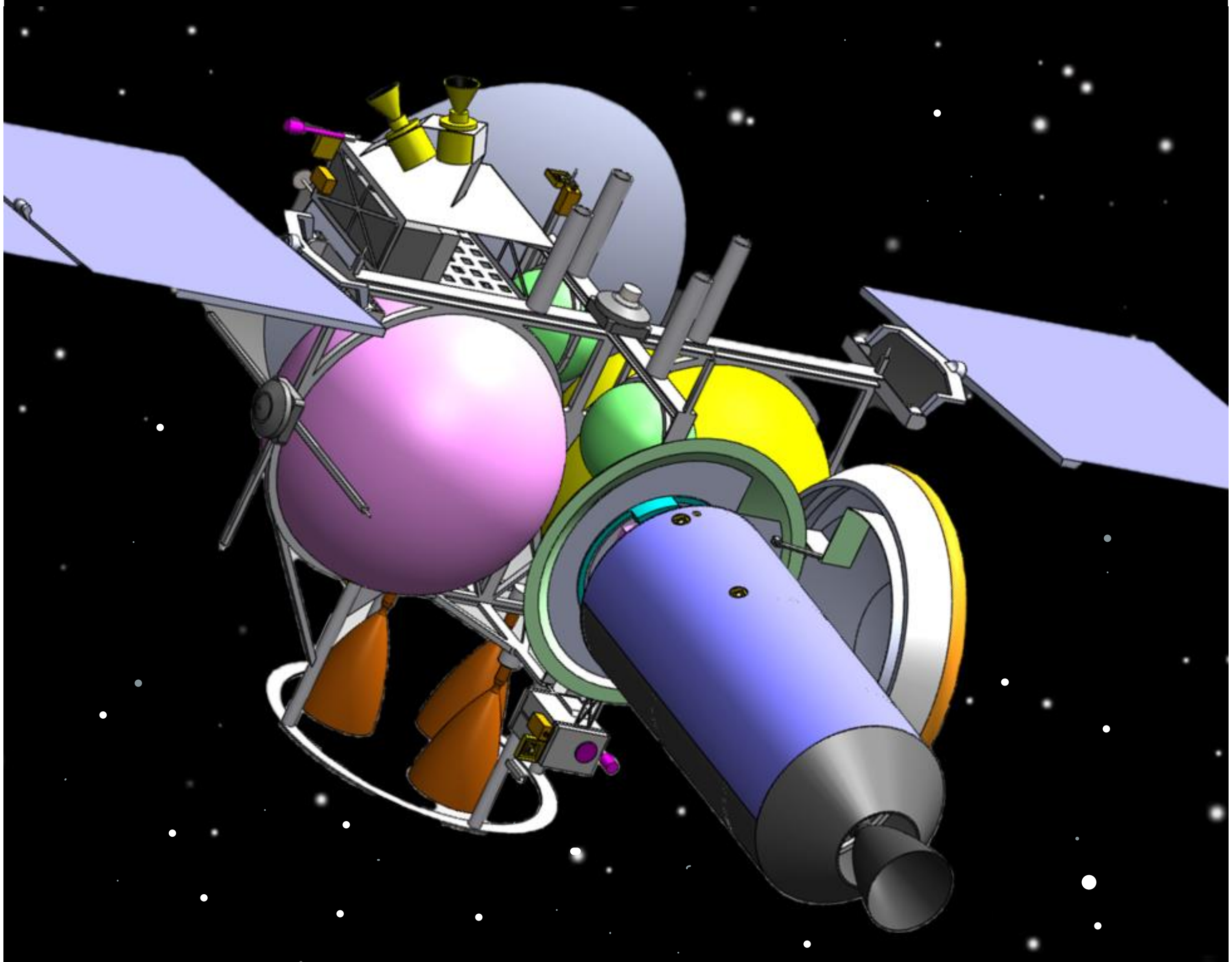

In Response to AIAA Mars Ice Core Sample Return Design Competition

Mars Ice Core Key Exploration Yacht

MICKEY

California State Polytechnic University, Pomona
Department of Aerospace Engineering

Team Organization Chart

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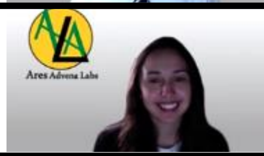	**Structures Deputy** Cheyenne Gonzalez AIAA ID: 1206895
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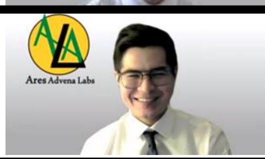
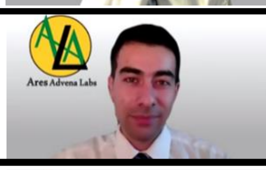
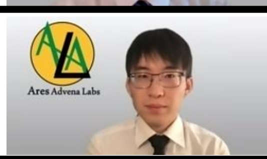
	**Business Lead** Kevin Finney AIAA ID: 1230738
	**CAD Lead** Adrian Torres AIAA ID: 985687
	**CAD Deputy** Jingfeng Di AIAA ID: 1181311
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List of Acronyms

ACS	Attitude Control System
AR	Aeroject Rocketdynbe
BIG MACT	BIG Mars Ascent ConcepT
CAD	Computer Aided Design
COSPAR	Committee of Space Research
COTS	Commerical Off-the-Shelf
C&DS	Command and Data System
DSN	Deep Space Network
eMMRTG	Enhanced Multi-Mission Radioisotope Thermoelectric Generator
ERV	Earth return vehicle
EDL	Entry Descent and Landing
EDLS	Entry Descent and Landing System
FTA	Fault Tree Analysis
FOV	Field of View
FRR	Flight Readiness Review
GNC	Guidance, Navigation, and Control
GLOM	Gross Lift Off Mass
I_{sp}	Specific Impulse
JPL	Jet Propulsion Laboratory
KBPS	Kilo Bits Per Second
LGA	Low Gain Antenna
LIDAR	Light Detection and Ranging
LMSS	Lockheed Martin Space Systems
LV	Launch Vehicle
LQR	Linear Quadratic Regulator
MAV	Mars Ascent Vehicle
MAHLI	Mars Hand Lens Imager
MER	Mass Estimating Relation
MI&T	Manufacture, Integration, and Test
MICKEY	Mars Ice Core Key Exploration Yacht
MMH	Monomethyl Hydrazine
MON–3	Mixed Oxides of Nitrogen, 3% Nitric Oxide
MS	Margin of Safety
NGSS	Northrop Grumman Space Systems
NTO	Nitrogen Tetroxide
O/F	Oxizier/Fuel
OODM	On-Orbit Dry Mass
OS	Orbiter Spacecraft
PD	Proportional/Derivative
P&ID	Parts and Instrumentation Diagram
PCEC	Price Cost Estimating Capability
PDR	Preliminary Design Review
PICA	Phenolic Impregnated Carbon Ablator
RCS	Reaction Control System
RFP	Request for Proposal
SRC	Sample Return Capsule
SDR	System Design Review
SIR	System Integration Review
SRP	Solar Radiation Pressure
SSPA	Solid State Power Amplifier
SSTO	Single Stage to Orbit
TEI	Trans Earth Injection
TOF	Time Of Flight
TRL	Technical Readiness Level
TS	Time Stamp
TSTO	Two Stage to Orbit
TVAC	Thermal Vacuum Chamber
TWTA	Traveling Wave Tube Amplifier

1.0 Executive Summary

Ares Advena Labs pursues the global effort to make human civilization a multi-planetary civilization by designing a system to study, collect, and deliver Martian ice cores back to Earth. The system is named: **M**ars **I**ce **C**ore **K**ey **E**xploration **Y**acht (**MICKEY**). Studying these ice cores will bring insight into if life has existed on Mars. Humans will need to colonize other celestial objects to preserve the longevity of the human race, and Mars is the best candidate for colonization. The research gained from these ice cores will allow scientists to understand if life could be sustained with the Marian water. The ice cores will also allow scientists to research methods to develop propellant and life support systems with oxygen and hydrogen. Successful recovery of the ice cores with the MICKEY architecture will benefit future exploration to Mars and beyond.

The MICKEY architecture contains two main spacecraft, an orbiter and an entry, descent, and landing system (EDLS), to complete the mission. The cruise stage will house the lander, the rover, and the Mars ascent vehicle (MAV); the orbiter will house the sample return capsule (SRC). This mission duration is planned to be 935 Earth days and will end in May 2029. The mission cost was estimated using the NASA Price Cost Estimation Capability (PCEC) and was found to be $985 Million, which includes a 10% Margin. Each vehicle was designed to complete its intended mission with efficiency and cost in mind. We also considered planetary protection requirements as the samples must be returned in pristine condition and will remain frozen throughout the entire mission.

There were several challenges the MICKEY design faced, but the most significant was dealing with the Mars ground operations. The first step to recover ice cores successfully is to land the rover, Mars ascent vehicle, and lander on the ice in Louth Crater. The EDLS requires precise sensors; therefore, the lander utilizes three systems: Doppler LiDAR, Flash LiDAR, and a Laser altimeter. Together, these systems allow the lander to find the best landing site in Louth Crater and navigate the system to touchdown safely with a calculated landing velocity of 1.7 m/s. There are twelve MR-80B thrusters to assist with the powered descent, which are split into four clusters of three engines. Each cluster contains a redundant engine since we designed the lander to use only eight for its powered descent. The lander accomplishes its primary mission when it touches down safely and deploys the ramp for the rover to conduct its mission. The intended landing site will be under five months of total darkness. Therefore, the lander uses an enhanced multi-mission radioisotope thermoelectric generator (eMMRTG) to provide reliable power and heat for itself and the MAV

The rover is the most crucial vehicle because it collects 3.0 kg of ice cores and keeps them frozen for the entire mission. The rover utilizes sensors and cameras to detect the ice samples before retrieval. The rover will operate in the cold, icy environment as it uses an onboard eMMRTG to provide constant power and heat throughout the mission. The ice cores will be collected expeditiously with the coring drill, and the samples will be studied with the onboard Planetary Instrument for X-ray Lithochemistry (PIXL). The ice cores will be packed into sealed sample tubes to prevent sublimation and melting. These tubes will be manufactured to meet the cleanliness requirements and prevent Earth contamination to the Martian ice cores. There will be three witness tubes onboard the rover to test the contamination levels of the sample tubes on Mars to meet the cleanliness requirements. The rover completes its mission when the cryocooler containing the 3.0 kg of ice cores is placed into the MAV.

The Mars ascent vehicle will be brought down to the Martian surface with the lander and is housed inside a launch silo. The MAV is tasked with the most challenging phase of our mission because it must autonomously launch from Mars, jettison its nose, and dock with the orbiter. The MAV launches the 24 kg cryocooler payload into a 300 km Low Mars Orbit to rendezvous with the orbiter. The male docking adapter is placed below the sample container onboard the MAV, and the female docking adapter will be inside the sample return capsule. The orbiter rendezvous with the MAV and utilizes its docking sensor to conduct the docking maneuvers. We designed the docking maneuver to occur even if the docking sensor were to fail, using telemetry between the two vehicles.

The orbiter will be in operation for the entire mission, 935 Earth days, and has the task of transmitting data and delivering the sample return capsule back to Earth. The orbiter will spend its entire mission in Low Mars Orbit and interplanetary space, requiring the use of radiation-hardened electronics to ensure they will survive the harsh environment. The 3-axis stabilized orbiter will use one degree of freedom solar arrays to generate power. A two-meter, fixed telecommunications dish will provide 57 kbps with a minimum margin of 5.6 dB with the Deep Space Network throughout the mission.

Ares Advena Labs designed each vehicle using over 170 requirements derived from the customer requirements. We used model-based systems engineering to ensure the traceability of the customer requirements throughout the system's design. Trade studies were used for each vehicle and subsystem to determine the best solution. We developed unique figures of merit for each vehicle since they all have different missions and working environments. The orbiter must operate in interplanetary space, while the rover and lander must operate on Martian ice. There will be no sunlight for five months at our landing site, therefore, we ensured that the ground operations are completed before this event occurs. However, for redundancy purposes, we designed the rover and lander to hibernate during this event by utilizing eMMRTGs.

We initially designed two different architectures to complete this mission and continued each design for several months. We developed a conventional and radical design and down-selected after we conducted our System Design Review (SDR). The conventional design was the MICKEY architecture since it included vehicles with high TRL and low risk. The second architecture, BIG Mars Ascent ConcepT (BIG MACT), was the radical design since it included a two-stage Mars ascent vehicle and a large EDLS. The lander in the BIG MACT concept was designed to land around 8,000 kg on the Martian surface and have a small spacecraft on the MAV deliver the samples back to Earth. This design was risky since it included low TRL vehicles and concepts. The cost was also a concern because the large vehicles would be expensive to test and manufacture according to the NASA PCEC tool. We performed a trade study to down-select to a single architecture; the figures of merit included cost, mass, and complexity. We quantified complexity by counting the number of single points of failure and the number of moving parts. The higher the value, the greater the complexity was for each design. The team found different methods to reduce the level of complexity with each vehicle during each design iteration. One method to reduce risk was to simplify mechanisms and maneuvers. The orbiter initially had a robotic arm to transfer the samples, but the complexity of this was reduced when we transferred the samples during the docking maneuver. We found the conventional design with MICKEY was the best fit for the mission with this trade study.

While designing each architecture, we held many design reviews with our advisors and industry representatives to ensure our design was complete. In total, we had six reviews, four of which were with our advisors. The other two reviews were with Northrop Grumman Space Systems and NASA JPL. Each design review included panelists of scientists and engineers with experience in spacecraft and rover design. The advice we received helped our team improve the design to ensure the requirements are met with acceptable risk. These design iterations guided the team to our final MICKEY architecture.

We utilized multiple programs to design our architectures. Microsoft Excel was used to track all design changes and calculate corresponding values. This spreadsheet helped the team understand how a simple change could alter the entire design. Using this spreadsheet, the team could get mass and power estimates for each architecture when we started this design project. The team used these estimates to form CAD models with SOLIDWORKS 2020. Visualizing the designs was an integral step for the design iterations. When specific components were selected for each subsystem, they were also modeled in SOLIDWORKS and added in. The vehicles were assigned their respective material to determine their structural integrity and mass. Using the SOLIDWORKS FEA simulations, the team analyzed the margin of safety for the structures. The team analyzed worst-case loads, such as the Earth launch and the EDL phase of the mission. Also, we utilized ANSYS for our thermal analysis for each vehicle. Since the ground operations phase of the mission will take place directly on the ice, we found the worst-case scenario and used ANSYS to design the proper thermal control system for the lander and rover. The orbiter went through a similar process; however, its operating environment is interplanetary space and Mars orbit. Next, AGI Systems Tool Kit (STK) was used to model the telecommunications systems. STK allowed us to visualize and analyze communication links between our system and the DSN. Finally, the NASA PCEC tool generated an accurate system cost. Manual inputs for the program included: number of science instruments, number of RTGs, TRL, duration of mission phases, LV costs, and structure material. Our team updated these inputs with every design iteration to refine our cost estimate to improve our accuracy. The vehicles' masses would also increase the overall cost, so the team cut mass wherever possible without jeopardizing the derived requirements. The cost for the system only included launch, design, development, test, and evaluation for each vehicle; this excludes the ground operations from the total cost.

The system life cycle ensures ample time to manufacture and test each vehicle before launch. There are seven phases of the mission that incorporate more design reviews to ensure each vehicle will be manufactured and integrated correctly and meet the cleanliness requirements defined for planetary protection. We allocated different vehicles to different industries such as Lockheed Martin and NASA JPL to ensure the best quality of work. There is also a timeline specified for proper disposal for each vehicle. The orbiter will be placed into a heliocentric orbit after it delivers the sample return capsule to Earth. The rover and lander will continue to study the Martian ice until their sensors and mechanisms wear out. The eMMRTGs on the lander and rover will continue to provide daily power for 14 years post-landing. The MAV will remain in its Low Mars Orbit until its orbit decays, and the MAV crashes into the Martian surface if it does not disintegrate during reentry. After the sample return capsule has completed its mission, we plan to donate the capsule to Cal Poly Pomona to showcase for the entire community.

Our completed MICKEY architecture is a robotic, autonomous system that will retrieve 3.0 kg of ice samples from Mars. The EDLS system will conduct a powered descent to safely bring the lander, rover, and MAV on top of the ice deposit located within Louth Crater. The ice mound is 15 km in diameter and will be the target for our landing site. The rover's coring drill will extract samples that are 50 mm in diameter and 150 mm in length. All samples would be sealed after it has been retrieved and studied by the PIXL to preserve their frozen state. Each sample would be stored inside a cryocooler to maintain the ice cores temperature throughout the entire mission until Earth recovery. The sample packing method would maximize the scientific data for the ice cores as it prevents contamination on Mars and Earth. The PIXL data would also provide additional information on the composition of the ice upon retrieval. The Mars Hand Lens Imager (MAHLI) would also provide detailed images on the sample retrieval site and the surrounding environment. Our landing site in Louth Crater was chosen amongst four other candidates after a trade study was conducted. We compared the different elevations, amounts of fresh water ice, and terrain variations amongst the Utopia Planitia, Arcadia Planitia, North Pole, and the South Pole. Trade Studies dictated our decisions throughout this project and chose specific subsystems and components for our vehicles; common figures of merit included cost and mass. This was to conserve cost as the entire budget should not exceed $1 billion (FY20).

Our system life cycle includes phases for design, assembly, test, evaluation, and integration for the subsystems and vehicles. Flight-ready sensors and components were chosen for each vehicle. The MICKEY architecture includes details for every phase of the mission from Earth launch, interplanetary coast, EDL, ground operations, Mars launch, sample transfer, Earth EDL, and sample recovery. The mission schedule will return the samples before December 31, 2030.

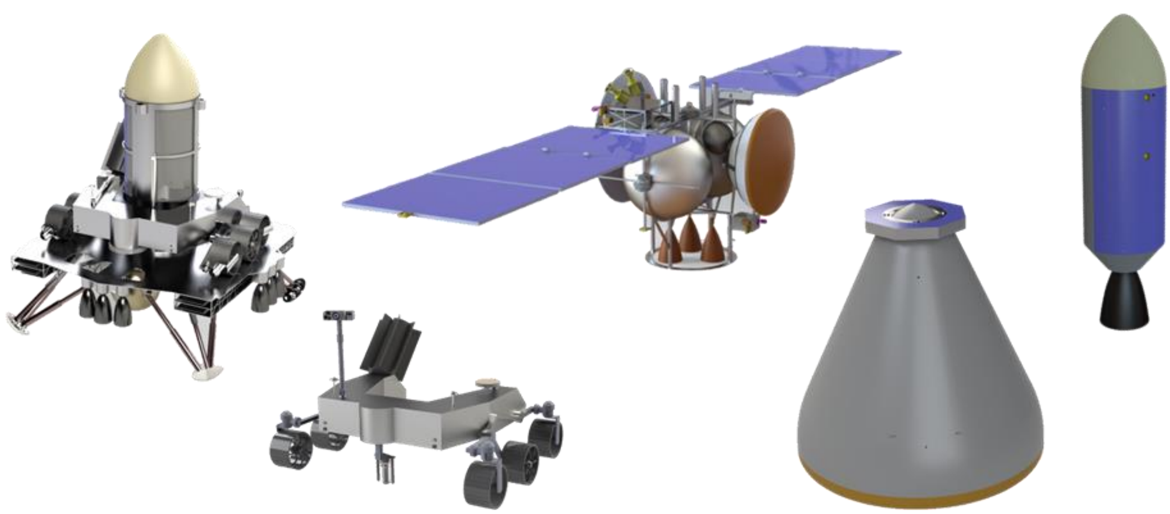


Figure 1.0-1 Lander, Rover, Orbiter, and Cruise Stage Rendered Images

2.0 Mission Overview

2.1 Needs Analysis

A system to recover Martian ice cores is needed to help establish a human presence outside Earth's sphere of influence. These ice cores will provide insight into Mars and help us understand if the Martian water could sustain life. This research would also pave the way for future exploration and crewed missions since water could provided propellant for propulsion systems and breatheable oxygen for life support systems.

2.2 Customer Requirements

The request for proposal (RFP) supplied requirements, constraints, and deliverables. We transformed the design requirements and constraints into customer requirements, and we have shown the them in **Table 2.2-1**. Better traceability of derived requirements to the customer requirements was made possible by developing system level requirements. **Appendix B** shows the traceability between customer, system level, and derived requirements.

Table 2.2-1 Customer Requirements

ID	Requirement Statement
CUST.1	The system shall use robotic vehicles to accomplish the mission
CUST.2	The system shall operate on the surface of Mars
CUST.3	The system shall land on or near Martian ice deposits
CUST.4	The system shall drill ice cores on Mars
CUST.5	The system shall return ice cores with a diameter greater than 25 mm and a length greater than 100 mm
CUST.6	The system shall keep ice cores frozen during entire mission operations
CUST.7	The system shall return at least 2.5kg of Mars ice cores to Earth
CUST.8	The system shall accommodate the safe transfer of ice cores to laboratories on Earth
CUST.9	The system shall return ice cores to Earth no later than December 31, 2030
CUST.10	The system's launch, DDT&E, and flight unit shall cost no more than $1 billion (FY 2020)

2.3 Mission Objective

The key mission objective is to drill and deliver Martain ice core samples to Earth before the end of 2030. Each vehicle is designed to survive its respective environments, endure expected loadings from launch and reentry, and rapidly complete its mission. The ice samples gathered must be kept frozen throughout the entire mission and be properly sealed to preserve scientific data and avoid contamination. The mission will be completed with expeditiously intent as the samples must be returned before December 2030. We will be using the most up-to-date technology available with a budget that does not exceed $1 Billion (FY20). Avoiding risk in the design was essential, so TRL levels above six were desired for all vehicles and their subsystems.

3.0 Mission Design

3.1 Design Method/Process

When we received AIAA's Request for Proposal for the Mars Ice Core Retrieval mission, we created two architectures with different approaches to meet the customer requirements. The two designs consisted of a conventional design and a radical design. Ares Advena Labs continued each design for several months, ensuring each vehicle will complete its mission with acceptable risk. We presented our designs in several design reviews with advisors and industry professionals from Northrop Grumman and NASA JPL, which helped us further improve our design.

3.2 Summary of MICKEY Architecture

The first architecture, Mars Ice Core Key Exploration Yacht (MICKEY), is our conventional design to accomplish the RFP. The key components of MICKEY are an orbiter, lander, Mars ascent vehicle, and a rover. The lander is powered by an eMMRTG and uses a robotic arm to transfer the sample container from the rover to the MAV. The orbiter acts as a relay for the ground vehicles, performs docking maneuvers with the MAV, and uses a liquid bi-propellant propulsion system. The Mars ascent vehicle is a single-stage-to-orbit launch vehicle, performs a docking maneuver with the orbiter, and uses a liquid bi-propellant propulsion system. The rover is powered by an eMMRTG and was uniquely designed in a U shape to fit around the MAV. The mission will launch from Earth on November 1, 2026, arrives at Mars on August 18, 2027, departs from Mars on October 6, 2028, and returns to Earth on May 24, 2029.

3.3 Summary of BIG MACT Architecture

The second architecture, BIG Mars Ascent ConcepT (BIG MACT), is our radical design to accomplish the RFP. The vehicles of BIG MACT are a two-stage-to-orbit (TSTO) MAV, oblate lander, Earth return vehicle, sample return capsule, and a rover. The oblate lander will act as a launch erector for the MAV and uses a mono-propellant propulsion system to land. The rover uses an eMMRTG and uses a conventional rover design. The TSTO MAV uses a solid propellant first stage and a liquid bi-propellant second stage. After the MAV launches the Earth return vehicle (ERV) to orbit, the ERV will support the sample return capsule and return it to Earth. The mission will launch from Earth on November 1, 2026, arrives at Mars on August 18, 2027, departs from Mars on October 6, 2028, and returns to Earth on May 24, 2029.

3.4 Down Selection

When analyzing the down-selection of the architectures, we considered the number of moving parts and single points of failure to quantify each system's complexity. It was important to reduce risk when possible; therefore, fewer single points of failure would benefit the mission. The trade study also considered the TRL of each vehicle to ensure reliability and ease of manufacturing and integration. The cost was a heavily weighted figure of merit as the total budget of the mission shall not exceed $1 Billion (FY2020). The NASA PCEC tool was used to estimate each mission cost for this trade study. The down-selection trade study results are shown in **Figure 3.4-1** and illustrate that the MICKEY architecture is the best design for this mission.

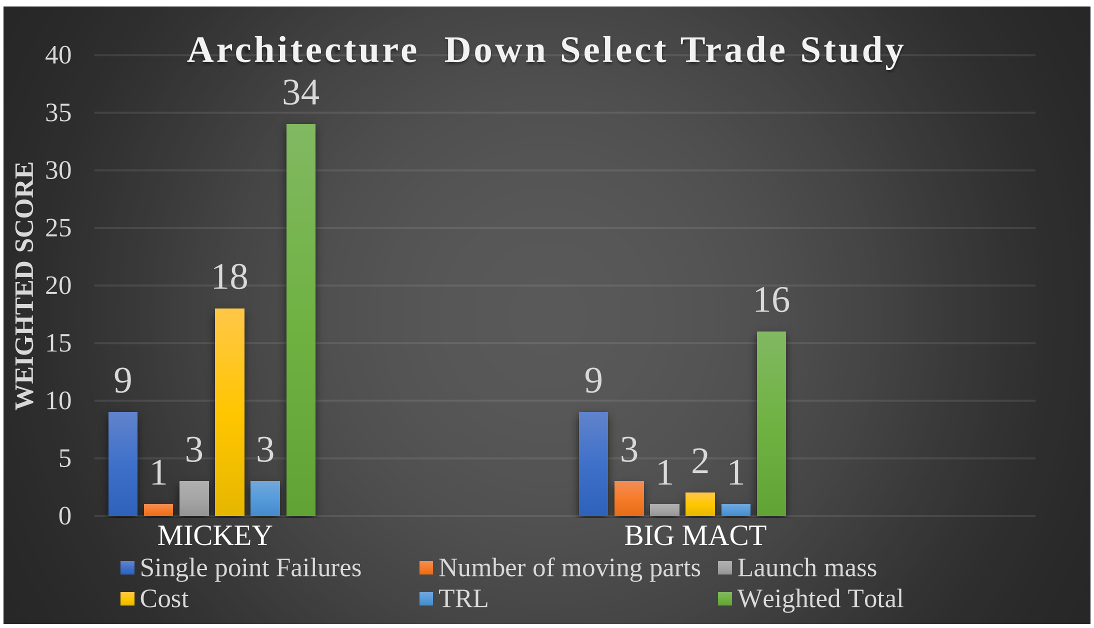


Figure 3.4-1 Architecture Down Selection Trade Study Results

3.5 Concept of Operations

The following section will discuss the concept of operations for the MICKEY architecture. **Figure 3.5-1** below illustrates the key phases that will occur throughout the mission. Ares Advena Labs will utilize an expendable Falcon Heavy to bring the orbiter and cruise stage into low Earth orbit and then inserts them into a trans-Mars injection with a C_3 of $9.19\ \frac{km^2}{s^2}$. This launch will occur on November 1, 2026, with a launch window of -40 to +10 days.

Once the system achieves a Mars rendezvous trajectory, the lander and orbiter will separate. After a cruise time of 290 days, both spacecraft will arrive at Mars on August 18, 2027. A ΔV of 0.9 km/s is required from the orbiter to obtain a 300 km x 45,000 km Mars orbit. The orbiter will aerobrake into a circular 300 km parking orbit during its 414 day stay. Concurrently, the lander will enter the Martian atmosphere and begin the EDL phase of the mission.

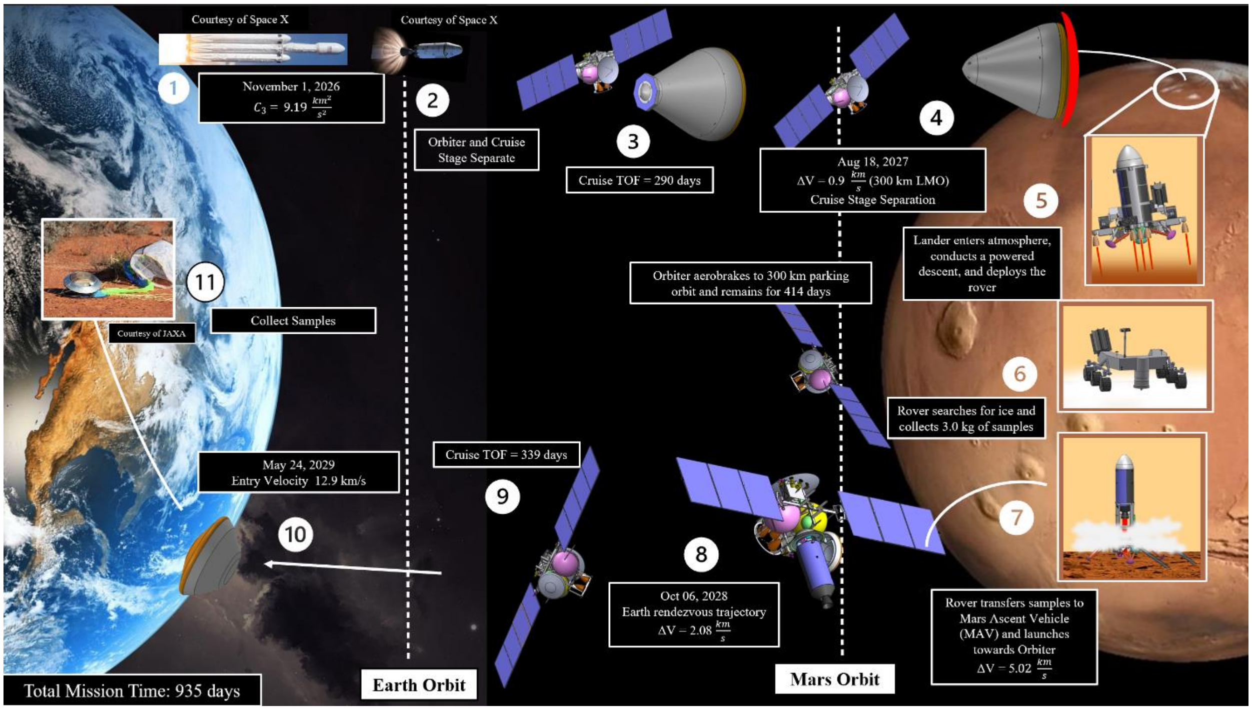


Figure 3.5-1 MICKEY Concept of Operations

When the lander reaches an altitude of 4.2 km, an 18 m diameter supersonic parachute will deploy at 165 seconds after entry. At 170 seconds, the heat shield ejects. At 178 seconds, the lander will conduct a powered descent utilizing 12 MR-80B thrusters. This descent will consume 256 kg of fuel, leaving a 27% fuel margin. Touchdown will occur at 235 seconds at a speed of 1.7 m/s. Once a safe landing has been confirmed, the lander will deploy its spring-loaded ramps. The rover will then spend some time testing its instruments and electronics before starting its mission. There will be a one Earth week period where the rover must test its systems and prove all systems are operating nominally. When tests are completed, the rover will begin its mission, and the lander will conduct several tests of its instruments.

Once deployed, the rover will begin its search for ice within Louth Crater. Sensors will confirm the presence of ice, and the rover will deploy the drill to begin sample retrieval. The drill will collect an ice sample with a diameter of 100 mm and a length of 50 mm. The collected sample will be studied by the onboard Planetary X-ray Lithochemistry sensor in the rover's body and then placed into a sample tube. The samples will be stored within an onboard cryocooler that will keep the samples frozen throughout the remainder of the mission. Each sample would take about two Earth days to retrieve, seal, and store into the sample container. There are 37 samples that will fit into the container; therefore, it will take 74 Earth days for the rover to successfully collect a full set of samples. However, we plan for 90 Earth days to include margin. There is also an additional 14 Earth days added in for travel time. With this timeline, in

December 2027, the rover will deliver the sample cryocooler to the lander. The lander's robotic arm will grab the sample container and place it within the nosecone of the MAV. The rover will move away from the vicinity as the MAV prepares for launch. There are 14 days allocated for the lander to transfer the samples to the MAV and prepare for launch to low Mars orbit.

A ΔV of 5.02 $\frac{km}{s}$ is required to reach the orbiter's 300km orbit for rendezvous and docking. After the MAV achieves orbit, its nosecone will eject to expose a male docking adapter. The MAV will act as the passive target, and the orbiter will actively perform the rendezvous maneuvers as the "chaser" spacecraft. The 3-axis controlled orbiter will align the sample return capsule's docking adapter with the MAV docking adapter. Once docked, the cryocooler with the ice core samples will be securely transferred from the MAV to the SRC. After confirmation that the sample cryocooler has been transferred, the SRC and MAV undock. The orbiter will remain in its parking orbit until its Earth transfer window arrives.

On October 6, 2028, the orbiter will perform an escape burn with a ΔV of 2.18 $\frac{km}{s}$ to enter an Earth rendezvous trajectory. After a cruise time of 230 days, the orbiter will arrive at Earth on May 24, 2029, with a V_∞ of 4.27 $\frac{km}{s}$. The orbiter will release the SRC, which will enter Earth's atmosphere. After entry, a six-meter diameter parachute will deploy at an altitude of five km. Then, the heat shield is ejected, and the SRC will touchdown at a speed of 8.3 m/s. Overall, the total mission time is 935 days.

3.6 Landing Site Trade Study/Research

When assessing a potential landing site, we prioritized avoiding steep and rugged terrain. The landing site needed to be near surface-level water ice to allow the rover to quickly acquire samples. The site needed to be of high scientific interest and have the potential to preserve evidence related to habitability. It was necessary that the landing site was free of large rocks to protect the lander during the powered descent and allow the ramps to deploy fully. The environment of the landing site significantly impacted the lander and rover thermal design. In the colder conditions, we designed the lander and rover to provide power for heating onboard components, electronics, and propellant.

Five possible landing sites were considered for our ice sample return mission. Utopia Planitia is located at 46.6 °N 117.5 °E. It can be considered the very largest plain within Utopia; also the largest impact crater on Mars which its estimated diameter is around 3,000 kilometers. A large amount of underground ice can be found in the Utopia Planitia area, and the volume of water ice is equivalent to the volume of water in Lake Superior. The second possible landing site is Arcadia Planitia, located at 47.2 °N 184.3 °E, and lies just south of the northern polar ice cap. A high possibility of ground ice is in the low regions of Arcadia. Planum Boreum is located at the northern polar plain on Mars. A large region of polar ice caps covered up to 100 kilometers wide and 2 kilometers high, and these permanent ice caps mainly consist of water ice. Planum Australe is at the southern polar plain on Mars. But is partially covered by a permanent ice cap composed of frozen water and carbon dioxide around 3 kilometers thick. There will be seasonal ice caps forms on top of the permanent ice caps during winter. The Louth crater is located at 70.0 °N 103.2 °E and around 36 kilometers in diameter. The ice mound inside the crater is estimated to be 15 km in diameter.

The trade matrix below compares all five possible landing sites with weight based on the water ice abundance, water ice depth, landing surface environment, landing elevation, and environmental challenges. Water abundance shows how much water ice exists within those potential landing sites, and it increases the chance of a successful mission by having access to a large number of samples. Water ice depth is determined to be an important figure of merit because it is difficult to extract ice if it is very deep under the surface. The landing surface environment is another important figure of merit because it is vital to the success of the landing. A rough landing surface environment might cause an off-balance on surface touch down and lead to a crash. Landing elevation is important because lower altitudes allow for easier landing, and ground ice's presence is found to be near the subsurface level at higher latitude.[14] Furthermore, terrain challenges such as cliffs, rugged terrains, and large rocks should be avoided when selecting landing sites because it is difficult for the Rover to travel through such terrain challenges and might be prone to mission failure.

The water ice mound within the Louth crater has the largest surface water ice deposit at this latitude and has mound temperatures between 173 K to 246 K. Ice exposed at the center of this deposit is older than exposed within the edge. So higher quality of the ice core samples at the surface of the Louth crater perfectly meets our ice sample return mission. As a result, the Louth crater located at will be the landing site for our mission. **Figure 3.6-1** summarizes our trade study results, and **Figure 3.6-2** and **Figure 3.6-3** illustrate the elevation characteristics of Louth Crater and its location relative to the North Pole, respectively.

