Team Cupid

Moons of Mars Mission

2022 AIAA Space Design Competition

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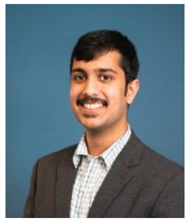

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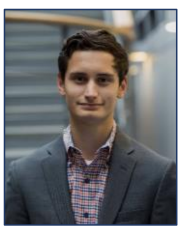

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List of Acronyms

AIAA	American Institute of Aeronautics & Astronautics
APXS	Alpha Particle X-Ray Spectrometer
ATCS	Active Thermal Control System
CER	Cost Estimating Relationship
DST	Deep Space Transport
EEV	Exploration Excursion Vehicle
EVA	Extravehicular Activity
FY	Fiscal Year
ISS	International Space Station
JAXA	Japanese Aerospace Exploration Agency
MLI	Multi-Layer Insulation
NASA	National Aeronautics and Space Administration
NPR	NASA Procedural Requirements
PCEC	Project Cost Estimation Capability
RF	Radio Frequency
RAD	Radiation Assessment Detector
RFP	Request for Proposal
US	United States of America
UWMS	Universal Waste Management System
WBS	Work Breakdown Schedule

Nomenclature

h	=	Height of the dome
P_{in}	=	Internal Pressure
R_{out}	=	Outer Radius
SF	=	Safety factor
t	=	Thickness
α	=	Cone angle
σ_c	=	Compressive stress
σ_t	=	Tensile Stress
σ_y	=	Yield stress

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1 Executive Summary

Access to space has never been easier than it is right now. The growth of the private space industry and its optimization of the for-profit model has allowed costs associated with launching and maintaining spacecraft to fall drastically. To that end, NASA has been able to concentrate further on the science of its missions rather than labor over how they'll even get up into space. These factors have led to the active goal of trying to establish a human presence on the Martian surface. NASA's Artemis mission is fully underway attempting to land humans back on the moon by the end of the decade [1]. The next logical step after Artemis is sustaining life in deep space. Just as Apollo did back in the 1960s, NASA plans to send astronauts into orbit around Mars. This paper presents a proposed mission architecture to send 2 humans to Martian orbit, explore both moons, and return safely to earth.

The overall goals of this mission are to sustain a human presence in Martian orbit and to gain significant scientific understanding of the two Martian moons: Phobos and Deimos. The deepest we have ever sent astronauts into the solar system was lunar orbit at around 400,000 km. Our mission is set to send humans 62 million kilometers which brings along with it a whole host of challenges but large opportunities to learn about life in deep space and how to endure the harsh environment. The secondary goal seeks to advance our understanding of Phobos and Deimos and improve our capabilities to explore future destinations across the solar system.

The design requirements and constraints for this mission came from the American Institute of Aeronautics and Astronautics (AIAA). These requirements request the design of an Exploration Excursion Vehicle (EEV), which will travel uncrewed to Martian orbit to rendezvous with a Deep Space Transport (DST) in the summer of 2040. Once in orbit, the EEV will autonomously dock with the DST and 2 crew members will transfer through a pressurized tunnel to begin the mission. The crew members will then explore the Martian moons, Phobos and Deimos, for 30 days upon which they'll return to the DST and travel back to Earth. With them will come at least 100 kg of sample material, 50 kg minimum from each moon. The total end-to-end mission cost, including launch, will be $818.1 million USD, which does not exceed the maximum of $1 billion USD (FY21).

The EEV is cylindrical in shape with chamfered ends to fit the constraints of the launch vehicle, which the Falcon Heavy has been selected. Before October 13th, the EEV will depart from Earth. It will arrive in low earth orbit and wait for the booster stage. The booster stage will be launched from earth into LEO such that it intersects the orbit of the EEV. Once they are docked together, on October 13th, 2039, the booster stage will activate towards Mars. The entire maneuver to get to Mars takes about nine months. On July 20th 2040, the EEV will arrive in 5-sol

Commented [TFB1]: LSM

orbit at Mars, then will make a 26 degree inclination change to get on the same orbital plane as Phobos. After that, the EEV will wait until July 1st to begin the mission. On July 1st the EEV will transfer to Deimos which is the lowest orbiting moon about Mars. This orbit will take about 4 hours. Once at Deimos, the EEV will perform scientific experiments and core sampling at five different sites.

To perform precise navigation on and around the moons the EEV is equipped with 12 reaction control system (RCS) thrusters. The RCS thrusters are equipped with a methane oxygen propulsion system. To aid in determining position and orientation in reference to the moons, mounted in various places around the EEV are star sensors which use the location of reference points to determine orientation. Also equipped are LIDAR sensors so that the distance from each moon or any other object is known and can be compensated for. The EEV is controllable via a series of two joysticks which will control rotational and translational motion.

As part of the scientific experiments, the EEV will use instruments to measure chemical composition of the lunar soil, measure the radiation levels on and around the moons, and measure the seismic activity of the moons. To deploy the scientific instruments, an instrument deployment mechanism has been designed which will be mounted outside of the habitable volume but inside the outer shell. The deployment mechanism will use a rack and pinion device to extrude itself in and out of the EEV to perform its scientific objectives. This will allow it to get full operational performance.

A core sampler will also be mounted to the edge which is exposed to the lunar soils. While drilling, the RCS thrusters will be used to counteract the force onto the EEV from the drilling core. The sampling core will collect seven samples from various locations on each of the moons and will be strong enough to cut through high-strength rocks which may be under the regolith surface. This is so that later analysis of the soil will be more effective in determining the age of the layers.

Commented [TFB2]: Core sampling

The EEV will use ultra-high-frequency signals for essential communications, relayed to the Deep Space Network on Earth via the Mars Relay Network. There will also be a video stream from the EEV sent to Earth using an optical communications system. This system will use the DST to relay the video data to Earth.

The EEV stays on Phobos for 13 days while performing these scientific experiments. The two crew members, while aboard the EEV during the 30-day mission, will be assisting in performing scientific experiments, observing the moons of Mars to look for in places of interest to visit, be communicating with the crew members aboard the

DST, and troubleshooting any minor difficulties that may arise. Additionally, the crew will be documenting their experience on the Martian moons to aid in planning future missions and settlements.

Commented [TFB3]: Crew systems

After performing experimentation and sampling on Phobos, the EEV will begin a transfer maneuver on July 13th from Phobos to Deimos. This transfer orbit will take about 8 hours. The EEV well perform the same scientific experiments and soil sampling. On July 26th, the EEV will depart from Deimos and rendezvous with the DST on July 28th.

2 Mission Overview

This section discusses the overall system requirements for the mission as well as the covering some overview on Phobos and Deimos to support the reasoning for the scientific objectives.

2.1 System Requirements

The following are the system requirements derived from the AIAA Request for Proposal. Further subsystem requirements which have been derived from Table 2.1.1 can be found in the appendix.

Table 2.1.1 Overall System Requirements

REQ #	Requirement
S – 1	The EEV shall maintain a habitable environment for the full duration of the mission in accordance with NASA NPR 8705.2C.
S – 2	The EEV shall protect the crew members from the harmful environment of space.
S – 3	The EEV shall be capable of maneuvering to Phobos and Deimos.
S – 4	The EEV shall be capable of staying in constant contact with an Earth-based mission control center.
S – 5	The EEV shall provide all power necessary to complete mission objectives.
S – 6	The EEV shall have adequate contingencies to minimize mission failure in the case of emergency or catastrophic failure.

2.2 Phobos and Deimos Overview

For the Moons of Mars mission, our target bodies are Mars' two moons: Phobos and Deimos. Named after Greek mythological characters, Phobos and Deimos are interesting in that they are more similar in size and shape to asteroids, rather than moons. Table 2.2 provides a summary of their physical characteristics, using data from NASA's Mars Exploration Program website [24]. Phobos and Deimos have been objects of interests by scientists for many years, but there has never been any formal NASA mission to study them. There has been a push to study these moons since there is a possibility they could be used as bases of operations for deep exploration of Mars, since humans have not been sent to Mars yet and it may be easier to send them to the moons first. Moreover, scientists are still unsure about their origin and evolution. There are theories that they may have been formed at the same time as Mars, within the planet itself, or are just captured asteroids. According to Tomohiro Usui, a robotic planetary exploration expert at JAXA, Mars has a weak gravitational pull due to being a tenth of Earth's mass. However, if Phobos and Deimos were formed from a debris disk launched up from Mars after a colossal impact, Deimos should be orbiting closer to Mars than it is today [31]. By studying these moons, it would allow us to better understand

Mars' past and perhaps give insight on how the early solar system was formed. In addition, there is a possibility that water and/or other organic material that could be found. This would suggest that Phobos and Deimos were once capable of sustaining life or even that they still are. Finding evidence of this would also suggest that the same could hold true for Mars as well. There are theories that Earth and other terrestrial planets formed dry, and that water was delivered by icy meteorites. Dr. Usui also suggests that if the moons are indeed captured asteroids, they would be evidence of this, and their composition could show what materials were delivered to early Earth [31]. In addition, meteors crashing into Mars could have spread Martian dust or other minerals common on the surface of Mars to Phobos and Deimos. If a mission to the moons was able to collect and analyze samples, scientists could learn how Mars transformed from a habitable world into an inhabitable one.

Table 2.2.1 Phobos and Deimos Characteristics

Characteristic	Phobos	Deimos
Mean distance from Mars (km)	9,377	23,436
Orbital period (Mars days)	0.31891	1.26244
Major axis (km)	26	16
Minor axis (km)	18	10
Mass (x 10^{15} kg)	10.8	1.8
Mean density (kg)	1,900	1,750

2.3 Scientific Objectives

The aim of our mission is to either support or debunk the theories mentioned above. We also want to take advantage of having crew onboard. To do this, we want to be able to answer the following questions:

1. What is the internal composition of Phobos and Deimos?
2. Are there phyllosilicates or other Martian minerals present on Phobos and Deimos?
3. Is there water and/or other organic compounds on Phobos and Deimos?
4. What is the terrain and weather like on Phobos and Deimos?
5. What are the radiation levels on Phobos and Deimos?
6. What is the effect of deep space on human organs and reproductive cells?

Once the EEV lands on each moon, it will travel to different geological features in order to take pictures, take instrumental data, and collect samples. Since we want a wide variety of data, we chose features of assorted size and location. These include Stickney (9 km crater on Phobos), Laputa Regia (large area on Phobos marked by reflectivity or color distinctions from adjacent areas, Lagado Planitia (low plain on Phobos), Swift (1 km crater on Deimos), Voltaire (1.9 km crater on Deimos), and the north and south poles of each moon. These features are mapped in Figure 2.1 and Figure 2.2. At each moon, the EEV will visit one location per day and 4 hours will be allotted so that the EEV has substantial time to take the necessary actions required to complete the scientific objectives.

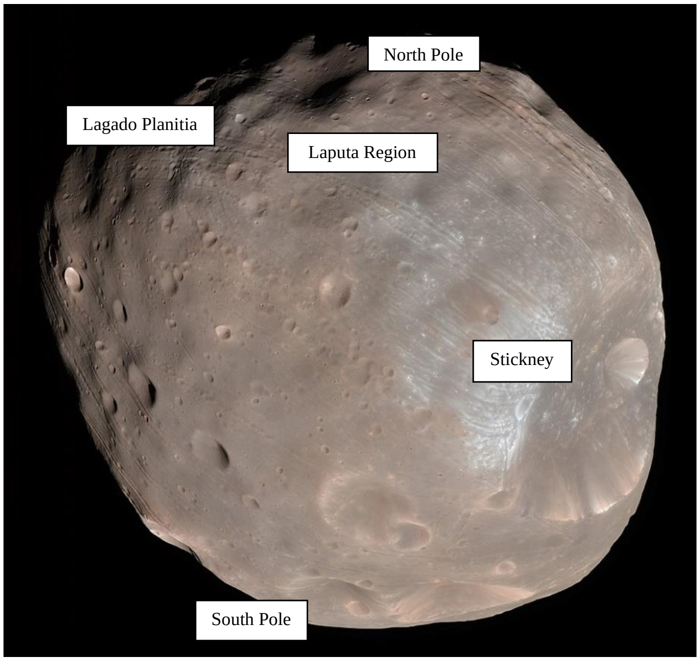


Figure 2.3.1 Phobos Target Mission Features [24]

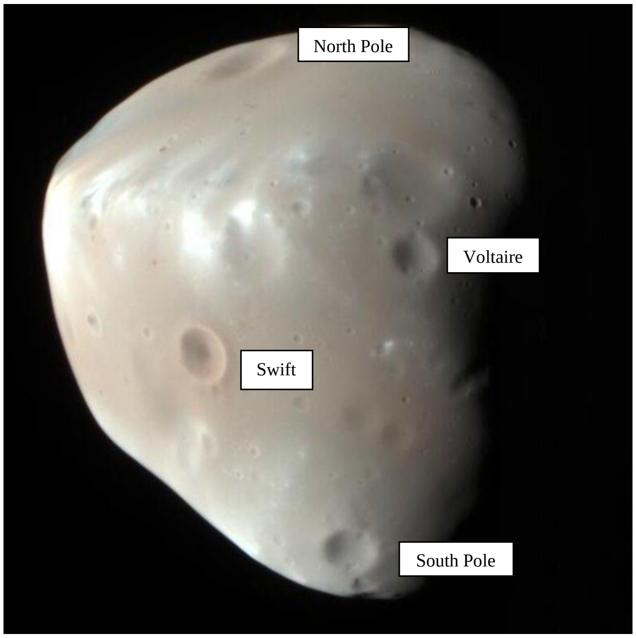


Figure 2.3.2 Deimos Target Mission Features [24]

3 Mission Design

3.1 Mission Planning

Mission Planning can be broken down into two subcategories – The EEV's transfer from Earth to Mars, and the EEV's transfers within the sphere of influence of Mars to the moons Phobos and Deimos, and the DST.

3.1.1 Earth to Mars Transfer

Transfers from Earth to Mars only occur in 26-month intervals. As such, timing is paramount in achieving minimal velocity change, which will achieve minimum fuel necessary. To aid in the selection of the launch date, various porkchop plots were generated, which graph the launch date from the Earth vs the arrival date at Mars, with various contours that represent important transfer parameters such as time of flight, total delta V, and right ascension and inclination at departure and arrival.

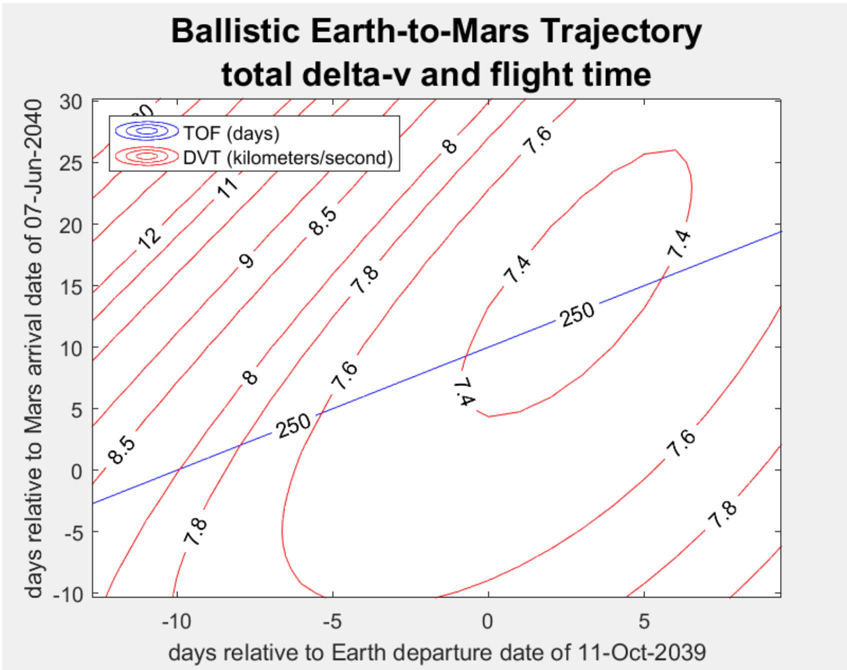


Figure 3.1.1 Earth-to-Mars Porkchop Plot

A 1-way delta V of 7.4 km/s has been identified to occur around Oct 13, 2039, which will make the arrival date on June 15th, 2040, before the 'mission start' of July 1st, 2040. Once the EEV is in a 5-sol orbit at Mars, it will make an inclination change to put it approximately in plane with both Phobos and Deimos.

3.1.2 Mars Sphere of Influence Transfers

First, it is helpful to look at the characteristics of each moon to first understand what needs to be done.

Table 3.1.1 Orbit Parameters

Orbit	Semi-major axis (km)	Inclination (°)	Eccentricity	RAAN (°)	Orbital Speed (km/s)
DST	59,800	26	0	276	0.846
Phobos	9376	26.04	0.0151	276	2.14
Deimos	23,500	27.58	0.00033	276	1.35

Phobos is the inner-most moon, with a semi-major axis of 9376, followed by Deimos. The DST is about 2 times further away from Mars than Deimos.

To accurately find optimal transfer times, the moons' ephemeris data was extracted from Johns Hopkins Propulsion Lab (JPL) Horizons software. This software gives a user the ability to find the precise state (position and velocity) of any satellite/observed asteroid in the solar system. The ephemeris data for Phobos and Deimos was partitioned into 10 minute increments during the entire time of the mission, July 2040. Next, the orbits were plotted using MATLAB. The most optimal method of transfer is the Hohmann Transfer, where the change in velocity occurs at periapsis and apoapsis of the respective orbits to transfer to/from. Because the moons are on approximately the same inclination, Hohmann maneuvers could very nearly be achieved. Since the two moons are at different radii away from Mars, and therefore have different periods, precise times, to the nearest 10 minute interval, could be attained to remove the need to do rendezvous transfers. There were several times within the 30 day period, and the ones chosen to optimize the time on both moons while having approximately the same time for scientific experiments on each of the moon, about 13 days, while keeping the mission within the 30-day maximum. To visualize the orbits of the DST, Phobos, and Deimos, each orbit was propagated and then animated in MATLAB, in the ecliptic frame. Here is an example of one timestamp, with the DST in purple, furthest out, followed by Deimos in orange, Phobos in yellow, and Mars at the center of the orbit in blue.

