Preliminary Design Review

Team 16: L'Space Invaders

July 30, 2020

Contents

1 Introduction and Summary

1.1 Team Introduction

Team 16, otherwise named L'Space Invaders, consists of twelve individuals who worked with passion and dedication to develop this mission concept. Their personal biographies can be seen below.

Christian Acosta has completed an undergraduate degree at Illinois Institute of Technology in Chicago, Illinois. He brings experiences to the team such as the coding languages Python and MATLAB. Also, he has a higher level understanding in mathematics and astrophysics.

Kejsi Bishaj is an aerospace engineering student at Illinois Institute of Technology in Chicago, Illinois. She is the Safety Officer for the team. She is ardent about science and engineering, and she has had anterior experience with teamwork and programming. In her leisure time, she enjoys studying aerodynamics and doing sports.

Angirasa Darbha is a student at the Illinois Institute of Technology, Chicago. He is an aerospace engineering and applied mathematics major. He brings to the table his proficiency in Computer-Aided Design and programming.

Riley Ellis is majoring in mechanical engineering and is minoring in computer science while attending Illinois Institute of Technology in the city of Chicago, Illinois. He is the Deputy Project Manager of the team. Riley has experience in Python, Java, and Inventor and brings his open mind to the team.

Michael Gromski is a dual-enrollment student at Lewis University in Romeoville and Illinois Institute of Technology in Chicago studying physics and aerospace engineering. Michael brings years of research experience from material science to aerospace technology. He has presented research titled "Development of Long Endurance Solar Unmanned Aerial Vehicle" at industrial conferences, and "Using Self-Assembled Nanophotonic Arrays to Enhance Photodetectors" at university wide conferences. Michael's experience with working in groups on research, and passion for aerospace science and technology, were beneficial to this team.

Roberto Hernandez Jr studies at Illinois Institute of Technology in Chicago, Illinois. He largely partakes in the academy for the learning opportunity and to gain beneficial exposure to software and experience in working on team-led projects.

Patrick Kozyra studies at Elgin Community College in Elgin, Illinois. He has graduated with his associate in science degree in May 2020. Patrick has participated in previous NASA programs such as the NASA Community College Aerospace Scholars Program, detailing how NASA could expand its use of Microbial Monitoring and Medical and Diagnostic Equipment in future NASA missions. Patrick is assigned to be the Science Team Leader for Team 16.

Adrian Manrique is a dual-degree student obtaining his bachelor's degrees in mechanical engineering and physics from Illinois Institute of Technology and Lewis University. Adrian brings to the team his CAD background, experience in a professional research setting at Argonne National Laboratory, and dedication.

Anusha Mody is enrolled in a dual-degree program. She is earning bachelor's degrees in aerospace engineering from Illinois Institute of Technology in Chicago, Illinois and mathematics from Dominican University in River Forest, Illinois. She is the Project Manager of the team. She benefits the team with her extensive experience in team leadership, public speaking, and written communication. Anusha also entered the project with developed skills in CAD, programming, research, and professional experience in data analysis.

Yogi Patel graduated with a bachelor's and master's degree in aerospace engineering from Illinois Institute of Technology, Chicago in May 2020. He is the Lead Engineer of the team. He was a board member of the ASME organization on campus, and he served as a member of ASME National Student Advisory Panel. Being an Engineering team lead, Yogi brings his strong technical background and his skills in research, organization, planning, decision-making, communication, problem-solving, and leadership.

David Wardein studies at Illinois Institute of Technology in Chicago, Illinois. He is a mechanical engineering major with experience in drafting software, mainly Autodesk Inventor. He has previously earned a bachelor's degree in psychology and has past experience working in teams in the insurance industry.

Will Yarbrough currently attends University of Arkansas at Little Rock where he is majoring in computer science. He is the Lead Administrator of the team. Before pursuing computer science, Will earned an associate's of science in business from University of Arkansas Pulaski Technical College. While at University of Arkansas Pulaski Technical College, he was accepted into the NASA Community College Aerospace Scholars Program (NCAS) and was invited to the On-Site Experience. NCAS is what inspired him to change majors from business to computer science. As a freshman, Will enlisted in the Army as a Combat Engineer and served in the National Guard. In the military Will learned the fundamentals of leadership and team cooperation. Will currently has worked in the finance sector for the past several years and holds a position processing loans. He specializes in finance, leadership, budgeting, and working in a team environment.

1.2 Mission Overview

1.2.1 Mission Statement

The goal of the mission is to gain insight into or indirectly detect the presence of microbial life on Mars, whether past or present, by analyzing the geology of Jezero Crater for life-relevant minerals and compounds. A mobile rover, named Fortitude, will be used to analyze soil samples containing evidence of biosignature preservation from Jezero Crater, compare various samples, and analyze them with infrared (IR) spectrometry. This mission will improve the understanding of life beyond Earth by determining whether life is possible on Mars or discovering signs that there was life previously. It will determine if Jezero has a variety of rocks and soils, including those from an ancient time when Mars could have supported life and if the rock types and molecules at the site are able to preserve physical, chemical, mineral, or molecular signs of past life. The mission expectation is to develop humankind's understanding of exobiology and the evolution of possible life on Mars.

1.2.2 Mission Constraints and Requirements

The mission must be completed without exceeding 180 kg in mass, 415,776 cm^3 (61cm x 71cm x 96 cm) in volume, or the cost of $100 million. The Entry, Descent, and Lander system is allowed an additional 72 kg (40% of the original mass limit), and the budget only refers to the payload. The landing site was to be chosen from the four finalists of the Mars 2020 Rover landing site, which included Columbia Hills, Jezero Crater, Midway, and NE Syrtis. The mission requires the spectral resolution for rock detection to be 4-12 nm, spatial resolution greater than 2 mm, and land scanning span of 2-3 m.

1.2.3 Mission Success Criteria

The minimum expectations for a successful mission are:

- Analyze two soil samples, one selected from an area that might have preserved signs of life within Jezero Crater (such as near the river delta), and one from outside the river delta or an area with a smaller likelihood of preserving signs of microbial life.
- Receive data on Earth from the rover.
- Compare the two samples from Jezero Crater on Earth by analyzing the retrieved data from the samples.
- Take at least 1 picture with NavCam of samples, geological features, or landmarks.
- Analyze 1 sample with IR spectroscopy.
- Travel into the crater without complication.
- Confirm minerals such as carbonates and perchlorates.

Through this mission, the following objectives are expected to be resolved:

- Objective 1: Does the area have a variety of rocks and "soils", including those from an ancient time when Mars could have supported life?
- Objective 2: Are the rock types at the site able to preserve physical, chemical, mineral, or molecular signs of past life?

The focus for Objectives 1 and 2 is to find and study rock containing carbonate because they have higher potential for preserving microorganisms.

1.2.4 Concept of Operations

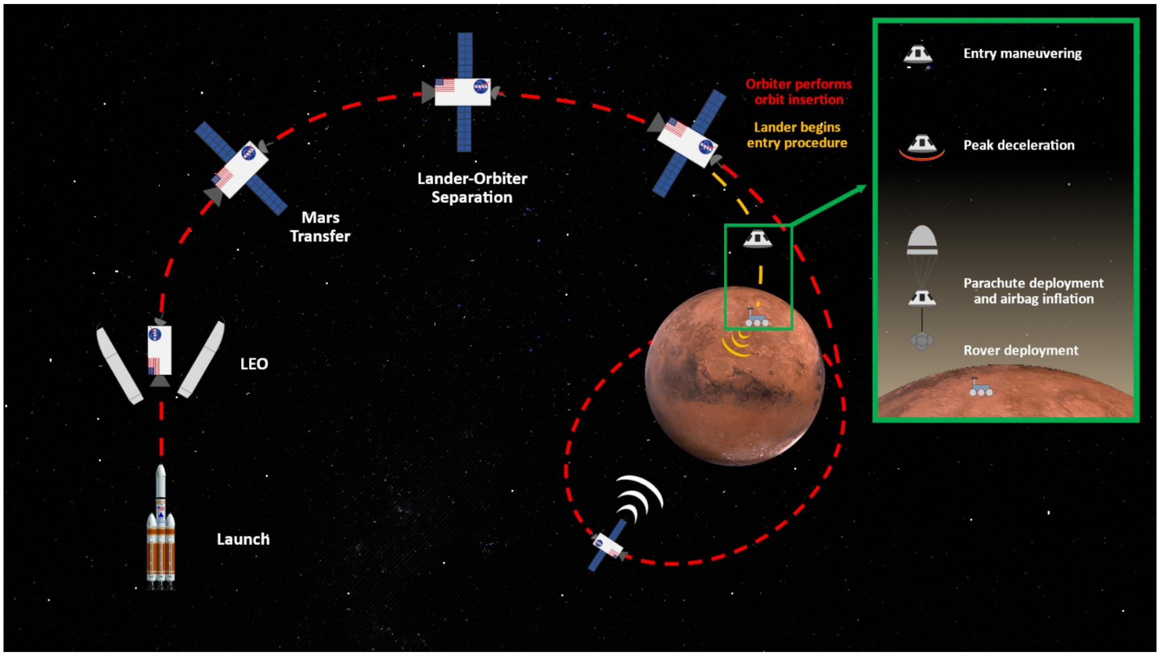


Figure 1: Sequence of operations involved in the mission from Launch to Land.

The mission starts when the rocket launches carrying the primary payload and the rover. After the launch, the upper stage will enter a circularized Low-Earth-Orbit (LEO). Once the interplanetary stage reaches the Mars transfer window, the engines will ignite, sending the payload into a transfer trajectory to Mars. When the interplanetary stage reaches Mars, the lander will detach from the primary payload and begin the descent to the surface. The interplanetary stage will enter an orbit around Mars to serve as a telecommunication relay between the rover and Earth. The lander will fire its engines to enter Mars atmosphere at the proper speed and angle. Shortly after, the lander will enter the outer martian atmosphere, experiencing maximum deceleration and heating. When the lander is close to the surface, the parachute and impact stage will deploy. Once the impact stage lands on the surface of Mars, the rover will deploy and start its mission.

1.2.5 Major Milestones Schedule

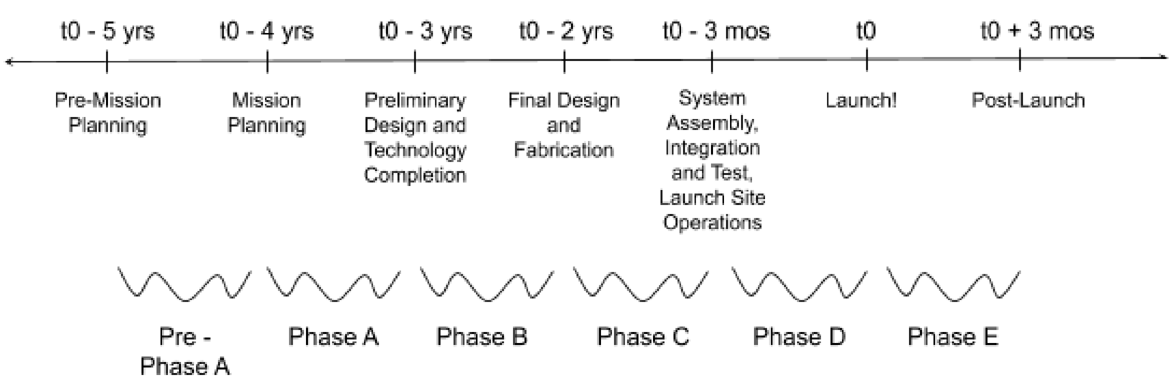


Figure 2: Timeline of everything from Pre-Mission Planning to Post-Launch analysis.