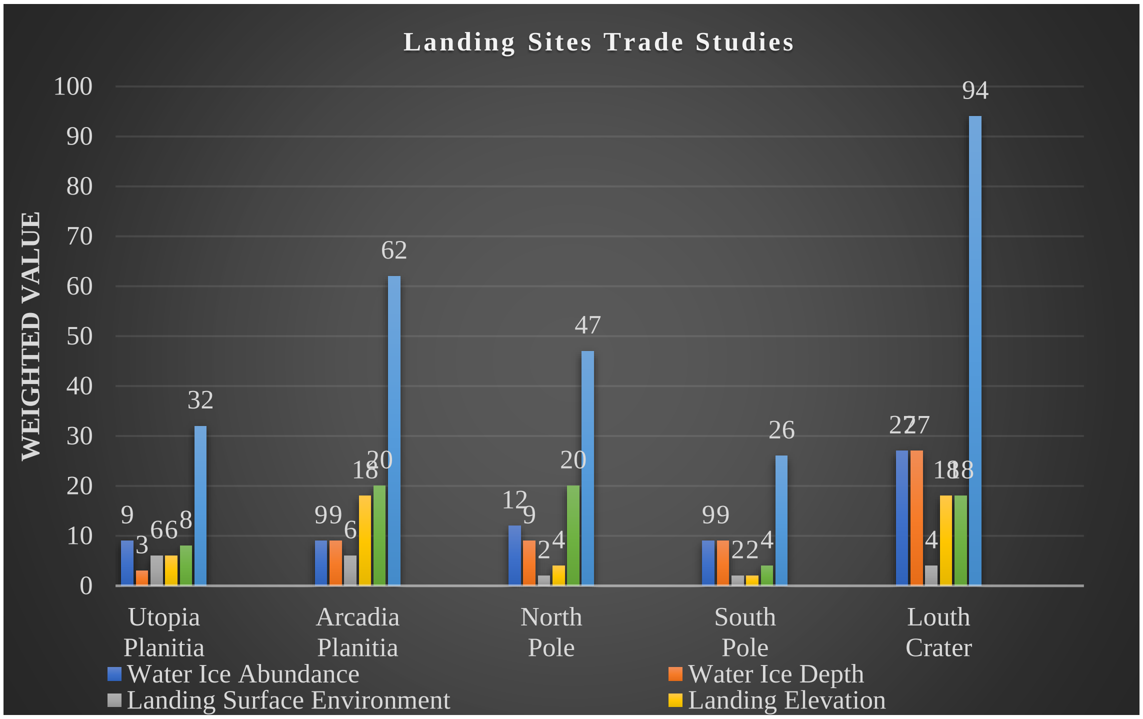


Figure 3.6-1 Landing Site Trade Study

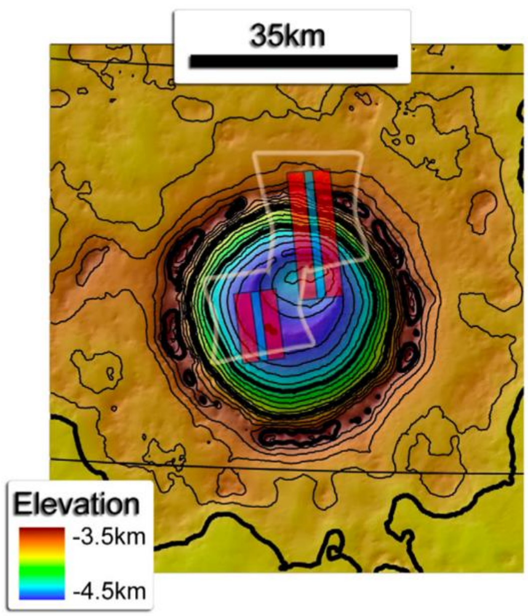


Figure 3.6-2 Polar Stereographic Projection of Louth Crater

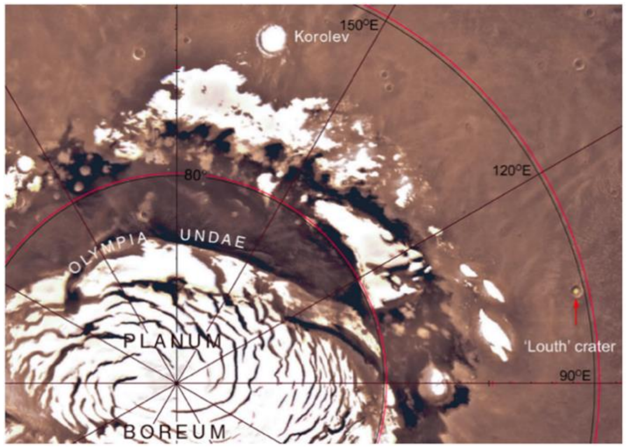


Figure 3.6-3 Location of Louth Crater

3.7 Trajectories

We designed our interplanetary trajectories using the patched conic approximation and a Lambert problem solver in MATLAB. MATLAB swept through multiple departure and arrival dates to compare the required ΔVs. Porkchop plots were created to visualize the launch windows and find optimal departure and arrival dates by comparing launch C_3s, inset and escape ΔVs, and entry velocities. An example of one of the porkchop plots used is shown below in **Figure 3.7-1**. MATLAB was also used to simulate the ballistic entries of the lander at Mars and the sample return capsule at Earth. These were used to estimate g-loads and heating loads. The chosen planetary trajectories were also simulated in Freeflyer to verify their accuracy.

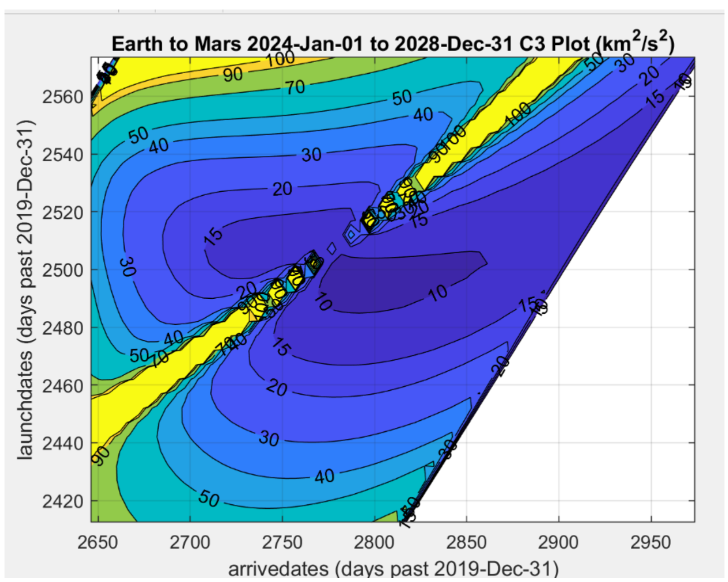


Figure 3.7-1 Earth Launch Porkchop Plot

First, the goal of the trajectory from Earth to Mars is to maximize delivered mass to Mars by minimizing launch C3 and insertion ΔV. To further decrease the amount of fuel needed, the orbiter will capture into a highly elliptical orbit of 300 x 45,000 km at Mars and aerobrake into a 300 km circular orbit. Later launch dates were preferred to provide ample time to develop, manufacture, and test the vehicles. **Figure 3.7-2** shows the resulting trajectory is a Hohmann Transfer, which is expected since launch C3 and insertion ΔV were minimized. **Table 3.7-1** below shows the important parameters of this trajectory. Since the Falcon Heavy has a high payload mass capacity and the orbiter is designed with an appropriate mass margin, the launch window is -40 to +10 days from the date listed in the table. Late 2026 is the latest launch opportunity to Mars that allows the samples to be returned before Dec 31, 2030. The next opportunity to launch after 2026 will be late 2028, which would not allow the system to return to Earth before Dec 31, 2030. This trajectory also provided an acceptable entry velocity for the lander and is comparable to past Mars lander missions. In sum, this trajectory maximizes the mass delivered to Mars while providing an acceptable entry velocity for the lander and maximizing the amount of time available to develop, manufacture, and test the vehicle.

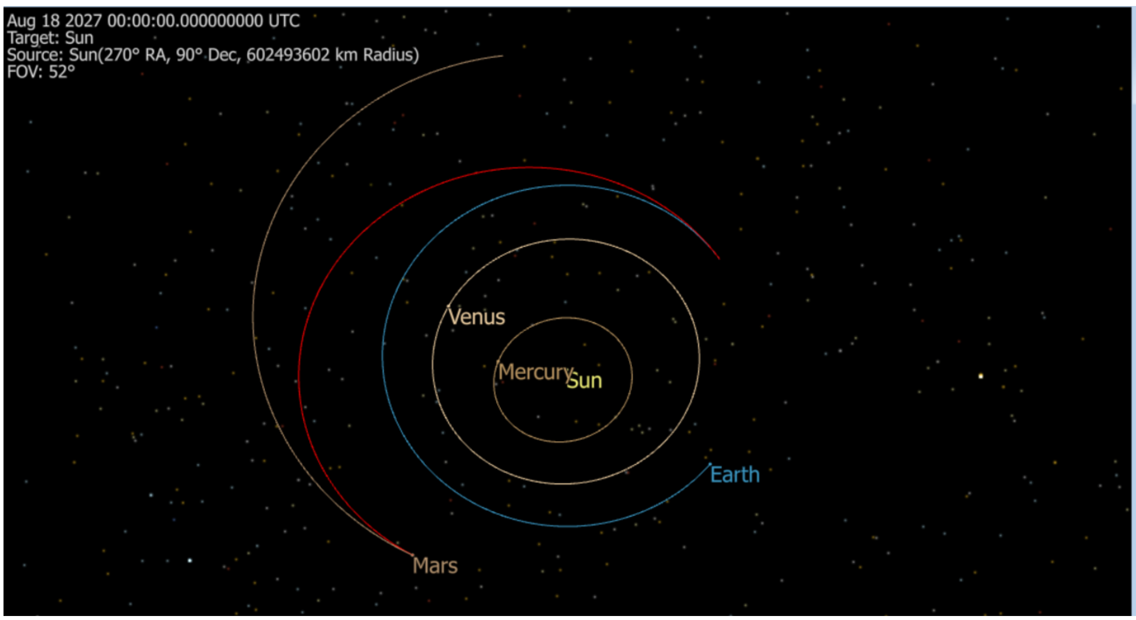


Figure 3.7-2 Earth to Mars Trajectory

Table 3.7-1 Earth to Mars Trajectory Parameters

Departure Date	November 1, 2026
Arrival Date	August 18, 2027
Time of Flight	290 days
Launch C3	9.19 km^2/s^2
Orbiter Insert ΔV	0.902 km/s
Lander Entry Velocity	5.6 km/s

Next, the EDL trajectory for the lander is discussed. **Figure 3.7-3** below shows the simulation results for the lander conducting a ballistic entry. There are four events that happen in succession: parachute deployment, heat shield separation, back shell separation, and touchdown. Based on the Curiosity and Perseverance missions, it is likely that a lifting entry will be needed to land at Louth Crater accurately. However, at this point in the design, the ballistic entry was easier to simulate. It allowed the team to size the subsystems for the lander, such as its parachute, propulsion system, and structure. The lander will use terrain relative navigation to avoid obstacles when it approaches its landing zone. Multiple simulations were run to determine appropriate values for the size of the parachute and the amount of fuel needed. The goal was to achieve a low touchdown velocity. As shown in **Figure 3.7-3**, with all the events happening at the respective times, an 18 m diameter parachute and 256 kg of fuel are needed to achieve a touchdown velocity of 1.7 m/s. The lander will have 350 kg of fuel allocated for the powered descent, which provides a 27% fuel margin. This also provides margin if the EDL trajectory is changed in the future and will help prevent the need to resize the fuel tanks. Also, MR 80-B thrusters were chosen to perform the powered descent because of their deep throttle capabilities. The simulations confirmed that the thrusters are more than capable of providing the thrust range needed to perform the powered descent. Next, as seen in the top right of **Figure 3.7-3**, the maximum g-load experienced is 12 Earth g's when the parachute opens; therefore, the lander's structure was designed to survive those loads. Finally, the heat shield was chosen to be made of Phenolic Impregnated Carbon Ablator (PICA) and was sized similar to Curiosity and Perseverance's heat shields, which could survive heating rates above 200 W/cm^2; this was so that a lifting entry like those missions could be used. However, this heat shield could also survive the ballistic entry as the maximum heating rate from the simulation was found to be about 75 W/cm^2, as seen below in **Figure 3.7-4**. The lander's heat shield, parachute, propulsion system, and structure have been appropriately sized and designed to survive the rigors of EDL at Mars. The next trajectory that will be discussed is the MAV trajectory.

The designed trajectory and other performance metrics can be seen below in **Figure 3.7-4**. This trajectory assumes that there will be no velocity bonus from Mars' rotation and takes the MAV from the Martian surface to a 300 km orbit. This is seen on **Figure 3.7-4** as represented by the green line and the left axis values. The flight path angle is set to 90° until 290 seconds into the flight, where a pitch kick angle of just under 0.08° is applied to the MAV. A constant pitching maneuver of 0.3° per second is applied to the MAV in addition to the minor pitching effects of gravity. The pitching maneuver ends at engine shutoff at 550 seconds. Throughout the entire flight, the S3K engine continuously fires at full throttle with a thrust of 3,500 Newtons. This can be seen through the smooth velocity curve until engine shutoff at a final velocity of 3,409 meters per second. Maximum dynamic pressure is 145 pascals and occurs at 137 seconds into flight. At the end of the flight, there is about 5.3 kg of MON-3 and about 18 kg of MMH, not including any usage of MMH for the ACS. This leaves a remaining ΔV of about 270 m/s.

1-Parachute Deploy- 18m diameter- Opens at 4.2 km

2-Heat Shield Separation

3-Back Shell Separation – Begin Powered Descent

4- Touchdown – velocity = 1.7 m/s

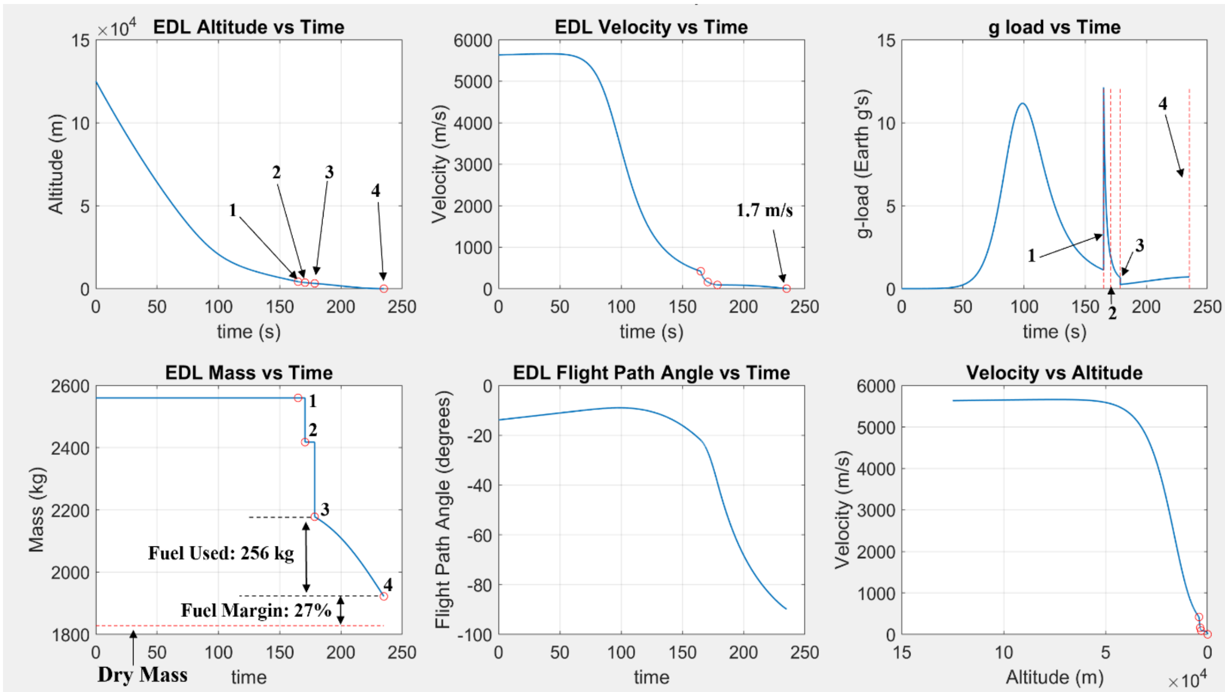


Figure 3.7-3 EDL Trajectory at Mars

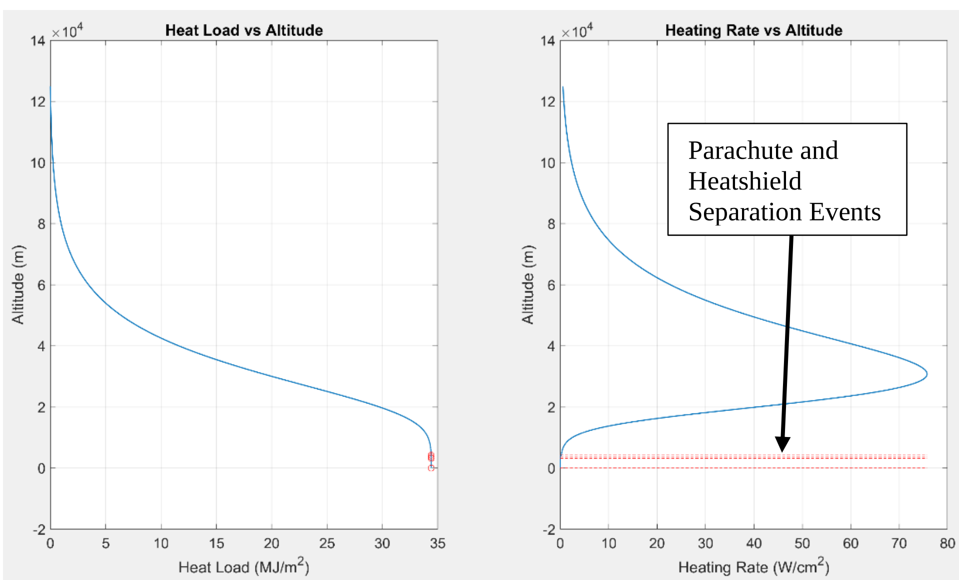


Figure 3.7-4 EDL Heat Load and Heating Rates during EDL at Mars

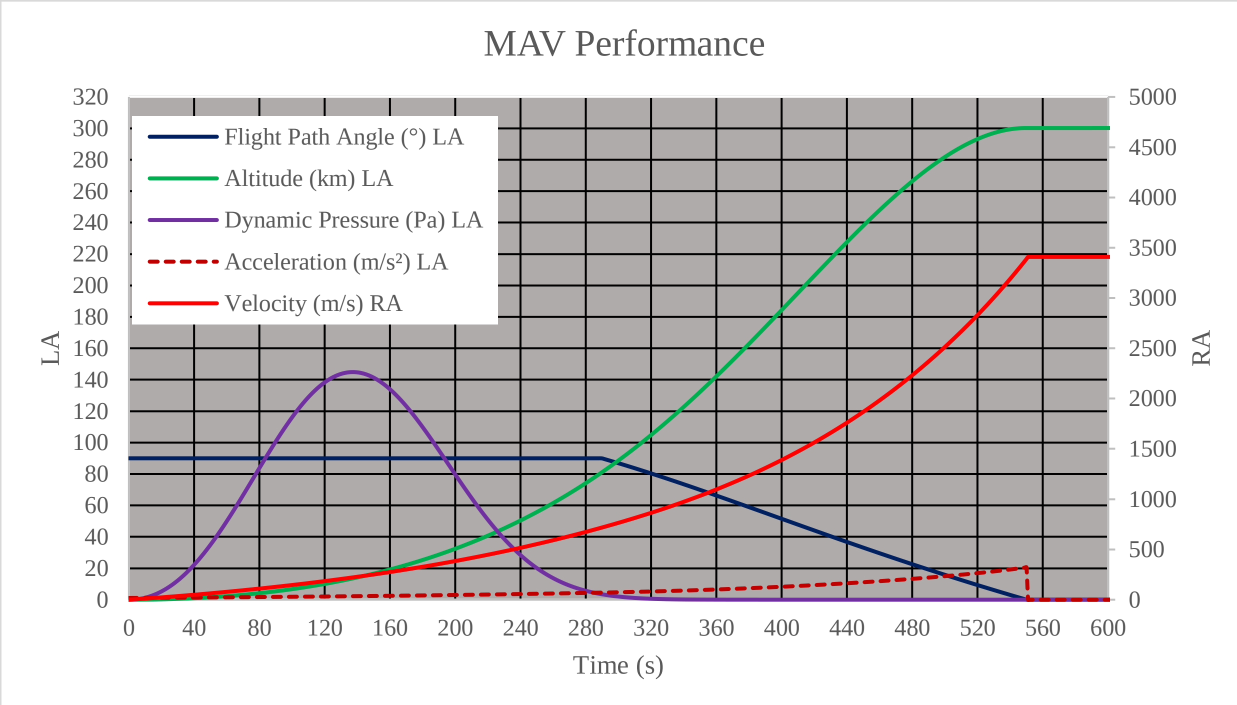


Figure 3.7-5 MAV Performance Graph

Next, the orbiter's trajectory from Mars back to Earth will be discussed. The goal was to minimize the escape ΔV and ensure that we returned before December 31, 2030. Minimizing the amount of fuel needed to escape would minimize the amount of fuel needed to capture the orbiter into Mars Orbit. The insertion ΔV at Earth was not a concern since the sample return capsule enters directly into the atmosphere, and the orbiter performs a flyby and enters its disposal phase. It was, however, pertinent to ensure that the entry velocity for the capsule at Earth was not too high. As seen in **Figure 3.7-5**, the resulting trajectory is a type 1 transfer since only the escape ΔV was minimized.[8] The orbiter was designed with an appropriate mass margin; therefore, there is a ± 20-day escape window from the day listed in **Table 3.7-2**. Also, as seen in **Table 3.7-2**, the entry velocity of the return capsule is 12.9 km/s which is the same as NASA Stardust's entry velocity. The entry velocity was required to remain below 14 km/s. The mission also benefits from the trajectory's shorter flight time by slightly reducing operations costs. In sum, the return trajectory from Mars ensures the samples return to Earth before December 31, 2030, while minimizing escape ΔV and providing an appropriate entry velocity for the sample return capsule.

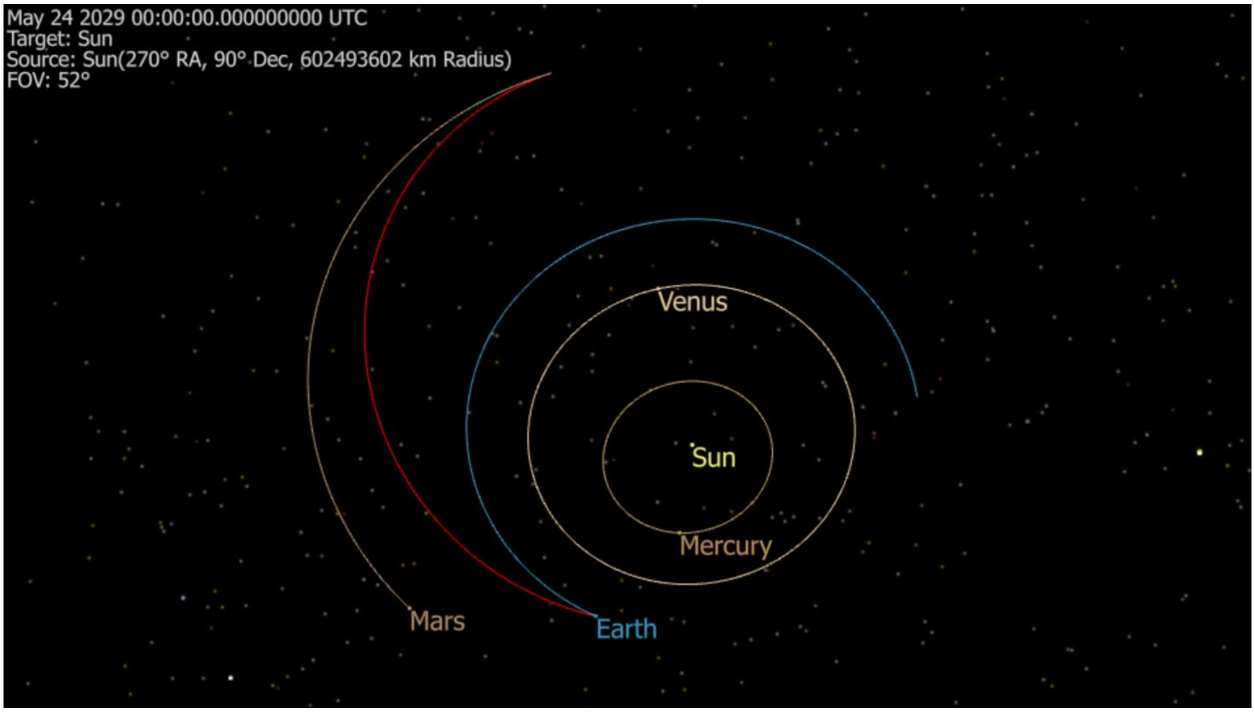


Figure 3.7-6 Mars to Earth Trajectory

Table 3.7-2 Mars to Earth Trajectory Parameters

Departure Date	October 6, 2028
Arrival Date	May 24, 2029
Time of Flight	230 days
Orbiter Escape ΔV	2.181 km/s
Sample return capsule Entry Velocity	12.9 km/s

Finally, upon arrival at Earth, the sample return capsule will need to survive entry into the atmosphere. This trajectory was designed and simulated in MATLAB, similar to the EDL trajectory. The goal of simulating the trajectory was to size the parachute and heat shield and analyze the g-loads. The factor that affects the g-loads and heating loads the most is the entry velocity and flight path angle; this was why the entry velocity was required to remain below 14 km/s. With the entry velocity being 12.9 km/s, an entry flight path angle of -5.7 degrees was chosen. As seen in **Figure 3.7-6**, this kept the maximum deceleration to 25 g's to reduce the loads on the ice samples. Using a steeper angle would result in higher g-loads (35-50 g's), but using a shallower angle would increase the total heat load, which would increase the mass of the heat shield. This angle was found to be a good compromise. We also determined that the capsule would touch down at 8.3 m/s using a six-meter parachute, which was the largest parachute that could be reasonably fit into the capsule. The heat shield separates from the capsule after parachute deployment to further reduce the descent velocity. The capsule has cushioning to protect the samples upon touchdown. Next, as seen in **Figure 3.7-7**, the capsule will experience a total heat load of 130 MJ/m^2 and a max heating rate of 340 W/cm^2. Once again, PICA is chosen for the heat shield material since it is more than capable of handling this; these heating values were used to size the heat shield based on Mass Estimating Relations developed by Sepka and Samareh[3]. Overall, the samples return capsule's heat shield and parachute are properly sized to survive entry, and a shallower flight path angle is used to reduce the g-loads on the samples.

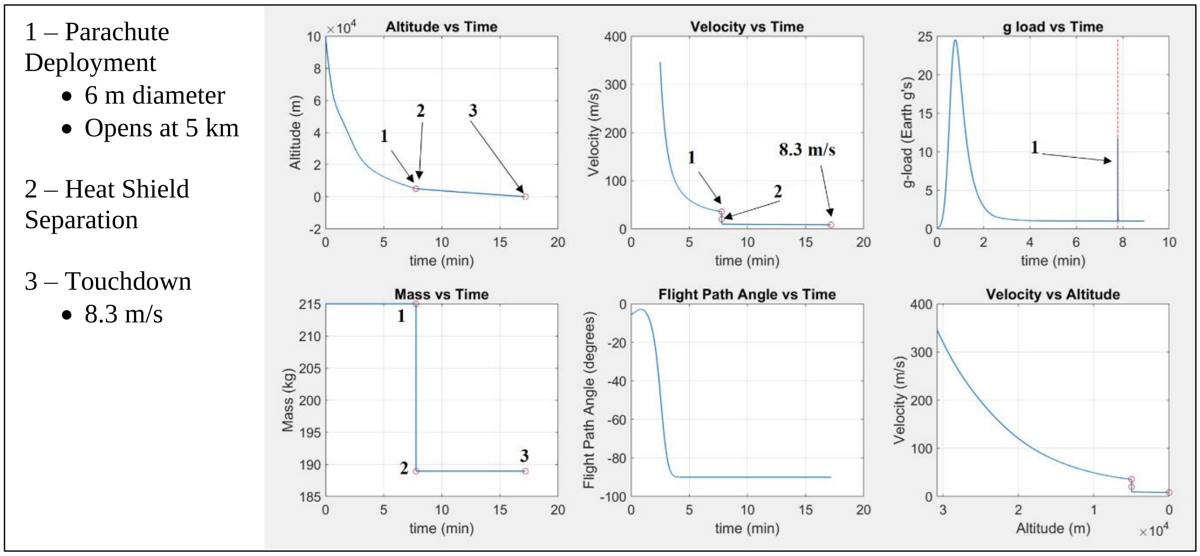


Figure 3.7-7 Earth Entry Trajectory

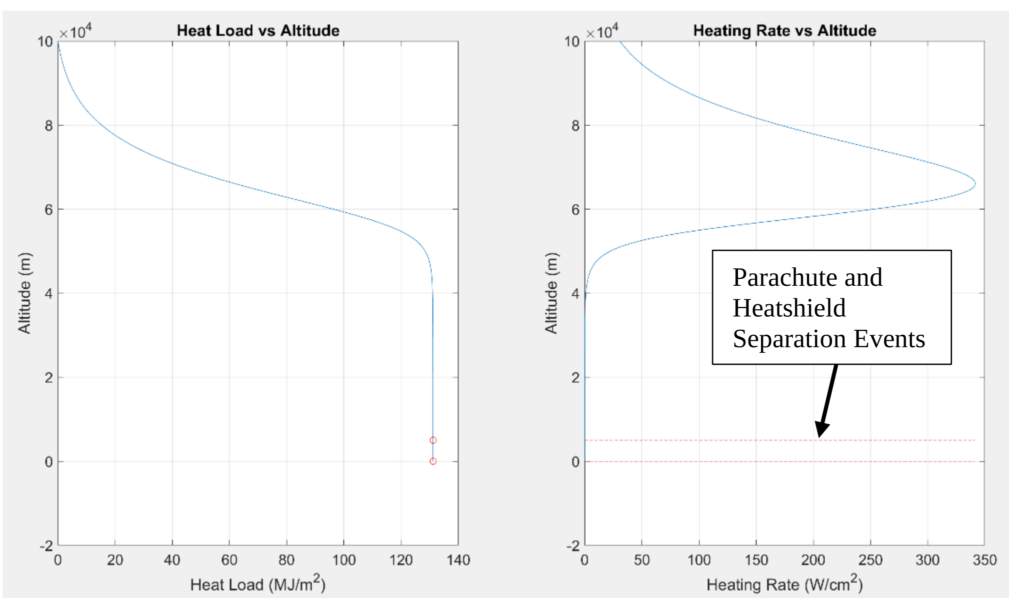


Figure 3.7-8 Earth Entry Heating

3.8 Launch Vehicle Selection

In determining the optimal launch vehicle to utilize for Earth launch to Mars, the following figures of merit were considered: the payload fairing capacity, the payload mass capacity to Mars, and the cost. The launch vehicle needed to support the system dimensions and mass at a minimal cost. **Table 3.8-1** details the total launch mass of the system and highlights the mass requirement for launch.

Table 3.8-1 Total Masses of System

Total Masses	Budget (kg)	Current (kg)
OODM	3,473	4,297
On Orbit Wet	6,887	7,568
Adapter	570	621
Launch Mass	7,457	8,189

The launch vehicles assessed are the Falcon Heavy, Atlas V-551, and the Falcon 9. **Table 3.8-2** presents each launch vehicle's capacities and costs. Payload capacities for each vehicle are provided for a C3 of 9.19 $\frac{km^2}{s^2}$. The Falcon Heavy is the only launch vehicle that met the system's dimensions and mass. The graph shown in **Figure 3.8-1** below details the trade study results, displaying the Falcon Heavy with the highest weighted total. The Falcon Heavy will provide an available mass margin of 4,984 kg for the budget and 4,251 kg for the current design.

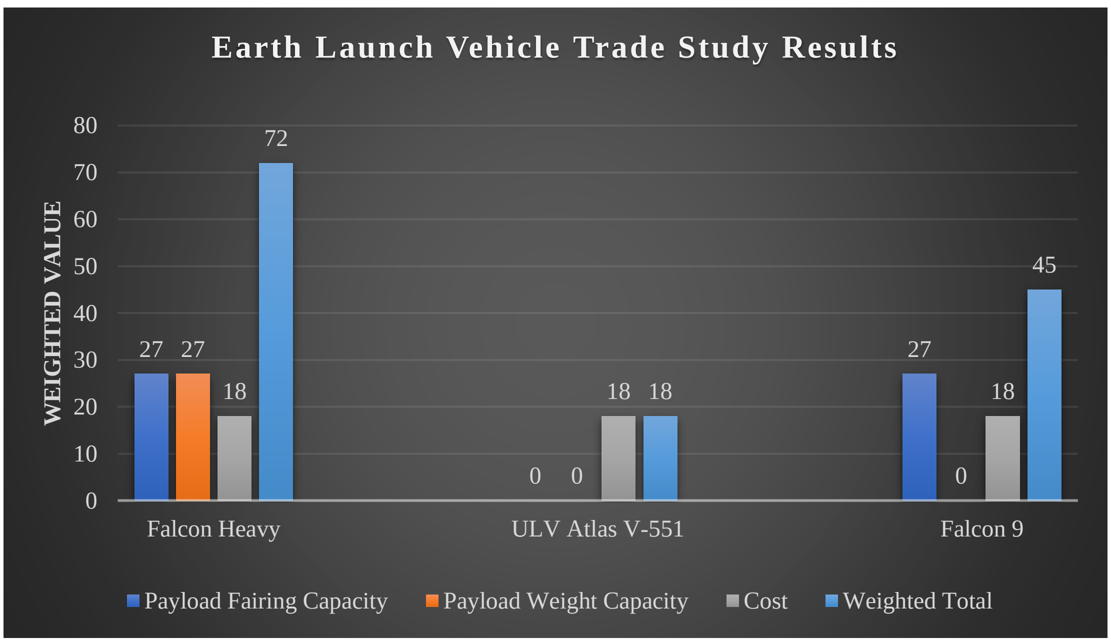


Figure 3.8-1 Earth Launch Vehicle Trade Study Results

Table 3.8-2 Earth Launch Vehicle Trade Study

	Falcon Heavy	**Atlas V-551**	**Falcon 9**
Payload Fairing Capacity:	Height: 13.1 m Diameter: 5.2 m	Height: 10.18 m Diameter: 4.52 m	Height: 13.1 m Diameter: 5.2 m
Payload Weight Capacity to C3 of 9.19 km^2/s^2	12,550 kg	5,145kg	2,305 kg
Cost	$150M	$73M	$62M

A risk associated with the Falcon Heavy is that it may not be RTG certified before launch in 2026. A launch vehicle must be Launch Service Program Category 3 certified to carry RTGs. While the Atlas V and Falcon 9 are certified at this level, they do not meet the requirements for this design. The expendable Falcon Heavy is currently not certified at this level; however, it is on track to become certified after one additional successful flight. Ares Advena has chosen to move forward with the Falcon Heavy and begin the launch approval process to support the current design. NASA Procedural Requirements document NPR 8715.3D lists steps to get a mission approved based on the amount of radioactive material being launched.[13] For two rtg payloads using ^{238}PU, NASA Procedural Requirements states that the launch needs to be reported to and approved by the Nuclear Flight Safety Assurance Manager, reported to Office of Science and Technology Policy, and a concurrence letter from the Nuclear Flight Safety Assurance Manager. This process must be completed no later than 4 months before the expected launch date for the launch to be approved.

