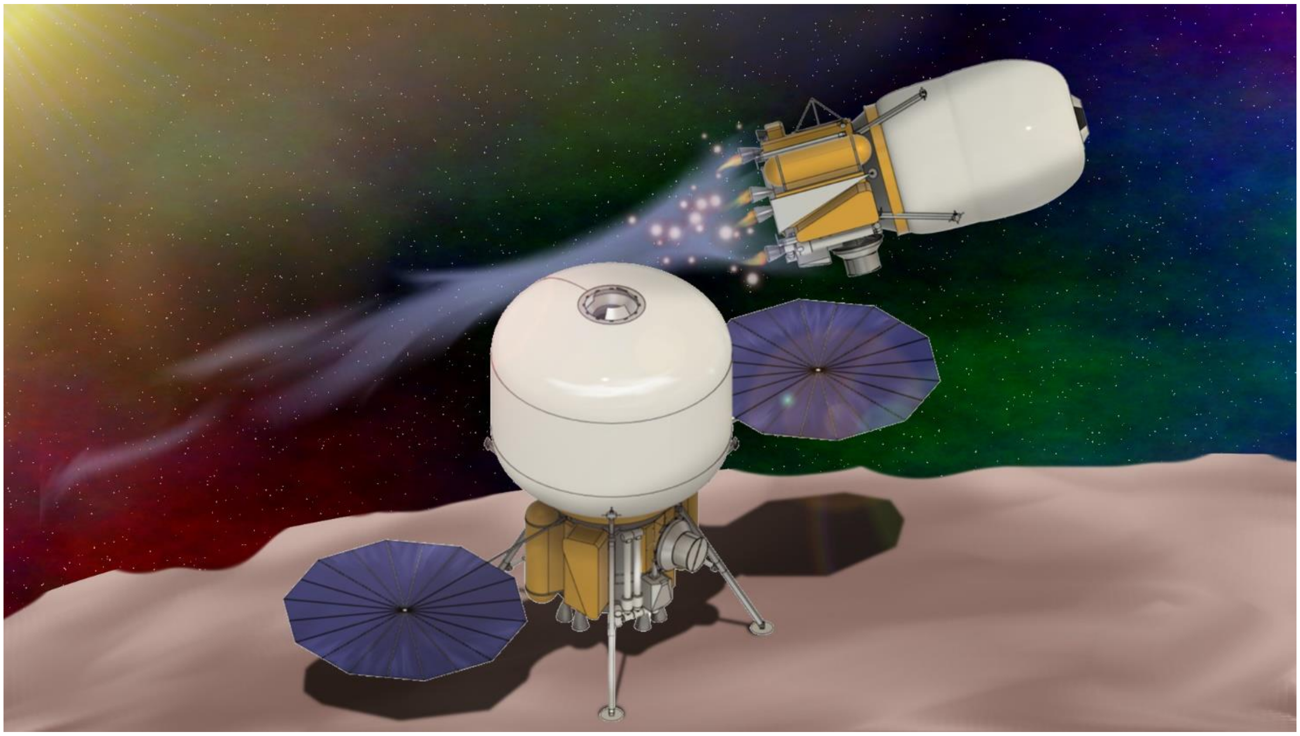

California State Polytechnic University, Pomona Department of Aerospace Engineering

ETS

AIAA

The World's Forum for Aerospace Leadership

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Team Organization Chart

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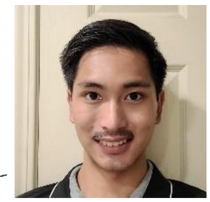
Systems Lead Christopher Morales AIAA ID: 1342731 	Engineering Lead Dominic Victoriano AIAA ID: 1356265 
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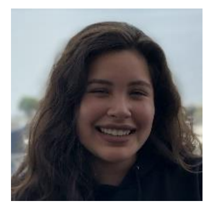
Crew Systems Lead Sameed Mobin AIAA ID: 1356856 	Environmental Control Lead Jessica Hernandez AIAA ID: 1180308 
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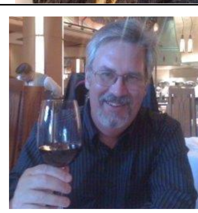
Advisor Steven Nanning AIAA ID: 147851 	

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List of Acronyms

AIAA	American Institute for Aeronautics and Astronautics
BOL	Beginning of Life
BOM	Beginning of Mission
CCSDS	Consultive Committee on Space Data Systems
CME	Coronal Mass Ejections
CO2	Carbon Dioxide
COPV	Composite Overwrapped Pressure Vessel
COTS	Commercial off the Shelf
DSN	Deep Space Network
DST	Deep Space Transport
ECLSS	Environmental Control and Life Support System
EEV	Excursion Exploration Vehicle
EOL	End of Life
EOM	End of Mission
ETS	Extraterrestrial Solutions
FH	Falcon Heavy
GCR	Galactic Cosmic Rays
HGA	High Gain Antenna
IDSS	International Docking System Standard
IPA	Isopropyl Alcohol
Isp	Specific Impulse
ISS	International Space Station
KBPS	Kilobits Per Second
LGA	Low Gain Antenna
LH2	Liquid Hydrogen
Li-Ion	Lithium Ion
LILT	Low Intensity Low Temperature
LiOH	Lithium Hydroxide
LV	Launch Vehicle
MLI	Multi-layer Insulation
MMH	Monomethyl Hydrazine
NASA	National Aeronautics and Space Administration
NTO	Dinitrogen Tetroxide
O2	Oxygen Gas

OOMS	On-Orbit Maneuvering System
OODM	On-Orbit Dry Mass
OOWM	On-Orbit Wet Mass
OPR	Oral Presentation Review
ORE	Outside Review Evaluation
ORG	Organization Structure
PBA	Portable Breathing Apparatus
PFE	Portable Fire Extinguisher
RAD	Radiation Assessment Detector
REID	Radiation Exposure Induced Death
RTG	Radioisotope Thermoelectric Generator
SDR	System Definition Review
SOI	Sphere of Influence
SPE	Solar Particle Events
T/W	Thrust Weight
TOCA	Total Organic Carbon Analyzer
TRL	Technology Readiness Level
TWR	Thrust to Weight Ratio
USCV	Unmanned Sample Collection Vehicle
WBS	Work Breakdown Structure

1.0 Executive Summary

Extraterrestrial Solutions (ETS) aims to enable human exploration of the Solar System, for which the next step is a crewed exploration of Mars. By responding to this request for proposals (RFP), ETS is fulfilling that mission statement by providing the design for a system that will bring humans to the Martian moons of Phobos and Deimos, whereupon samples will be collected from each moon. This will provide the necessary experience from which more complex crewed missions can be conducted in the Martian sphere of influence (SOI) that will eventually allow for a human presence on the surface of the red planet. The moons themselves are also of scientific interest, and samples collected from them will help the scientific community further their research into the origins of the Solar System and the history of Mars.

The system designed by ETS is the Exploration Excursion Vehicle (EEV), which will bring the crew to the moons within the Martian SOI. This will be sent to Mars uncrewed, where it will then wait in a 5-sol parking orbit for the Deep Space Transport (DST) to bring the crew. The DST is not designed or produced by ETS. After docking, transferring the two crew members and supplies from the DST, and undocking, the EEV will begin its mission to explore and collect samples from the Martian moons.

As the EEV is a crewed vehicle, accommodations for the crew are the key design driver of the vehicle. It was found that a pressurized volume of greater than 30 m^3 would be sufficient for a 30-day mission. The EEV is equipped with an environmental control and life support system (ECLSS), which consists of atmospheric control and revitalization, water management, waste management, and fire detection and suppression systems. Food and other crew supplies have also been considered, and radiation shielding vests are provided to the crew to enhance radiation protection in addition to what is inherently provided by the structure. A shelter for the crew is also present in case of radiation events. The ECLSS will have a mass of 763 kg on launch, but 377 kg of food and water will be brought aboard by the crew upon docking with the DST. The entire system will draw about 1750 W of continuous power.

A coring-based sample collection mechanism has been devised for the mission. Like the sample collection drill on the Mars Perseverance rover, a core is extracted by a mechanism and hermetically sealed before being placed in the sample caching system. This contains two separate areas that isolate samples from Phobos and Deimos from each other. 19 sample containers have been allocated for each moon. However, due to the differing average density of the moons, only 11 are estimated to be required from Phobos while 13 will be required from Deimos. After the samples are collected and the EEV returns to the DST, the sample caching system is detached from the EEV and is docked to the DST using an IDSS docking adapter. The crew is then able to access the sealed sample containers from the interior of the DST. This sample collection system is projected to have a mass of 320 kg and draw 200 W of power.

Since there is very little known about the surfaces of Phobos and Deimos, additional payloads are necessary. To determine the viability of specific landing sites, the EEV is equipped with a thermal imaging camera and a thermal emissions spectrometer. In order to save on development costs, these payload instruments will consist of the Thermal Emission Imaging System (THEMIS) used on the 2001 Mars Odyssey orbiter and the OSIRIS-REx Thermal Emission Spectrometer (OTES). THEMIS will draw 14 W of power and OTES will draw 11 W, with both operating simultaneously while the EEV is in its scanning mission mode. The EEV is also equipped with the Radar Imager for Mars' Subsurface Experiment (RIMFAX) as its ground penetrating radar that determines the viability of sample collection from the area around the EEV after landing. This will draw up to 10 W while it is in use. Exterior engineering cameras will also be used in order to monitor the vehicle and as give the astronauts an ability to see the outside space. These cameras will always be on and continuously use 100 W of power during the mission. In total these payloads will contribute 24 kg to the vehicle mass.

These payload components produce scientific and engineering data and will be processed by the EEV's command and data handling system. This will use proven radiation hardened hardware similar to what is currently in use on Mars on the Perseverance rover. Four independent computers will be used, allowing for redundancy in the hardware in case of any single event effects. A hard drive will also be present for data storage. While significant amounts of non-mission critical data can be stored for processing after the mission, such as non-time sensitive video feed, there is still some data that needs to be processed in a timely manner, such as potential landing site parameters or critical engineering data about the health of the crew and system. For this the EEV is equipped with a communications system that consists of two antennae, with one each for a low-gain and a high-gain. The low-gain antenna primarily serves as a receiver and will downlink commands from the Deep Space Network (DSN) on the 7.16 GHz band. The high-gain antenna will primarily transmit data to the DSN and DST, which will have two more crew members who may assist in some data processing or forward it back to the DSN. This antenna operates on a transmitting frequency of 8.42 GHz. These two systems have a combined mass of 73 kg and draw 1000 W of power.

In order to meet the thermal requirements of the spacecraft, the vehicle is covered in MLI and has powered heaters strategically placed to ensure that the temperature remains within acceptable ranges throughout the mission. The entire system will require a maximum of 706 W of power for the heaters and a mass of 155 kg.

Between all the subsystems and flight modes, the EEV will have a maximum power usage of 5870 W. To meet these energy demands, the EEV is equipped with solar panels, with batteries used for power during the eclipse periods. The solar arrays will have a total area of 42.8 m^2 in the shape of two decagonal folding panels. These solar arrays will be gimballed to ensure sure that they are properly oriented and always pointing towards the Sun. With some redundancy built in for one battery out or minor solar array damage, the mass of the power system mass totals 1020 kg.

The EEV will utilize 3-axis attitude control with hydrazine monopropellant thrusters, as well as reaction wheels to provide extra stability for the scanning payloads, as they have narrow fields of view. This will enable the spacecraft to manage its attitude in order to maximize solar panel exposure, dock with the DST, and point its scanners or thrusters. Star trackers and magnetometers will allow for attitude determination. This system will require 368 W and have a mass of 16 kg.

The EEV's concept of operations will involve it moving from the DST parking orbit to both moons, and then back to the DST, requiring a minimum of 2.15 km/s of Δv. This necessitates a very large, bi-propellant propulsion system. As the EEV will be in orbit around Mars for years before the arrival of the crew, storable hypergolic propellants were selected, with 8170 kg of propellant mass estimated to be required for the mission. The propellant tanks are integrated into the primary service module structure to maximize the internal volume of the tanks. In order to minimize the mass of the main tanks, they will not be equipped with propellant management devices (PMDs) and will be made of stainless steel due to its high strength-to-weight ratio and lack of reactivity to the propellants. To ensure propellant flows into the engines prior to burns, there will be smaller header tanks equipped with PMDs that will provide the propellant to the engines until the main tanks are settled. When also considering the mass of the engines and plumbing, the system will have a dry mass of 1750 kg.

The remaining structural mass comes out to 2870 kg, which is mainly in the inflatable crew habitation module. This maximizes the amount of internal volume of the habitation module for the allotted mass. The structure uses redundant bladders with restraint layers for rigidity, with micrometeoroid and orbital debris protection added on top. During launch, the bladders are deflated and restrained on a central aluminum structure that doubles as the radiation shelter. After separation from the launch vehicle, the bladders inflate, and the crew module expands to its maximum diameter. The other large source structural mass is in the landing legs, which will enable the spacecraft to remain stable on the surface of the moons, despite the extremely low gravity.

As this is a sample collection mission to other bodies, cleanliness of the spacecraft will be maximized in accordance with planetary protection requirements. The vehicle will remain in a high orbit at the end of its mission to minimize the risk of any contamination that may come with it crashing into either Mars or one of the moons.

The EEV is projected to cost $2 billion, which is not compliant with the $1 billion requirement. However, this budget requirement will not ensure a safe crewed mission, and it has been found that a crewed mission of this magnitude would require an increase in funding. However, a robotic sample collection mission would greatly reduce this cost, as it would not require the large habitation module, ECLSS and associated power systems, or large propulsion system. This could still interface with the DST and use the same sample collection mechanism that transfers the samples to the DST. If this mission architecture were to be used instead, the engineering and scientific objectives of the mission would still be met, as human spaceflight and mission operations in the Martian SOI would still be conducted, while samples from the Martian moons would still be retrieved and returned to Earth.

Other than the cost requirement, the crewed EEV design is compliant with each technical and managerial requirement given in the RFP.

2.0 Mission Overview

2.1 Needs Analysis

Crewed exploration of Mars is the next frontier in humanity's quest for the stars. However, no crewed vehicles have ever ventured beyond Earth's sphere of influence (SOI). To ensure a safe landing on Mars, an incremental approach to human missions should be utilized. A key step in this is in crewed missions in the Martian SOI. This will provide invaluable experience in developing procedures for these missions, as well as data in testing systems in the Martian environment.

The two Martian moons, of Phobos and Deimos, provide an opportunity for crewed exploration of the Martian sphere of influence, as their exploration does not necessitate the development of heavier vehicles for landing on and taking off from the surface of Mars itself. Both moons are scientifically interesting and can provide insights into the history of Mars and the Solar System, as they share many similarities with class C and D-type asteroids. Samples obtained from both moons for examination on Earth, therefore, will provide data that would advance studies into the formation of the Solar System and allow for a greater understanding of its history and what resources are present.

2.2 Mission Objective

In order to meet the needs of the engineering and scientific communities in the Martian SOI, the key mission objective will be to bring a crewed vehicle to both Martian moons to collect samples and bring them back to Earth. This will provide the procedural and testing data that will enable future missions to Mars and eventually enable a human landing on the planet's surface while also providing the material needed for extensive studies into Phobos and Deimos.

2.3 Customer Requirements

The customer has included several system level requirements in their request for proposals (RFP), which have been tabulated in Table 2.3-1. Each one has been labelled with where in the RFP it comes from and assigned a reference ID, which is done to enhance the traceability of the requirements.

The reference ID consists of three parts. The first is a letter to describe the type of requirement: T for technical requirements, C for a cost requirement, and M for a managerial requirement. The next section refers to what part of the WBS (Appendix A) the requirement is relevant to. The last number, following a dash, numbers the requirement in relation to the other requirements from that WBS section and type. For example, T4.1-1 is the first technical requirement for the life support system (ECLSS).

The RFP paragraphs are labelled where the main sections in the RFP contribute the first number, the subheadings contribute the second, and the paragraph number itself contributes the third. Section 4.1 is for the Design Requirements and Constraints section of the RFP.

Table 2.3-1: Customer System Level Requirements

RFP paragraph #	Reference ID	Requirement Statement
4.1.1	T4.1-1	The EEV must support two crew members when visiting both moons
4.1.1	T6.2-1	The total mission duration must not exceed 30 days following departure from the DST
4.1.1	T5.0-1	The EEV must be able to collect at least 50 kg worth of samples from each moon
4.1.1	T6.0-1	The crew must remain inside the EEV for the entire mission duration
4.1.2	T6.0-2	The mission must produce significant scientific data from the moons
4.1.2	T7.0-1	The samples must be quarantined from the crew until arrival at Earth
4.1.3	T4.0-1	The EEV must autonomously dock with the DST
4.1.4	T6.0-3	The EEV must launch on an existing launch vehicle

4.1.7	C0.0-1	The vehicle and its launch cost shall not exceed $1 billion (2021 equivalent)
4.1.4	M6.2-1	The EEV must be in a 5-SOL parking orbit around Mars by Summer 2040
4.1.5	M4.0-1	Preferably utilize system and subsystem levels with higher TRL

3.0 Mission Design

3.1 Architecture Considerations

To ensure that an optimal vehicle design was selected, multiple mission architectures were considered. These were differentiated in how the EEV's operations for the Martian moons would play out. Architecture 1 would use a single space vehicle that would land the crew on the moon itself, while architecture 2 would use an unmanned sample collection vehicle (USCV) to collect samples and bring them back to the EEV, which would be stationed in space nearby.

3.1.1 Architecture 1

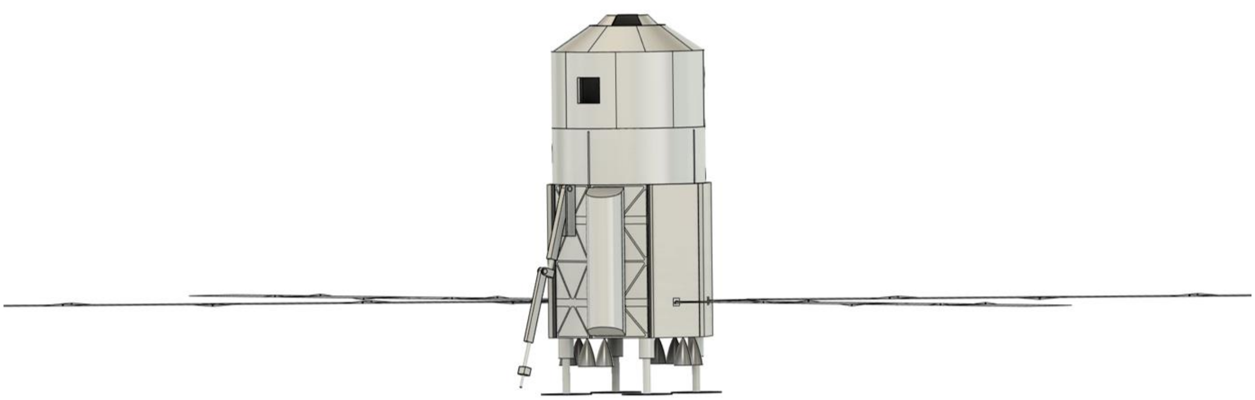


Figure 3.1.1-1: Architecture 1

3.1.2 Architecture 2

Figure 3.1.2-1: Architecture 2

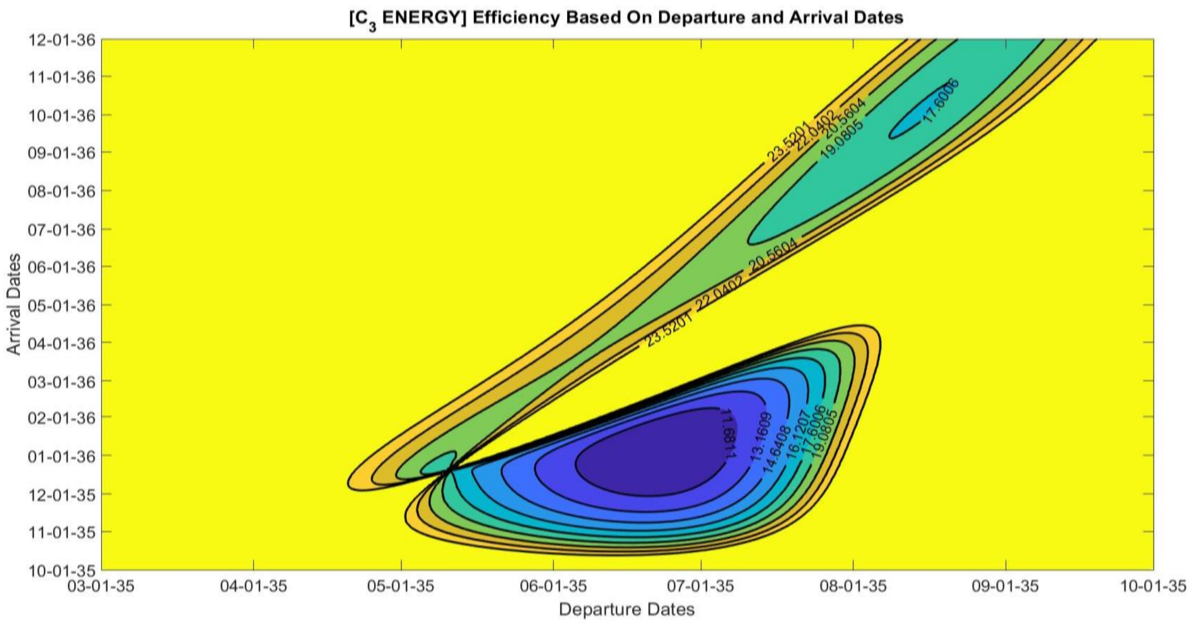
3.1.3 Down Selection

Architecture 1 was selected as it was estimated that the addition of the second vehicle would increase launch mass and mission complexity, ultimately driving up the cost of the mission and making the EEV non-compliant with the $1 billion requirement. After selecting architecture 1, the EEV was redesigned to incorporate additional crew habitation space, decreased solar panel length, increased leg length for stability, and an outer inflatable crew module insulation system.