A brief explanation of each phase shown in the Major Milestones Schedule is included below:

- Pre-Phase A includes a suggested proposal to NASA and a definition of the mission and scientific opportunities, goals, mission requirements, and system concepts. During this phase, the scientific quality of the proposal is assessed.

- Phase A consists of the preliminary design and project plan. This includes the build plan, launch date, spacecraft course, process during flight, mission timeline, system test location, delegating mission operations, determining the ground data system capabilities required, and scientific experiments. The experimenters are also decided.
- Phase B reviews and solidifies the systems requirements, schedules, and system designs. Additionally, experimenters modify and develop scientific ideas through the development of other experiments. Personnel teams are created to develop scientific instruments and analyze the returned data, and team leaders are established.
- Phase C includes the complete detailed design and realization of the systems. The systems are produced and software is prepared for integration. Studies continue through this phase to validate the design. Phase C also consists of the Preliminary Design Review, Critical Design Review, and safety considerations.
- Phase D is the process of integrating components and performing validation and verification tests. During Phase D, the team begins to prepare for launch by checking operations, data processing, the system and supporting elements, and developing a Test Readiness Review. The safety considerations of the mission are also updated through this phase.
- Phase E is the implementation of the mission operations plan. The launch vehicle's performance is assessed, science and engineering data is collected and processed, and the team prepares for deactivation and disassembly if applicable. Any failures or problems are addressed, and a final mission report is developed.

1.3 Descent and Lander Summary

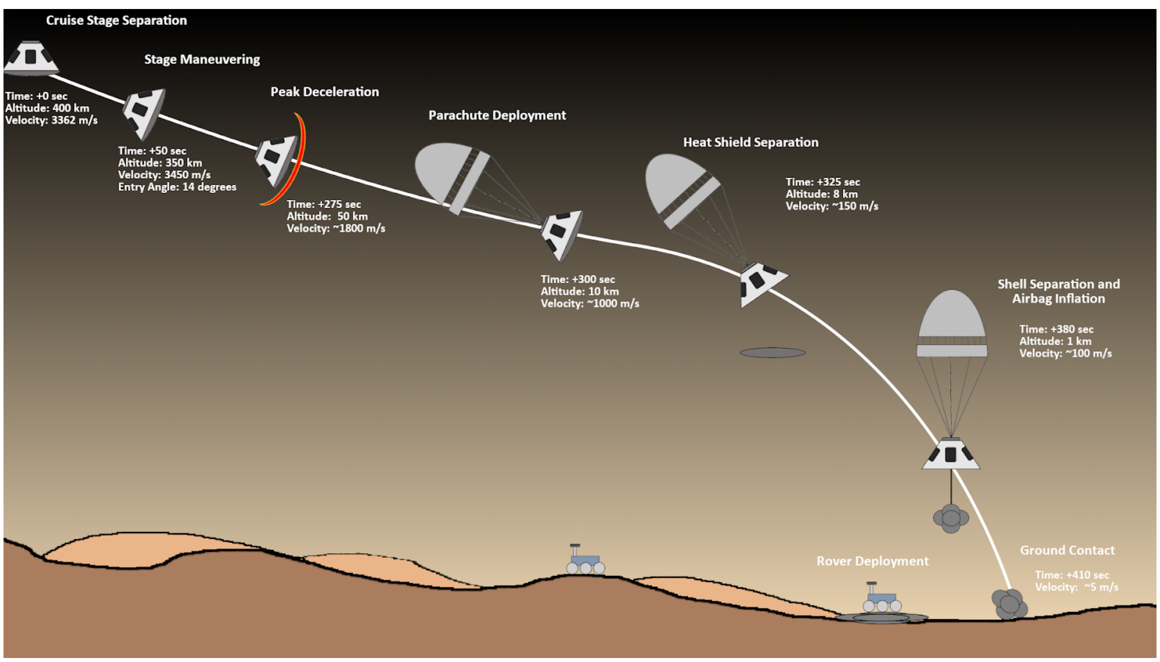


Figure 3: Graphic describing Entry, Descent, and Landing strategy.

The rover will be contained within a two stage lander, including a tetrahedral protective airbag impact stage within the shell used during the entry stage. The lander will initially have a mass of 180 kg. The lander will separate from the primary payload at 400 km above ground level (AGL) and conduct an entry burn. This will provide a trajectory for the lander to enter the martian atmosphere at an angle of 14 degrees, a velocity of 3450 m/s, and an altitude of 350 km AGL. An ablative heat shield will decelerate the lander as it proceeds through the initial entry. Once the lander reaches an altitude of 10 km AGL, a parachute will be deployed to decelerate the descent of the lander to 150 m/s. At an altitude of 8 km AGL, the heat shield will separate from the bottom of the lander, exposing the airbag impact stage. At approximately 1 km AGL, a cable will suspend the impact stage below the entry stage, and the airbags will inflate. The Rocket Assisted Descent (RAD) motors will fire to slow the lander to 5 m/s at 10-15 m AGL. Then, the impact stage will be released from the entry stage, and the airbags will cushion the lander as it reaches the surface. The lander stage will fly away to a safe distance, greater than 500 m from the impact stage. Once the lander comes to a complete stop, the airbags will deflate, and the walls of the lander will fold open to expose the rover. Fortitude will start its mission as it drives off the lander.

1.4 Payload and Science Summary

The following instruments will be used to satisfy the mission's science deliverables:

Acousto-optic Tunable Filter InfraRed (AOTF IR) Spectrometer ($15 million)

The AOTF IR spectrometer satisfies both objectives with its ability to easily identify carbonates and other minerals in the martian soil (Wiens et al.). The IR spectrometer will be used to discover mineral compounds and molecules that may have supported past life on Mars (J. -M. Reess et al.). The IR spectrometer has a volume of 10.4 cm x 6 cm x 4 cm, for a total volume of 249.6 cm^3, and a mass of 330 g (Bain).

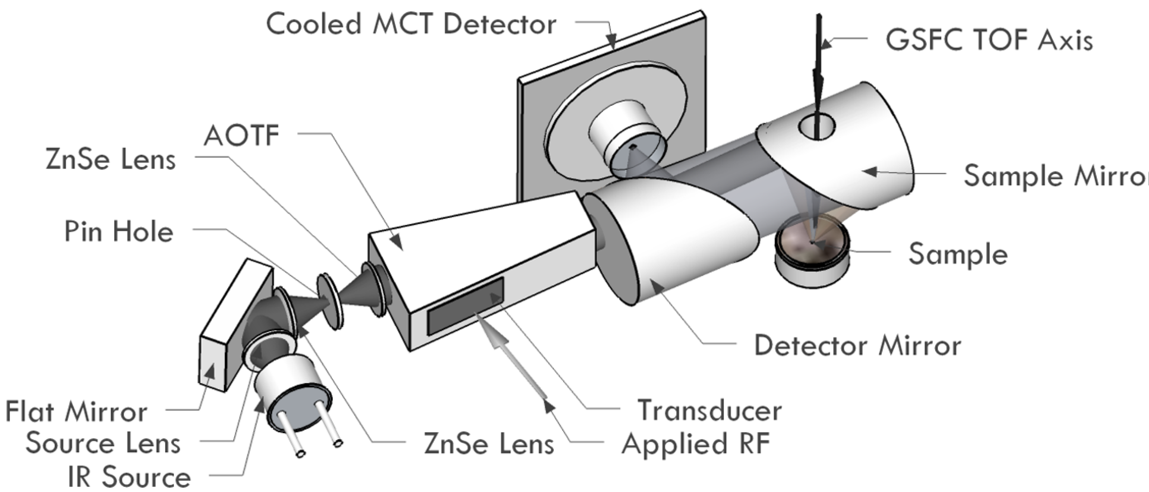


Figure 4: AOTF system

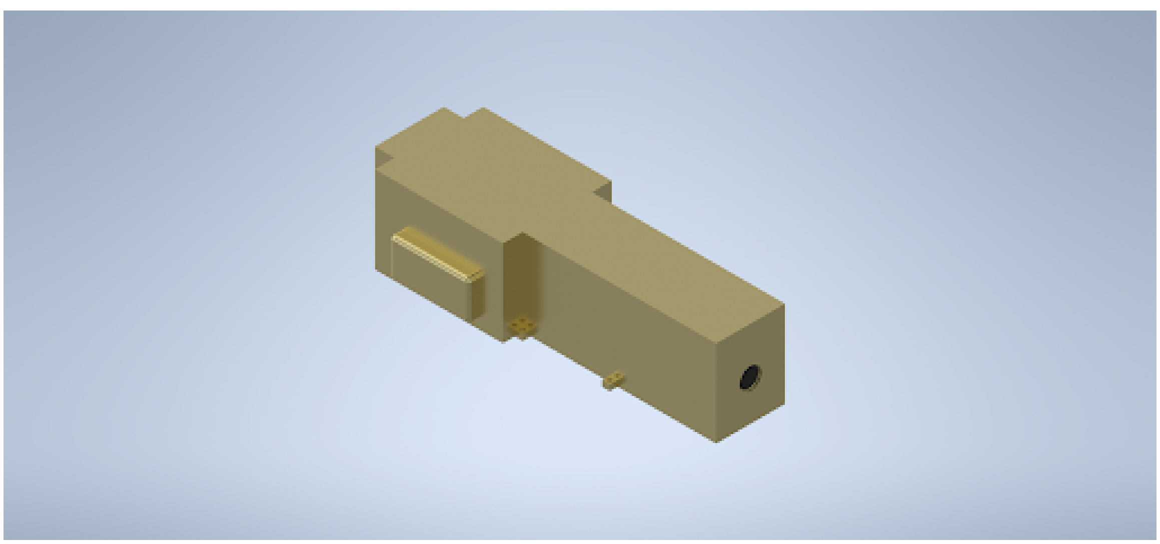


Figure 5: 3D model of the AOTF IR spectrometer

Silicon Rubber Thermofoil Heater ($0.1 million)

The thermofoil heater will keep the science instruments, stored in the payload, warm enough to operate in the martian environment using heat from electrical power. These would be wrapped around the rover, instruments, and components to provide them with heat and will efficiently control the rover's temperature. Fifty grams of heater uses 11.83 W to increase the temperature of the rover by 20°C in 60 seconds, and the heater would cover an area of 1,200 cm^2 ("Flexible Heaters Design Guide").