3.9 Vehicle Configurations

To successfully conduct the mission, we have identified and modeled multiple vehicle configurations. The first vehicle configuration occurs during Earth launch. The fully assembled launch configuration must fit within a Falcon Heavy payload fairing while not encroaching on the fairing's allotted vibrational envelope.[12] **Figure 3.9-1** shows the orbiter fastened to a standard SpaceX launch vehicle adapter. The orbiter also supports the EDLS utilizing its four main supporting struts. In this configuration, the orbiter's solar arrays are folded in and pose no threat towards violating the payload fairing's volume constraints.

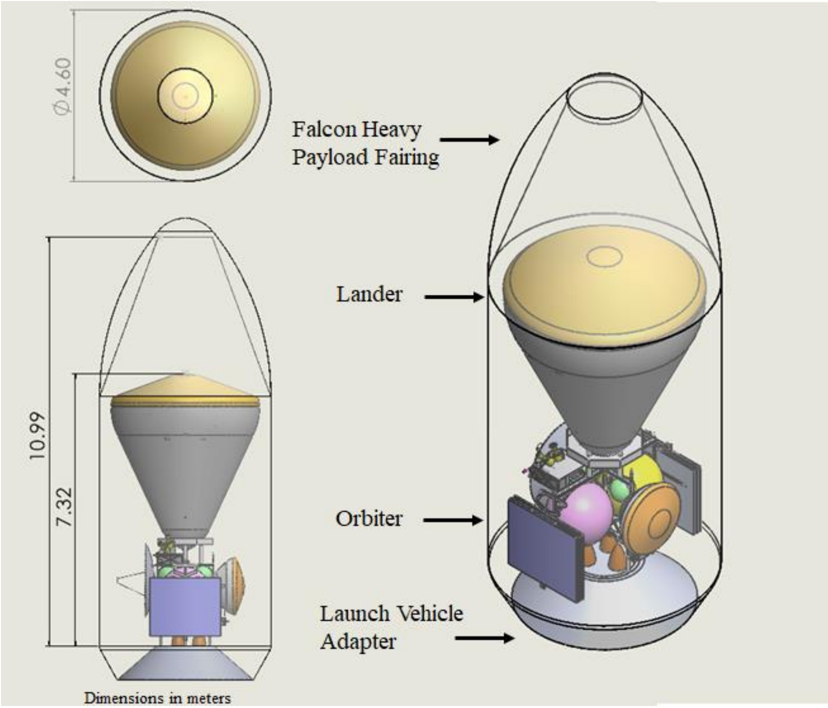


Figure 3.9-1 Launch Configuration

The EDLS cruise stage, shown in **Figure 3.9-2**, represents its configuration after it separates from the orbiter and before entering the Martian atmosphere. The lander structure and supporting equipment were placed and sized to fit within the capsule. We included extra space within the capsule to accommodate the addition of parachutes and their accompanying mortars. In the cruise configuration, the lander's landing legs are retracted inside the landing leg mounts, and the rover is stowed to be nearly lying on its belly. This stowed configuration of the rover allows its structure to be easily attached to the lander's structure using short hardpoints. After landing on the Martian surface, these hardpoints will be released to allow the rover to begin its ground operations.

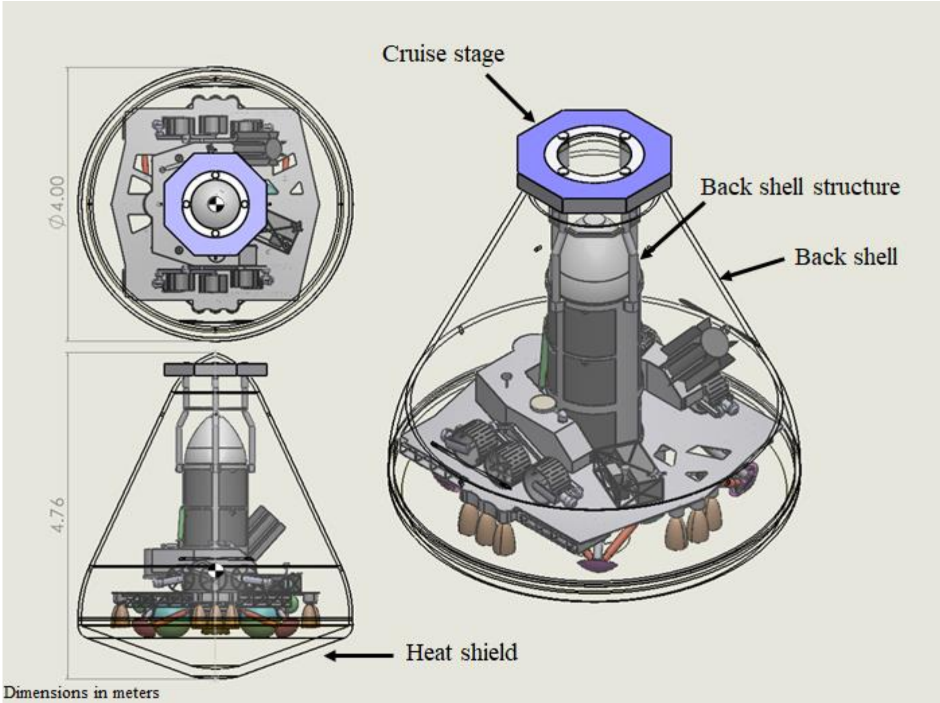


Figure 3.9-2 EDLS Cruise Configuration

Once the orbiter separates from the second stage of the Falcon Heavy, it decouples from the EDLS and becomes an independent spacecraft. **Figure 3.9-3** shows the orbiter in its configuration for the remainder of the Mars ice sample return mission. This configuration shows the orbiter fully operational with its solar arrays deployed.

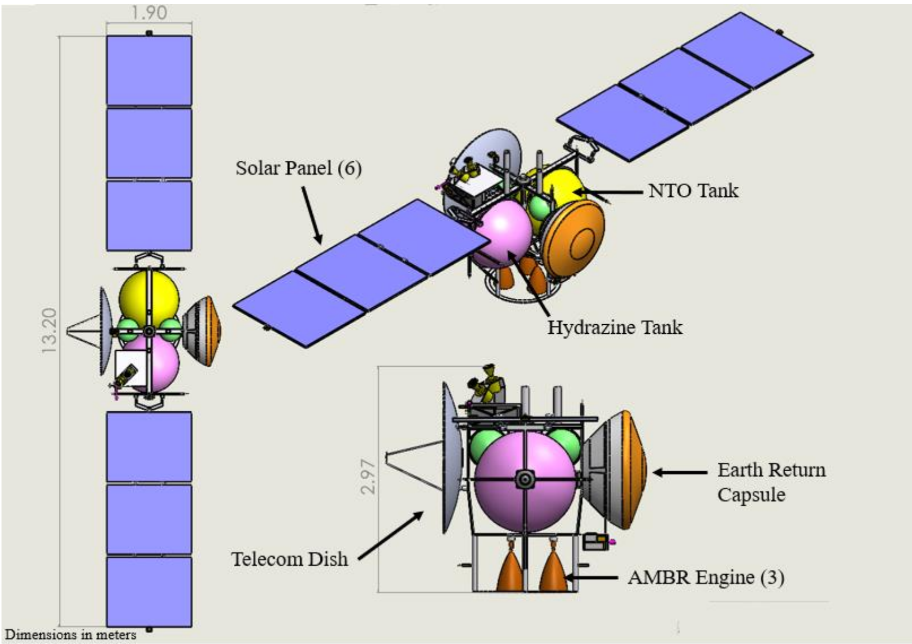


Figure 3.9-3 Orbiter Mission Configuration

4.0 Vehicle Descriptions

4.1 Orbiter

Figure 4.1-1 illustrates the layout for the orbiter's components and subsystems.

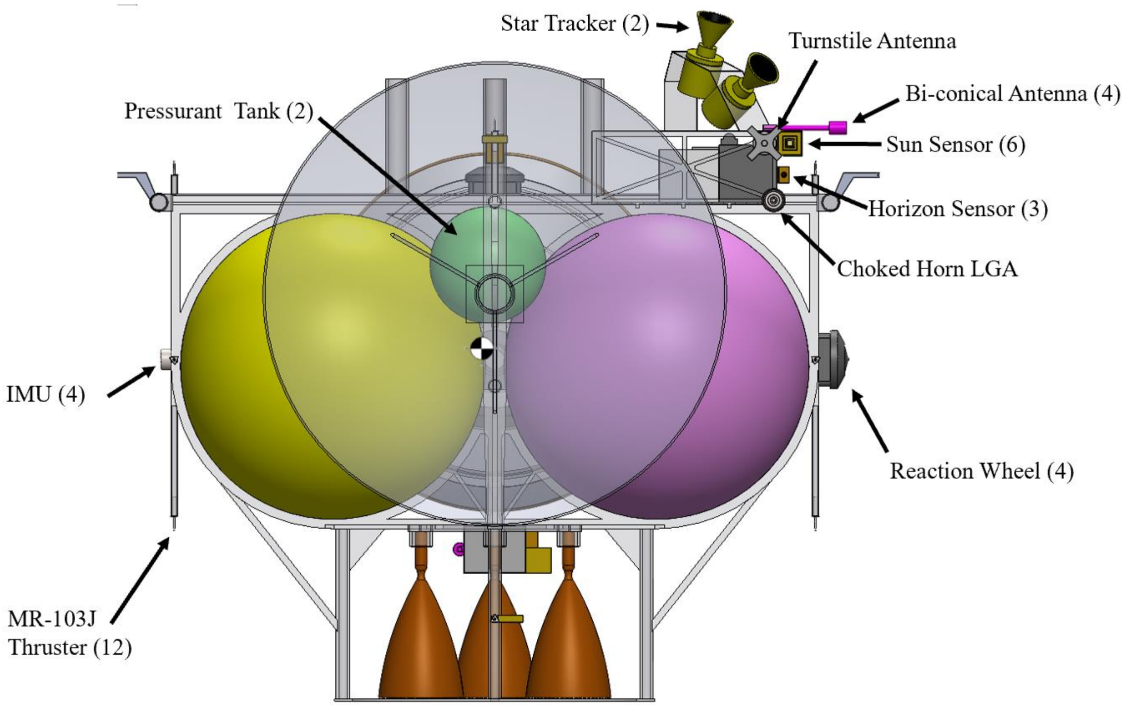


Figure 4.1-1 Orbiter Bus Layout

4.1.1 Derived Requirements

The derived requirements for the orbiter are listed in **Table 4.1.1-1**. The requirements were derived for our orbiter to ensure the design met the customer requirements.

Table 4.1.1-1 Orbiter Derived Requirements

ID	Requirement Statement
ORB.1	The orbiter shall enter an Earth return trajectory no later than October 6, 2028 (± 20 days)
ORB.2	The orbiter shall enter Earth's SOI on May 24, 2029
ORB.3	The orbiter shall be an autonomous vehicle
ORB.4	The orbiter's wet mass shall not exceed 4,184.1 kg
ORB.5	The orbiter shall have a delta-V of 3172 m/s
ORB.6	The orbiter shall perform a 180° turn in five minutes or less.
ORB.7	The orbiter's power summation should not exceed 308.9 W.

4.1.2 Payload

The payload on the orbiter consists only of the sample return capsule. The remaining hardware shown in **Figure 3.9-3** is considered non-payload equipment since they fall under the category of supporting subsystems. The sample return capsule has a mass of 204 kg, which includes an additional 12% mass margin. The capsule uses a 70° sphere-cone PICA heatshield and a Silicone Impregnated Reusable Ceramic Ablator tile aft body. The sample return capsule is responsible for opening before rendezvous, docking, receiving the ice sample container from the MAV, and surviving the Earth atmospheric EDL for collection by Ares Advena personnel.

4.1.3 Propulsion

We conducted a trade study to compare the different propulsion systems for the orbiter. We compared the following systems: electric propulsion, solid propulsion, liquid bipropellant system, and a mono-propellant system. Some figures of merit that drove the trade study included cost, mass, TRL, and storability. Since the orbiter will be stationed in a low Mars orbit, the propellant must be storable with minimal loss for the entire mission. A high TRL and low mass would reduce the overall cost of the vehicle. The winning result of the trade study was a pressure-fed liquid bipropellant system. The orbiter will contain 1,297 kg of nitrogen tetroxide as an oxidizer, 1,360 kg of hydrazine as fuel, and 14.47 kg of helium as pressurant. This provides a ΔV of 3.18 km/s. The orbiter's parts and instrumentation diagram (P&ID) is shown in **Figure 4.1.3-1**. The orbiter will utilize 3 AMBR engines that will be manufactured by Aerojet Rocketdyne and provide up to 125 lb_f of thrust each.[5]

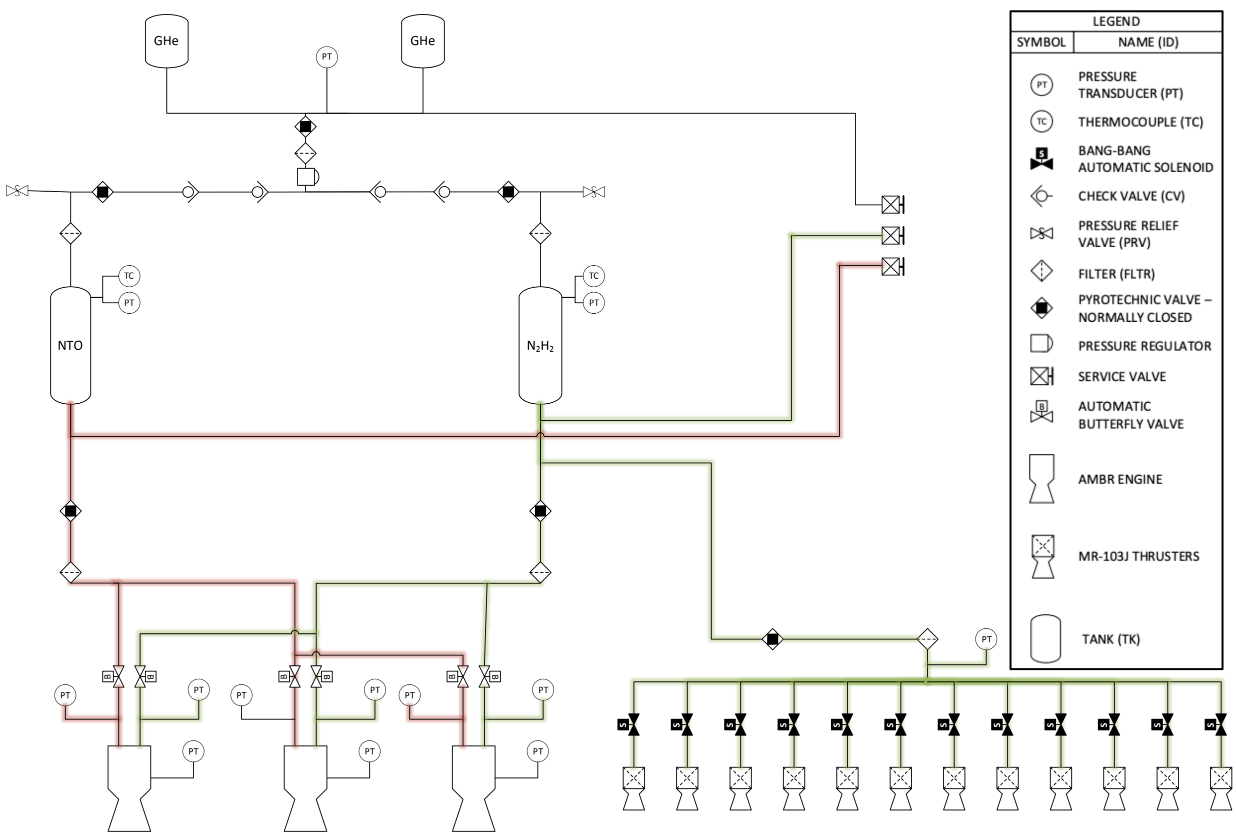


Figure 4.1.3-1 Orbiter's Propulsion Parts and Instrumentation Diagram

4.1.4 Structures

The orbiter's structure material was completely modeled using the computer-aided design software SOLIDWORKS 2020. The structure's construction utilizes I-beams, T-sections, hollow cylinders, and gussets to maintain a high strength to mass ratio. We performed structural load simulations in the SOLIDWORKS 2020 software by using a static load simulation tool. The methodology for conducting these simulations was to find every significant force that would be applied to the vehicle's structure and place that force in the simulation. The orbiter will experience a max load of six Earth g's and must withstand the mass of the EDLS assembly sitting on the four main supporting struts. It must also retain the mass of the fuel tanks, pressurant tanks, telecommunications dish, and the sample return capsule. These forces were simulated and shown in **Figure 4.1.4-1.**

The material used for the structure of the orbiter is an aerospace-grade 2195 aluminum-lithium alloy. This material was selected for its higher strength properties and lower density when compared to the commonly used 6061-T6 aluminum. In **Figure 4.1.4-1**, the highest encountered stress was located at a gusset near the rocket engine housing. It experienced a stress of 505 MN/m^2, which is sufficiently below the 600 MN/m^2 of the ultimate yield stress of the material and gives the overall build a margin of safety of 0.188 for yield with a mass of 191 kg.

Figure 4.1.4-1 Static Load Analysis of the Orbiter's Primary Structure

4.1.5 Attitude Determination and Control System

For the orbiter to successfully complete its mission, it needs to perform maneuvers to capture into Martian orbit, accurately point its high gain antenna (HGA) at Earth, dock with the MAV, and perform the escape burn. To satisfy these requirements, the orbiter was designed to be three-axis stabilized using reaction wheels and reaction control system (RCS) thrusters. Reaction wheels will be used for the fine pointing accuracy needed to communicate with Earth, and the RCS thrusters will be used for large slew maneuvers and reaction wheel desaturation. Based on the requirements stated above, requirements for the orbiter's attitude determination and control system (ADCS) were derived and shown below in **Table 4.1.5-1**.

Table 4.1.5-1 Attitude Determination and Control System Requirements

ID	Requirement Statement
ORB.6	The orbiter shall perform a 180° turn in five minutes or less.
ORB.6.1	The RCS thrusters should provide a minimum thrust of 0.11 N
ORB.6.1.1	The RCS thrusters should provide a maximum thrust of at least 0.23 N
ORB.6.1.2	The RCS thrusters should provide a minimum of 70,000 N-s of total impulse.
ORB.6.1.3	The RCS thrusters shall provide a minimum of 10,000 pulses.
ORB.6.2	The orbiter shall have a pointing accuracy of 0.119° or less.
ORB.6.3	The reaction wheels shall store 10-12 N-m-s of angular momentum, allowing about one desaturation per orbit.
ORB.6.3.1	The reaction wheels shall provide up to 0.05 N-m of torque.

The design of the attitude control system began with finding the worst-case disturbance torques that would be acting on the orbiter in Mars orbit. Next, those disturbance torques were used to aid in the design of a linear attitude control system for the orbiter. Running simulations with the control system allowed an appropriate reaction wheel to be selected, which determined the number of needed desaturation maneuvers to be calculated. We then established durability requirements for the RCS thrusters. Finally, simple equations were used to determine the thrust the RCS thrusters needed to perform a 180° turn in five minutes or less. Results and trade studies for the design of the ADCS are shown throughout this section.

First, the worst-case disturbance torques for the orbiter were calculated using the methods described in Chapter 5 of Brown.[1] Torques from solar radiation pressure (SRP), gravity gradient, drag, and magnetism were all considered. The values of the torques are shown below in **Table 4.1.5-2**.

Table 4.1.5-2 Disturbance Torques for Orbiter on Mars Orbit

	Gravity Gradient	SRP	Drag	Magnetic	Total
x-component	-1.44E-03	7.4E-05	5.47E-04	1.6E-07	-8.26E-04
y-component	1.22E-05	7.4E-05	5.47E-04	1.6E-07	6.34E-04
z-component	1.19E-03	7.4E-05	5.47E-04	1.6E-07	1.81E-03

Next, the torques shown above were inputted into the linear equations of motion for a spacecraft with three reaction wheels in Mars orbit. Since the equations account for gravity gradient, only the sum of the SRP, drag and magnetic torques were treated as external step disturbances. The control system was designed with an optimal control method called Linear Quadratic Regulator (LQR) control. We used this method to tune the controller to minimize the reaction wheel torque needed while still achieving acceptable response times and steady-state errors. The loop is closed by feeding back the full state of the spacecraft (angles and angular rates) and multiplying it by an optimal gain matrix. A block diagram of the closed-loop control system is shown below in **Figure 4.1.5-1**. Minimizing the reaction wheel torque would allow a smaller reaction wheel to be used, saving mass, power, and cost. The implementation of the control system during the mission will be handled by the orbiter's command and data handling system.

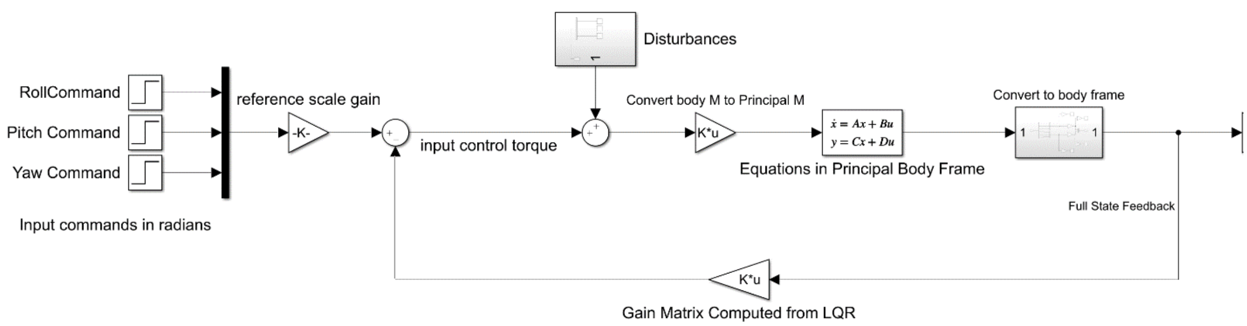


Figure 4.1.5-1 Orbiter Control System Block Diagram

The control system was modeled in MATLAB/Simulink. After we tuned the controller and ran multiple simulations, the system achieved performance requirements. Response characteristics for a 1° step response in each axis are shown below in **Table 4.1.5-3** and **Figure 4.1.5-2**. The steady-state error is below the maximum allowable values of 0.119 degrees to point the HGA at Earth. The low error is also beneficial for docking with the MAV. The settling time is short compared to the orbit period, which would not interfere with the amount of time available for telecommunications. Finally, the input torque needed is not enough to saturate most reaction wheels available on the market. Requirement **ORB6.3.1** from **Table 4.1.5-1** was derived based on this analysis.

Table 4.1.5-3 Step Response Characteristics for Orbiter (1° Step Command)

	Steady State Error (deg)	Settling Time (s)	% Overshoot	Max Input Torque Needed (N-m)
Roll	0.022	244	2.212	0.039
Pitch	0.016	184	2.408	0.039
Yaw	0.008	232	0.833	0.032

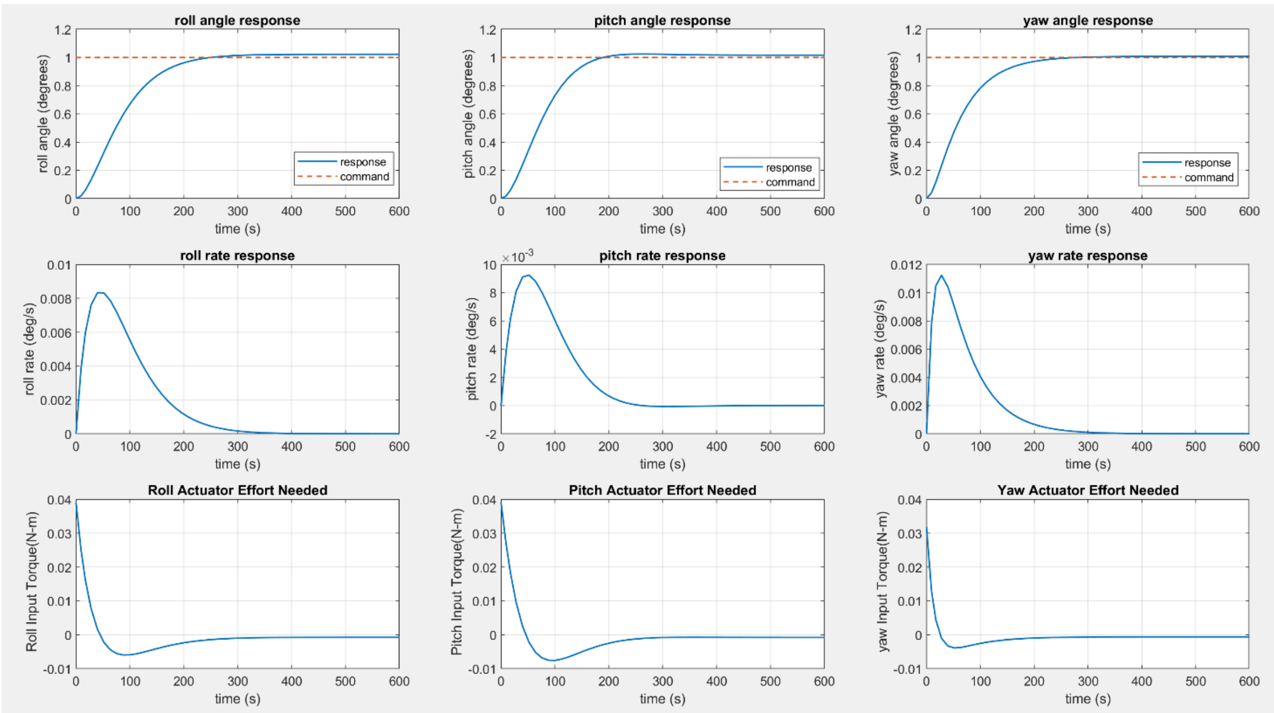


Figure 4.1.5-2 Step Response Characteristics for Orbiter (1° Step Command)

With the reaction wheel performance defined, we conducted a trade study to decide which reaction wheel would be best suited for our mission. We considered four commercial off-the-shelf (COTS) reaction wheels, and we performed a trade study based on the figures of merit shown below in **Table 4.1.5-4**. Minimizing mass and power usage to reduce cost was a priority for the entire mission, so both were weighted more heavily in this study. Since all the considered reaction wheels' operational life was much greater than the length of the mission, lifetime was not chosen to be a figure of merit. A visual of the trade study results is shown in **Figure 4.1.5-3**, and the trade matrix can be found in **Table A.2** in **Appendix A**. We selected the reaction wheel with the highest weighted score.

As seen in **Figure 4.1.5-3**, the HR0610 won because its torque capability matched closest to what the control system required while still having the lowest mass and power usage. It was also the only one to meet the angular momentum storage requirement.

Table 4.1.5-4 Reaction Wheel Trade Study Figure of Merit

FOM #	Figures of Merit		Weight Factor	
FOM.1	Power Usage	The less power, the better.	3	Attitude Control is taking up a lot of power, so it needs to be minimized.
FOM.2	Mass	The lower the mass, the better.	2	The vehicle mass should be minimized to lower cost and to meet LV capability.
FOM.3	Max Torque	The max torque should be around -0.05 N-m or more.	2	The max torque does not need to be very high to meet control requirements.
FOM.4	Temperature Range	The higher the temperature range, the better.	1	A higher operating temperature range is ideal but not as important as there will be a thermal control system.
FOM.5	Angular Momentum Storage	The wheel should be able to store 10-12 N-m-s.	1	The wheel should store enough momentum to require only about one desaturation per orbit.

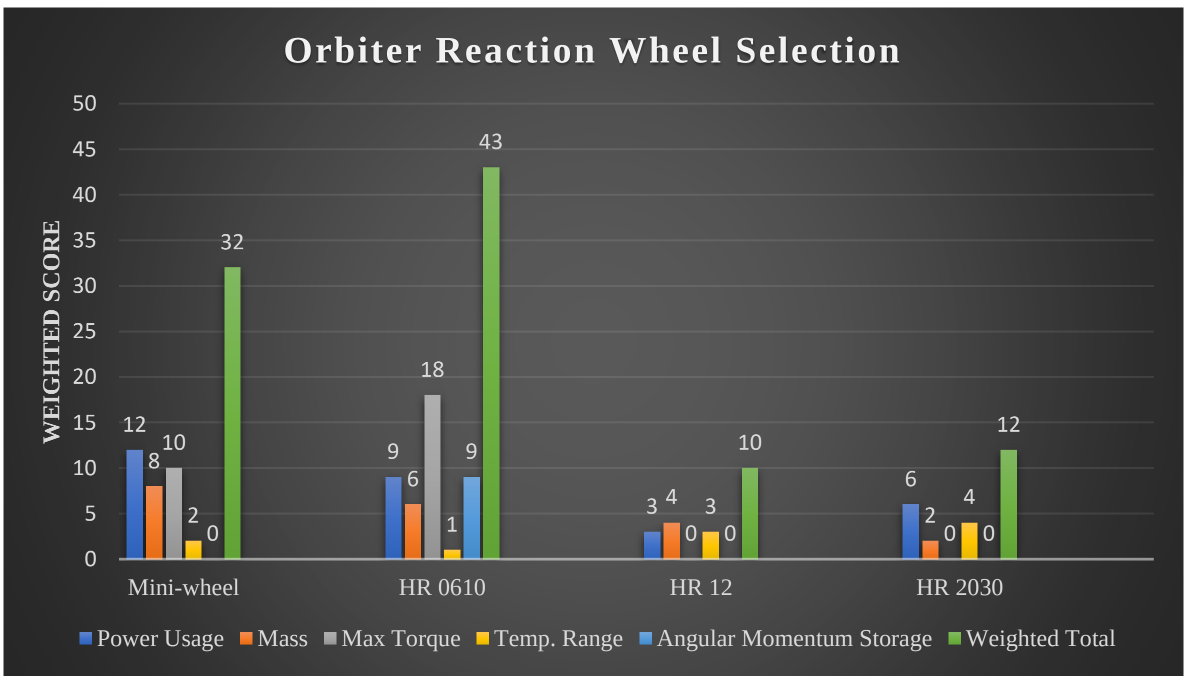


Figure 4.1.5-3 Orbiter Reaction Wheel Trade Study Results

With the reaction wheel selected, the amount of desaturation maneuvers could be calculated using the wheel's angular momentum storage and the calculated worst-case disturbance torques shown in **Table 4.1.5-2**. Using methods outlined in Chapter 5 of Brown[1], we found that the RCS thrusters would need to perform, at most, 5,400 desaturation maneuvers for the entire 414 days on orbit at Mars. The thrusters would have to provide at least 5,400 pulses and 35,000 N-s of impulse. The number of pulses and the total impulse drove the thruster life requirements. The thrusters were required to provide at least double the number of pulses and total impulse. Finally, for the orbiter to perform an orbit insertion and escape at Mars, it should be capable of performing a 180° turn in 300 seconds or less. Using simple dynamics equations outlined in Chapter 4 of Brown[1], it was determined that the range of thrust needed to perform this maneuver in each axis was 0.11 N to 0.23 N. Utilizing the defined thrust and life requirements for the orbiter, we performed a trade study to select an appropriate RCS thruster. Four COTS thrusters that matched the requirements were found and considered for the trade study. The figures of merit are shown below in **Table 4.1.5-5**. Thruster life was weighted the heaviest in this trade study because thruster failure would likely lead to loss of mission. Power usage and specific impulse were weighted second heaviest because it was a major focus to minimize power and mass. The thrust range was not weighted as high as the life because the orbiter could still meet maneuver requirements if the thrust range is not perfect. A visual graph of the results is shown in **Figure 4.1.5-4** and the trade matrix is shown in **Table A.3** in **Appendix A**.

Table 4.1.5-5 Orbiter RCS Thruster Trade Study Figures of Merit

FOM #	Figures of Merit		Weight Factor	
FOM.1	Total Pulses	Engine should be able to provide at least 10,000 pulses.	3	The engines need to be able to start and stop enough to do all commanded and desaturation maneuvers.
FOM.2	Total Impulse	Each engine should be able to provide a total impulse of 70,000 N-s.	3	The engines should be able to provide enough impulse for all maneuvers.
FOM.3	Power Usage	Each engine should use 17W or less.	2	The engines should have a lower power usage to minimize the spacecraft power needs.
FOM.4	Specific Impulse	The engine should have an average I_{sp} of 215s.	2	The engine should have a high I_{sp} to lower fuel consumption.
FOM.5	Thrust Range	The engines range of thrust should include 0.11-0.23 N.	1	The engine should provide the minimum thrust needed to meet the maneuver requirements.
FOM.6	Fuel Hazard Level	It is ideal for the fuel used to have a low hazard level.	1	Low hazard level fuels simplify fuel loading and are safer. This could potentially reduce cost.
FOM.7	Operating Temperature Range	The system with the larger temperature range is better.	1	The system should be usable at a high range of temperatures for reliability.

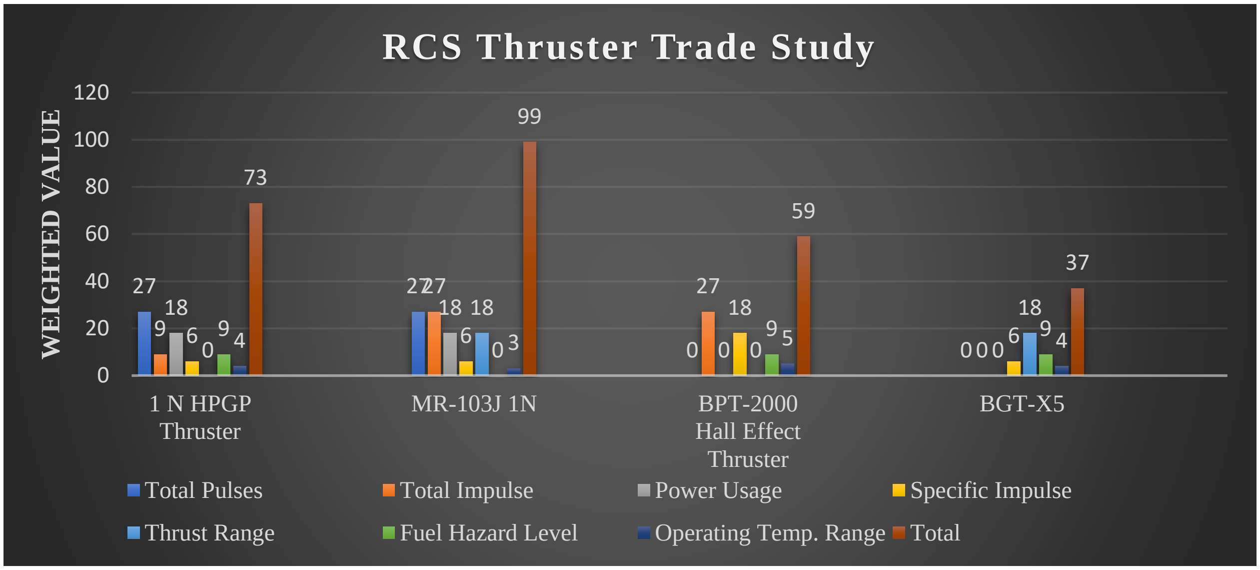


Figure 4.1.5-4 Orbiter RCS Thruster Trade Study Results

In **Figure 4.1.5-4**, the MR-103J from Aerojet Rocketdyne was chosen. It had the best lifetime, and its thrust range matched closest with the requirement. It also had relatively low power usage and high specific impulse. **Figure 4.1.5-5** below shows the 180° maneuvers that the orbiter achieves in each axis using the MR-103J. Note that the response for the z-axis is faster because its moment of inertia is smaller, and the thruster cannot throttle down to a low enough thrust, but still meets the maneuver requirement. The response is the same as the x- and y-axes since the thruster is capable of throttling to the desired thrust. With the thruster selected, the amount of fuel needed for all the desaturation maneuvers was found to be 50 kg. This fuel, along with extra fuel needed for other maneuvers, will be included in the orbiter's main hydrazine tank since both the MR-103J and the orbiter's main engines use hydrazine. The MR-103J meets the desaturation and slew maneuver requirements and has the durability to be used for our entire mission.