3.2 Concept of Operations

3.2.1 Porkchop Plots

In order to depart Earth at an optimal date, Porkchop Plots were created upwards of 5 years prior to mission start. It is important to note that every 11 years, the departure and arrival window between Earth and Mars becomes most efficient. With respect to the mission, that window occurs in the Summer of 2035. Figures 3.2.1-1,2,3 demonstrate Porkchop Plots 5, 3, and 1 year prior to mission start, respectively. It is important to select a departure and arrival opportunity that requires characteristic energy (C_3) within the bounds of the launch vehicle.



<u>Figure 3.2.1-1 Departure and arrival efficiency 5 years prior to mission start</u>

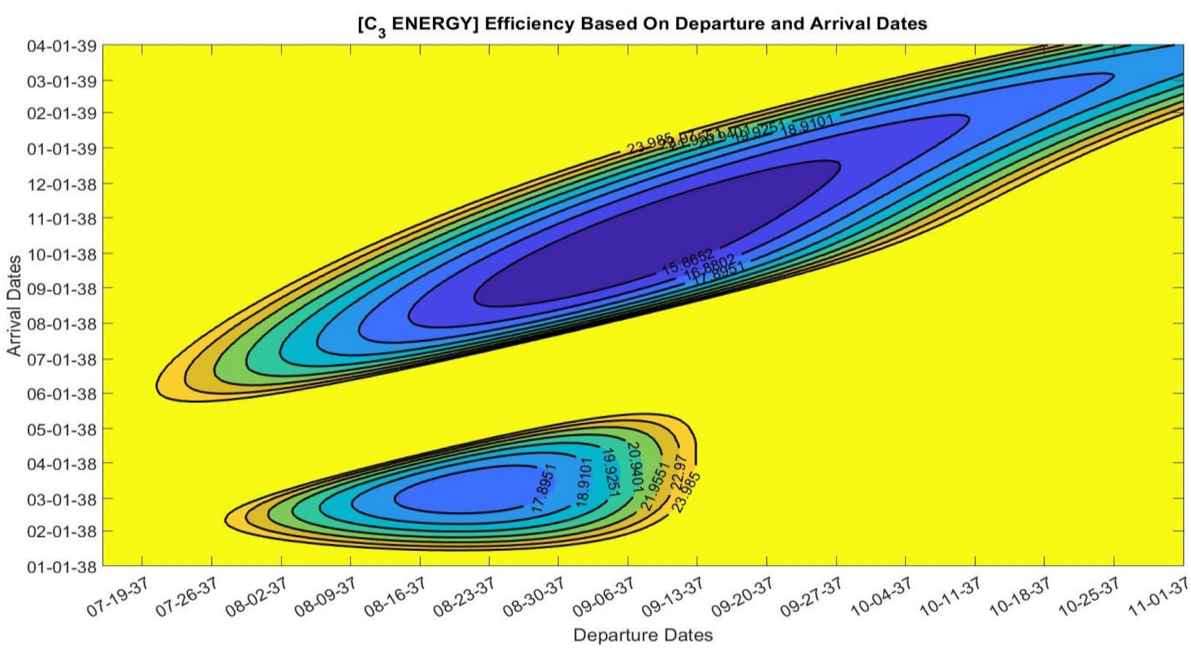


Figure 3.2.1-2 Departure and arrival efficiency 3 years prior to mission start

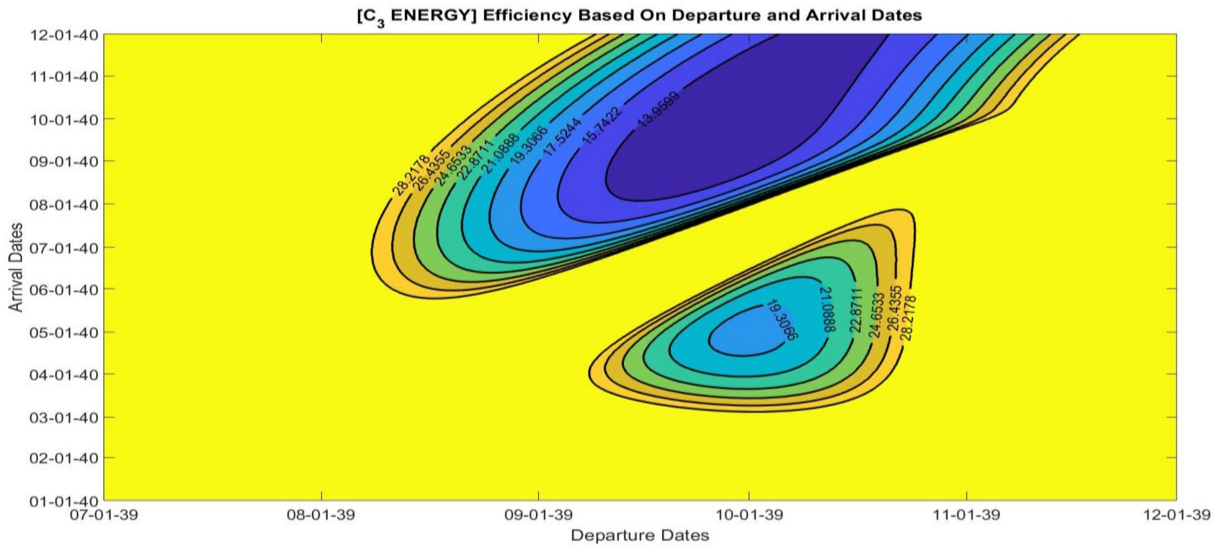


Figure 3.2.1-3 Departure and arrival efficiency 5 years prior to mission start

From the plots, launching 1 year prior to mission start is not feasible as the most efficient opportunity would occur after the arrival of the DST. Per the RFP, this launch window is disqualified as the EEV must arrive prior to the arrival of the DST (does not meet SLR M6.2-1).

Launching 3 years prior to mission start requires a minimum C_3 of 15.7 km^2/s^2, whereas launching 1 year prior requires a minimum C_3 of 11.7 km^2/s^2. Therefore, the spacecraft will launch Summer of 2035 to reduce C_3 as much as possible (aids in meeting SLR T6.0-3). As a result, the spacecraft will arrive around January of 2036.

3.2.2 Trajectories

STK was used to display the sequence of events from the moment the spacecraft departs Earth to landing on both Phobos and Deimos. The spacecraft will launch from Cape Canaveral, Florida at Complex 39A. This launch pad is leased to SpaceX by NASA. The launch date is set to June 15th, 2035. After Earth departure, the launch vehicle will enter a 300 km circular parking orbit, as demonstrated in Figure 3.2.2-1,2.

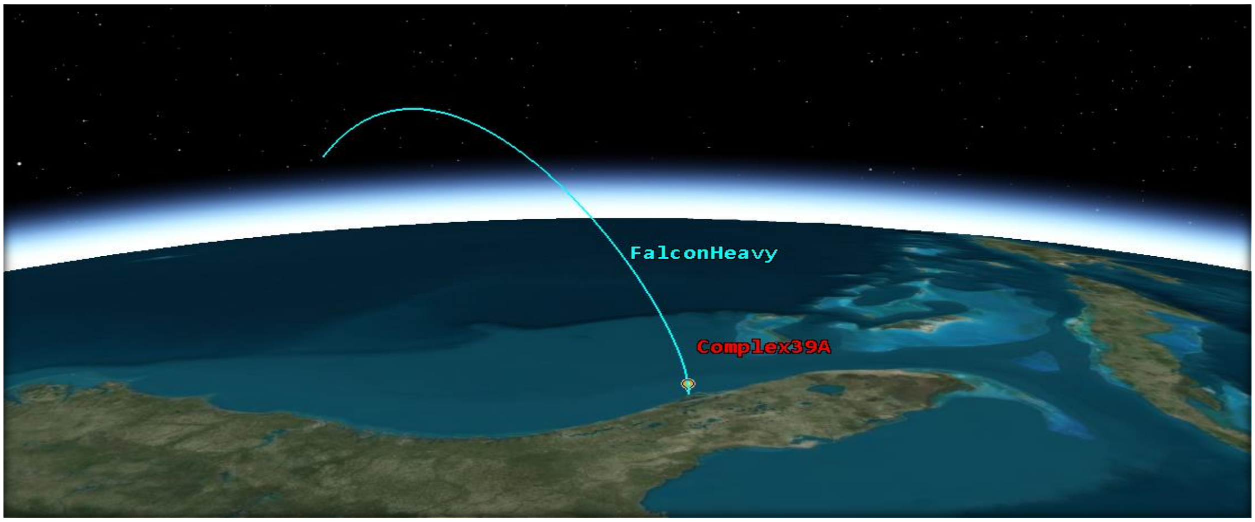


Figure 3.2.2-1 The Falcon Heavy departing Earth

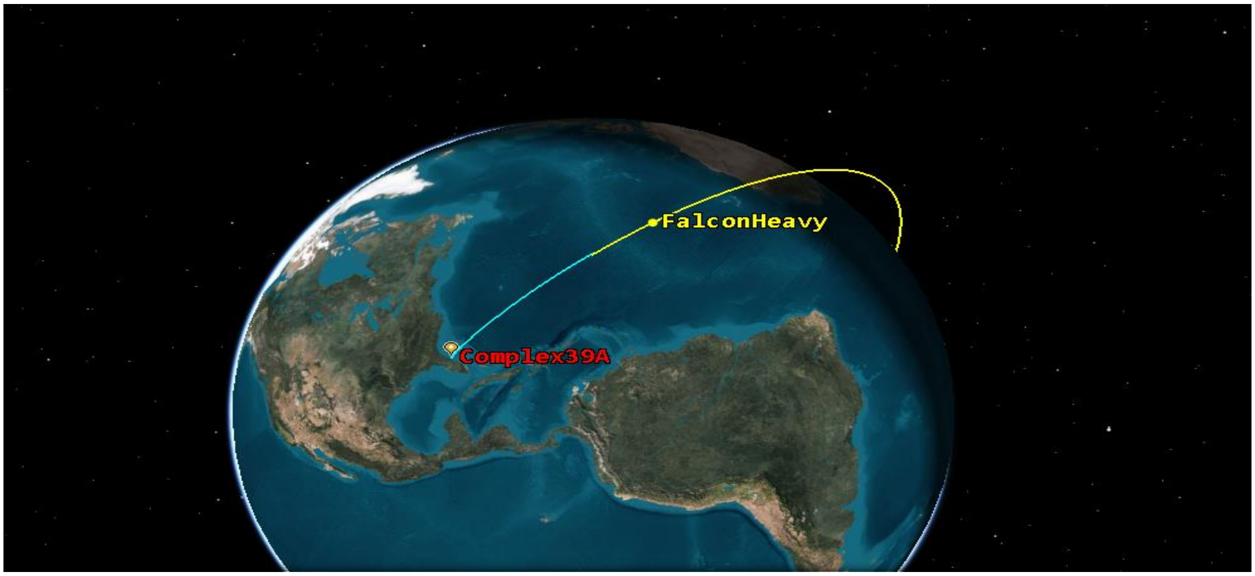
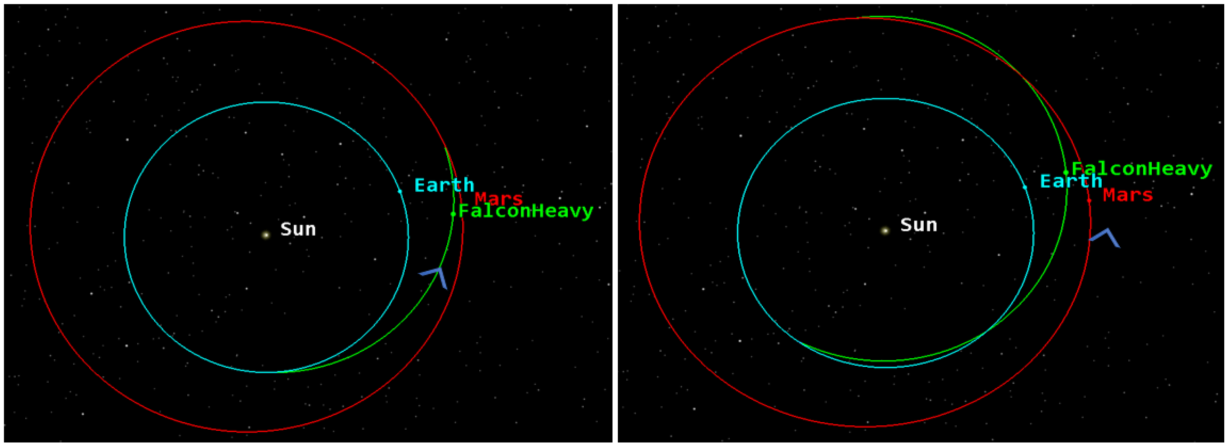


Figure 3.2.2-2 Circular parking orbit insertion around Earth

Once a circular orbit is obtained, the Falcon Heavy will perform a burn to initiate a Hohmann Transfer and propel itself towards the Martian 5-SOL parking orbit. To verify the accuracy of the Porkchop Plots, 2 different interplanetary trajectories were found using STK. These trajectories can be seen in Figure 3.2.2-3. Interplanetary Transfer #1 falls within the lower C_3 range from Figure 3.2.1-1, whereas Interplanetary Transfer #2 does not. A trade study was conducted to select between the two proposed trajectories. However, it is clear that Interplanetary

Transfer #1 is ideal due to the required C_3. Interplanetary Transfer #1 has a TOF of 186 days and requires a total Δv of 3.75 km/s. Mars arrival is estimated to occur on November 11th, 2035.



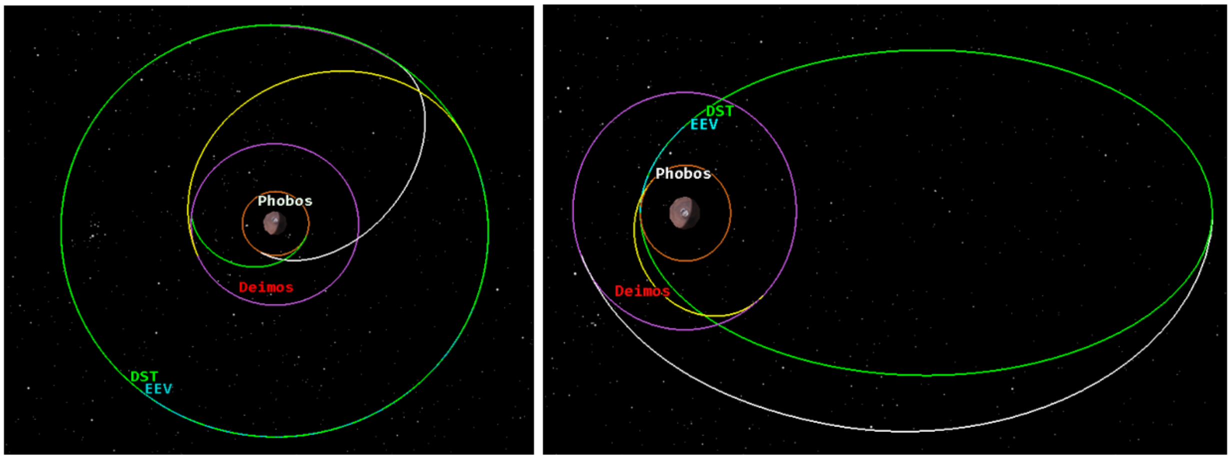
Interplanetary Transfer #1 Interplanetary Transfer #2

Figure 3.2.2-3 Interplanetary Hohmann transfer to Mars from Earth

Table 3.2.2-1 Trade study to select optimal interplanetary trajectory

FOMs / Trajectories	Required Δv ≤ 3.5 km/s Weight = 3		Required C3 ≤ 20 km²/s² Weight = 2		Time in Mars Orbit ≤ 365 days Weight = 1		Weighted Total = SUM(W)
	U	**W**	**U**	**W**	**U**	**W**	
Interplanetary Trajectory #1	3.75 km/s	9	11/6 km²/s²	6	≤ 186 days	9	**24**
	3		3		9		
Interplanetary Trajectory #2	4.40 km/s	3	22.9 km²/s²	2	≤ 323 days	9	**16**
	1		1		9		

After arriving at the Mars SOI, 2 different Martian trajectories were proposed. A trade study was conducted to select the most optimal Martian trajectory. Figure 3.2.2-4 illustrates the 2 trajectories, and Table 3.2.3-1 showcases the trade study used.



Circular Parking Orbit   Elliptical Parking Orbit

Figure 3.2.2-4 Different 30-day Martian trajectories

Table 3.2.2-2 Trade study to select optimal 30-day Martian trajectory

FOMs / Martian Trajectories	Required Δv ≤ 2 km/s Weight = 3		Average Planet Stay-Time ≥ 10 days Weight = 2		Mission Duration ≤ 30 days Weight = 1		Weighted Total = SUM(W)
	U	W	U	W	U	W	
Circular Parking Orbit	2.41 km/s	3	11 days	6	29 days	3	12
	1		3		3		
Elliptical Parking Orbit	2.15 km/s	9	12 days	18	29 days	3	30
	3		9		3		

It is ideal to limit the required Δv as a higher Δv requires more propellant mass. A Δv of 2 km/s for a Martian mission of this nature would be considered efficient. Also, because the mission requires the EEV to obtain 50 kg worth of samples from each moon, it is ideal to reduce the required maneuver time to allow for more sampling time. Therefore, it is best to have a minimum of 10 days' worth of sampling time for each moon (aids in meeting SLR T5.0-1). Finally, per the

RFP, the Martian mission time must also not exceed 30 days as required by SLR T6.2-1. Using the stated metrics, it was clear the elliptical parking orbit was the best option.

The elliptical parking orbit has an eccentricity of 0.844. This eccentricity gives the orbit its distinct, elongated shape. The EEV and DST will rendezvous at the elliptical parking orbit. Once the 2 crew members have boarded the EEV with all the required equipment, the EEV will undock and begin the 30-day Martian mission.

The first burn after crew boarding will occur at the parking orbit's perigee. Due to location of Phobos and the parking orbit, the EEV will only have to perform a retro burn to enter an orbit around Phobos. After the EEV has landed on Phobos and collected its samples, the EEV will depart Phobos and perform a Hohmann transfer to Deimos. When the EEV has landed on Deimos and collected samples, a final Hohmann transfer is performed to arrive at the 5-SOL parking orbit for a second rendezvous with the DST. Table 3.2.2-3 lists the different steps required for a successful 30-day trajectory along with the required Δv for each maneuver. Figures 3.2.2-4 illustrates the EEV insertion into an orbit around Phobos and Deimos. It is important to note that each lunar insertion will require the EEV to scan the surface of the moon prior to landing. This is done to ensure the surface composition of the landing location will not jeopardize mission success.

Table 3.2.2-3 Trade study to select optimal 30-day Martian trajectory

Date	**Maneuver**	**Δv, km/s**
06/01/2040	DST Rendezvous w/EEV	0.001
06/01/2040	EEV Rendezvous w/Phobos	0.763
06/13/2040	EEV Initiates Hohmann Transfer to Deimos	0.415
06/13/2040	EEV Ends Hohmann Transfer to Deimos	0.332
06/25/2040	EEV Initiates Hohmann Transfer to DST	0.383
06/28/2040	EEV Ends Hohmann Transfer to DST	0.258
06/29/2040	EEV Rendezvous w/DST	0.001
TOTAL		**2.15**

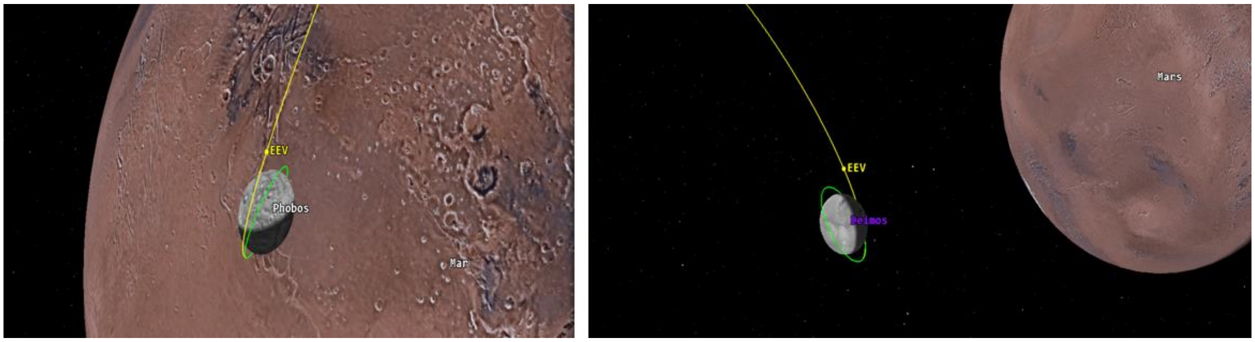


Figure 3.2.2-4 EEV rendezvous with Phobos and Deimos

3.3 Landing Site Selection

The surface composition of both Phobos and Deimos is largely unknown. Both moons have visibly discolored surfaces covered in regolith. These discolorations are theorized to be caused by space-weathering. The exterior (visible) surface of the moons is one distinct color, whereas the rock underneath this exterior surface is another color. Reasons for why deep rock is exposed range from landslides to meteor impacts. Figure 3.3-1 demonstrates the discoloration on both moons.

Phobos has two distinct colors: red and blue. The red surface is thought to be old, space-weathered rock, whereas the blue surface is thought to be rock that lies underneath the red rock. There are over 15 named craters on Phobos, with the largest crater being the Stickney crater. The Stickney crater contains both blue and red surfaces, making it an excellent landing site as the EEV has the capability of easily collecting the red and blue rock.

Deimos, like Phobos, also has two distinct colors. The red areas are radiation-stained rock, whereas the blue surface color is the rock underneath the exterior (visible) surface. Unlike Phobos, there are only 2 named craters: Swift and Voltaire. Both craters are roughly 1 km apart and both contain red and blue rock, making for an excellent sampling site location as the opportunity to collect red and blue rock is much higher in this general area.