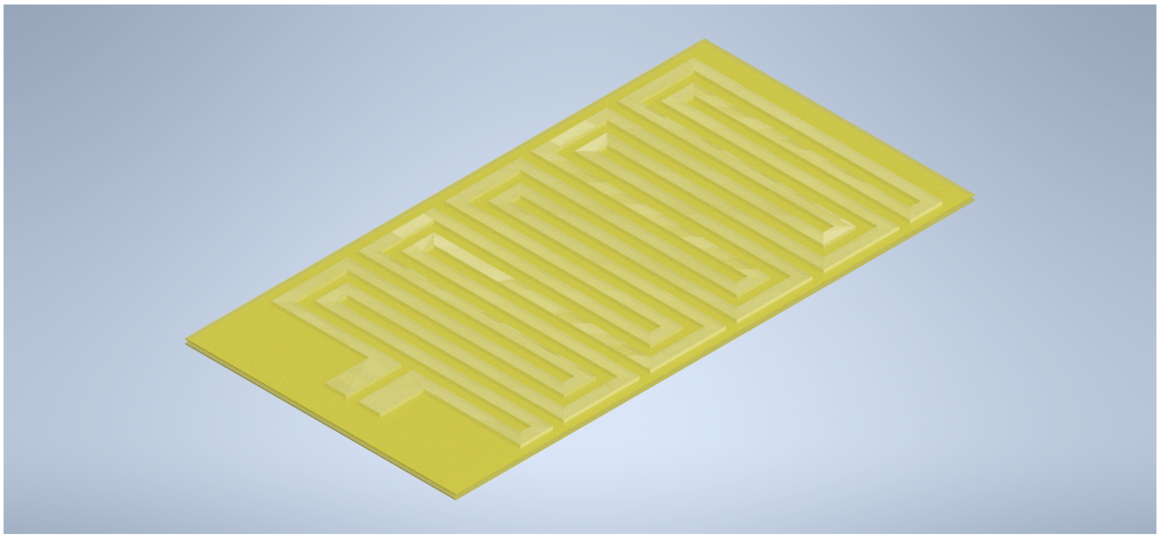


Figure 6: 3D model of the silicon rubber thermofoil heater

NavCam ($10 million)

The NavCam will help the rover take pictures of samples, geological features, and landscapes of interest, as well as help the rover navigate Jezero's precarious terrain. The NavCam will be mounted below the IR spectrometer. It consists of an electronics box about 6.7 cm x 6.9 cm x 3.4 cm, a detector head of 4.1 cm x 5.1 cm x 1.5 cm, and a 3.5 W heater (Maki, J. N. et al.). The NavCam weighs about 220 g, uses 2.15 W on its own, 5.65 W total with the heater on, and operates at a temperature range between -55°C and 5°C (Maki, J. N. et al.).

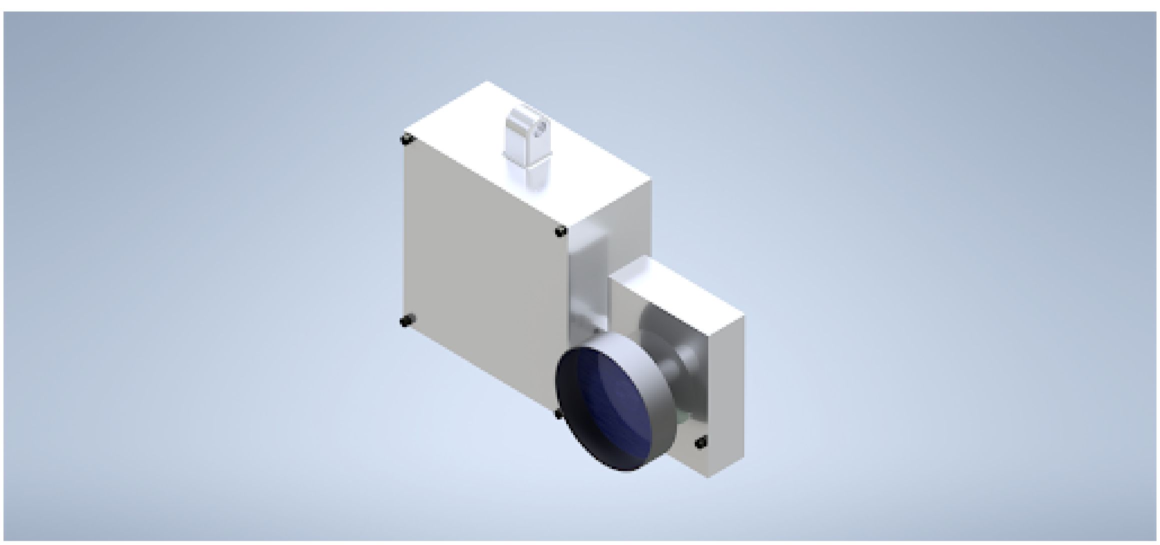


Figure 7: 3D model of the NavCam

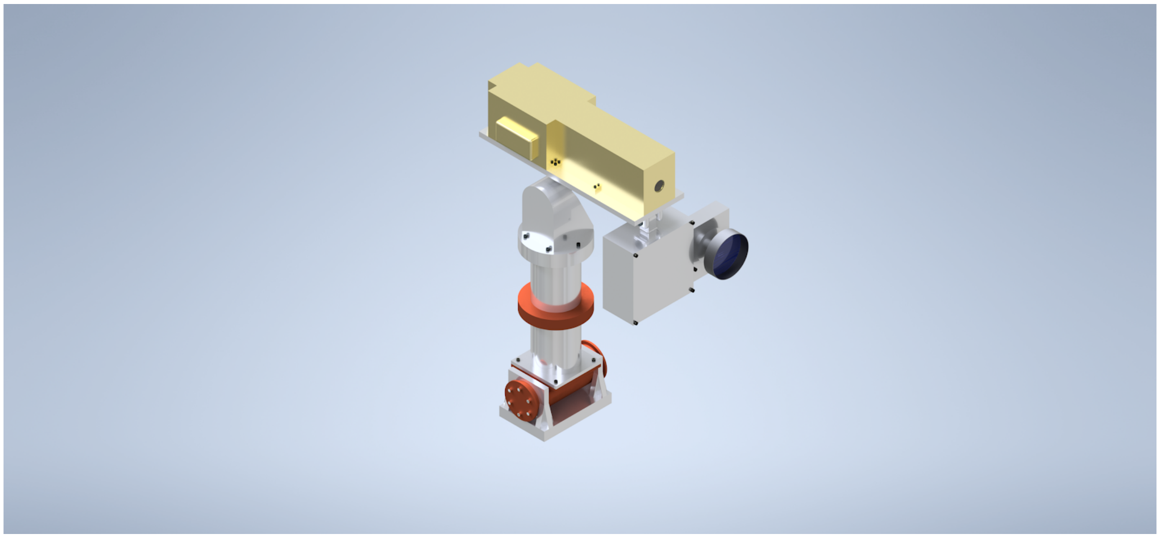


Figure 8: 3D model of the mast

Helios Quadrifilar Helical Deployable Antenna ($0.011 million)

The Helios Quadrifilar Deployable Antenna is a helical antenna used to satisfy the communication requirements and send data from the rover to the orbiter and Earth. It is space rated by the United States Air Force and has had flight heritage since 2018 ("Helios Deployable Antenna"). The antenna has a length of 100 mm, a width of 100 mm, a stored height of 35 mm with a deployed height of 330 mm, and the antenna uses 1 W of power to operate ("Helios Deployable Antenna").

Once the payload safely lands on Mars, the antenna will receive a deployment command from one of the RAD750 processors on board. During this command, the control system will tell the power system to allocate eight volts of direct current at seven amps to the antenna. This is the power required by the communications system to deploy the helical antenna. During the deployment, the antenna's height will rise from 35 mm to 330 mm. This power will be supplied for the entire duration of deployment spanning a rough time of 60 to 90 seconds. Once the antenna has been deployed, the power command will cease ("Helios Deployable Antenna"). Below is the CAD model for the antenna in its stored position.

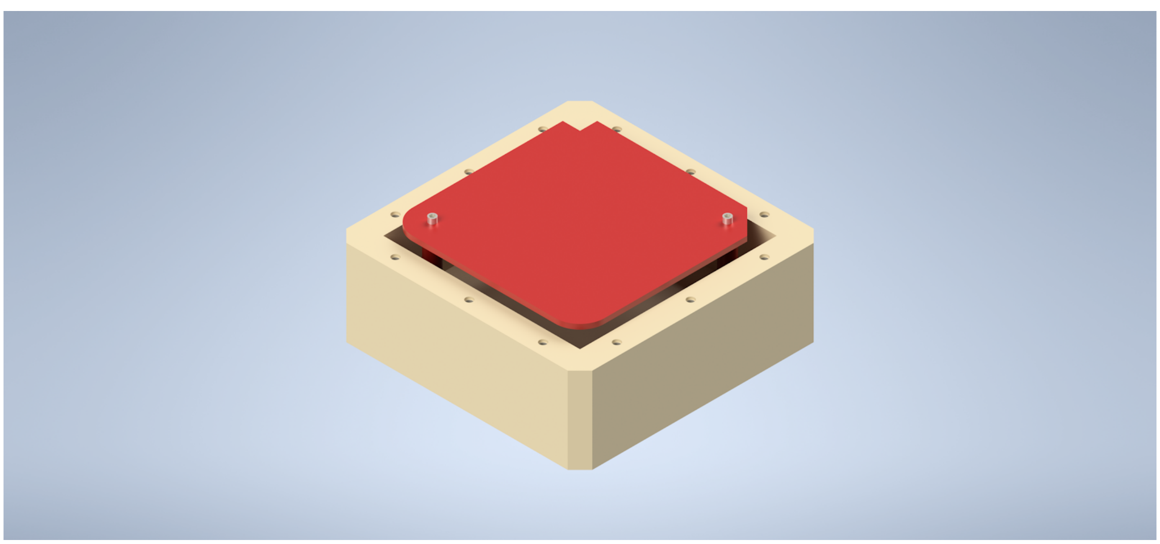


Figure 9: 3D model of the antenna

2 Evolution of Project

2.1 Evolution of Descent and Lander

2.1.1 Descent

Initially, the given conditions for start of the mission were 2.38 km/s initial velocity at an altitude of 400 km. It was assumed that the craft was already in a stable, circular orbit and heading in the direction of the target landing site. It was not discovered until later in the design process that the initial velocity was not appropriate for the given release altitude. The velocity of a spacecraft in a circular orbit around a body of mass is defined by the equation:

$$V = \sqrt{\frac{G \times M_{\mathrm{m}}}{R_{\mathrm{m}} + R_{\mathrm{alt}}}},$$

where V is the velocity of the spacecraft, G is the gravitational constant ($6.673e^{-11}$ Nm^2kg^{-2}), M_m is the mass of Mars ($6.39e^{23}$ kg), R_m is the radius of Mars ($3.3895e^9$ m), and R_{alt} is the altitude of the spacecraft above the surface of Mars. Using this formula and assuming that the mission starts at 400 km altitude, the velocity of the spacecraft will have to be 3,362 m/s. This error required the trajectory descent profile to change halfway through the design process.

Before the initial trajectory velocity was corrected, the planned descent profile was primarily determined by a parabolic entry without an entry burn. The entry burn was deemed unnecessary as the lander would already be entering the martian atmosphere. A computer program was written to aid in deciding the best descent profile. The program used equations of motion (EOM) of high altitude reentry, derived by NASA for the Apollo space program. The EOM were originally derived for Earth's atmosphere, but were deemed acceptable for the martian atmosphere as the conditions are similar.