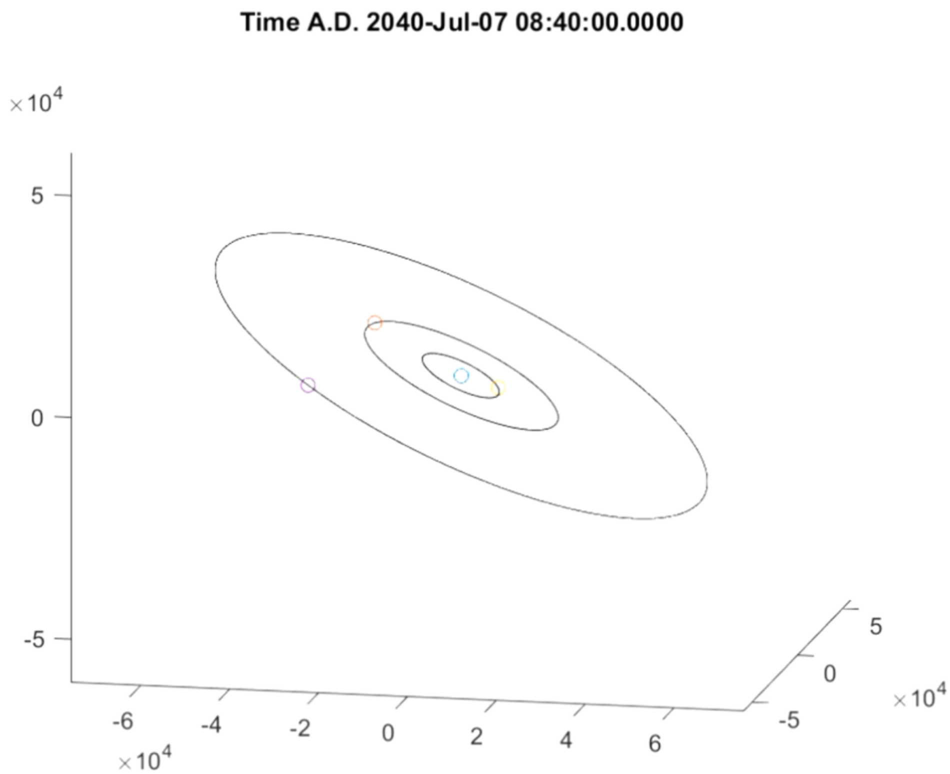


Figure 3.1.2 DST, Deimos, Phobos, and Mars

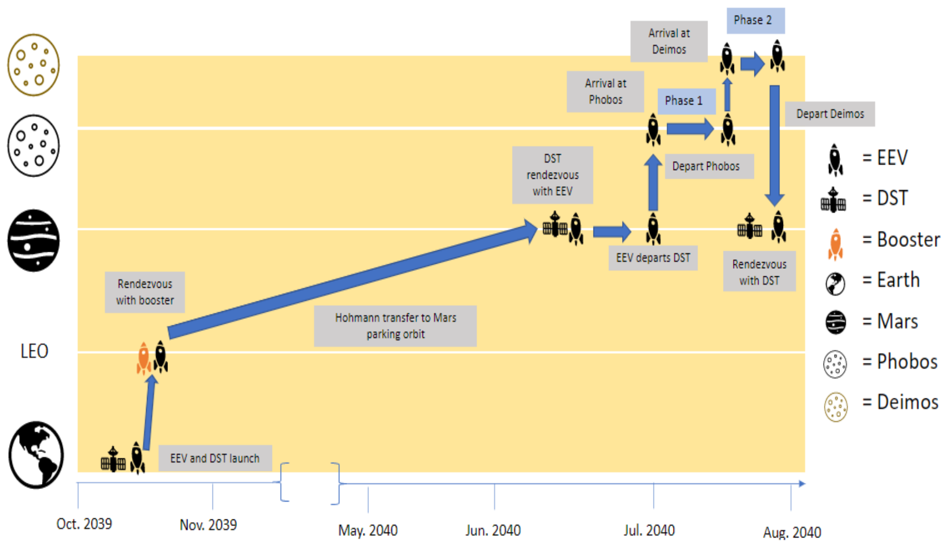
Table 3.1.2 Mission Timeline

Date	Event
October 11-16, 2039	EEV and DST launch date range
June 7-20, 2040	EEV arrives in Mars 5-sol parking orbit, afterwards crew arrives in DST to transfer to EEV
July 1, 2040 01:00	EEV leaves DST
July 2, 2040 04:10	EEV arrives to Phobos
July 13, 2040 05:10	EEV leaves Phobos
July 13, 2040 14:00	EEV arrives to Deimos
July 26, 2040 17:50	EEV leaves Deimos
July 28, 2040 05:40	EEV arrives back to DST, crew transfers into DST back to Earth

Table 3.1.2 provides the timeline of events for the mission. Per the AIAA RFP, we were constrained by a 30 day total mission period starting from when the EEV arrives in Mars' orbit. The EEV and DST will launch at the same time from Earth on the Falcon Heavy, our chosen launch vehicle. However, we were only tasked with designing the EEV according to the RFP. After launch, the EEV will rendezvous with a booster that allows it to arrive at Mars first. The EEV will then rendezvous and dock with the DST after it arrives in a 5-sol parking orbit. On July 1, 2040, then primary mission begins as the EEV departs the DST and heads towards the first moon, Phobos. While on Phobos, the EEV performs the necessary scientific objectives. Next, the EEV departs Phobos and performs a similar procedure on Deimos. The EEV will roughly spend 13 days on each moon to complete all scientific objectives. After leaving Deimos, the EEV will rendezvous and dock with the DST for the last time to transfer the crew back and return to Earth, ending the mission.

3.1.3 Concept of Operations

Figure 3.1.3 Concept of Operations Diagram



During the transfer to Mars' orbit, the EEV performs minor corrections to its trajectory to adjust for minor differences from the nominal transfer trajectory, using the RCS. Phases 1 and 2 represent the periods of time during which the EEV will collect the necessary data on Phobos and Deimos to complete the scientific objectives.

3.1.4 Launch Vehicle Trade Study

There were many factors to take into consideration when selecting a launch vehicle such as price, mass to LEO capability, fairing volume, and fairing diameter. The specifications in Table below are taken from the launch vehicle user guides. However, the costs for Starship and SLS were estimated using NASA PCEC. Falcon Heavy was the best choice for the Moons of Mars mission as it is one of the cheapest heavy lift launch vehicles and large enough to house both the EEV and DST while still providing enough lift to carry them to Mars' orbit.

Figure 3.1.4 Launch Vehicle Specifications

Launch Vehicle	**Estimated Price ($M)**	**Inner Fairing Diameter (m)**	**Mass to LEO (t)**	**Fairing Volume (m^3)**	**Estimated Price/Volume ($M/$m^3$)**
Starship	150	8	100	893	0.22
New Glenn	N/A	6.35	45	487	N/A
Falcon Heavy	150	145	63.8	145	1.03
SLS Block 1B	900	621	97.2	621	1.45
SLS Block 2	2000	1320	104.7	1320	1.52

3.2 Vehicle Structure

3.2.1 General Considerations

The EEV structure underwent two major redesigns over the course of the project. While the first design was primarily cylindrical and the second design was primarily conical, the final design hybridizes a cylindrical and conical design. Because the mission takes place in microgravity with no directional reference, the flat faces of a cylinder are sufficient for relative orientation. The negligible atmosphere on the moons reaffirmed the sufficiency of a cylinder because aerodynamic considerations are low priority. The motivation behind the conical component is to fully utilize of the launch vehicle's fairing space.

The structure has three main components: an exterior shell, interior habitable volume, and a storage trunk. Each of these components were designed to fairing, environmental, and human limitations. The Falcon Heavy launch vehicle was chosen to transport the EEV into its Mars trajectory. The fairing and its dynamic envelope are pictured below:

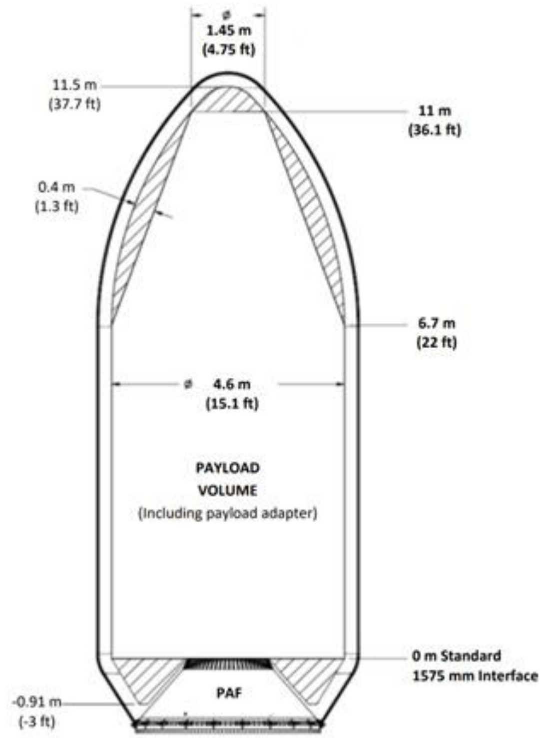


Figure 3.2.1 Fairing Dimensions and Dynamic Envelope of the Falcon Heavy Launch Vehicle [5]

The general shape of the EEV was designed to the dynamic envelope to take advantage of the available space. 0.25 meters of radial clearance is left around the envelope within the cylindrical section, allowing a maximum vehicle diameter of 4.1 m. this was to provide room for solar panels and other exterior components which would occupy the remaining space.

The material chosen for the structure is Aluminum 7150-T7751. It is a zinc alloy with traces of copper and magnesium, and the treatment includes heat treatment and overaging [6]. The primary considerations for choosing the structural material were the strength-to-weight ratio, weldability, and machinability of various alloys. More than 50 alloy-treatment combinations were considered, and they were narrowed down to the top 11 by strength-to-weight ratio [7, 8]. A table of the top choices is shown below:

Table 3.2.1 The Top 11 Choices of Aluminum, Ranked by Strength-To-Weight Ratio

Alloy	Treatment	STW (Nm/kg)
7150	T7751	0.181
7475	T651	0.165
7010	T7651	0.161
7050	T7651	0.161
2024	T861	0.159
7475	T61	0.158
7050	T7451	0.156
7010	T7451	0.151
2090	T83	0.151
7475	T761	0.148
7475	T7651	0.148

The top 11 were found to have comparable machinability and weldability, so the final choice was determined by the strength-to-weight ratio. Choosing a high ratio has allowed the vehicle to remain light while offering ample impact protection and reducing concerns about stress fractures.

With a thickness of 1 cm, the exterior shell offers protection from orbital debris up to about ¾ cm in diameter. This was determined from the following equation:

$$m_c = \left(\frac{2.54t}{K_t p^{\frac{1}{6}} V^{\frac{7}{8}}} \right)^{\frac{1}{0.352}}$$

Equation 3.2.1 Solution for the Critical Meteoroid Mass of Penetration [9]

M_c represents the minimum meteoroid mass that can penetrate a wall of thickness t. The diameter was calculated from the mass assuming spherical debris. The material constant K_t is 0.54 for aluminum alloys. The mass density of the meteoroid p is approximated as 2.5 g/cm^3, and the impact velocity V is calculated from the orbital velocity of Phobos – 2.24 km/s.

3.2.2 Exterior Shell

The exterior shell, which closely matches the shape of the fairing, is pictured below:

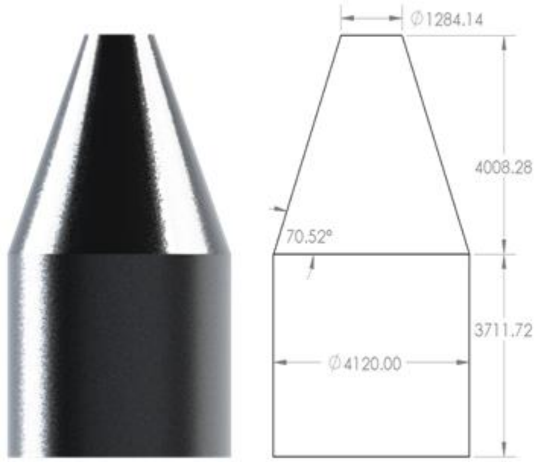


Figure 3.2.2 Render and Dimensions of Exterior Shell (mm)

The main purpose of the shell is to protect the EEV's science instruments in a non-pressurized but radiation- and impact-shielded region. The thickness of the shell is 1 cm, which can shield against orbital debris around Mars up to 0.76 cm in diameter [9]. Outside of the habitable volume, the exterior shell offers 24 m^3 of storage.

3.2.3 Habitable Volume

The interior habitable volume is pictured below:

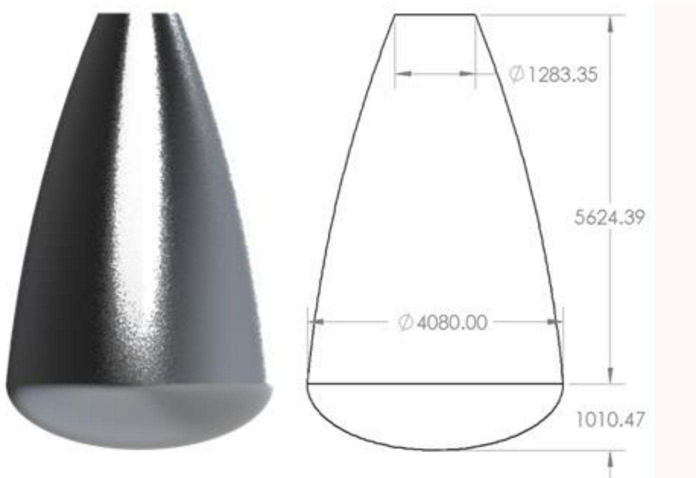


Figure 3.2.3 Render and Dimensions of Habitable Volume (mm)

The habitable volume hosts the operation and living quarters for the two crew members. It is designed to have no flat edges, except for where it meets with the exit hatch. There are 50 m^3 available for the crew members and facilities. For an internal pressure of 6 psi, the minimum possible thickness for a safety factor of two is 0.35 mm. However, the thickness remains at 3.5 mm, which is an order of magnitude above the minimum thickness. The justification for this is two-fold. First, Aluminum 7150 is typically machined at a thickness of 0.7 mm and above, so designing for a 0.35 mm thickness would require special techniques. Second, increasing the thickness decreases the maximum strain by a factor of 10. With a thickness of 3.5 mm, the maximum strain is $3.6 \cdot 10^{-5}$.

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In analyzing the stresses on the domed portion of the interior shell the following equation for axial stress was used:

$$t = \frac{P_{in} \cdot SF \cdot R_{out}^2}{\sigma_y \cdot 2 \cdot h}$$

Equation 3.2.1 Thickness of the dome

The reason for choosing a domed shape for the shell was to maximize the usable volume towards the bottom of the shell without going for a completely flat plane which would mean maximizing the stress due to pressure. The dome has a height of 1.01 m and a radius of 2.04 m. Using Eq. 1 the minimum allowable thickness considering an internal pressure of 6 psi, a material yield strength of 510 MPa, and a safety factor of 2 was 0.35 mm.

$$t = \frac{P_{in} \cdot SF \cdot R_{out}}{\sigma_y \cdot cos(\alpha)}$$

Equation 3.2.2 Thickness of the cone

For the purpose of analyzing the structural rigidity of the shell's curved cone, a cone with a base radius of 2.94 m and a height of 8.76 m was used giving a cone angle of 18.55°. This simplification of the conical shape was made in order to obtain a reasonable estimate for the required shell thickness of the cone considering an internal pressure of 6 psi, a material yield strength of 510 MPa, and a safety factor of 2. Obtained from Eq. 2 was a thickness of 0.356 mm which is close to the 0.35mm obtained from the dome and nearly a factor of 10 lower than the actual thickness used.

3.2.4 Trunk

The trunk is pictured below:

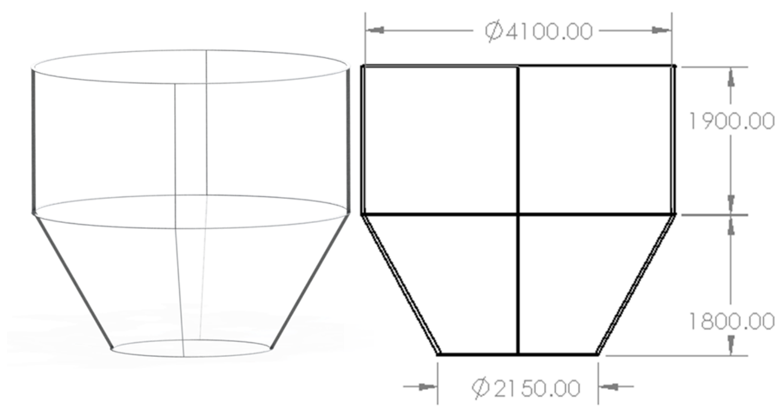

Figure 3.2.4 Render and Dimensions of Trunk (mm)

This is primarily a containment space for the propellant and oxidizer tanks, below which the main EEV thrusters are attached. Each circular ring is 5 cm in annular thickness and 1 cm in depth; each connecting strut matches these dimensions with a 5 x 1 cm cross section. Previously, the trunk had a solid shell to offer extra micrometeoroid protection. However, a heavy-duty trunk is not required because the tanks have wall thicknesses closer to 10 cm, which is sufficient for orbital debris protection. Without any additional insulation, the tanks can withstand impacts from orbital debris more than 6 cm in diameter.

3.2.5 Windows

For a crewed mission, NASA requires that all visual objectives can be completed via windows [10]. The visual objectives on this mission are navigation across each moon and scientific tasks. The minimum required field of view for ground navigation in poor visibility conditions is 32 degrees [11]. The field of view also begins to drop off rapidly at around 25 cm of lateral offset, so 25 cm was the most extreme case that was considered [12]. Lateral offset refers to the viewer's horizontal distance from the center of the window. For the common setback distance of 15 cm, the minimum window diameter that suffices a 32-degree field of view in the worst-case scenario is 29 cm. Two windows of this size on opposite sides of the craft allow for navigation, and a third window halfway between them will be used for scientific analysis. The relative window sizing can be seen below.

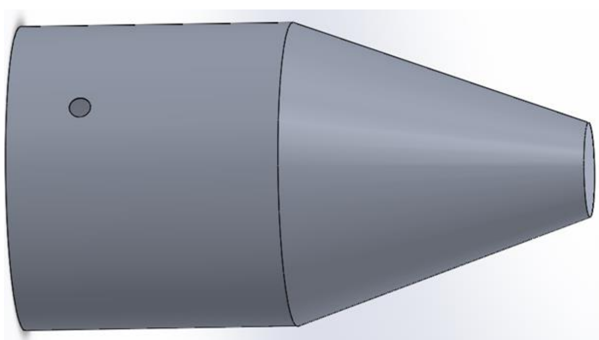

Figure 3.2.5 Relative Sizing of Window to Exterior Shell

3.3 Sample Collection

3.3.1 Overview of the Moons

Phobos and Deimos are known to have layers of regolith in many areas with most of the layers predicted to be non-native [2], so in choosing the sample collection method much thought was placed on the scientific objectives and the role the sample collection mechanism would play on those said objectives. Objectives aimed to be achieved by the collection of the sample were carbon dating to determine the age on the regolith collected as well as material analysis of the collected sample. Material analysis of the regolith and/or rock samples can give insight as to what they are made of, how they were formed, and how they got to their respective orbits. It can also help answer whether the moons are trapped asteroids, broken-off pieces of a larger moon that was once a part of the Martian orbit, or parts of Mars that were launched out of its atmosphere due to impact.

3.3.2 Sample Collection Methods

Assuming that Phobos and Deimos' regolith are like that of our moon in compressive and tensile strength then, they would have tensile strengths σ_t of 1kPa as well as compressive strengths σ_c of 11 kPa [3]. Knowing this, three sample collection mechanisms were considered. The three considered were a claw, an auger, and a drill core.

Figure 3.3.1 Claw

The claw design would have done reasonably in collecting loose and fine soil while being great for any hard rock found during the collection process. It would have offered a great range of motion allowing for retrieval in a variety of condition that may not have been possible for the other methods whilst also given the crew more freedom in deciding what to take with them. However, it was apparent that such a system would be heavy, would require myriad of mechanical components mainly multiple linear actuators, would draw a lot of power or excess material through its linear actuators whether electric, hydraulic, or pneumatic, and would require a multitude of reactionary forces to counter the digging forces which could be at various angles given the design, as such, this method was not chosen.

Figure 3.3.2 Auger

The auger was the next thought for collecting samples. It could drive itself into the lunar soil and would allow it to move upwards and be collected in a storage tank limiting the need for retracting the auger as well as the need for linear actuators. Though, there is a concern for the auger being stuck in the soil, clogging of soil in the auger, the effect high cohesive stress has on the auger and its helix, as well as the difficulty to sanitize the auger and handle the sample after each sample collected a problem that can also be attributed to the claw. Thus, this also was not chosen as the mission sample collection mechanism.

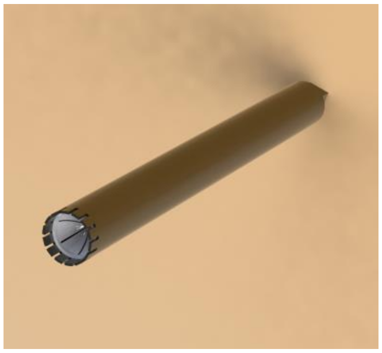

Figure 3.3.3 Drill Core

Ultimately, the mechanism that was chosen for the mission was the drill core. It works great at collecting fine soil and regolith, there is no need for sanitation as the drill core shell can be replaced after each sample collected as such the shells will work as both the sample collection method as well as the sample containment. It does not work well in collecting rock or very hard soil but as per expectation most of the sample being collected will be loose regolith or fine soil which brings up the main reason for choosing this method that reason being its ability to keep the sedimentary layers intact. This will help with better determining the origin of both moons and having those layers could help date them.