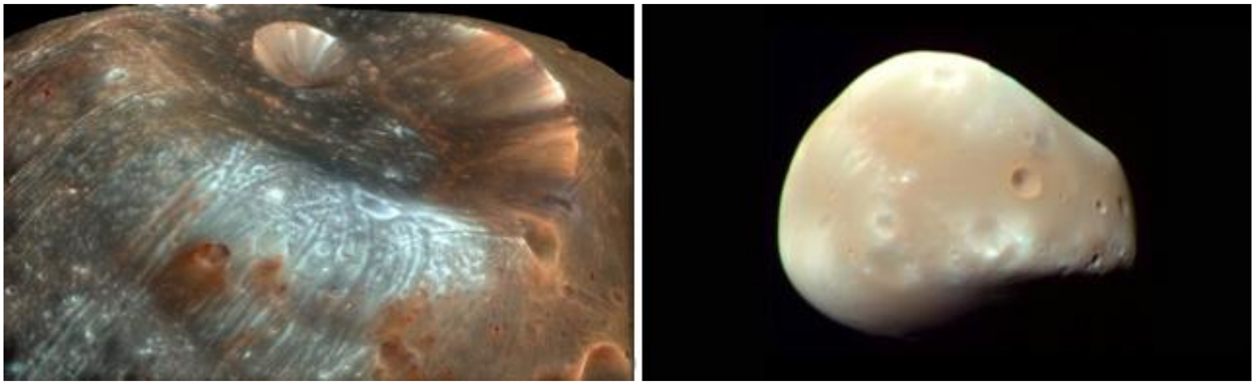


Figure 3.3-1 The surface discoloration on Phobos (left) and Deimos (right)

It’s important to note that formal data on the surface composition of Phobos and Deimos is non-existent. Although the Stickney and Swift-Voltaire craters have characteristics that make them both attractive landing sites, the EEV’s payload system will scan these areas prior to landing to ensure the safety of the crew and mission success. Figure 3.3-2 illustrates the required lunar surface scanning.

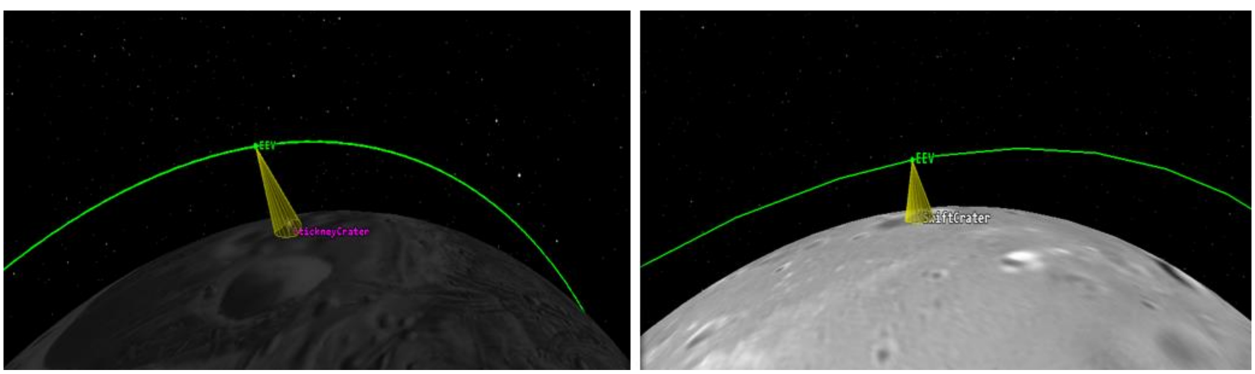


Figure 3.3-2 Moon scanning prior to landing on Phobos and Deimos

3.4 Launch Vehicle Trade Study/Selection

Multiple launch vehicles were considered to get the EEV to Mars. These candidates were the Falcon 9, Atlas V, Delta IV Heavy, Falcon Heavy, Vulcan-Centaur, New Glenn, Starship-

Superheavy, and the Space Launch System (SLS). The figures of merit used were payload mass to Mars orbit, reliability, cost, and payload fairing size.

When the trade study was conducted, it was quickly determined that the Atlas V and Delta IV Heavy would not be available for our mission due to the remaining LVs being sold. The Falcon 9 could be rolled into the Falcon Heavy for consideration, since the latter has a much closer mass capability to the EEV's requirement. SLS was not considered due to its high cost per launch exceeding the total budget requirement.

Falcon Heavy	Starship	New Glenn	Vulcan
Pros • Low cost (\$90 M to \$150 M) • Can launch the EEV with a 1.8% margin • Based on Falcon 9 (high success rate of 98%) **Cons** • Small 4.6 m payload fairing constrains the size of the crew compartment	**Pros** • >100 t to Mars (597% margin) • Extremely high payload volume (9 m diameter) • Low projected cost (<\$10 M) **Cons** • Unproven vehicle • Very complex system	**Pros** • Very high payload capacity (unknown mass to Mars) • Extremely large payload fairing (7 m diameter) **Cons** • Unknown cost per kg • Unproven vehicle • Developer has no orbital flight record	**Pros** • High payload capacity (unknown mass to Mars) • Large 5.4 m payload fairing • Low cost (\$82 M to \$200 M) • Developer has very good flight record **Cons** • Unproven vehicle

Figure 3.4-1: LV Trade Study Breakdown

Falcon Heavy was selected from the remaining four options as it has the capability to launch the EEV directly to its orbit around Mars in a fully expendable configuration. As the Falcon Heavy has never had an unsuccessful flight and Falcon 9 has had a very high success rate of 98%, it has a far higher score for reliability than the other three LVs, which do not have orbital track records. However, it has the smallest payload fairing of the LVs that were considered, which constrains the geometry of the spacecraft.

Using the spacecraft geometry, it has been determined that no ballast will be needed for the launch, as the spacecraft center of mass is located 2.7 m above the payload attach fitting plane. This is within the allowable distance for the 2624 mm PAF that is offered by SpaceX for a 16.5-ton spacecraft.

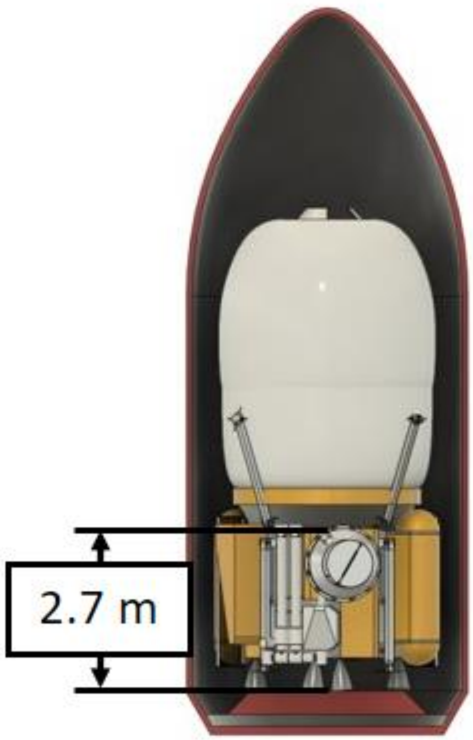


Figure 3.4-2: Spacecraft Launch Configuration

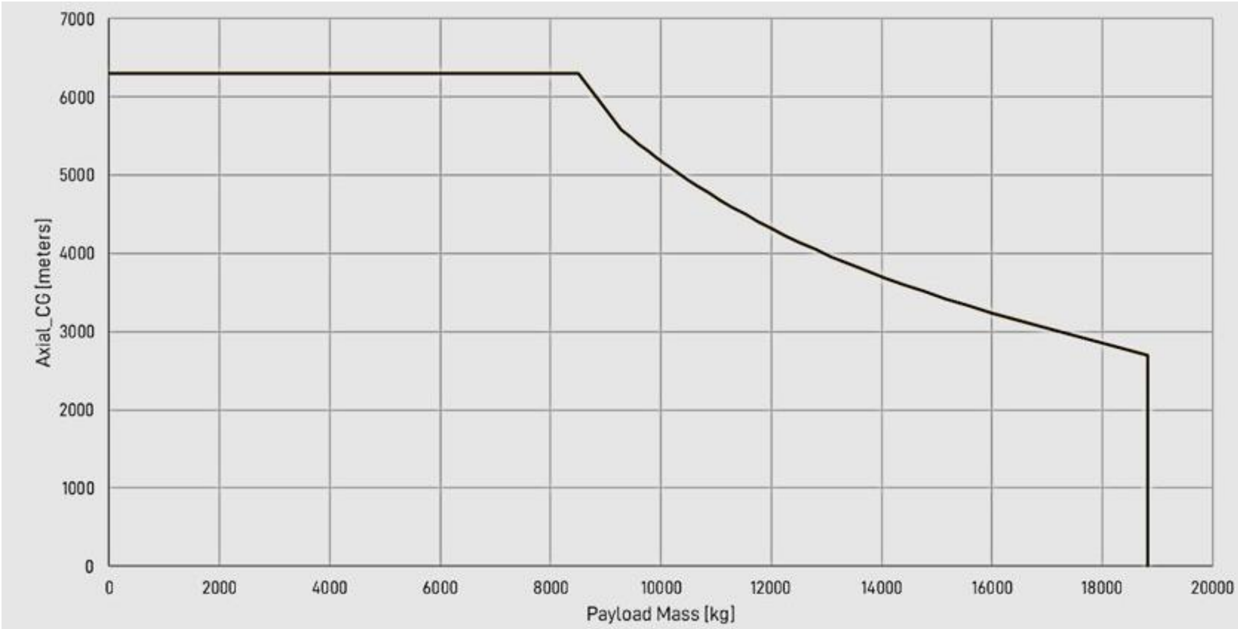
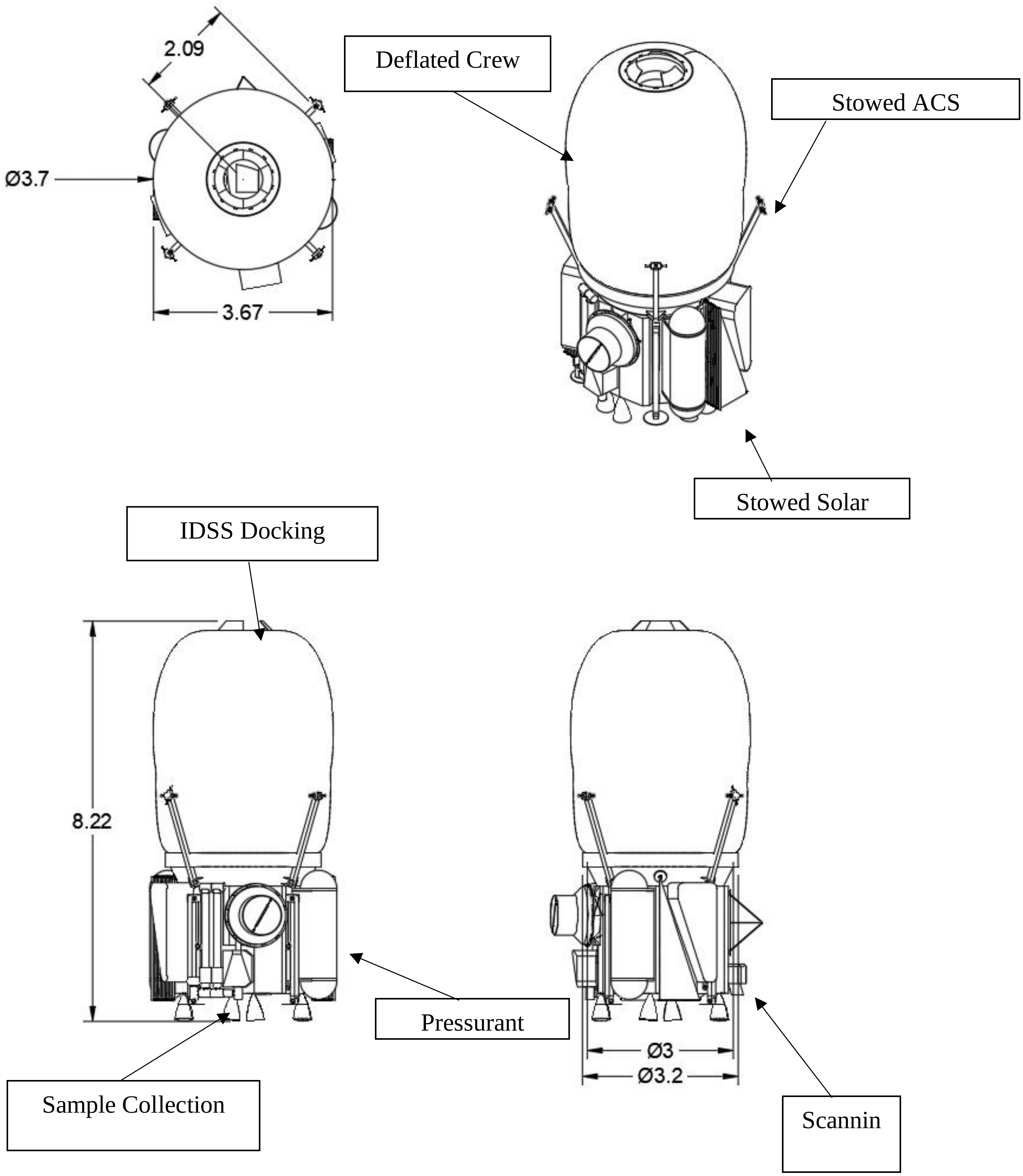


Figure 3.4-3: SpaceX PAF 2624 Capability [1]

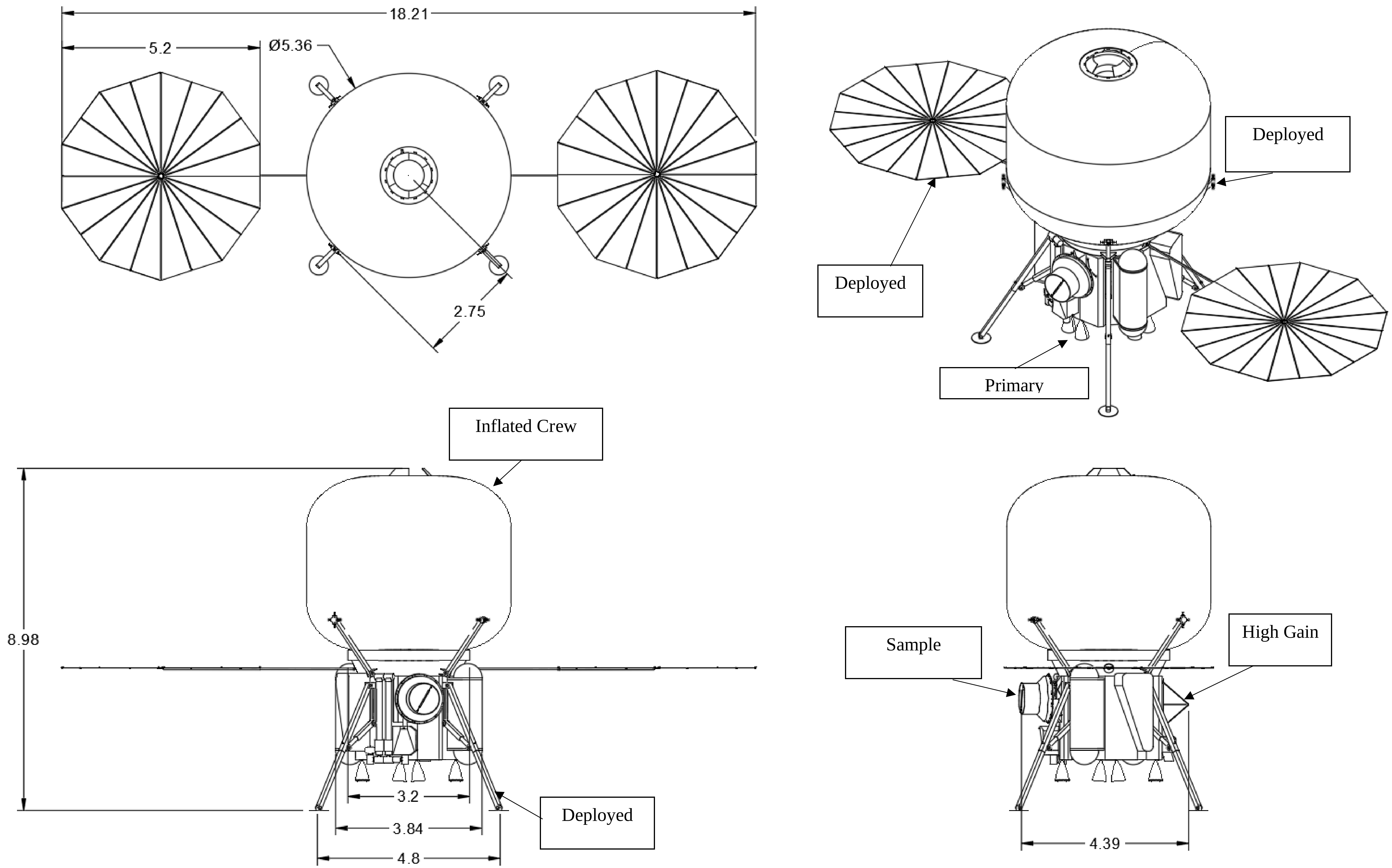
As the Falcon Heavy only offers a 1.8% mass margin, a contingency has been planned if the mass of the EEV increases or a less optimal transfer time is required due to a schedule slip. This plan would be in the form of using a STAR 48B solid rocket motor as a kick stage. It has been found that this will increase the payload mass to Mars by 1000 kg and cost an additional $10 million.

3.5 EEV Configurations

3.5.1 EEV Stowed Configuration



3.5.2 EEV Deployed Configuration



4.0 Vehicle Design

4.1 Environmental Control and Life Support System

The Environmental Control and Life Support System (ECLSS) is an essential subsystem needed to sustain the lives of the EEV's astronaut crew of two for the duration of the 30-day manned mission. The ECLSS must fulfill the crew's nutrition, waste management, and medicinal/first aid needs. The ECLSS also provides a safe and comfortable environment for the crew, i.e., atmosphere control, water management, fire mitigation, and radiation mitigation. ECLSS feature selection is based on the mission duration of 30 days plus a 15-day margin to provide a buffer for the purposes of safety. This 45-day duration fits within the 12 days to 3-month window shown in **Figure 4.1-1**. The EEV's environmental control and life support system consists of the features in green and are configured in an open loop system where all consumables will be transferred aboard the EEV from the DST directly prior to the EEV's manned mission start. These consumables include oxygen, nitrogen, carbon dioxide (CO2) scrubbers, food, and water. This open loop design reduces complexity and financial cost for increased system mass.

Figure 4.1-1 Life Support features as a function of mission length

4.1.1 Atmosphere Control and Revitalization

The open-loop atmosphere control and revitalization subsystem is designed to provide the EEV with a viable atmosphere with-in the crew cabin for a duration of 45 days.

The crew will expel an estimated 97.2 kg of CO^2 throughout the mission. Figure 4.1.2-1 shows one of the 24 canisters of lithium hydroxide (LiOH) that is used to scrub the CO^2 from the cabin atmosphere. 75.6 kg of oxygen and 130 kg of nitrogen will be brought aboard the EEV from the DST in COPV tanks like the one shown in Figure 4.1.2-2, enough for 45 days' worth of supply for 2 average humans. The oxygen and nitrogen supply is monitored and regulated by total pressure sensors and oxygen pressure sensors. A Major Constituent Analyzer, similar to the one aboard the International Space Station (ISS), will be used to monitor the atmospheric distribution of nitrogen, oxygen, carbon dioxide, methane, hydrogen, and water vapor.

Figure 4.1.2-1 (left)Canister of Lithium Hydroxide (LiOH)

Figure 4.1.2-2 (right) COPV tank

Given the crewed nature of the mission, additional care must be taken for the prevention of, monitoring of, and protection from particulates, biological microorganisms, and toxic substances that may pollute the cabin atmosphere. This is done through passive preventative measures that scrub and control the atmosphere environment, and active methods such as a rigorous housekeeping schedule. The ECLSS provides several trace contaminant sensors and portable air samplers placed throughout the habitation module to monitor air quality and provide sufficient information and time for timely corrective action in the event of atmospheric anomalies. Trace contaminant levels are controlled using activated carbon filters with acid impregnation. Biological contaminants are curbed using HEPA filters which limit microbe and particulates in the air. Additionally, the relative humidity of the atmosphere is kept beneath 70% to prevent the growth of microbes.

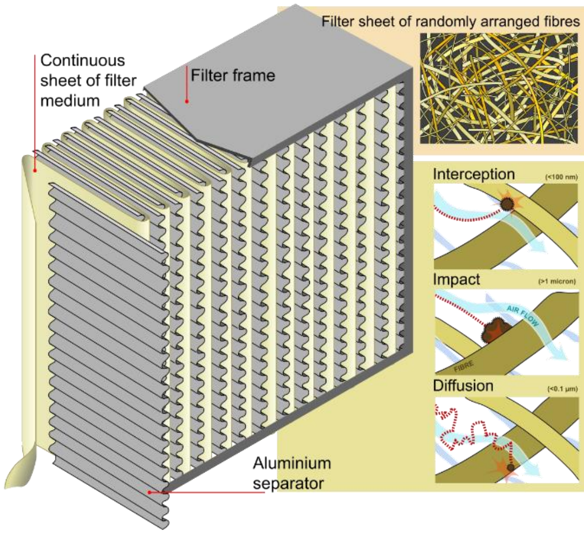


Figure 4.1.2-3 HEPA Filter Corrugated Internal Structure

For comfortable and safe habitation, the crew cabin must maintain a temperature between 18°C and 26°C and a humidity level between 25% and 70%. These levels are maintained using the dehumidifier in Figure 4.1.2-3 and the cabin fans in Figure 4.1.2-4. The dehumidifier uses the vacuum of space to draw humidity across its water-permeable (but air-impermeable) Nafion membranes. The water vapor is then expelled into space in the same orbital direction as the EEV to mitigate collisions with the spacecraft's external surfaces and equipment.

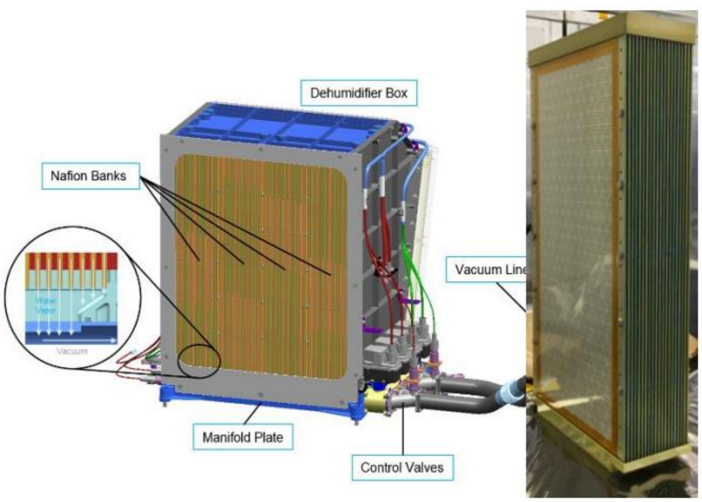


Figure 4.1.2-4 Dehumidifier box (left) and Nafion blade bank (right).