After the trajectory correction, it was clear that an entry burn would be required to enter the atmosphere. It was also made clear that a proper descent profile had to be chosen. If the spacecraft came in at an overly shallow entry angle, the spacecraft runs the risk of "bouncing" off of the atmosphere and overshooting the landing site. There is also a risk of entering at too steep of an angle, in which case the spacecraft would overheat from the atmosphere and would decelerate too quickly, damaging critical mission equipment. After analyzing the descent profiles of previous Mars missions and running simulations with the written program, an entry angle of 14 degrees, a speed of approximately 3450 m/s, and an altitude of 350 km was chosen as an acceptable descent profile. As a result, the spacecraft and its payload were designed with an expected maximum G loading of 8 g and a maximum temperature of 2000°C.

2.1.2 Lander

The design of the lander was inspired by previous successful Mars missions like the Spirit, Opportunity, and Curiosity rovers. The initial design of the lander was a conical-shape with a blunt base and a tapered top. The top of the lander would house a parachute assembly that would deploy shortly after maximum heating to stabilize and decelerate the lander. The base would have a heat shield that would protect the lander from the immense heat of atmospheric entry. After the lander has experienced maximum aerothermodynamic heating, the lander would jettison the heat shield and expose the bottom of the lander. Much like the Spirit and Opportunity missions, the lander assembly contained a smaller impact assembly that would house the rover. The impact assembly would be a protective cage that encases the rover, and would release the rover when it safely lands.

During descent, the impact stage would be lowered below the lander stage with a cable. This is done to prevent the entry stage from colliding with the rover during impact. The impact stage contains airbags on the outside that would inflate a few hundred meters above the ground. This would protect the impact stage from the initial shock of hitting the terrain and allow the rover to decelerate any remaining velocity. To reduce the impact velocity of the impact stage, rocket engines will fire just before ground contact. Once the rover is completely still, the airbags would deflate, and the impact stage would fold open like a flower bud and release Fortitude. All of the design iterations were based off of this staging procedure.

Rejected Lander Design #1

The first design of the lander takes the primary shape of the landers included with the Spirit, Opportunity, and Curiosity rovers. The conical-shape is stable during reentry, the bottom has a large blunt surface to allow most of the heating during reentry to be dissipated into the atmosphere, and it is a legacy design that has proven to work reliably.

The impact stage inside the entry stage is shaped as a pyramid as it provides a good balance of shared volume between the impact stage and landing equipment. The pyramid design is different from the tetrahedral design of a legacy mission like Spirit. This was done to maximize the volume allocated for the rover. The parachute cap on top of the capsule is designed to protect the parachute deployment system (PDS) and comes off shortly before parachute deployment to reduce mass. The rocket assisted deceleration (RAD) motors sit just below the PDS and above the impact stage. The RAD engines would fire just before ground contact to help decelerate the payload. The heat shield is on the bottom of the capsule and is designed to eject shortly after parachute deployment for further mass reduction and to expose the impact stage. One difficulty faced with this initial design was that there was very little volume left for the payload. The design was largely inefficient with the volume limitations and it was difficult to find any scientific instruments that would properly fit inside the remaining volume.

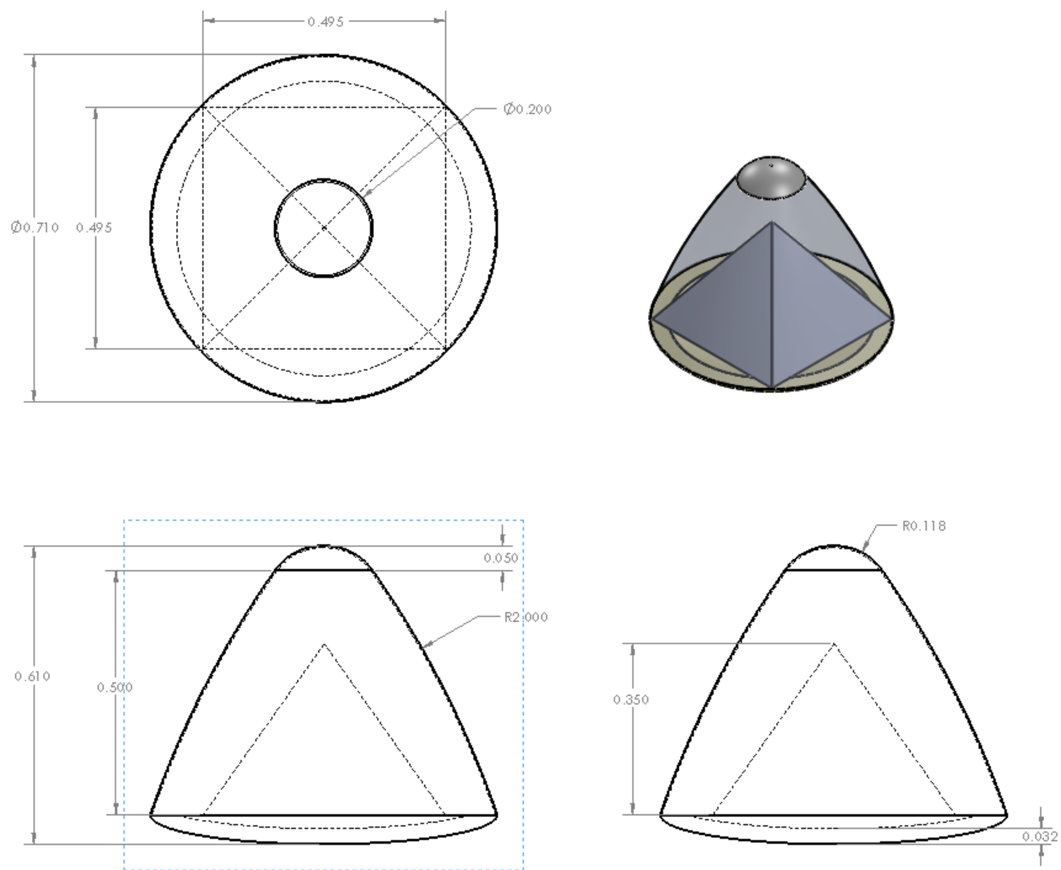


Figure 10: First lander design

Rejected Lander Design #2

The second iteration of the design was focused on providing more volume for both the payload and lander equipment. The base has a smaller cross-sectional area compared to the initial design, changing from 0.40 m^2 to 0.29 m^2. This was a trade-off to allow a longer entry stage and provide more volume for the stage subsystems. The cylindrical portion at the top of the lander was created to house the PDS. Having the PDS located further above the rest of the subsystems avoided potential conflict with the rocket engine subsystem as the top of the entry stage became very narrow. As with the previous design, the PDS features a cap to protect the subsystem during entry and would jettison shortly before parachute deployment. The impact stage retains the same pyramid shape as before, but the base of the pyramid is smaller, changing from 0.495 m to 0.426 m square side length. The base of the impact stage had to be adjusted to accommodate the smaller cross-sectional area of the lander. The height of the impact stage was elongated from 0.35 m to 0.475 m to provide more volume for the rover as the base was made smaller. The heat shield is the same system as before but was made smaller to accommodate the smaller cross-sectional area.

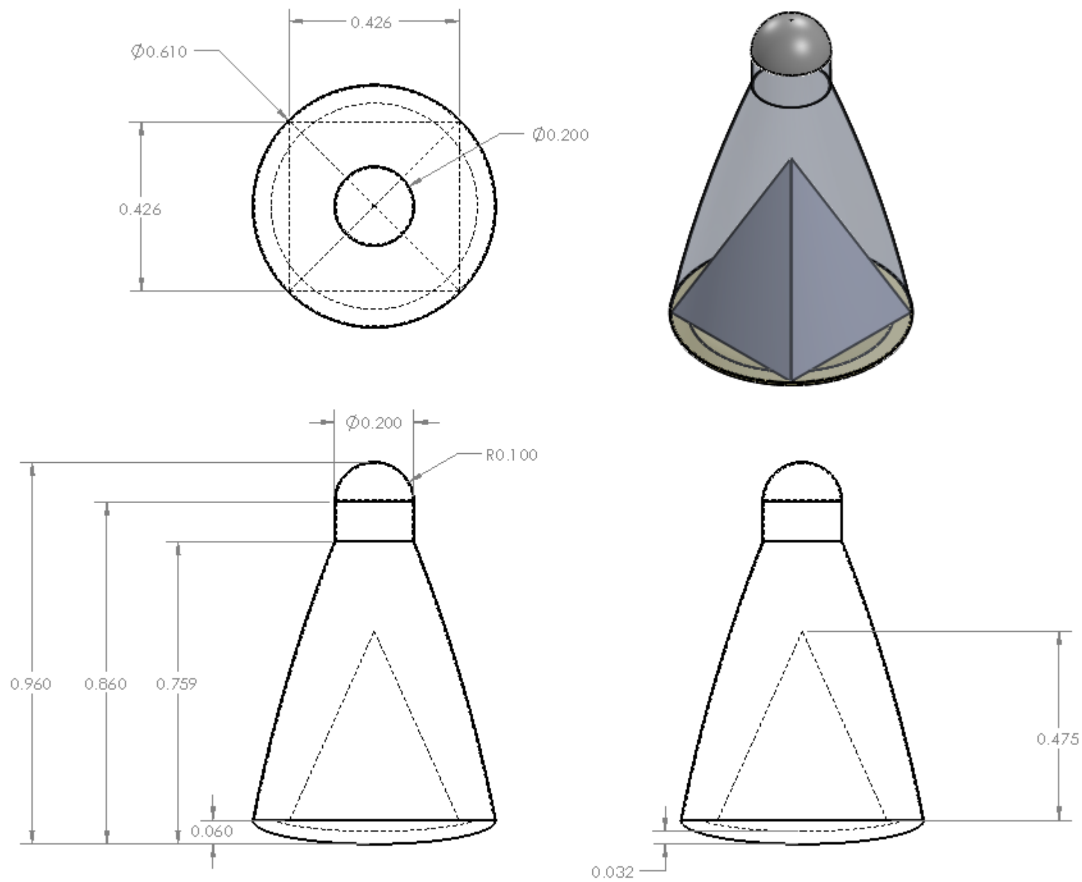


Figure 11: Second lander design

With the second design iteration, there were many trade-offs to increase the volume of the lander. The first trade off was sufficient volume for the payload inside the impact stage. With a much more narrow base and a taller design, it would be difficult to create a rover design to use the volume. With a less symmetrical design of the impact stage, there were also concerns of the impact stage rolling on the surface of Mars with the airbags deployed. A less symmetrical design, with the pointed edge, poses a threat to the payload when rolling on the surface. How the craft rolls on the ground is harder to predict with a non-symmetrical shape, and the protruding edge could puncture through the airbags and hit the rocky terrain. This would lead to a larger than expected shock to the payload and potentially damage the instruments. The next trade-off with this design would be how the spacecraft enters the martian atmosphere. With the narrower cross-sectional area and a more aerodynamic shape of the lander, the ballistic coefficient increases substantially. The ballistic coefficient is a measure of how a body can overcome air resistance in flight; this is also correlated with the maximum heating endured during entry. A higher ballistic coefficient means higher maximum heating necessary for the spacecraft to endure.