3.3.3 Drill Core

The drill core shells will be 1.01 m long with an inner radius of 51 mm and an outer radius of 46 mm. They will be made up of 304 stainless steel. Though the stainless steel does not have as high of a tensile strength as the EEV's aluminum, the stainless steel can better the lower temperatures of the moon without losing much of its tensile and shear strength by becoming brittle and it has a much higher shear modulus, 77.0 GPa versus 26.9 GPa and thus it's better suited for drilling. The drill shell will have teeth at the end to better obtain the sample as well as a core catcher to keep the regolith in. The plan is to take 14 core shells for the mission with 7 being used on Phobos and the other 7 being used on Deimos. It will collect 12.6 kg of sample with each shell on Phobos and 9.9 kg of sample on Deimos for a total of 157.4 kg samples acquired during the mission. With each core being 12.4 kg that is a net total of 157.4 kg for the samples and the shells that the EEV would transport to the DST.

As the core will be experiencing a σ_t of 1 kPa as well as a σ_c of 11 kPa, it will need to output a force of 13.1 N normal to the soil surface with a downward velocity of 5.6 mm/s and a rotational velocity of 0.932 rpm to tackle the cohesive stress and tensile strength of the soil. It will use an electric linear actuator to bring it down into the soil and a rotational motor for the motion. The electrical linear actuator will need $3.64*10^{-4}$ kWh of energy and the motor will need $5.5*10^{-3}$ kWh of energy. They will both be running for 3 min for each sample for a total run time of 42 min.

The plan is to collect the sample and while the core is still spinning quickly pull the core shell and given that the samples will be collected near zero g environments the core catcher and the rotation will be enough to prevent significant loss of material. The shell will then be capped off and stored into the storage area. When ready to sample another area, the mechanism will then pick out an empty drill core shell, attach it to the rotational motor, and begin sampling.

3.4 Power, Propulsion and Thermal

This section discusses the power requirements for the EEV and how those will be met. Next, it describes the propulsion system which was designed for the EEV to maneuver between Earth and Marts orbit and then between the two moons and the DST to complete the mission. And lastly it details the thermal considerations that were taken.

3.4.1 Power Requirements

The current estimate for the power requirements of our mission is as follows:

Table 3.4.1 Power Requirements by Subsystem

Subsystem	Nominal Power Required (W)	Peak Power Required (W)	Mission Critical Systems (W)
Crew Systems	450	450 + 2080*	400
Scientific Instruments	45.4	80.9	-
Communications	70	150	50
Propulsion	285	300	285
Computers	680	1000	100
Thermal Control	40	3760	40
Total	**1,570**	**5,741 + 2080**	**875**

* One time power requirement of 2080 W when vaporizing cabin atmosphere

To satisfy this power budget, it is necessary to identify the nominal solar power input over a full day cycle. It is estimated by NASA that the average solar flux in the Mars sphere of influence is ~586 W/m^2 (NASA, 2022). Furthermore, the longest period of darkness experienced by the EEV will be when conducting sample collection on the dark side of the Martian moon Deimos. Deimos is tidally locked and has an orbital period of 30 hours and 18 minutes [4]. Due to being tidally locked, a point on the surface of the moon will be exposed to sunlight at a maximum 90° incidence for no more than half the orbit of Deimos, or 15 hours and 9 minutes. This duration is the maximum amount of time the EEV could experience darkness and thus becomes a critical design requirement for energy storage and power generation requirement.

With these numbers in mind, our estimated energy requirement for this period of darkness is 87 kWh under peak power of 5.8 kW for the total duration of darkness.

3.4.2 Energy Storage

Energy will be stored using rechargeable batteries. These have the following desired characteristics:

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1. Reliability and Robustness
2. Light weight and compact

Comparing various battery chemistries and energy density leads to choosing a Li-ion battery chemistry due to their high specific energy density. More specifically, Li-Cobalt batteries have the current highest specific energy density between 150 and 200 Wh/kg [5]:

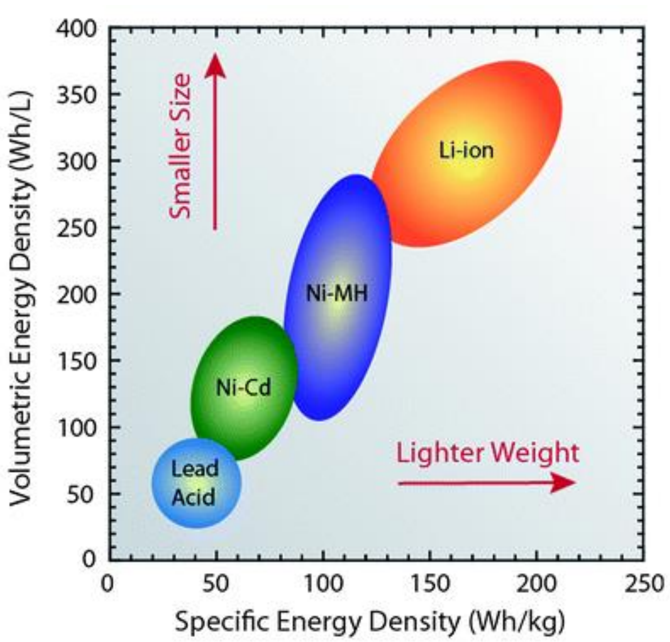

Figure 3.4.1 Volumetric Energy Density vs. Specific Energy Density [6]

Furthermore, these batteries have cells which can deliver up to 3.6 Volts, require little to no maintenance regarding our mission duration. Li-Cobalt batteries also have a low self-discharge rate of 1.5-2% per month (CEI, University of Washington). At end of mission, it is expected that these batteries would be holding no less than 80% of their original capacity. As such, for the life of our EEV (~9 months) these batteries are the ideal solution with current technology.

To accommodate for potential future development of onboard systems (such as new scientific instruments) which may require higher voltage or amperage than the legacy 28 Volt DC, the power supply is being designed for 120 Volts DC at 30 Amps-hour. This means each battery will be storing 3.6 kWh of energy with 34 cells at ~3.6

Volts. To respond to the 104.4 kWh energy requirement (assuming worst case of 80% battery capacity at end of mission), 29 of these Li-Cobalt batteries will be needed for the main power bank. Assuming a specific energy density of 200 Wh/kg, this will result in the main power bank having a mass of ~522 kg.

For redundancy, two additional power banks will be available to run mission critical components of the EEV requiring a total of 850 W of power. Mission critical has been defined as the life support, crew, propulsion, thermal, and avionics systems. For this, each emergency power bank is on its independent network and is designed to sustain the EEV during the longest period of darkness that could be experienced, 15 hours and 9 minutes. As such, each emergency power bank has a capacity of 15.8 kWh, accounting for end of mission 80% capacitance. This results in each emergency power bank being comprised of 5 batteries at 3.6 kWh and having a mass of ~79 kg.

Table 3.4.2 Power Bank Parameters

Characteristic	**Main Power Bank**	**2x Emergency Power Bank**
Peak Power Req	5,741 W	875 W
Longest Eclipse	15 hours 9 minutes	
Max Energy Req (During darkness)	86.2 kWh	13.2 kWh
Single Battery	3.6 kWh	3.6 kWh
# Batteries	29	5
Total Battery (80% capacitance)	104.4 kWh	15.8 kWh
Total Mass	522 kg	2 x 79 kg

3.4.3 Power Generation

Power will be generated by two solar panels which must adhere to the following requirements:

1. Generate power to satisfy peak power and energy storage charging requirements.
2. Dust mitigation capability
3. Deployment risk management

To satisfy the first requirement, a minimum of 6 kW of power must be generated to run the EEV at peak power solely using our power generation method. Furthermore, when the EEV has drained its main power bank during the worst-case eclipse scenario of 15 hours and 9 minutes it must be able to recharge this power bank in less time than the duration of the eclipse. As such, 104.4 kWh of energy must be generated in less than this time. For safety, a one-hour buffer zone after sunrise and before sunset will be used. This means 104.4 kWh to be generated in less than 13 hours and 9 minutes to charge the main power bank. Recalling the necessary 6 kW of power to run the

EEV, the total energy to be generated over 13 hours and 9 minutes is 182.4 kWh. Finally, this means approximately 14 kW of power per hour must be generated.

To achieve this, the EEV will have two solar panels each capable of producing 7 kW of power each at the Martian sphere of influence where solar flux is on average ~586 W/m^2. Solar cell efficiency is expected to reach >40% for multi-junction cells by 2040. Furthermore, power density for solar panels is estimated to be 200-250 W/kg by 2040 as well [7].

The current solar panel design is as follows:

Table 3.4.3 Solar Panel Characteristics

Characteristic	Solar Panel
Power Output	7 kW
Power Density	~200 W/kg
Total Area	30 m^2
Cell Efficiency	~0.40
Mass	~100 kg (70 + 30)
Dimensions	2.5x12 m

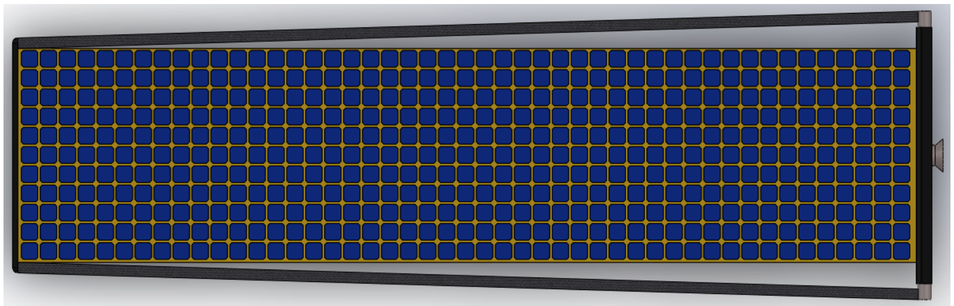

Figure 3.4.2 Single Solar Panel

The design is inspired by the Roll Out Solar Array (ROSA) experiment done on the ISS. Although the current ROSA deployment system is slightly bulky, we expect this to become more compact and lightweight by the mission date. This design is perfectly adapted to our mission as it is passively and elastically deployable, responding to our third requirement.

Finally, to respond to the dust mitigation requirement, we plan on incorporating the project currently in development by NASA's Jet Propulsion Laboratory and Colorado University Boulder called Moon Duster. This promising technology uses electron beams to remove dust on sensitive surfaces. Although currently a handheld device, it is imagined that this could be incorporated into the structure of the solar panels.

3.4.4 Propulsion

The propulsion system for the EEV was designed in two steps. First an initial design was proposed with certain fixed parameters. Then, after validation and review, a more precise design was generated using updated parameters and iterative calculations done with Rocket Propulsion Analysis tool.

The following design requirements/constraints were considered:

1. Target delta-V of 2.8 km/s
2. Redundancy
3. Reliability
4. Efficiency

These were the primary points of focus while designing the propulsion system of the EEV. Due to the mission profile, multiple considerations were made when making the design:

Our first approach was considering the type of propulsion system appropriate for this mission: ionic, nuclear, or chemical. The EEV has 30 days to travel to the two moons of Mars which would require a delta-V of ~2.8 km/s. This consideration already eliminated ionic propulsion due to its inadequate thrust to weight ratio. Furthermore, the EEV would be in transit to the Martian SOI for ~8 months after launching from Falcon Heavy. With the mission duration and the mass constraint of launching on Falcon Heavy, nuclear propulsion was also eliminated.

Having chosen chemical propulsion a new set of constraints arose: long term propellant storage, type of engine cycle, and impulse. The first hurdle was choosing the type of fuel to use for our propulsion system.

We initially considered multiple fuels however, after research it was narrowed down to liquid hydrogen – LH2, or liquid methane – LCH4. First, we considered the long-term storage of the fuel. Both LH2 and LCH4 are cryogenics, meaning they must be kept at extremely low temperatures as well as high pressure to be stored as liquid as this is the most efficient way to store them when volume is limited. Important to note that LH2 must be stored at a much lower temperature than LCH4, see figure 3.3.4-1 and 3.3.4-2:

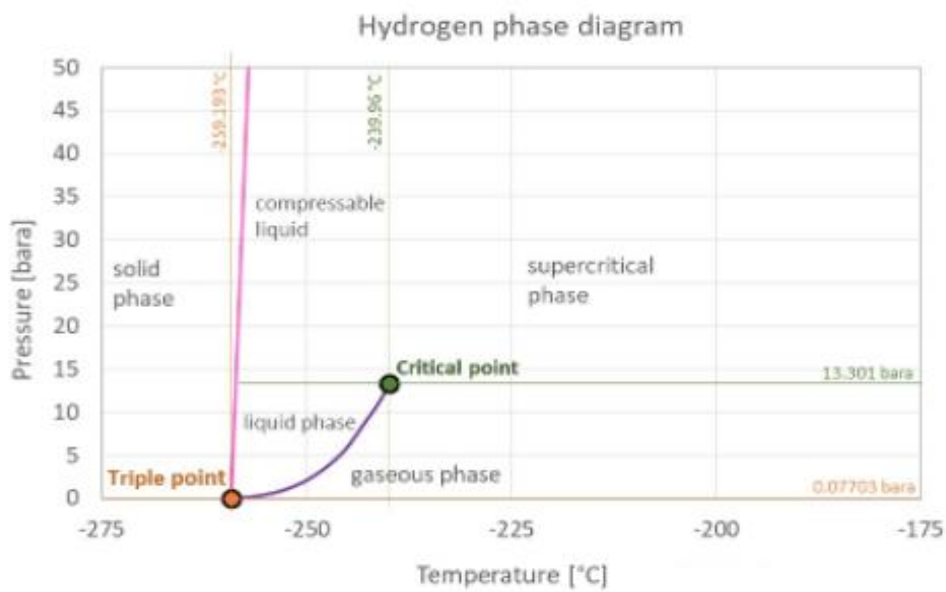

Figure 3.4.3 Hydrogen Phase Diagram [8]

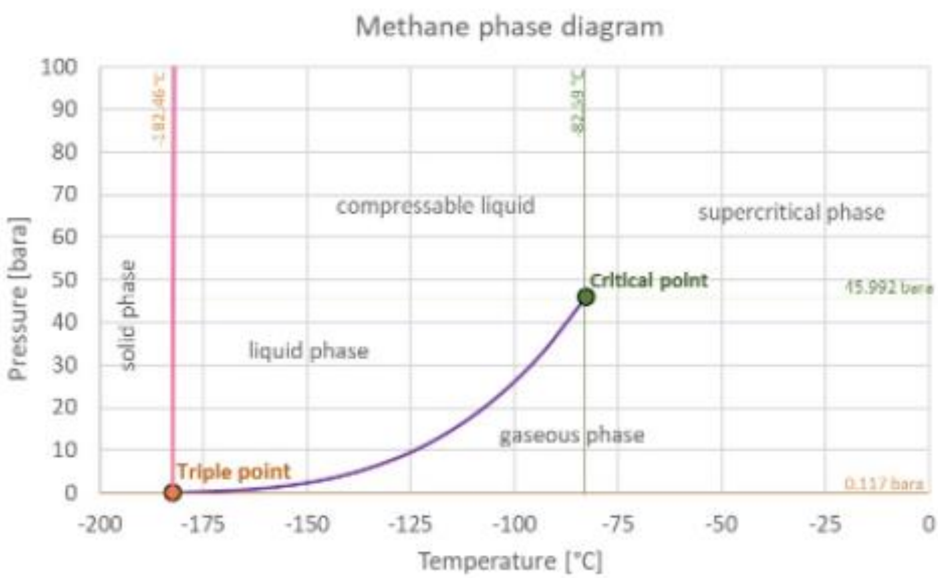

Figure 3.4.4 Methane Phase Diagram [8]

Comparing the two diagrams we can observe that LCH4 can be kept in liquid phase at a higher temperature than LH2 (at equivalent pressure). This is important to note since lower temperatures require more cooling power.

Furthering our analysis of propellant, we compared LCH4 to LH2. Methane has multiple advantages to Hydrogen, namely:

- Higher vapor pressure (excellent vacuum ignition)
- 6 times denser (great for storage volume)
- Safer to transport: non-toxic, non-corrosive, self-venting [9].

The main advantage of liquid hydrogen compared to liquid methane is its specific energy. LH2 has approximately 120 MJ/kg compared to LCH4 55.6 MJ/kg [8] [10]. This essentially means that hydrogen can achieve a higher ISP than methane. While this is a strong argument for choosing LH2 over LCH4 in general, it is not appropriate for our mission. Storing the equivalent amount of LH2 and LCH4 to achieve the same amount of delta-V will require fuel tanks 6 times the size. This unfortunately would make the EEV oversized and unable to launch on Falcon Heavy. Ultimately liquid Methane was chosen as the fuel and liquid oxygen as the oxidizer for the propulsion system.

Before designing the propellant tanks, we began the process of designing the EEV main engine. For this design we started with a few fixed parameters to allow us to manipulate the isentropic equations.

- Thrust of 50,000 Newtons
- ISP value of 365 seconds
- Area of exit / Area of throat ratio of 80
- Oxidizer to Fuel ratio of 3.4

We chose to have 4 engines in quad formation for redundancy. Furthermore, we chose a thrust of 50 kN for a single engine. This is because we desired a thrust to weight ratio of at least 1.0 in the event two engine failed. This is to allow for timely maneuvering back to the DST in the event of engine failures. Two engines can provide 100 kN of thrust, enough for the estimated 10,000 kg dry mass of the EEV.

The ISP of 365 seconds was determined by studying other Methalox (LCH4/LOX) vacuum engines such as Space X’s Raptor engine (vacuum variant) or the European Space Agency’s M10 engine. Similarly, the initial combustion chamber pressure was also chosen to be 6 MPa and O/F mixture ratio of 3.4. From these initial values we utilized the Rocket Propulsion Analysis tool (a CEA – NASA equivalent tool) to better refine our engine.

First, we looked at ISP vs Ae/At:

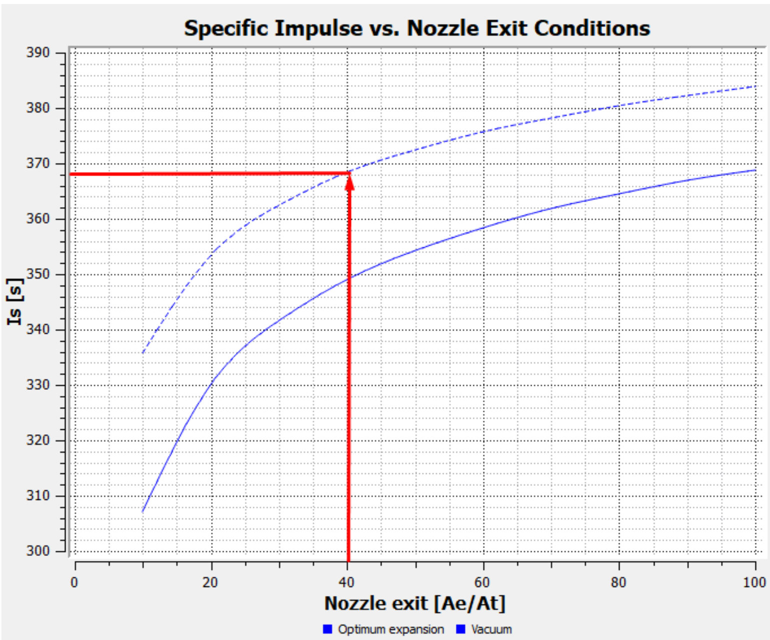

Figure 3.4.5 Specific Impulse vs. Nozzle Exit Conditions

Highlighted on figure 3.3.4-3 is our initial choice for the Ae/At ratio of 40. However, after development on the EEV progressed it was found that the height of the EEV within the fairing of the Falcon Heavy allowed us to increase the size of our engine nozzles, ultimately netting the highest overall gain in ISP while keeping other parameters constant.

