32	0
	TCS of SEV	0.18	0.11
	Science equipment on SEV	0.07	1.5
	Public Outreach Payload	0.26	0
	Sample Cryo-containment Locker	0.38	0.5
	Attitude Determination & Control System (ADCS)	0.7	1
	Medical Kit	0.1	0
	Communications and Avionics	0.1	0.28
	Robotic arm	0.06	1

	PPM	0.65	0
	Propellant	4.3	0
	Total	11.94	**4.64**
	Total SEV Wet Mass (incl. 15% margins)	**12.52**	
PROP STAGE 1	NTP Engine	13.33	4
	Propellant tank with engine module	7.7949	0
	Propellant with engine	28.87	0
	5 Tanks dry mass	49.8825	0
	5 Tanks propellant	184.75	0
	Battery	0.2	0
PROP STAGE 2	NTP Engine	13.33	4
	Propellant tank with engine	7.7949	0
	Propellant with engine	28.87	0
	2 Tanks dry mass	19.953	0
	2 Tanks propellant	73.9	0
	Battery	0.2	0
	TOTAL PROP (inlc. margins)	**428.88**	**8**
Phase 3	Taxi		
	Crew module	7.9	
	Service module	20	9.15
	TOTAL MPCV (inlc. margins)	**27.9**	**9.15**

TOTAL MASS IN LEO (inlc. margins) 513.5 t

Manufacturing Scheme

With a majority of the vehicle modules available for direct purchase (by the expected timeline), structural modifications/designs are largely related to retrofitting tasks required to tailor the vehicles to the Asaph mission profile.

One aspect of the mission that will require large-scale manufacturing is the generation of the fairings for the modified launch program. More detailed information of the alterations is available in Section 3 of Propulsion and Launch Vehicle Selection.

Assembly in Orbit

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. While this manner of task has already been undertaken during the assembly of the ISS (NASA, 1999) at reduced range, the core nature of the procedure remains common. One of the driving requirements to ensure the transition from in-situ assembly, such as the ISS, to completely remote orbit docking is visibility, which is an achievable task with current technology. Sufficient training on remote assembly will be required for mission success, though given the scope of this project an exact timeline has not been determined.

Collapsible Structures

Given the size of some of the solar arrays required for mission power supply (134 m^2), collapsible structures represents an attractive option to increase the flexibility of payload allocation on scheduled launch vehicles and thus decrease the gross number of launches. 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 this technology.

Attitude Determination & Control System

Essentially all control systems require two types of hardware components: sensors and actuators. Sensors are used to sense or measure the state of the system, and actuators are used to adjust the state of the system. Similarly, the attitude determination and control system for the proposed design typically uses a variety of sensors and actuators. For a better modularization, ADCS has further been divided into Attitude Determination and Attitude Control.

Attitude Determination

In the proposed mission, the attitudinal states of all physical stages are described by three angular variables along x, y and z axes. The coordinate frame is always Body-Centered-Inertial (BCI). A brief trade study was performed to select appropriate Attitude sensors. The trade was conducted with the single point requirement of reliability and redundancy. Below is the summarized result of the same:

ATTITUDE DETERMINATION

Sensor	Accuracy	Characteristics and Applicability
Magnetometers	1.0° (5000 km alt) 5.0° (200 km alt)	Attitude measured relative to Earth's local magnetic field. Magnetic field uncertainties and variability dominate accuracy. Usable only below $\sim$ 6,000 km.
Earth sensors	0.05° (GEO) 0.1° (LEO)	Horizon uncertainties dominate accuracy. Highly accurate units use scanning.
Sun sensors	0.01°	Typical field of view ±30°
Star sensors	2 arc-sec	Typical field of view ±6°
Gyroscopes	0.001 deg/hr	Normal use involves periodically resetting reference.
Directional antennas	0.01° to 0.5°	Typically 1% of the antenna beamwidth

Figure 39

At the end of the study it was decided that all stages of mission on both precursor and main mission 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

The system will primarily depend on star tracker for the attitude determination. As a preliminary choice, it is proposed that a system similar to sensing system onboard Clementine Star Tracker Cameras [NSSDC ID: 1994-004A-07] be used. The sensor on board has an extremely low mass of about 300 grams. The star tracker will have full sky map due to the nature of the mission, involving multiple orbital configuration. During times of sun saturation, the system will fall back on sun position sensor which can derive heritage from GOES-15 [NSSDC ID: 2010-008A].