Rejected Lander Design #3

The third design iteration focused on decreasing the ballistic coefficient of the lander and increasing the amount of useful payload volume. The cross-sectional shape was changed from circular to elliptical, the top of the lander was widened to match the lower cross-sectional area of the lander, and the impact stage shape was changed from a pyramid to a rectangular prism. The bottom of the lander was stretched to fit with the maximum volume constraints to provide the most volume for the payload and to increase the cross-section area from 0.29 m^2 to 0.34 m^2. Widening the top of the lander stage provided more volume for the lander subsystems and increased the coefficient of drag of the vehicle. Increasing the coefficient of drag lowers the ballistic coefficient of the lander and lowers the peak heating of the craft. The impact stage was changed from a pyramid to a rectangular prism to optimize the volume allocated for the rover. It also provided a symmetrical profile when the airbags are inflated and the stage is on the surface of Mars. The PDS still has a cap that would eject before parachute deployment, but instead of the hemisphere shape, the cap is a flat circle that would eject during parachute deployment.

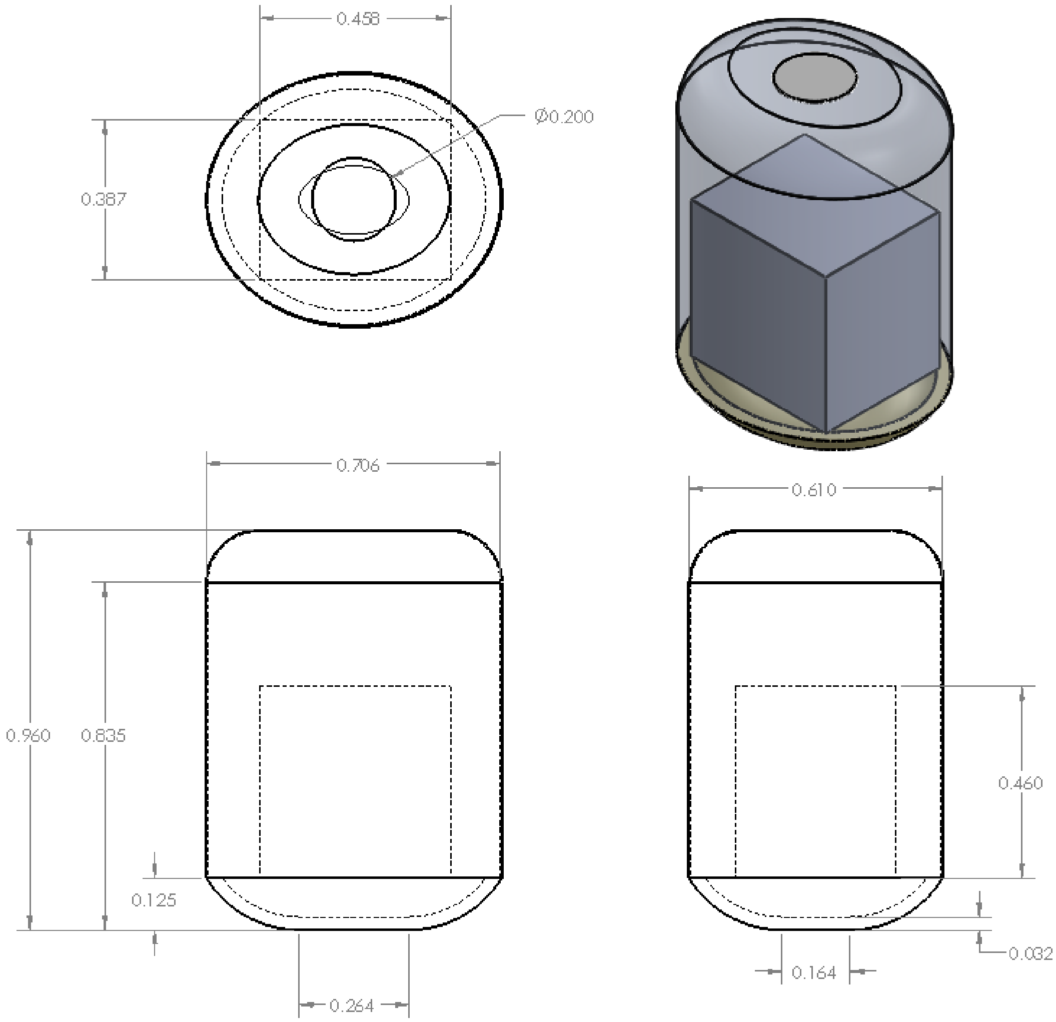


Figure 12: Third lander design

The main concerns with the third design iteration is stability of the lander and atmospheric heating on the sides as it enters the atmosphere. With an unconventional lander design, it is unknown how the craft will behave. The primary concern with the spacecraft is tumbling during entry. There will need to be plenty of testing to be conducted to ensure that the aerodynamic behavior is well-known. The heat shield is relatively small compared to the rest of the lander, and the side walls are prone to exposure to the airstream. Exposure to the airstream would lead to the walls melting and the craft breaking up during entry.

Accepted Lander Design #4

The fourth and final design iteration addresses the concerns presented in the previous iterations. The lander is oriented to the side, providing the maximum cross-sectional area, fitting inside the volume limitations, and expanding the area from 0.34 m^2 to 0.65 m^2. This large increase in cross-sectional area lowered the ballistic coefficient of the spacecraft even further, reducing the maximum temperature the spacecraft would endure. The size of the heat shield was increased to prevent the sides of the lander from being exposed to the high temperatures. The impact stage was increased to allow more volume for the payload and to account for all of the subsystems that would need to be contained inside the impact stage. The PDS is located to the side of the lander, instead of being in the center of the craft. This was done because the PDS system required a cylindrical tube construction, and placing a cylinder in the center of the craft would constrict the volume for the payload. To get around this, the PDS was made into an elliptical shape and placed to the side of the lander stage. To have the parachute line still connected above the center of mass and to allow the lander to descend in the horizontal orientation, the outer surface of the lander shell has a groove to allow the parachute line to be stowed away before deployment. The stability of the lander is still widely unknown due to the unconventional shape of an elliptical cylinder. To address stability concerns, the lander is designed to have a center of mass lower than the center of pressure, and to further address any stability concerns, a cold gas reaction control system was introduced.

Overall, the descent and lander stage went through multiple design iterations before settling on the final design. The final descent profile was chosen to prevent the lander from skipping off the atmosphere and missing the landing site or entering at too steep of an angle and exceeding the heating and load limits of the lander. The lander design was optimized to reduce the heating load during reentry, efficiently use the volume given the volume criteria, and provide the most volume for the scientific payload.

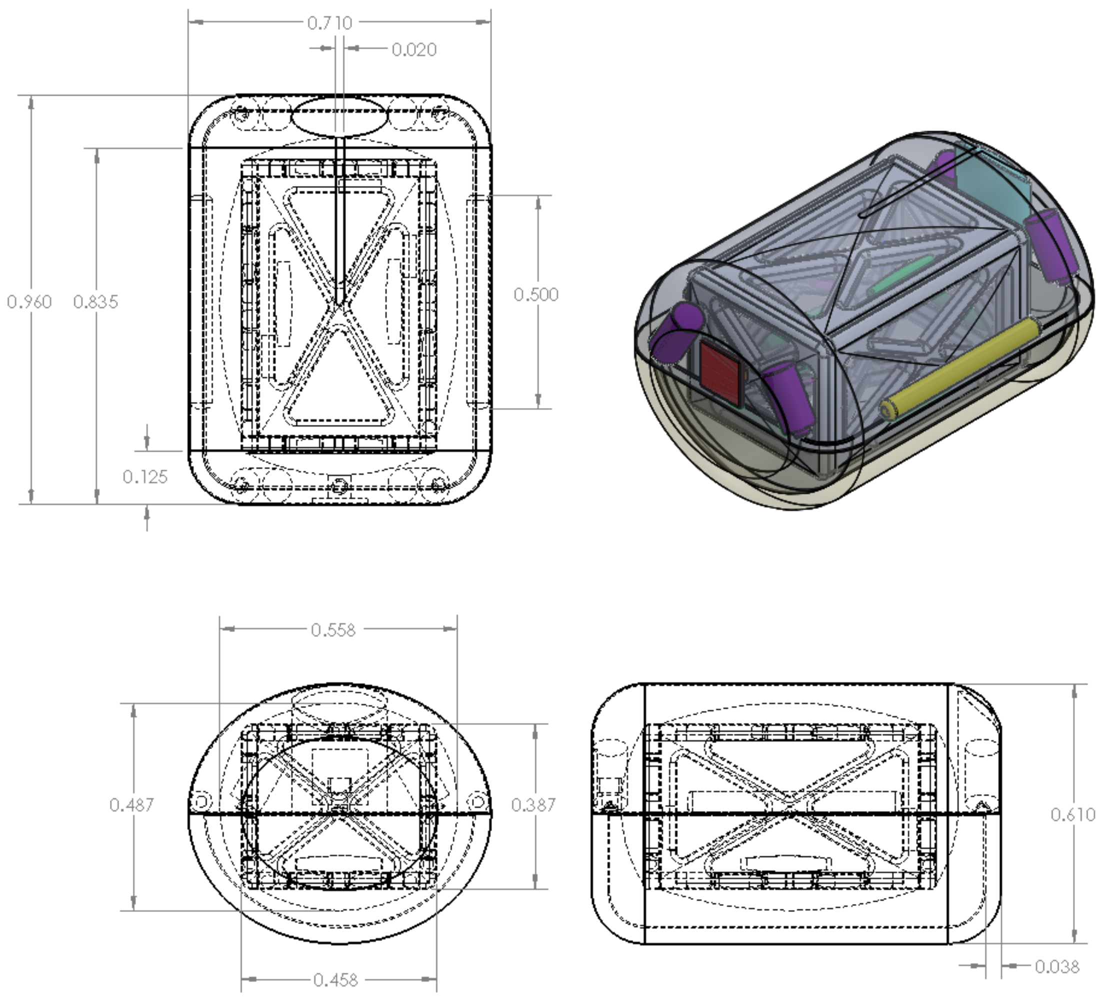


Figure 13: Final lander design

2.2 Evolution of Payload

Rejected Payload Design #1

The first payload design included Scanning Habitable Environments with Raman & Luminescence for Organics & Chemicals (SHERLOC), the Energy Dispersive X-Ray Spectrometer (EDS), and the Multi-Hundred-Watt Radioisotope Thermal Generator (MHW-RTG).