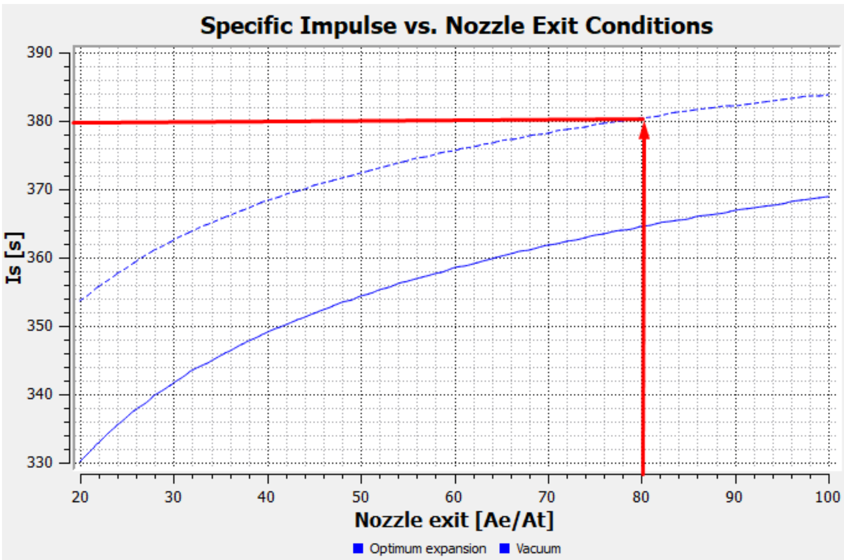

Figure 3.4.6 Specific Impulse vs. Nozzle Exit Conditions

With this new Ae/At ratio, our ISP jumped to 380 s for a chamber pressure of 6 MPa and O/F ratio of 3.4.

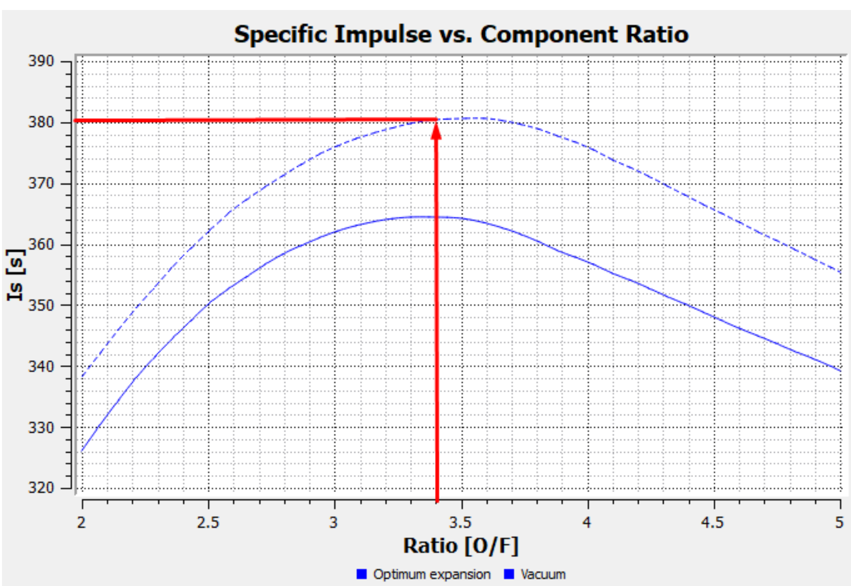

Figure 3.4.7 Specific Impulse vs. Component Ratio

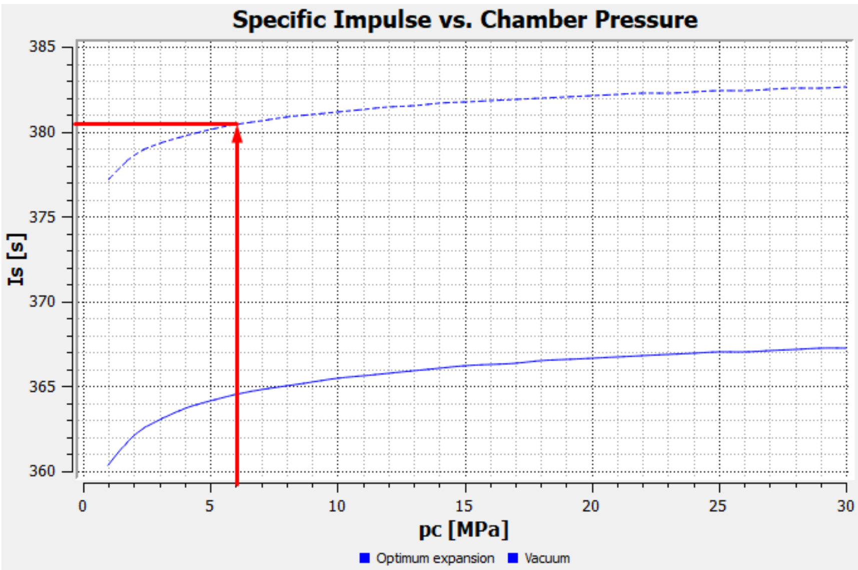

Figure 3.4.8 Specific Impulse vs. Chamber Pressure

Figures 3.3.7 and 3.3.8 show that adjusting the chamber pressure and O/F ratio do not affect the ISP as much as varying the Ae/At ratio. From these, our engine performance metrics are as follows:

Table 3.4.4 Single Engine Performance Characteristics

Single Engine Performance								
Characteristic	Pc	Tc	Pe	Te	Me	Thrust	Isp	Ae/At
Value (unit)	6.0 MPa	~3530 K	0.0064 MPa	~1579 K	4.37	50 kN	380 s	80

Due to our mission, multiple engine restarts will be required. In vacuum, this can be difficult if not impossible when using an expander cycle design due to the fuel needing to be regeneratively heated by the nozzle to drive the turbopumps. Therefore, we have opted for a gas generator augmented expander cycle designed by W. Greene for the US government in 2013. Although theoretical, this engine cycle design would allow for reliable engine restarts in vacuum thanks to its gas generator cycle. It also provides the added functionality of turning off the gas generator once full ignition is achieved. This allows for the engine to run fully as an expander cycle, increasing reliability. See figure 3.3.9 for schematic of engine.

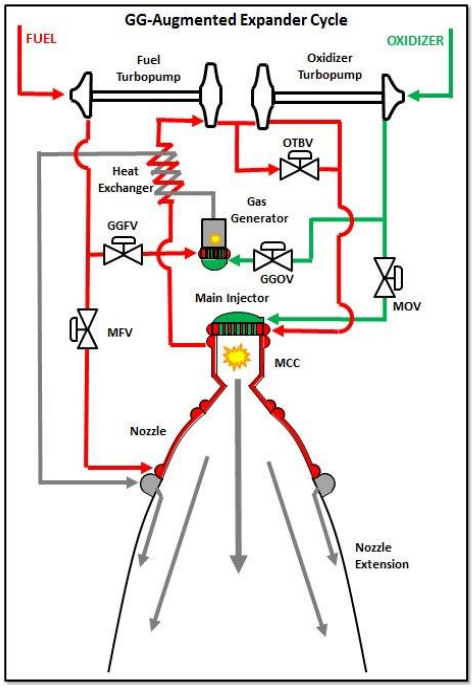

Figure 3.4.9 GG-Augmented Expander Cycle

Table 3.4.5 GG-Augmented Expander Cycle

Dimension	Value (unit)
Throat diameter	73 mm
Throat radius	36.5 mm
Exit diameter	655 mm
Throat Area	0.0042 m^2
Exit Area	0.336 m^2
Throat to Exit	~736 mm
Chamber length	~250 mm

In the table above, you will find the estimated dimensions of a single engine based off parabolic approximation by G.V.R. Rao. To effectively fuel the four engines for this ~9 month mission, the propellant will need to be kept and maintained in liquid phase. The liquid oxygen will be stored at 6 MPa and 90 K and the liquid methane at 6 MPa and 90 K. To achieve and maintain these temperatures, the propellant tanks will be covered in multi-layered insulation. Note, due to the Delta-V requirement of the 30 day mission and the size of the payload bay of the Falcon Heavy, we are limited in the thickness of the tanks and insulation. As such, cryocoolers, 5 per tank for redundancy, will provide active thermal control. The peak cooling power required will be experienced in low earth orbit where the solar flux is greatest. Each of the cryocoolers have 30 W of cooling power. See below for the tank characteristics:

Table 3.4.6 Tank Characteristics

Delta-V (2,745 m/s)		
Characteristic	Value (unit)	
Prop mass	10,010 kg	
Oxidizer mass	7,738 kg	
Fuel mass	2,271 kg	
Oxidizer flow rate (100% throttle)	10.80 kg/s	
Fuel flow rate (100% throttle)	3.18 kg/s	
Tank Properties	**LOX**	**LCH4**
Tank Volume	3.03 m^3	3.25 m^3
Tank Area	10.15 m^2	10.62 m^2
Tank Diameter	1.79 m	1.84 m
Tank Thickness	9.0 cm	9.2 cm
MLI Thickness	1.27 cm	
Peak Cooling Power (LEO)	147 W	153 W
Total Peak Power	300 W	
Tank Mass	2 x 96 kg	2 x 55 kg
Cryocooler Mass	10 x 20 kg	
Total Mass	**502 kg**	

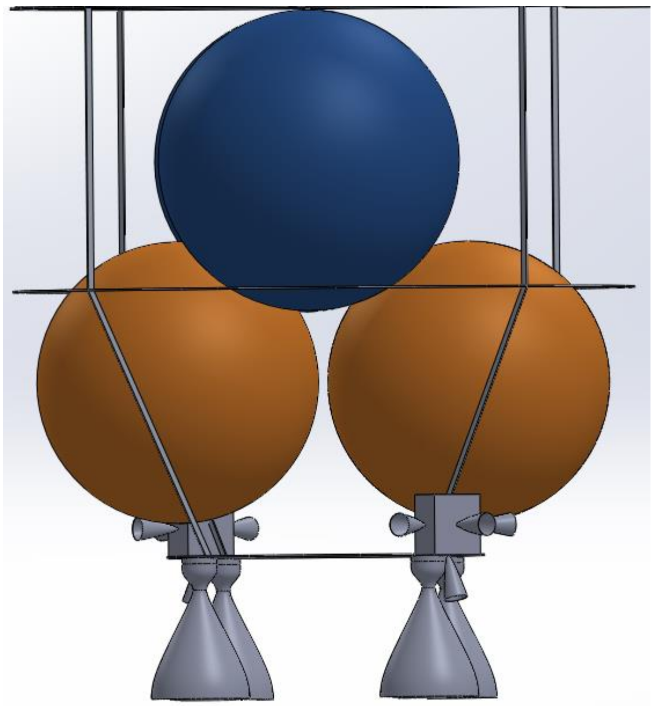

Figure 3.4.10 EEV CAD of Propulsion System (CH4 in blue, LOX in orange)

For the booster stage, the exact same design method was adopted as for the EEV. However, the mission time for the booster stage is no more than 1 month in LEO. This stage can be described as simply propellant tanks with a nozzle attached, hence redundancy and reliability is not as important as for the EEV. Its sole purpose is to send the EEV on a trajectory to Mars. To accomplish this, the booster stage requires ~4.11 km/s. However, Methalox cannot be used as the maximum achievable ISP is too low and the required volume of propellant does not fit on the Falcon Heavy. As such, a LH2/LOX engine must be used. Inspiring our engine off the RL-10 and Vinci, we chose a O/F ratio of 5.2, and chamber pressure of 15 MPa. We also picked an Ae/At ratio of 160 since the booster stage would utilize a single engine.

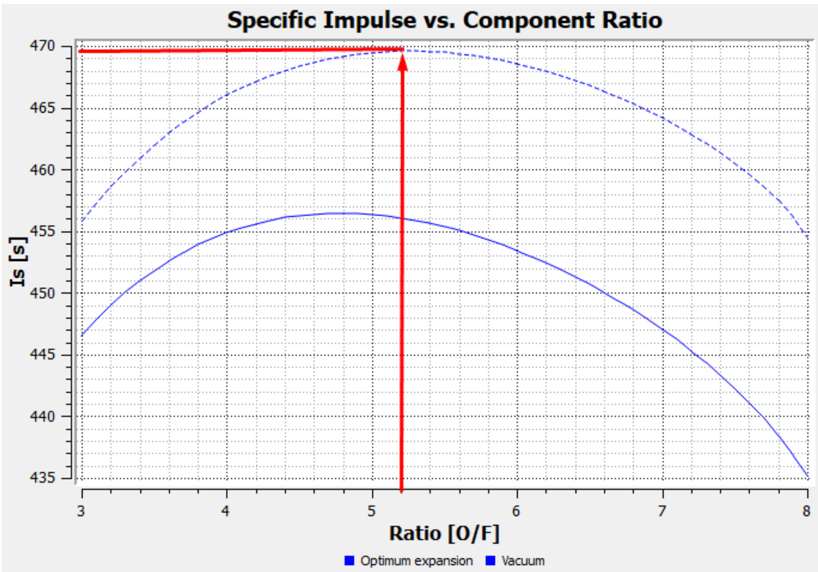

Figure 3.4.11 Booster Specific Impulse vs. Component Ratio

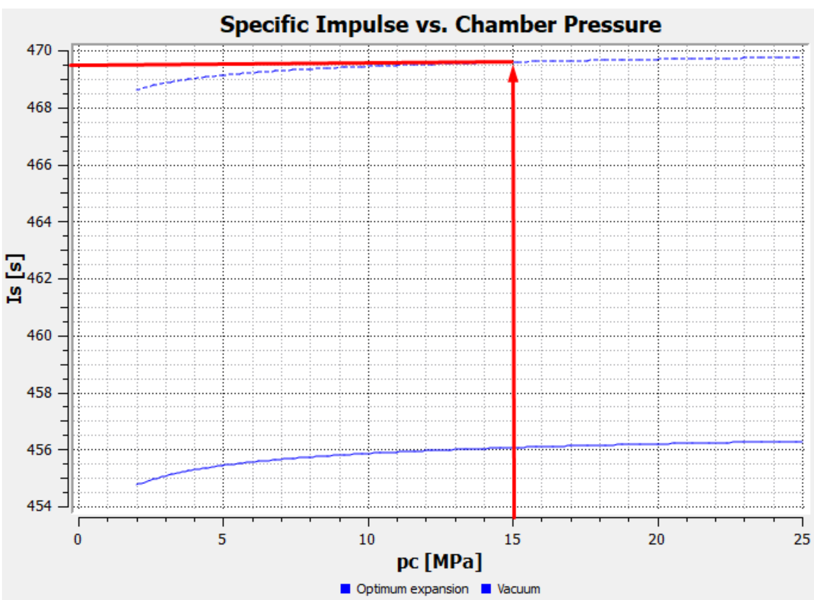

Figure 3.4.12 Booster Specific Impulse vs. Chamber Pressure

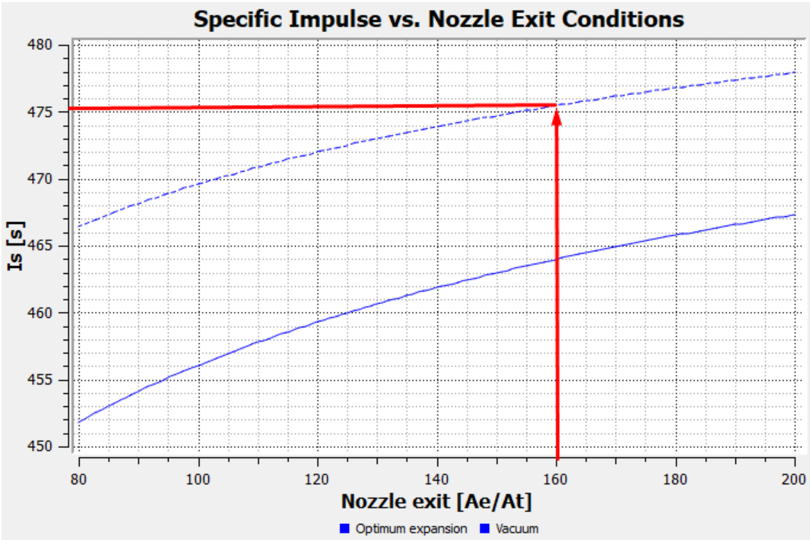

Figure 3.4.13 Booster Specific Impulse vs. Nozzle Exit Conditions

Table 3.4.7 Booster Single Engine Performance Characteristics

Single Engine Performance								
Characteristic	Pc	Tc	Pe	Te	Me	Thrust	Isp	Ae/At
Value (unit)	15.0 MPa	~3392 K	0.0045 MPa	~769 K	5.56	600 kN	~475 s	160

The engine for the booster stage will be a staged combustion cycle engine, see figure 3.3.14.

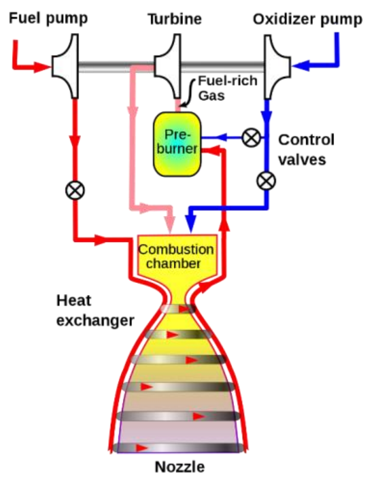

Figure 3.4.14 Staged Combustion Cycle

To fuel the engine for this ~1 month mission, the propellant will need to be kept and maintained in liquid phase. The liquid oxygen will be stored at 6 MPa and 90 K and the liquid hydrogen at 10 MPa and 35 K. To achieve and maintain these temperatures, the propellant tanks will be covered in multi-layered insulation. As such, cryocoolers, 5 per tank for redundancy like the EEV, will provide active thermal control. The peak cooling power required will be experienced in low earth orbit where the solar flux is greatest. Each of the cryocoolers have 30 W of cooling power. Unlike the EEV, since most of the booster structure is fuel tanks, the MLI thickness was increased to 10.27 cm allowing to reduce the cooling power required to 337 W. See below for the tank characteristics:

Table 3.4.8 Tank Properties for Booster

Delta-V (4,110 m/s)	
Characteristic	Value (unit)
Prop mass	47,133 kg
Oxidizer mass	39,534 kg
Fuel mass	7,602 kg
Oxidizer flow rate (100% throttle)	110.66 kg/s
Fuel flow rate (100% throttle)	21.28 kg/s

Tank Properties	LOX	LH2
Tank Volume	31.78 m^3	99.75 m^3
Tank Area	48.53 m^2	113.5 m^2
Tank Height	3.12 m	4.4 m
Tank Radius	1.56 m	2.1 m
Tank Thickness	15.6 cm	21 cm
MLI Thickness	10.27cm	
Peak Cooling Power (LEO)	101 W	337 W
Total Peak Power	337 W	
Tank Mass	497 kg	1,147 kg
Cryocooler Mass	10 x 20 kg	
Total Mass	**1,844 kg**	

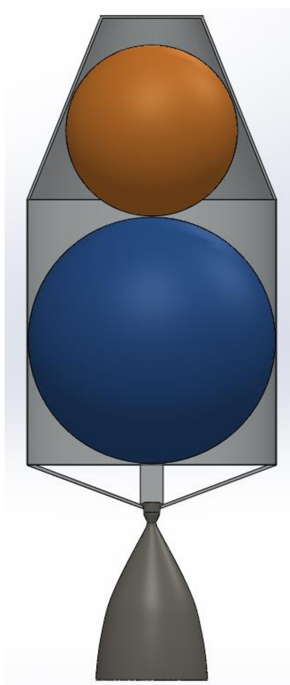

Figure 3.4.15 Booster Stage CAD (LH2 in blue, LOX in orange)

3.4.5 Thermal Balance and Control

Within the orbit of Mars, equilibrium temperatures range from 209.6 K (-63.55 °C) to 4 K (-269.15 °C) [11]. With this wide range of temperatures, it was crucial to design the EEV thermal subsystem to be able to heat the craft at the coldest point and cool it down in the hottest environment. These environments were summarized in the following table:

Table 3.4.9 Thermal Sources and Losses

Subsystem/Element	Heat (W)
Intercepted Solar Flux (max)	+ 4880
Radiation out (full sunlight)	- 4170
Radiation out (full shadow)	- 5570
Heat from Crew (highest)	+ 800
Heat from Crew (least)	+ 234
Heat from Electrical Components (nominal)	+ 1570
Heat from Electrical Components (peak)	+ 3825
Most Heat Rejected	**5335**
Most Heat Needed	**3760**

In the initial stages of the design process, multi-layer insulation (MLI) was considered for the housing of the crew cabin, but after a few design iterations it was found to not be necessary for the EEV. Between the two layers of aluminum and the amount of heat being produced by other subsystems it was found that the MLI wouldn't need to prevent heat from escaping as radiating heat away from the EEV was the bigger concern for this mission. Additionally, without the added weight of the MLI layers, the EEV was able to budget more weight to other subsystems.