Finally, the orbiter has multiple sun sensors, star trackers, gyroscopes, and horizon sensors for determining its attitude. Each piece of attitude determination hardware has at least one redundant backup. There is also a docking sensor to help the orbiter dock with the MAV. This equipment provides the full state feedback that the control system needs and provides telemetry for mission controllers on Earth. A list of all the hardware that the ADCS needs is shown below in **Table 4.1.5-6**. All the equipment listed is COTS and radiation-hardened, and their field of view plots can be seen in **Figure 4.1.5-6**.

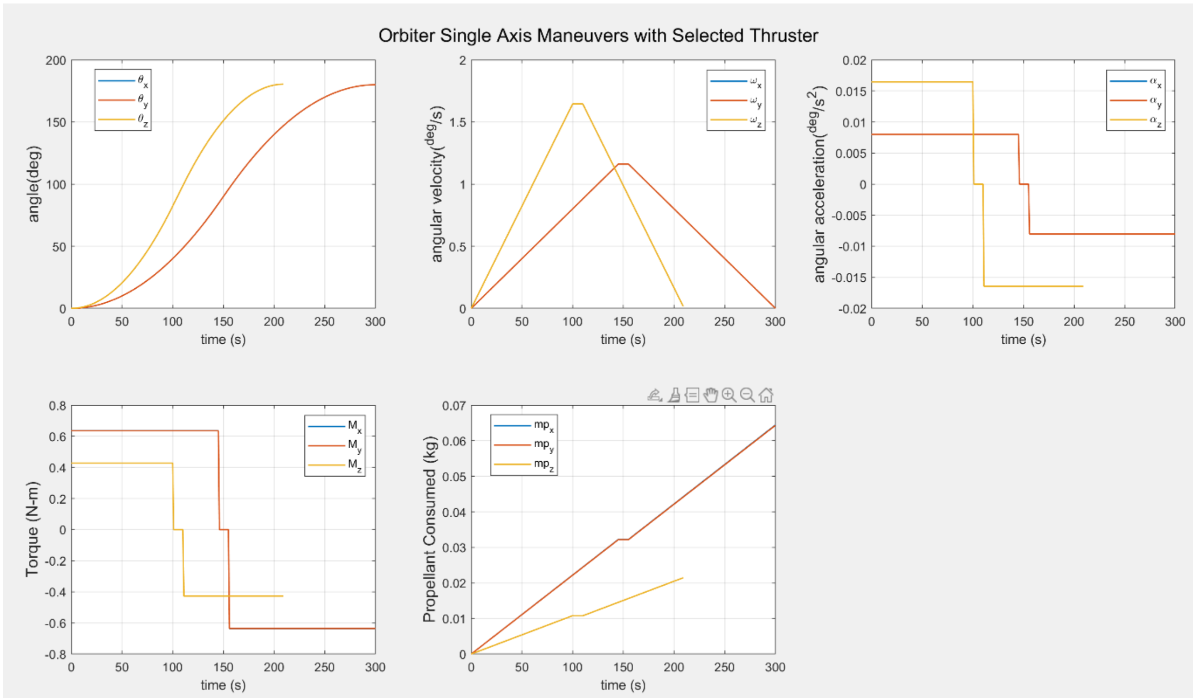


Figure 4.1.5-5 Orbiter 180° Slew Maneuvers in Each Axis with MR-103J Thrusters

The redundant reaction wheel will be mounted 45° to all the axes to provide partial control if one of the wheels fails. All data will be fed to the command and data handling system to be implemented in the control system and stored to be downlinked when possible. The orbiter's ADCS has been designed to successfully complete the mission efficiently within the given constraints.

Table 4.1.5-6 Orbiter ADCS Hardware

Item	Quantity	Individual Mass (kg)	Individual Power (W)	Summed Mass (kg)	Summed Power (W)
MR-103J Thruster	12	0.110	0.00	1.320	0.00
Engine valves	12	0.200	8.250	2.400	99.0
Valve/Catalyst Heater	12	0.065	7.860	0.780	94.32
Bradford Coarse Sun Sensor	6	0.215	0.00	1.290	0.00
GG1320AN gyro	6	0.454	1.600	2.724	9.60
SITAEL S.p.A Digital Horizon Sensor	4	0.400	2.00	1.60	8.0
Leonardo AASTR Star Tracker	2	2.60	12.60	5.20	25.2
HR 0610 Reaction Wheel	4	5.00	15.00	20.0	60.0
RVS 3000-3D Docking Sensor	1	14.00	85	14.0	85
			Total	**49.3**	**381.1**

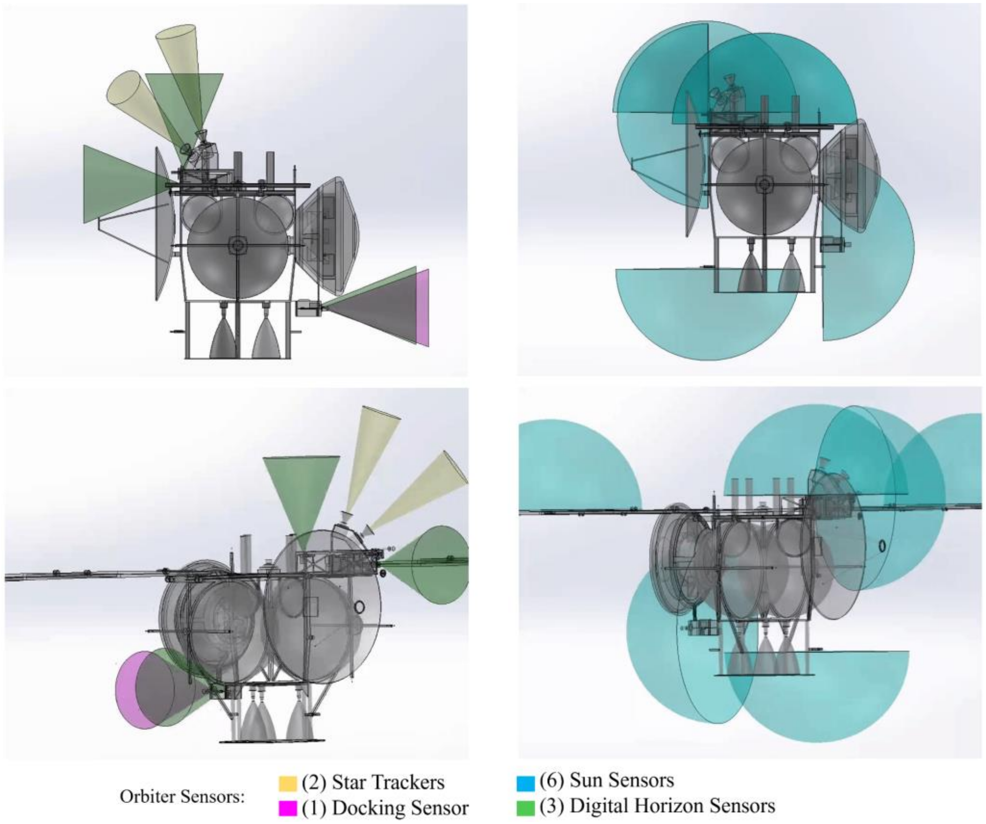


Figure 4.1.5-6 Field of View Plots for the Orbiter's Sensors

4.1.6 Thermal Control and Analysis

The orbiter's thermal control system is designed with passive thermal control components by utilizing multilayer insulation (MLI). This prevents heat from solar radiation from entering into the orbiter and minimizes heat loss to interplanetary space. The thermal analysis on the vehicle was performed by utilizing ANSYS to determine the temperature on external surfaces and internal components of the orbiter. The thermal analysis is carried out by assuming the presence of MLI to maintain the internal components of spacecraft at room temperature and analyze the temperature gradient for multiple cases. A total of five cases have been carried out for this thermal analysis. Four scenarios of worst-case hot occur when the spacecraft is directly subjected to solar flux at Earth at four different angles. An additional scenario is the worst-case cold when the orbiter is in complete darkness at Mars. The details of the analyses are shown in **Figure 4.1.6-1** and **Figure 4.1.6-2**. The temperature range is expected to vary from 13ºC inside the orbiter to the maximum of 80ºC on the solar panels. The analysis showed the orbiter satisfies the temperature range requirements of all the orbiter's internal components, as shown in **Figure 4.1.6-3**.

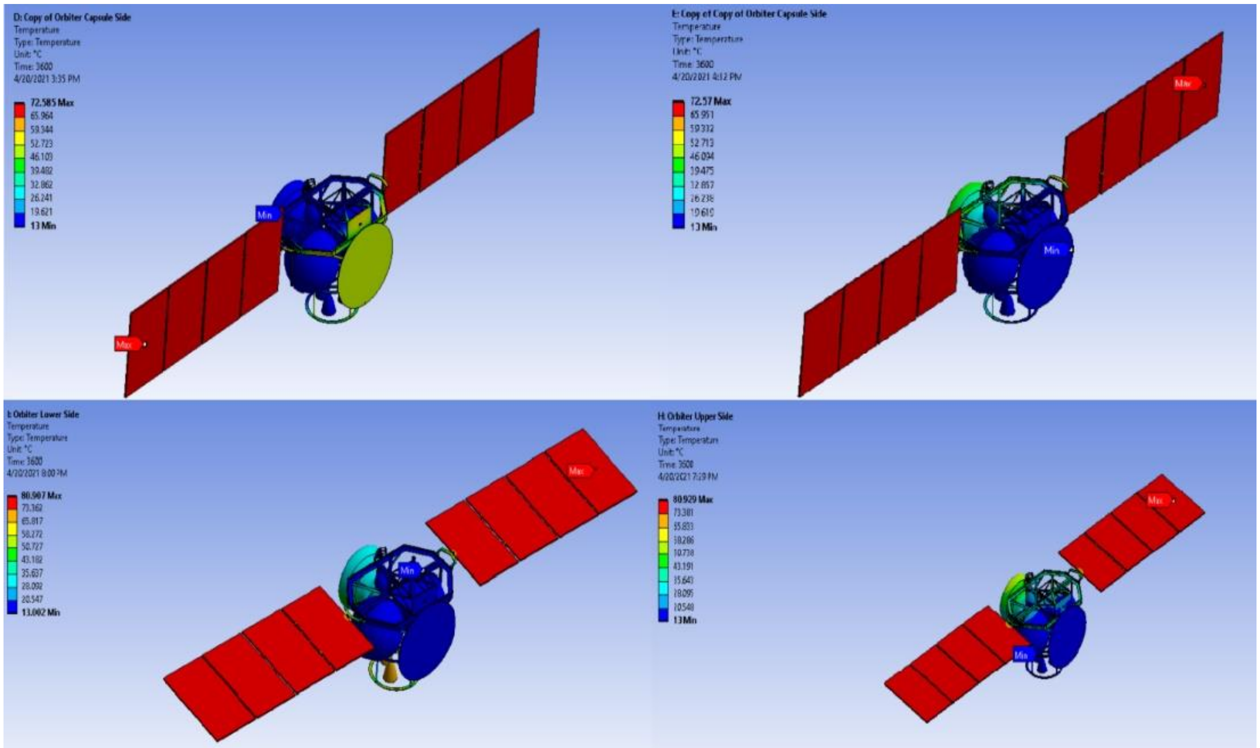


Figure 4.1.6-1 Orbiter Worst-Case Hot

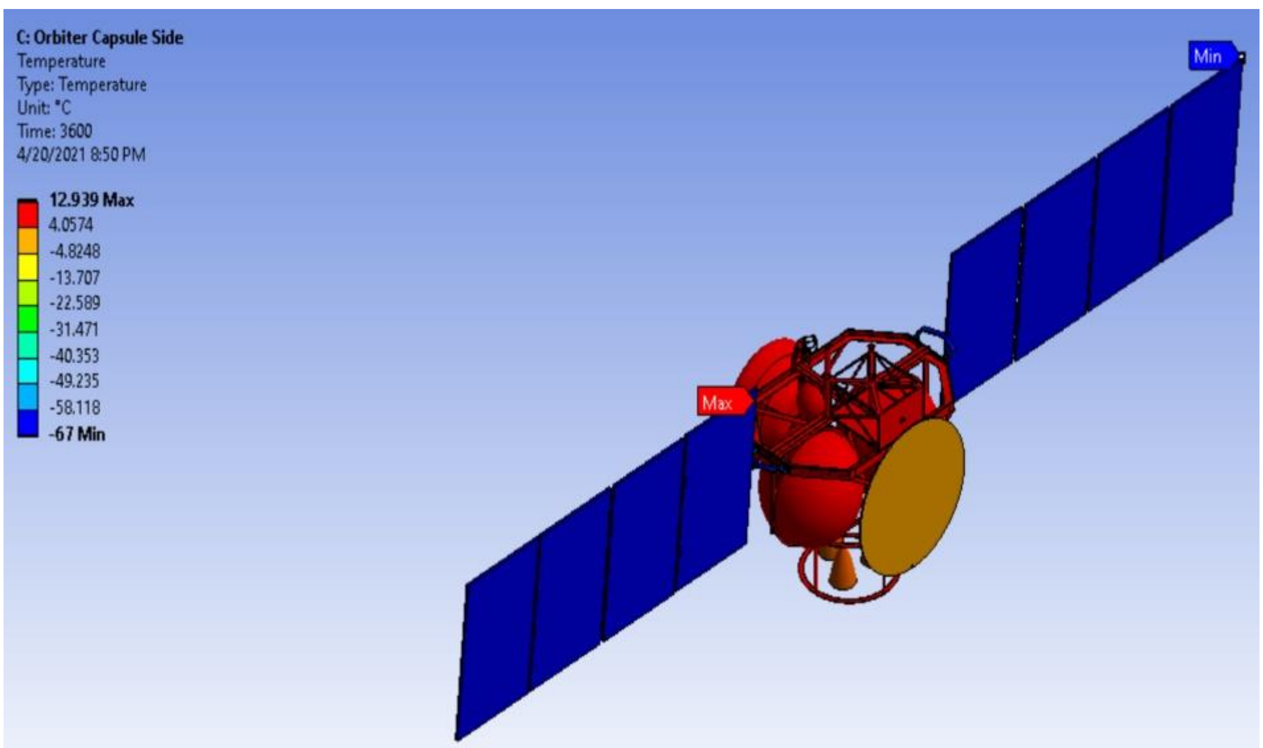


Figure 4.1.6-2 Orbiter Worst Case Cold

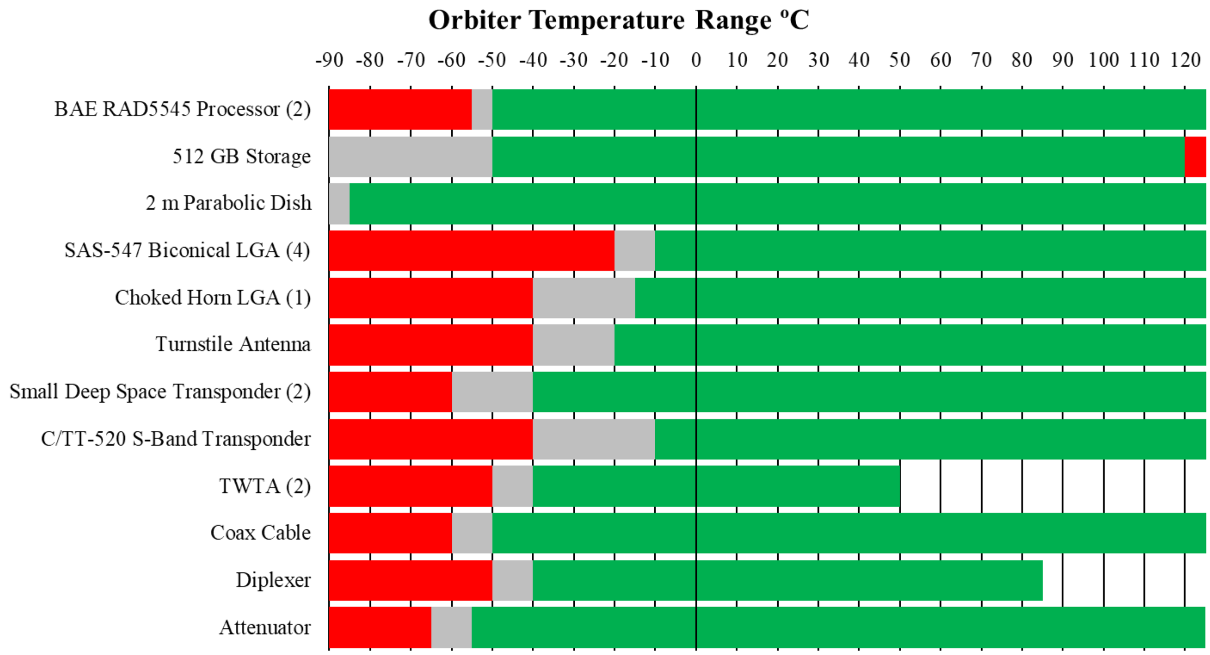


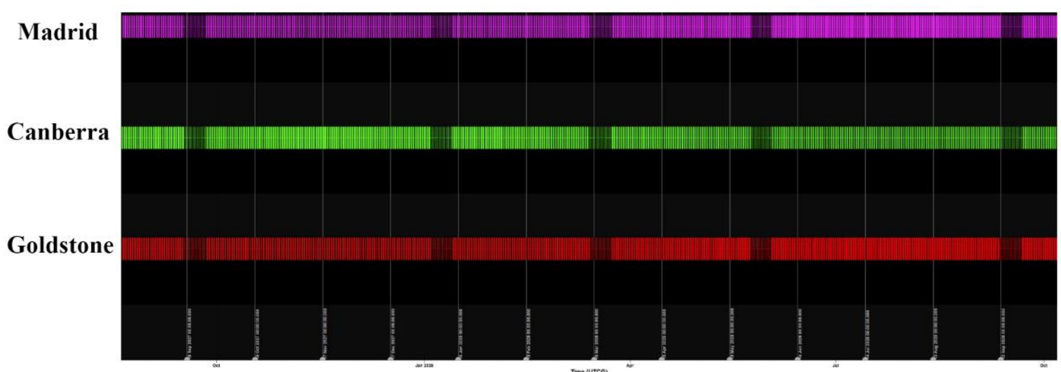
Figure 4.1.6-3 Orbiter Components Temperature Range

4.1.7 Telecommunications

The orbiter will be equipped with a two meter parabolic HGA and multiple low gain antennas (LGAs). The HGA will be used to communicate at a high data rate with the DSN. The LGAs will be used to send and receive emergency signals from the DSN; they will also be used to receive data at a high rate from the rover. The telecommunication links and equipment were sized based on standard link equations outlined in Chapter 9 of Brown[1]. Then the vehicle positions and antenna parameters were modeled in Systems Tool Kit (STK) to validate their performance. STK also provided reports of line-of-sight times from the orbiter to the DSN, rover to the orbiter, and from the rover to the DSN for the entire mission at Mars. These reports showed that the orbiter will have daily contact with the DSN and the rover. However, the rover will not have direct access to the DSN for emergency communications from mid-January to mid-May of 2028 due to the tilt of Mars. During this time, the orbiter will still be able to relay data to and from the rover. **Figure 4.1.7-1** below shows the results from the work done by STK for line-of-sight from the orbiter to the DSN. The graph shows access intervals for the entire mission from August 18, 2027, to October 06, 2028. The report confirms that there is daily contact over the entire mission with no long-term blackouts. The brighter parts of the graph are when the orbiter has shorter access intervals of about an hour, and the dimmer parts are when the orbiter has contact for longer periods of time (10-14 hours per day). Below the graph is a sample of the line-of-sight report and to the right is a summary of the average and total times the orbiter can communicate with the DSN.

This data was helpful for scheduling with the DSN. Since the DSN almost always operates at capacity, minimizing the amount of time needed to communicate with the DSN was a priority. Estimations of the amount of data generated per day by the orbiter and rover were done to find out what data rate the orbiter needed to transmit at and for how long each day. Choosing a higher data rate meant shorter transmission times but would require a larger antenna or increased transmitting power. Using a smaller antenna or using less power would result in longer transmission times. It was determined that the orbiter needed to transmit for 2 hours a day at about 57 kbps to downlink all the data that would be generated. This would require a 2 m dish transmitting a 30 W signal in X-band frequencies. This was an acceptable compromise between transmission time and antenna size, and power. The orbiter can access the DSN for an average of five to eight hours per day, so there is plenty of margin.

To ensure that the DSN can accommodate the mission's needs, we reviewed scheduling issues and recommendations discussed by Johnston et. al.[2] The DSN requirements for scheduling requests are shown below in **Table 4.1.7-1**. We will keep the DSN schedulers updated on the mission's communication requirements by providing them with the information in **Figure 4.1.7-1** and **Table 4.1.7-1.** We will request three hours a day (one hour of margin). By taking these measures, the DSN should be able to provide the required communication time.



Sample From Report of Line of Sign Times:

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (hr)
1	19 Aug 2027 20:00:00.000	19 Aug 2027 20:58:46.133	0.979
2	20 Aug 2027 09:57:33.094	20 Aug 2027 10:16:22.786	0.314
⋮	⋮	⋮	⋮
2662	6 Oct 2028 11:40:12.721	6 Oct 2028 12:35:44.513	0.925
2663	6 Oct 2028 13:34:05.670	6 Oct 2028 14:29:37.640	0.926

Total Contact Time on Mars Orbit	8,855 hrs
Avg Daily Contact Time	5-8 hrs
Avg Interval Contact Time	1 hr

Figure 4.1.7-1 Orbiter to DSN Line of Sight Analysis from STK

Table 4.1.7-1 DSN Scheduling Constraints

Requested Time= 3 hr/day		
Constraint	**Description**	**How It Applies to System**
Reducible	Whether and by how much the requested time can be reduced to fit in an available opportunity	Can be reduced by 1 hour at most
Extensible	Whether and by how much the requested time can be increased to take advantage of available resources	Can be increased by 2-3 hours, for higher quality images
Splitable	Whether the requested time must be provided in one unbroken track, or can be split into two or more	Split is acceptable, most access intervals are 1 hour long.
Split Duration	If splitable, the minimum, maximum, and preferred durations of the split segments; the maximum number of split segments	15 minutes duration minimum. 8 splits max
Split Segment Overlap	If the split segments must overlap each other, the minimum, maximum, and preferred duration of the overlaps	Split segments do not need to overlap
Split Segment Gaps	If the split segments must be separated, the minimum, maximum, and preferred duration of the gaps	Segments not required
View periods	Periods of visibility of a spacecraft from a ground station, possibly constrained to special limits (rise/set, other elevation limits)	orbiter will be blocked by Mars multiple times a day. View periods typically last 1 hour.
Events	General time intervals that constrain when tracks may be allocated; examples include: day of week, time of day (for accommodating shift schedules, daylight), orbit/trajectory event intervals (occultations, maneuvers, surface object direct view to Earth,)	EDL phase and MAV launch and docking phase should be uninterrupted. EDL occurs on Aug 18, 2027. MAV launch and docking occurs on Oct 06, 2028. Emergency downlinks will require long periods of use with the 70 m antennas.

Next, **Figure 4.1.7-2** below shows the data link margin that the orbiter's HGA achieves over the entire mission around Mars; the data was generated with STK. The minimum data link margin achieved is 5.6 dB when Mars is at its maximum range of 360 million km from Earth. The minimum data link margin required is 3.0 dB. Note that this link only requires use of the DSN's 34-meter dishes which are more available than the 70-meter dishes.

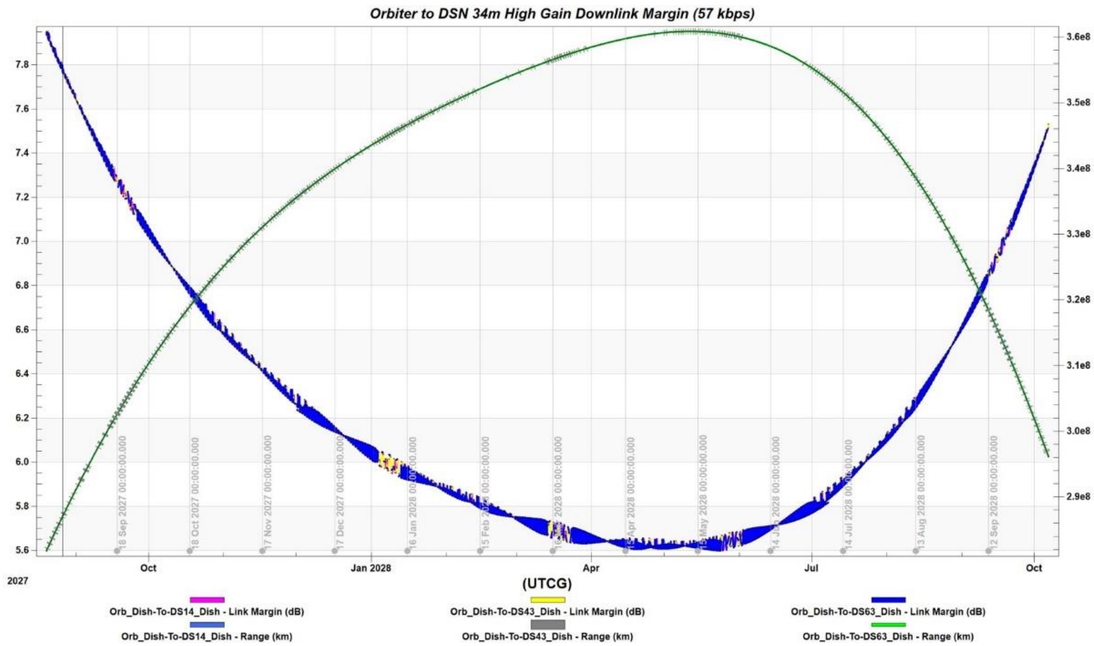


Figure 4.1.7-2 Orbiter to DSN High Gain Downlink Margin

Next, the orbiter's LGAs will provide emergency communications with the DSN as well as high-rate communications with the rover. Emergency situations could include main antenna failure or loss of attitude control. The orbiter will have four biconical antennas to receive emergency X-band signals from the DSN as well as high-rate X-band signals from the rover. It will also include a choked horn LGA to receive S-band signals from the rover's LGAs if the rover's main antenna fails. Finally, the orbiter has a turnstile antenna to send emergency downlink signals in S-band to the DSN; this will require the use of DSN's 70 m dish. It will also use the turnstile antenna to communicate with the lander and MAV. Similar to what was done for the HGA, these antennas were modeled in STK to verify their link performances. A summary of all the links that include the orbiter is shown below in **Table 4.1.7-2**. The right-most column lists the lowest value the link margin will be during the entire mission. This occurs when Earth and Mars are farthest apart or when the orbiter is low on the horizon with respect to the rover. The MAV and lander will never be more than 2,000 km away from each other while they have line-of-sight. The worst-case scenario occurs between April and May of 2028 as shown above in **Figure 4.1.7-2.** Higher data rates can be achieved outside these dates without increasing transmission power.

Table 4.1.7-2 Orbiter Link Performance

Link	Link Type	Data Rate (kbps)	Min Link Margin (dB)
Orbiter - DSN 34 m	High Gain Downlink	57	5.6
Orbiter - DSN 70 m	Emergency Downlink	0.100	3.2
Rover-Orbiter	High Gain Uplink	2,320	3.0
Rover-Orbiter	Backup High Gain Uplink	2,320	3.0
DSN 34 m - Orbiter	Emergency Uplink	0.257	3.9
MAV-Orbiter	Telemetry (for docking)	7	12.6
Lander-Orbiter	Telemetry (for EDL)	8	16.5

Finally, a list of all the telecommunications hardware that the orbiter requires is shown below in **Table 4.1.7-3**. All the equipment is COTS or easily manufactured. Four biconical antennas are needed to provide 360˚ coverage in all directions; they were chosen because they provide relatively high gain (4 dB) in a toroidal pattern. They will only be used to receive transmissions during an emergency since transmitting on all of them would require too much power. The biconical antennas will receive X-band signals. The orbiter needs small deep space transponders (SDSTs) and an S-band transponder because the SDSTs only work on X-band frequencies. With both sets, signals can be received in both frequencies. There is a redundant SDST but only one S-band transponder because it will only be used in emergencies since it draws twice the power. A traveling wave tube amplifier (TWTA) will excite X-band signals to be sent out of the HGA; there is a redundant TWTA in case of failure. It was important to keep the HGA's transmission power low because the TWTA is only about 22% efficient. To achieve the HGAs 30 W transmission power, over 130 W needs to be input into the TWTA. X-band solid-state amplifiers (SSPAs) were not chosen for this task because they are not much more efficient than TWTAs and cannot produce the transmission power needed. Next, an S-band SSPA will be used to excite signals to be sent out of the turnstile antenna for emergency downlinks. This is because S-band SSPAs are more efficient (53%) than X-band SSPAs or TWTAs. They can achieve the 175 W transmission power needed to send the emergency signals. S-band was chosen to be the frequency range to send the emergency signals because there is less path loss than with X-band frequencies; less power can be used to achieve acceptable link margins. The high data rate is not a priority with emergency communications, so higher frequencies are not needed. Finally, the diplexer allows the system to send and receive signals at the same time. This is needed so no time is wasted with the DSN; commands can be uplinked while data is downlinked. All data links will utilize Reed-Solomon coding to reduce bit errors.

Table 4.1.7-3 Orbiter Telecommunications Hardware

Component	Mass (kg)	Power (W)
2 m Parabolic Dish	21.2	0.0
SAS-547 Biconical LGA (4)	1.8	0.0
Choked Horn LGA	2.1	0.0
Turnstile Antenna	0.3	0.0
Small Deep Space Transponder (2)	6.4	15.8
S-Band SSPA (1)	4.5	0.0
C/TT-520 S-Band Transponder	4.0	39.0
TWTA (2)	9.9	133.7
Coax Cable	7.8	0.0
Diplexer	0.5	0.0
Attenuator	0.10	0.0
Totals	**58.6**	**188.5**

The orbiter will have all the equipment required to provide reliable communications with Earth and the rover. It has redundant communication methods and hardware; all data links were modeled and verified in STK to ensure data link margins stayed above the 3 dB requirement. Scheduling with the DSN is also considered and accounted for.

4.1.8 Command and Data Handling Systems

The orbiter will utilize two BAE RAD5545 multi-core processors to manage the data and commands for the mission. The two will be wired in parallel for redundancy in the case that one fails. They are certified for flight and radiation-hardened. In addition to the processors, the orbiter will use an additional 512 GB solid-state recorder for its data. The orbiter will utilize a Deep Space Atomic Clock to help with the timekeeping for ground control. **Table 4.1.8-1** lists the summation of the masses and powers for each component the orbiter will use for its command and data handling system.

Table 4.1.8-1 Command and Data Handling System Components for Orbiter

Component	Mass (kg)	Power (W)
Deep Space Atomic Clock	17.5	44
BAE RAD5545 Processor (2)	4.0	35.4
512 GB Storage	14	35
Totals	**35.5**	**114.4**

4.1.9 Power Systems

A trade study between solar, RTG, and fission was conducted to determine the power generation system for the orbiter. The figures of merit included cost, mass, and power generation. The winning design was the quadruple junction solar panel due to it staying under five million dollars (FY20), under a mass of 150 kg, and generating 1,467 W. The solar panel area needed to satisfy these power needs is 18 m^2 and has a mass of 111.3 kg. We will use the QL015KA battery cells that are manufactured by Quallion for power storage. These cells have an energy capacity of 142 Wh/kg and will be wired in 4 parallel strings of 8 cells in series. The design only calls for 3 parallel strings, but it includes an additional cell for redundancy. The power system design can provide up to 1,670 Wh and will only be discharged 60% to preserve the long-term capacity of the cells with this configuration.

4.1.10 Mass and Power Statements

The orbiter's mass and power summations are tabulated in **Table 4.1.10-1** and **Table 4.1.10-2** below.

Table 4.1.10-1 Orbiter Mass Statement

Subsystems	Budget (kg)	Current (kg)	Status
Structure	278.4	191.0	C
Thermal	28.8	18.3	C
ACS	96.0	49.3	C
Power	201.6	109.5	C
Cabling	76.8	102.8	C
Propulsion	144.0	164.0	C
Telecom	67.2	58.6	C
CDS	67.2	35.5	C
Total	**960.0**	**728.9**	**C**
Margin	336.0	336.0	C
Payload	216.0	203.9	C
OODM	**1,176.0**	**1,268.8**	**C**
Propellant	2,800.1	2,657.1	C
Pressurant	140.0	14.5	C
On-Orbit Wet Mass	**4,116.1**	**3,940.3**	**C**

Table 4.1.10-2 Orbiter Power Statement

Subsystems	Budget (W)	Current (W)	Status
Thermal	46.7	56.5	C
ACS	15.6	381.1	C
Power	2.8	683.1	C
CDS	21.2	114.4	C
Comms	42.5	188.5	C
Propulsion	5.7	5.7	E
Mechanisms	7.1	7.1	E
Total	**141.5**	**1,436.4**	**E**
Margin	127.4	127.4	E
Payload	40.0	40.0	E
On-Orbit Power	**308.9**	**1,603.8**	**E**

4.2 Cruise Stage and Lander

Figure 4.2-1 illustrates the layout for the lander's components and subsytems.

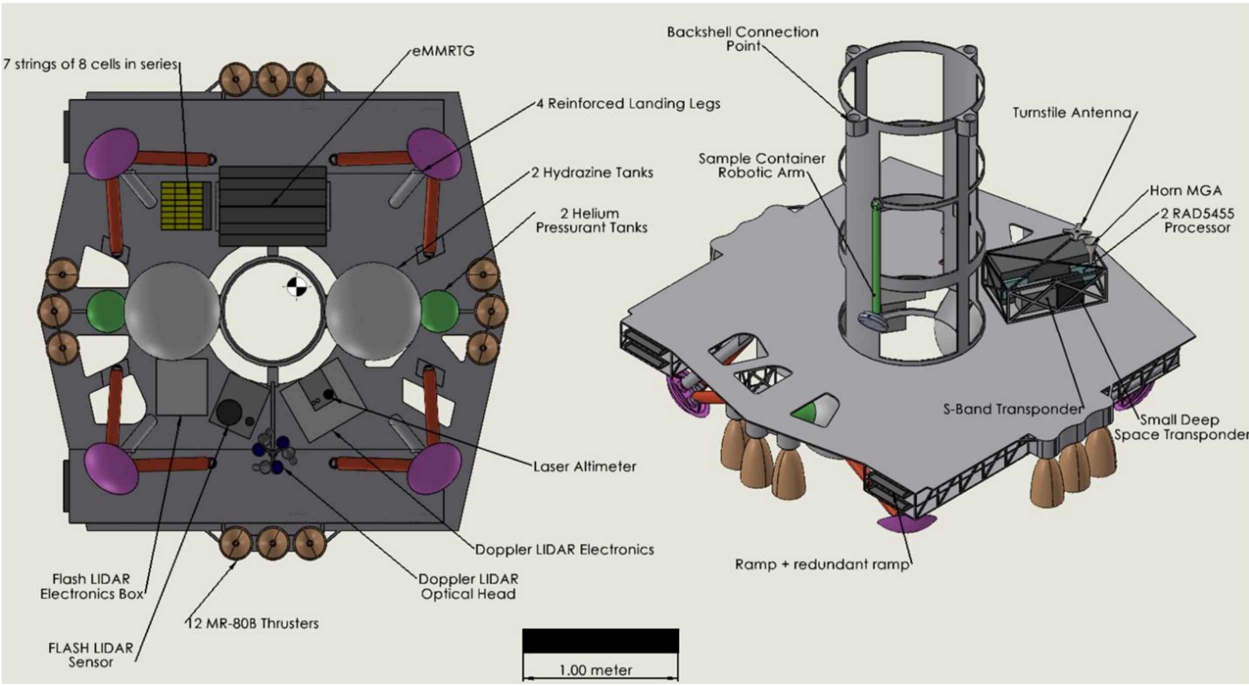


Figure 4.2-1 Lander Component Layout

4.2.1 Derived Requirements

Table 4.2.1-1 shows the derived requirements satisfied by the lander.