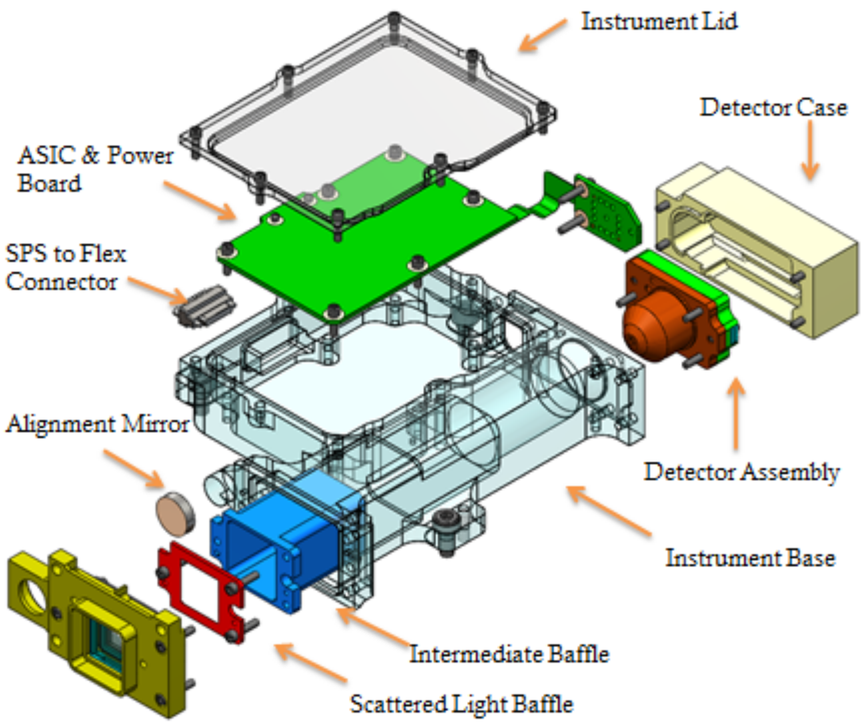


Figure 40: Exploded view of GOES-SPS

It is believed with a high level of confidence that the above stated two-line system will be reliable. However, as a last line of defense, in case of complete temporary failure, attitude determination can still be performed to within reasonable accuracy using 'see and follow' philosophy depending on sight-sextant.

Attitude Control

The difference between the desired and measured attitude states is fed into an Attitude Control System which in turn physically corrects the attitude. Several strategies can be employed to achieve this. Some of the options considered for the proposed mission are as follows:

ATTITUDE CONTROL

Method	Accuracy	Characteristics and Applicability
Thruster	High	High fuel usage, engine wear, but proven and easy
Spin Stabilization	Medium	Lack of high accuracy, difficult on multiple axes
Momentum Wheels	High	Very heavy and low energy efficiency
Control Moment Gyros	Very High	Costly and little massive, but very reliable and efficient
Solar Sails	Very High	Relatively cheap, deployment dependent, extremely weak
Gravity Gradient Stabilization	Very High	Applicable only for long objects, needs high gravity environment
Magnetic Torquers	High	Dependent on Magnetic field

Figure 41

At the end of the study it was decided that SEV and DSH be equipped with a combination of a Control Moment Gyroscope (CMG) and Monopropellant Hydrazine Thrusters and all stages of phase I and smaller stages of phase II be equipped with just Monopropellant Hydrazine Thrusters. Rationale behind the selection was:

1. Non-dependence on magnetic field, gravity gradient,
2. Tried and tested nature of technologies involved
3. Sufficiently capability for relatively fast maneuvers
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 CMG's be used on SEV and DSH. In addition there should also be a backup system of monoprop thrusters using Hydrazine. This will provide a three-axis control with built-in contingency fall back while maintaining simplicity and reliability. It is believed with a high level of confidence that the above stated two-line system will be reliable. However, as an absolute last line of defense, in case of complete failure, SEV's attitude can be somewhat controlled by robotic arm in the proximity of the Phobian surface.

The following is a broad quantitative justification for the above decision using first order approximations:

SEV

Mass: 14 metric tonne

Torque capacity of modern day CMG with a 100 kg is about ~ 2000 Nm

Along perpendicular Axis (y -axis and z-axis)

Moment of Inertia along principal axis: 38,573 kg·m^2

Start Slew Rate: ~ 2.5 °/sec

Along Transverse Axis

Moment of Inertia along principal axis: 53,235 kg·m^2

Start Slew Rate: ~ 6 °/sec

DSH

Mass: 47 metric tonne

Torque capacity of modern day CMG with a 100 kg is about ~ 2000 Nm

Along perpendicular Axis (y -axis and z-axis)

Moment of Inertia along principal axis: 661,917 $kg \cdot m^2$

Start Slew Rate : ~ 0.1 °/sec

Along Transverse Axis

Moment of Inertia along principal axis: 237,938 $kg \cdot m^2$

Start Slew Rate: ~ 0.24 °/sec

Communications

The communication system for our mission will be similar to the Mars Science Laboratory (MSL) mission. Owing to the proximity of Phobos and Deimos to Mars, the cruise trajectory was found to be similar to MSL, and the distance over which surface operations must be monitored from Earth will also be similar. The key difference is that our mission is a manned mission, and there needs to be a reliable radio link between Earth and the vehicles in the Martian system. The long latency time for radio communication will require that the crew for this mission be trained to be nearly autonomous. Contact with Earth will be maintained the entire time (except occultation periods), but the number of instructions from Earth will be kept to a minimum. It would also make sense to group instructions from Earth into batches before they are sent as a packet. The optimal frequency of these instruction packets would be an operational issue that will need to be established as required.