To control the heating and cooling of the EEV, a scaled down version of the Active Thermal Control System (ACTS) from the International Space Station (ISS) will be used. This system uses a series of water loops around the cabin to transfer the heat from the various sources within the habitable volume [12]. This water is then pumped to a heat exchanger on the outer surface of the EEV where the water loop heats up a separate loop of ammonia. Only the ammonia will run on the outside of the EEV and be subject to the outside environment. Ammonia is ideal for heat exchanging in the sunlight as the freezing point (196.15 K) is below the expected average temperature in full sunlight in Mars orbit.

The ammonia loop will be run through a radiator which is another scaled down version of the one on the ISS. To dissipate the 5.4 kW of energy during max heating the radiator needed to be 15.0 m^2. Using a report on the radiators from the ISS, this radiator would weight about 118 kg and be able to pump 1.5-200 kg/hr of ammonia through the system [13].

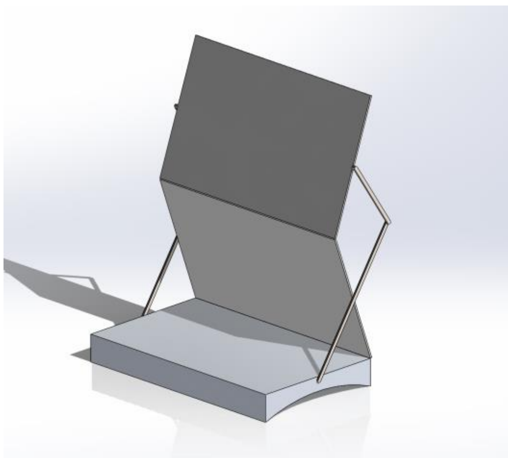

Figure 3.4.16 Radiator Assembly detached from the EEV

In the situation where the EEV needs to be heated, off-the-shelf standard patch heaters will be purchased from one of NASA's suppliers Minco. These heaters are electrical resistance elements sandwiched between two pieces of Kapton [14]. They can be ordered in many different sizes and have a near 100% efficiency rating. There will be various circuits for redundancy and the mass is virtually negligible.

3.5 Scientific Instrumentation

To accomplish the scientific objectives, a variety of instrumentation must be delivered and deployed onto the moons. A table containing each scientific instrument along with its high-level purpose is given below.

Table 3.5.1 Scientific Instruments

Instrument	Purpose
Seismometer	Measure the seismic activity on Phobos and Deimos
Radiation Assessment Detector	Measure radiation levels of the surrounding area of the moons
Alpha Particle X-ray Spectrometer	Measure the chemical composition of the soil of the moons

Of these instruments, the Seismometer, Radiation Assessment Detector (RAD), and the Alpha Particle X-ray Spectrometer (APXS) are all adapted from previous Martian rover missions. Because our intents for scientific instruments align very well with that of the Mars missions, it was determined to be advantageous to recycle many of the designs from these Mars missions. These instruments have the advantage of being already rated for extreme environmental conditions, as well as the ability to be deployed entirely autonomously, a necessity for our mission. In addition to this, we also plan to have a 4k camera to capture images and live video of Phobos' and Deimos' terrain.

3.5.1 Seismometer

The seismometer will be used to measure the seismic activity of the moons of Mars. Below is a cross-sectional image of the device.

Figure 3.5.1 Seismometer cross-sectional image

The device uses seismic perturbations to measure the activity. This instrument was originally deployed during Mars Rover missions, where seismic perturbations occurred naturally through meteor collisions [15]. Alternatively, we cannot rely on natural events to trigger seismic waves on the moons of Mars and will instead need to actively launch projectiles. Quillen, through a research article, presents a method of determining the size and speed of projectiles to generate meaningful results.

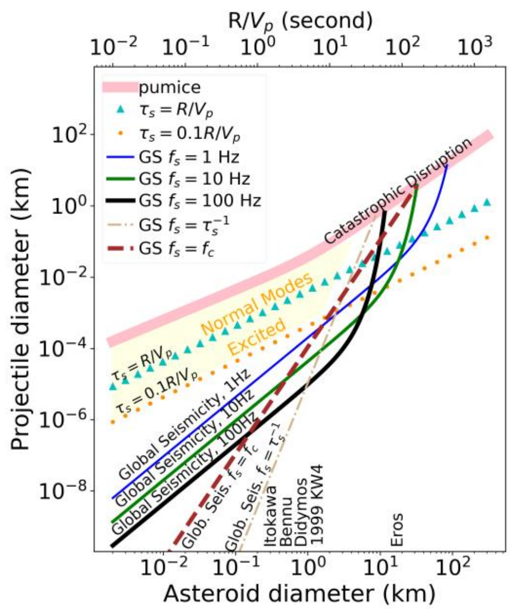

Figure 3.5.2 Seismic activity in the presence of projectiles

The yellow area shows "regime where a strong impact lies just below catastrophic disruption, but above that giving global seismic reverberation" [15]. As Phobos and Deimos's diameters are both on the order of 10 km, meaningful seismic data should be attainable. As Quillen writes, "Solar system formation and evolution theories have changed over the years, and in the last two- or three-years, theory has leaned toward fragmented interiors for small bodies, but still no direct data is available." Doing seismic tests on the moons will help with understanding what their interior really looks like. Knowing the interior of the moons is important, because through them we can deduce what the interior of other similar asteroids are, which will help in determining important factors such as if Earth collisions would be detrimental.

3.5.2 Radiation Assessment Detector

The RAD was previously used on the Mars Curiosity Rover. It is similar in size to a small toaster and is composed of silicon detectors, a small block of cesium iodide, and an internal signal processor [19]. The RAD's main function is to monitor radiation in space on the way to Mars but can extend to Phobos and Deimos as well. Knowing this information would aid future human missions to Mars by measuring how much radiation astronauts need shielding from for protection. The RAD works by measuring and identifying all high-energy radiation on the Martian surface, including protons, neutrons, gamma rays, direct radiation from the sun and space, and secondary radiation produced from the interaction between the Martian atmosphere and surface rocks and soils [19]. After collecting data, scientists can calculate an equivalent dose of radiation that humans would be exposed to on Mars and its moons. The effect of radiation on potential microbial life and chemical and isotopic composition of regolith can also be assessed, helping scientists to get a full picture of Phobos and Deimos' environment. Figure 3.9 shows a computer model of the RAD [19].

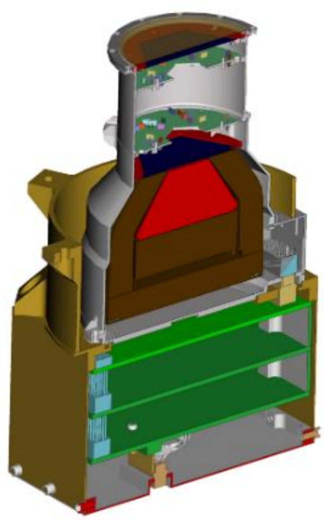

Figure 3.5.3 RAD Computer Model

3.5.3 Alpha Particle X-ray Spectrometer

The APXS was used on both the Mars Curiosity and Perseverance Rovers and is the about the size of a cupcake. Its main function is to analyze chemical elements present in Phobos and Deimos' regolith. The APX can take both day and night measurements with a runtime of two to three hours to reveal all elements or ten minutes to reveal major elements [20]. With the samples collected on each moon, the APXS will expose them to alpha particles and X-rays emitted during radioactive decay of curium [20]. Energies produced by the X-rays allow scientists to identify rock-forming elements, such as sodium [20]. Data from the APXS lets scientists characterize and examine the interiors of rock and soil samples. Through analysis, scientists will be able to understand material in the moons' regolith formed and if it was later altered by water, wind or ice, helping to support the theories previously mentioned above. Figure is an image of the actual APXS used on the Mars Exploration Rovers.

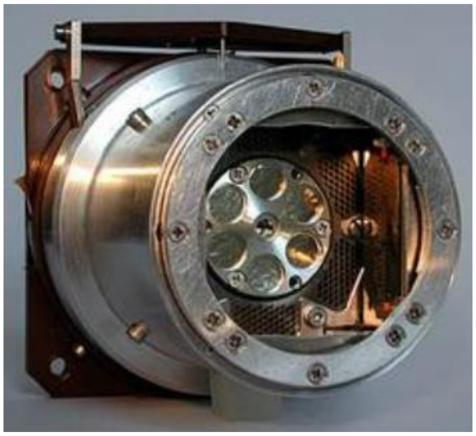

Figure 3.5.4 APXS on Mars Exploration Rovers

3.5.4 Deployment Mechanism

In designing the deployment mechanism for the scientific instruments, the physical constraints and necessary functionality of the instruments had to be considered. First, both the seismometer and APXS must be in direct contact with the soil. As such, the EEV must land with them facing into the soil. The RAD must be exposed to the environment, but not in direct contact with soil. With these constraints in mind, the Instrument Deployment Mechanism (IDM) was designed. On the left the deployment mechanism is shown in full, and on the right shows the relative placement of the instruments within the IDM.

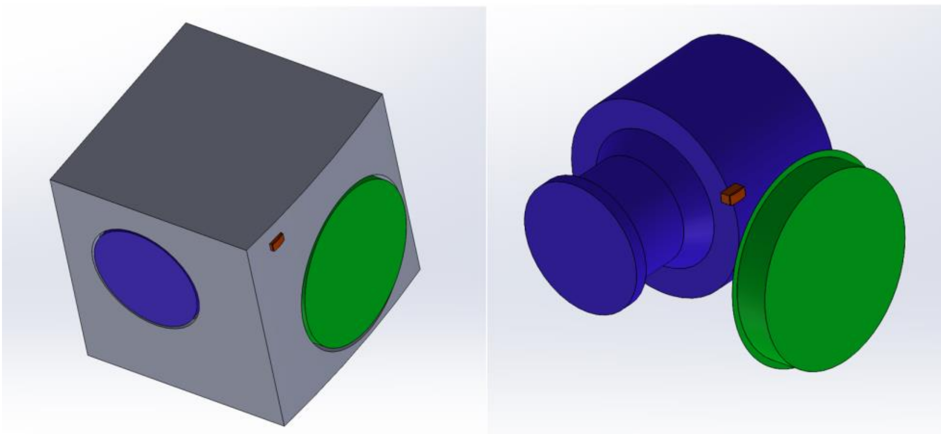

Figure 3.5.5 Instrument Deployment Mechanism (IDM)

Below is pictured the IDM within the EEV. It will be deployed using a rack-and-pinion mechanism with a latch.

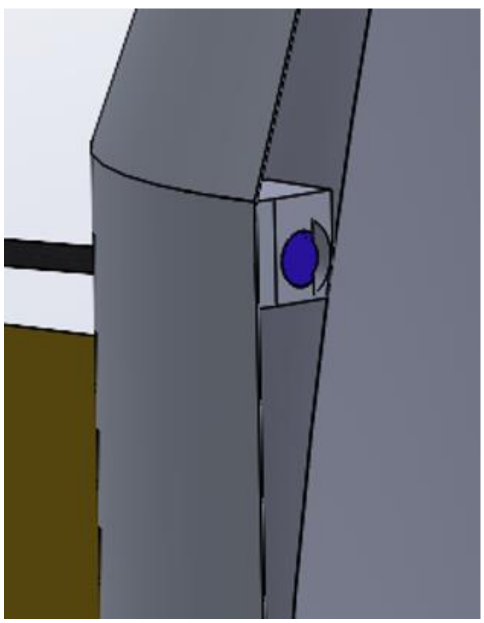

Figure 3.5.6 IDM within EEV

3.6 Avionics

3.6.1 Sensors and Telemetry

In order to keep the crew alive, the EEV will need to have systems to monitor conditions in the cabin: temperature, pressure, carbon monoxide level, and oxygen level, for instance. The EEV will also have several smoke detectors around the interior to ensure the crew catch any dangerous fires quickly.

For navigation, the craft will have sun and star sensors to help orient itself in space. A gyroscope and accelerometer will help track rotational and translational motion. a LIDAR-based altimetry system will allow the EEV to map notable parts of the lunar terrain from orbit, and help the crew navigate during descent and landing.

3.6.2 Inertial Frame of Reference

The Inertial Frame of Reference of the EEV is defined as the *y* axis in the direction of the right solar panel, the *z* axis in the axial direction towards, and the *x* axis completing the frame in a right-handed system. The origin is centered on the center of mass of the spacecraft.

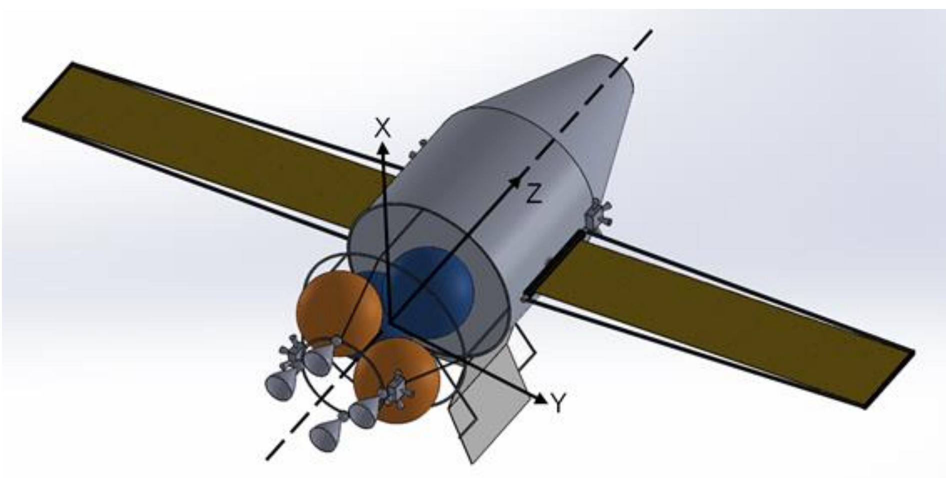

Figure 3.6.1 Inertial Frame of Reference

3.6.3 Spacecraft Dynamics

When determining how the spacecraft will move in the vacuum of space, two important parameters needed to be found: center of mass, and moment of inertia. From these two parameters, the strategy for controlling the spacecraft to a desired location or orientation can be determined.

The center of mass was determined by averaging the point masses of each subsystem. The assumption of completely full tanks, which add most of the fuel. Since this is the 'worst case' for maneuvers, the state of full tanks was designed around. Using this approach, we found a center of mass of 2.7 meters from the bottom of the spacecraft, less than a meter above the local fuel-propellant tank center of mass.

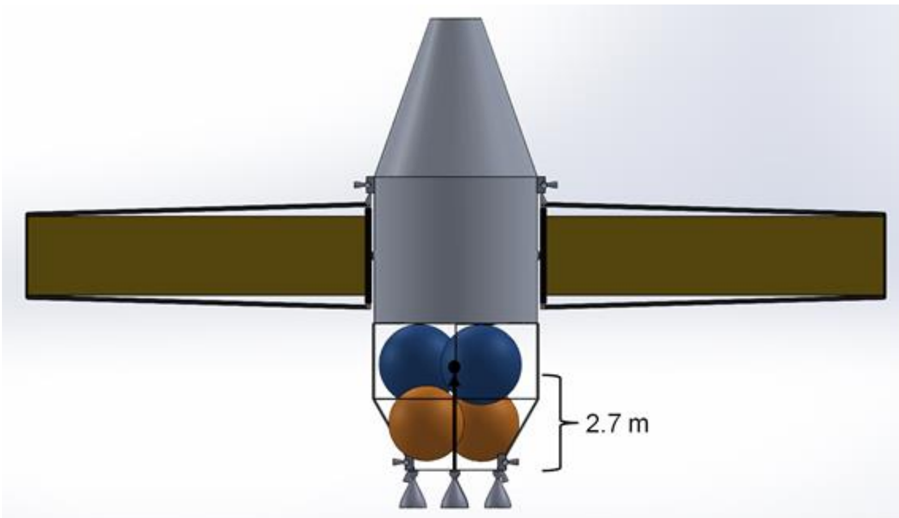

Figure 3.6.2 Center of Mass Depicted on the EEV

To calculate the Moment of Inertia (MoI), the geometric MoI of the structure of the EEV was taken, and added to it was point mass assumptions for all other components, using the Parallel Axis Theorem. Given below is the MoI matrix.

$$\boldsymbol{I} = \begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & 0 \\ 0 & 0 & I_z \end{bmatrix} = \begin{bmatrix} 41500 & 0 & 0 \\ 0 & 41500 & 0 \\ 0 & 0 & 87600 \end{bmatrix} kg - m^2$$

Equation 3.6.1 Matrix Moment of Inertia

The highest moment of inertia is about the *z* axis, which will dictate the design of the Reaction Control System.

3.6.4 Reaction Control System

A series of reaction control thrusters (RCT) will be used to perform orientation maneuvers. Four RCT modules of four nozzles each will be placed on the EEV, two located on the bottom, next to the main thrusters, in line with the solar panels. Two will be located on the same *x-y* plane, next to the solar panels. These locations were chosen to maximize torque about the *z* axis, the axis that is most difficult to rotate about. Here is a look at a single RCT module, followed by a look at the placing of each RCT module.

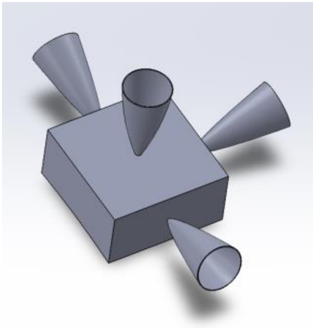

Figure 3.6.3 RCT module

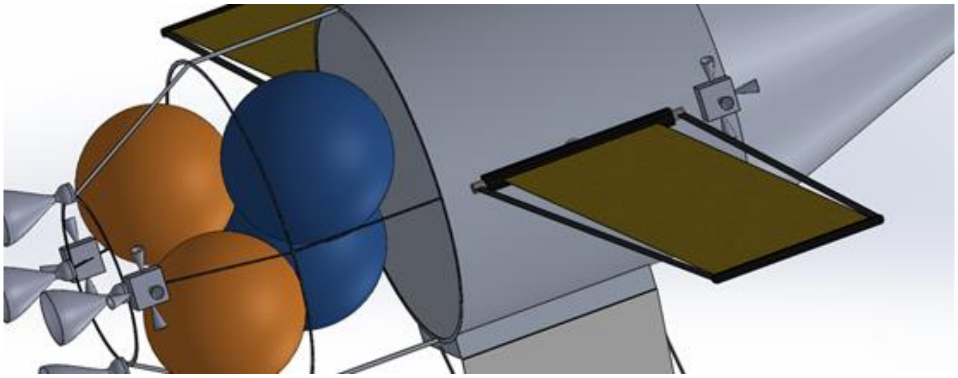

Figure 3.6.4 RCT mounting

The RCT modules do not include inward facing *z* axis thrusters, as the exhaust plume would be directed into the solar panels. As such, maneuverability about the *x* axis will be slightly more difficult. However, the trade-off from getting the passive power generation of the solar panels was deemed to be more valuable than extra maneuverability about the *x* axis. The RCS thrusters will share the methane-oxygen fuel-propellant combination with the main thrusters, which will allow for fewer total amount of fuel/propellant tanks. The RCS thrusters have a thrust of 440N and a specific impulse of 395 seconds.