Table 4.2.1-1 Lander Derived Requirements

ID	Requirement Statement
LND.1	The lander shall separate from the orbiter no later than November 1, 2026.
LND.2	The lander shall be an autonomous vehicle
LND.3	The lander shall operate for five months in total darkness on Mars
LND.4	The lander shall operate during a minimum of seven months of day/night cycles
LND.5	The mass of the lander in the cruise stage configuration should not exceed 2737.5 kg
LND.6	The lander shall perform a 90-degree roll in five seconds
LND.7	The lander's power summation during cruise should not exceed 1,046 W.
LND.8	The lander shall have a ΔV of at least 250 m/s
LND.9	The lander shall arrive at Mars no later than August 18, 2027.

4.2.2 Propulsion

To land the system safely and to fulfill requirement **LND.10**, the lander has 12 MR-80B engines which have the potential to provide 43,200 Newtons of thrust. These engines are fueled by 200.4 kg of hydrazine held in two tanks. In addition, there are 12 MR-107S engines connected to the hydrazine lines as part of the ACS. The hydrazine tanks are pressurized by the two pressurant tanks, which hold a total of 3.2 kg of gaseous helium at 34.5 MPa. This can be seen in **Figure 4.2.2-1**. The red highlighted lines are representative of the fuel lines present in the system, while the non-highlighted lines are for the pressurant.

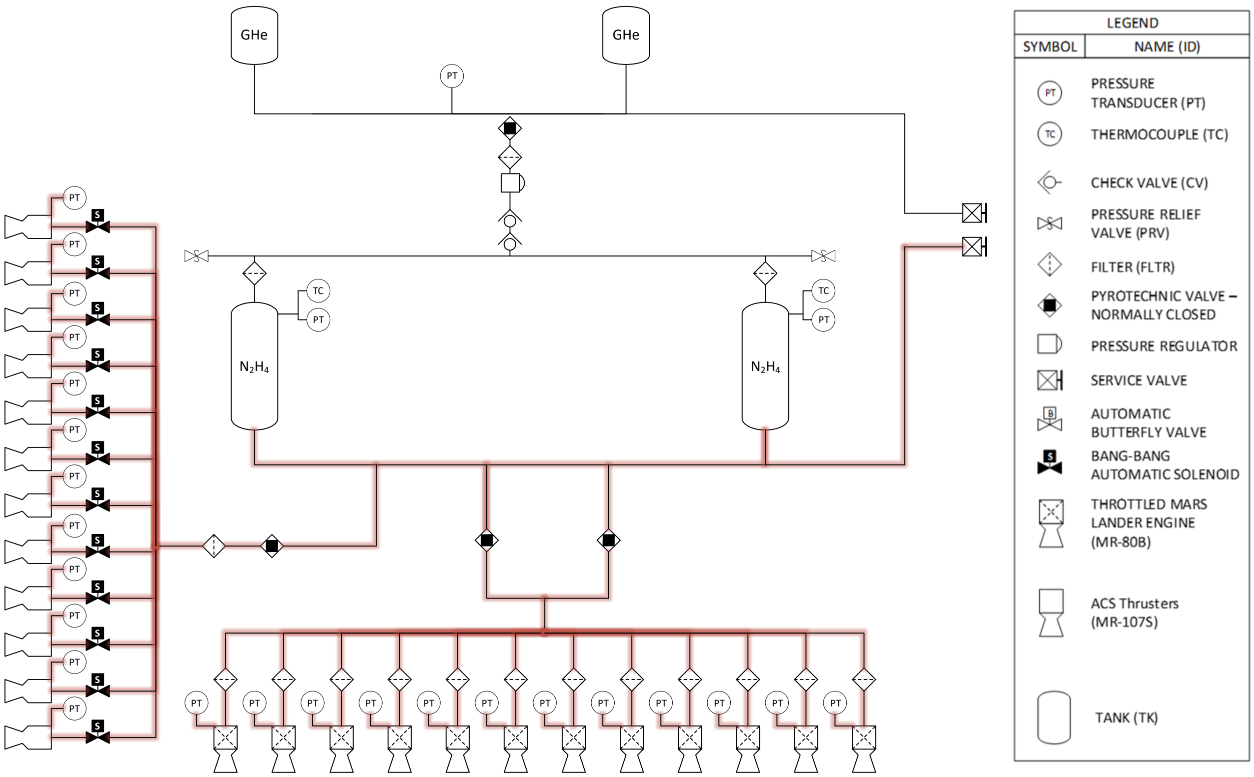


Figure 4.2.2-1 P&ID for the Lander

4.2.3 Structures

A similar methodology as used with the orbiter was used with the structure of the lander for simulating worst-case static loads. The lander experiences its max load of 12 Earth g's during the Mars EDL phase. It experiences stresses from having to support the mass of the back shell structure, MAV, rover, fuel tanks, and supporting equipment. Also built using the 2195 aluminum-lithium alloy, the lander's structure experiences a maximum stress of 430 MN/$m^{2,}$ which gives it a calculated margin of safety for a yield of 0.395 and has a total mass of 430 kg.

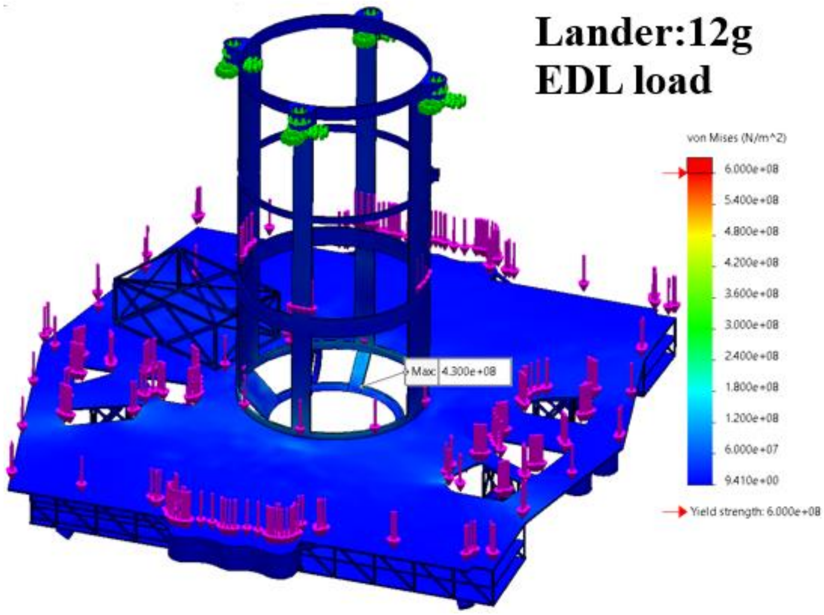


Figure 4.2.3-1 Static Load Analysis of the Lander's Structure

4.2.4 Attitude Determination and Control System

During transit to Mars, the lander's ADCS has the task of keeping the solar panels pointed at the sun with an acceptable error of 10°. However, once it gets to Mars, it has the challenging task of ensuring that the initial entry attitude is correct, then it must change its attitude depending on the changing atmospheric conditions in real-time to accurately guide the lander to the landing site. We determined that the lander must be capable of performing a 90° roll within 5 seconds and have an attitude error of less than 3° during the entry phase.[4] To achieve these requirements, we determined the lander should be 3-axis stabilized using thrusters. Based on the lander's moments of inertia, the MR 107-S thrusters from Aerojet Rocketdyne were chosen to perform the required maneuvers. **Figure 4.2.4-1** below shows that the selected thrusters can perform the 90° roll in 5 seconds. The errors in the other axes are also less than the required 3°. These thrusters will also provide attitude control on transit to Mars and provide course corrections.

Next, the entry attitude controller will utilize a feedforward path, a Proportional/Derivative (PD) controlled feedback path, and deadbands to achieve the required attitude to guide the lander to the landing zone. A block diagram of the controller is shown below in **Figure 4.2.4-2.** The feedforward path is used to achieve fast response times, while the feedback path is used to stabilize the system with PD control. Deadbands are used to account for errors between the actual and predicted trim angle of attack and other attitude errors. They prevent the controller from fighting the actual trim angle of attack, which wastes fuel [4]. The gains and deadbands are scheduled appropriately to meet the diverse needs of EDL.

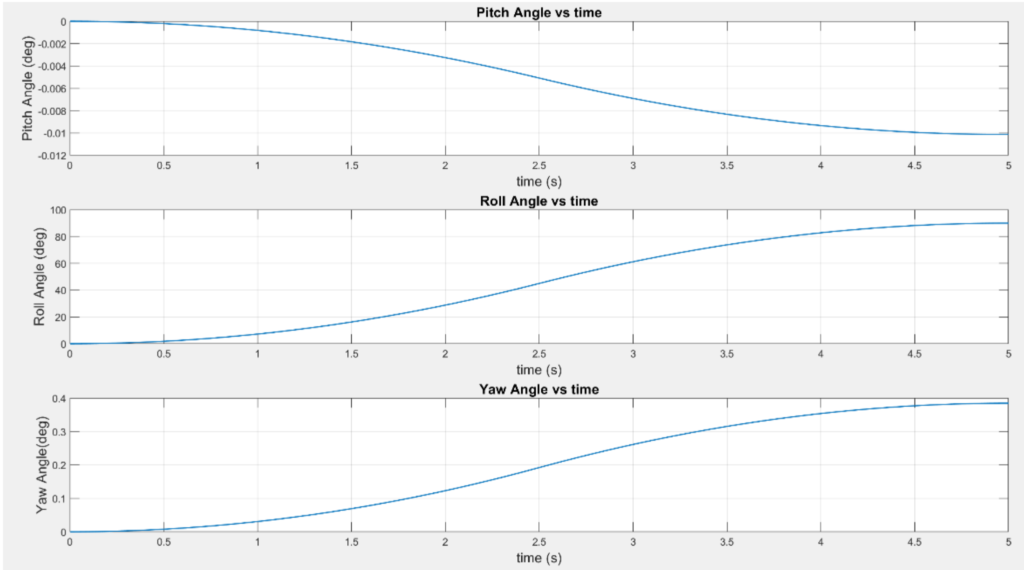


Figure 4.2.4-1 Lander Roll Maneuver for Entry

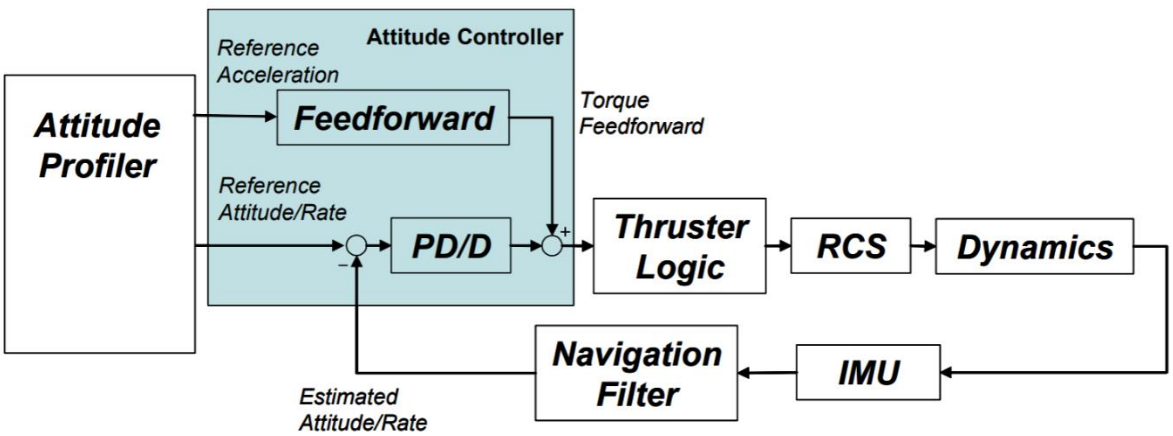


Figure 4.2.4-2 Lander Entry Attitude Control Block Diagram (Courtesy JPL)

After the entry phase is complete and the lander separates from the back shell, it will utilize terrain relative navigation to avoid obstacles and find a safe landing spot during the powered descent. The lander has an appropriate fuel margin in case it needs extra time to find a suitable landing zone. A listing of the hardware that the lander will use on transit to Mars and during EDL is provided below in **Table 4.2.4-1**. The star trackers and sun sensors will be mounted on the cruise stage since they are only needed in space. The gyros will determine the attitude during entry, and the LIDAR equipment will be used during the powered descent for terrain relative navigation. All the hardware is commercially available, and there are backups for the items that are not too heavy. Overall, the lander has the necessary equipment to successfully guide itself from Earth to the landing site on Mars.

Table 4.2.4-1 Cruise Stage and Lander ADCS Hardware

Item	Quantity	Mass (kg)	Power (W)	Total Mass (kg)	Total Power (W)
MR-107s Thruster	12	1.01	0.0	12.12	0.0
Engine Valves	12	0.00	34.8	0.0	417.6
Valve/Catalyst Heater	12	0.00	17.2	0.0	206.4
Bradforfd Coarse Sun Sensor	6	0.215	0.0	1.29	0.0
GG1320AN Gyro	6	0.454	1.6	2.72	9.60
SITAEL S.p.A Horizon Sensor	3	0.40	2.0	1.2	6.0
Leonardo AASTR Star Tracker	2	2.60	12.6	5.2	25.2
Doppler LIDAR – Electric Chassis	1	108.05	145.0	108	145.0
Doppler LIDAR - Optical Head	1	24.26	0.0	24.3	0.0
Flash LIDAR - Sensor Head	1	78.28	100.0	78.3	100.0
Flash LIDAR - Electronics Box	1	79.38	0.0	79.4	0.0
Laser Altimeter	1	52.92	70.0	52.92	70.0
			Total	**365.4**	**979.8**

4.2.5 Thermal Control and Analysis

The lander's thermal system is designed to keep its subsystems and payloads within their nominal operating range and keeping the MAV propellants warm during ground operations. The lander's electronic equipment is covered with MLI to prevent alpha and beta particles from the eMMRTG from damaging the system. The thermal analysis was performed to determine the temperature of the lander and MAV at the landing site of Louth Crater. During ground operations, the eMMRTG provides heat to sustain the MAV's propellants and equipment at their operating temperatures. **Figure 4.2.5-1** and **Figure 4.2.5-2** show the details of the thermal analysis using a solar flux of 180 W/m^2 and a generated heat of 1,200 W. We analyzed three cases: direct sunlight and complete darkness during ground operation, and during the cruise phase[9]. The analysis shows that the system satisfies the temperature ranges shown in **Figure 4.2.5-3**.

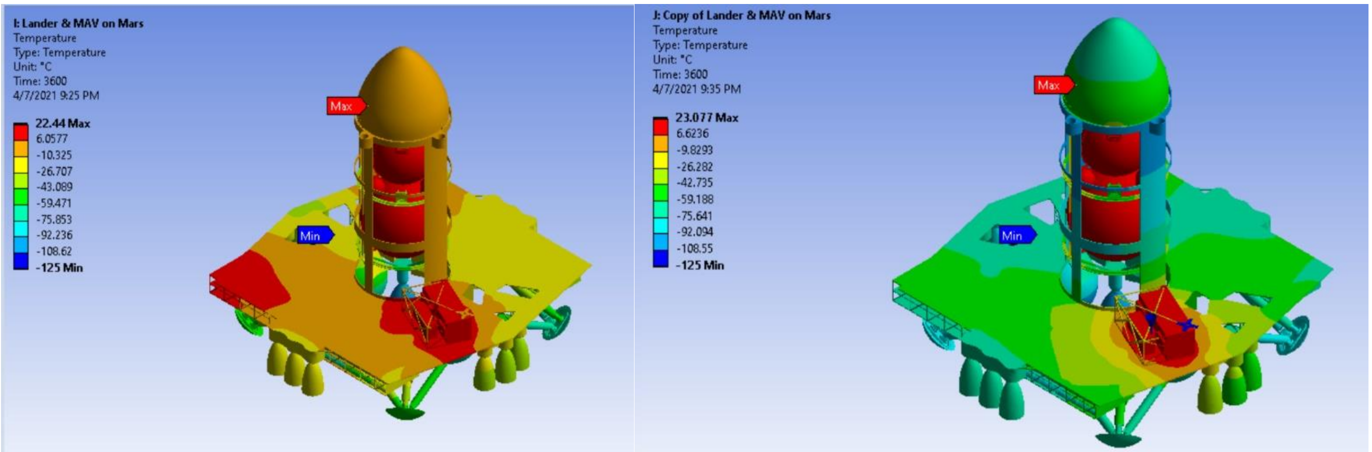


Figure 4.2.5-1 Lander & MAV Ground Operation Thermal Analysis

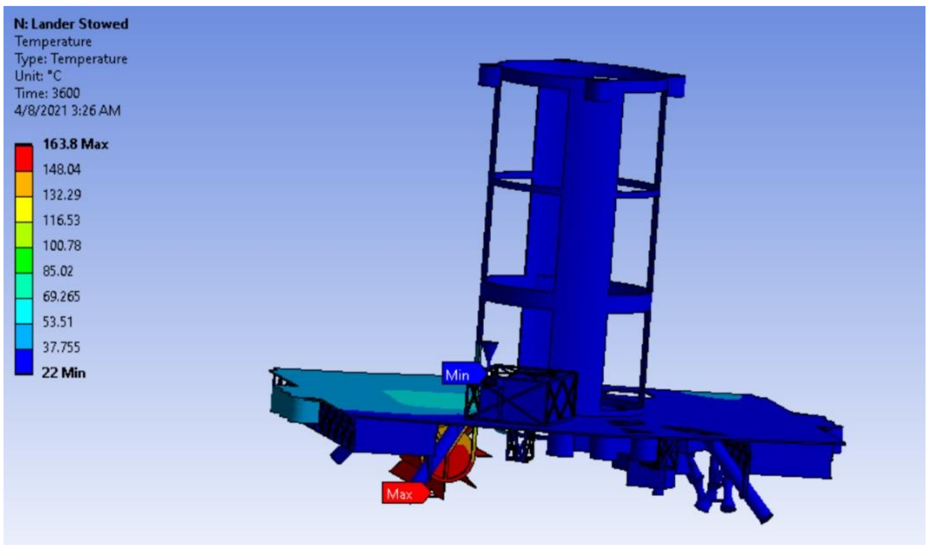


Figure 4.2.5-2 Lander Thermal Analysis Cruise Stage

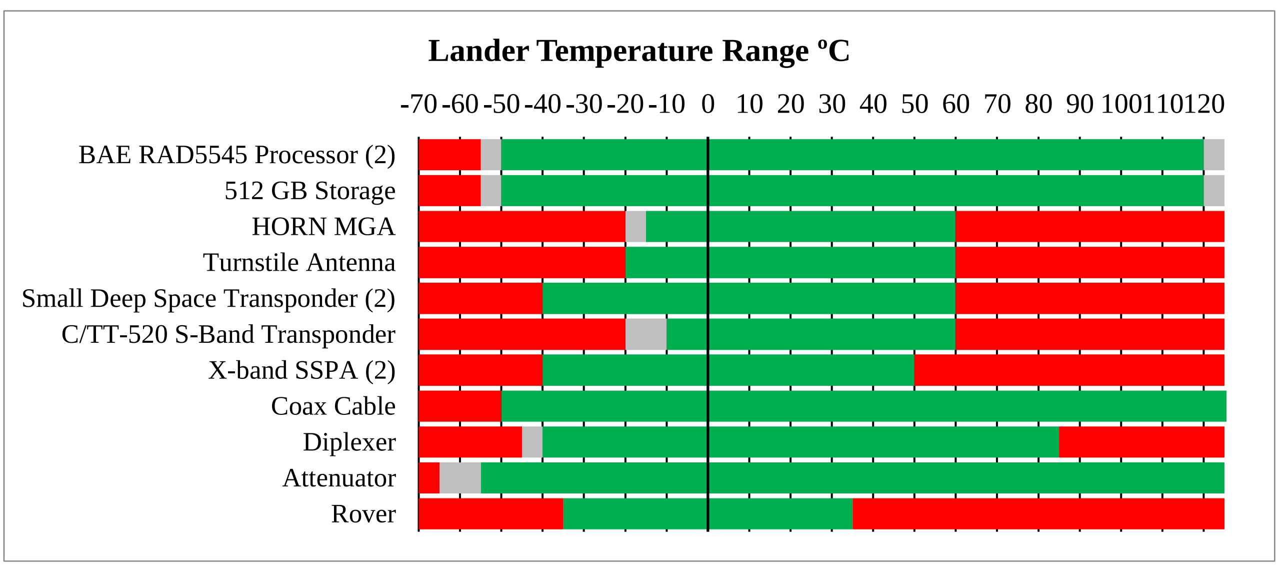


Figure 4.2.5-3 Lander Equipment Temperature Range

4.2.6 Telecommunications

The lander will use the orbiter as a relay to transmit and receive signals from Earth. Since the orbiter will be close to the lander during transit, the lander can utilize a low-power turnstile antenna to transmit information to the orbiter. A list of the lander's communication hardware is listed below in **Table 4.2.6-1**. The listed equipment is COTS or easily manufactured. The turnstile antenna will communicate in S-band with the orbiter's LGA, and the horn medium-gain antenna will be used if it becomes necessary for the lander to send and receive signals directly from the DSN. Again, we used standard link equations from Chapter 9 of Brown [1] and STK to verify the link performances. A summary of the links is shown in **Table 4.2.6-2.** All the links have acceptable link margins. In sum, the lander has the appropriate hardware to stay in contact with Earth on the way to Mars.

Table 4.2.6-1 Lander Telecommunications Hardware

Component	Mass (kg)	Power (W)
Turnstile Antenna	0.3	0.0
Horn MGA	2.1	0.0
Small Deep Space Transponder (2)	6.4	15.8
X-band SSPA (2)	2.7	60.0
C/TT-520 S-Band Transponder	4.0	39.0
Coax Cable	7.8	0.0
Diplexer	0.5	0.0
Attenuator	0.100	0.0
Totals	**23.9**	**114.8**

Table 4.2.6-2 Lander Link Performance

Link	Link Type	Data Rate (kbps)	Min Link Margin (dB)
DSN 34 m - Lander	Emergency Uplink	0.500	3.2
Lander – DSN 70 m	Emergency Downlink	0.140	3.2
Lander-Orbiter	Telemetry	8	16.5

4.2.7 Command and Data Handling Systems

The lander utilizes two BAE RAD5545 multi-core processors to process data and commands for the mission. The two processors are connected in parallel for redundancy in case one fails. They are certified for flight and radiation-hardened. In addition to the processors, the lander uses a 512 GB solid-state recorder for storing the data. **Table 4.2.7-1** lists the summation of the masses and powers for each component the lander uses for its command and data handling system.

Table 4.2.7-1 Command and Data Handling System Components for Lander

Component	Mass (kg)	Power (W)
BAE RAD5545 Processor (2)	4.0	35.4
512 GB Storage	14	35
Totals	**18.0**	**70.4**

4.2.8 Power Systems

A trade study between solar, RTG, and fission was conducted to determine the power generation system for our lander. The winning design was an eMMRTG due to the lander needing constant heat while potentially being in continuous darkness for five months. The lander is responsible for heating the propellant for the MAV, so the eMMRTG benefits both vehicles. The system must power the EDL sequence, survive Martian nights, and heat the electronics and propellant onboard; therefore, the battery configuration is designed to have seven parallel strings of eight cells wired in series. The design only needs six parallel strings, but an extra string was added for redundancy. The battery cells used for the lander will be the QL015KA cells from Quallion. The total battery configuration can provide 2,923 Wh and will weigh 20.6 kg. The EDL phase of the lander's mission will require the most power with 268 Wh. The power system design will only discharge the batteries 87% to preserve their long-term life.

4.2.9 Mass and Power Statements

The lander's mass and power budgets are listed below in **Table 4.2.9-1** and **Table 4.2.9-2**.

Table 4.2.9-1 Lander Mass Statement

Subsystems	Budget (kg)	Current (kg)	Status
Structure	447.9	905.0	C
Thermal	46.3	112.4	C
ACS	154.5	365.4	C
Power	324.4	55.5	C
Cabling	123.6	210.1	C
Propulsion	231.7	110.9	C
Telecom	108.1	23.9	C
CDS	108.1	18.0	C
Total	**1,544.7**	**1,801.3**	**C**
Margin	463.4	427.4	C
Payload	752.1	990.0	E

Table 4.2.9-2 Lander Power Statement

Subsystems	Budget (W)	Current (W)	Status
Thermal	132.0	0.0	C
ACS	94.3	979.8	C
Power	47.2	100.0	C
CDS	80.2	70.4	C
Comms	108.5	114.8	C
Propulsion	4.7	4.7	E
Mechanisms	4.7	4.7	E
Total	**471.6**	**1,274.4**	**E**
Margin	424.4	424.4	E
Payload	150.0	150.0	E
On Orbit Power	**1,046.0**	**1,848.9**	**E**

OODM	**2,296.7**	**3,218.7**	**C**
Propellant	451.2	625.8	C
Pressurant	22.6	15.7	C
On Orbit Wet Mass	**2,770.5**	**3,860.2**	**E**

4.3 Rover

4.3.1 Derived Requirements

Table 4.3.1-1 shows the derived requirements developed to help the rover satisfy the needs of the mission.

Table 4.3.1-1 Derived Requirements for the Rover

RVR.1	The rover shall be an autonomous vehicle
RVR.2	The rover shall operate for five months in total darkness on Mars
RVR.3	The rover shall operate during a minimum of seven months of day/night cycles
RVR.4	The rover mass should not exceed 280 kg
RVR.5	The rover shall store between 2.5 kg and 3 kg of ice core samples
RVR.6	The rover's power summation should not exceed 1,350 W
RVR.7	The rover shall keep the ice core samples' temperature less than -5 degrees Celsius
RVR.8	The rover shall drill ice cores with a diameter between 25 mm and 50 mm
RVR.9	The rover shall drill ice cores with a length between 100 mm and 150 mm

4.3.2 Payload

Our rover is equipped with the following: MAHLI, PIXL, NavCams, and HazCams for various fields of view during the operation. **Figure 4.3.2-1** below displays the rover's internal layout.

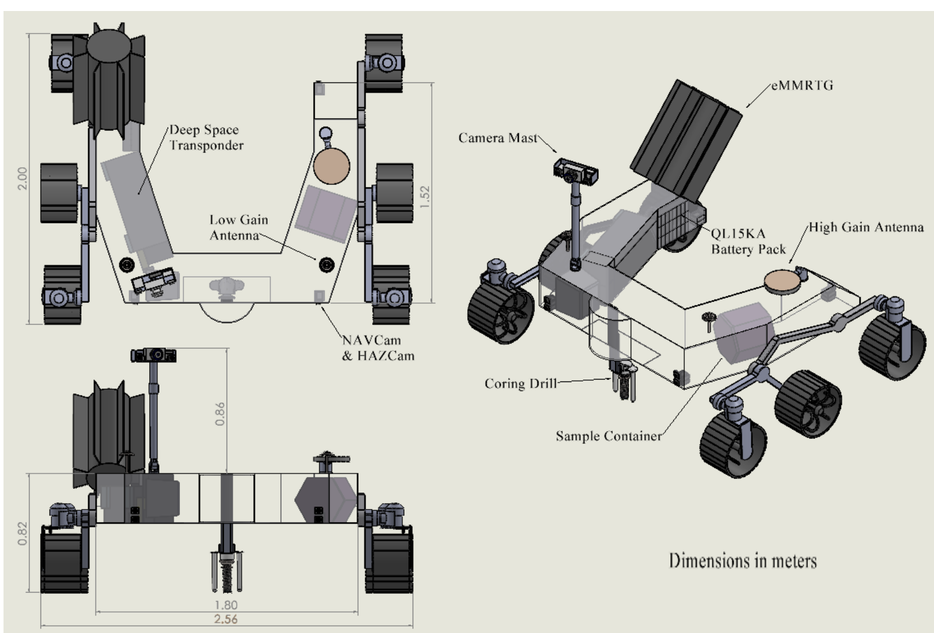


Figure 4.3.2-1 Rover Internal Layout

Drilling is a vital part of this mission to collect Martian ice cores. The rover will be equipped with an internal body-mounted drilling system which has the benefit of providing additional stability during drilling. This also reduces system complexity and can minimize hardware malfunction during the drilling phase. To meet the ice core dimension requirements, a mechanical drill will be used to drill through the overburden layer. If the layer is relatively thin, the coring drill bit will drill a sufficient depth into the ice. The ice core samples shall be at least 25 mm in diameter and 100 mm in length. As shown in **Figure 4.3.2-2**, the team has designed the coring drill to have the ability to collect samples that are 50 mm in diameter and 150 mm in length. This satisfies the derived requirement listed in **Table 4.3.1-1**. The coring drill is made with AISI 316 Annealed Stainless Steel and is equipped with three drill cutters for a better penetration rate. The rotary power required is 110 W, and the percussive power required is 65 W. After the sample has been collected, the drill will be raised to the PIXL via telescoping sliders to have its composition studied and verified. If the sample is found to not comply with requirements, the drill can dispose of the sample and search for new drilling locations. Once sample verification is completed, the drill will transfer the collected ice core into the sample tubes to be sealed. Inside the rover's body, a robotic arm will transfer the sealed sample tubes into the cryocooler.

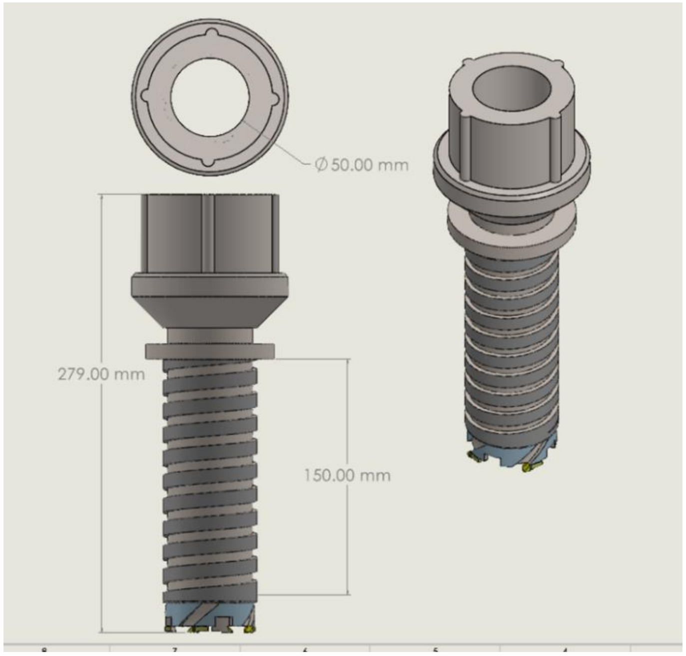


Figure 4.3.2-2 AISI 316 Annealed Stainless Steel Coring Drill Configuration

To safely keep the ice core samples in a frozen state back to earth, aluminum is one of the materials which would be satisfactory for the inner bag of the sample tubes. They are constructed of a four-ply lamination of a 0.50

mm aluminum foil inner liner, a 0.50 mm polyester film second ply, a 0.50 mm polyester film third ply, and a 0.50 mm aluminum foil outer ply[11]. After the ice samples are collected, an internal barometer will ensure proper pressure is maintained. The cryocooler will be used to keep the ice samples at a constant temperature of -80°C.

4.3.3 Propulsion

The rover is equipped with a rocker-bogie suspension system to aid in its search for ice cores. There are four HazCams, four NavCams, and a single MAHLI to help navigate the rover autonomously. **Figure 4.3.3-1** illustrates the sensors' fields of view with no obstruction to any.

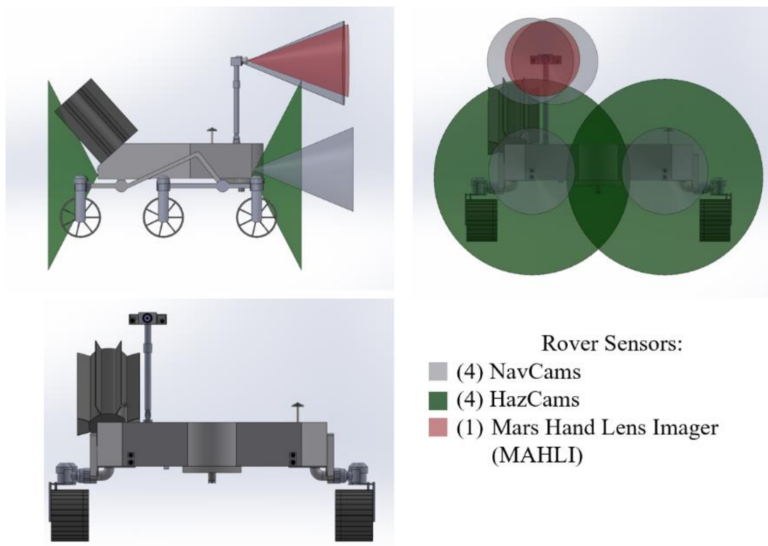


Figure 4.3.3-1 Field of Views of the Rover's Sensors

4.3.4 Structures

The rover structure, like the other vehicles, was modeled in SOLIDWORKS 2020 with the 2195 aluminum-lithium alloy. As shown in **Figure 4.3.4-1**, the rover was uniquely built in the shape of the letter "U." This design choice was a main driver in the appearance of the rover. It was selected to allow the rover to fit on the lander without disturbing the positioning of the MAV while also keeping the center of mass close to the geometric center of the lander.

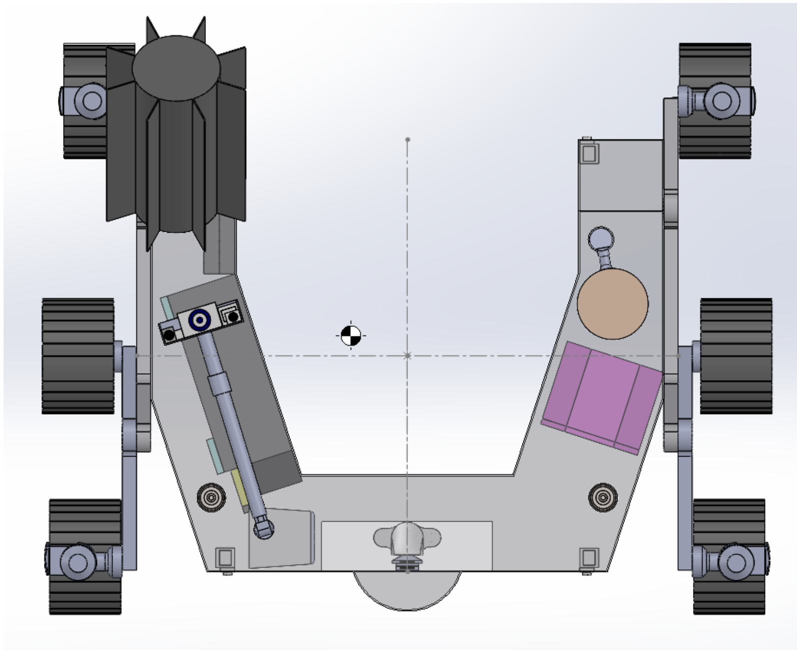


Figure 4.3.4-1: Rover's Center of Gravity Compared to Geometric Center

4.3.5 Thermal Control and Analysis

The rover is designed to survive the worst-case thermal conditions at the landing site. The main heat generator for the rover during ground operation is the eMMRTG that constantly generates heat which keeps the rover's electronic components within their operating ranges. Thermal analysis was performed on the rover under three cases: ground operation with sunlight, ground operation with complete darkness, and cruise configuration. **Figure 4.3.5-1** and **Figure 4.3.5-2** show the details of the thermal analysis using a solar flux of 180 W/m^2 and a generated heat of 1,200 W. The results show that the system satisfies the temperature range requirement for the rover components through all mission phases shown in **Figure 4.3.5-3**.