SHERLOC

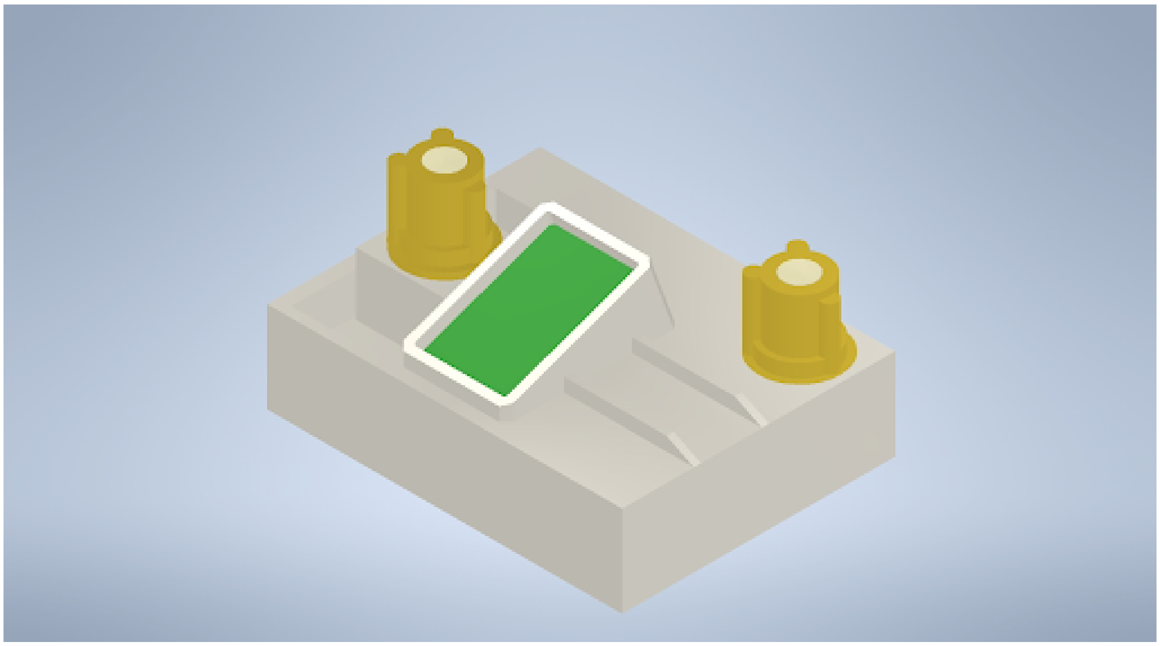


Figure 14: 3D model of SHERLOC

SHERLOC, from Perseverance, is a Deep UV spectrometer that was one of the first instruments considered in the first payload design to fulfill the mission and the first iteration of science objectives. SHERLOC has a mass of about 4.72 kg total and a volume of 26.0 cm x 20.0 cm x 6.7 cm (Beegle).

SHERLOC's function would have been to detect organic molecules, minerals, and compounds critical to show that Jezero might have supported past microbial life. The goals of SHERLOC were to "assess the availability of key elements and energy sources for life (C, H, N, O, P, S, etc.)" and "assess the habitability and potential of a sample and its aqueous history" (Beegle et al. 1).

This instrument was rejected because SHERLOC took up a significant amount of volume, power, and the cost budget, despite satisfying the originally chosen objectives. The instrument's hefty volume, cost ($80M), and power consumption (48.8 W in total) made this instrument unable to be utilized in the payload (Beegle). This was a sign that the science team had to start downsizing the mission and instruments. This led the team to consider breaking down instruments and bundles of instruments into their working components to reduce inhabited volume.

Energy Dispersive X-Ray Spectrometer (EDS)

The Energy Dispersive X-Ray spectrometer can detect atomic composition using x-rays. It is composed of a detector head and a Scanning Electron Microscope (SEM), allowing for imaging of the sample. The volume of a SEM in a small spectrometer is 330 mm x 614 mm x 547 mm.

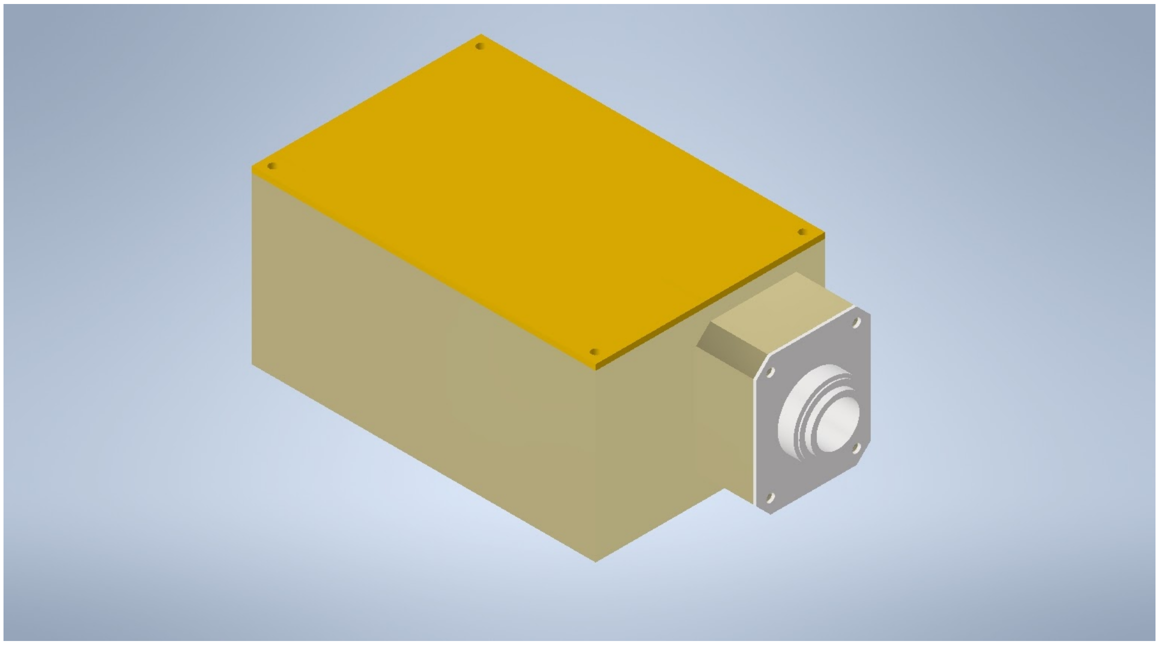


Figure 15: 3D model of the EDS Detector

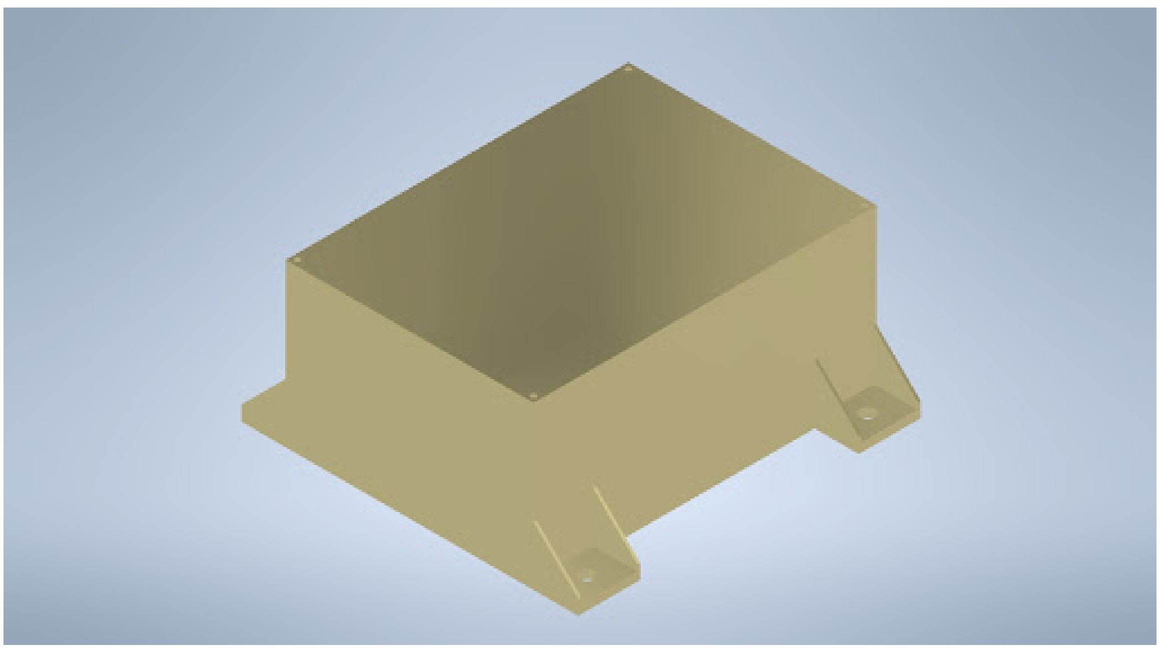


Figure 16: 3D model of the SEM

The Energy Dispersive X-ray spectrometer would excite a sample using x-rays, causing energy to be released as x-rays (UW Clean Energy Institute). That energy is characteristic of the element, allows the determination of the atomic composition of a sample, and with the Scanning Electron Microscope, images the sample with electron microscopy, giving context for the composition of the sample. The spectrometer was originally included to measure elements heavier than sodium and the distribution of elements in the sample, giving a means to answer all three previously chosen objectives.

This instrument was rejected mainly due to the volume constraint. Imaging samples with microscopy (especially electron microscopy) would prove useful for searching for signs of life, particularly potential remains of microbial life. However, the Scanning Electron Microscope was calculated to take up about 27.6% of the total volume. Despite the wealth of data that could be collected from the instrument on the chemical context of Jezero Crater, it would have been impractical to include it on the mission.

Multi-Hundred-Watt Radioisotope Thermal Generator (MHW-RTG)

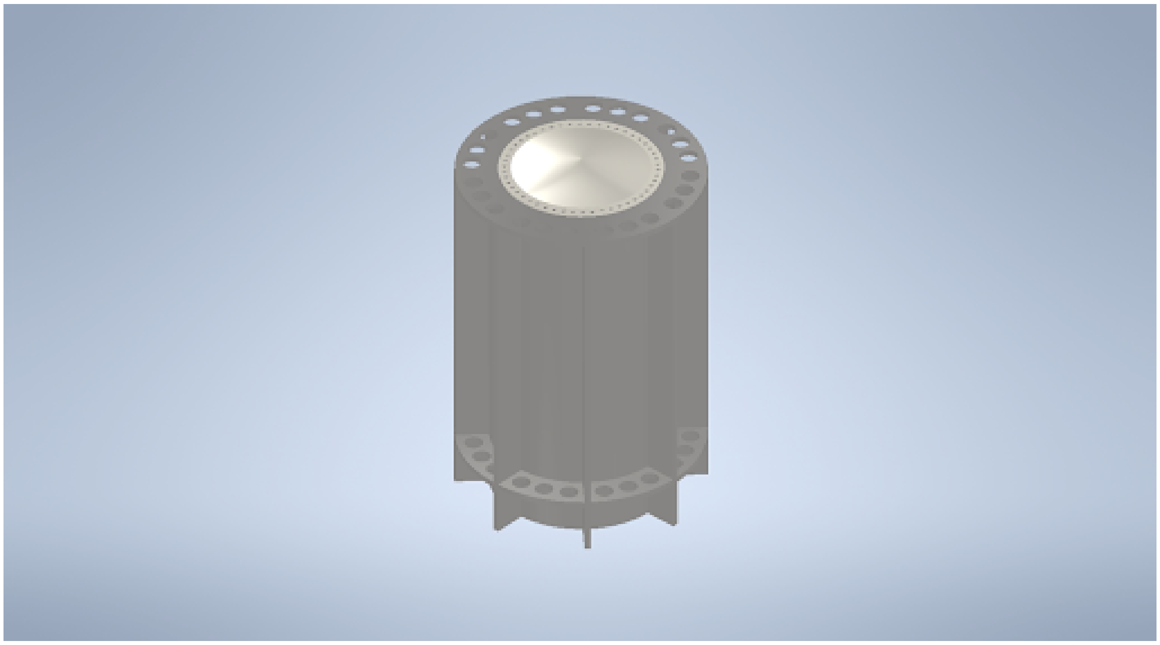


Figure 17: 3D model of the MHW-RTG

The Multi-Hundred-Watt Radioisotope Thermal Generator is a power generator that was considered in early design iterations, and has proven longevity through its use on the Voyager spacecraft. It produces 158 W of electrical power through the decay of Plutonium-238, has a length of 58.31 cm, a diameter of 39.73 cm, and weighs 37.69 kg (Cataldo and Bennett 10).