Commented [TFB12]: RCS geometry here

3.6.5 Compensator Design

In designing the compensator for attitude control, the following approach was taken: since small perturbations and precise dynamics are not known, use the idealized model for rotation and design large margins, using strategies that mitigate the effects of sensor noise, time delay, and so on. As such, the plant transfer function follows simple Newtonian dynamics, where *I* is the mass MoI:

$$G = \frac{1}{I * s^2}$$

Equation 3.6.2 Plant transfer function for attitude control

Several compensators with varying phase margin and magnitude crossover, including Lead, Proportional-Integral (PI), Proportional-Derivative (PD), and Proportional-Integral-Derivative (PID). Since rate data for attitude is available through gyroscopes, intuitively it made more sense to go with compensators that include derivative terms. This was supported by simulations that showed better response characteristics, such as lower settling time and peak overshoot. After testing the potential options with a variety of compensator Ultimately, a PD compensator with a phase margin of 85 degrees and a crossover frequency of 0.3 rad/sec was chosen, as it proved to perform best given while minimizing controller saturation. The compensator is shown below.

$$H = 2.618 * 10^4\, s + 687.1$$

Equation 3.6.3 Attitude control compensator

Below is shown a step response of 1 radian:

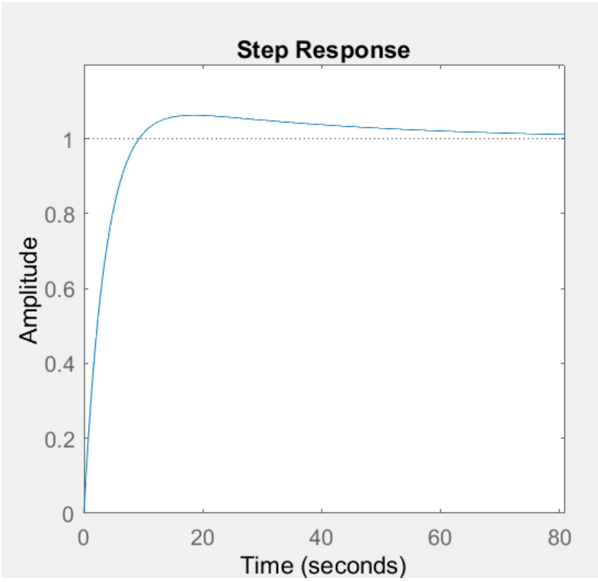

Figure 3.6.5 Step Response

The compensator exhibits a settling time of 61.9 seconds, and a low peak overshoot of 6.29%. While the settling time may seem high, the rise time is quite low, at under 10 seconds. Considering that many of the maneuvers necessary at each moon do not need to be extremely precise, and that the crew has about 13 days at each moon, this compensator will perform satisfactorily in most cases, while having the ability to give precise attitude control when necessary.

Two important parameters when discussing the RCS performance are the time required to maneuver an angle or a distance in the most optimal sense, which is defined as a doublet function of the control system, where the system saturates the controller in one direction for half of the total time, and then saturates in the other direction for the remaining half. The *z* axis was chosen to do this analysis, as this is the most difficult axis to rotate about (highest MoI). The required degree change was plotted against the time required to do the degree change.

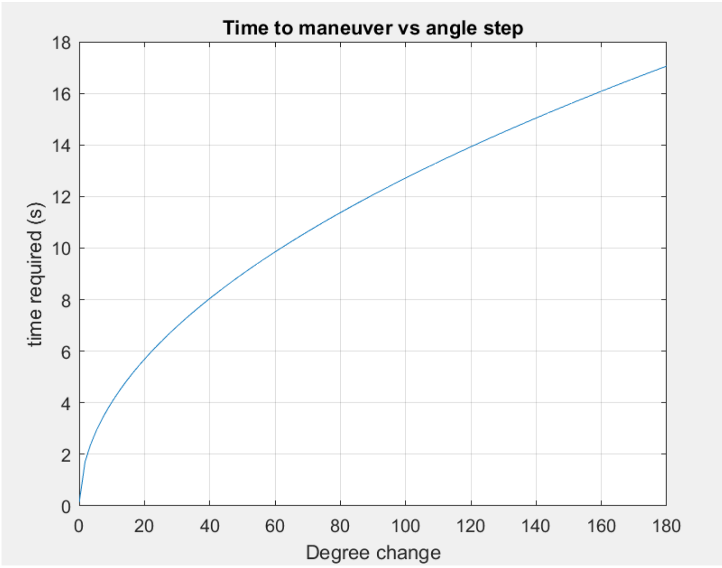

Figure 3.6.6 Time to maneuver vs angle step

Similarly, the required location change was plotted against the time required to do the location change.

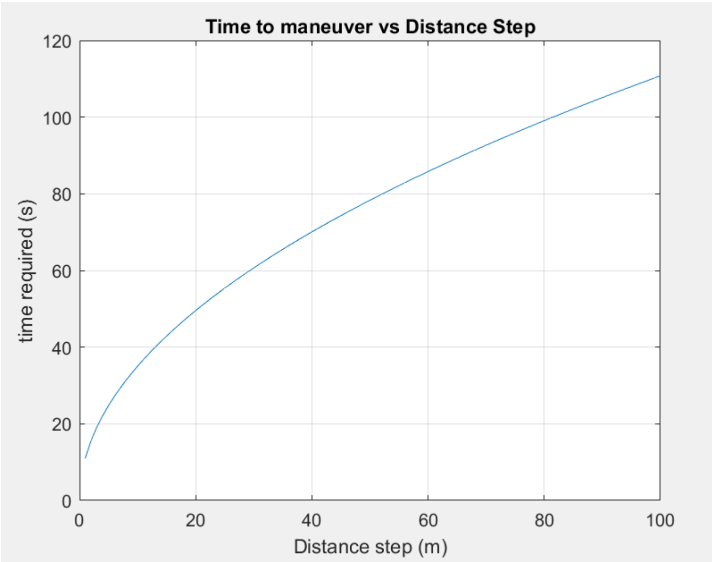

Figure 3.6.7 Time to maneuver vs distance step

The system exhibits satisfactory performance with a 180-degree change taking 17 seconds as the minimum possible time, and a distance step of 80m taking 100 seconds. Both of these are well within the required time to visit several locations on each moon within the 13 days on each of them.

3.6.6 User Control

As part of the mission requirements, the EEV must be user controllable. If a crew member sees something interesting on the surface of the moons, such as an unexpected crater, or an unexpected material, the crew member will be able to navigate the EEV using the RCS thrusters. Below is shown the user controller, a series of two 3-dimensional joysticks which control rotation and translation.

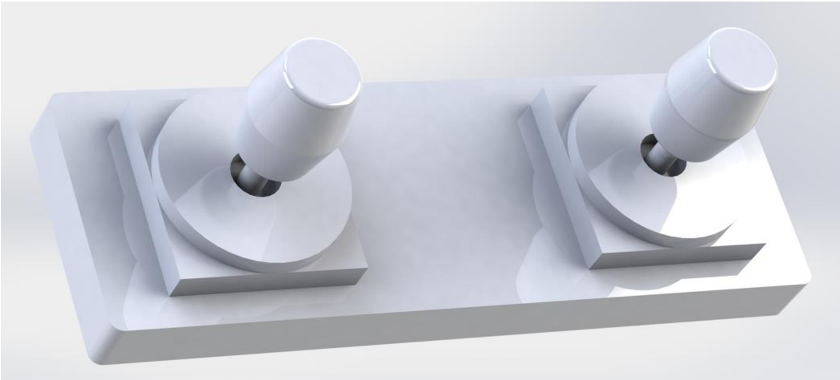

Figure 3.6.8 User Controller, Isometric View

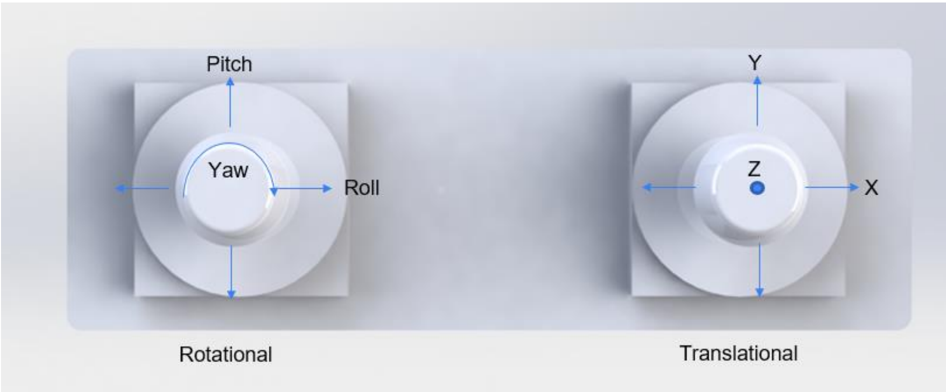

Figure 3.6.9 User Controller, Top-down view

The rotational joystick is twistable, which will control yaw. The translational joystick is able to be pushed in and out to control z-axis motion.

3.5.7 Communications System

The communications system consists of several subsystems. Firstly, an ultra-high frequency (UHF) radio system is necessary for short-range communications. This includes communication with the DST and with the Mars Relay Network (which relays information between the spacecraft and Earth). This can be used for essential communications, including telemetry and voice communication. It will primarily serve as a backup system. The primary communication system will be laser-based. In addition to allowing the crews on the DST and EEV to communicate, the DST will serve as a relay to facilitate communication between Earth and the EEV.

With optical communications, it becomes possible to transmit an amount of data that surpasses any radio-based communications. With this, we plan to transmit a live 4K video to Earth for viewers to watch. Assuming 8-bit color video at a resolution of 4096x2160 and 30 frames per second, the live video will require about 795 megabytes per second of data uplink. Laser communications systems have been proven to support up to 622 megabytes per second, with a maximum theoretical rate of 1 gigabyte per second. With this data rate, two simultaneous channels of optical communication would reach 1200 megabytes per second or more. Thus, the EEV and DST will be equipped with dual-channel laser communications systems (replicated twice for the DST), operating in parallel.

3.7 Crew Systems

3.7.1 Internal Layout

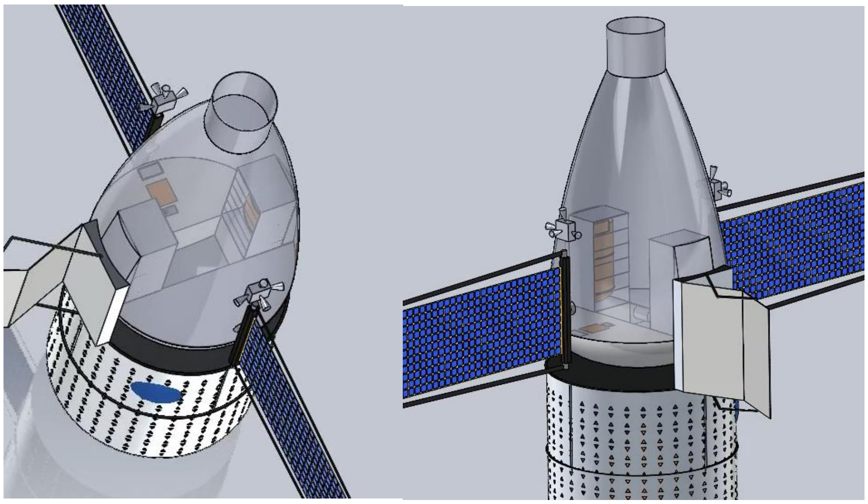

Figure 3.7.1 Internal Crew Systems Layout as seen from Exterior

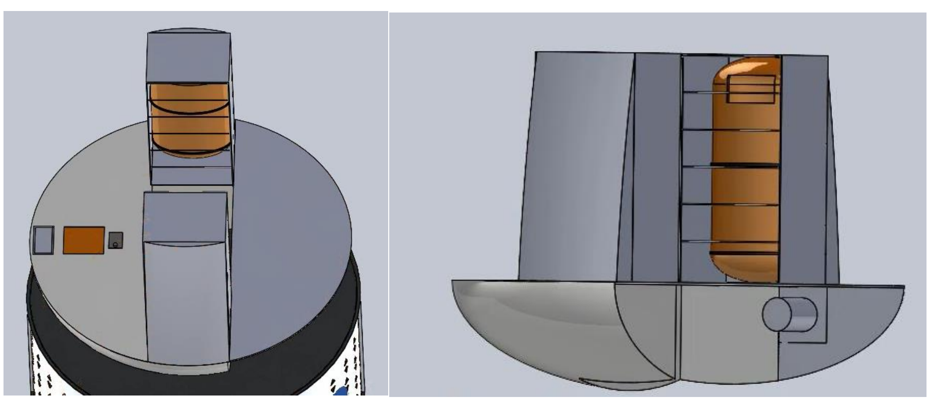

Figure 3.7.2 Internal Crew Quarters Layout

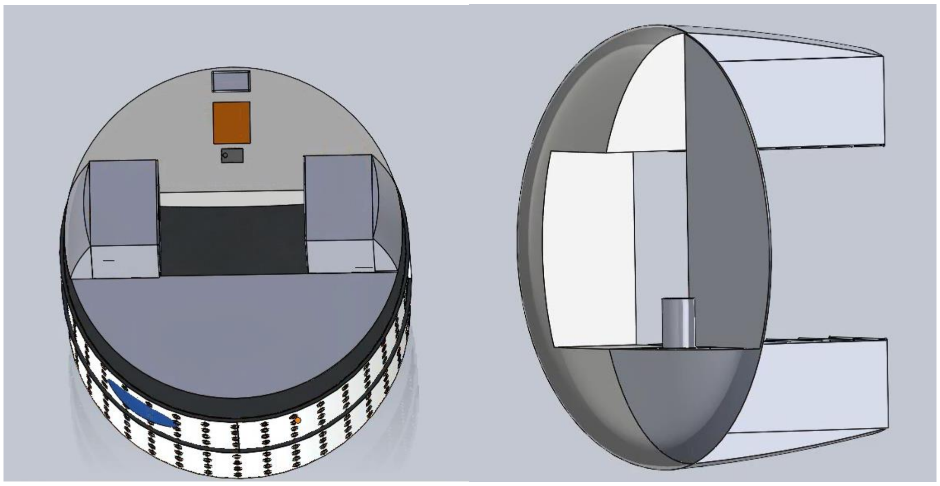

Figure 3.7.3 Top (left) and Bottom (right) Views of Crew Quarters

The EEV is split into two sections: the mission section and the crew section. The mission section contains everything crucial to the operation of the EEV and the completion of the objectives. This includes flight controllers, scientific instruments, the sample collection mechanisms, on-board computers, and communications systems. In the crew quarters is everything needed to sustain the crew and maintain their comfort over 30 days. Here, the food and water will be stored, as well as the systems to prepare meals, sleeping quarters, waste disposal, storage, and the toilet. Separating the cabin in this way will allow the astronauts to have a physical break between their working time and their leisure time.

The vertical features are the sleeping quarters, which measure 2 meters in height to account for nearly any sized astronaut. These are made of polyethylene to aid in radiation shielding, which will protect the crew while sleeping and act as an emergency “bunker” in the event of a solar flare or other increased radiation event. The flat surface next to the sleeping quarters contains the galley and crew storage. Inserted into the face are a food warmer, water dispenser, and small refrigerator (as seen from top to bottom on the right side of Figure 3.6.3). These will provide everything necessary for the astronauts to prepare meals and receive drinking water. The water will be stored in a bladder behind these fixtures. Food will be stored in NASA standard storage lockers, which line the rear facing of the interior. Additionally, personal effects, supplies, and anything else the crew might need will also be stored here. The right side of Figure 3.6.3 shows the underside of the crew quarters, which features the toilet. This system utilizes the NASA Universal Waste Management System and will be separated from the rest of the crew cabin by a locking door.

3.7.2 Atmosphere

As the EEV evolved, one of the subsystems that underwent the most change was the atmosphere of the cabin. Several decisions went into the overall system which comprised the atmosphere, namely the air concentrations, filtration, and circulation. Temperature control was another important consideration and was designed as a joint venture between the Crew Systems and Power, Propulsion, and Thermal sub teams. As such, this will be discussed in more detail later. The first decision was selecting the composition of the atmosphere inside the EEV. Early in the design process, when the DST design was still a consideration, the idea was to mirror the atmosphere composition of the two craft to aid docking compatibility and remove the need for an airlock to be used when moving between the vehicles. To sustain a years-long deep space mission, the DST would likely have used an Earthlike atmosphere of 21% oxygen and 79% nitrogen, pressurized to 14.7 psi. However, as the design process progressed, focus shifted away from developing the DST, and it was left essentially as a "black box." This meant its exact specifications and features would not be directly known, but for design purposes could be implied or stated as requirements for mission success. As a result, the atmosphere on the DST could no longer be used as justification for the EEV atmosphere, and the team decided to go back to the drawing board to see if other atmosphere selections would be suitable.

For the revamped atmosphere selection, the team thought it would be best to approach it by looking at the minimum requirements for NASA spacecraft environments. From this, the biggest design limit was the oxygen concentration. To reduce fire risk, NASA craft cannot have more than 30 percent oxygen present in the atmosphere [16]. So, this was the starting point for the new atmosphere design. From here, a pressure of 8.7 psi was selected, keeping the craft just above the hypoxic boundary. This minimized the mass of the gasses stored, the mass and volume of the storage tanks, and the energy needed to keep the gasses cooled. Additionally, having the minimum amount of gas would minimize the amount of energy needed to vaporize it upon crew arrival, which further lowers the power requirements of the vehicle. However, external review of this design revealed a major flaw in the logic. Typically, spacecraft only utilize reduced pressure atmospheres when performing extra vehicular activities. The reason for this is to drastically reduce or eliminate the prebreathe time required prior to conducting the EVA. Having almost no prebreathe allows for less downtime before and after spacewalks and greatly simplifies the process. However, one of the foremost requirements for this Martian moon exploration mission is no EVA can occur. Thus, there was no real justification for having an atmosphere wholly unlike one experienced on Earth or the ISS. But the trade study performed on the atmosphere, shown in Table 4, verified utilizing a 14.7 psi environment. This only added around 23 kg of gas mass, 15 kg of tank mass, and 0.03 cubic meters of tank volume. All these additions are minimal in the overall design and allow for full justification of the 21/79% atmosphere.

Table 3.7.1 Atmosphere Selection Comparison

Atmosphere (% O_2)	Pressure (psi)	Pressure (kPa)	Gas Mass (kg)	Total Tank Volume (m^3)	Total Tank Mass (kg)
100	5	34.5	22	0.02	15.4
70	5	34.5	21.2	0.021	14.9
50	7	48.3	28.9	0.03	20.3
30	8.7	60	35	0.04	24.5
26	10.2	70.3	40.8	0.046	28.6
21	14.7	101.3	58.4	0.067	40.9

With the atmosphere selected, the next design feature is the purification method. Each astronaut will produce around 1 kilogram of carbon dioxide every day during the mission, which amounts to 60 total kg over the 30-day duration. This must be removed to allow the astronauts to keep breathing and continue the mission. Several methods of CO_2 removal were examined, primarily lithium hydroxide, molecular sieves, solid water amine disposal, and electrochemical depolarization decomposition (EDC). All of these systems had their strengths, however only one seemed most suitable for use on the EEV. The technical comparison of each method of CO_2 scrubbing can be seen in Table 3.6.2. In comparing the data, two options stuck out as frontrunners. Lithium Hydroxide and EDC seemed the most suitable for a small, short-term vehicle like the EEV. However, the perks of LiOH made it the slight favorite. As a passive system, it requires no power. Carbon dioxide scrubbing with LiOH works through the chemical equation below. For every mole of carbon dioxide produced, 2 moles of lithium hydroxide are needed to break it down.

$$2LiOH + H_2O + CO_2 \Leftrightarrow Li_2CO_3 + 2H_2O$$

Equation 3.7.1: Carbon Dioxide and Lithium Hydroxide Chemical Reaction

This results in a system which requires 1.09 kilograms of LiOH for every kilogram of CO_2 produced. Over a 30-day mission, this equates to around 65.4 kg of LiOH needed for a two-person crew. However, the capsules which store the lithium hydroxide add around 1 kilogram per kilogram of CO_2, bringing the total mass to 125.4 kg. Though it has a higher overall mass, the simplicity of the system outweighs the few dozen extra kilos, which are miniscule compared to, say, the outer shell or the fuel tanks. Additionally, a failure while using LiOH is much easier to mitigate than with EDC. Either a secondary system or backups of every part would need to be used for EDC systems, which could double or triple the mass of the system depending on resiliency requirements. Lithium hydroxide is stored in canisters which sit inside the air circulation system and changed out by hand when they absorb their limit of carbon dioxide. For a two-person crew, this would be approximately every 12-14 hours. A failure in a LiOH system would be a failure of one cannister, which would easily be taken out, replaced, then discarded. Additionally, bringing enough space cannisters to be resilient to multiple failures would only increase the total system mass by a fraction of the overall mass. Lastly, a crucial benefit of LiOH scrubbers is their ability to work in the event of full power failure. Under normal circumstances, air is moved throughout the cabin by a circulation system, which pulls air in, passes it through the scrubbers, and pumps it back into the cabin. If there is no power to circulate the air, the cannisters can be opened and the crew can remove the LiOH inside. By placing the chemicals in breathable curtains and attaching these around the cabin, the crew can then manually move air over the LiOH. This method provides about half the scrubbing efficiency as the powered system but would be enough to keep the crew alive in the event of a catastrophic failure and will be discussed in more detail in a later section [17].