The cabin fans provide ventilation throughout the EEV to ensure proper thermal gradients are maintained, contaminant buildup is reduced, and equipment is properly cooled. Any recirculation within the 3-fan mechanism is eliminated by a shuttle valve called a Flapper box. The Flapper Box creates a streamlined duct downstream of the active fan that minimizes pressure drop.

Figure 4.1.2-5 Flapper box. Operating modes with center fan and left-hand fan active

4.1.2 Water Management

Water is an essential consumable that will be brought aboard the EEV as per the open loop design. The water will be stored in large rubber bladder tanks and piped through plumbing made of stainless steel and Teflon wrapped stainless steel mesh. Pumps and fans propel the water throughout the plumbing system with filters installed at points of use as needed. The filters provide further water conditioning prior to crew usage. Emergency-use contingency water carriers are also provided in the event of plumbing failure. Water usage is further defined in section 4.1.6.

Figure 4.1.3-1 Contingency Water Carriers

Monitoring the quality of the crew's potable water is of the utmost importance. The ECLSS will monitor the crew's potable water quality using a Second-generation Total Organic Carbon Analyzer (TOCA), shown in Figure 4.1.3-2. The TOCA oxidizes organic carbon species present in the water to carbon dioxide gas and measures the concentration using nondispersive infrared spectroscopy. The TOCA will assess the quality of the water supply on a weekly basis.

Figure 4.1.3-2 Second Generation Total Organic Carbon Analyzer

4.1.3 Waste Management

A crew of two is projected to produce, on average, 16.4 kg of fecal and urine waste over a 45-day period. To dispose of this waste in a comfortable and hygienic manner, our ECLSS includes a waste management system much like the one aboard the ISS, i.e., a fancy space toilet. The inside of the toilet will be lined with pretreated bags which will capture and store fecal matter. The bags will be vacuum sealed with the waste and then stored aboard the spacecraft. Urine will be captured through a suction cup mechanism, which will then be vented overboard much like the water vapor from the atmosphere's humidity. Vents will be heated to prevent urine from freezing during the venting process. In case of a waste management system failure, the crew will be provided with backup waste bags as an alternative. Hygiene and cleaning equipment will be included.

Figure 4.1.4-1 Space Toilet Aboard the ISS

4.1.4 Fire Detection and Suppression

A fire detection and suppression system is necessary for the crew's safety and health. This subsystem detects and eliminates fire threats using a photoelectric fire detector and two portable fire extinguishers (PFE). Additionally, four portable breathing apparatuses (PBA) are provided for crew safety to prevent smoke inhalation. Should a fire occur on the EEV, the air will be scrubbed using post-fire air revitalization filters and the crew will use cleaning pads to remove any toxic particles from surfaces. Figure 4.1.5-1 shows the PBA and PFE aboard the Columbus in the European Science Laboratory. This same equipment is stored on the EEV.

Figure 4.1.5-1 PBA and PFE aboard the Columbus that are equipped on the Mars EEV

4.1.5 Food Storage, Preparation, and Nutrition

Throughout the duration of the mission, the crew will be supplied with 2000-3000 calories per person per day, depending on the astronaut's specific nutritional needs. Including the 15-days of contingency supplies, 159.3 kg of dried food has been allocated for and will be transported to the EEV from the DST. Figures 4.1.6-1 and 4.1.6-2 show the dry food packets the crew will receive, and the bulk overwrap bags that they will be stored in, respectively. The crew's water supply is allocated for 72 kg of food rehydration water and 145.8 kg of drinking water, which will all be stored in the rubber bladder system mentioned in Section 4.1.3.


<u>Figure 4.1.6-1 (left) Dry food packets</u>

<u>Figure 4.1.6-2 (right) Dry food stored in bulk overwrap bags</u>

The European Space agency has observed bone loss to occur at 1-2% per month and a muscle loss of 10-20% to occur even on short missions. To lessen the effects of microgravity on the EEV's crew, they will be provided with resistance bands like the ones shown in Figure 4.1.6-3. Because of the mission's limited mass and monetary budgets, larger equipment is not feasible.

<u>Figure 4.1.6-3 Crew exercise resistance bands</u>

4.1.6 Radiation Protection and Monitoring

Radiation is one of the primary concerns for the crew's safety. The EEV's radiation mitigation strategies therefore have been designed to follow the ALARA principle (as low as reasonably achievable). The ECLSS supplements the shielding provided by the inflatable walls of the habitation module with AstroRad vests, which are being tested aboard the Orion Spacecraft's crew module. These vests selectively shield the most sensitive organs and tissues categorized by high Radiation Exposure Induced Death (REID) probabilities while also shielding stem cells which enable the recovery of damaged tissue, significantly reducing probability radiation induced-cancer Additionally, the habitation module includes a central radiation shelter constructed from 7075-T6 Aluminum, providing 2.7 g/cm^2 of extra shielding for the crew during high radiation events. To detect such radiation events, the spacecraft is equipped with Radiation Assessment Detectors throughout the habitation module to monitor for the presence of high-energy charged particles. These are silicon detectors equipped with small cesium iodide blocks which were utilized on the Mars Curiosity Rover.

Figure 4.1.7-1 AstroRad Vest

4.1.7 Crew Accommodations

Other accommodations include essentials such as personal hygiene equipment, clothing, medicine, etc., which are all itemized in Table 4.1.10-1 under the Human Accommodations subsection. These accommodations and their masses were defined and calculated by design specifications found in NASA's Human Integration Design Handbook and Space Mission

Engineering: The New SMAD. The key accommodation to note is medication. The EEV will provide basic medicine and supplements to facilitate crew operation during the mission duration, including items on par with cold and sleep medicine. Supplements will also be allocated based on the individual needs of the selected astronauts. As for emergency care, the EEV will only provide basic emergency care and medical supplies on a first aid basis. Any major emergency incidents will require treatment at the DST.

4.1.8 Crew Compartment Design

The crew compartment design is made from five main layers and can be seen in figure 4.1.9-1. The first layer is a made out of Nomex which is flame resistant and puncture resistant. Since this is the inner layer where the two crewed members can physically touch, it is important that it is also soft so that no one gets hurt. The second layer is a three bladder layer made out of a polyurethane-saran laminate. This layer is mechanically integrated at cold temperatures to ensure that they are flexiable and puncture resistant. These three bladder layers are pressureized with nitrogen at four atmospheres and seperated by bleeder cloth and then sealed to the interface. After that they are restrained by the third layer. A kevlar felt cloth layer that reinforces the bladder layer and provides more puncture resistance which protects the bladders from any abrasion. After the kevlar restaint layer there are 4 micrometeroid orbital debris (MMOD) protection layers which are made from kevlar and nextel. These shield layers are separated by foam spacers which act as their own thick layers. This MMOD layer acts as an insulation layer and causes space debris particles to shatter and lose energy the deeper it penetrates. The fifth and final layer is a thermal protection layer that is based on the ISS standard multi-layer insulation (MLI) design. A mylar layer which has an inner and outer covered layer called kapton. Both of these layers are reinforced with double aluminized layers. Then as an outer layer attached to the mylar there is a beta cloth layer connected which helps protect against any atomic oxygen and is also aluminzed on the inside to help block any light transmitted. These five main layers were put together to create the crew compartment were the two crewed members will live in, and be protected from the outerspace environment.

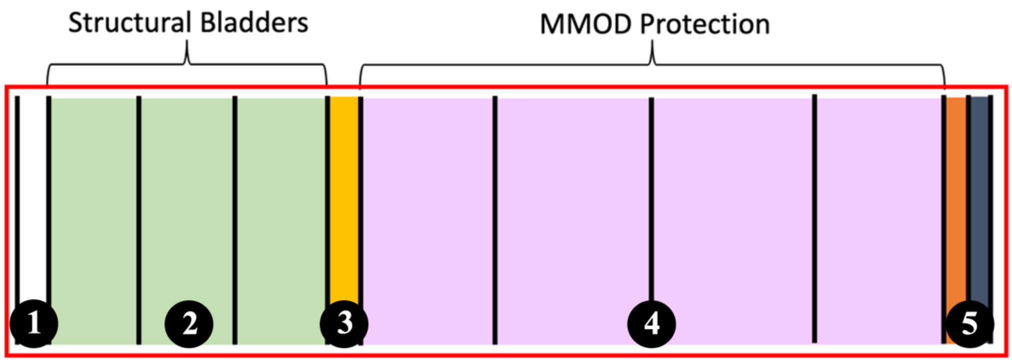


Figure 4.1.9-1 Inflatable Crew Design

4.1.9 Mass and Power Statements

Table 4.1.10 outlines the mass, volume, and power of the EEV's ECLSS. Note that all of the consumables and most of the accommodations are planned be brought aboard the EEV from the DST prior to the start of the mission in the Martian sphere of influence. This will reduce the launch mass of the spacecraft and allow for the inflatable structure design.

Table 4.1.10-1 ECLSS Mass, Volume, and Power Statement

Component	Mass (kg)	Volume (m^3)	Power (W)
Atmosphere Control & Revitalization			
oxygen	75.6	0.19	-
oxygen tank- COPV	3.5	0.2	-
oxygen supply hardware	1.4	0.45307	-
CO2 removal - LiOH	96	0.108	50
nitrogen	130	0.3035	-
nitrogen tank - COPV (x6)	62.268	3	-
nitrogen supply hardware	0.72	0.679605	-
atmosphere filtration	9.6	0.339802	900
atmosphere monitoring	35	0.509703	
ventilation fans	1	0.0396436	100
dehumidifier	4.5	0.254852	500
Subtotal	419.588	6.0781756	1550
Water Management			
water (total)	217.8	0.22	-
water tank - rubber bladder	3	0.24	-
water supply hardware	0.48	0.509703	-
water quality monitor	4	0.01415842	60

water filtration	13.2	0.00707921	-
Subtotal	238.48	0.99094064	60
Waste Management			
waste management system	45	0.254852	110
waste management backup	5	0.0566337	-
waste hygiene	2	0.0283168	-
Subtotal	52	0.3398025	110
Fire Detection & Suppression			
photoelectric fire detectors	0.36	0.00284	9
portable breathing assemblies	1.2	0.00708	-
portable fire extinguishers	2.35	0.0588	-
Subtotal	3.91	0.06872	9
Radiation Mitigation			
AstroRad vests	6.26	0.0545	-
radiation detectors	1.5	0.003	4.2
Subtotal	7.76	0.0575	4.2
Human Accommodations			
food	159.3	0.126	-
personal hygiene	23	0.169901	-
clothing	12	0.0849505	-
recreational equipment	10	0.00530941	-
housekeeping	65	0.141584	-
operational restraints	33	0.113267	-
maintenance*	72	0.181584	-
sleep accommodations	18	0.254852	-
medical kits + health care	20	0.147345	-
lighting	6.27	0.113267	20
Subtotal	418.57	1.33805991	20
Total	**1140.31**	**8.87319864**	**1753.2**

4.2 Payload

The payload onboard the EEV consists of a Sampling and Caching System, a Thermal Emission Imaging System (THEMIS), an OSIRIS-REx Thermal Emission Spectrometer (OTES), a Radar Imager for Mars' subsurface experiment (RIMFAX), an ECLSS, and ten exterior visual cameras. The Sampling and Caching System will function similarly to the sampling and caching system on the Perseverance rover. It will be responsible for collecting and storing the samples while keeping them quarantined away from the crew. This system will weigh around 320 kg and have a maximum power usage of 200 watts. The Thermal Emission Imaging System will be responsible for determining the thermal properties of potential landing zones. This system will weigh around 11.2 kg and have a maximum power usage of 14 watts. It is located on the bottom of the EEV and has a FOV of 1.43 degrees, as can be seen in Figure 4.2-1 as the blue FOV. The Thermal Emission Spectrometer will be responsible for determining the mineralogy of potential landing zones. This system will weigh around 6.3 kg and have a maximum power usage of 10.8 watts. It is also located on the bottom of the EEV next to THEMIS and has a FOV of 0.37 degrees, as can be seen in Figure 4.2-1 as the red FOV.

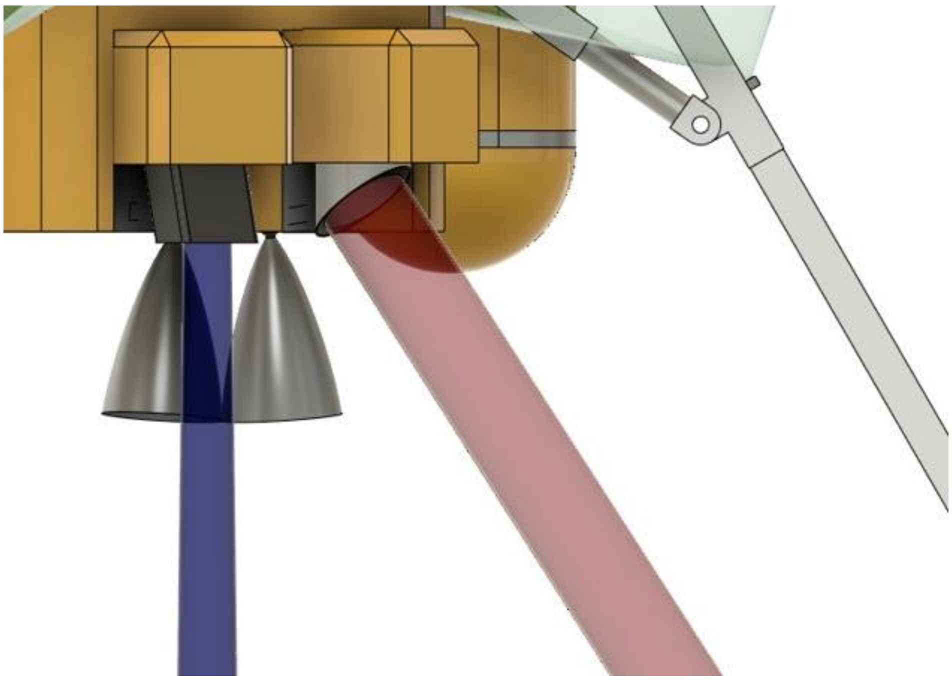


Figure 4.2-1 THEMIS and OTES FOVs

The Radar Imager for Mars' subsurface experiment will be responsible for determining the geologic features below surface after the EEV lands. This system will weigh around 3 kg and have a maximum power usage of 10 watts. It is located on the sample collection arm of the EEV and has a FOV of 2.71 degrees, as can be seen in Figure 4.2-2 as the blue FOV.

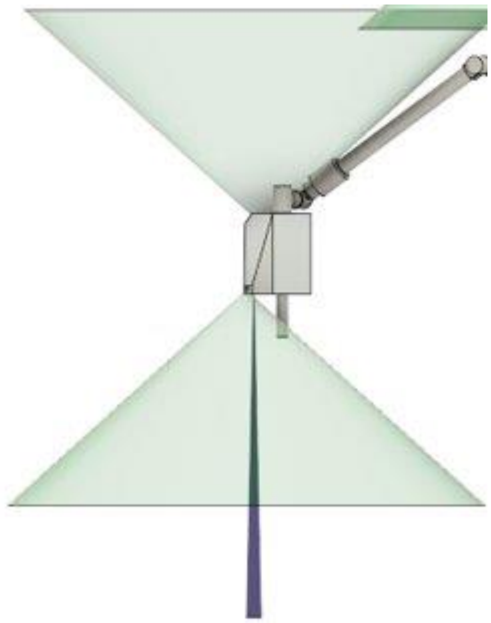


Figure 4.2-2 RIMFAX and Engineering Cameras FOVs

The ECLSS will be responsible for keeping the crew alive, as previously discussed. This system will weigh around 589 kg and have a maximum power usage of 1142 watts. The exterior visual cameras will be responsible for obtaining images of the exterior of the vehicle, as well as keeping the crew away from any hazards they otherwise wouldn't be able to see. They have a FOV of 96 degrees, and their locations can be seen in Figure 4.2-3. This system will weigh around 4 kg and have a maximum power usage of 99 watts.

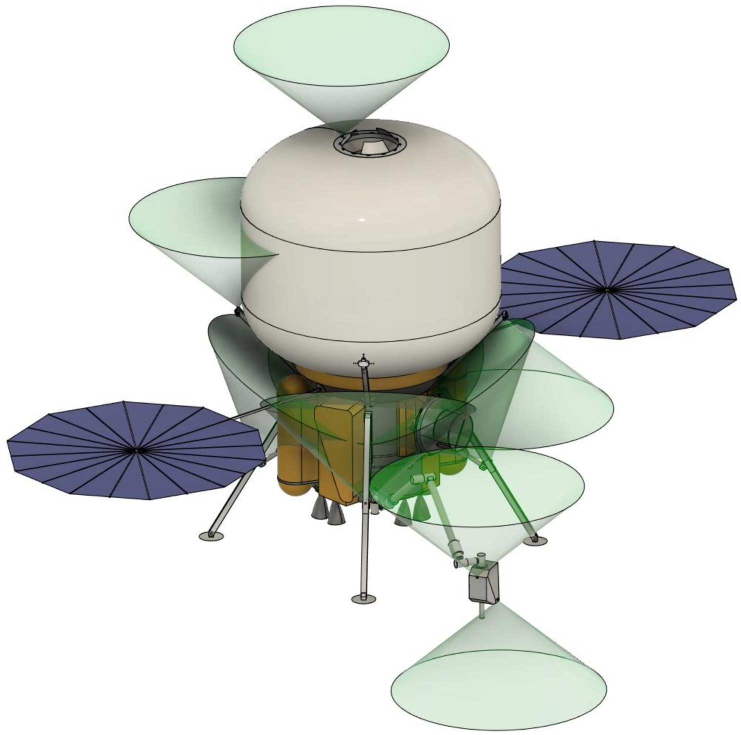


Figure 4.2-3 Engineering Cameras Locations and FOVs

In total, the payload will weigh 933.5 kg and use a maximum power of 1475.8 watts when every system is running at peak power.

4.2.1 Sample Collection Mechanism

The sample collection mechanism on the EEV consists of three primary components: the corer, the robotic arm, and the sample caching system. This is similar to the Perseverance rover and was sized based on that system, scaled up linearly from two different sized variants that were considered by Honeybee Robotics for the mission [2], using their one bit one core system. This is so that the EEV will be able to cache 50 kg of samples from each moon. In order to estimate how much volume is required for this, the average density of each moon was used: 1.88 g/cm^3 for

Phobos and 1.47 g/cm^3 for Deimos. An initial design of the sample containers estimates the interior dimensions at 100 mm in diameter and 360 mm in length. This would mean that 11 samples are required from Phobos and 13 from Deimos. For redundancy in the case of a failed sample collection or insufficient mass collected, extra bits are provided to bring the total to 19 for each moon.

The corer is mounted to the end of the robotic arm and works on the principle of inserting a bit into the surface of the moon and sealing the sample inside. The collected sample cores are then moved into the sample caching system. To ensure that enough sample mass is collected, the IDSS adapter is actuated, and the inertia of the sample caching system can be measured.

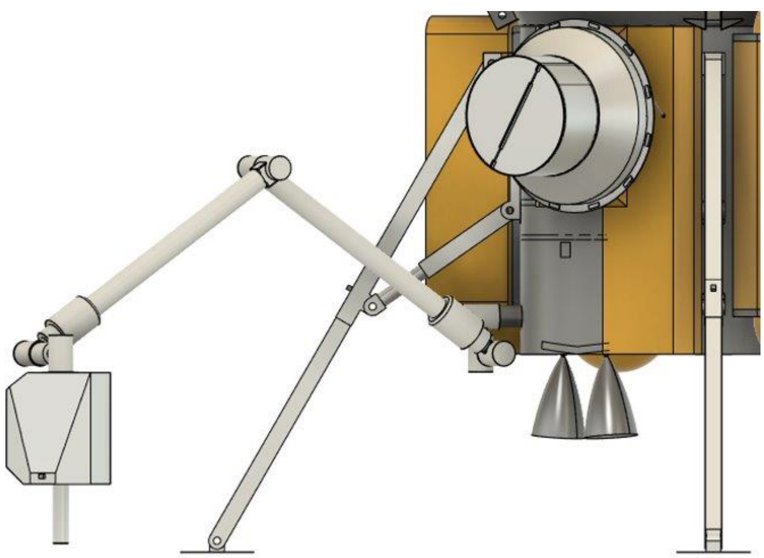


Figure 4.2.1-1: Sample Collection Mechanism

The bits are stored in the sample caching system on launch then removed and replaced one at a time for sample collection. An IDSS docking adapter on linear actuators provides the interface for the sample caching system with the EEV and allows the assembly to be removed for transfer of the samples into the DST. This is done by the robotic arm, which will manipulate it so that the adapter on the sample caching system's end can be attached to the adapter on the DST's end and the crew can open it from the inside to retrieve the samples.