The following communication links are required:

1. **Video link** - The video link will be used only occasionally during this mission, that is, during landing and exit of astronauts from the SEV, for periodic video contact with Earth for psychological reasons, and in the event of an emergency that may mandate a video link. The primary reason for this is that video linking requires a high data rate, which increases the power requirements. Recorded video messages rather than streaming video would be a better method to establish video communication.
2. **Voice communication link with Earth** - This link will also be used only occasionally, but more frequently than the video link. Voice transfer requires lesser data rate, but the high latency time will make live voice communication problematic. The astronauts will be sent messages from Earth periodically, primarily for psychological reasons, but also during mission critical events.
3. **Telemetry link** - The telemetry link is the most important communication link between the Earth and the exploration vehicles. The telemetry link transmits flight data pertaining to vehicle and astronaut health and critical data gathered on-site. Instructions will be sent to the astronauts as parsed text messages through the telemetry link. This is a far more efficient way to pass messages between Earth and the spacecraft as it requires a much lower data rate and reduces

the requirement for synchronicity that exists for voice communication between the Earth station and the astronauts. This also reduces the possibility of cross-talk.

4. **Crew voice communication link** - The crew member in the command module needs communicate with the two astronauts in the SEV when they are separated. This link will provide voice communication, and is expected to work well, given the short distance between the two vehicles. Voice communications between all three crew will be on the same link, in order to maintain three-way communication.

Communication Infrastructure

As mentioned before, the communication system on this mission would be modeled after MSL, which has direct-to-earth (DTE) X-band radio link and also has a high-volume, parallel UHF link via the Mars Reconnaissance Orbiter (MRO) (Makovsky et al., 2009). On Earth, the Deep Space Network is used to establish communication, either directly with Curiosity or through MRO. The DTE is link is used to transmit Multiple Frequency-Shift Keying (MFSK) tones containing software updates and commands to Curiosity. Twice a day, MRO is visible from Curiosity, which transmits high volume data over the UHF band for relay back to Earth from MRO.

We propose to use a similar infrastructure for our mission. The Phobos-Deimos Surveyor (PDS) from the precursor mission will serve as the communication mediator to Earth from the spacecraft in the Martian system. PDS will have communication facilities on-board for the UHF and Ka-bands. While the UHF band will primarily be used for communication between the two spacecraft and between the spacecraft and PDS, the Ka-band link will be used by PDS to communicate with Earth through a high gain antenna to provide fast data rates. On Earth, we will use 34m and 70m facilities on the Deep Space Network for high data rate communication with PDS and the exploration vehicles. The Ka-band on the Deep Space Network operates between 31.8 GHz and 34.7 GHz and was shown to provide a downlink data rate of about 6 Mbps from MRO {NASA JPL MRO Website}. Over a time span of 20 years, we expect this data rate to become much better; as shown in Figure 42 below (DESCANSO report, Yuen and Taylor, 2002).

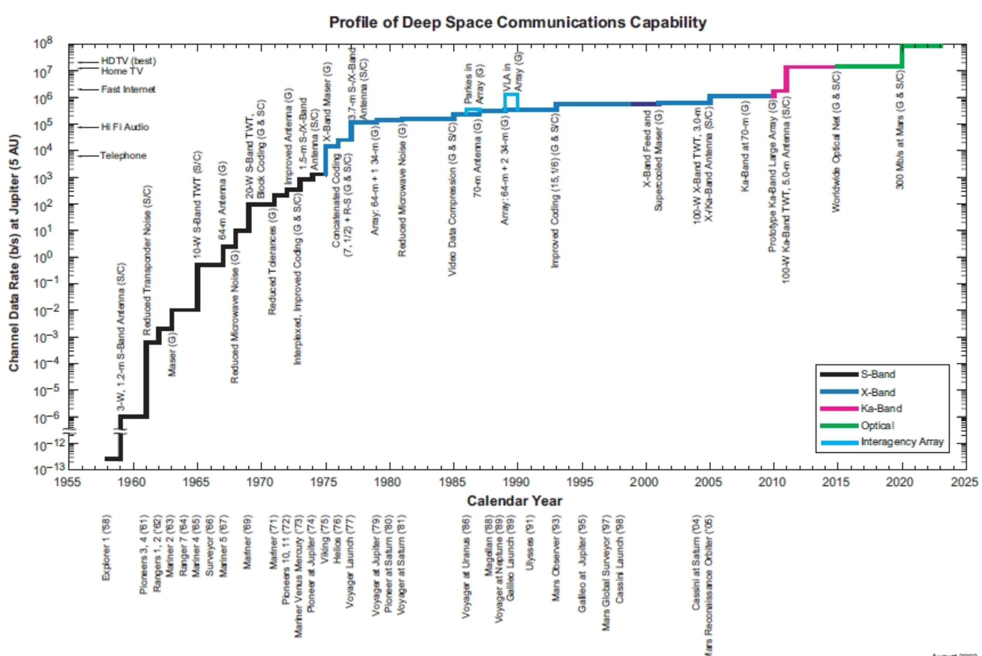


Figure 42: Increase in deep space communication capability

The Mars Reconnaissance Orbiter and Curiosity use pairs of Electra Lite transponders to communicate with Curiosity over the UHF band. The DTE data rate on Curiosity is between 500-32000 bits per second and is 100-250 megabits per second via the UHF link to MRO. MRO itself has a DTE data rate of 0.5-4 megabits per second. {NASA JPL MRO and MSL website} This data rate is sufficient for the fast transfer of low quality video. The same system can be used to establish communication between the CM, SEV and PDS.