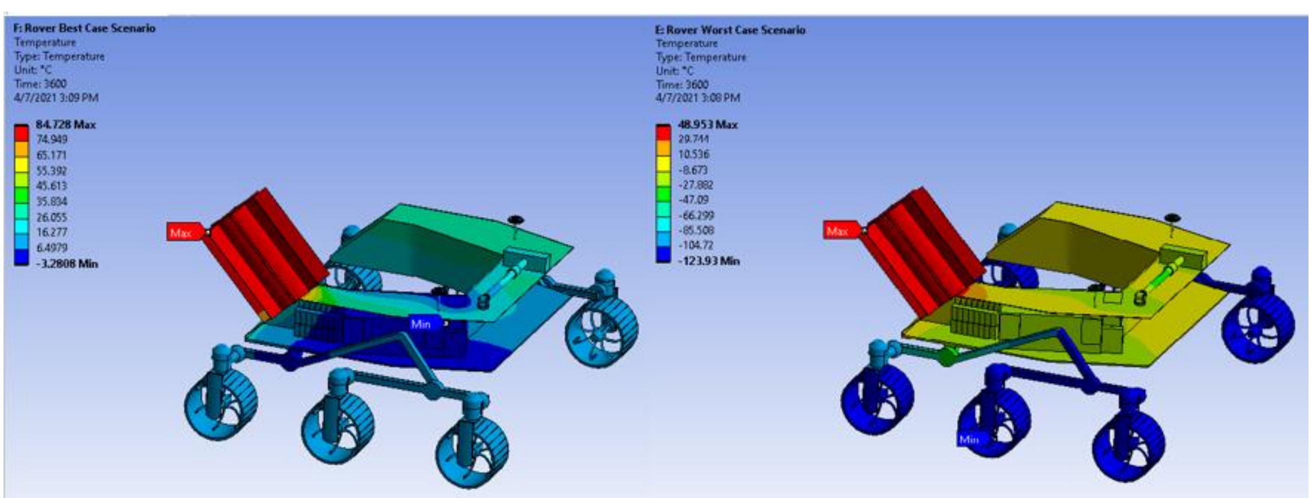


Figure 4.3.5-1 Rover Ground Operation Thermal Analysis

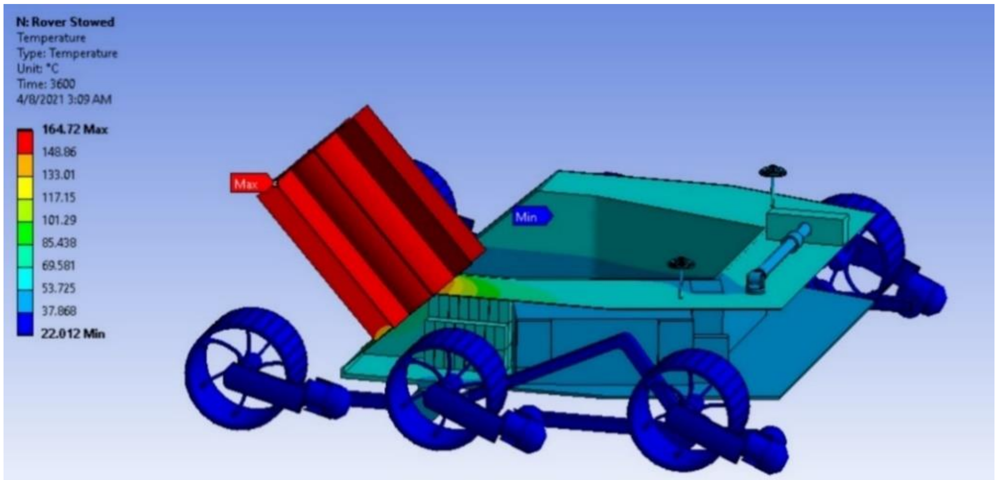


Figure 4.3.5-2 Rover Stowed Configuration Thermal Analysis

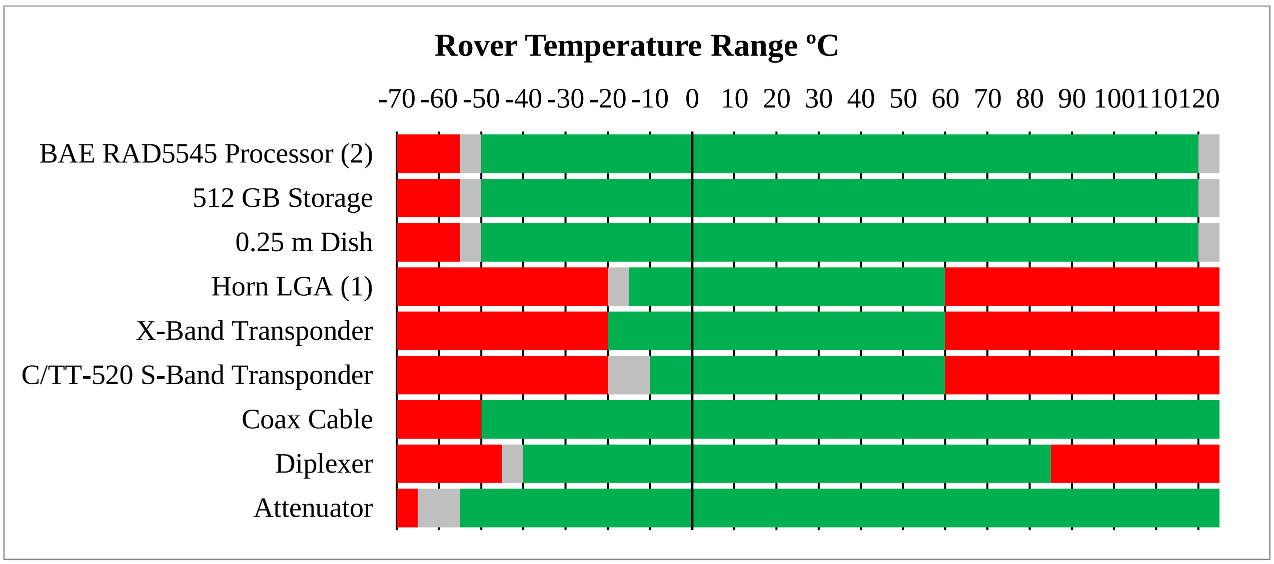


Figure 4.3.5-3 Rover Components Temperature Range

4.3.6 Telecommunications

The rover needs to communicate with the orbiter to uplink all the data and pictures it collects. We determined that the rover must transmit to the orbiter at 2.32 Mbps for 8 minutes a day to transmit all the data it would generate. It may also need to communicate directly to the DSN in the case of an emergency. To achieve this, it uses COTS or easily manufactured hardware listed below in **Table 4.3.6-1**. The rover will mainly use the 0.25 m HGA to send data to the orbiter in X-band. The two choked horn LGAs provide a 180° field of view to provide emergency communications with the DSN and orbiter in S-band. The hardware had to meet the communication requirements without using more than the 150 W that the eMMRTG provides.

Table 4.3.6-1 Rover Telecommunications Hardware

Component	Mass (kg)	Power (W)
0.25 m Dish	1.1	0
Choked Horn LGA (2)	4.2	0.0
Small Deep Space Transponder (2)	6.4	15.8
X-band SSPA (2)	2.7	60.0
S-Band SSPA (1)	4.5	0.0
Coax Cable	0.1	0.0
Diplexer	0.5	0.0
Attenuator	0.100	0.0
Totals	**19.6**	**75.8**

The links were modeled in STK to verify their performance. The link margin for the HGA to the orbiter is shown below **Figure 4.3.6-1** for a typical sol during the mission**.** As the sol goes on, the orbiter gets more overhead and the link margin increases. Each interval has about ten minutes when the link margin is above the required 3 dB. The rover has multiple opportunities per day to transmit for the eight minutes that are required. Only 60 W is needed to be input into the X-band SSPA to achieve this link performance; the rover can still do other activities while it transmits the data since the eMMRTG provides 150 W. However, if the HGA fails, the rover's LGAs are also capable of transmitting to the orbiter in S-band at the required 2.32 Mbps. It achieves a very similar performance to that shown in **Figure 4.3.6-1**. However, since the antennas are low gain, it requires 132 W to be input into the S-band SSPA to achieve this performance. The rover would have to cease most other activities to transmit with the LGAs. Next, the LGAs provide emergency downlink at 120 bps to the 70 m DSN dishes. This requires the same amount of power as the link with the orbiter, so the rover will need to cease most activities to transmit to the DSN. The LGAs are also able to receive 233 bps emergency signals from the 34 m DSN dishes. The link performance over the entire mission is shown below in **Figure 4.3.6-2.** As seen in the figure, the rover does not have direct access to the DSN from mid-January to May 2028 due to Mars' tilt. The orbiter will remain in communication with the rover during this time. The rover will also be in constant darkness during this time, so it will not be able to do much except keep in contact with the orbiter.

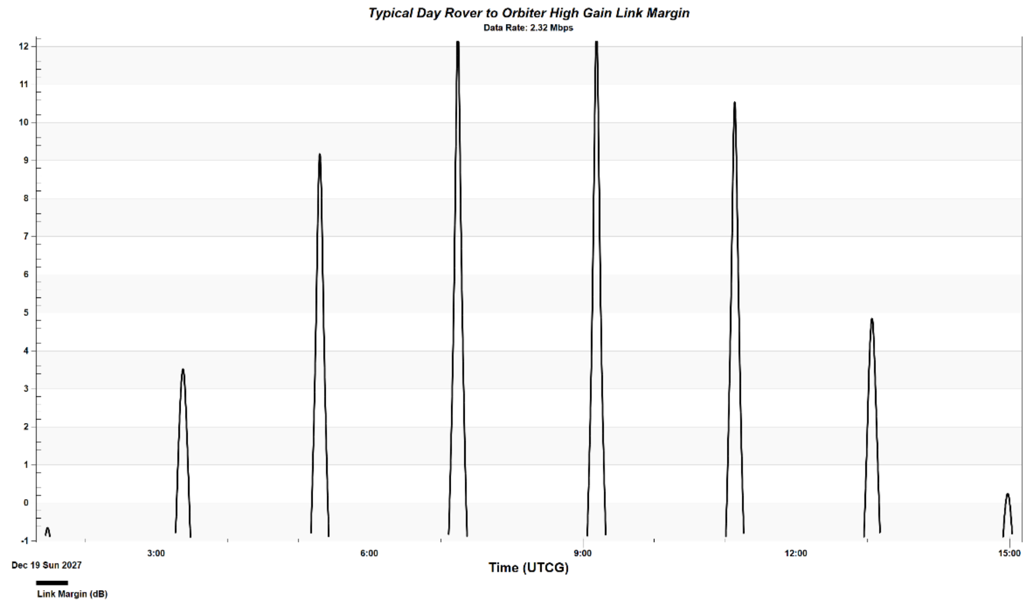


Figure 4.3.6-1 Rover to Orbiter High Gain Uplink Margins on a Typical Sol

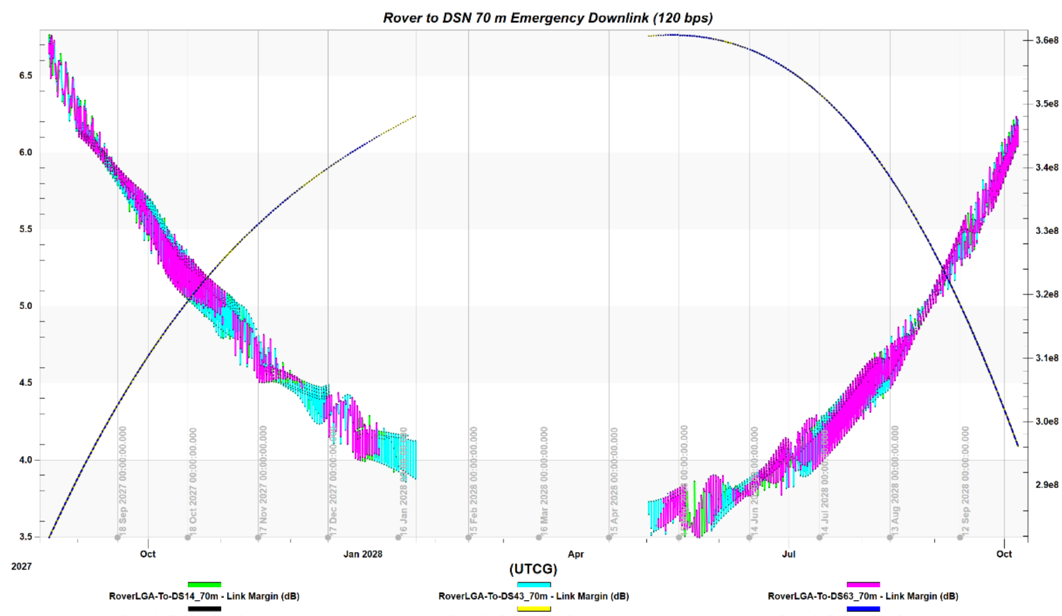


Figure 4.3.6-2 Rover to DSN 70 m Emergency Downlink Margins

Finally, a summary of the rover's links and their performance is provided below in **Table 4.3.6-2.** The minimum link margins achieved throughout the entire mission for each link are above the required three dB, and the margins are usually higher. This means better performance can be achieved without increasing power consumption. The rover's telecommunication system provides reliable and redundant communications that meet requirements within the tight mass and power constraints.

Table 4.3.6-2 Rover Link Performance

Link	Link Type	Data Rate (kbps)	Min Link Margin (dB)
Rover - DSN 70 m	Emergency Downlink	0.120	3.4
DSN 34 m - Rover	Emergency Uplink	0.233	3.3
Rover-Orbiter	High Gain Uplink	2,320	3.0
Rover-Orbiter	Backup High Gain Uplink	2,320	3.0

4.3.7 Command and Data Handling Systems

The lander utilizes two BAE RAD5545 multi-core processors to process the data and command for the mission. The two processors are connected in parallel for redundancy in case one fails. They are certified for flight and are radiation-hardened. In addition to the processors, the lander uses a 512 GB solid-state recorder for storing the data. **Table 4.3.7-1** lists the summation of the masses and powers for each component the lander uses for its command and data handling system.

Table 4.3.7-1 Command and Data Handling System Components for Rover

Component	Mass (kg)	Power (W)
BAE RAD5545 Processor (2)	4.0	35.4
512 GB Storage	14	35
Totals	18.0	70.4

4.3.8 Power Systems

A trade study between solar, RTG, and fission was conducted to determine the power generation system for the rover. The winning design ended up being an eMMRTG due to low solar insolation at the landing site and the rover's need to have a constant heat source while potentially being in continuous darkness for five months. The current mission design will have the rover completing its mission before the dark period, but it was essential to plan for it just in case. The QL015KA battery cell from Quallion will store the power for the rover and will be wired with eight cells in series with six strings wired in parallel. The design includes one redundant string and can store up to 2,506 Wh. The system design will only discharge the batteries 87% to preserve their long-term life.

4.3.9 Mass and Power Statements

The rover's mass and power budgets are tabulated in **Table 4.3.9-1** and **Table 4.3.9-2**.

Table 4.3.9-1 Rover Mass Statement

Subsystems	Budget (kg)	Current (kg)	Status
Structure	32.8	23.6	E

Table 4.3.9-2 Rover Power Statement

Subsystems	Budget (W)	Current (W)	Status
Thermal	154.7	0.0	E

Thermal	4.4	3.1	E	Power	55.2	20.0	C
Power	41.6	61.6	C	CDS	93.9	70.4	C
Cabling	17.5	10.2	C	Telecomm	127.1	75.8	C
Telecomm	6.6	19.6	C	Propulsion	116	112.9	E
CDS	6.6	18.0	C	Mechanisms	5.5	5.4	E
Total	**109.5**	**136.2**	**E**	**Total**	**552.4**	**284.5**	**E**
Margin	38.3	27.5	E	Margin	497.2	484.0	E
Payload	60.4	43.4	E	Payload	300.8	239.2	E
Total Mass	**169.8**	**254.0**	**E**	**Total Power**	**1,350.4**	**1,007.7**	**E**

4.4 Mars Ascent Vehicle

Figure 4.4-1 illustrates the layout for the MAV's components and subsystems.

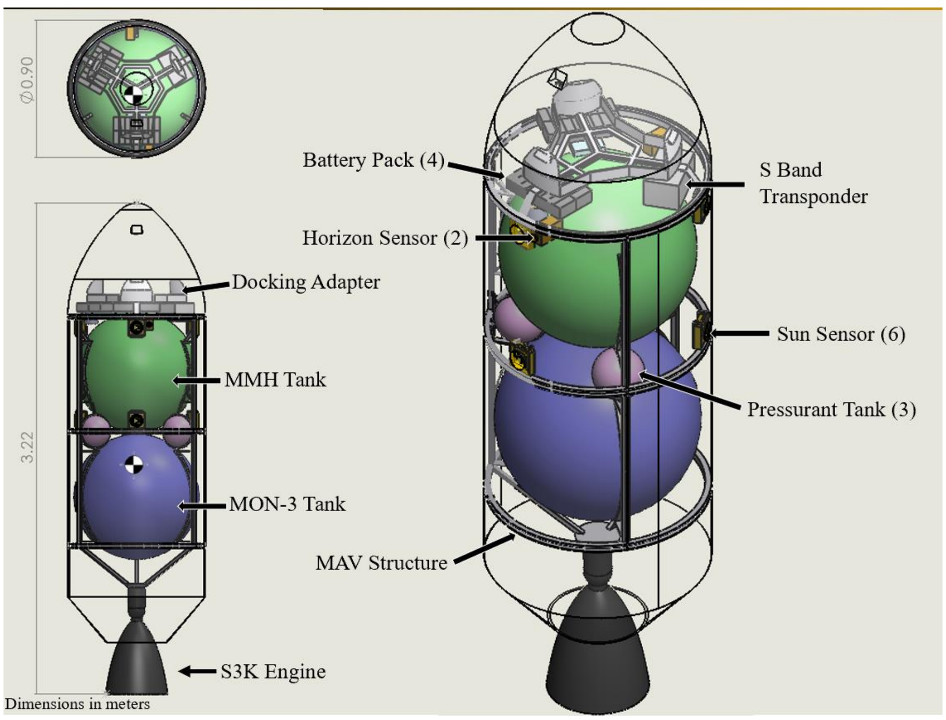


Figure 4.4-1 MAV Internal Layout

4.4.1 Derived Requirements

Table 4.4.1-1 shows the derived requirements developed to help the MAV satisfy the needs of the mission.

Table 4.4.1-1 Derived Requirements for MAV

MAV.1	The MAV shall be an autonomous vehicle
MAV.2	The MAV shall be able to launch in total darkness
MAV.3	The MAV shall be able to launch during a minimum of seven months of day/night cycles
MAV.4	The MAV shall withstand wind speeds of 30 m/s
MAV.5	The MAV shall deliver the sample cryocooler with samples to the orbiter

MAV.6	The MAV mass should not exceed 416.4 kg
MAV.7	The MAV power summation should not exceed 419.4 W
MAV.8	The MAV shall have a ΔV of at least 4,500 m/s
MAV.9	The MAV shall autonomously dock with the orbiter within four hours of launch

4.4.2 Payload

To safely keep the ice core samples in a frozen state back to earth, an oxford cooler will be used to keep the ice samples at a stable temperature. The MAV will receive the sample container from the rover and launch it into a low Mars orbit.

4.4.3 Propulsion

To bring the MAV into orbit, the S3K engine was selected due to its ideal I_{sp} and thrust capability of 352 seconds and 3,500 N, respectively. This engine required the use of MON-3 as the oxidizer and MMH as the propellant with an O/F mixture ratio of about 1.9. The MON-3 and MMH are fed into the S3K engine by helium pressurant. In addition, the MMH feeds into the 12 MR-103J thrusters. On the P&ID, which is **Figure 4.4.3-1**, the red and green highlighted lines show the oxidizer and fuel lines, respectively, and the non-highlighted lines are the pressurant lines.

In the MAV, there is about 371 kg of MON-3 and about 210 kg of MMH for a total of about 581 kg of propellant. In addition, there are 2.25 kg of helium spread throughout the three pressurant tanks which are pressurized to 34.5 MPa. As seen from **Section 3.7**, this configuration allows the MAV to accomplish requirements **MAV.8** and **MAV.9** as it gives it the capability of reaching orbit and docking with the orbiter within 4 hours of launch.

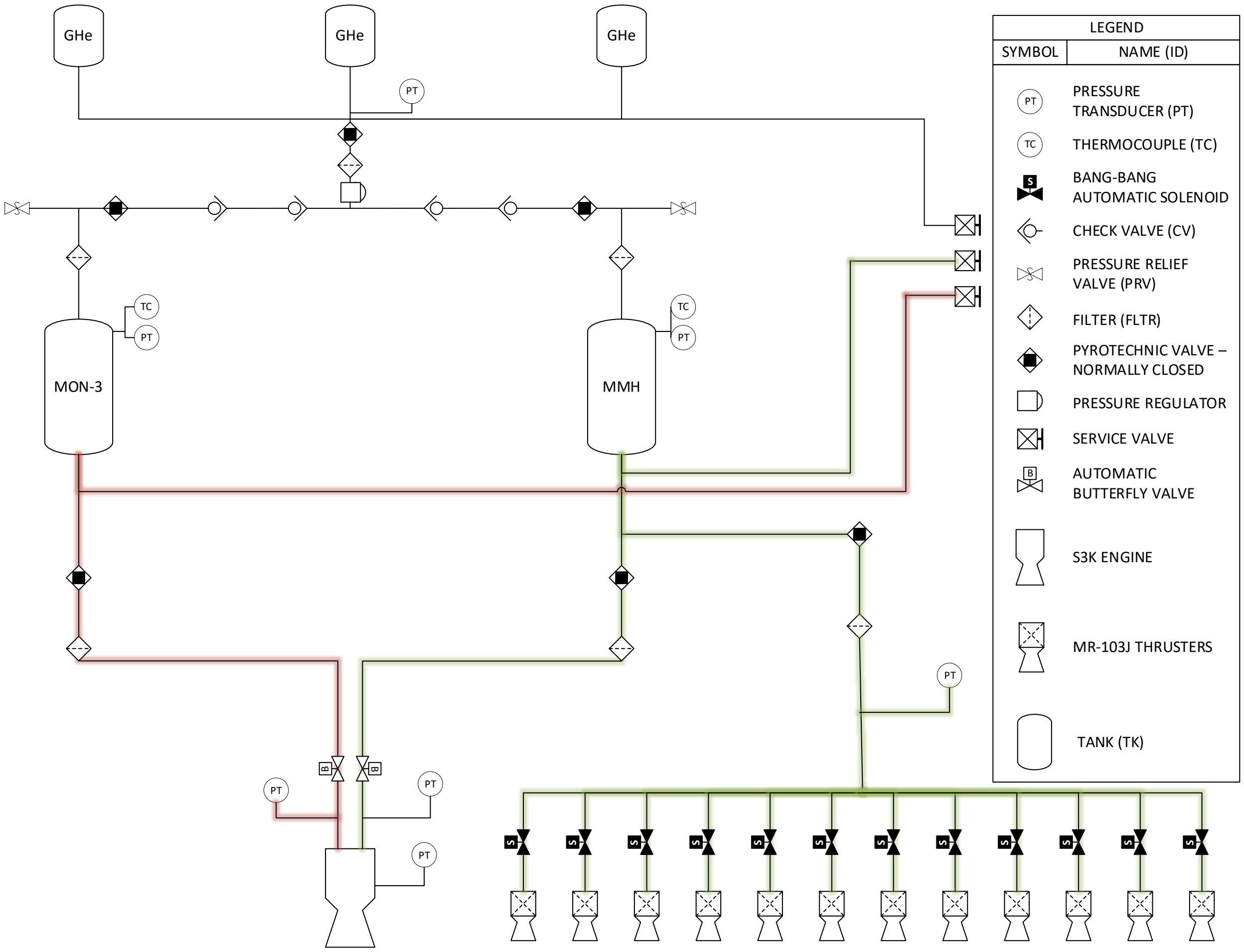


Figure 4.4.3-1 P&ID for the MAV

4.4.4 Structures

The worst-case MAV loads of 12 Earth g's occur during EDL. This was simulated through SOLIDWORKS 2020 as seen in **Figure 4.4.4-1**. This structure supports the whole MAV to ensure that it would not collapse or buckle under its own weight. It is constructed from aluminum-lithium 2195 C-channels and T-sections to fulfill the requirement **MAV.4**. The majority of the structure is meant to retain the spherical propellant and pressurant tanks while supporting the remaining subsystems at the top of the MAV structure.

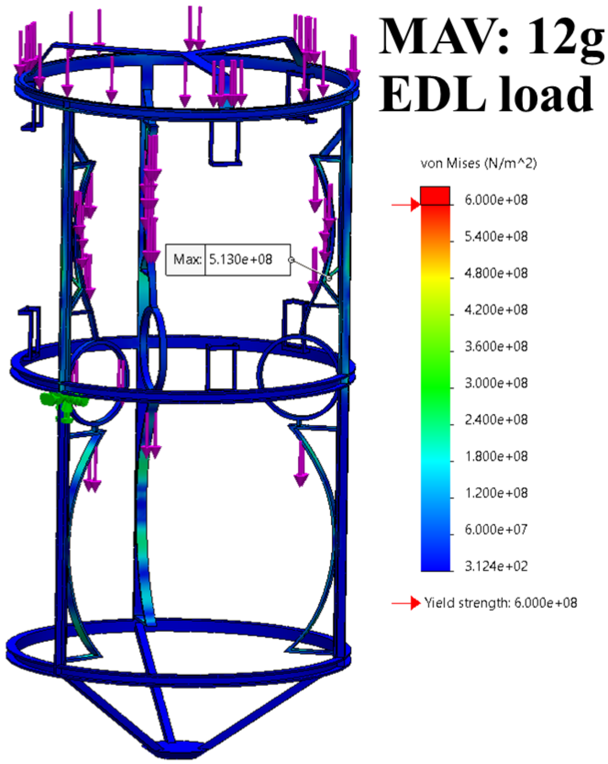


Figure 4.4.4-1 MAV 12 Earth G Static Load

4.4.5 Attitude Determination and Control System

As shown in **Section 4.4.3**, the thrusters for the ACS consist of 12 MR-103J thrusters by Aeroject-Rocketdyne. These thrusters are located around the MAV and are fueled by the MMH tank. An attitude control system needed to be designed for the pitch maneuvers during launch and for operations on orbit. The MR-103J was chosen as it could provide high thrust for fast response times but also has a small enough impulse bit for fine maneuvers. The control system for orbit was designed just like the orbiter. Disturbance torques were accounted for, and a control system was designed using LQR control. The control system for launch also uses LQR, and the only difference is that the plant dynamics are different for launch. The challenge with designing this control system was producing fast response times without saturating the thrusters. With the placement of the MR-103Js on the MAV, the maximum amount of torque that can be produced in the pitch and yaw axes is 1.875 N-m, while in the roll axis, it is 0.904 N-m. The results for a 1° pitch step response during launch is shown below in **Table 4.4.5-1.** This maneuver achieves an acceptable settling time while the maximum torque needed does not saturate the thrusters in the pitch axis since it is below 1.875 N-m.

Table 4.4.5-1 MAV Launch 1° Pitch Kick Step Response

	Steady State Error (deg)	Settling Time (s)	% Overshoot	Max Input Torque Needed (N-m)
Roll	7.16E-07	20	0.00	0.000
Pitch	9.93E-06	7	0.09	1.814
Yaw	4.31E-08	20	0.00	0.000

Next, a 1° step response in all axes is shown for the MAV in orbit below in **Table 4.4.5-2.** The errors are low with acceptable settling times and exceptionally low overshoot, which is important for docking. Also, none of the needed torques saturate the thrusters. The MAV's computer will be able to switch the control system when the launch phase is over. Since the feedback loops for both controllers are the same, the computer must only load a separate set of gains.

Table 4.4.5-2 MAV 1° Step Response in all Axes on Orbit

	Steady State Error (deg)	Settling Time (s)	% Overshoot	Max Input Torque Needed (N-m)
Roll	9.93E-04	12	0.10	0.901
Pitch	1.74E-05	6	0.91	1.866
Yaw	2.57E-03	9	-0.26	1.866

Finally, the MAV uses low mass and power-efficient equipment to determine its attitude. **Table 4.4.5-3** below shows the COTS hardware that the MAV uses. It uses sun sensors, horizon sensors, and IMUs to determine its attitude since they are low mass and use little power. The MAV will rely on the IMU during eclipse. While the thrusters use a lot of power, the MAV's batteries can power them for long enough to dock. Overall, the MAV's attitude control systems have been appropriately designed, and the MAV has the ADCS hardware it needs to launch into orbit and dock with the orbiter to transfer the ice sample container.

Table 4.4.5-3 MAV ADCS Hardware

Item	Quantity	Mass (kg)	Power (W)	Total Mass (kg)	Total Power (W)
VN200 SMD	2	0.016	0.40	0.032	0.80
SITAEL S.p.A Horizon Sensor	2	0.400	2.00	0.80	4.00
Bradford Coarse Sun Sensor	6	0.215	0.00	1.290	0.0
MR-103J Thruster	12	0.110	0.00	1.320	0.0
Engine Valves	12	0.200	8.25	2.40	99.0
Valve/Catalyst Heater	12	0.065	7.86	0.78	94.32
			Total	**6.62**	**198.1**

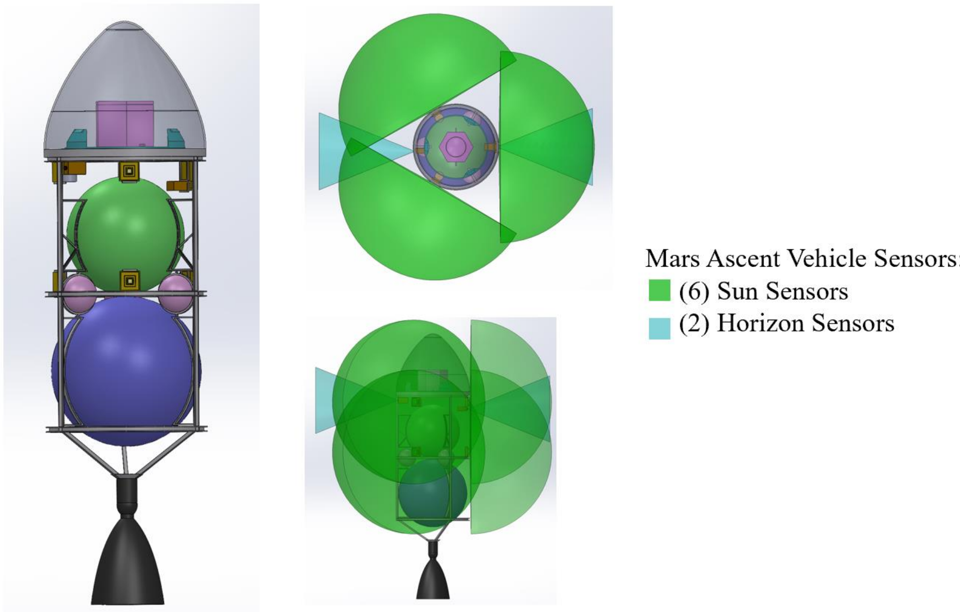


Figure 4.4.5-1 Field of View Plots for the MAV

4.4.6 Thermal Control and Analysis

The MAV’s thermal system is designed to keep its subsystems and payloads within their nominal operating range and keep the MAV propellants warm during ground and space operations. The detailed results of thermal analysis for MAV during ground operation have been shown in **Section 4.2.5**. The thermal analysis in this section focuses on the MAV when it launches to low Mars orbit and performs docking maneuvers to transfer the ice samples to the orbiter. There are three cases corresponding with three directions of the heat flux from the sun toward the MAV during docking operation. The detailed result of the thermal analysis was performed using ANSYS shown in **Figure 4.4.6-1** and shows that the system satisfies the temperature requirement shown in **Figure 4.4.6-2**.

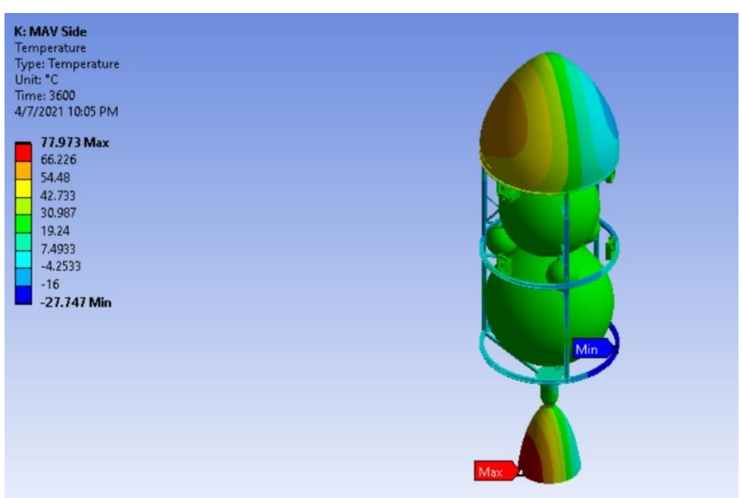


Figure 4.4.6-1 MAV Docking Operation Thermal Analysis

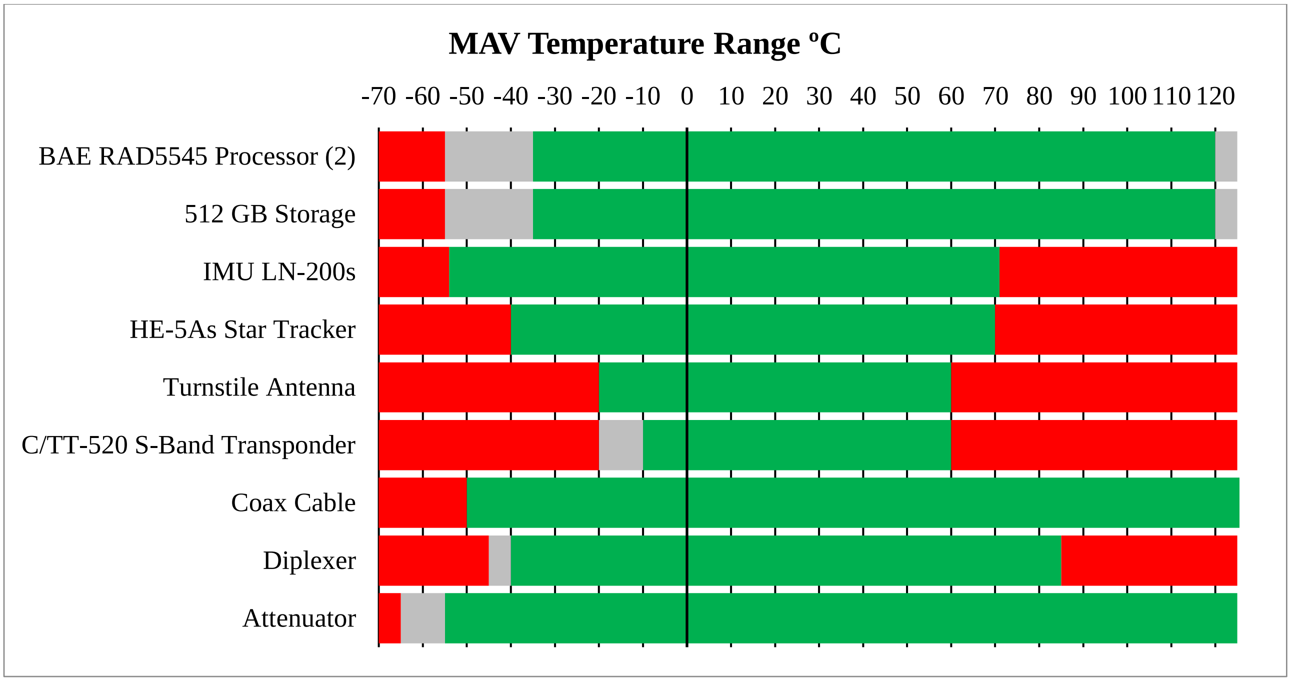


Figure 4.4.6-2 MAV Components Temperature Range

4.4.7 Telecommunications

The MAV will need to transmit its telemetry to the orbiter to successfully dock with it. We found that the MAV generates seven kbps of telemetry that it needs to transmit to the orbiter. With both the vehicles being in a 300 km orbit, the maximum distance that they have line of sight is only 1500 km. This is a relatively short range, and the MAV is able to achieve a high link margin with the orbiter by transmitting a 1 W signal out of a turnstile antenna in S-band. This is convenient since the MAV's subsystem mass and power usage needed to be reduced as much as possible. Using a turnstile antenna also allows the vehicles to communicate no matter what their attitude is. This link was verified using standard link equations from Chapter 9 of Brown[1]. **Table 4.4.7-1** below shows the telecommunications hardware that the MAV uses. It uses lightweight and low-power equipment that is commercially available.