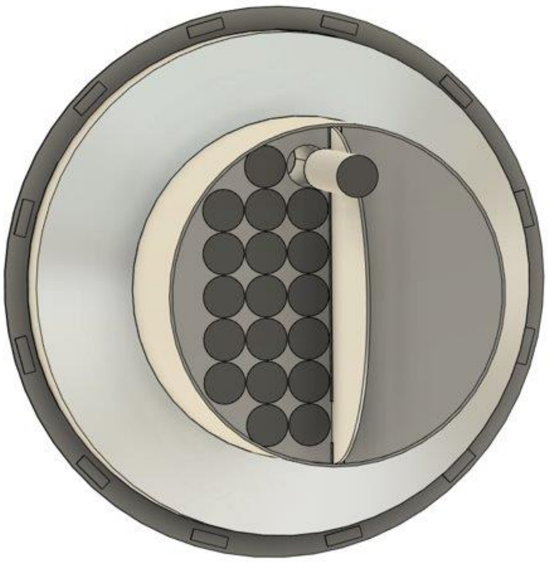


Figure 4.2.1-2: Sample Caching System

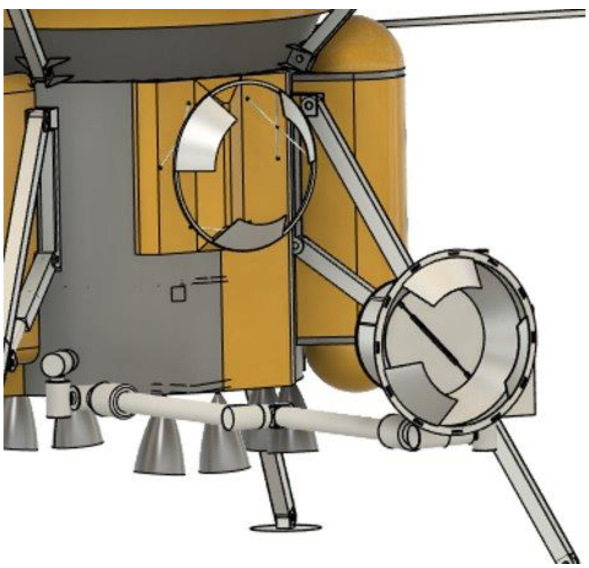


Figure 4.2.1-3: Detached Sample Caching System

4.3 Command and Data Handling

The Command and Data handling (C&DH) subsystem is concerned with ensuring accurate readings from spacecraft sensors, equipping the spacecraft with sufficient processing power, and ensuring that encoding procedures are optimal for smooth data transmissions.

4.3.1 Computers and Storage

In order to meet derived requirement M4.6-1, the BAE RAD750 was selected for the EEV's processing unit. This 21-year-old processor's architecture was designed by IBM and is currently manufactured by BAE systems. It can withstand a maximum of 10,000 grays and has proven itself for use in interplanetary missions (i.e., Mars Perseverance Rover). The processor will be paired with 128MB of dynamic random-access memory (DRAM), 2GB of non-volatile flash memory (NAND), and 256KB of electrically erasable programming read-only memory (EEPROM). This specific memory allocation is also used in the Mars Perseverance Rover. To capture and store the video obtained during the 30-day mission, a Phison PSS4A111-8G 480GB SSD will be equipped. This specific SSD has a built-in Intel Atom processor to aid in photo handling. The 480GB SSD will allow the EEV to store about 25 hours of video or 8000 photos at 4K resolution.

Four of these computers will be integrated into the EEV. One of these computers will be used for attitude determination and engineering data handling, another for payload data handling, and the final two will be incorporated for redundancy. A network onboard the EEV will help connect these computers together to boost efficiency whenever possible. Figure 4.3.1-1 demonstrates the RAD750 and Phison SSD.

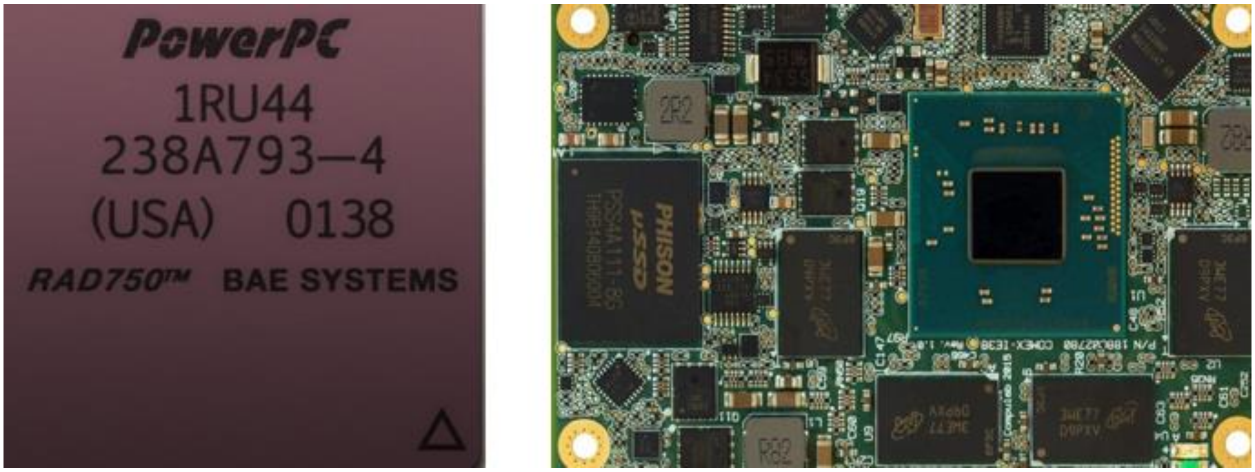


Figure 4.3.1-1 BAE RAD750 (left) and Phison 480GB SSD (right)

4.3.2 Encoding, Overhead Rates, and Quantization Error

Encoding is used to send clean, error-free data to and from the spacecraft. Different encoding schemes exist, some schemes being faster than others. To meet derived requirement M4.6-3, the EEV will encompass encoding schemes involving Reed-Solomon (R-S), Bose-Chadhuri-Hocquehem (BCH), and convolutional coding as these are the recommendations set forth by the Consultive Committee for Space Data Systems (CCSDS). For uplinks (transmission from DST and Earth to EEV), the encoding scheme will be BCH(63, 56). BCH(63, 56) is best optimized to correct multiple errors at a time by increasing the transmission bit rate by a factor of 1.13. For downlinks, (transmission from EEV to DST and Earth), the encoding scheme will be R-S(255, 233) paired with convolutional coding with a 0.5 rate and 7-bit register. In theory, the R-S(255, 223) would be the outer code helping to correct large groups of errors, whereas convolutional coding with be the inner code helping to correct smaller errors. Both R-S(255, 223) and convolutional coding increase the transmission bit rate by a factor of 2.29.

Overhead rates are added to collect relevant sensor data like corresponding dates, access times, run-times, etc. However, the higher the overhead rates, the more corrupted the sensor data can become. Because derived requirement T4.6-2 requests overheads to remain at or under 10%, the maximum overhead included in the EEV will be 10%.

Quantization error is error that stems from the conversion from analog to digital. Prior to spacecraft downlink, data is converted to digital. This conversion introduces quantization error (synonymous with round-off error). To avoid this error, the number of bits per data sample transmitted will be increased to 14. This will bring quantization error down to 0.015%, meeting derived requirement T4.6-3.

4.4 Telecommunications Systems

A telecommunication system is necessary to control and relay data from the spacecraft to ground stations back on Earth. For the design it was decided that the spacecraft should be compatible with NASA's Deep Space Network (DSN) as well as the Deep Space Network (DST). This allows the spacecraft to transmit or receive data at high or low speeds depending on the mission portion.

4.4.1 Ground Stations

The EEV will be communicating with the DSN's 70-m ground stations for some portions of the mission particularly the uncrewed portions. Utilizing the larger 70-m dishes from the DSN complexes allows the EEV to transmit data at high data rates despite the large distance of 78.4 million km. Based on the antennas' transmitting frequencies, the EEV will use the DSN's channel 16.

4.4.2 DST Relay

Like how the Perseverance rover communicates with the Mars Reconnaissance Orbiter, the EEV will use the DST as a relaying system. This will come into play during the crewed portion of the mission, with a much shorter range of 36.3 thousand km. By doing this, the EEV will be capable of transmitting at a rate three times faster than that of the EEV to DSN rate. Using STK, access intervals were found along with how much data can be downlinked and uplinked during those time periods. Table 4.4.2-1 shows an average of the results for the 30-day manned portion of the mission.

Table 4.4.2-1 EEV – DST Access Intervals

Moon	**Location**	**Average Access**	**Max Downlink$^{\Omega}$**	**Max Uplink$^{\Omega}$**
Phobos	Stickney Crater	2.71 hours	19.5 GB	585 KB
Deimos	Swift Crater	9.09 hours	65 GB	1.96 MB

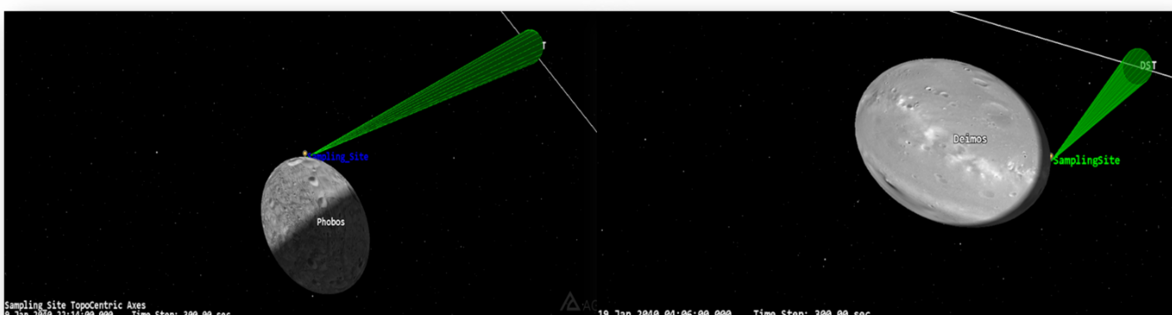


Figure 4.4.2-1 Visualization of EEV – DST Access Intervals for Phobos (Left) and Deimos (Right)

4.4.3 System Configuration

The EEV will utilize a HGA and a LGA both of which operate on the X-band frequency range. The HGA is a 1.3-meter parabolic dish that operates with an 8.42 GHz transmitting frequency and will be exclusively used for downlinking. Capable of communication with the DSN and DST, the HGA will downlink with rates ranging from 0.5 to 2000 kbps depending on the flight mode during the mission. The LGA is a 0.25-meter parabolic dish that operates with a 7.16 GHz and will be used for uplinking commands with rates ranging from 60 to 500 bps. Table 4.4.3-1 shows the four flight modes and their corresponding data rates.

Table 4.4.3-1 Flight Modes and Corresponding Data Rates

Flight Mode	Max Data Sampling Rate, kbps	Max Uplink Rate, bps	Max Downlink Rate, kbps
Sampling	670	60	2000
Mapping	670	62.5	625
OOMS	670	62.5	300
Emergency	670	500	0.5

4.4.4 Downlink and Uplink Rate Results

Looking at the two most important downlinking scenarios of Sampling and Mapping, it was desired to obtain a minimum data link margin of 10 dB. These two cases require the most power for either high speed downlinking or downlinking for a large range. For uplinking during an emergency scenario, it was also desired to obtain a minimum data link margin of 10 dB. Table 4.4.4-1 shows the two downlinking scenarios while Table 4.4.4-2 shows the emergency uplinking scenario.

Table 4.4.4-1 Downlinking Data Rates and Results

DOWNLINK: EEV to DSN			**DOWNLINK: EEV to DST**		
Parameter	**Value**	**Unit**	**Parameter**	**Value**	**Unit**
Transmitting Frequency	8.42	GHz	Transmitting Frequency	8.42	GHz

Range	7.84E7	km	Range	36.3	km
Data transfer rate	625	Kbps	Data transfer rate	2.00	Mbps
Symbols/s	1.43	MSps	Symbols/s	4.57	MSps
$E_b/N_{0,\ achieved}$	14.7	dB	$E_b/N_{0,\ achieved}$	79.9	dB
$E_b/N_{0\ required}$	4.5	dB	$E_b/N_{0\ required}$	4.5	dB
Data Link Margin	10.2	dB	Data Link Margin	75.4	dB
HGA Diameter	**1.35**	**m**	**HGA Diameter**	**1.35**	**m**
HGA Required Power	**60**	**W**	**HGA Required Power**	**60**	**W**

Table 4.4.3-2 Uplinking Data Rate and Result

UPLINK: DSN to EEV		
Parameter	**Value**	**Unit**
Transmitting Frequency	7.16	GHz
Range	7.84 E7	km
Data Transfer Rate	500	bps
Symbols/s	572	Sps
$E_b/N_{0,\ achieved}$	30.5	dB
$E_b/N_{0\ required}$	5.5	dB
Data Link Margin	25	dB
LGA Diameter	**0.25**	**m**

4.4.5 Hardware Table

The telecommunication system utilizes various components that have been used in previous Mars related missions. With the two manufacturers of many of the components being L3Harris or General Dynamics. The subsystem will weigh about 61 kg and require 265 W of power. Table 4.4.5-1 lists the components along with their masses and power requirements.

Table 4.4.5-1 Telecommunication Subsystem Hardware List

Telecommunication Equipment		
Component	**Mass, kg**	**Power, W**
Small Deep Space Transponder	3.2	19.5
Mars Ultra High Frequency Transceiver	3	65
Diplexers	0.6	0
X-Band Solid State Power Amplifier	1.37	60
RF Network Components	5.3	0
RF Network Subsystem	21.8	0
HGA (D=1.4m)	12	60
LGA (D=0.25m)	3	0
Wideband Transmitters	3.5	60
Cabling	7.8	0
Total	**61.2**	**265**

4.5 Thermal System

The EEV's Thermal System had to be designed to meet the temperature requirements found in both Earth and Martian space environments. This temperature range is between -65°C to +125°C for the Earth's space environment in low orbit, and -112°C to -4°C for the 5-SOL Martian parking orbit. However, the EEV's operational temperature is also based on the components' temperature specifications, as shown in Table 4.5-1, and constrains the EEV's temperature limits to an even smaller margin within the space environment temperature ranges. As a result, the EEV's passive thermal control was designed with Multi-layer Insulation (MLI) to maintain an operational temperature range. The MLI is composed of an outer cover that resists shedding, flaking, and particulate generation, acting as the protective layer for the components and insulation. The MLI is further decomposed with reflector layers that reduce incoming exterior emissions, separator layers that create spacings that minimize heat conduction between layers, and an inner cover that reflects interior emissions, insulating the EEV and preventing excessive leakage of heat to the exterior atmosphere. Materials for each of these layers were chosen based on their optimal absorptivity and emissivity as shown in Table 4.5-2. To complement the EEV's passive thermal control system, about 706 W of required powered heating was calculated for the spacecraft's instrumentation and ECLSS to operate within their operational temperature range, while the solar arrays can operate without additional heating. This gives a total thermal system mass of 155 kg, with a detailed breakdown shown in Table 4.5-3.

Table 4.5-1 EEV Component Operational Temperature Ranges

Component	Operational Temperature Range (°C)
Sampling and Caching System	-70 to +70
Thermal Emission Imaging System	-55 to +50
Thermal Emission Spectrometer (THEMIS)	+10 to +40 (Operational) -25 to +55 (Inactive)
Radar Imager	-50 to +50
Exterior Visual Cameras	-55 to +50
ECLSS	19.2 to 23.4

Table 4.5-2 Multilayer Insulation Materials Breakdown

Part	Material	Thickness (mm)	Absorptivity α	Emissivity ε
Outer Cover	Beta Cloth	0.20	0.45	0.80
Reflector Layer	Aluminized Kapton	0.127	0.14	0.05
Separator Layer	Nomex Netting	0.16	-	-
Inner Cover	Aluminized	0.01	-	0.06 (aluminized side) 0.4 (reinforced side)

Table 4.5-3 Thermal System Component Mass Breakdown

Thermal System Components	**Mass, kg**
MLI (12 layers with separators)	3.22
Heaters (installed weight)	53.6
Radiators (250 W rating)	72.6
Paint	25.7
TOTAL	**155**

4.6 Power System

The power subsystem is the lifeline of a spacecraft. As such it is necessary to have a design that will ensure the spacecraft's power needs are met throughout the mission. Previous Mars-related missions have used solar panels or radioisotope thermoelectric generators, while previously manned spacecraft typically used fuel cells. It is with this knowledge that a trade study was conducted between solar arrays, RTGs, and fuel cells.

4.6.1 Power System Trade Study

The three candidates were evaluated with figures of merit based on reliability, power generation, safety, price, and availability. Solar arrays came out as the winning candidate and were utilized for the design of the spacecraft. Figure 4.6.1-1 shows the listed pros and cons that were considered, and Figure 4.6.1-2 shows the scores for the three candidates

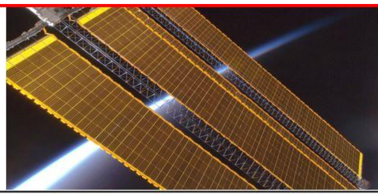
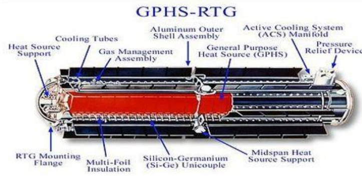
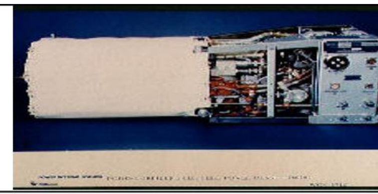
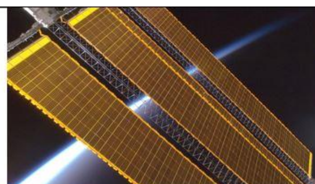
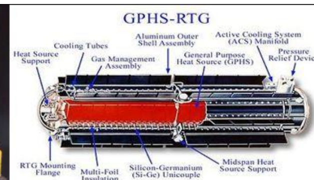
		
Solar Arrays	**Radioisotope Thermoelectric Generator**	**Fuel Cells**
Pros: • Proven heritage & reliability • Large power demands can be met • Can be pointed toward sun for efficient use • Can have numerous configurations Cons: • Must be stowed at launch • No use during an eclipse • Requires deployment & rotation mechanisms • Requires sun sensors for positioning • Requires a power storage system	Pros: • Proven heritage & reliability • Provides continuous long-term power • Eliminates need for Sun on distant missions Cons: • Costly & difficult to handle • Radiation affects instruments & electronics	Pros: • Proven heritage • High power levels available • Water is a byproduct Cons: • Thermal insulation possibilities limit mission duration • Fuel cell chemicals must be stored as cryogenics for volume efficiency

Figure 4.6.1-1 Power System Comparison

Candidates: / Performance:										
		Solar Arrays			**RTGs**			**Fuel Cells**		
		Perf.	**U**	**W**	**Perf.**	**U**	**W**	**Perf.**	**U**	**W**
Reliability	**Wt. = 3**	>1 yr	9	27	>1 yr	9	27	>1 month	9	27
Power Generation (kW)	**Wt. = 3**	0.2-300	9	27	0.2 - 10	9	27	0.2-50	9	27
Safety	**Wt. = 2**	Safe	3	6	Dangerous	0	0	Safe	3	6
Price ($/W)	**Wt. = 2**	800-3,000	9	18	16K – 200K	0	0	50K – 100K	0	0
Availability	**Wt. = 1**	Flight proven	3	3	Flight proven	3	3	Flight proven	3	3
Total		**81**			**57**			**63**		

U = Utility Value: 9 = better than requirement >15%; 3 = within ±15% of meeting requirement; 1 = 16% to 25% worse than requirement; 0 = requirement not met by >25%
W = Weighted Value = U x W; W = 2 means two times more important than the average FOM

Figure 4.6.1-2 Power Candidates and Their Respective Scores

4.6.2 Spacecraft Power Usage

As previously seen from the spacecraft's power statement, the vehicle's max power is 6105 Watts. To better analyze how much power would be required for the mission, a load analysis was conducted for different stages of the mission. The mission is divided into two main portions, those being the uncrewed and crewed portions with the uncrewed consisting of four stages and the crewed consisting of nine. Table 4.6.2-1 shows the different stages along with power usage and margin from the 6105W limit.

Table 4.6.2-1 Power Usage and Margin at Different Stages

Mission Portion	Mission Stage	Power Usage (W)	Margin (%)
Non-Crewed	Deployment/Earth Departure	2947	52
	Transfer Course Corrections	4041	34
	Transfer w/ECLSS Check	4851	21
	Mars Arrival/Aerobraking	3371	45
Crewed	DST Docking	5121	16
	DST Departure Burn	5467	10
	Moon Rendezvous	5467	10
	Moon Scanning	5085	17
	Landing	5871	4
	Sample Collecting	4511	26
	Moon Departure	5537	9
	Moon Transfer	5791	5
	DST Rendezvous	5861	4

4.6.3 Subsystem Configuration

A solar panel surface area of 42.8m^2 will provide the spacecraft with 6105W of power. A battery configuration of 19 parallel and 8 series cells will ensure that the spacecraft can have its peak power usage requirement of 5871 W met during expected eclipse periods. An extra battery cell installed in parallel is added for a battery-out scenario. The batteries will be discharged to a 60% depth of discharge for 10,000 cycles. Table 4.6.3-1 shows a mass overview for the power system's components.

Table 4.6.3-1 Power System Mass Overview

Power System Components	Mass (kg)
Battery	58
Solar Panel Mass	225
Attachment Mass	25
Drive Mechanism	31
Control Electronics	347
Cabling	393
TOTAL	**1023**

4.7 Attitude Determination and Control System

The attitude control system (ACS) is a subsection of the spacecraft that allows for the ability to rotate and adjust spacecraft attitude. Different ACS methods exist, with some requiring no feedback (open-loop), and others requiring sensors to correct error in attitude measurements (closed-loop). The complexity of the ACS depends heavily on the spacecraft mission and payload pointing requirements, as well as the type of maneuvers for a successful mission.

4.7.1 Major ACS Maneuvers and Disturbance Torques

There are several maneuvers required to guarantee mission success. Table 4.7.1-1 lists the maneuvers in order.

Table 4.7.1-1 List of different ACS maneuvers required for mission success

Major Attitude Adjustments Required	Change in Attitude (°)
Retro Thrust to Phobos Orbit	180
Scanning and Landing Repositioning on Phobos	90
Retro Thrust to Deimos Orbit	180

Scanning and Landing Repositioning on Deimos	90
Retro Thrust to 5-SOL Orbit	180
Align EEV Docking Adapter to DST	180

From the table, retro burns require 180° rotations, while scanning and landing on the moon surfaces require a 90° rotation. For example, multiple retro burns (or anti-thrust burns) require the main engine thrusters to point opposite of the direction the spacecraft is traveling (a 180° change in attitude). During moon landing, the spacecraft must position its landing legs perpendicular to the moon surface (a 90° change in attitude). It is important to note that these attitude changes assume the spacecraft body frame z-axis is aligned with the spacecraft direction of motion.

During the 30-day mission, the spacecraft will have its attitude affected by disturbance torques. These disturbance torques, although small, originate from drag, solar pressure, magnetic fields, and gravity acting on different sections of the spacecraft. Table 4.7.1-2 lists the disturbance torques.