Antennas

For the cruise stage on our mission, we propose to communicate directly with Earth using the Ka-Band through a low gain antenna (LGA) on DSH. A high gain gimballed (HGA) (parabolic) antenna serves as a backup and provides a higher data rate connection to an earth station when the boresight of the HGA can be pointed at the earth. A duplicate set of these antennas will be installed on the SEV. These antennas provide good redundancy and are also required in the event that the modules need to independently contact Mission Control when the two parts are separated.

Aside from the Ka-band antennas, two low gain UHF antennas (ULGA) are installed on the DSH and SEV in order provide for communication between the two modules and the Mars orbiters.

The output power at the ground station antenna is turned down to 200 W during launch and the early part of cruise, but is later increased up to 2kW as the spacecraft moves further away.

Infrastructure mass

The mass and size of the RF systems on MSL were found to be a good reference for the communication system for our mission. From the MSL communications report, it was found that the mass of the UHF subsystem is 3 kg for the UHF radios. The weight of the low gain UHF antennas is negligible compared to this. The weight of the Ka-band system on our spacecraft would be comparable to that on MRO. The Ka-band subsystem weight on MRO is about 25 kg (NASA JPL MRO Website). We have two of each of these systems on the spacecraft (DSH+SEV). With the inclusion of miscellaneous weight of about 10 percent, the first estimate of the total mass of the RF system on board the spacecraft is 61.6 kg.

Power consumption

MSL and MRO communication systems reports (Makovsky et al., 2009 and Taylor et al., 2006) also provide us good first estimates of power consumption by the RF system on board the spacecraft. The Electra Lite UHF subsystem on MSL consumes 69 W of power when transmitting and 21 W in standby mode. The Ka-band subsystem on MRO consumes 81W of power when transmitting. These are good estimates for power for the communication system for our mission.

With improvements in technology over the next 20 years, these systems are expected to have an even lower footprint than calculated.

Redundancy

Communication through multiple antennas over the two bands provides good redundancy on 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 spacecraft may also be rotated (or the antenna swiveled) in order to establish a radio link with a different antenna in the event of an antenna failure.

Propagation delay, radio blackouts and occultations

At its furthest point, the spacecraft is roughly 390 million kilometers away from the Earth. This implies a maximum propagation delay of 22 minutes.

There will be Earth occultations when the spacecraft are in the Martian system. STK simulations of the orbits of PDS, the command module and the SEV showed that the primary landing site on Phobos would be visible from the Command Module and PDS twice a day each. The average length of each of these encounters would be 3 hours. Aside from this, the primary site would also be visible from DSN twice a day, the average encounter lasting about 4 hours each. The parking orbit would be visible from DSN twice a day, each encounter lasting a maximum of 8 hours. This means that despite the presence of additional UHF communication facilities on Deimos, there will be parts of the day when the crew will be cut off from the Earth and maybe from each other for short periods of time. This is undesirable, but can be solved through operational means. The crew would need to be trained to be mostly autonomous to perform well during these occultations, whatever their duration is. This situation can also be made much better by providing coverage can be provided by sending more communication satellites into high inclination Mars orbits that provide a good view of Phobos and the Earth. This will also help a future manned mission to Mars. We also propose to utilize resources on MRO-type missions that may exist 15-20 years from now in order to reduce occultation times.

The transfer orbit has been designed in a way which ensures that the Earth and the spacecraft never have the Sun between them. This means that we never have the problem of solar occultation.

Communication Protocol

Communication will be carried out using Binary Phase-Shift Keying (BPSK) protocol with different antennas. Forward Error Coding (FEC) will be used for error detection. Error detection robustness is extremely important for the telemetry link, but not so much for the voice and video links. Therefore, in the event of limited resources, more computational power must be devoted to maintaining an error-free telemetry link.

Post-human mission communications

The SEV will be parked at Phobos after the astronauts leave the Martian system. The Ka-band antennas and UHF antennas on board the SEV can also serve as a communication hub with a DTE link and a node on the UHF network (during non-occulted hours). This system will provide the communication link for experiments that the astronauts leave on the surface of Deimos. The UHF network provides capability for hosted student payloads (ChipSats etc.) to perform experiments and transfer data back to Earth. A small amount of bandwidth will be allocated on the UHF network for these payloads.

Link Budget

A first estimate of the uplink and downlink budgets is shown below. These calculations are made for the Ka-band antenna for a direct to Earth link for the farthest distance from Earth on this mission. The downlink budget calculation was performed using Jan King's "Mars Micro-Spacecraft Link Budget" spreadsheet (Jan King, 2007). This assumes a 34-m DSN ground station. Therefore, the link margin may be improved by using a 70-m station. The uplink budget calculation was performed using Jan King's "Uplink Budget Calculation" spreadsheet (Jan King, 2003). The gain and aperture efficiency values for the spacecraft were taken from Makovsky et al. and for DSN were taken from the NASA DSN website.