This instrument was chosen initially to resolve power concerns that might have arisen. Given the cold environment of Jezero and the observation that Perseverance is nuclear-powered, this power system would theoretically be the most viable, given the MHW-RTG's large power output. It would have been able to satisfy most power demands on the rover for a substantial amount of time independent from the time of day on Mars.

However, there were multiple problems with the RTG that produced significant complications. The instrument was rejected due to volume, cost, and heat complications. Although this instrument fit within volume and mass constraints, its length did not leave enough space for the remaining instruments and the EDL system. Also, the MHW-RTG would produce 2243 W of heat through the decay of Plutonium-238 that needed to be rejected (Lee and Bairstow 8). This was excessive to heat the instruments, and the magnitude of heat produced would certainly lead to problems with the mission's and instruments' functionalities. If this power system was included, just by the heat produced alone, this gave ample reason to reject this power source. Finally, a radioisotope thermal generator such as the MHW-RTG cost, at minimum, about $109 million (Werner et al. 4), immediately disqualifying it from the payload. Instead, solar power was considered as a more viable option.

Rejected Payload Design #2

The second payload consists of Alpha Particle X-Ray Spectrometer (APXS), SuperCam, Wide Angle Topographic Sensor for Operations and Engineering (WATSON), the General Purpose Heat Source (GPHS), and the Lightweight Radioisotope Heater Units (LW-RHU).

Alpha Particle X-Ray Spectrometer (APXS)

The Alpha Particle X-Ray spectrometer is an X-ray fluorescence spectrometer that measures the atomic composition of samples through the use of x-rays and alpha particles. APXS is in a cylindrical enclosure, 53 mm in diameter, and 84 mm in length, ending in an insulating flange of 68 mm x 68 mm (Rieder et al. 3). The instrument has a volume of 185.319 cm^3, a mass of 370 g, and its power usage is about 0.3 W (Rieder et al. 7).

APXS uses a combination of terrestrial standard methods, Particle-Induced X-ray Emission (PIXE) and X-ray Fluorescence (XRF) to determine elemental chemistry (Gellert). Like the EDS, atoms in a sample are excited by x-rays or alpha particles emitted by the instrument. It can detect elements from sodium to bromine, with some capability in detecting Germanium, Gallium, Lead, and Rubidium (Gellert). This would have allowed for a more thorough analysis of samples, gaining high quality data about the composition and geological history of Jezero, as well as the possibility of detecting constituents of organic compounds from clay and gaining insight into the history of the geology of the area.

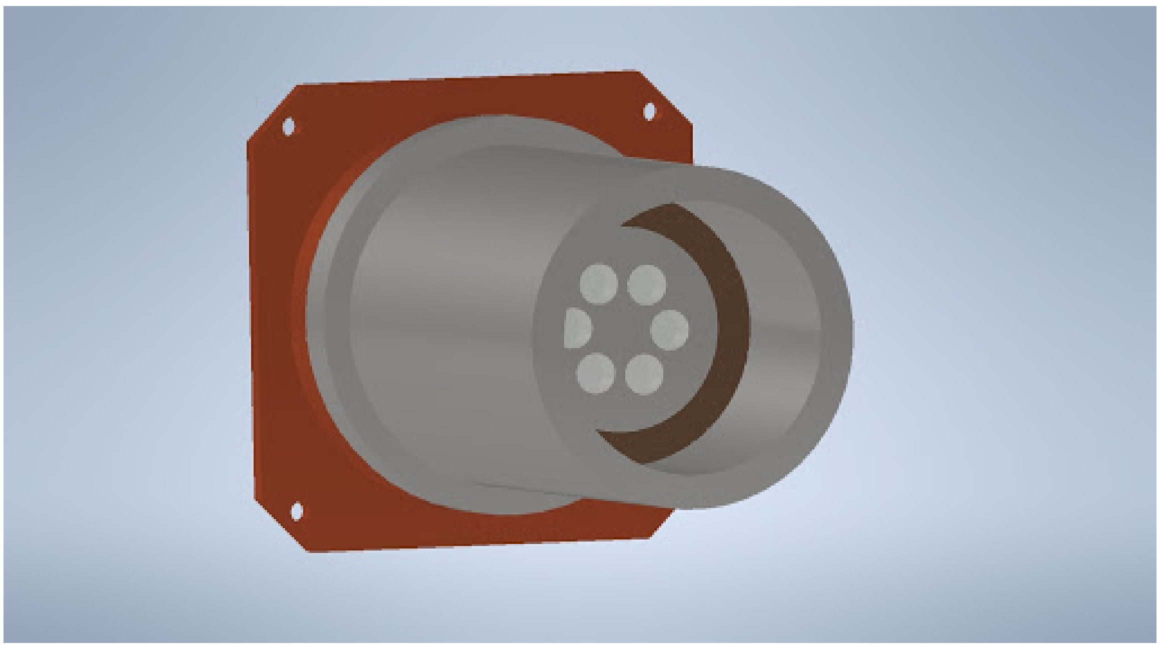


Figure 18: 3D model of the APXS

This instrument was rejected due to budgetary and volume constraints. The instrument cost \$30M according to estimates, and there were concerns about the budget of personnel and other potentially unforeseen budget considerations. Though the instrument itself is small, it was designed to function attached to a turret at the end of an arm, where the sensor head would be against the sample to collect data (Rieder et al.). This led to a concern of how much volume an arm would take, as the extended arm of MERS, which works with APXS, fully extended to around 90 cm (*The Rover's "Arm"*). This instrument would prove versatile in answering the three original science objectives which hoped to find signs of microbial life, characterize the geology of Jezero, and determine if there was sufficient geochemistry and geohistory to support life, but how realistically this instrument could be incorporated had to be considered. There was not any information found on how large the arm would be flexed and pulled inward. Given the payload's limiting volume constraints, it was appropriate to sacrifice this instrument and remove a mission objective in order to have room for other components (EDL and other instruments), rather than have a volume complication when finalizing the design. Again, this emphasized how small-scale the mission is meant to be and that downsizing from a top-down approach was the best way to do this until complications minimized and a workable, useful design emerged.

SuperCam

SuperCam is a collection of spectrometers that could provide a diverse array of data in the investigation of Jezero. It is about 10.7 kg in total mass and occupies a volume of 38 cm x 24 cm x 19 cm when mounted on a mast, as it is on Perseverance (Wiens).

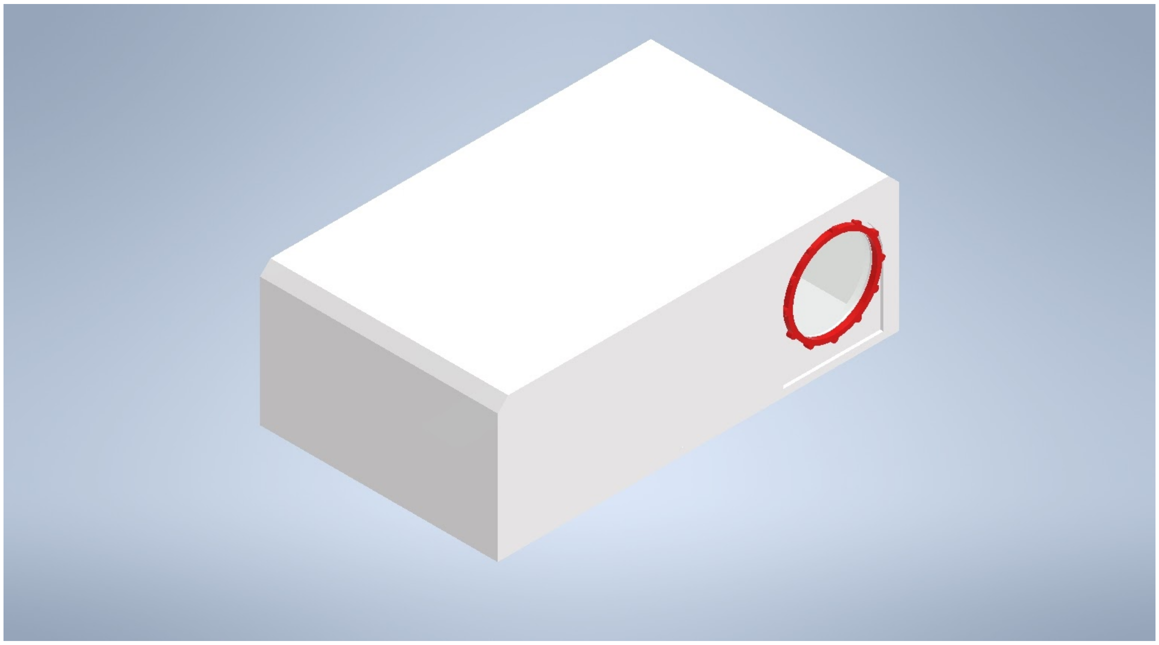


Figure 19: 3D model of SuperCam

SuperCam can perform Laser-Induced Breakdown Spectrometry (LIBS), Visible and Infrared Spectrometry, and Raman Spectroscopy. SuperCam can detect organic molecules, important minerals and life-significant molecules, and atomic composition of a sample. It is versatile, has multiple spectroscopy methods that could be performed at a distance (UV/Raman for organic compounds, LIBS for general composition, IR for minerals and molecules), could take color images of samples, remove dust from samples, and could fit with constraints (Wiens et al. 50). Using a color Remote Microimager, SuperCam also has the ability to take images of samples before beginning analysis, as well as removing dust from potential samples (Wiens et al. 51).

SuperCam was rejected due to budget constraints. Although a typical mass spectrometer was estimated to cost $80 million, there was an additional $15 million cost for an IR spectrometer. It also took up a significant portion of the allotted volume, which could potentially cause future complications for the other subsystems in the Fortitude rover. Also, as originally formulated on Perseverance, with SuperCam on the mast, most of its electronics were contained in the body of the rover, which could lead to other potential volume and instrument complications when connecting the body unit to the mast unit. Essentially, SuperCam would be contained to the mast, and its electronics would be on the rover's body. Given the large volume of the mast unit, the team wanted to avoid taking up so much volume in the payload in an effort to conserve the volume budget. Although the spectrometer from SuperCam was initially chosen to be the main, final spectrometer in the accepted payload, SuperCam's IR spectrometer could not determine the minerals necessary to detect and establish a context for investigating potential life. Overall, SuperCam was too bulky, large, and expensive to include in the payload. The budget would dictate that, if included, SuperCam would be the only science instrument, leading the team to consider taking instrument components from SuperCam instead to reduce cost and volume.