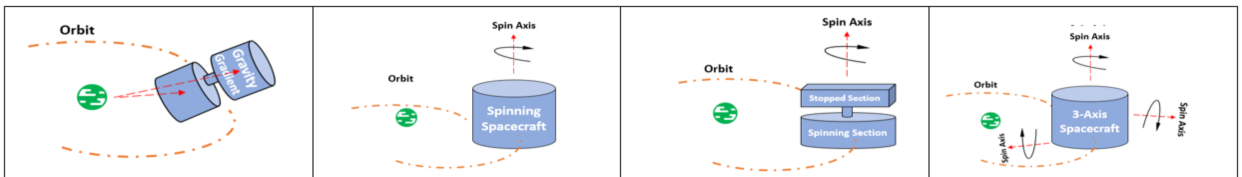
Table 4.7.1-2 List of disturbance torques experienced at Mars

Total Disturbance Torques at Mars	**N-m**
Magnetic Torque	2.31E-08
Gravity-Gradient Torque	1.18E-04
Solar Torque	4.54E-04
Drag Torque	6.13E-04
Total Torque	**1.18E-03**

The listed values represent the highest possible torque at any given point in the spacecraft's trajectory. For example, drag, magnetic, and gravity-gradient torque are greatest when the EEV is nearest to the Martian surface, whereas solar torque is greatest when the EEV is nearest to the Sun.

4.7.2 Comparing Different ACS Methods

Different ACS configurations exist. The most common methods include gravity-gradient stabilized, spin-stabilized, dual-spin stabilized, and three-axis stabilized. The greater the control over all the spacecraft's axes, the more complex and expensive the system becomes. Figure 4.7.2-1 demonstrates a comparison table between the different ACS methods.



Gravity-Gradient Stabilized	Spin-Stabilized	Dual-Spin Stabilized	Three-Axis Stabilized
Pros	**Pros**	**Pros**	**Pros**
• Cheapest System Here	• 2nd Cheapest System Here	• Maneuver Speed (Slow)	• Pointing Accuracy (0.001°)
• Power-Obtainability (90% Eff.)	• Scanning Motion is Inherent	• Scanning Motion is Inherent	• Maneuver Speed (Fastest)
• Most Reliable System Here	• 2nd Most Reliable System Here	• Pointing Accuracy (0.1-1°)	• Power Obtainability (99% Eff.)
• Nadir Pointing (Passive)	• Planetary Mission Capable	• Planetary Mission Capable	• Planetary Mission Capable
Cons	**Cons**	**Cons**	**Cons**
• Pointing Accuracy (5-10°)	• Pointing Accuracy (0.3-1°)	• 3rd Cheapest System Here	• Most Expensive System Here
• Maneuver Speed (Slowest)	• Maneuver Speed (Slow)	• 3rd Most Reliable System Here	• Least Reliable System Here
• Not Planetary Mission Capable	• Power Obtainability (32% Eff.)	• Power Obtainability (32% Eff.)	• No Inherent Scanning Motion

Figure 4.7.2-1 Comparing different ACS methods

In order to meet derived requirements T.4.4-1 and T.4.4-5, the three-axis stabilized system was selected. Derived requirement T.4.4-2 requires the spacecraft to be agile and adaptable to changes in mission sequence. Examples of this agility include attitude changes to perform retro burns and positioning the EEV to land on the moons. Derived requirement T.4.4-5 requires pointing accuracy capabilities that many of the ACS methods can achieve, but most methods do not possess the agility required for a successful mission. For example, the Thermal Emission

Spectrometer has a FOV of 0.23°. Spin-stabilized and dual-spin stabilized both meet this pointing accuracy requirement, but neither can meet requirement T.4.4-2.

4.7.3 Required ACS Equipment

To promote spacecraft agility and adaptability, while also having high pointing accuracy and the ability to correct disturbance torques, thrusters and reaction control wheels are required. To meet derived requirement T4.4-4, thrusters must run using monopropellant hydrazine. This allows for tank sharing between the main engines and ACS engines. Adding reaction wheels allows the spacecraft to meet derived requirements M4.4-1, allowing it to correct disturbance torques that would otherwise cause instability and an unsteady 5-SOL parking orbit.

To measure spacecraft attitude accurately, star trackers are required. Star trackers allow the spacecraft to measure the position of the spacecraft relative to stars. Other attitude measuring sensors include Earth sensors and Sun sensors, however these sensors cannot achieve the accuracy of a star tracker. During the spacecraft's transportation from Earth to Mars, the initial deployment of the spacecraft from the launch vehicle's payload fairing requires another attitude sensor. This additional sensor is needed to obtain initial attitude measurements of the spacecraft relative to Mars. This initial attitude measurement can be accomplished by a three-axis magnetometer. Once the spacecraft understands where it is with respect to Mars, the star trackers will then be used.

A final sensor is required to measure spacecraft body rates. Body rate sensors (or gyroscopes) are used to boost spacecraft pointing capability, while also providing another measurement for use in control laws (body rates). Also, in the event the spacecraft's attitude sensors are not functioning, body rate sensors act as redundancy in determining attitude. This redundancy also works the other way; when the body rate sensors cease to work, attitude sensors can estimate spacecraft body rates.

4.7.4 ACS Equipment Trade Studies and Selection

Trade studies were conducted to select equipment from a variety of brands. Tables 4.7.4-1,2,3,4 showcase trade studies related to the selection of a star tracker, gyroscope, thruster, and three-axis magnetometer.

Table 4.7.4-1 Trade study process for ACS thruster

FOMs / Component	Propellant Consumption/180° Maneuver < 25 Weight = 3		Burn Time/180° Maneuver < 350 seconds Weight = 3		Mass < 0.5 kg Weight = 1		Weighted Total = SUM(W)
	U	W	U	W	U	W	
RD MRM-106F	26 kg	9	100 s	27	2.23 kg	0	36
	3		9		0		
RD MR-106L	24 kg	9	200 s	27	0.59 kg	1	37
	3		9		1		
Moog Monarc-5	17 kg	27	550 s	0	0.49 kg	3	30
	9		0		3		

An ideal ACS thruster is one that meets derived requirement T.4.4-2. The thrusters must be placed in accordance with the spacecraft design, meaning the farthest the thrusters can be located is 2.74 meters away from the spacecraft center of mass. To calculate the required burn time, it was assumed that a pair of thrusters were required to fire to rotate the spacecraft about its highest mass moment of inertia axis. Based on this analysis, the RD MR-106L meets requirement T.4.4-2 and wins the trade study for also being the most efficient option, requiring only 24 kg of propellant for every 180° maneuver.

Table 4.7.4-2 Trade study process for ACS star tracker

FOMs / Component	Max Slew Rate > 2˚/s Weight = 3		FOV > 10˚ x 10˚ Weight = 2		Mass < 0.5 kg Weight = 1		Weighted Total = SUM(W)
	U	W	U	W	U	W	
Ball CT-2020	8˚/s	27	33˚ x 33˚	18	3.00 kg	0	45
	9		3		0		
Rocket Lab ST-16RT2	3˚/s	27	15˚ x 20˚	18	0.158 kg	9	54
	9		9		9		
Vectronic VST-68M	3˚/s	27	14˚ x 14˚	18	0.470 kg	3	48
	9		9		3		

To select an ideal star tracker, max slew rate, FOV, and mass were considered. Max slew rate is the max allowable angular rate of change to prevent the star tracker from producing distorted images and thus poor attitude measurements. A slew rate of 2°/s is common and ideal. Having a larger FOV allows the spacecraft to use multiple stars to measure attitude. A FOV of 10° x 10° is common and optimal. Finally, to reduce spacecraft mass and improve launch vehicle payload mass margin, a mass less than 0.5 kg per tracker is desired. Of the options listed in Table 4.7.4-2, Rocket Lab's ST-16RT2 scored highest.

Table 4.7.4-3 Trade study process for ACS three-axis magnetometer

FOMs / Component	Accuracy ≤ ±0.8 gauss Weight = 3		Range ≥ ±1 gauss Weight = 2		Mass < 0.1 kg Weight = 1		Weighted Total = SUM(W)
	U	W	U	W	U	W	
Honeywell HMR2300	±0.8 gauss	9	±2 gauss	18	0.098 kg	3	30
	3		9		3		
Bartington Spacemag-Lite	±1.0 gauss	3	±0.60 gauss	0	0.094 kg	3	6
	1		0		3		
Meisei 3-Axis Mag	±1.0 gauss	3	±0.64 gauss	0	0.220 kg	0	3
	1		0		0		

In selecting a magnetometer, an important consideration is the distance from Mars to where the spacecraft will arrive in its 5-SOL parking orbit. If the magnetometer does not have sufficient range, the spacecraft may not have the ability to measure attitude after deploying from the launch vehicle fairing, causing severe issues in mission sequence. Therefore, a magnetometer with a range of ±1 gauss was desired. Even if the EEV is within range, it is important to have an accurate initial attitude measurement. For that reason, an accuracy smaller than ±0.8 gauss was desired. Finally, to reduce mass, individual magnetometers with mass less than 0.1 kg were preferred. With that, the Honeywell HMR2300 scored the highest due to its great range and accuracy.

Table 4.7.4-3 Trade study process for ACS gyroscope

FOMs / Component	Sample Rate > 100 Hz Weight = 3		Dynamic Range ≥ ±200˚/s Weight = 2		Mass < 0.15 kg Weight = 1		Weighted Total = SUM(W)
	U	W	U	W	U	W	
Honeywell HG4934SRS	**600 Hz**	27	**±158˚/s**	2	**0.145 kg**	3	**32**
	9		1		3		
POLARIS	**200 Hz**	27	**±400˚/s**	18	**2.00**	0	**45**
	9		9		3		

Table 4.7.4-3 selects an optimal rate sensor (gyroscope). An ideal rate sensor has a high sample rate to allow for quick data collection and a high dynamic range to allow the spacecraft to capture high and low changes in rotation speed. A sample rate of 100 Hz is common, and a dynamic range greater than 200°/s is preferred. The POLARIS rate sensor scored the highest.

4.7.5 ACS Hardware Quantity and Placement

Table 4.7.5-1 shows the list of actuators and thrusters used, as well as the quantity, total power, and total mass. Extra sensors were added for redundancy.

Table 4.7.5-1 Complete list of ACS equipment

Equipment Name	**Equipment Type**	**Amount**	**Total Required Power, W**	**Total Mass, kg**	**Uses**
RD MR-106L	Thruster	12	300	7.1	Adjusts attitude directly
RL ST-16RT2	Star Tracker	2	2	0.32	Measures relative attitude to stars
HW HMR2300	3-Axis Magnetometer	2	0.9	0.2	Initial attitude measurements after deployment from fairing
Vectronic VRW-D-6	Reaction Control Wheel	4	15	0.58	Stabilizes spacecraft during THEMIS operation
POLARIS	Rate Sensor	3	40	8	Measures body rates directly\improves pointing stability
TOTAL			**368**	**16.2**	

To size the number of actuators, is it typical to use 12 or 16 thruster configurations. The spacecraft is more than adequately designed to use 12 thrusters. Also, 2 star trackers and 2 three-axis magnetometers were equipped, 1 as a backup in the event the primary star tracker/magnetometer failed. 4 Vectronic reaction control wheels were also added. Each principal axis would have 1 reaction wheel, with a fourth added as back up and placed 45 degrees between 2 principal axes. Finally, 3 rate sensors were added, 1 for each principal axis. Redundancy was not included here for the rate sensor as body rates can also be found using the star tracker (however having a dedicated sensor to measure body rates produces less noisy feedback, thus improving control law accuracy). Figure 4.7.5 illustrates the placement of the ACS thrusters, star trackers, and gyroscopes.

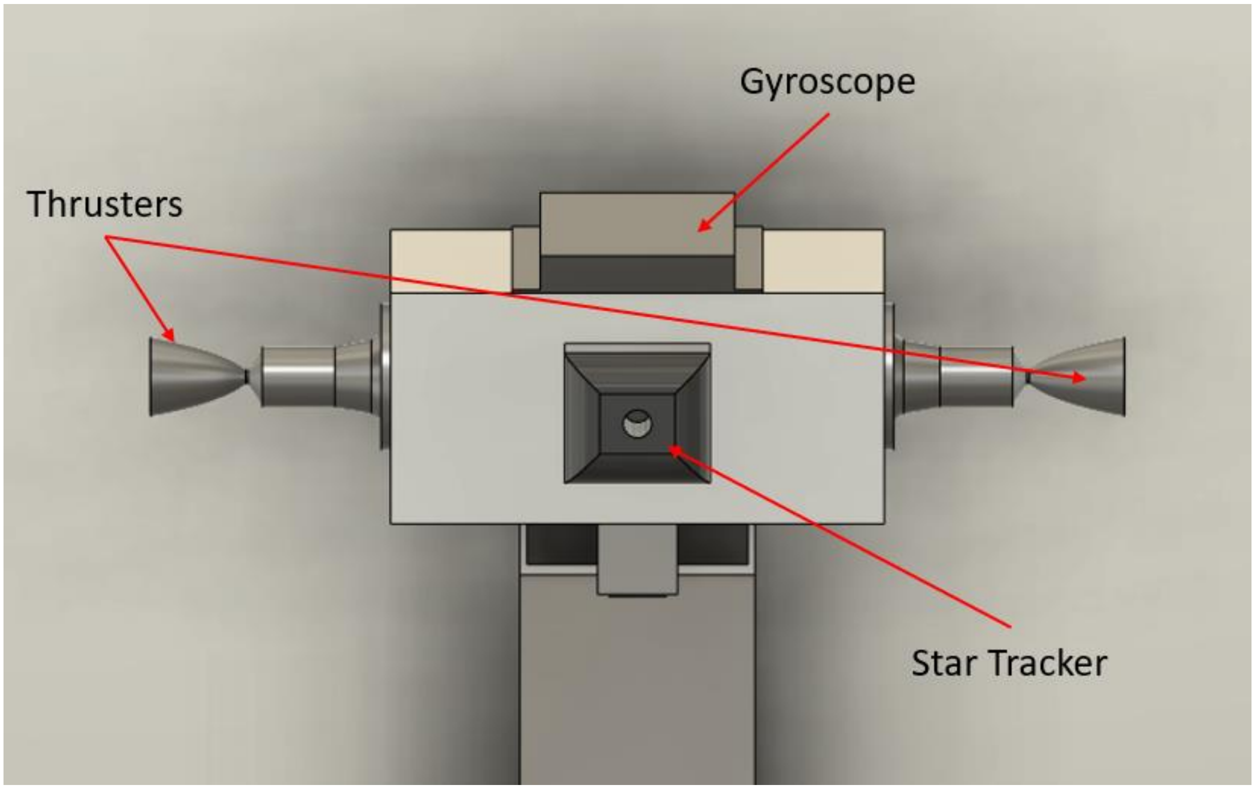


Figure 4.7.5-1 Placement of the thrusters, star tracker, and gyroscope on the EEV

4.7.6 ACS Piping Diagram

To decrease spacecraft mass, derived requirement T4.4-4 was created to allow for propellant tank sharing between the main engine and the ACS thrusters. Figure 4.7.6-1 demonstrates the ACS flow schematic implemented into the spacecraft.

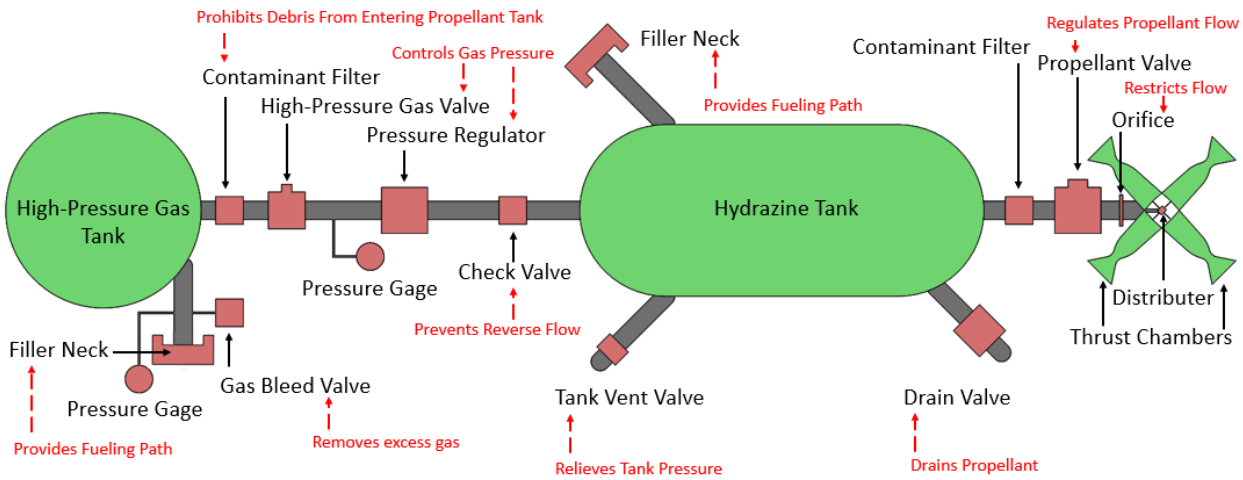


Figure 4.7.6-1 ACS Piping Diagram

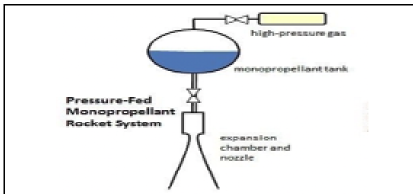
Although the main engines and ACS thrusters share the propellant tank, ACS piping requires a high-pressure gas tank to feed the propellant into the thrust chambers. Between the high-pressure tank and the propellant tank, there is a filter to remove contaminants, a pressure regulator to control gas tank pressure, and a check valve to prevent reverse flow. Both tanks have filler necks to add gas/propellant, and bleed/drain valves to remove any excess gas/propellant. A final contaminant filter is added past the propellant tank, with a final valve to control propellant flow into the thrust chambers.

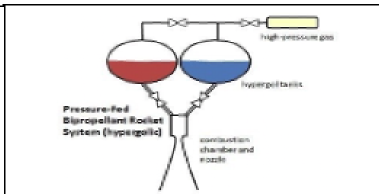
4.8 Propulsions Systems

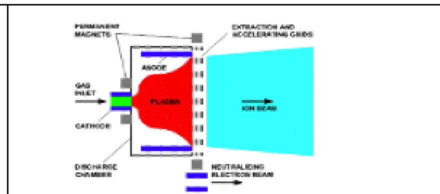
The propulsion system is a subsystem of the EEV. The propulsion system consists of thruster engines, propellant tanks, pressurant tanks, feed lines, propellant pumps, and pressure control valves. The propulsion system of the EEV was designed to remain functional throughout its idle period prior to crew arrival and to meet the customers' needs of a 30-day mission window. Other considerations for the design of the propulsion system were an ability of reigniting its engines, for the orbital insertion maneuver that would need to be performed upon arrival to the Martian SOI and liftoff/rendezvous of Phobos and Deimos.

4.8.1 Propulsion System Classification

Spacecraft generally utilize either monopropellant, bi-propellant, or electric-ion propellant propulsion systems. These three types of propulsion systems were considered and compared for reliability, functionality, and efficiency in Figure 4.8.1-1.







Monopropellant System	Bi-propellant System	Electric Propulsion System

Pros:
• Simple to design
  • High system reliability
  • Low cost
Cons:
• Low efficiency (I_{sp} of ~220 s)
  • Requires more propellant mass than Bi-Prop and electric
• Most existing engines provide low thrust (<500 N)
	Pros:
	• Decent efficiency (I_{sp} of ~300-465 s)
	• High thrust engines available (>2 MN)
	Cons:
	• More complexity due to use of two propellants
	• May require special considerations for propellants due to corrosivity, temperature, or instability
		Pros:
		• Very high efficiency (I_{sp} of ~2000-5000 s)
		• Simple feed system due to use of singular propellant
		Cons:
		• Requires high amounts of electrical power (~1-7 kW)
		• Produces extremely low thrust (~25-250 mN)
		• Have high system mass due to power supplies and high-pressure propellants
		• Trajectory would exceed 30-days

Figure 4.8.1-1 Propulsion System Comparison

From the comparison, the bi-propellent system was chosen as the type of propulsion system to be utilized upon the EEV. This is due to the bi-propellent architectures combination of I_{sp} and thrust levels produced. The low I_{sp} and thrust produced by monopropellant would not be optimal for landing and liftoff of the Martian moon surfaces. The electric propulsion architecture had the highest I_{sp}; However, the low thrust levels would not allow for the mission to be completed within the 30-day mission window. For these reasons, the EEV will be equipped with a bi-propellant system for travel within the Martian SOI.

4.8.2 Thruster Engines

When selecting thruster engines, the attributes that were considered were the thrust-weight ratio, I_{sp}, mass, restart capabilities, and propellant compatibility. Figure 4.8.2-1 compares three engine thrusters that were considered for use on the EEV. The three engine thrusters that were selected for comparison are all designed and manufactured by Aerojet Rocketdyne.

R4-D	AJ10	RL10
Pros: • Large number of restarts • Decent I_{sp} (up to 329 s) • Low mass (5.44 kg) • Storable hypergolic propellants **Cons:** • Low thrust (445 N) • Low T/W ratio (8:1)	**Pros:** • Large number of restarts • Decent T/W ratio (23:1) • Storable hypergolic propellants • Built-in engine gimbal **Cons:** • Heavy (118 kg) • Low I_{sp} (316 s) • Not currently in production	**Pros:** • Very high I_{sp} (450-465 s) • Very high TWR (57:1 for RL10C-1) **Cons:** • Requires cryogenic propellants • Heavy (190 kg)

Figure 4.8.2-1 Thruster Engine Comparison

The R4-D-15 High Performance Apogee Thruster was selected as the main engine for its combination of low mass, I_{sp}, ability for multiple restarts, and compatibility with hypergolic propellants. The EEV is expected to arrive roughly 5 years prior to the crew aboard the DST. To ensure the EEV propulsion system will be functional after an extended period of being dormant in its parking orbit, thruster engines that are compatible with hypergolic propellant types are highly desirable. Hypergolic propellant engines are known to be repeatedly ignited in a reliable manner. Furthermore, hypergolic propellants can be stored at room temperature, eliminating the need for a cryogenic system. Hypergolic fuels and oxidizers also react on contact; therefore, no catalyst would be needed to produce optimal thrust levels. The R-4D engine is specifically compatible with the hypergolic propellant combination of a Monomethylhydrazine (MMH) fuel and Nitrogen Tetroxide (NTO) oxidizer.