Downlink budget:

Carrier frequency	32.3 GHz
Transmitter RF output	35 W
Antenna diameter	2m
Antenna aperture efficiency	0.6
Antenna gain	54.6 dBi
Antenna half power beam width	0.18 degrees
Spacecraft EIRP	66.44 dBW
Path length	390 million km
Path line loss	-294.47 dB
Ground station antenna gain	74.70 dBi
Data rate	120000 bps
Link margin	5.98 dB

Uplink Budget:

Ground station transmitted power	2000 W
Antenna gain	53.3 dB
Ground station EIRP	82.3 dBW
Path length	390 million km
Path line loss	-274.4 dB

Spacecraft antenna gain	56.4 dBi
Data rate	2000 bps
Link margin	8.3 dB

Electrical Power System

The major objective of the Electrical Power System (EPS) is to provide electrical energy to the spacecraft. Energy sources are commonly distinguished whether they are inside (e.g. a nuclear power source) or outside the spacecraft (e.g. the sun radiation). In a first step the energy of the energy source has to be converted into electrical energy. Afterwards, it is stored, conditioned and distributed among all spacecraft components according to their respective requirements. In the following the electrical power converters and the electrical power storages for the EPS of the DSH and the SEV are designed.

First, possible electrical power converter technologies are identified. Amongst others the following power converters exist:

- Photovoltaic (PV)
- Solardynamic (SD)
- Nuclear Power Plant (NPP)
- Radioisotope Thermoelectric Generator

PV systems convert solar radiation into direct current electricity using semiconductors that exhibit the photovoltaic effect. SD converters consist of a series of components. A solar radiation collector focuses the solar radiation onto a heat exchanger that heats up a medium that is fed to a gas turbine. The latter drives a generator which finally provides electrical energy. NPPs are commonly doing exactly the same, however the heat emerges from radioactive radiation. Inside RTGs the heat is not converted into mechanical energy but directly into electrical energy through a thermocouple.

SD converters can have higher efficiency compared to PV converters despite involving two more intermediate energy conversion steps. However they are more complex, have moving parts and lower TRL. PV converters are mature and have been used frequently aboard space stations, satellites and interplanetary probes. Solar power converter output varies linearly with the incident solar flux which decreases with the squared distance to the Sun. Thus nuclear power sources become more important the higher the distance to the Sun gets. Solar power converters are totally sufficient for spacecraft with a total power demand ranging between 10 and 100 kW (Wertz, 2011). Since in the past their capabilities did roughly double every four years it may be assumed that their efficiencies will be further approved in the near future (Dankanich, 2012). This will result in a mass specific power of about 150-300 W/kg. In contrast, nuclear power plants are supposed to only reach 13-40 W/kg. Having only 6-8 W/kg RTGs are suitable for low power space applications like space probes and rovers. Since all space elements require less than 15 kW PV converters are chosen for energy supply.

The solar panel design is based on the Ultraflex solar panels of the MPCV. These panels are superior to conventional solar panels as 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. Table 28 lists the key EPS characteristics. They are conservatively extrapolated or copied from various References.

Table 28: Characteristics of the PV (Messerschmid, 1999; Maral, 1993; Surampudi, 2011; Dankanich, 2012; Wertz, 2011)

Characteristics	**Value**	**Unit**
Collection efficiency	85	%
Solar cell conversion efficiency	26	%
Power distribution efficiency	90	%
Solar cell degradation factor	2.75	%/year
Solar constant at Mars	590	W/m^2
Mass specific power of the PV system	150	W/kg
Volume specific power (packed)	70	kW/m^3
Specific mass of solar panels	1.21	kg/m^2

The solar panel area per module has to be designed to fit the power requirements of the module at all times. They are dimensioned for the minimum solar flux available which is about 590 W/m^2 at Mars. The power demand of the DSH and the SEV are 12.6 kW and 4.6 kW respectively. Considering the efficiencies given in Table 28 and an estimated solar cell life time of three years, the solar panel areas sum up to 134 m^2 and 50 m^2 respectively, including a 15% margin. Using these values, the panel can be computed to 163 kg and 63 kg.

In a next step the electrical storage components are dimensioned. Only chemical storage systems are considered. These are Li-Ion batteries and a Regenerative Fuel Cell System (RFCS), which comprises a Polymer Electrolyte Membrane Fuel Cell (PEFC) and a Static Feed Water Electrolyzer (SFWE) as well as tanks for water, hydrogen and oxygen. Table 29 lists the key storage component characteristics which are conservatively extrapolated or copied from various References.

Table 29: Characteristics of the electrical storage components (Altmann, 2011; Wertz 2011)

Characteristics	**Value**	**Unit**
Mass specific energy of Li-Ion batteries	150	Wh/kg
Mass specific power of the PEFC	0.20	kW/kgH2
Mass specific power of the SFWE	0.33	kW/kgH2
DSH & SEV shall store electrical energy for	24	h
All storage shall be recharged within	3	h
Specific H2 storage tank mass	10	kg/kg
Specific O2 storage tank mass	0.5	kg/kg
Specific H2O storage tank mass	0.6	kg/kg

The required amount of stored energy is calculated to 302 kWh and 110 kWh for the DSH and SEV respectively. The key driver for the dimensioning of the batteries and RFCS tanks is the time period in that energy has to be supplied by storage in case the solar panels cannot supply electrical power. Requiring this time period to be 24 hours the battery masses sum up to 692 kg and 255 kg, including a 15% margin. In reality, a much longer power outage could be handled since a complete failure of all PV converters at once is unlikely and the power consumption would be significantly reduced in this case of emergency. The RFCS masses are calculated to 854 kg and 315 kg, respectively. Based on the assumptions made battery storage would be preferred as it results in less mass in both modules. However, if inside the DSH the EPS is linked with the ECLSS which already comprises a SFWE as well as the required periphery, only a fuel cell would have to be added in order to establish a RFCS. Synergistically sharing the periphery, comprising tanks for water, hydrogen and oxygen as well as the contained consumables, would allow significant mass saving. Thus for the DSH the RFCS is chosen. The SEV is equipped with battery storage. Table 30 summarizes the electrical power system budgets.