Table 4.4.7-1 MAV Telecommunications Hardware

Component	**Mass (kg)**	**Power (W)**
Turnstile Antenna	2.1	0.0
ISISPACE S-Band Transceiver	0.2	13.0
Coax Cable	2.6	0.0
Diplexer	0.5	0.0
Attenuator	0.10	0.0
Totals	**5.5**	**13.0**

Overall, the MAV's telecommunication is simple and minimal, but has reliable communication with the orbiter to successfully dock and transfer the ice samples.

4.4.8 Command and Data Handling Systems

The MAV utilizes two BAE RAD750 processors to process the data and commands for the mission. The two computers are connected in parallel for redundancy in case one fails. They are certified for flight and radiation-hardened. In addition to the computers, the MAV uses a 1 GB solid-state recorder for storing data. **Table 4.4.8-1** lists the summation of the masses and powers for each component the MAV uses for its command and data handling system.

Table 4.4.8-1 Command and Data Handling System Components for MAV

Component	Mass (kg)	Power (W)
BAE RAD750 processor (2)	1.1	20.0
1 GB storage	0.0017	0.234
Totals	**1.1**	**20.2**

4.4.9 Power Systems

A trade study between solar, RTG, and fission was conducted to determine the power generation system for the MAV. The winning design ended up being quadruple junction body-mounted solar panels due to the MAV needing to generate 103 W. The selected design of the solar panels resulted in a mass of 6.2 kg and a solar array area of 2.2 m^2. The MAV will use the Quallion QL015KA battery cells and will be wired with eight cells in series and two parallel strings. There is one additional string of eight cells for redundancy, and the battery will only be discharged 85%.

4.4.10 Mass and Power Statement

The mass and power budget for the MAV is listed below in **Table 4.4.10-1** and **Table 4.4.10-2**.

Table 4.4.10-1 MAV Mass Statement

Subsystems	Budget (kg)	Current (kg)	Status
Structure	11.6	20.3	C
Thermal	2.8	2.8	E
ACS	4.6	6.6	C
Power	16.2	6.4	C
Cabling	1.8	1.8	E
Propulsion	4.6	43.4	C
Telecom	1.8	5.5	C
CDS	2.8	1.1	C

Table 4.4.10-2 MAV Power Statement

Subsystems	Budget (W)	Current (W)	Status
Thermal	59.9	0.0	E
ACS	42.8	198.1	C
Power	21.4	55.9	C
CDS	36.4	20.2	C
Comms	49.2	13.0	C
Propulsion	2.1	2.1	E
Mechanisms	2.1	2.1	E
Total	**213.9**	**291.6**	**E**

Total	**46.2**	**87.9**	**E**
Margin	40.2	40.2	E
Payload	24.0	24.0	E
OODM	**70.2**	**152.0**	**E**
Propellant	329.7	603.1	C
Pressurant	16.5	3.3	C
On Orbit Wet Mass	**416.4**	**736.0**	**E**

Margin	192.5	192.5	E
Payload	13	13.0	E
On Orbit Power	**419.4**	**497.1**	**E**

4.5 Sample Return Capsule

Figure 4.4-1 illustrates the layout for the sample return capsule's components and subsystems.

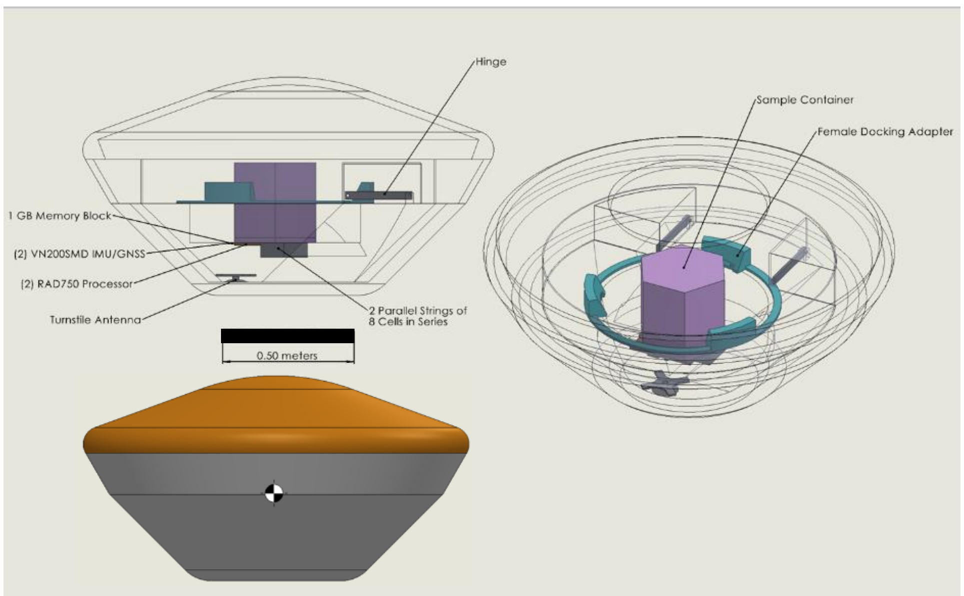


Figure 4.5-1 Sample Return Capsule

4.5.1 Derived Requirements

Table 4.5.1-1 shows the derived requirements developed to help the sample return capsule satisfy the needs of the mission.

Table 4.5.1-1 Sample Return Capsule Derived Requirements

ID	Requirement Statement
SRC.1	The Earth Return Capsule shall land on Earth on May 24, 2029
SRC.2	The SRC shall be an autonomous vehicle
SRC.3	The SRC mass shall not exceed 211 kg
SRC.4	The SRC touchdown velocity shall not exceed nine m/s
SRC.5	The SRC shall use a parachute with a diameter of six meters

4.5.2 Payload

The sample return capsule will have the sample storage container as its payload. The sample container has a mass of 21 kg and will contain 3 kg of ice cores, which will be kept frozen by a cryocooler. The total mass of the payload will be 24 kg. The sample container will be sealed, and the samples will be housed in tubes that will also seal the ice cores. The sample tubes will prevent any contamination from Mars, Earth, and interplanetary space.

4.5.3 Structures

The sample return capsule will have a carbon composite back shell and a heat shield that will be made from PICA tiles. The female docking ring adapter is located within the sample return capsule to allow the MAV to dock and transfer the sample container. When the transfer is completed, the female docking ring releases and allow the MAV to detach. This sample return capsule will seal shut with a hinged door to preserve the sample container within its heat shield. The structure of the sample return capsule will have a mass of 130 kg, and the total mass of the SRC with all subsystems and payload is 204 kg.

4.5.4 Attitude Determination and Control System

The sample return capsule will not utilize an active Attitude Control System but will have an IMU/GPS on board to determine its position at all times. The SRC contains two VN-200 SMD, which totals 8 grams and will draw 210 mA at 3.3V. The VN-200 connects to the Global Navigation Satellite System (GNSS) to provide the SRC's position when it lands. The sensor all gives yaw, pitch, and roll data with a frequency that can reach up to 1 kHz and transmits its position with a frequency of 400 Hz. The SRC will only need one VN-200 sensor but has an extra for redundancy.

4.5.5 Telecommunications

There will be a low gain turnstile antenna to continuously transmit data from the VN-200 to ground control so the SRC could be recovered. The additional components to help transmit the data are listed in **Table 4.5.5-1**.

Table 4.5.5-1 Telecommunication Components List

Component	Mass (kg)	Power (W)
Turnstile Antenna	2.1	0.0
Coax Cable	0.1	0.0
Diplexer	0.5	0.0
Attenuator	0.10	0.0
Totals	**2.7**	**0.0**

4.5.6 Command and Data Handling Systems

The sample return capsule will use a BAE RAD750, which is radiation-hardened. The SRC will be in interplanetary space for 935 days, so we chose a processor that was flight-proven. There is an additional 1 GB storage block to record data. **Table 4.5.6-1** tabulates the components' masses and power consumptions.

Table 4.5.6-1 Command and Data Handling Components List

Component	Mass (kg)	Power (W)
BAE RAD750 processor (2)	1.1	20.0
1 GB Storage	0.0017	0.2346
Totals	**1.1**	**20.2**

4.5.7 Power Systems

The SRC will use the QL015KA battery cells from Quallion, which will be wired with eight cells to a string and two parallel strings wired together. This battery configuration will survive for 14 hours post-orbiter separation and weighs 5.9 kg. This was based on the Hyabusa 2 sample return mission that took 14 hours for the capsule to separate and be recovered successfully.

4.5.8 Mass and Power Budget Statement

The mass and power budget for the SRC is listed below in **Table 4.5.8-1** and **Table 4.5.8-2**.

Table 4.5.8-1 SRC Mass Statement

Subsystems	Budget (kg)	Current (kg)	Status
Structure	55.4	130.00	E
Thermal	5.7	0.00	C
ACS	19.1	0.01	C
Power	40.1	5.88	C
Cabling	15.3	15.29	C
Propulsion	0.0	0.00	C
Telecom	13.4	2.74	C
CDS	13.4	1.1	C
Total	**162.5**	**155.0**	**C**
Margin	24.8	24.8	
Payload	24.0	24.0	
Total Mass	**211.3**	**203.9**	**C**

Table 4.5.8-2 SRC Power Statement

Subsystems	Budget (W)	Current (W)	Status
Thermal	0.0	0.0	C
ACS	0.0	0.9	C
Power	21.4	0.0	C
CDS	36.3	20.2	C
Comms	49.1	0.0	C
Propulsion	0.0	0.0	C
Mechanisms	2.1	2.1	C
Total	**108.9**	**23.3**	**C**
Margin	27.8	27.8	
Payload	12.0	12.0	
On Orbit Power	**148.7**	**63.0**	**C**

5.0 Systems Engineering

5.1 System Summary

The use of a systems engineering process throughout our design was crucial for ensuring our design met every requirement defined in the RFP. Our team utilized model-based systems engineering (MBSE) to verify that our design satisfies the customer, system, and derived requirements. We used the SysML modeling language in Cameo Systems Modeler to track our requirement satisfaction and verification.

In the MBSE model, we developed system level requirements that would trace back to the customer requirements. We developed derived requirements from our system level requirements based on the analysis we performed on what the system needs to do and how well it needs to do it. In addition to modeling our requirements, we decomposed the system into vehicles, subsystems, and components using a block definition diagram. We connected the blocks to the requirements they were responsible for through a satisfy relationship. After we set the relationships, we developed verification methods for each requirement. We ensured the customer requirements were traceable through the systems, and every requirement could be verified. **Appendix B** shows every requirement the MICKEY system has and the relations mentioned above.

Most derived requirements related to vehicle and subsystem mass and power estimates were written using "should" rather than "shall" due to them being guidelines during early design rather than strict requirements. As the design matured, we began designing around the real mass and power values rather than the estimated values in the requirements. Therefore, these guideline requirements no longer needed to be met, but we still prioritized minimizing mass and power to help lower the total cost. The key mass requirements we still enforced were the Earth launch mass and the sample return capsule mass. Overall, the customer and "shall" system-level and derived requirements have been satisfied by the system.

5.2 System Life Cycle

Figure 5.2-1 shows the system life cycle from conceptual design until mission closeout. Detail design will span from June 2021 until June 2022. Manufacturing, integration, and testing will last from June 2022 until June 2026 with a year of margin. There will be ample time for testing, verification and TRL development of the MAV. The system will launch on November 1, 2026, and return to Earth on May 24, 2029. Mission closeout will last from mid-2029 until the end of 2030 to allow time for post-mission planetary protection procedures and delivery of samples to laboratories.

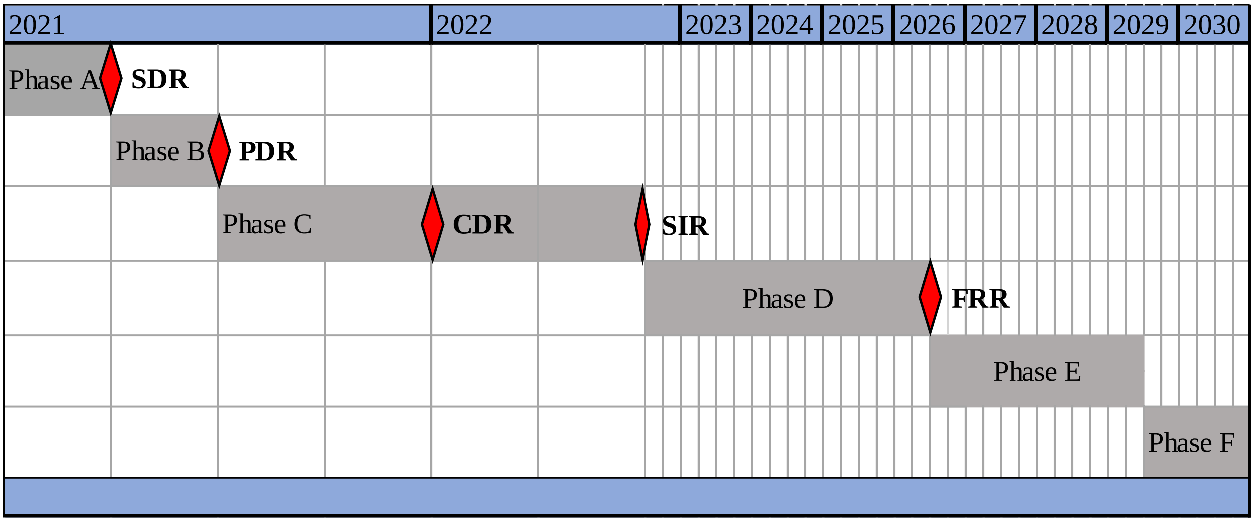
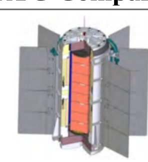


Figure 5.2-1 System Life Cycle

5.3 Mission Lifetime Assessment

The mission relies on two eMMRTGs to supply power and heat for both the rover and lander. The radioisotope thermoelectric generators' power generation degrades as the plutonium breaks down. However, the eMMRTG was designed to operate for 17 years, which includes three years for on Earth assembly. In **Table 5.3-1,** the comparison between the eMMRTG and MMRTG depicts that the power generation will degrade 2.5% every Earth year but will provide 24% more power at the beginning of the mission [6]. The rover and lander will conduct surface operations within one Earth year but will have the capability to conduct more science experiments for 15 years post mission completion.

Table 5.3-1 MMRTG and eMMRTG Comparison



	MMRTG	enhanced MMRTG	*enhancement*
BOL Power (W)	124 *	~154 **	24%
Power at EODL (W)	55	~101	84%
Degradation Rate ***	4.8%	2.5%	~ 2x
No. of GPHS Bricks	8	8	
System Mass	44.1	44.1	
Hot Junction Temp	520 °C	600 °C	
Cold Junction Temp	~200 °C	~200 °C	
Mission Usage	Multi-Mission	Multi-Mission	
Development Risk	None	Low – Moderate	
Addressed Program	MSL, Mars 2020	Discovery, New Frontiers, Flagship	

BOL – Beginning Of Life, fueling
EDL – Entry, Descent, and Landing
EODL – End Of Design Life, 17 yrs from BOL

* 28V, Thermal Inventory = 244 Wth; 4k thermal sink
** 32V, Thermal Inventory = 244 Wth, 4k thermal sink
*** Steady-state thermal sink, Mars hot case

5.4 Manufacturing, Integration, and Test Concept

In our decision of who would manufacture, integrate, and test (MI&T) our vehicles, we focused primarily on previous success with similar vehicles and the ability to perform tests on the vehicles in-house. **Table 5.4-1** shows each vehicle and the company selected to manufacture, integrate, and test it as well as their location.

Table 5.4-1 Companies Selected for MI&T

Vehicle	Company	Location
Orbiter SRC	Lockheed Martin Space Systems	Littleton, Colorado
Rover Lander	NASA JPL	Pasadena, California
MAV	Lockheed Martin Space Systems And Aerojet Rocketdyne	Littleton, Colorado

Lockheed Martin Space Systems (LMSS) has been successful with their interplanetary spacecraft, such as MAVEN, JUNO, and OSIRIS-Rex. NASA JPL has mastered rovers, landers, and the EDL process with their many missions to Mars, such as Curiosity, Perseverance, and Phoenix. With Lockheed Martin's potential acquisition of Aerojet Rocketdyne (AR), their combined experience in space, rockets, and missiles will aid in the MI&T of a first-of-its-kind MAV. In the case of the acquisition not going through, we would still desire both companies to work together for MI&T.

Many components in the design are also COTS components, which decreases manufacturing and testing costs. This is due to the desire to use as many high TRL components as possible. Therefore, the companies selected to manufacture the vehicles can manufacture the rest of the components or contract out the manufacturing.

The orbiter and sample return capsule will be integrated at Lockheed Martin's Gateway Center's High Bay. The rover and lander will be integrated at NASA JPL's High Bay. Since Lockheed Martin and Aerojet Rocketdyne's acquisition process is still ongoing, we assumed MI&T would occur at Lockheed Martin's Gateway Center. **Table 5.4-2** shows a high-level vehicle testing plan.

Due to the mission returning samples from Mars, COSPAR V Mars requirements for planetary protection must be met to minimize contamination.[10] COSPAR requires category IVb requirements to also be met, which included restricting surface bioburden level to less than 30 spores. [10] Every vehicle will need to be assembled in clean rooms and be sterilized to meet these requirements. If a vehicle is able to, it will also be baked to kill microbes.

Table 5.4-2 High-Level Vehicle Testing Plan

Vehicle	Location	Comprehensive Systems Tests	Vibration Tests	Acoustic Tests	Deploy & Shock Tests	Thermal Vacuum Tests
• Orbiter • SRC	LMSS	Gateway Center: High Bay	Lockheed Denver Facilities	Lockheed Denver Facilities	Gateway Center: High Bay	Gateway Center: TVAC
• Lander • Rover	JPL	High Bay 1	NASA JPL Facilities	Acoustic Test Chamber	High Bay 1	Space Simulator Facility
• MAV	LMSS & AR	Gateway Center: High Bay	Lockheed Denver Facilities	Lockheed Denver Facilities	Gateway Center: High Bay	Gateway Center: TVAC

5.5 Maintenance Concept

Each vehicle was designed with electronics placed in a location with easy access if repairs are needed before launch. The MAV will have all electronics below the nosecone, aside from the sun sensors, which are accessible from outside and inside the skin. The rover will have all the main electronics next to the eMMRTG to help with heating and to help with accessibility. The lander will have all the main EDL electronics placed below the landing platform, and the other electronic systems will be placed above the platform to help with accessibility.

5.6 Disposal/End of Mission Concept

If budget allows, the rover and lander will continue science experiments post-mission completion. The rover will continue to collect ice cores and study their composition using the PIXL onboard. The rover will transmit data to nearby orbiters to relay back to Earth. The continuous science data on the ice cores will allow scientists to further their understanding of the composition of the ice on Mars. The MAV will stay in its low Mars orbit of 300 km until it deorbits due to aerodynamic drag. The MAV will then burn up in the atmosphere, and the surviving components of the vehicle will crash land on the surface of Mars. The orbiter will deliver the sample return capsule to Earth and enter a heliocentric orbit post-delivery. The orbiter will not have enough propellant to continue with another mission. Once the sample return capsule has been successfully recovered, it will be donated to California State Polytechnic University, Pomona, and will be placed on display for the community.

5.7 Risk Analysis

We performed risk analysis on our system, focusing primarily on what could impact the mission the most. Each risk has a likelihood and consequence value assigned to it. We developed risk statements defining the cause of a major risk as well as the overall consequence if it were to occur. These risks were mitigated to acceptable levels.

5.7.1 Risk Statements

Table 5.7.1-1 shows the risks as well as their likelihood and consequence values. "L" stands for likelihood, and "C" stands for consequence. "Pre" are the values before any mitigation, and "Post" are the values after mitigation.

Table 5.7.1-1 Risk Statement Table

ID	Risk Statement	Pre		Post	
		L	C	L	C
RSK.1	If the system is unable to launch during the launch window due to delays in design and I&T, then the system will fail to meet the required return date	2	4	1	4
RSK.2	If the Falcon Heavy is unable to be launch service program category 3 certified to carry RTGs, due to not reaching three consecutive successful flights, then the mission could still be performed, but at an increased cost and delayed schedule	3	5	1	5
RSK.3	If a vehicle is unable to complete its mission-critical tasks, due to component failure, then the samples would not be delivered to Earth	3	5	1	3
RSK.4	If the ice core samples are contaminated due to coming into contact with Earth contaminants, then the mission will be compromised	2	5	1	5
RSK.5	If the rover is unable to find deep enough ice to drill, due to landing in the wrong place, then the ice cores will not meet the required dimensions	1	3	1	1
RSK.6	If the MAV is unable to launch, due to the rover not being able to complete drilling operations before five months of total darkness, then the samples would not be delivered on time or delivered at all	3	5	1	4
RSK.7	If the orbiter is unable to dock with the MAV, due to the failure of the docking sensor, then the samples would not be delivered to Earth	1	5	1	3
RSK.8	If the lander is unable to transfer the samples from the rover to the MAV, due to the robotic arm failing, then the samples would not be delivered to Earth	2	5	1	5
RSK.9	If the MAV is unable to achieve orbit, due to inaccuracy in Mars atmospheric models, then the samples would not be delivered to Earth	2	5	1	5

5.7.2 Risk Mitigation

Table 5.7.2-1 shows the mitigation steps which we took for each risk defined in **Table 5.7.1-1**. The bolded values in the yellow cells show the final likelihood and consequence value for its respective risk.

Table 5.7.2 Risk Mitigation Steps

Risk ID	Mitigation	L	C
RSK.1	Push launch date back two years to allow more time for design and MI&T	**1**	**4**
RSK.2	The mitigation step of pushing the launch date back two years from RSK.1 also allows more time for the Falcon Heavy to be certified	2	5
	During CDR, follow NPR 8715.3D approval steps by requesting a concurrence letter and approval by the Nuclear Flight Safety Assurance Manager and by reporting to the Office of Science and Technology Policy.	**1**	**5**
RSK.3	Redo selection of components and prioritize built-in redundancy and low failure rates	2	4

RSK.3	Add redundant components when mass and power allow	**1**	**3**
RSK.4	Implement planetary protection requirements into the system	**1**	**5**
RSK.5	Reanalyze and reselect landing equipment	1	2
	More propellant margin for lander	**1**	**1**
RSK.6	Design system to hibernate and survive five months of total darkness	2	5
	Reanalyzing the power subsystem for lander and rover with total darkness duration in mind resulted in an RTG being selected for both vehicles to provide heat and power.	**1**	**5**
RSK.7	Design orbiter and MAV to communicate and send telemetry to each other during docking	**1**	**3**
RSK.8	Perform a large amount of testing with the lander arm and the sample transfer sequence	**1**	**5**
RSK.9	Add more margin into the fuel to cover unexpected atmospheric losses	**1**	**5**

5.7.3 Risk Cubes

Table 5.7.3-1 shows a risk cube for the risks defined in **Figure 5.73-1** before mitigation and after mitigation.

LIKELIHOOD						
	5					
	4					
	3					**2, 3, 6**
	2				**1, 5**	**4, 8, 9**
	1			**5**		**7**
		1	2	3	4	5
	CONSEQUENCE					

LIKELIHOOD						
	5					
	4					
	3					
	2					
	1	**5**		**3, 7**	**1, 6**	**2, 4, 8, 9**
		1	2	3	4	5
	CONSEQUENCE					

Figure 5.7.3-1 Risk Cube Before and After Mitigations

5.8 Cost Analysis

5.8.1 NASA PCEC

A cost estimate was performed for the MICKEY Architecture using the NASA Price Cost Estimating Capability (PCEC) Tool. The PCEC tool uses Cost Estimation Relationships (CERs) based on various inputs into an excel file consisting of many sheets. The CERs were compiled using the entire history of NASA space missions, taking a known parameter from the vehicle and graphing them. Once all of the missions are listed on the graph, a trendline is generated and that trendline is used to estimate the cost of something based on that parameter. The software does this automatically and compiles the results into a Work Breakdown Structure (WBS). A graphical WBS was constructed based on those numbers and shown in **Figure 5.8.1-1**. The dollar amounts shown are all in the fiscal year 2020. The PCEC tool is able to adjust for inflation and was used on all parts of the cost analysis.

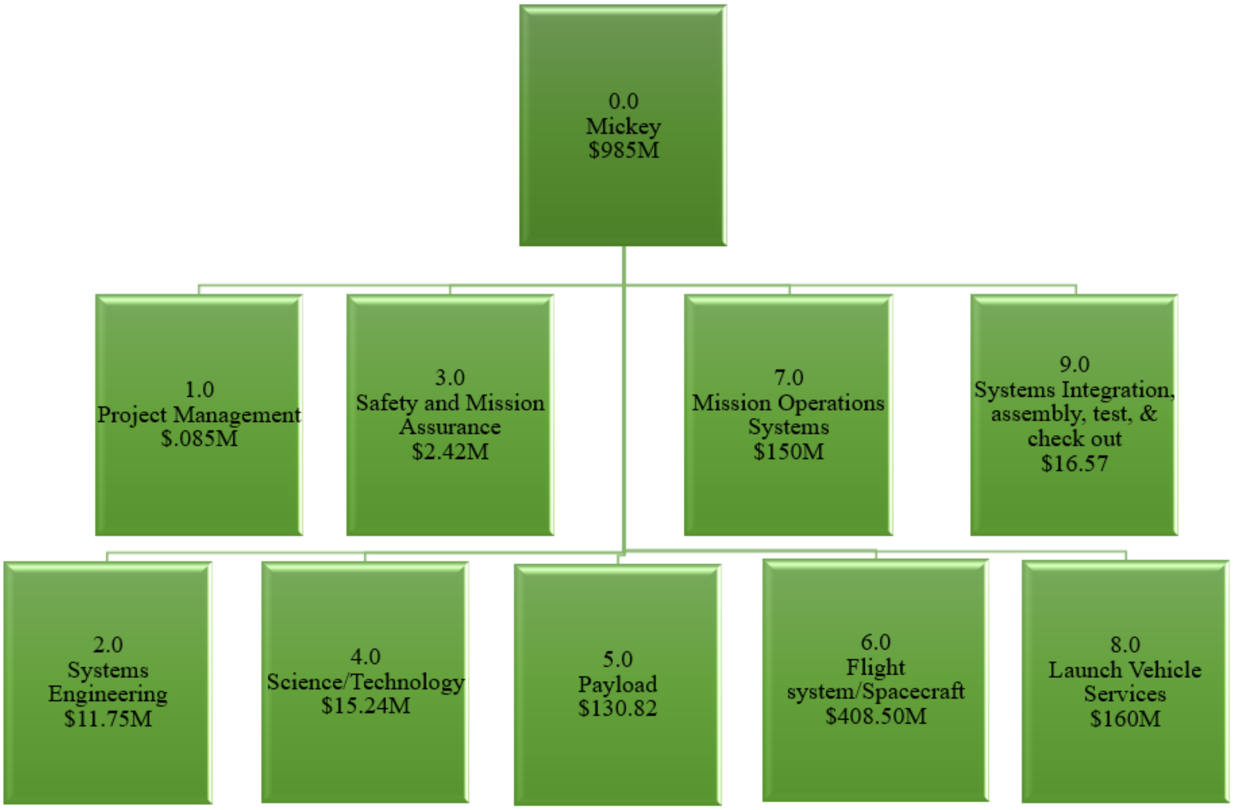


Figure 5.8.1-1 Cost Analysis WBS

5.8.2 Cost Analysis Summary

The WBS separated the total system cost in the mission and was placed into **Table 5.8.2-1**. Unfortunately, the value for the Ground Data System was unable to be calculated by the NASA PCEC tool; additionally, Mission Ops & Data Analysis were excluded from the overall costs based on the RFP requirements. Some assumptions were also made that went into the overall cost analysis of this design. The NASA PCEC tool did not have the expendable Falcon Heavy listed as a possible launch vehicle, so the price was manually inputted with a margin included for insurance costs. The PCEC tool does not have the option for eMMRTG power so to estimate the cost of having two eMMRTGs power our mission, we input the amount of power needed for our mission and added the cost, in addition to manually inputting the cost of two eMMRTGs. The cost of two eMMRTGs was found to be $94 million dollars*, a 32% margin was added to the cost for safety, testing, and various other costs associated with having eMMRTGs on a mission.* The sample acquisition drill cost was manually inputted into the spacecraft portion of the cost analysis.

*according to our inquiry with JPL technical person on eMMRTGs

The cost of the drill was assumed to be $25 million. Once the PCEC tool had calculated the cost of each category, a 10% margin was added to the total cost resulting in the final price of $985 million based on FY2020. This can be seen in **Table 5.8.2-1**.

Table 5.8.2-1 Cost Breakdown Summary with Total Cost

WBS #	Level	WBS Element	Total (FY2020 $M)
0	1	System Name- MICKEY	-------
1.0	2	Project Management	$ 0.85
2.0	2	Systems Engineering	$ 11.70
3.0	2	Safety and Mission Assurance	$ 2.42
4.0	2	Science/Technology	$ 15.24
5.0	2	Payload(s)	$ 130.82
6.0	2	Flight System/Spacecraft	$ 408.50
7.0	2	Mission Operations System (MOS)	$ 150.00
8.0	2	Launch Vehicle/Services	$ 160.00
9.0	2	Ground Data System (GDS)	$ -
10.0	2	System Integration, Assembly, Test & Check Out	$ 16.57
--	3	Mission Ops & Data Analysis (Phase E)	$ 95.53

	Total (FY2020 $M)
Margin = 10%	$ 985.71

5.9 Compliance Matrix

The RFP provided sections on design requirements, constraints, and deliverables. **Table 5.9-1** shows design requirements and constraints defined in the RFP, their completion status, the sections of the report that meet the requirement, and an ID we assigned based on the section and bullet point/indentation shown in the RFP.

Table 5.9-1 Design Requirements and Constraint Compliance Matrix

ID	Statement	Comply	Section(s)	Page(s)
DRC.1	Design a robotic mission to the surface of Mars with the primary goal of returning a minimum of 2.5 kg of ice core samples back to Earth	Yes	3.5,4.3.2	8-10, 53-55
DRC.1.1	The designed system should deliver a robotic system that can land on or near Martian ice deposits	Yes	3.7, 4.2	15-21, 43-51
DRC.1.2	The system should be capable of perform drilling operation on the Martian surface with the express purpose of retrieving ice core samples	Yes	4.3.2	51-53
DRC.1.3	The ice core samples should be at least 25 millimeters in diameter, and 100 millimeters in length	Yes	4.3.2	51-53
DRC.1.4	The robotic system needs to be capable of storing the ice cores in a frozen state during surface operations	Yes	4.3.2	51-53

DRC.1.5	The system must return a minimum of 2.5 kg of ice cores in its frozen state back to Earth and accommodate the safe transfer of the ice cores to Earth-based laboratories in their frozen state"	Yes	3.7, 4.3.2, 4.5	18-19, 51-53, 67-69
DRC.2	Design and define the end-to-end mission operations, including launch, transit to Mars, entry/descent/landing, surface operation, ascent, and return to Earth	Yes	3.5	8-10
DRC.2.1	Select a mission architecture and vehicle design that maximizes the science data return within the cost and schedule constraints	Yes	3.4	8
DRC.2.2	Discuss the selection of target locations and the values of the selected site, including the assessment criteria	Yes	3.6	10-13
DRC.3	Perform trade studies on system options at the system and subsystem level to demonstrate the fitness of the chosen mission design. It is highly desirable to use technologies that are already demonstrated on previous programs or currently in the NASA technology development portfolio. Advanced technology can be used; however, cost, schedule, and risk consideration of utilizing advanced technology must be included in proposal.	Yes	4.0, Appendix A	24-69, 83-84
DRC.4	Discuss selection of subsystem components, including mass, power, and volume, and how the design requirements drove the selection of the subsystem	Yes	4.0	24-69
DRC.5	The cost for end-to-end mission shall not exceed $1 Billion US Dollars (in FY20), including launch, design development test and evaluation (DDT&E) and flight unit costs for the mission	Yes	5.8	75-77
DRC.6	If advanced technology options are utilized in the design, estimation of technology advancement cost must be included	Yes	5.8	75-77
DRC.7	The ice core sample must be returned to Earth for scientific analysis no later than December 31, 2030.	Yes	3.5, 3.7	8-10, 13-19

Table 5.9-2 shows the deliverables and the same categories as **Table 5.9-1**. We assigned the ID numbering based on the number defined in the RFP, and the sentence number.

Table 5.9-2 Deliverables Compliance Matrix

ID	Statement	Comply	Section(s)	Page(s)
	Requirement Definition			
DLV.1.1	The report should include the mission and design requirements at the vehicle, system, and subsystem level.	Yes	4.0, Appendix B	24-69, 85-91
DLV.1.2	The requirements definition should demonstrate the team's understanding of the RFP Design Requirements and Constraints and lay the foundation for the design decisions that follow.	Yes	Appendix B	85-91
	Concept of Operations			
DLV.2.1	A detailed concept of mission operation should be included to describe all phases of the mission and to demonstrate the realization of the mission requirements in Design Requirements and Constraints.	Yes	3.5	8-10

DLV.2.2	The report must discuss how each subsystem level decision is made, with description of the selection metrics and their associated weightings when appropriate, and provide detailed discussions on how each decision impact system level metrics such as cost, schedule, and risk.	Yes	4.0	24-69
Trade Studies				
DLV.3.1	The report should include the trade studies for the vehicle architecture, mission operations, and subsystem selections, and must discuss in detail how the system level requirement are developed from mission requirements by describing the pro and cons of each subsystem options.	Yes	4.0, Appendix A	24-69, 83-84
DLV.3.2	The report must discuss how each subsystem level decision is made, with description of the selection metrics and their associated weightings when appropriate, and provide detailed discussions on how each decision impact system level metrics such as cost, schedule, and risk.	Yes	4.0	24-69
Design Integration and Operation				
DLV.4.1	The report should discuss how the trades selected in section 3 are integrated into a complete architecture.	Yes	4.0	24-69
DLV.4.2	This section should discuss design of all subsystems: structures, mechanisms, thermal, attitude control, telemetry, tracking, and command, electric power, propulsion, payload and sensors, and the mission concept of operations.	Yes	3.5, 4.0	8-10, 24-69
DLV.4.3	Discussion on the extensibility of the overall system design and how it can support future exploration mission should be included.	Yes	5.3, 5.6	71, 73
DLV.4.4	A mass and power budget must be included, broken down by subsystem, with appropriate margins assigned to each system based on industry standards.	Yes	4.1.10, 4.2.9, 4.3.9, 4.4.10, 4.5.8	42, 50-51, 58-59, 66-67, 69
DLV.4.5	The report must clearly describe all of the tools and methods utilized for the system and subsystem design and provide brief description of the inputs, outputs, and assumptions for the design.	Yes	3.1, 4.0, 5.1	7, 24-69, 70
DLV.4.6	A discussion on the validation of the tools and methods must be included.	Yes	3.7, 4.0	13, 24-69
DLV.4.7	A summary table should be prepared showing all mass, power, and other resource requirements for all flight elements/subsystems with the appropriate mass and power margins clearly labeled and discussed.	Yes	4.1.10, 4.2.9, 4.3.9, 4.4.10, 4.5.8	42, 50-51, 58-59, 66-67, 69
Cost Estimate				
DLV.5.1	Top level cost estimate covering the life cycle for all cost elements should be included.	Yes	5.8.1	75
DLV.5.2	Work Breakdown Structure (WBS) should be prepared to capture each cost element including all flight hardware, ground systems, test facilities, and other requirements for the design.	Yes	5.8.1	75
DLV.5.3	Estimates should cover design, development,	Yes	5.8	75-77

	manufacture, assembly, integration and test, launch operations and checkout, in-space operations, and final delivery to the Martian surface and return to the Earth.			
DLV.5.4	Use of existing/commercial off-the-shelf hardware is strongly encouraged.	Yes	4.0, 5.8	24-69, 75-77
DLV.5.5	Advanced technology utilization must be fully costed with appropriate cost margin applied.	Yes	5.8	75-77
DLV.5.6	A summary table should be prepared showing costs for all WBS elements distributed across the various project life cycle phases.	Yes	5.8.2	76
DLV.5.7	The report should discuss the cost model employed and describe the cost modeling methods and associated assumptions in the cost model.	Yes	5.8	75-77
DLV.5.8	The cost analysis should provide the appropriate cost margin based on industry standards.	Yes	5.8	75-77
Schedule				
DLV.6.1	A mission development and operation schedule should be included to demonstrate the mission meets the schedule deadline established in the RFP.	Yes	3.5, 3.7, 5.2	8-10, 13-20, 70-71
DLV.6.2	Schedule margin should be applied to appropriate areas with funded schedule reserve detailed in the cost estimate.	Yes	5.2, 5.8	70-71, 75-77
DLV.6.3	Any advanced technology assumption should have corresponding technology development schedules and costs associated with the technology and appropriate contingency plans should be discussed.	Yes	5.2	70-71
Summary and References				
DLV.7.1	A concise, 5 page “Executive Summary” of the full report must be included and clearly marked as the summary at the beginning of the report.	Yes	1.0	1-5
DLV.7.2	The executive summary should provide a clear sense of the project’s motivation, process, and results.	Yes	1.0	1-5
DLV.7.3	References should be included at the end.	Yes	7.0	82
DLV.7.4	A compliance matrix, listing the page numbers in the report where each these section as well as the items identified under the Design Requirements and Constraints and Deliverables sections can be found, is mandatory.	Yes	5.9	77-80

6.0 Conclusion

We designed the Mars Ice Core Key Exploration Yacht (MICKEY) to recover 3.0 kg of Martian ice cores within acceptable risk. Each vehicle went through multiple design iterations to improve its overall efficiency and to account for possible failures. The rover and lander were designed to expeditiously recover ice cores but could hibernate through 5 months of darkness in the case that the mission would not be completed before the dark period. The mission could be completed successfully even if the vehicles hibernate together. The rover is equipped with a PIXL camera which will provide useful information on the ice cores' composition when they are collected. The rover also has the capability to continue this after it has completed its mission post-MAV launch. The vehicles were all designed to incorporate instruments that were only necessary to complete the mission to reduce mass and costs. The entire mission will cost $985 million (FY20), which meets the $1 billion (FY20) budget requirement. The mission will return the samples on May 24, 2029, which is before the December 31, 2030 deadline. Each subsystem for every vehicle was chosen with trade studies that used common figures of merit such as cost, mass, and TRL. The mission required the least amount of risk because the mission will be completed autonomously due to the One Way Light Travel time. Our team has also determined the best locations for manufacturing, assembly, and testing for each vehicle. Ares Advena Labs is pleased to present an efficient and reliable system to retrieve ice cores from Mars. When we successfully recover the ice cores, it will pave the way for future exploration missions to Mars and beyond and help humans become a multi-planetary civilization.