4.8.3 Propellant Tanks

Propellant tank design and integration is crucial to the EEV propulsion system. The EEV main propulsion system would need a minimum of three propellant tank volumes: including tanks for the fuel, oxidizer, and pressurant. To determine the volume of propellant needed the spacecraft mass, engine effective exhaust velocity, and required ΔV (per maneuver) were considered. Table 4.8.3-1 shows the calculated ΔV needed per engine ignition, along with the estimated propellant burned for said engine ignition.

Maneuvers	Δv (m/s)	Propellant Mass (kg)
Course Corrections	40	207
Orbital Insertion	120	606
EEV Rendezvous with DST	10	56
Phobos Arrival	763	3584
Phobos Departure	415	1633
Deimos Arrival	332	1159
Deimos Departure	383	1208
EEV Rendezvous with DST	269	763
Margin	**106**	**283**

Table 4.8.3-1 EEV Propellant Budget

The entirety of the mission is estimated to burn 9216 kg of propellant mass of the total 9499 kg initial propellant mass of the EEV. This leaves the EEV with 283 kg of propellant for use in emergency maneuvers. To accommodate this amount of propellant, a total volume of 10.2 m^3 would be needed for the combination of fuel, oxidizer, and pressurant. Of the 10.2 m^3 volume, roughly 30% will be reserved as ullage space. To reduce the complexity of a propulsion system with multiple tank structures and flow paths, the EEV will be designed with a custom propellant tank structure capable of storing the needed volumes of fuel, oxidizer, pressurant, and ACS propellant. The main propellant tank structure will be custom fabricated to house COTS titanium tanks within the structure. The COTS titanium tanks will be equipped with PMDs and used as header tanks. The purpose of integrating header tanks within the main propulsion structure is not only to simplify the propulsion tank integration to the vehicle, but also to help reduce the amount of boil-off and sloshing of the propellants. Furthermore, having header tanks equipped with PMDs with a smaller volume of propellant will ensure the propellent remain readily available in low gravity environments. Figure 4.8.3-1 shows the internal configuration of the propellent tank structure.

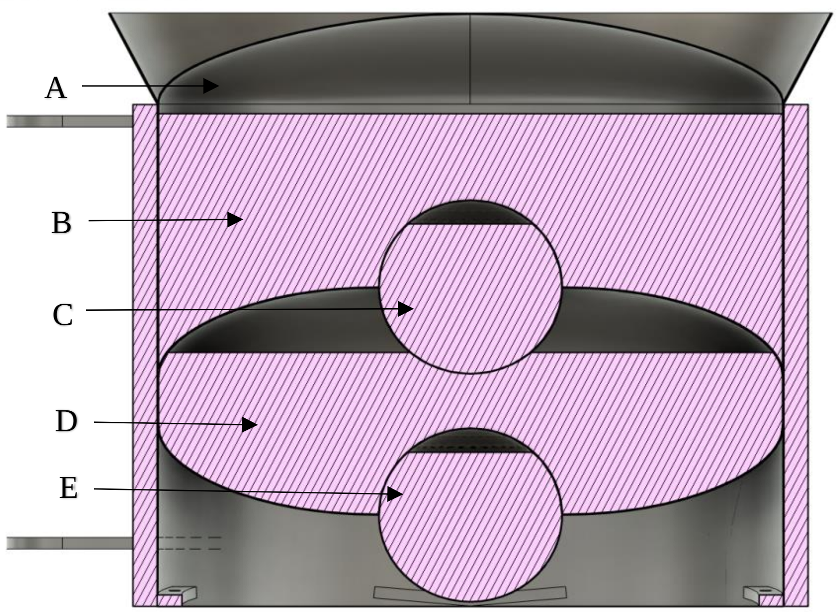


Figure 4.8.3-1 Propellant Structure Internal Configuration

The volume labeled A in figure 4.8.3-1 has been designated to contain the ACS propellant. The sections labeled B and C the volumes for the fuel. Section C is the titanium header tank with

PMDs, which will be used during the initial engine ignitions. Volumes D and E are the volumes that will be used to store the oxidizer for the mission. Section E will be a similar header tank for as the one used in section C of the propellant tank.

4.9 Structures

The spacecraft structure is broken up into two areas, the crew habitation module, and the service module. The former is the primary design driver for the EEV's size, while the latter is sized based on the propellant requirements of the EEV.

The crew habitation module consists of an aluminum support structure and an inflatable exterior. Aluminum 7075-T6 was selected for the internal support structure, as it is lightweight for the provided strength, provides decent radiation protection, and is easy to use for sheet metal manufacturing. The highest loading period is during launch, with maximum loading occurring at 6 g of axial acceleration and 0.5 g of lateral acceleration.

The inflatable portion of the crew habitation module uses Kevlar to restrain the bladders that provide structural rigidity. As the bladders are pressurized to 4 atm, the Kevlar is sized in order to constrain this pressure using hoop stress calculations. A factor of safety of 4 is used to maximize safety in the retention of the structural bladders.

The service module structure is made of stainless steel 304L, which was selected for its strength-to-weight ratio and compatibility with the hypergolic propellants. It will be pressurized throughout the entire mission, up until being depressurized for the decommissioning of the spacecraft and was designed for a maximum pressure of 33 bar. The electronics, solar panels, pressurant tanks, payloads, landing legs, and propulsion systems will all be mounted to the exterior of the service module.

4.9.1 Structural Analysis and Simulations

To ensure the safety of the EEV and crew, structural simulations were conducted on the crew module of the EEV and the propellant tank using SolidWorks static force simulation. The forces estimated in the simulations were calculated using the estimated mass and maximum acceleration of the structures in operation. The crew module was analyzed for the estimated bending moment force of 123,606 N that would be induced during operational maneuvering, as well as the anticipated compressive force of 61,803 N that would occur during launch. The propellant tank structure was analyzed for bending moment force of 13,734 N and a compressive force of 164,808 N. However, the propellent structure was also analyzed with its rated maximum internal pressure of 33 bar. When conducting these simulations, the forces were assumed as point loads in lieu distributed loads. This assumption was made due to the constraints of the utilized simulation software package. In operation, the EEV will be subject to uniform distributed loads rather than point loads. Therefore, this assumption has led the resultant factors of safety to be slightly lower than anticipated when designing the crew module and propellant structure; but still survivable. Figures 4.9.1-1 through 4.9.1-8 show the resultant stress, strain, displacement, and factors of safety plots for the crew module and propellant structure, respectively.

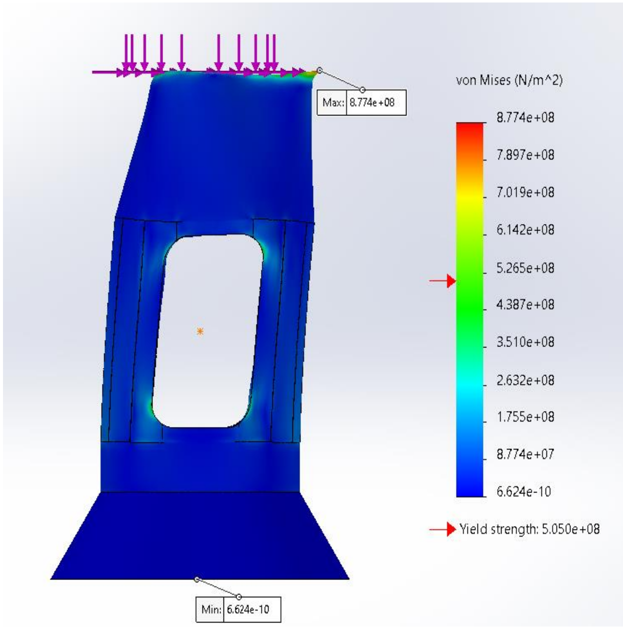


Figure 4.9.2-1 Crew Module Stress Plot

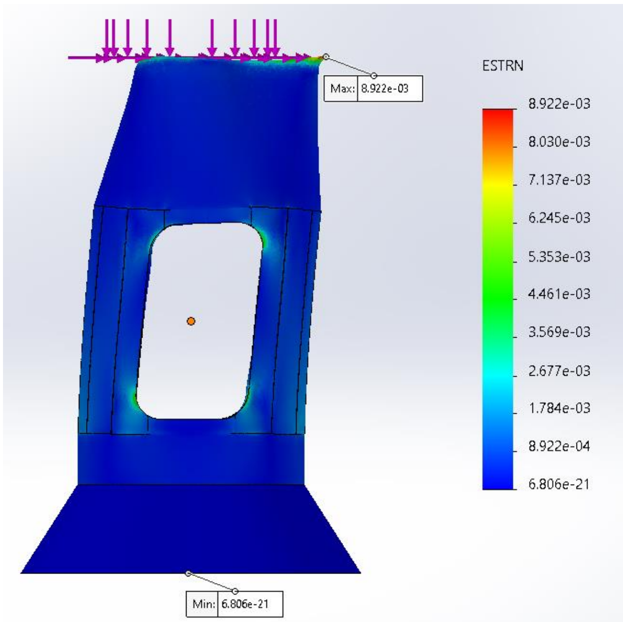


Figure 4.9.2-2 Crew Module Strain Plot

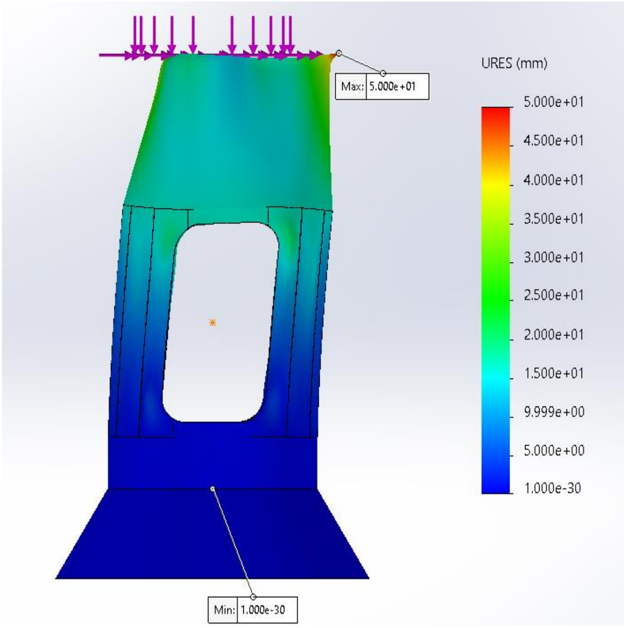


Figure 4.9.2-3 Crew Module Displacement Plot

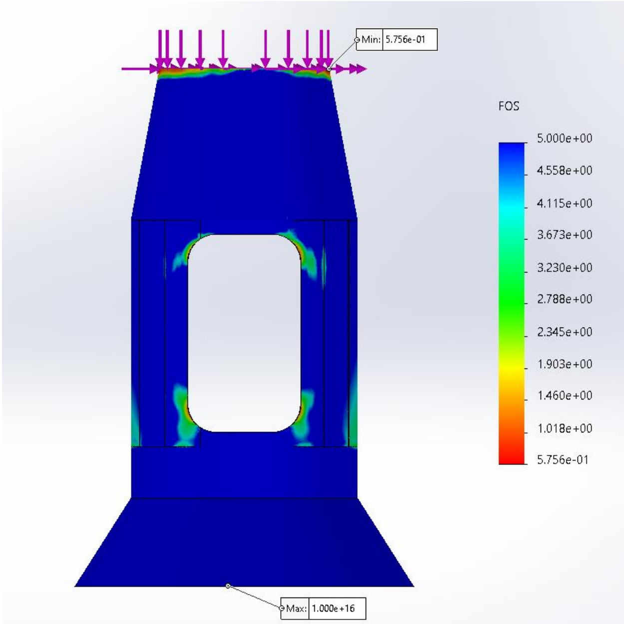


Figure 4.9.2-4 Crew Module Factor of Safety Plot

From the simulation to the crew module, the aluminum structure has the strength to survive the mission. It should be noted that the area near the top of the crew module is a weak point. However, this simulation was done without the docking hatch installed. The installation of the docking hatch will add to the structural integrity of the crew module, reducing the stresses and strains felt on the upper opening of the crew module. Therefore, although the simulation suggests the crew module only has a 58% safety margin, the installation of additional structural components will increase the safety margin.

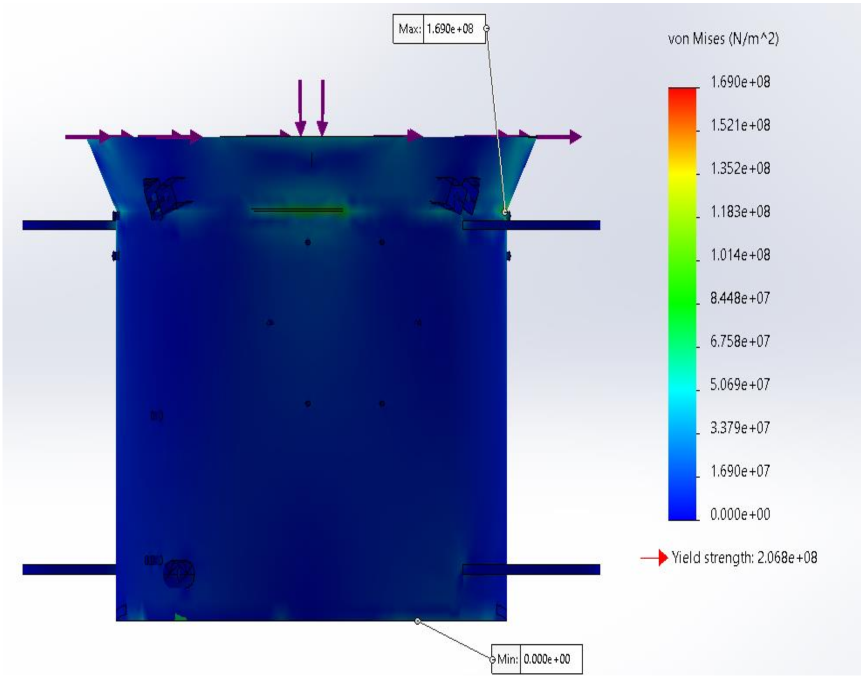


Figure 4.9.2-5 Propellant Structure Stress Plot

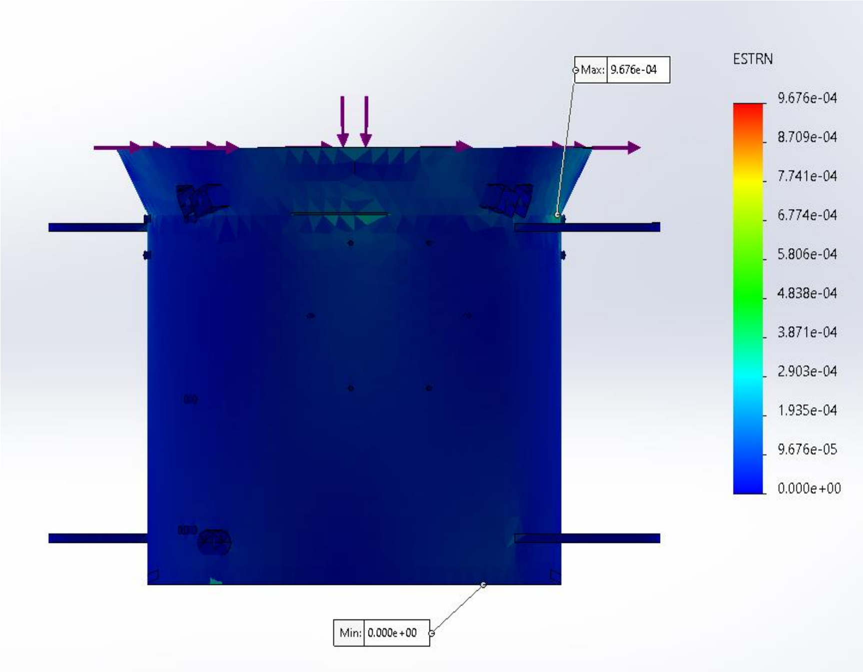


Figure 4.9.2-6 Propellant Structure Strain Plot

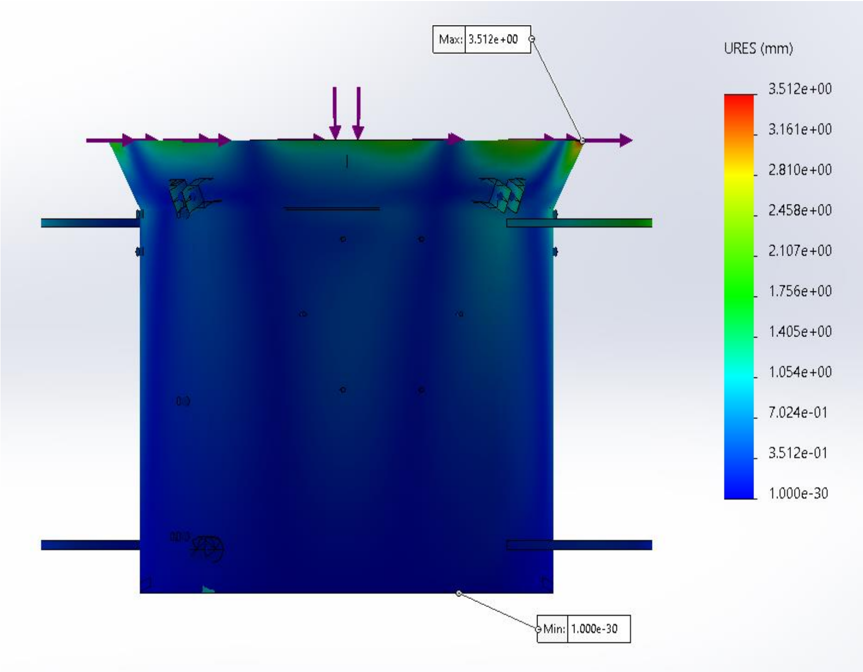


Figure 4.9.2-7 Propellant Structure Displacement Plot

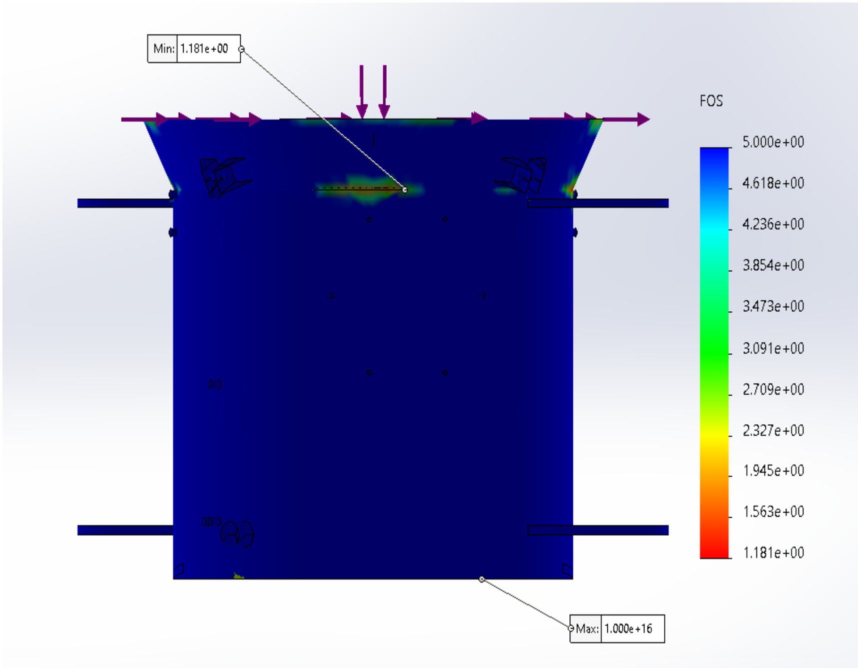


Figure 4.9.2-8 Propellant Structure Factor of Safety Plot

From the analysis of the stainless steel propellent structure, the propellant structure has been designed to withstand loads 118% higher than are expected for the EEV to encounter. The area of the propellent structure that is most likely to fail is the top-center of the structure. However, in operation, the forces will be more uniformly distributed. Therefore, the tank is estimated to be safer than the analysis suggests.

5.0 Systems

5.1 Disposal

Before the EEV can be disposed of after the mission is over, there is one thing that needs to be done first, passivation. Passivation is a very important part of the disposal process since the EEV will have components that can explode at any moment with time. Extraterrestrial Solutions was aware and followed the Inter-Agency Space Debris Coordination Committee guidelines there are for EOM disposal in outer space. To be within guidelines the propulsion system and power system were passivated. For the power system, there was built-in ground safety disconnect relays that prevent any discharging and recharging of the lithium batteries. The solar arrays will also be completely disconnected for safety redundancy to prevent any power going into the power bus. As for the propulsion system all propellants and pressurants will be depleted by either burning or venting. The EEV will then be left in the parking orbit of Mars where it will stay there for the remainder of time. Ready to be refueled and powered for any other mission revolving Mars.

5.2 Risk Analysis

In analyzing potential risks to the EEV, a failure modes and effects analysis (FMEA) was conducted. This was done through a bottom-up approach, where each individual component is analyzed in determining what effect it will have on the EEV. The FMEA analyzes the severity, chance of occurrence, and detectability of each potential failure mode for the components on the EEV. These are then used to calculate a risk priority number, which is used to categorize the risk posed by the failure mode as low, medium, or high. For medium and high-risk components, mitigation steps are implemented into the design. Note, since this is a manned mission, the

Environmental Control and Life Support System (ECLSS) needs to be included in the FMEA process.

Severity relates to the level of severity that each failure mode has and its effect on the mission, while considering the mitigation steps that need to be taken. This does not necessarily mean that one failure mode will have a direct effect on other subsystems, but it does consider how close the mission will be to failure, if the threat is not minimized or eliminated. It also considers what effect it has on its surrounding subsystems if the failure mode was to occur.

Occurrence relates to the level of likelihood that a failure mode can occur, while also considering mitigation steps that need to be taken. Although there are levels in likelihood that a failure mode can occur, this deals with the failure mode itself and not the effects on the mission. That is, how frequently can this failure mode happen. To determine the frequency of the failure mode happening, the team needs to find out what could cause the failure mode to occur. For example, if there is an excess of propellant flow, due to the propellant valve failing to shut off, then the cause would be a propellant valve control malfunction. Another example would be if the tank is over pressurized, due to the tank vent valve failing to open, the cause would be a gas valve control malfunction.