Table 30 : Electrical power system budgets (including a 15% margin)

Characteristic	**DSH**		**SEV**	
EPS power requirements	12.6	kW	4.64	kW

Solar panel areas	134	m^2	50	m^2
Solar panel masses	163	kg	60	kg
Electrical power storage masses	290	kg	107	kg
EPS mass budgets	1017	kg	315	kg

Thermal Control System

The objective of the Thermal Control System (TCS) is to ensure certain temperature range requirements at all times. The requirements are driven by the crew as well as the systems onboard the spacecraft. Thermal loads have to be transported, distributed, and eventually radiated. 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. The TCS has to be able to additionally radiate thermal loads from inside the modules. The latter is assumed to equal the electrical power demand of the modules. One has to keep in mind that the irradiating loads are calculated using the projected areas of the modules, whereas outgoing radiation can be emitted by the whole surface area of the cylindrical modules. It is assumed that they top and bottom areas do not emit radiation.

Table 31: Radiation fluxes at LEO (Messerschmid, 1999; Wertz 2011)

Radiation fluxes	**Value**	**Unit**
Solar constant at LEO	1366	W/m^2
Earth albedo	37	%
Earth reflected sunlight at LEO	505	W/m^2
Earth IR flux density at LEO	244	W/m^2
Module surface attenuation coefficient	20	%
Module surface emission coefficient	80	%
Maximum surface temperature	318	K

Based on the assumptions and the parameters listed in Table 31 the radiator areas can be calculated. They are 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 are directly mounted onto the surfaces. In order to transport the heat loads emerging inside the modules to the radiators outside liquid cooling loops are to be installed. The specific radiator system mass is estimated to be 7 kg/m² including periphery like e.g. pumps, liquids and valves. The radiator system masses can then be calculated to 564 kg and 185 kg for the DSH and SEV respectively. Assuming a specific TCS power demand of 0.07 kW/kg, the systems require 290 W and 107 W respectively. Table 32 summarizes the thermal control system budgets.

Table 32: Liquid radiator system budgets (including a 15% margin)

Characteristic	**DSH**		**SEV**	
Radiator areas	81	m²	26	m²
Liquid radiator system mass budget	564	kg	185	kg
Liquid radiator system power budget	290	W	107	W

Risk and Cost

The risk matrices for mission and program risks below show threats and possible mitigation strategies. There are multiple precursor scenarios considered to insure the best chance of usable data for the manned mission. Also many scenarios for a new deep space mission are considered. Contamination with Phobian material was considered the most severe mission risk, due to the classification of back contamination.

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

		Contingencies	Contingencies	Risk	
A1	Precursor Mission Failure to reach Phobos (launch failure, failure in transit)		New precursor mission could be launched to study Phobos before crew launch window closes		
A2	PE failure to reach and study Phobos		DE still able to be used for Phobos study		
A3	DE and PE failure to reach and study Phobos		Precursor EVA studies partially compromised, DPS still able to perform some preliminary studies of Phobos. Manned mission capability for successful EVA will be assessed.		
A4	DPS failure 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	DPS, DE, PE failure 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		DPS 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 catastrophic failure with Nuclear propulsion cargo		Requires Launch site with sufficient safety radius. Active monitoring of radioactivity at launch site. Will require further testing to place in matrix because it is a new, emerging technology.		
A8	Initial Launch catastrophic failure with Crew		Orion abort capability in place		
A9	Failure during launch in atmosphere of cargo		Requires Launch site with sufficient safety radius.		
A10	Failure during launch in atmosphere of crew		Orion abort capability in place		
A11	Staging Failure in Orbit		Launch available rescue mission given severity of situation. Nuclear propulsion stages and Cargo options will be assessed given situation and severity		
A12	Return propulsion capabilities compromised		Rescue mission missions will be assessed based on severity of failure and remaining supplies available		
A13	MPCV Reentry Compromised		Difficult launch rescue capabilities.		
A14	Minor fire on Mothership or SEV		Fire Extinguisher available on both Mothership and SEV		

A15	Life Support in HAB compromised	High TRL but Redundancies in place
A16	Life Support in SEV compromised	EVA on Phobos Capabilities Compromised
A17	Disease because of Food/Water Contamination	High TRL but will be monitored
A18	Crew Psychology Compromised	Crew selection and composition, High HAB 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 Mothership	Limited Helio proximity, Shielding in place
A21	SPE while in SEV	Shielding in place
A22	SPE during EVA	Robust SPE warning system assumed in place. EVA activities rescheduled to avoid SPE exposure.
A23	Micrometeorite Impact in Deep Space on Mothership	Unscheduled EVA options
A24	Micrometeorite Impact in Deep Space on SEV	Scheduled and Unscheduled EVA Options
A25	Minor Failure of External Mechanisms of Mothership	Unscheduled EVA options
A26	Minor Failure of External Mechanisms of SEV while separate from Mothership	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
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 with each other.
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