Table 3.7.2 Carbon Dioxide Scrubber Analysis

Method	LiOH Absorption	2-Bed Molecular sieve	4-bed molecular sieve	Solid amine water desorption	Electrochemical Depolarization Concentration
Regenerable	No	Yes	Yes	Yes	Yes
Mass (kg)	125.4	32	60	34	33
Power (W)	0	154	340	300	120
Volume (m^3)	0.086	0.09	0.11	0.07	0.04
Concerns	Exothermic Reaction	High heat requirement (>350 C)	High heat requirement (>350 C)	Corrosive, Chemically Unstable	Combustible

For the safety of the crew, carbon dioxide is not the only thing that needs to be removed from the atmosphere. Other harmful particles, known as trace contaminants, are present in the atmosphere and can arise from a variety of sources. However these particles make their way into the atmosphere, they must be removed before they have the potential to cause harm to the astronauts. As with carbon dioxide scrubbing, there are a number of methods for removing trace contaminates. The most advanced system is likely the one deployed on the ISS. As there are hundreds of cubic meters of air to purify, the system is massive, taking up several square meters of space and utilizing hundreds of kilowatts of power [18]. A system such as this would be impossible to use on the EEV so, just like the CO_2, scrubbing method, the simplest options were examined. Activated charcoal screens will be added into the air filtration system to capture and remove the contaminates from the atmosphere. As with the LiOH cannisters, this is a passive system, and the screens only need to be changed periodically. These charcoal filters work by creating an affinity for contaminates, which are usually metallic or heavy inorganic compounds, which causes them to bond to the filter and be permanently removed from the atmosphere.

The last facet of the atmosphere design for the EEV is the air circulation system. This will allow the air to move throughout the cabin and through the various purifiers to maintain the flow of clean, breathable air. OSHA standards in the United States limit workplace carbon dioxide exposure to a maximum of 5,000 ppm, or 0.5%, over an 8-hour working period [19]. Each astronaut aboard the EEV will produce around 1 kilogram of carbon dioxide each day, or an average of 42 grams per hour. The total amount of gas in the atmosphere will be around 58.4 kilograms, as shown in Table 3.6.1. To maintain levels less than 0.5% CO_2, there must be no more than 292 grams of carbon dioxide present in the atmosphere at any time. With a two-person crew, 242 grams of carbon will be produced in just under 3.5 hours. This means the atmosphere must be fully recirculated and purified at most once every 3 hours and 25 minutes. This will be done using vents in the fore and aft of the craft, which will create a mild flow of air throughout the craft and keep the air clean.

3.7.3 Radiation

Protecting the astronauts from radiation while in Martian orbit and on the moons is one of the most crucial functions of the EEV. In fact, doing so fulfills one of the primary mission requirements of bringing 2 crew to the Martian moons and returning them safely to Earth. As such, careful planning went into selecting materials, designing shelters, and developing radiation-related procedures. First, different methods of mitigating radiation dosage for the crew were examined. This involves creating a physical barrier containing a radiation-blocking material separating the crew quarters from the exterior of the spacecraft. These materials increase in efficacy as they increase in thickness, as shown in Figure 3.7.4. Three materials were examined for use on the EEV, which were aluminum, water, and polyethylene. Aluminum is highly standardized in the industry, being used in nearly every spacecraft as structural material. Conveniently, aluminum also works well to block radiation in small doses, such as those seen by spacecraft within the Earth's sphere of influence. However, as a spacecraft heads into deep space, a much thicker barrier is needed to maintain safe levels of radiation for the crew. Aluminum has an average density of 2.7 grams per cubic centimeter, which greatly contributes to the weight of the craft. Thus, it works well at blocking radiation in a deep space or Martian environment, but at the cost of increased structural mass. The second material, polyethylene, has similar efficacy to aluminum, but at a fraction of the mass. Additionally, it is currently in use by NASA on the ISS in crew quarters to provide extra protection to the astronauts while in their bunks. However, it has not yet been deployed as the main shielding device for an entire spacecraft. This likely could change soon. Not only is it 2.4 times lighter than aluminum, but polyethylene is also 50 percent better at blocking solar flares and 15 percent more effective against cosmic radiation than aluminum [20]. Figure 12 compares the shielding efficiency of polyethylene and aluminum as they vary in thickness. This means even less polyethylene is required to achieve the same level of radiation defense, which drastically reduces the weight. For all these reasons, polyethylene was an early frontrunner for radiation control on the EEV. Lastly, water was considered. Also a material commonly used for radiation control, many craft designs employ a so-called water wall around sleeping quarters to increase protection while astronauts sleep. Though it is highly effective for this purpose, it is rather impractical to scale to a full-sized craft. The mass would be excessively large, and lots of power would be required to keep the water from freezing around the hull. Additionally, water surrounding the hull would need to be kept separate from wiring and other systems stored externally and could cause problems if contamination occurs.

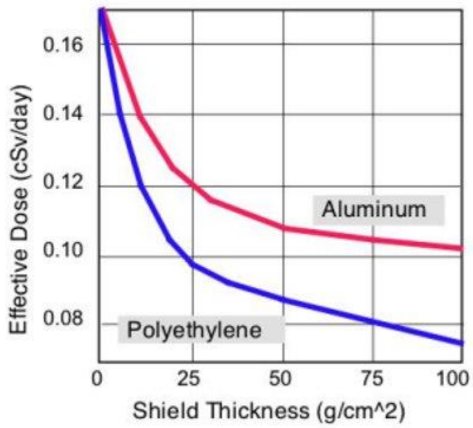

Figure 3.7.4 Polyethylene and Aluminum Shielding Efficacy [21]

In the initial design of the spacecraft, a pressurized aluminum hull was surrounded by a layer of polyethylene approximately 5 $\frac{g}{cm^2}$ thick. This would reduce the absorbed radiation dose to 1.4 mSv/day, or just under 45 mSv for the duration of the mission. Over the entire surface of the EEV, this would add a shell with a mass of 6,420 kg. For comparison, aluminum shielding with the same dosage would require a thickness of 12 $\frac{g}{cm^2}$

and have a mass of 15,400 kg. Clearly, polyethylene was selected for radiation shielding. However, as the design progressed, a major issue with the spacecraft dry mass arose. The team realized that the mass would require the spacecraft to use more fuel than could be fit into the limited space of the EEV. As such, it became crucial to shed mass wherever possible. The biggest target for mass elimination quickly became the more than 6400 kilograms surrounding the craft. When reexamining radiation, the team realized that having a full exterior shield could be excessive. For only 30 days, the total amount of radiation absorbed by each astronaut will be miniscule compared to the dosage received on the flight to and from Mars. Typically, NASA astronauts are limited to absorbing 1 Sievert of radiation over the course of their careers. A mission to Mars and back would likely max out this limit, and thus the maximum radiation the crew can receive on the Moons of Mars Mission is 1 Sievert. As stated above, the bulk of this will occur while the crew are onboard the DST, which limits the amount of extra radiation the two members on the EEV mission can endure.

As a NASA-designed deep space craft, the DST would likely have radiation shielding of around 20 g/cm^2. Regardless of the material of this shielding (aluminum or polyethylene), the dose inside the craft would be anywhere from 1.1-1.3 mSv/day, or anywhere from 600-700 mSv total dosage. This leaves a high margin of safety in the event of unexpected radiation events or solar storms and allows for much less efficient shielding on the EEV. Though the polyethylene shielding was removed, the redesigned craft gained an outer aluminum shell, which is used to protect external instruments and other systems stored outside of the pressurized vehicle. This shell, combined with the pressure hull, is 1.35 cm thick, and provides an average shielding thickness of around 3.6 g/cm^2. This will expose the astronauts to a radiation dose of around 1.7-1.9 mSv/day, which will only add 50-60 mSv total to their mission dosage. In the event of a solar flare, solar particle event, or other radiation situation, the crew will have added protection in the form of radiation shelters. The sleeping quarters will be made of polyethylene and will serve as a safe space for the astronauts if a radiation event takes place. Though these shelters will only be a few grams per square centimeter thick, it will be enough to keep them safe and healthy during an event. Additionally, the bunks will be large enough to contain emergence food and water reserves, as well as media hookups and lighting in case an extended shelter is required. Since these shelters double as the sleeping quarters, the overall radiation dosage will be limited, since the crew members will be spending at least 6-8 hours sleeping each day.

3.7.4 Crew Life

As the mission is only 30 days long, the crew will be extremely busy keeping the mission on track and collecting scientific data. For consistency, the crew will be following the standard NASA mission workday, which goes from 6:00 GMT to 21:30 GMT. This is a very long day, but the astronauts will not be “working” the entire time. All meals are included in that time, as well as activities such as cleaning, meal preparation, and designated relaxation time. Typically, astronauts also receive around 1.5 workdays off out of every 7. For a mission as brief as the Moons of Mars Mission, this would likely be reduced to 1 out of every 7 days and would be staggered between the crew so there is no idle time on the spacecraft.

For the working periods during the day, crew members will be conducting scientific observations or having update meetings with the DST crew or mission control. This will mainly involve one crew member collecting samples and storing them, while the other deploys instruments such as the seismometer or radiometer. Completing these objectives simultaneously prevents downtime and ensures the mission time is being maximized

For the mission goals, however, non-working periods will be just as important. A robotic craft can collect soil samples and gather scientific data, as demonstrated beautifully by the Perseverance and Curiosity rovers. Only a human can provide a detailed description of life in deep space and give a firsthand account of the Martian moon environment. This will be vital in planning future missions, as this mission is the first time humans will spend an extended period in deep space and on a foreign body. Eventually astronauts and even civilians will be sent to Mars proper and having an understanding of what living in deep space does to the mind and body will be crucial. Routine mental and physical checkups will be crucial, both for data collection and crew wellbeing.

Another important aspect of the crew member's lives will be mealtime. Unlike typical missions such as the ISS or Space Shuttle, meals on the EEV will be eaten at the same time. This will allow for social interaction time outside of the work schedule and help provide a sense of normalcy to the mission. Additionally, in an effort to stave off boredom, a variety of food options (as much as can be provided in dehydrated capsules) will be on the menu. This will also aid in reducing meal prep time, as all meals will be prepared simultaneously rather than each astronaut preparing then eating their meals separately. To prepare food, an ISS standard food warmer will be used to heat rehydrated food packets. This machine is capable of simultaneously heating 6 rehydrated packets in as little as 35 minutes [22] [22]. Water will be dispensed using the ISS Potable Water Dispenser, which is currently being developed to meet expanding crew needs on the ISS. This system is capable of dispensing both hot and cold water, making it easier for the preparation of food and beverages. The refrigerator will be used to chill food and drinks but will likely be rarely used and is merely there for astronaut comfort in the event they want to use it.

Lastly, maintaining physical fitness while in space is crucial. While on the ISS or other NASA missions, astronauts typically spend anywhere from 2.5-3 hours exercising each day. This prevents muscle and bone atrophy while in a zero-gravity environment and keeps the astronauts in shape for high-exertion activities such as spacewalks. Though no EVA will occur (barring total spacecraft failure), maintaining this level of fitness will still be important. To provide for all the astronaut's needs, the newly developed ROCKY system will be deployed. This system has been designed for deployment on Orion missions and is intended to handle full body workouts. Ideally for the EEV, this system only weighs 20 pounds and can store into one cubic foot [23]. This small size means one astronaut can be exercising while another is working or sleeping, and it will not take up the limited space of the interior.

Table 3.7.3 Astronaut Daily Schedule

Time	Activity
06:00	Morning Check-in with DST/Mission Control
06:30	Exercise Period 1

07:20	Meal 1 Prep
07:30	Astronauts eat Meal 1 together
08:00	Astronaut 1 starts sample collection; Astronaut 2 cleans, inspects EEV for maintenance
10:00	Astronaut 2 finishes tasks, begins deploying scientific instruments. Astronaut 1 helps once sample collection finished
12:00	Exercise period 2
12:50	Meal 2 Prep
13:00	Astronauts eat meal 2 together
13:30	Astronauts clean up meal 2, return to experiments and ensure sample storage
18:00	Exercise period 3
18:50	Meal 3 prep
19:00	Astronauts eat meal 3
19:30	Meal 3 cleanup, astronauts store scientific instruments and survey EEV
20:30	End-of-day debrief with DST crew/mission control
21:30	End of debrief end of official workday
Post 21:30	Leisure time

3.7.5 Safety and Contingency Planning

While the crew is on board the EEV, steps must be taken and practices must be in place to always ensure the safety and wellbeing of the crew. This includes things such as medical procedures, fire prevention, and evacuation plans. For medical consideration, the crew will be using the full ISS medical kit, scaled down to 2 crew members and 30 days. This kit is extensive and covers everything the astronauts could run into medically while on board the craft. Mental health will be just as important as physical health for the crew. Regardless of how fit they maintain themselves or how prepared they are medically, mental health issues could happen at any time, especially in deep space. The astronauts will be isolated millions of miles from Earth, alone in a 60 cubic meter vessel for a month. This could lead to serious issues with the wellbeing of the crew members, and steps must be taken to ensure that nothing affects the mission or the safety of the astronauts. Routine check-ins with the members of the DST crew will help stave off isolation in the EEV, and fully enclosed bunks will allow the astronauts to have private space where they can decompress. Additionally, the downtime built into the daily and mission-long schedules will keep the crew from being too overworked. With these precautions in place, the crew should be fine mentally and physically during the mission, but if something serious happens, the DST will be sent to retrieve affected crew members and salvage the mission if possible. The biggest threat facing the crew members inside the EEV is a fire. In space, even a small fire can spell disaster for a mission if not caught in time or handled properly. As such, protocols will be in place for different types of fires and how to handle them. In the event of any fire, oxygen masks will be put on by the crew until the air is properly purified. Small fires, the least damaging group, are easy to put out by hand and cause minimal to no harm to the spacecraft or anything contained within. These fires would take less than 10 minutes to extinguish and release a minimal amount of contamination into the atmosphere. Use of a handheld extinguisher will easily put out small fires. A moderate fire is somewhat increased in intensity. A fire of this scale would cause minor damage to isolated areas of the craft but leave all major systems intact and have no severe impact on crew safety or mission success. These fires would take anywhere from 10-20 minutes to extinguish and would reduce the quality of the internal air for a brief period. However, the craft's major systems would still be intact, and the mission could continue as planned, albeit with minor adjustments. Continued use of handheld extinguishers could suffice to stop the fire, but partial evacuation of the atmosphere may also be needed. Last are severe fires. Fires in this category would involve multiple systems or a majority of the EEV and cause severe structural damage. Additionally, these fires would take over 20 minutes to extinguish and would severely hamper the ability of the crew to safely continue the mission. In the event of a severe fire, the crew would don emergency space suits and call for retrieval by the DST. If the craft is safe to remain in following the fire, the astronauts would shelter in place, otherwise they would be forced to wait for evacuation outside the craft. At this severity, the fire would only be extinguished by a full evacuation of the atmosphere and would leave everything except the most crucial life support systems inoperable.

In the event of a hull breech or failure of major life support systems (air circulation, food and water systems, EEV power) the crew would need to put on emergency space suits, alert the DST of the situation, and wait for retrieval. As long as backup power systems are functional, the crew will be able to survive for an extended period of time. The Orion emergency suits will be utilized on the EEV and can provide up to 6 days' worth of life support while connected to vehicle power [24]. This will give the DST crew plenty of time to perform their rendezvous and successfully rescue the crew. In the unlikely event of total power loss, the astronauts would only have around 24-36 hours of life support in the suits, making recovery much more difficult.

The biggest threat stemming from a total power loss would be the lack of air circulation. Without circulation, the air cannot be moved through the purification system, which would lead to an unsafe buildup of carbon dioxide in under four hours. However, the use of lithium hydroxide mitigates this risk. Due to the nature of the chemical reaction, air merely needs to interact with the LiOH, as discussed in the atmosphere section. This means that in the event of a power failure, the LiOH cannisters can be opened and exposed to the atmosphere, which would allow for continued functionality. But in space, loose chemicals cannot be moving about the cabin, which is what would happen if a LiOH cannister was opened in a zero-gravity environment. To rectify this, the team looked to the biggest user of lithium hydroxide scrubbing: the US Navy. Many US submarines use LiOH cannisters as either a primary or secondary method of CO2 removal while on board. As such, they have contingencies in place in the event of a total loss of power to the submarine. Battelle curtains are the main technology used here. They work by placing an open LiOH cannister in a hollow curtain, which fills with particles and allows for interaction with the atmosphere [17].

Figure 3.7.5 Lithium Hydroxide cannisters attached to Battelle Curtains [17]

The curtains are then hung throughout the interior of the craft, which then passively work for hours to reduce CO2 levels. In a study conducted by the Naval Submarine Medical Research Laboratory, lithium hydroxide-filled Battelle curtains were found to have a scrubbing capacity of 0.756 kg CO2 per kg LiOH [17]. Considering the chemical reaction limit is 0.919, this is highly effective. The below graph shows CO2 levels present after multiple Battelle curtain hangings over an 18-hour period.

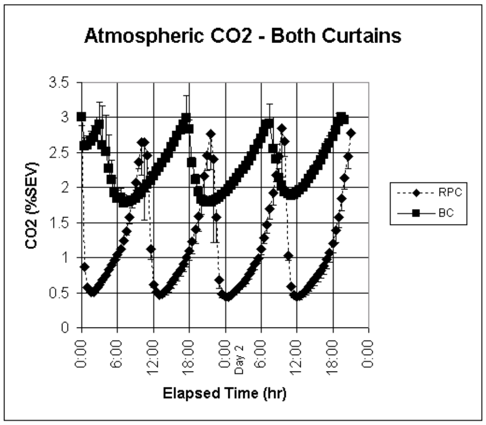

Figure 3.7.6 Battelle Curtain (top) Carbon Dioxide Levels Over Time [17]

This data was conducted assuming scrubbing started at 3% CO2, which is the Naval safe limit. Additionally, CO2 production was assumed to come from a standard nine-man crew, which is far greater than the conditions present on the EEV.

Something that indirectly impacts crew safety, while still being crucial, is waste management. Without proper waste storage and disposal mechanisms, unhealthy and unsafe conditions can emerge and cause harm to the crew. While on board the EEV, there will be three major types of waste: human waste, solid waste, and liquid waste. Now, human waste contains both solid and liquid waste, but since they will be handled similarly and pose a unique biological hazard to the crew, will be discussed as one type of waste. Human waste involves then things which leave the human body, primarily urine and feces. To safely remove these wastes, the EEV will be utilizing NASA's Universal Waste Management System. This is a new toilet system being designed for use on Orion missions and beyond, making it a perfect fit for the Moons of Mars Mission. One of the best features this system has is the ability to automatically separate solid and liquid waste, treat them separately, and dispose of them accordingly. Since solid and liquid human waste often go hand in hand, having a system that can simultaneously deal with each is crucial. Once separated, the UWMS introduces chemicals to prevent bacterial buildup prior to disposal or storage. After urine is collected and treated, it will be vented out of the spacecraft, rather than take up mass and volume on the spacecraft and pose a health threat [25]. Solid feces will also be chemically treated but cannot be vented in the same way as urine. Thus, it will be collected and stored in compartments outside of the pressure hull to prevent the possibility of contamination with any crew systems.