Detectability relates to how likely it is for a crew member to detect whether a failure mode has occurred once again while considering any mitigation steps that need to be taken. As for scoring, the lower the Detectability scores, the more likely the failure mode is to be noticed, thus causing the failure to be dealt with and minimized.

As there are multiple steps throughout the mission, the process step is an important inclusion in the FMEA, as it will determine the presence of crew or other important mission parameters.

Table 5.2-1: Process Step Configurations

	Process Step
1	ACS Firing and Control
2	Tank Pressurization
3	EEV Docking
4	EEV Landing
5	Life Support
6	Communications System Operation
7	Power Supply
8	EEV Computers
9	(RIMFAX) Radar Imager
10	Operation
11	Exterior Camera Operation
12	Thermal Emission Spectrometer Operation
13	Visual Emission Spectrometer Operation
14	Thermal Emission Imaging System (THEMIS) Operation
15	Sample Collection Operation
16	Mission Configuration Deployment
17	Shielding
18	Solar Array Deployment

Using the failure mode, it was determined how each subsystem could fail during the applicable process. This mode may contain more than one way that the subsystem can fail. If that is the case, conduct separate analysis for each failure mode. Conducting separate analysis can minimize alternate risks that may otherwise be overseen.

As mentioned earlier, RPN was determined using the four steps in analyzing a failure mode for each component. It does not consider potential risk mitigation methods that can be used to minimize the risk of mission failure. An equation was used to determine the value for RPN, which took the product of the severity, occurrence, and detectability. Each of the three attributes are rated from 1 to 4, with the lowest possible score for RPN being 1 and the maximum being 64. Ideally, this value should be the lowest possible, but the higher the score, the more critical the failure mode becomes to the mission. Below is the method for determining the value for the first Risk Priority Number, which is calculated as follows:

RPN = (Severity) x (Occurrence) x (Detectability)

The product of these three categories determines an accurate score for each failure mode under RPN and was deemed necessary to allocate the adequate amount of mitigation for the most important RPN. The following will show how each component in the equation was determined (i.e., Severity, Occurrence, and Detectability).

5.3 Cost Analysis

5.3.1 NASA PCEC Tool

The preliminary cost estimate for the EEV was done using the NASA Price Cost Estimating Capability (PCEC) Tool. This tool works by implementing cost estimating relationships (CER) into a Microsoft Excel add-in. NASA does this by compiling its entire mission history and using known parameters. The estimation is started by choosing a pre-built work breakdown structure, and for this mission, the "Crewed" option was selected. Once a pre-built work breakdown structure is selected, the WBS can be customized based on what subsystems the EEV will use. After this, the masses of each subsystem are manually inserted, as the cost is calculated on a first-pound cost basis. Using this process, a cost estimate was found for the EEV, as seen below in Table 5.3.1-1.

Table 5.3.1-1 EEV Price Cost Estimation

EEV Price Cost Estimation ($M, FY 2015)						
Item Name/Description	**DDT&E**	**Design & Development**	**System Test Hardware**	**Flight Unit**	**Production**	**TOTAL**
EEV	$1439.5	$1,272.6	$166.9	$166.4	$166.4	**$1,605.9**
Crewed Vehicle Management	$97.1	$97.1	$-	$13.7	$13.7	$110.8
Crewed Vehicle Systems Engineering	$149.1	$149.1	$-	$18.9	$18.9	$168
Crewed Vehicle	**$817.3**	**$650.4**	**$166.9**	**$111.1**	**$111.1**	**$928.4**
Primary Crew Structures	$401.4	$325.7	$75.7	$58.3	$58.3	$459.7
Structures/Mechanisms	$401.4	$325.7	$75.7	$58.3	$58.3	$459.7
Adapters	$36.9	$27.8	$9	$7	$7	$43.8
Thermal Control	$23.5	$21.5	$2	$1.6	$1.6	$25.1
Propulsion	$25.7	-	$25.7	$2.5	$2.5	$28.2
Liquid Engines	$25.7	-	$25.7	$2.5	$2.5	$28.2
Avionics	**$41.8**	**$29.7**	**$12.1**	**$9.3**	**$9.3**	**$51.1**
Guidance, Nav, & Control	$7.4	$5.8	$1.6	$1.2	$1.2	$8.6
Telemetry & Tracking	$15.4	$10.5	$4.9	$3.8	$3.8	$19.2
CDH	$19	$13.4	$5.6	$4.3	$4.3	$23.4
Electric Power	**$236.7**	**$218.1**	**$18.7**	**$14.4**	**$14.4**	**$251.1**
Crew Systems	$51.2	$27.7	$23.6	$18.1	$18.1	$69.4
Integration, Assembly, Checkout	$21.6	$21.6	$-	$31.2	$31.2	$61.8
System Test Operations	$118.8	$118.8	$-	$-	$-	$118.8
Ground Segment	$235.7	$235.7	$-	$-	$-	$235.7
Ground/Test Support Equip	$235.7	$235.7	$-	$-	$-	$235.7

Highlighted and bolded in the table are the cost drivers for this estimation, which are the structures, avionics, and the electrical power of the EEV. These are essential for the success of the mission. This program did not take into account the cost of the payload or launch vehicle, so the price for those two was manually added to the total cost afterward. The disposal of the EEV was also not calculated by the program but is assumed to cost $0 as it will only consist of passivation as discussed in Section 5.4. The total cost for the EEV after inflation comes out to $2.07 billion. This is 107.2% over the RFP's budget of $1 billion, meaning requirement SLR C0.0-1 is not met.

5.3.2 Budget Increase Proposition

Because the preliminary cost estimate is extremely over budget, a budget increase proposition is being made. This will be the first crewed mission outside of Earth's sphere of influence. Robotic missions going to Mars are generally over 1 billion dollars, as you can see in Table 5.3.2-1. After some research, it was found that crewed missions are about 3 times more expensive than robotic missions, because of factors such as more space needed to accommodate the crew and the higher levels of safety that are required. So, if an unmanned mission to Mars is already over $1 billion, it seems unfeasible to stay under a budget of $1 billion when there will be a crew on board. Thus, a request for at least a 2.5-billion-dollar budget that will account for some margin in order to achieve a successful mission.

Table 5.3.2-1 Cost of Similar Missions in 2021

Cost of Similar Vehicles in 2021 (M)	
Perseverance Rover	$2,959
Apollo	$1,998
Phobos 1 and 2	$1,035
Mars Reconnaissance Orbiter	$907

5.4 Compliance Matrix

Table 5.4-1: System Level Requirements Compliance Matrix

Reference ID	Requirement	Compliance
T1.2-1	The EEV must support two crew members when visiting both moons	Yes
T1.1-1	The total mission duration must not exceed 30 days following departure from the DST	Yes

T1.3-1	The EEV must be able to collect at least 50 kg of samples from each moon	Yes
T1.2-2	The crew must remain inside the EEV for the entire mission duration	Yes
T1.3-2	The mission must produce significant scientific data for the moons	Yes
T1.3-3	The samples must be quarantined from the crew until arrival at Earth	Yes
T1.1-2	The EEV must autonomously dock with the DST	Yes
T0.0-1	The EEV must launch on an existing launch vehicle	Yes
C0.0-1	The vehicle and its launch cost shall not exceed $1 billion (2021 equivalent)	No
M1.1-1	The EEV must be in a 5-sol parking orbit around Mars by 01/01/2040	Yes

6.0 Conclusion

The ETS designed EEV is compliant with all technical and managerial requirements but is projected to run overbudget. Because of this, a request is being made to either increase the budget allotment for the mission to $2.5 billion or to remove the requirement for the EEV itself to be crewed. A robotic mission has been found to still meet the engineering and scientific requirements of the mission if crew is still on board the DST, and thus should be considered in the place of the crewed EEV.

However, the crewed EEV design will still have every capability required and will be ready for launch by 2035. This will use as much existing hardware as possible in order to reduce the cost and development time of the vehicle, especially in the manner of the payload components. High TRL concepts are also implemented throughout the entire design, as flight proven technologies have been utilized throughout the entire design. This will ensure the lowest possible risk categories on the mission.

A full plan has also been devised for the mission throughout integration, launch, operations, and disposal. This will not only ensure mission safety, but planetary protection, that will ensure the quality of these samples and any future samples from the Martian moons.

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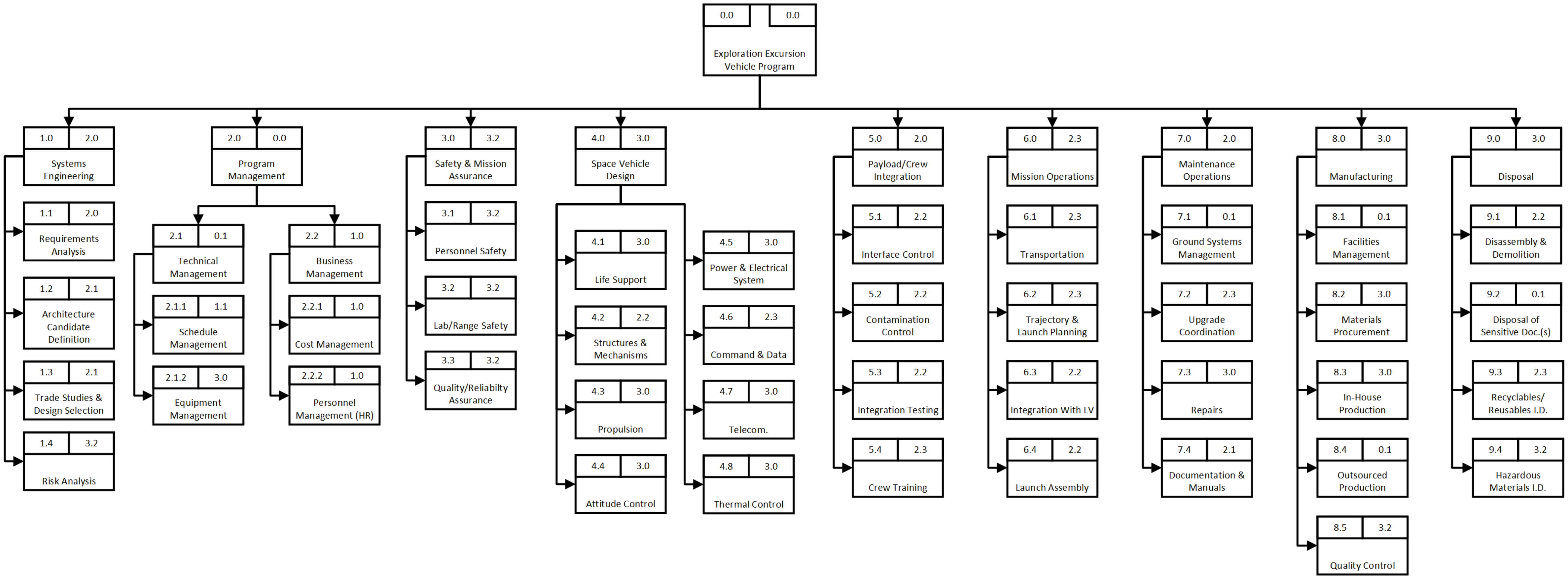
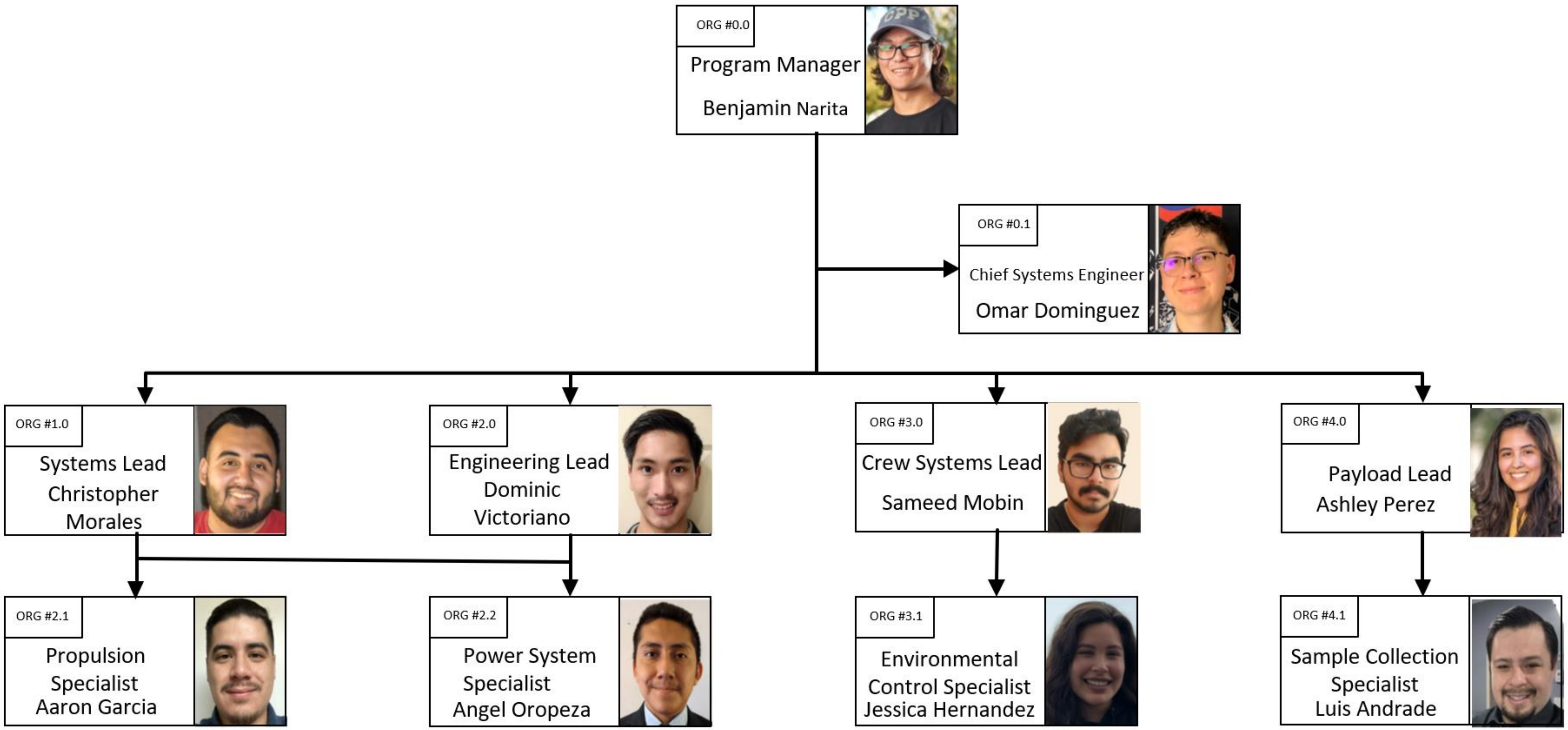
Appendix A

Functional WBS



Organizational WBS



Appendix B

System Level Requirements

Reference ID	Requirement
T1.2-1	The EEV must support two crew members when visiting both moons
T1.1-1	The total mission duration must not exceed 30 days following departure from the DST
T1.3-1	The EEV must be able to collect at least 50 kg of samples from each moon
T1.2-2	The crew must remain inside the EEV for the entire mission duration
T1.3-2	The mission must produce significant scientific data for the moons
T1.3-3	The samples must be quarantined from the crew until arrival at Earth
T1.1-2	The EEV must autonomously dock with the DST
T0.0-1	The EEV must launch on an existing launch vehicle
C0.0-1	The vehicle and its launch cost shall not exceed $1 billion (2021 equivalent)
M1.1-1	The EEV must be in a 5-sol parking orbit around Mars by 01/01/2040

Derived Requirements

Reference ID	Requirement
M1.0-1	The EEV must be compliant with Planetary Protection requirements
T1.1-3	The EEV will be left in the Mars 5-Sol parking orbit when mission is complete.
T1.0-1	The EEV must be capable of withstanding extreme temperatures, debris, and radiation while in the 5-Sol parking orbit and during the Martian moon mission
M3.1-1	The EEV must abort mission and return to the DST immediately in the event of main power source failure
M3.1-2	The crew shall return to Earth in good mental and physical health with minimized significant impacts to their lifelong health
T3.1-1	The crew shall not be exposed more than 0.2 roentgen per day
T3.1-2	The crew shall experience no more than 1% bone loss by mass during the mission
T3.1-3	Human inhabited environment shall have a maximum average sound intensity of 75 dB and have a maximum sound intensity exposure of 115 dB for 2 minutes per 24 hours
M3.1-3	Medical emergencies shall be treated according to NASA STD 3001 4.1.1.6.5

M3.1-4	Crash cart shall provide tools and components to support 2 crew members at a minimum of one event per 1 months
M3.1-5	Food shall be rationed to provide at minimum of 2000 calories per crew member per day for the 30-day mission duration
T4.0-2	The EEV shall be command-able by the crew in the event of a loss in communications
T4.1-2	Oxygen system shall supply a minimum 2 kg of breathable oxygen per day
T4.1-3	CO2 scrubbing shall be capable of removing a minimum of 3 kg of CO2 per day
T4.1-4	System shall be able to safely isolate and store a minimum of 2 kg of waste per day
T4.1-5	Human inhabited environment shall have an average internal pressure of 1 atm +/- 0.025 atm
T4.1-6	Crew exposure to ionizing radiation shall not exceed 3 percent Risk of Exposure-Induced Death (REID) for cancer mortality at a 95 percent confidence level to limit the cumulative effective dose (in units of Sievert) received by an astronaut throughout their career
T4.1-7	Human inhabited environment shall have a humidity range from 50% - 60% relative humidity
T4.1-8	Spacecraft shall detect and extinguish fires autonomously within a maximum of 1 minute
T4.1-9	Water recovery system shall be capable of recycling a minimum of 90% of the water contained in feces, urine, and air
T4.2-1	The EEV shall be able to withstand all induced forces (i.e. from launch, in-space maneuvers, DST docking, and landing of moon surfaces)
T4.2-2	The EEV shall have a vibration dampening system to alleviate acoustic, launching, landing, and other maneuvering loads that can bring vibrational patterns to a structurally dangerous level
T4.2-3	Human inhabited environment shall have a maximum average vibrational rating of 0.025 g and a maximum vibrational frequency of 0.05 g
T4.3-1	The EEV shall have a propulsion system capable of completing the 30-day mission and maneuvering in the Martian SOI
T4.3-2	The EEV shall have a propulsion system capable of providing at least 0.01 m/s^2 of acceleration to liftoff from the Martian moon surfaces.
T4.4-1	The EEV shall be equipped with an attitude determination and control system
M4.4-1	The EEV shall maintain a steady Mars 5-Sol parking until arrival of the DST
T4.4-1	The EEV shall be equipped with an attitude determination and control system
T4.4-2	The EEV’s attitude control system shall be able to rotate the spacecraft 180° in under 350 seconds
T4.4-3	The EEV’s attitude control systems shall detect and correct shifts in attitude smaller than 2°
T4.4-4	Any equipped EEV attitude control system thruster(s) shall run using monopropellant hydrazine
T4.4-5	The EEV’s attitude control system shall have pointing-accuracy smaller than 0.1°

T4.5-1	The EEV shall be capable of supplying 90% power usage during a 30-day period
T4.5-2	The EEV shall incorporate power systems with TRL 7 or higher to minimize potential opportunities for failure and LOC
T4.5-3	The EEV shall incorporate redundancy in its power system to allow a continued power usage of 35% of total power during a 30-day period in the event the main source of power fails
T4.5-4	The EEV must never experience momentary power outages that last more than 200 milliseconds
T4.6-1	The EEV shall incorporate a redundant communications system in order to transmit engineering data at a minimum bit rate of 500 bits per second
T4.6-2	The EEV shall have the capability to return to the DST autonomously in the event of a 3-day loss in communications
T4.6-3	The EEV shall have 3 control options, autonomous, from the EEV crew, and control by a remote crew
T4.7-1	The EEV shall be capable of communication with the DSN at a range of 78.4 million km with a minimum data rate of 500 000 bits/second
M4.7-1	The EEB shall have the ability to communicate with the DST
M4.6-1	The EEV shall be equipped with radiation-hardened computer systems
M4.6-2	The EEV shall incorporate 2 redundant computer systems for use in attitude determination and data handling.
M4.6-3	The EEV's command and data handling system shall use recommended encoding procedure for uplinks and downlinks provided by the Consultative Committee for Space Data Systems (CCSDS).
T4.6-1	The EEV shall have the ability store a minimum of 400 GB of photos and videos captured throughout its 30-day mission.
T4.6-3	The EEV's command and data handling system shall reduce quantization error to less than 0.5%.
T4.8-1	The EEV shall utilize localized thermal control methods to maintain operating conditions for each instrument
T5.0-2	The EEV shall be capable of determining the locations of potential landing sites
T5.0-3	The EEV shall be capable of determining the surface composition of potential landing sites
T5.0-4	The external instruments shall be able to withstand temperature variations
T5.0-5	The EEV shall have an inhabited volume for the crew that maintains a temperature of 18.3 C to 26.7 C
T5.0-6	All instruments must have a reliability of 1/1000 chance of failure
T5.0-7	The EEV shall be capable of probing a minimum of 10 meters under the Martian moons' surfaces
T5.0-8	The sample collection instrument must be able to accommodate a variety of sample types (i.e., soil, rocks, pebbles, and chip samples)
T5.0-9	The sample storage container shall be able to store 0.2 m^3 in samples
T5.0-10	The EEV shall be capable of collecting 50 kg of samples from a moon in a maximum of 7 days

T5.1-1	The sealing station shall be removable to transfer collected samples from the EEV to the DST for transport back to Earth.
T5.2-1	The EEV shall be equipped with a sample sealing station to prevent contamination.
M5.2-1	Human inhabited environment shall be disinfected at a minimum frequency of once every 8 hours
M5.2-2	Personal sanitation items shall be provided to disinfect targeted areas of contamination with up to 99.99% disinfection rates
M5.2-3	Any food sourced for the mission must be irradiated according to NASA STD 3001B
M5.4-1	Flight crew shall be trained and designated as crew medical officers (CMOs) according to NASA STD 3001
T6.3-1	The EEV shall have a launch configuration that fits within the payload fairing of an existing launch vehicle