7.0 References

[1] Brown, C. D. (2002). *Elements of spacecraft design*. Reston, VA: American Institute of Aeronautics and Astronautics.

[2] Johnston, Mark D, et al. *Request-Driven Scheduling for NASA's Deep Space Network*. California Institute of Technology, 2009, https://ai.jpl.nasa.gov/public/documents/papers/johnston_iwpss09_request.pdf

[3] Sepka, S. A., & Samareh, J. A. (2015, June 26). Thermal Protection System Mass Estimating Relationships For Blunt-Body, Earth Entry Spacecraft https://ntrs.nasa.gov/api/citations/20150018864/downloads/20150018864.pdf

[4] Brugarolas, P. B., San Martin, A., & Wong, E. C. (2006, December 8). Entry Attitude Controller for the Mars Science Laboratory. Retrieved from https://citeseerx.ist.psu.edu/viewdoc/download?doi=10.1.1.820.7743&rep=rep1&type=pdf

[5] Aerojet Rocketdyne, "In-Space Propulsion Data Sheet," April 8, 2020

[6] Woerner, David. (2015). *30W Boost in MMRTG Power Delivered at the Beginning of Life & 50W at End of Design Life*. NASA and US Dept. of Energy.

[7] Brown, Adrian J., Bryne, Shane, Tornabene, Livio L., Roush, Ted. "Louth crater: Evolution of a layered water ice mound." Icarus, Volume 196, Issue 2, 2008,

[8] Curtis, Howard D. Orbital Mechanics for Engineering Students Fourth Edition. Butterworth-Heinemann, An Imprint of Elsevier, 2020.

[9] Young H., et al. Radioisotope Power System Effects on Science Instruments and Measurements. Jet Propulsion Laboratory, California Institute of Technology,NASA Glenn Research Center, www.lpi.usra.edu/opag/meetings/aug2016/posters/Lee.pdf.

[10] Kminek, G., et al. "COSPAR's Planetary Protection Policy." COSPAR, 2017.

[11] Ralph Stone and Company, Inc. Investigations of Lunar Sampling and Sample Return Methods.

[12] Falcon User's Guide. SpaceX, Apr. 2022, www.spacex.com/media/falcon_users_guide_042020.pdf.

[13] "NPR 8715.3D - Chapter6." NASA, NASA, nodis3.gsfc.nasa.gov/displayDir.cfm?Internal_ID=N_PR_8715003D&page_name=Chapter6.

[14] Hoffman, Stephen, et al. "Mining" Water Ice on Mars An Assessment of ISRU Options in Support of Future Human Missions . NASA, www.nasa.gov/sites/default/files/atoms/files/mars_ice_drilling_assessment_v6_for_public_release.pdf.

Appendix

Appendix A: Trade Matrices

Table: A.1 Landing Site Trade Study Matrix

FOM # / Arch	Water Ice Abundance WF = 3 U		W	Water Ice Depth WF = 3 U		W	Landing Surface Env. WF = 2 U		W
Utopia Planitia	Arch. Value	Medium	9	Arch. Value	0.5 m	3	Arch. Value	Smooth	6
	Score	3		Score	1		Score	3	
Arcadia Planitia	Arch. Value	Medium	9	Arch. Value	< 0.1 m	9	Arch. Value	Smooth	6
	Score	3		Score	3		Score	3	
North Pole	Arch. Value	Medium	12	Arch. Value	< 0.1 m	9	Arch. Value	Rough	2
	Score	4		Score	3		Score	1	
South Pole	Arch. Value	Medium	9	Arch. Value	< 0.1 m	9	Arch. Value	Rough	2
	Score	3		Score	3		Score	1	
Louth Crater	Arch. Value	High	27	Arch. Value	0	27	Arch. Value	Rough	4
	Score	9		Score	9		Score	2	

Landing Elevation WF = 2 U		W	Terrain/Environmental Challenges WF = 3 U		W	Weight Total
Arch. Value	Likely	6	Arch. Value	Likely	8	32
Score	3		Score	4		
Arch. Value	Very Likely	18	Arch. Value	Very Likely	20	62
Score	9		Score	10		
Arch. Value	Unlikely	4	Arch. Value	Very Likely	20	47
Score	2		Score	10		
Arch. Value	Unlikely	2	Arch. Value	Unlikely	4	26
Score	1		Score	2		
Arch. Value	Very Likely	18	Arch. Value	Very Likely	18	94
Score	9		Score	9		

Table A.2 Reaction Wheel Trade Matrix

Reaction Wheel \ FOMs	Power Usage (W) WF = 3			Mass (kg) WF = 2			Max Torque (N-m) WF = 2			Temp. Range (°C) WF = 1			Angular Momentum Storage (N-m-s) WF = 1			Weighted Total
	U		W	U		W	U		W	U		W	U		W	
Mini-wheel	Arch. Value	6	12	Arch. Value	1.3	8	Arch. Value	0.028	10	Arch. Value	-25 - 60	2	Arch. Value	1	0	32
	Score	4		Score	4		Score	5		Score	2		Score	0		
HR 0610	Arch. Value	15	9	Arch. Value	5	6	Arch. Value	0.055	18	Arch. Value	-15-60	1	Arch. Value	12	9	43
	Score	3		Score	3		Score	9		Score	1		Score	9		
HR 12	Arch Value	22	3	Arch Value	7	4	Arch Value	0.2	0	Arch Value	-30-60	3	Arch Value	50	0	10
	Score	1		Score	2		Score	0		Score	3		Score	0		
HR 2030	Arch. Value	20	6	Arch. Value	11.2	2	Arch. Value	0.21	0	Arch. Value	-15-80	4	Arch. Value	45.6	0	12
	Score	2		Score	1		Score	0		Score	4		Score	0		

Table A.3 RCS Thruster Trade Matrix

Thrusters \ FOMs	Total Pulses WF = 3			Total Impulse (N-s) WF = 3			Power Usage (W) WF = 2			Specific Impulse (s) WF = 2			Thrust Range (N) WF = 2			Fuel Hazard Level WF = 1			Operating Temp Range (°C) WF = 1			Weighted Total
	U		W	U		W	U		W	U		W	U		W	U		W	U		W	
1 N HPGP Thruster	Arch. Value	60k	27	Arch. Value	67.5k	9	Arch. Value	10	18	Arch. Value	217.5	6	Arch. Value	0.25-1	0	Arch. Value	Low	9	Arch. Value	-5 - 60	4	73
	Score	9		Score	3		Score	9		Score	3		Score	0		Score	9		Score	4		
MR-103J 1N	Arch. Value	1m	27	Arch. Value	183k	27	Arch. Value	16.1	18	Arch. Value	213	6	Arch. Value	0.19-1.13	18	Arch. Value	High	0	Arch. Value	10-50	3	99
	Score	9		Score	9		Score	9		Score	3		Score	9		Score	0		Score	3		
BPT-2000 Hall Effect Thruster	Arch Value	6k	0	Arch Value	2.6m	27	Arch Value	2200	0	Arch Value	1765	18	Arch Value	0.12	0	Arch Value	Low	9	Arch Value	-100 - 301	5	59
	Score	0		Score	9		Score	0		Score	9		Score	0		Score	9		Score	5		
BGT-X5	Arch. Value	10k	0	Arch. Value	565	0	Arch. Value	20	0	Arch. Value	222.5	6	Arch. Value	0.08-0.5	18	Arch. Value	Low	9	Arch. Value	-5 - 60	4	37
	Score	0		Score	0		Score	0		Score	3		Score	9		Score	9		Score	4		

Appendix B: Requirements Table

ID	Requirement Statement	Derives	Satisfied By	Verified By
CUST.1	The system shall use robotic vehicles to accomplish the mission		System	System Ops Demo System Ops Sim
CUST.2	The system shall operate on the surface of Mars		System	Environment Tests Environment Sim
CUST.3	The system shall land on or near Martian ice deposits		System	Landing Sim
CUST.4	The system shall drill ice cores on Mars		System	Drilling Ops Demo Inspect Drill Ops Plan
CUST.5	The system shall return ice cores with a diameter greater than 25 mm and a length greater than 100 mm		System	Inspect Ice From Demo Drilling Sim Drilling Ops Demo
CUST.6	The system shall keep ice cores frozen during entire mission operations		System	Test Thermal Ranges Thermal Analysis
CUST.7	The system shall return at least 2.5kg of Mars ice cores to Earth		System	Mission Sim
CUST.8	The system shall accommodate the safe transfer of ice cores to laboratories on Earth		System	Earth Return Ground Ops Demo
CUST.9	The system shall return ice cores to Earth no later than December 31, 2030		System	Trajectory Sim
CUST.10	The system's launch, DDT&E, and flight unit shall cost no more than $1 billion (FY 2020)		System	Cost Calc Cost Estimation
SYS.1	The system shall be launched on November 1, 2026 (-40 to +10 days)	CUST.9	System	Trajectory Sim
SYS.2	The system shall arrive at Mars no later than August 18, 2027.	CUST.9	System	Trajectory Sim
SYS.3	The system shall leave Mars no later than October 6, 2028 (± 20 days)	CUST.9	System	Trajectory Sim
SYS.4	The system shall deliver samples to Earth no later than May 24, 2029.	CUST.9	System	Trajectory Sim
SYS.5	The system shall use autonomous vehicles to accomplish the mission	CUST.1	System	System Ops Demo
SYS.6	The system shall operate on the surface of Mars during a minimum of 7 months of day/night cycles	CUST.2	System	Mars TVAC Testing Mars Environment Sim
SYS.7	The system shall operate on the surface of Mars in total darkness for a maximum of 5 months	CUST.2 CUST.3	System	Mars Environment Sim Mars TVAC Testing
SYS.8	The system shall use a rover to drill ice cores samples on Mars	CUST.4 CUST.2 CUST.5	System	Drilling Ops Demo Inspect Drill Ops Plan ConOps Inspection
SYS.9	The system shall comply with COSPAR Category V Mars planetary protection requirements.		System	Trajectory Sim ConOps Inspection
SYS.10	The system shall use a lander to deliver the MAV and rover to the Mars surface near Martian ice Deposits	CUST.3 CUST.2	System	Landing Sim ConOps Inspection
SYS.11	The system shall use the lander to transfer a sample cryocooler from the rover to the MAV	CUST.4 CUST.2 CUST.7	System	ConOps Inspection
SYS.12	The system will drill ice cores will a diameter no more than 50mm	CUST.5	System	Drilling Ops Demo Inspect Ice From Demo
SYS.13	The system will drill ice cores with a length no more than 100mm	CUST.5	System	Drilling Ops Demo Inspect Ice From Demo

SYS.14	The system shall keep the ice core samples' temperature less than -5 degrees Celsius.	CUST.6	System	Vehicle TVAC Test Vehicle Thermal Analysis
SYS.16	The system shall use an orbiter to carry the SRC	CUST.7	System	ConOps Inspection
SYS.17	The system shall use an orbiter in orbit around Mars	CUST.7	System	Trajectory Sim ConOps Inspection
SYS.18	The system shall use the orbiter to return the SRC to Earth	CUST.7	System	Trajectory Sim ConOps Inspection
SYS.19	The system shall use the SRC to deliver the sample cryocooler to Earth's surface	CUST.8	System	ConOps Inspection
SYS.20	The system shall use a launch vehicle that cost no more than $200 million	CUST.10	System	Inspect Price from LV Provider Estimate Launch Cost
SYS.21	The system shall use a 10% cost margin until PDR is completed.	CUST.10	System	Check Cost Margin in Cost Estimation Results
SYS.22	The system shall use the SRC to receive the sample cryocooler from the MAV	CUST.7	System	System Ops Demo ConOps Inspection
SYS.23	The system shall use a MAV to deliver a sample cryocooler to the SRC on the orbiter in orbit around Mars	CUST.7	System	System Ops Demo ConOps Inspection
SYS.24	The system shall return between 2.5kg and 3kg of Mars ice cores to Earth	CUST.7	System	Test Mass with Different Config of Samples
SYS.25	The system's launch mass shall not exceed 12,440 kg	CUST.10	System	Launch Config Mass Meas Launch Config Mass Calc
SYS.26	The system's launch mass should not exceed 7456.5 kg	CUST.10	System	Launch Config Mass Calc Launch Config Mass Meas
LND.1	The lander shall separate from the orbiter no later than November 1, 2026.	SYS.2	LND	Trajectory Sim
LND.2	The lander shall be an autonomous vehicle	SYS.5	LND	LND Ops Demo
LND.3	The lander shall operate for five months in total darkness on Mars	SYS.7	LND	LND TVAC Test LND Thermal Analysis LND Power Config Sim
LND.3.1	The lander's TCS shall keep the vehicle's temperature between -10 and 35 degrees Celsius.	LND.3 LND.4	LND	LND TVAC Test LND Thermal Analysis
LND.4	The lander shall operate during a minimum of seven months of day/night cycles	SYS.6	LND	LND TVAC Test LND Power Config Sim LND Thermal Analysis
LND.5	The mass of the lander in the cruise stage configuration should not exceed 2737.5 kg	SYS.25 SYS.26	LND	LND Mass Calcs LND Mass Meas
LND.6	The lander shall perform a 90-degree roll in five seconds	SYS.10	LND	LND ACS Sim
LND.6.1	The lander's ACS system shall perform a 90-degree roll in 5 seconds	LND.8	LND ACS	LND ACS Sim
LND.6.2	The lander's ACS shall have a point accuracy of less than 3 degrees	LND.8	ORB ACS	LND ACS Sim
LND.7	The lander's power summation during cruise should not exceed 1046 W.	SYS.10	LND	LND Power Analysis
LND.7.1	The lander power margin's power summation during cruise should not exceed 424.4 W.	LND.9	LND Margin	LND Margin Power Analysis
LND.7.2	The lander PLDS's power summation during cruise should not exceed 150 W.	LND.9	LND PLDS	LND PLDS Max Power Test LND PLDS Power Analysis
LND.7.3	The lander PRPS's power summation during cruise should not exceed 4.7 W.	LND.9	LND PRPS	LND PRPS Max Power Test LND PRPS Power Analysis
LND.7.4	The lander TCS's power summation during cruise should not exceed 132 W.	LND.9	LND TCS	LND TCS Max Power Test LND TCS Power Analysis

LND.7.5	The lander ACS's power summation during cruise should not exceed 94.3 W.	LND.9	LND ACS	LND ACS Power Analysis LND ACS Max Power Test
LND.7.6	The lander CDS's power summation during cruise should not exceed 80.2 W.	LND.9	LND CDS	LND CDS Power Analysis LND CDS Max Power Test
LND.7.7	The lander EPS's power summation during cruise should not exceed 47.2 W.	LND.9	LND EPS	LND EPS Power Analysis LND EPS Max Power Test
LND.7.8	The lander TLCS's power summation during cruise should not exceed 108.5 W.	LND.9	LND TLCS	LND TLCS Max Power Test LND TLCS Power Analysis
LND.7.9	The lander SMS's power summation during cruise should not exceed 4.7 W.	LND.9	LND SMS	LND SMS Max Power Test LND SMS Power Analysis
LND.8	The lander shall have a delta-V of at least 250 m/s	SYS.10	LND	LND Prop Analysis
LND.9	The lander shall arrive at Mars no later than August 18, 2027.		LND	Trajectory Sim
LV.1	The launch vehicle shall launch the system on November 1, 2026 (-40 to +10 days)	SYS.1	LV	Trajectory Sim
MAV.1	The MAV shall be an autonomous vehicle	SYS.5	MAV	MAV Ops Demo
MAV.2	The MAV shall be able to launch in total darkness	SYS.7	MAV	MAV TVAC Test MAV Launch Sim MAV Thermal Sim MAV Power Analysis
MAV.2.1	The MAV's TCS shall keep the vehicle's temperature between -10 and 60 degrees Celsius.	MAV.2 MAV.3	MAV TCS	MAV TVAC Test MAV Power Analysis MAV Thermal Sim
MAV.3	The MAV shall be able to launch during a minimum of seven months of day/night cycles	SYS.6	MAV	MAV TVAC Test MAV Power Analysis MAV Thermal Sim MAV Launch Sim
MAV.4	The MAV shall withstand wind speeds of 30 m/s	SYS.6	MAV	MAV Wind Sim MAV Wind Tunnel Test
MAV.5	The MAV shall deliver the sample cryocooler with samples to the orbiter	SYS.23	MAV	Docking Sim ConOps Inspection
MAV.6	The MAV mass should not exceed 416.4 kg	SYS.25 SYS.26 SYS.23	MAV	MAV Mass Calc MAV Mass Meas
MAV.6.1	The MAV's PLDS's mass should not exceed 24 kg.	MAV.6	MAV PLDS	MAV PLDS Mass Calc MAV PLDS Mass Meas
MAV.6.2	The MAV's PRPS's dry mass should not exceed 4.62 kg.	MAV.6	MAV PRPS	MAV PRPS Mass Meas MAV PRPS Mass Calc
MAV.6.3	The MAV's PRPS's wet mass should not exceed 350.8 kg.	MAV.6	MAV PRPS	MAV PRPS Mass Calc MAV PRPS Mass Meas
MAV.6.3.1	The MAV's Pressurant's mass should not exceed 16.48 kg.	MAV.6.3	MAV Pressurant	MAV Pressurant Mass Meas MAV Pressurant Mass Calc
MAV.6.3.2	The MAV's Propellant's mass should not exceed 329.7 kg.	MAV.6.3	MAV PRPS	MAV Propellant Mass Calc MAV Propellant Mass Meas
MAV.6.4	The MAV's TCS's mass should not exceed 2.77 kg.	MAV.6	MAV TCS	MAV TCS Mass Meas MAV TCS Mass Calc
MAV.6.5	The MAV's CDS's mass should not exceed 2.77 kg.	MAV.6	MAV CDS	MAV CDS Mass Calc MAV CDS Mass Meas
MAV.6.6	The MAV's EPS's mass should not exceed 18.03 kg.	MAV.6	MAV EPS	MAV EPS Mass Meas MAV EPS Mass Calc
MAV.6.6.1	The MAV's Cabling's mass should not exceed 1.85 kg.	MAV.6.6	MAV Cabling	MAV Cabling Mass Calc MAV Cabling Mass Meas
MAV.6.7	The MAV's TLCS's mass should not exceed 1.85 kg.	MAV.6	MAV TLCS	MAV TLCS Mass Meas MAV TLCS Mass Calc

MAV.6.8	The MAV's ACS's wet mass should not exceed 4.62 kg.	MAV.6	MAV ACS	MAV ACS Mass Meas MAV ACS Mass Calc
MAV.6.9	The MAV's SMS's mass should not exceed 11.56 kg.	MAV.6	MAV SMS	MAV SMS Mass Calc MAV SMS Mass Meas
MAV.6.10	The MAV's EPS's mass without cabling should not exceed 16.18 kg.	MAV.6	MAV EPS	MAV EPS Mass Meas MAV EPS Mass Calc
MAV.7	The MAV's power summation should not exceed 419.4 W.	SYS.23	MAV	MAV Power Analysis
MAV.7.1	The MAV's power margin should be 192.5 W.	MAV.7	MAV Margin	MAV Margin Power Analysis
MAV.7.2	The MAV's PLDS's power summation should not exceed 13 W.	MAV.7	MAV PLDS	MAV PLDS Max Power Test MAV PLDS Power Analysis
MAV.7.3	The MAV's PRPS's power summation should not exceed 2.1 W.	MAV.7	MAV PRPS	MAV PRPS Power Analysis MAV PRPS Max Power Test
MAV.7.4	The MAV's TCS's power summation should not exceed 59.9 W.	MAV.7	MAV TCS	MAV TCS Max Power Test MAV TCS Power Analysis
MAV.7.5	The MAV's ACS's power summation should not exceed 42.8 W.	MAV.7	MAV ACS	MAV ACS Max Power Test MAV ACS Power Analysis
MAV.7.6	The MAV's CDS's power summation should not exceed 36.4 W.	MAV.7	MAV CDS	MAV CDS Power Analysis MAV CDS Max Power Test
MAV.7.7	The MAV's EPS's power summation should not exceed 21.4 W.	MAV.7	MAV EPS	MAV EPS Power Analysis MAV EPS Max Power Test
MAV.7.8	The MAV's TLCS's power summation should not exceed 49.2 W.	MAV.7	MAV TLCS	MAV TLCS Max Power Test MAV TLCS Power Analysis
MAV.7.9	The MAV's SMS's power summation should not exceed 2.1 W.	MAV.7	MAV SMS	MAV SMS Power Analysis MAV SMS Max Power Test
MAV.8	The MAV shall have a delta-v of at least 4,500 m/s	SYS.23	MAV	MAV Prop Sim
MAV.9	The MAV shall autonomously dock with the orbiter within four hours of launch	SYS.23	MAV	Trajectory Sim Docking Sim
ORB.1	The orbiter shall enter an Earth return trajectory no later than October 6, 2028 (± 20 days)	SYS.3	ORB	Trajectory Sim
ORB.2	The orbiter shall enter Earth's SOI on May 24, 2029	SYS.4	ORB	Trajectory Sim
ORB.3	The orbiter shall be an autonomous vehicle	SYS.5	ORB	ORB Ops Demo
ORB.3.1	The orbiter's TLCS shall communicate with Earth for at least 2 hours per day	ORB.3	ORB TLCS	ORB TLCS Analysis
ORB.4	The orbiter's wet mass should not exceed 4,116 kg	SYS.25 SYS.26	ORB	ORB Mass Meas ORB Mass Calc
ORB.4.1	The orbiter's PRPS's dry mass should not exceed 144 kg.	ORB.4	ORB PRPS	ORB PRPS Mass Calc ORB PRPS Mass Meas
ORB.4.2	The orbiter's PRPS's wet mass should not exceed 3084 kg.	ORB.4	ORB PRPS	ORB PRPS Mass Meas ORB PRPS Mass Calc
ORB.4.2.1	The orbiter's Pressurant's mass should not exceed 140 kg.	ORB.4.2	ORB Helium	ORB Pressurant Mass Meas ORB Propellant Mass Meas
ORB.4.2.2	The orbiter's Propellant's mass should not exceed 2800 kg.	ORB.4.2	ORB PRPS	ORB Propellant Mass Calc ORB Pressurant Mass Calc
ORB.4.3	The orbiter's TCS's mass should not exceed 28.8 kg.	ORB.4	ORB TCS	ORB TCS Mass Calc ORB TCS Mass Meas
ORB.4.4	The orbiter's CDS's mass should not exceed 67.2 kg.	ORB.4	ORB CDS	ORB CDS Mass Calc ORB CDS Mass Meas

ORB.4.5	The orbiter's EPS's mass should not exceed 278.4 kg.	ORB.4	ORB EPS	ORB EPS Mass Calc ORB EPS Mass Meas
ORB.4.5.1	The orbiter's Cabling's mass should not exceed 76.8 kg.	ORB.4.5	ORB Cabling	ORB Cabling Mass Calc ORB Cabling Mass Meas
ORB.4.6	The orbiter's TLCS's mass should not exceed 67.2 kg.	ORB.4	ORB TLCS	ORB TLCS Mass Meas ORB TLCS Mass Calc
ORB.4.7	The orbiter's ACS's wet mass should not exceed 96 kg.	ORB.4	ORB ACS	ORB ACS Mass Calc ORB ACS Mass Meas
ORB.4.8	The orbiter's SMS's mass should not exceed 278.4 kg.	ORB.4	ORB SMS	ORB SMS Mass Meas ORB SMS Mass Calc
ORB.4.9	The orbiter's EPS's mass without cabling should not exceed 201.6 kg.	ORB.4	ORB EPS	ORB EPS Mass Meas ORB EPS Mass Calc
ORB.5	The orbiter shall have a delta-V of 3172 m/s	SYS.18 SYS.17	ORB	ORB Prop Sim
ORB.6	The orbiter shall perform a 180° turn in five minutes or less.	SYS.16	ORB	ORB ACS Sim
ORB.6.1	The RCS thrusters should provide a minimum thrust of 0.11 N	ORB.6	ORB ACS	ORB ACS Sim
ORB.6.1.1	The RCS thrusters should provide a maximum thrust of at least 0.23 N	ORB.6.1	ORB MR-103J Thruster	ORB ACS Sim
ORB.6.1.2	The RCS thrusters should provide a minimum of 70,000 N-s of total impulse.	ORB.6.1	ORB MR-103J Thruster	ORB ACS Sim
ORB.6.1.3	The RCS thrusters shall provide a minimum of 10,000 pulses.	ORB.6.1	ORB MR-103J Thruster	ORB ACS Sim
ORB.6.2	The orbiter shall have a pointing accuracy of 0.119° or less.	ORB.6	ORB ACS	ORB ACS Sim
ORB.6.3	The reaction wheels shall store 10-12 N-m-s of angular momentum, allowing about one desaturation per orbit.	ORB.6	ORB ACS	ORB ACS Sim
ORB.6.3.1	The reaction wheels shall provide up to 0.05 N-m of torque.	ORB.6.3	ORB Reaction Wheel	ORB ACS Sim
ORB.7	The orbiter's power summation should not exceed 308.9 W.	SYS.17	ORB	ORB Power Analysis
ORB.7.1	The orbiter's power margin should be 127.4 W.	ORB.7	ORB Margin	ORB Margin Power Analysis
ORB.7.2	The orbiter's PLDS's power summation should not exceed 40 W.	ORB.7	ORB PLDS	ORB PLDS Max Power Test ORB PLDS Power Analysis
ORB.7.3	The orbiter's PRPS's power summation should not exceed 5.7 W.	ORB.7	ORB PRPS	ORB PRPS Max Power Test ORB PRPS Power Analysis
ORB.7.4	The orbiter's TCS's power summation should not exceed 46.7 W.	ORB.7	ORB TCS	ORB TCS Max Power Test ORB TCS Power Analysis
ORB.7.5	The orbiter's ACS's power summation should not exceed 15.6 W.	ORB.7	ORB ACS	ORB ACS Power Analysis ORB ACS Max Power Test
ORB.7.5.1	Each ACS thruster shall use less than 17 W	ORB.7.5	ORB MR-103J Thruster	ORB ACS Power Analysis
ORB.7.6	The orbiter's CDS's power summation should not exceed 21.2 W.	ORB.7	ORB CDS	ORB CDS Power Analysis ORB CDS Max Power Test
ORB.7.7	The orbiter's EPS's power summation should not exceed 2.8 W.	ORB.7	ORB EPS	ORB EPS Power Analysis ORB EPS Max Power Test
ORB.7.8	The orbiter's TLCS's power summation should not exceed 42.5 W.	ORB.7	ORB TLCS	ORB TLCS Power Analysis ORB TLCS Max Power Test
ORB.7.9	The orbiter's SMS's power summation should not exceed 7.1 W.	ORB.7	ORB SMS	ORB SMS Power Analysis ORB SMS Max Power Test

RVR.1	The rover shall be an autonomous vehicle	SYS.5	RVR	RVR Ops Demo
RVR.1.1	The rover's telecommunication subsystem shall communicate with the orbiter for at least eight minutes per day	RVR.1	RVR TLCS	RVR TLCS Sim
RVR.2	The rover shall operate for five months in total darkness on Mars	SYS.7	RVR	RVR TVAC Test RVR Power Config Sim
RVR.2.1	The rover's TCS shall keep the vehicle's temperature between -10 and 60 degrees Celsius.	RVR.2 RVR.3	RVR TCS	RVR Power Analysis RVR TVAC Test
RVR.3	The rover shall operate during a minimum of seven months of day/night cycles	SYS.6	RVR	RVR TVAC Test RVR Power Analysis
RVR.4	The rover mass should not exceed 169.8 kg	SYS.25 SYS.26 SYS.8	RVR	RVR Mass Meas RVR Mass Calc
RVR.4.1	The rover's PLDS's mass should not exceed 60.4 kg.	RVR.4	RVR PLDS	RVR PLDS Mass Calcs RVR PLDS Mass Meas
RVR.4.1.1	The Sample Storage Mass without samples should not exceed 21 kg	RVR.4.1	RVR Sample Cryocooler	Cryocooler Mass Calcs Cryocooler Mass Meas
RVR.4.1.2	The Sample Storage Mass with samples should not exceed 24 kg	RVR.4.1	RVR Sample Cryocooler	Test Mass with Different Config of Samples Cryocooler Mass Calcs Cryocooler Mass Meas
RVR.4.2	The rover's TCS's mass should not exceed 4.38 kg.	RVR.4	RVR TCS	RVR TCS Mass Calcs
RVR.4.3	The rover's CDS's mass should not exceed 6.57 kg.	RVR.4	RVR CDS	RVR Command and Data Handling Mass Meas RVR CDS Mass Calcs
RVR.4.4	The rover's EPS's mass should not exceed 59.11 kg.	RVR.4	RVR EPS	RVR EPS Mass Meas
RVR.4.4.1	The rover's Cabling's mass should not exceed 17.51 kg.	RVR.4.4	RVR Cabling	RVR Cabling Mass Meas RVR Cabling Mass Calc
RVR.4.5	The rover's TLCS's mass should not exceed 6.57 kg.	RVR.4	RVR TLCS	RVR TLCS Mass Meas RVR TLCS Mass Calcs
RVR.4.6	The rover's SMS's mass should not exceed 32.84 kg.	RVR.4	RVR SMS	RVR SMS Mass Calcs RVR SMS Mass Meas
RVR.4.7	The rover's EPS's mass without cabling should not exceed 41.6 kg.	RVR.4	RVR EPS	RVR EPS Mass Meas
RVR.5	The rover shall store between 2.5 kg and 3 kg of ice core samples	SYS.24	RVR	Test Mass with Different Config of Samples
RVR.5.1	The Sample Cryocooler shall store between 2.5 kg and 3 kg of ice core samples	RVR.5	RVR Sample Cryocooler	Test Mass with Different Config of Samples
RVR.6	The rover's power summation should not exceed 1350.4 W.	SYS.7	RVR	RVR Power Analysis
RVR.6.1	The rover's power margin should be 497.2 W.	RVR.6	RVR Margin	RVR Margin Power Analysis
RVR.6.2	The rover's PLDS's power summation should not exceed 300.8 W.	RVR.6	RVR PLDS	RVR PLDS Power Analysis RVR PLDS Max Power Test
RVR.6.3	The rover's TCS's power summation should not exceed 154.7 W.	RVR.6	RVR TCS	RVR TCS Max Power Test RVR TCS Power Analysis
RVR.6.4	The rover's CDS's power summation should not exceed 93.9 W.	RVR.6	RVR CDS	RVR CDS Max Power Test RVR CDS Power Analysis
RVR.6.5	The rover's EPS's power summation should not exceed 55.2 W.	RVR.6	RVR EPS	RVR EPS Max Power Test RVR EPS Power Analysis
RVR.6.6	The rover's TLCS's power summation should not exceed 127.1 W.	RVR.6	RVR TLCS	RVR TLCS Max Power Test RVR TLCS Power Analysis

RVR.6.7	The rover's SMS's power summation should not exceed 5.5 W.	RVR.6	RVR SMS	RVR SMS Max Power Test RVR SMS Power Analysis
RVR.7	The rover shall keep the ice core samples' temperature less than -5 degrees Celsius.	SYS.14	RVR	Cryocooler TVAC Test Cryocooler Thermal Analysis
RVR.7.1	The Sample Cryocooler shall keep the ice core temperature less than -5 degrees Celsius.	RVR.7	RVR Sample Cryocooler	Cryocooler Thermal Analysis Cryocooler TVAC Test
RVR.7.2	The Sample Cryocooler shall be powered through at least 99% of the mission once samples are inserted.	RVR.7	RVR MAV SRC	Vehicle Safe Mode Test Power Usage Analysis
RVR.8	The rover shall drill ice cores with a diameter between 25mm and 50mm	SYS.12 SYS.8	RVR	Inspect Ice From Demo Drilling Ops Demo Inspect Drill Dimensions
RVR.8.1	The drill shall drill ice cores with a diameter between 25 mm and 50 mm	RVR.8	RVR Drill	Drilling Ops Demo Inspect Ice From Demo Inspect Drill Dimensions
RVR.9	The rover shall drill ice cores with a length between 100mm and 150mm	SYS.13 SYS.8	RVR	Drilling Ops Demo Inspect Ice From Demo Inspect Drill Dimensions
RVR.9.1	The drill shall drill ice cores with a length between 100 mm and 150 mm	RVR.9	RVR Drill	Inspect Ice From Demo Drilling Ops Demo Inspect Drill Dimensions
SRC.1	The Earth Return Capsule shall land on Earth on May 24, 2029.	SYS.4	SRC	Trajectory Sim
SRC.2	The SRC shall be an autonomous vehicle	SYS.5	SRC	SRC Ops Demo
SRC.3	The SRC mass shall not exceed 211 kg	SYS.25 SYS.19 SYS.26	SRC	SRC Mass Meas SRC Mass Calcs
SRC.4	The SRC's touchdown velocity shall not exceed nine m/s	SYS.19	SRC	Earth Entry Sim
SRC.5	The SRC shall use a parachute with a diameter of six meters	SYS.19	SRC	Parachute Inspection Parachute Analysis