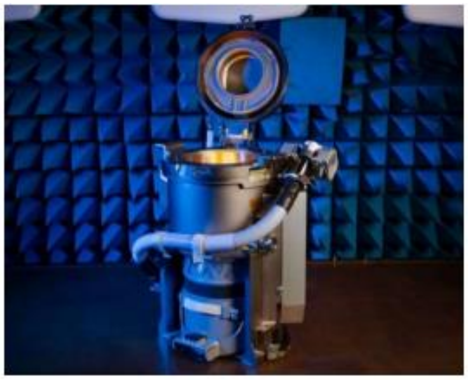

Figure 3.7.7: NASA Universal Waste Management System [26]

4 Budget and Cost Estimation

4.1 Budget

The mission's budget has been predefined by the requirements set forth in the AIAA RFP. According to the RFP, the vehicle designed cannot exceed $1 Billion US Dollar based on FY21, including launch cost.

4.1.1 Preliminary Top-Down Cost Estimation

At the beginning of the project, a budget for each sub-team was allocated using top down cost estimation. This was so that each sub-team could be informed of their expected budget and for the team overall to understand where major costs came from. Table 3.3.1 shows the budget for each sub-team and percentage of the total budget consumed. These values were assigned based on values from previous NASA planetary missions, keeping long term space travel in mind. As our design was finalized, parametric cost modeling was done using NASA's PCEC tool to calculate more accurate values.

Table 4.1.1 Preliminary Cost Estimation

Sub-team	Budget [$M]	Total Budget (%)
Avionics	100	10
Loads, Structures, Mechanisms	250	25
Power, Propulsion, Thermal	250	25
Mission Planning and Analysis	200	20
Crew Systems	100	10
Systems Integration	100	10
Total	1000	100

In this table, the budget accounts for all the design, development, test, evaluation, production, and operation costs for each sub-team. Avionics includes all sensors, reaction control systems, communications, and flight software associated with the mission. Crew Systems includes all components inside the EEV that are required to keep the astronauts inside alive. Loads, Structures, and Mechanisms includes all exterior parts of the EEV, structure, fairings, and other vehicle mechanisms. Power, Propulsion, and Thermal includes engines, fuel, oxidizer, batteries, solar panels, and thermal protection. Next, Mission Planning includes all scientific instruments required to complete the scientific objectives as well as cost associated with launch. Lastly, Systems Integration includes all lifetime support operations and crew mission operations costs.

4.1.2 Parametric Cost Modeling

Parametric cost modeling allows for a more accurate understanding of actual costs tied to components chosen for the mission. To estimate the cost of this mission, NASA's PCEC was used in the form of a Microsoft Excel add-on. The cost breakdown was spread across the mission's WBS, including the spacecraft and its subsystems, project management, systems integration and assembly, launch vehicle/services, and operations. To generate the full cost breakdown, PCEC uses NASA's library of CERs derived from historical mission data as well as user inputs such as mission architecture, subsystem mass, and heritage ratings. The mass inputs used were directly taken from the mass budget in Table __. PCEC also adjusts values for inflation, as the software initially makes calculations in FY2015 but can convert to any desired FY chosen. PCEC is very reliable because it is updated yearly, heavily detailed, and takes NASA historical data for CERs.

4.1.3 Cost Analysis

As shown in Table 4.1.2 below, the total cost of the Moons of Mars Mission in FY21 is $818.1 million and $900 million with a 10% margin. Both values are below $1 billion, indicating that the mission is within budget. Figure 4.1.1 shows the cost breakdown in percentages for the Level 2 WBS elements. The most expensive elements are spacecraft structures, launch vehicle/services, and operations, which is expected. Performing a cost analysis is critical to our mission since the RFP requires that the total cost cannot exceed $1 billion (FY21). This is low compared to other NASA Mars missions in the past, such as Perseverance and Curiosity. Therefore, having a cost analysis quantifies the cost of individual elements and shows how cost is distributed. If the mission goes over budget, a cost analysis can also clearly show where changes can be made to stay within budget. The total cost of the mission is close to $1 billion, so it's possible that delays or errors could cause the cost to increase. To cut down on cost, shortening the mission time span could be an option or reducing the number of scientific instruments and choosing cheaper ones. Overall, the biggest factor is the spacecraft mass. Minimizing mass as much as possible would also greatly reduce cost.

Table 4.1.2 NASA PCEC Work Breakdown Structure

WBS #	Level	WBS Element	Cost (FY21 $M)	Marginal Cost (10%)
0.0	1	System Name-Moons of Mars	-----	-----
1.0	2	Project Management	0.9	1.04
2.0	2	Systems Engineering	12.0	13.2
3.0	2	Safety and Mission Assurance	2.5	2.75
4.0	2	Science/Technology	15.3	16.8
5.0	2	Flight System/Spacecraft	460.9	507
5.01	3	Crewed Vehicle Management	16.5	18.15
5.02	3	Crewed Vehicle Systems Engineering	17.6	19.36
5.03	3	Crewed Vehicle Product Assurance	11.8	12.98
5.1	3	Crewed Vehicle	415	456.5
6.0	2	Mission Operations System (MOS)	150	165
7.0	2	Launch Vehicle/Services	160	176
9.0	2	System Integration, Assembly, Test & Check Out	16.5	18.15
		Total	**818.1**	**900**

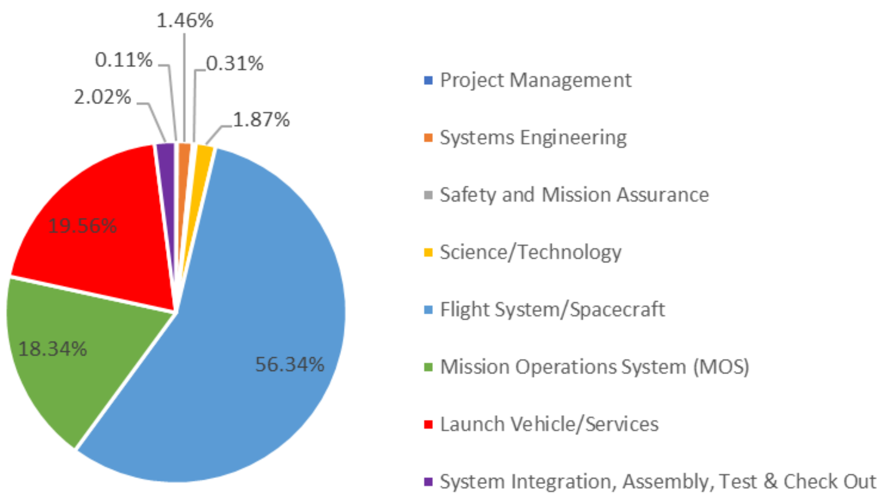

Figure 4.1.1 Moons of Mars Mission Total Cost Percentage

5 Risk Analysis

Every system of the vehicle entails potential failures. Various consequences can result from such failures, including crew injury, failure to achieve scientific objectives, and further system damage. The purpose of risk management is to anticipate these failures, what consequences they entail, how to prevent them, and how to lessen the consequences or remedy problems.

5.1 Overall Risk Analysis

In order to assess the relative importance of the various failures and understand their severity before and after mitigation measures are taken, a risk matrix was created. Each risk is rated on a scale of 1-5 for severity and likelihood at both the mission and crew level. The risks are also color coded by the sum of the severity and likelihood score. 2-3 denote green (lowest), 4-5 denote yellow, 6-7 denote orange, 8-9 denote red, and 10 denotes a deep red (highest). For severity, 1 corresponds to insignificant, 2 corresponds to low, 3 corresponds to moderate, 4 corresponds to high, and 5 corresponds to catastrophic. For likelihood, 1 corresponds to rare, 2 corresponds to unlikely, 3 corresponds to possible, 4 corresponds to likely, and 5 corresponds to very likely. Additionally, consequences for each failure were listed as well. Unmitigated risk ratings were determined first, then new risk ratings were assigned based on mitigations and contingencies. For better organization, the risks are categorized by each vehicle subsystem. Many risks come from crew systems due to ensuring the safety of the crew. Other risks come from spacecraft structure, propulsion, and power systems. Overall, every risk has lower mitigated risk ratings compared to unmitigated risk ratings, showing the efficiency of mitigation.

Table 5.1.1 Risk Matrix with Legend

		Severity				
Likelihood		Insignificant (1)	Low (2)	Moderate (3)	High (4)	Catastrophic (5)
	Very likely (5)	6	7	8	9	10
	Likely (4)	5	6	7	8	9
	Possible (3)	4	5	6	7	8
	Unlikely (2)	3	4	5	6	7
	Rare (1)	2	3	4	5	6

Legend
2
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10

5.2 Avionics

Table 5.2.1 Avionics Failures, Consequences, Mitigations, and Contingencies

Failure	Contingency	Mitigation	Consequences
Sensors	Restart and/or switch to working sensors	Redundant sensors	Spacecraft experiences loss of situational awareness and data
Reaction Control System	Restart and/or switch task to a working computer	Redundant flight computers	Spacecraft unable to move, adjust attitude and complete orbital transfer maneuvers
Communications	Restart software	Redundancy and ground testing	Spacecraft unable to transmit data and communicate with crew, spacecraft unable to move

Table 5.2.2 Avionics Mission and Crew Unmitigated and Mitigated Risks

Failure	Mission Unmitigated Risk (Likelihood, Severity)	Mission Mitigated Risk (Likelihood, Severity)	Crew Unmitigated Risk (Likelihood, Severity)	Crew Mitigated Risk (Likelihood, Severity)
Sensors	3,3	2,2	3,2	2,2
Reaction Control System	3,5	2,3	3,2	2,2
Communications	3,3	2,2	3,2	2,2

5.3 Loads, Structures, and Mechanisms

Table 5.3.1 Loads, Structures, and Mechanisms Failures, Consequences, Mitigations, and Contingencies

Failure	Contingency	Mitigation	Consequences
Sample Collection	Abandon current collection task and reset component	Ground testing	Failure of scientific objectives
Docking	Maintain some set distance out from station, restart docking routines and calibrate to current position	Implement alignment system and ground testing	Unable to transfer crew/supplies, docking accident can cause permanent vehicle damage
Scientific Instruments	Abandon current data collection task and restart	Ground testing and monitor power to scientific instruments	Failure of scientific objectives

Table 5.3.2 Loads, Structures, and Mechanisms Mission and Crew Unmitigated and Mitigated Risks

Failure	Mission Unmitigated Risk (Likelihood, Severity)	Mission Mitigated Risk (Likelihood, Severity)	Crew Unmitigated Risk (Likelihood, Severity)	Crew Mitigated Risk (Likelihood, Severity)
Sample Collection	3,2	1,2	3,1	2,1
Docking	3,5	2,3	3, 5	2,3
Scientific Instruments	3,2	2,2	3, 1	2,1

5.4 Crew Systems

Table 5.4.1 Crew Systems Failures, Consequences, Mitigations, and Contingencies

Failure	Contingency	Mitigation	Consequences
Lack of Oxygen	Wear pure oxygen respirators and masks	Redundant valves for environmental systems, vents for oxygen supply	Crew unable to breathe, possibility of death
Fire	Use CO2 extinguishers, turn on fire suppression system, astronauts evacuate EEV and wait for DST pickup	Have astronauts be trained on fire safety and prevention, ground testing of crew systems	Fire can rapidly spread, cause cascade of failures, deplete oxygen, explosion and flames can cause injury
Crew Injuries	Use ISS medical supplies	Have astronauts be trained on injury prevention, have a spotter for risky procedures	Less crew capacity to deal with problems, physical discomfort
Radiation	Use ISS medical supplies and/or evacuate EEV	Have astronauts be trained on radiation safety and prevention, have Geiger counters onboard, ground testing of crew systems	Low exposure can cause long term health problems, high exposure can cause immediate injury or death
Contamination	Sensors detect internal atmospheric composition and tell which systems are leaking based on composition, astronauts evacuate EEV	Have astronauts be trained on contamination safety and prevention, ground testing of crew systems	Immediate injury or death

Table 5.4.2 Crew Systems Mission and Crew Unmitigated and Mitigated Risks

Failure	Mission Unmitigated Risk (Likelihood, Severity)	Mission Mitigated Risk (Likelihood, Severity)	Crew Unmitigated Risk (Likelihood, Severity)	Crew Mitigated Risk (Likelihood, Severity)
Lack of Oxygen	2,5	2,2	3,5	1,2
Fire	2,5	2,3	3,5	1,2
Crew Injuries	3,5	2,2	3,5	2,2
Radiation	2,5	2,2	3,5	2,2
Contamination	2,5	2,2	3,5	2,2

5.5 Power, Propulsion, and Thermal

Table 5.5.1 Power, Propulsion, and Thermal Failures, Consequences, Mitigations, and Contingencies

Failure	Contingency	Mitigation	Consequences
Engine	Restart engine (computer reports conditions in engine prior to failure, advises restart procedures)	Redundancy and ground testing, computer prevents unsafe operation (temperature and pressure monitoring)	Spacecraft unable to move, adjust attitude, and complete orbital transfer maneuvers
Batteries	Reroute batteries	Redundant and parallel batteries, reports when batteries fail, design system so failures aren't correlated	No power
Thrusters	Restart thrusters, implement alternative control scheme	Redundancy and ground testing, reports when thrusters fail, design system so failures aren't correlated	Spacecraft unable to move, adjust attitude, and complete orbital transfer maneuvers
Solar Panels	Temporarily shut down non-essential systems so panels can collect power	Redundant solar panels, reduce power consumption	No power

Table 5.5.2 Power, Propulsion, and Thermal Mission and Crew Unmitigated and Mitigated Risks

Failure	Mission Unmitigated Risk (Likelihood, Severity)	Mission Mitigated Risk (Likelihood, Severity)	Crew Unmitigated Risk (Likelihood, Severity)	Crew Mitigated Risk (Likelihood, Severity)
Engine	3,5	2,3	2,2	1,1
Batteries	3,4	2,2	3,2	2,1
Thrusters	3,4	3,2	3,2	2,1
Solar Panels	2,3	1,2	2,3	1,2

6 Mass Breakdown

The top level mass breakdown for each EEV subsystem is shown below in Table 4.3. The current mass margin is 6.90%, which was calculated by taking the difference between the EEV initial estimated mass and final mass and dividing that value by the final mass.

Subsystem	Wet Mass (kg)	Dry Mass (kg)
Avionics	**641**	**641**
Thrusters	15	15
Docking	526	526
Computer Systems	20	20
Communications	80	80
Crew Systems	**1009**	**1009**
Bunks + Shielding	225	225
Food + Water	300	300
Hygiene Water	25	25
Food Preparation	68	68
LiOH Canisters	100	100
Charcoal Filters	50	50
Spacesuits	100	100
Miscellaneous	13	13
Atmosphere	70	70
Atmosphere Tanks	49	49
Exercise Equipment	9	9
Load, Structures, Mechanisms	**5689**	**5689**
Structure	5415	5415
Scientific Instruments	117	117
Payload	157	157
Propulsion	**10511**	**502**
Oxidizer (LOX)	7738	0
Fuel (LCH4)	2271	0

Oxidizer Tank	192	192
Fuel Tank	110	110
Cryocoolers	200	200
Engine	**400**	**400**
Power and Thermal	**460**	**460**
Solar Panels	54	54
Battery	288	288
Radiators	118	118
Initial Estimate	**20000**	
Total	**18710**	**8701**
Margin (%)	**6.90**	

7 Compliance Matrix

REQ #	Requirement	Page #
RFP – 01	The EEV should have the capability to support 2 crew members to visit both Martian moons	64
RFP – 02	The total mission shall not exceed 30 days, including transit time from the Deep Space Transport (DST) vehicle to the destination and back.	17
RFP – 03	The EEV should be able to support sample retrieval from each destination, with a minimum sample retrieval mass of 50 kg from each moon.	28
RFP – 04	The 2 crew member will remain inside the EEV during the mission, with no planned EVA capability	64
RFP – 05	The EEV should have the ability for the 2 crew members to conduct exploration of the moons to produce significant scientific understanding of the moons.	50
RFP – 06	These scientific objectives should advance our knowledge of both moons and improve our capability to explore future destinations across the solar system	15
RFP – 07	Describe scientific experiment equipment that are necessary to achieve these scientific goals	50
RFP – 08	Up to 200kg of science equipment can be delivered to the EEV with the crew on the DST, but they are limited to what the crew can carry into the EEV through the pressurized tunnel	50
RFP – 09	Describe the sample retrieval mechanism and how the samples will be stored during the sortie and how the sample will be transferred to the DST for the return trip to Earth. The sample must be quarantined from the crew until Earth arrival for scientific study	28
RFP – 10	The EEV shall autonomously dock with the DST, and 2 crew will	19

	transfer into the EEV to begin the mission sortie	
RFP – 11	Discuss the mission modes and maneuvers required to complete the roundtrip missions to visit both Martian moons	18
RFP – 12	Discuss the time and operation required at each destination to support the science objective as defined by the team	16
RFP – 13	Assume the Crew arrives in a DST vehicle in a Mars 5-sol parking orbit in the summer of 2040, between May and August, the EEV must already be in 5-sol orbit awaiting for Crew arrival before this date the crew arrive in this mission opportunity	21
RFP – 14	Discuss the launch opportunity for the EEV and the propulsion system required to deliver the EEV to Mar and the interplanetary trajectory for the EEV	36
RFP – 15	Describe the selection of launch vehicle and the selection process that led the team to the decision	22
RFP – 16	The cost for the vehicle shall not exceed $1 Billion US Dollar (in FY21), including the launch cost.	78

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9 Appendix

9.1 Subsystem Requirements

Table 9.1.1 Subsystem Requirements - 1

REQ #	Requirement
S1 – 1	The EEV shall provide food for 2 crew members for the duration of the mission.
S1 – 2	The EEV shall provide potable water for 2 crew members for the duration of the mission.
S1 - 3	The EEV shall provide private sleeping quarters for 2 crew members.
S1 – 4	The EEV shall have exercise equipment for crew members to maintain physical fitness.
S1 – 4	The EEV shall maintain an Earth-like atmosphere for the duration of the mission.
S1 – 5	The EEV shall filter air to prevent the buildup of toxic gasses.
S1 – 6	The EEV shall provide thermal control to maintain moderate Earth-like temperatures.
S1 – 7	The EEV shall provide waste management accommodations.
S1 – 8	The EEV shall provide circadian lighting to simulate earth days.

Table 9.1.2 Subsystem Requirements - 2

REQ #	Requirement
S2 – 1	The EEV shall protect the crew members from harmful radiation.
S2 – 2	The EEV shall provide a sealed, pressurized cabin.
S2 – 3	The EEV shall have insulated surfaces to protect from space temperatures.
S2 – 4	The EEV shall have an airtight hatch and airlock to dock with the DST.
S2 – 5	The EEV shall have adequate windows to protect the crew members from the environment.

Table 9.1.3 Subsystem Requirements - 3

REQ #	Requirement
S3 – 1	The EEV shall have full control over translational and rotational movement.
S3 – 2	The EEV shall have adequate thrusters to make all necessary transfers.
S3 – 3	The EEV shall have sufficient fuel to make all necessary transfers.
S3 – 4	The EEV shall have autonomous transfer burns.
S3 - 5	The EEV shall be compatible with an existing launch vehicle.

Table 9.1.4 Subsystem Requirements - 4

REQ #	Requirement
S4 – 1	The EEV shall have the ability to communicate with the Deep Space Network (DSN).
S4 – 2	The EEV shall have passive RF receivers to listen to communications.
S4 – 3	The EEV shall have transmission antennas.
S4 – 4	The EEV shall be capable of direct voice communication to the DST.

Table 9.1.5 Subsystem Requirements - 5

REQ #	Requirement
S5 – 1	The EEV shall provide enough power to maintain life support systems throughout the duration of the mission.
S5 – 2	The EEV shall have the capability to generate power.
S5 – 3	The EEV shall provide enough power to sustain all systems during peak operation.
S5 – 4	The EEV shall store enough power to maintain life support systems in the event of a catastrophic failure and an immediate return to the DST is necessary.
S5 – 5	The EEV shall have resiliencies for power generation.

Table 9.1.6 Subsystem Requirements - 6

REQ #	Requirement
S6 – 1	The EEV shall have appropriate medical equipment for crew members to treat themselves in the event of bodily injury.
S6 – 2	The EEV shall have abort procedures when failures or the existence of uncontrolled catastrophic hazards prevent continuation of the mission objectives.
S6 – 3	The EEV shall have procedures to suppress a fire should one arise.
S6 – 4	The EEV shall have procedures to handle loss of crew.