Likelihood	Program Risk Matrix				
	Insignificant	Minor	Moderate	Major	Severe

Almost Certain				B1, B2	
Likely				B3	
Possible				B4, B5	B6
Unlikely			B7		B8
Rare					
	Plan Changes	Minor Delay/Descope	Minor Delay/Redesign	Major Delay/Redesign	Mission Cancelled

B1	Political Delay of Precursor	Precursor EVA studies completely compromised. Mission and EVA goals will be reassessed based on future Mars missions unrelated to intimal precursor mission.
B2	Political Delay of Main Missions	If 2033 window for launch lost, backup for 2035 could be used, past those windows in mission cancellation
B3	Cost Growth	Use commercial readily available solutions where available. External constraints and reviews
B4	Mass Growth	Use commercial readily 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 EVA studies completely compromised. Mission and EVA goals will be reassessed based on future Mars missions unrelated to initial precursor mission.
B7	Precursor Partial Failure	Precursor EVA 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 EVA studies completely compromised. Mission and EVA goals will be reassessed based on future Mars missions unrelated to initial precursor mission.

Cost

First order cost estimation was done using the high-level Advanced Mission Cost Model. This model determines mission cost based on mass and type of mission or vehicle being designed. As such, the 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. This produces the following high-level cost estimate (Table 33):

Table 33: First order mission cost estimate calculation

Mission Segment	**Difficulty**	**Mission Segment Cost ($FY99 Billions)**
Precursor	Low	3.004
DSH	Very High	10.390
SEV	Very High	4.351
Orion MPCV	Low	3.701
PROP1,2	High	2.837
Launchers	Low	1.375
Total		**25.658**

Converting the total from fiscal-year 1999 dollars to current year (2013) dollars, the first-order cost estimate for this mission as a whole is approximately **$36 Billion**.

Planetary Protection

Phobos is classified as a Class II (restricted) object, reflecting NASA's Office of Planetary Protection's position that the Martian moon may harbor life from Mars, but does not warrant concern regarding possible 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. Restricting the vehicle to the surface minimizes the probability of the SEV deorbiting to the surface of Mars. The mission advisor for planetary protection warned that atmospheric entry of the SEV would be unlikely to eliminate the many forms of life left behind in the vehicle by its human occupants. The survival of organisms as large as one millimeter after atmospheric re-entry at Earth has been supported previously. While Earth organisms will be abundant in the SEV, widespread contamination of Phobos is unlikely given its environmental constraints.

However, Phobos's environment does not exclude the endurance of Martian organisms within the accreted material that coats the moon's surface. Hence, NASA's Office of Planetary Protection has assigned Phobos with "restricted" categorization, indicating that sample return should be undertaken with highly-elevated precautions. Martian organisms are considered to be a significant potential threat to human safety. Transporting these 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. It is unacceptable for a mission to place humanity at risk by not establishing procedures to minimize back contamination whenever possible.

Many mitigation technologies have been suggested to the Office of Planetary Protection, including the use of pyrophoric coatings to thermally sterilize spacecraft exteriors, peel-away exterior coating options, ethylene dioxide gas sterilization, dry heat sterilization, and plasma sterilization. However, many of these technologies have not been demonstrated in space. While some entail nearly intractable engineering and science challenges, Vapor Phase Hydrogen Peroxide Sterilization is a JPL-tested mechanism that can break the chain and mitigate back contamination. Upon returning to the DSH with the SEV, sealed metallic sample containers will be processed using Vapor Phase Hydrogen Peroxide Sterilization in a 1.0 m^3sample 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—by all feasible means—the possibility of transmission of pathogenic Martian microbes to Earth.

The complexities of sample return and the nature of close human interaction with the surface of Phobos may preclude the ability to completely break the chain of regolith exposures, including crew exposure 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 transmission to other humans. Hence, extended isolation and monitoring upon return to earth will also be necessary. Apollo isolation procedures 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 containment facility 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 (BSL 4) biological containment capabilities. All geological, chemical, and biological analysis must be conducted under BSL 4 conditions indefinitely or at least until a lack of infectious materials can be confirmed.

International and Commercial Partnerships

The mission has a very broad international aspect to it. The first being that there are many scientific payloads that will be coming from other countries, along with instruments from other countries (Table

34). This collaboration of these science payloads will spur an international community focused on acquiring the best science from this mission.

This mission will be the first of its kind, requiring 11 launches to complete the staging portion of the mission. In order to have a 2 to 3 week launch gap, multiple sites will be required to fulfill this goal. Should multiple sites not be obtained in America, international launch sites can be looked into as a solution. In order to obtain this goal, America can look to trade crew positions on the mission for site allocation. Having a multi-national crew will minimize the chance of a space race - with America trying to lead unilaterally – and enforce a scenario where the entire world comes together to complete a massive technological undertaking.

Table 34: International Science Instruments

Instrument	**Country**
Impactor Package	Japan (Lunar-A)
Asteroid Multi-band Imaging Camera	Japan (Hayabusa)
LIDAR	Japan (Hayabusa)
Retroreflector	European Space Agency (ERS)
VIS/NIR Spectrometer	Italian Space Agency (Dawn)
Visible Infrared Thermal Imaging Spectrometer	European Space Agency (Rosetta)

Public Outreach and Relations

One aspect of Public Outreach that will be critical to the overall success and lasting impact of this mission is to place a call to universities and corporations for instrumentation and experimentation that can be added to both the Precursor mission and the main mission. 1000 kg of mass has been allocated on the precursor mission for science that can include NanoSats, ChipSats, and CubeSats on the PE, DE and PDS portions of the mission. PE and DE portions can be used to include scientific payloads to help understand Phobos and Deimos from the surface, and the PDS portion can be used for scientific payload to study Phobos, Deimos and Mars. The public outreach payloads on the precursor mission will be selected from a competition which will allow enough time for payloads to be developed. Since the precursor mission will be launching in 2026, a call for experiments and payloads will be put in place as soon as possible, no later than 2024.

The same call for experiments will be placed for the main mission as well. These payloads will be placed on both the mothership and the SEV with 260kg being allotted for these experiments and payloads. The mothership experiments will be used as a way to give the Astronauts something to do on their way to the Martian system. This will also give the Astronauts the opportunity to interact with the designers of the experiments and report back findings and results and if the experiment needs to be redone. This will have the added benefit of boosting Astronaut moral. The SEV experiments will include both science instruments that can be placed on Phobos by the Astronauts and payloads that can be placed on Phobos to observe Mars. Because weight is more of a factor on the SEV, we have only allocated volume and mass for payloads being left on Phobos equivalent to the geological samples we will be bringing back. This way we maximize the space available.

Astronauts on the mission will be able to reach out to the public through Facebook, Twitter and other internet forums/communities. The internet community is vast, if it can be tapped into, interest will surge.

NASA TV can also be utilized to make all aspects of the mission into a celebrity. Mission design crew, mission control, and most importantly Astronauts can be utilized to inform the public about details of the mission. Importance of the mission along with risks can directly be expressed to the public.

STEM emphasis, especially in K-12 programs, is something that NASA already has a great focus on. This can be brought into this mission by making programs titled "Science from Space", and distributing them to public school to help engage and excite students into learning.

IV) What will be learned and what will the benefit(s) be if the project is successful? (Question 4)

- Better understanding of planetary systems, in particular the inner, rocky part of our solar system
- Better understanding of the formations of small planetesimal bodies.
- Investigating and understanding the internal structure of the bodies.
- Preparation for future missions to the Martian system, including the surface of Mars.
- PO payloads that will be on PE, DE, PDS, and placed on Phobos directly.
- Human long term duration conditions will be studied
- Advancement for further human deep space exploration missions.
- Contribution to the compendium of medical knowledge for use on Earth
- Physiological and psychological degradation characterized in the crew
- Countermeasure optimization for crew health maintenance
- Characterization of the health-relevant radiation profile at the Mars system and in transit
- Discovery of resources on the moons can be the base of further missions that can utilize the resources.
- Stimulation of the economy of the sponsor nation
- Inspiration of the next generation of scientists

V) How will the results change the future? (Question 5)

The Asaph 1 mission would be the farthest man has ever travelled into the solar system, giving significant historical bearing to the program. This is one of the key drivers to motivate international partners into collaboration, as the prestige of being able to claim participation in this achievement would be highly sought. This common goal also forms a strong foundation for further international collaboration, in aerospace and otherwise, as well as forming the fundamental financial and logistical infrastructure required to venture farther into the solar system.

Given the number and nature of modules remaining in the Martian system once the crew has departed, this mission has been developed to maintain presence and a measure of operational capability for years to come. This capacity may be measured in the form of further observation from the PDS, whether it be of Phobos, Deimos or Mars itself; or further exploration using the SEV module in a similar fashion to current assets such as the MSL. This is seen to extend mission capabilities and hence provide more justification for the required mission budget. The ability to monitor another planetary system such as Mars after a crewed mission has departed may also provide insight into manned mission impacts on different environments and assist in future mission planning.

The operating conditions of this mission are similar to that of missions to near Earth objects, hence the results from this program may assist in the development of parallel programs under NASA's current agenda.

Another benefit to conducting the Asaph 1 program is the potential to study to effect of deep space exploration on human beings and provide the foundation to develop future protection and habitation mandates. This will assist in a variety of missions, such as NEO mentioned above, and allow an environment for other detailed biological experimentation such as food production in low gravity environments.

Finally, this mission provides the opportunity to begin searching for outpost locations for In-Situ Resource Utilization (ISRU). As is clearly evident in the mission architecture, the majority of launches are assigned to transporting the necessary fuel from Earth to its destination. Should bodies such as Phobos or Deimos prove to contain resources such as water which may be feasibly extracted, the opportunity to establish fuel outposts would provide the basis for further exploration into the universe, given current propulsion mechanisms. This is potentially one of the most substantial contributions as it is essential to determine an alternate mission architecture to reach more distant bodies in the solar system.

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