Wide Angle Topographic Sensor for Operations and Engineering (WATSON)

WATSON is Perseverance's imaging system that was planned to be used complementary to SHERLOC and is based on Curiosity's Mars Hand Lens Imager (MAHLI) imaging instrument. It consists of a color camera extended about 1.2 meters from the end of an arm and provides images of samples analyzed by the rover (Eshelman et al.).

According to the Mars 2020 eoportal, "WATSON provides views of the fine-scale textures and structures in martian rocks and the surface layer of rocky debris and dust" (*Mars 2020 Rover Mission of NASA/JPL*). It can image samples before they are analyzed, giving context and a possible explanation for the sample's composition with spectrometry.

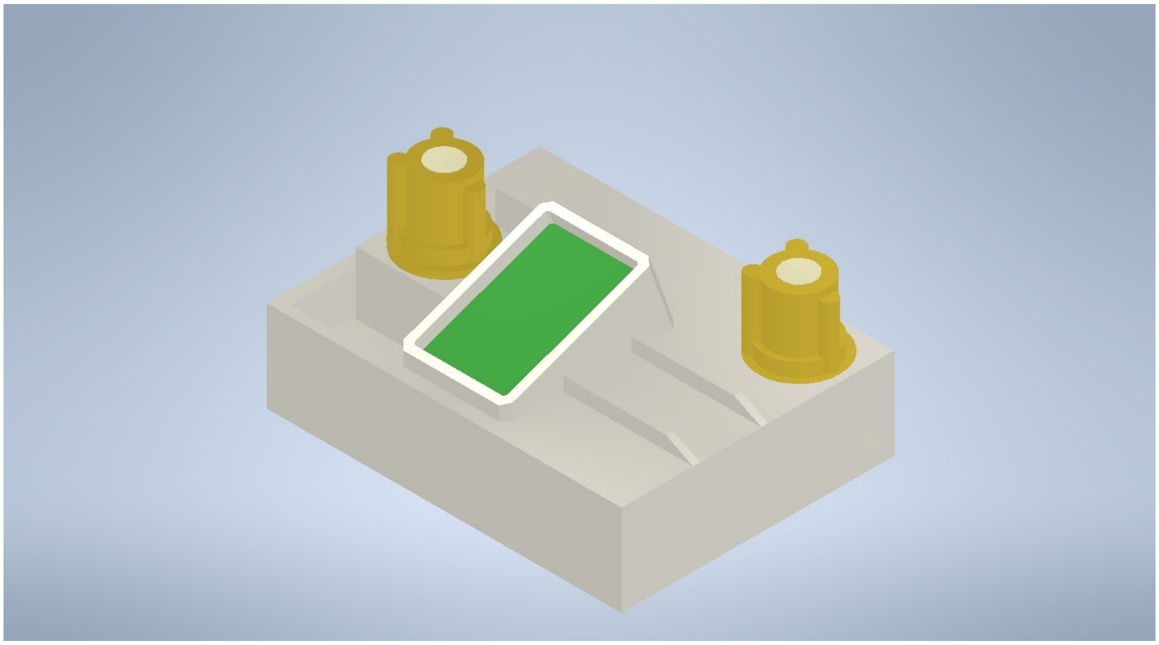


Figure 20: 3D model of WATSON

This was rejected because the length of WATSON violated the mission's dimensional constraints, making it impractical to include, especially given that there is not a method of reducing the size before deploying. The arm necessary to support WATSON as an imager also lead to further volume complications, similar to those encountered by APXS. Even though the team wanted to image samples and areas, it was deemed unnecessary and impractical if images can be viewed with a NavCam instead, which could be contained within a smaller volume.

General Purpose Heat Source (GPHS)

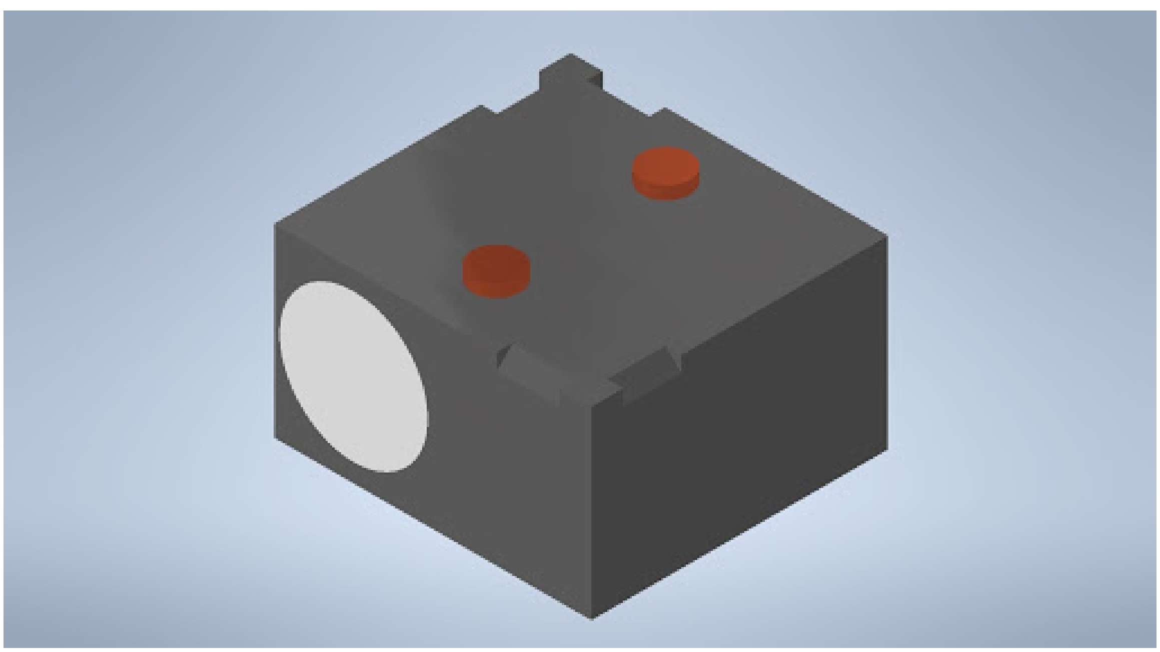


Figure 21: 3D model of the GPHS

The General Purpose Heat Source is an instrument that produces 250 W of heat through the decay of Plutonium dioxide. It contains 4 PuO_2 pellets, is encased in a 1.5 kg box of about 10.16 cm x 10.16 cm x 5.08 cm, and was originally positioned where the Li-ion battery and essential electronics were kept, to keep them warm ("General Purpose Heat Source"). The GPHS would act as a central heat hub where most of the heat comes from, and the heat would be distributed by the thermal and heat rejection systems.

The GPHS met the heating requirements for the instruments and rover. However, if the GPHS were implemented, it would have caused problems with the volume budget, as well as the heat excess and rejection through the continuous production of heat, therefore the GPHS was rejected.

Lightweight Radioisotope Heater Units (LW-RHU)

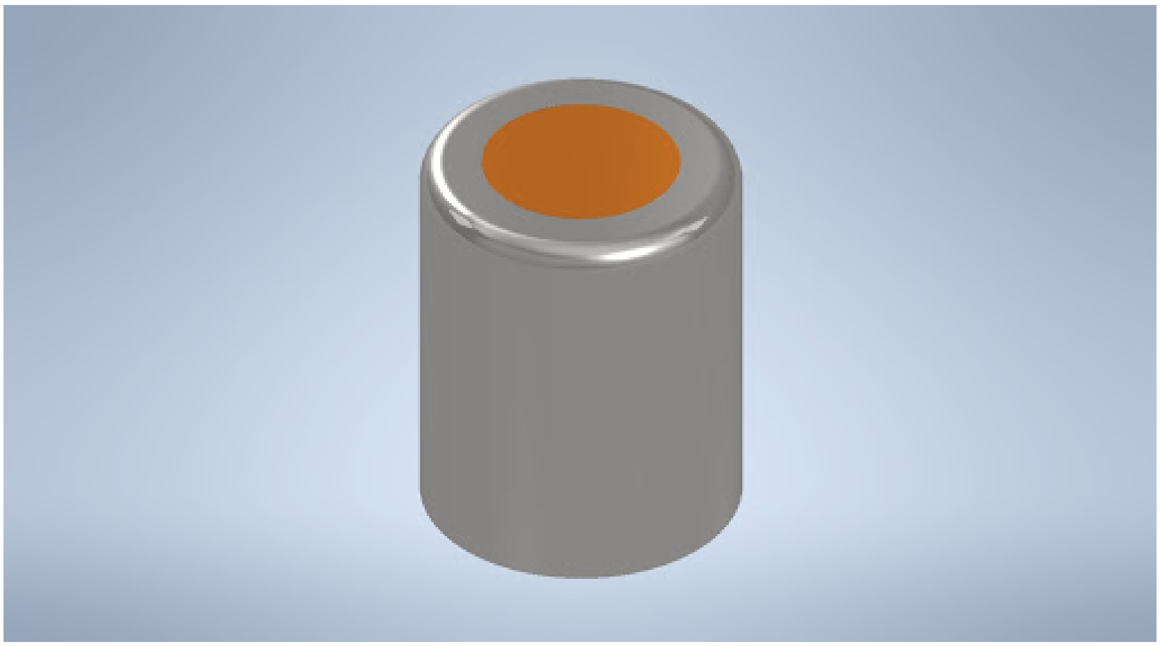


Figure 22: 3D model of the LW-RHU

Lightweight radioisotope heater units are small, 17 cm^3 cylindrical heaters, 3.2 cm long, 2.4 cm in diameter, and used to heat up components ("Light-Weight Radioisotope Heater Unit"). They each output 1 W of heat continuously through the decay of Plutonium dioxide.

The Radioisotope Heater Unit was one of the first heaters the team was considering, present in the teams' earliest designs. Given the larger volume of the GPHS, the 250 W heat output, and additional complexities included in the heat rejection system, multiple LW-RHUs were preferred for the mission. They were used in previous NASA missions and solar-powered rovers (MERS), and they are able to continuously produce heat to keep electronics and components working in the environment of Jezero. Their miniscule volume requirements also make them easily placeable as heat sources on a small-scale mission.

These were rejected because they were more expensive than initially calculated, and the thermofoil silicone heaters were more efficient and effective than the radioisotope heaters at meeting the thermal and heating demands of the rover. Multiple heater units, while effective through the continual production of 1 W of heat, might cause complications for the thermal system. This would be due to long heating times, insufficient heat, or the danger of too much heat being produced all at once, depending on the environmental conditions and temperatures experienced by Fortitude. There was also no way to turn off or adjust the amount of heat produced.

Accepted Payload Design #3

The final payload consists of an AOTF IR spectrometer, NavCam, 2 RAD750 processors, Silicone Thermofoil Heater, and a Helios Deployable Antenna.

AOTF-based IR spectrometer

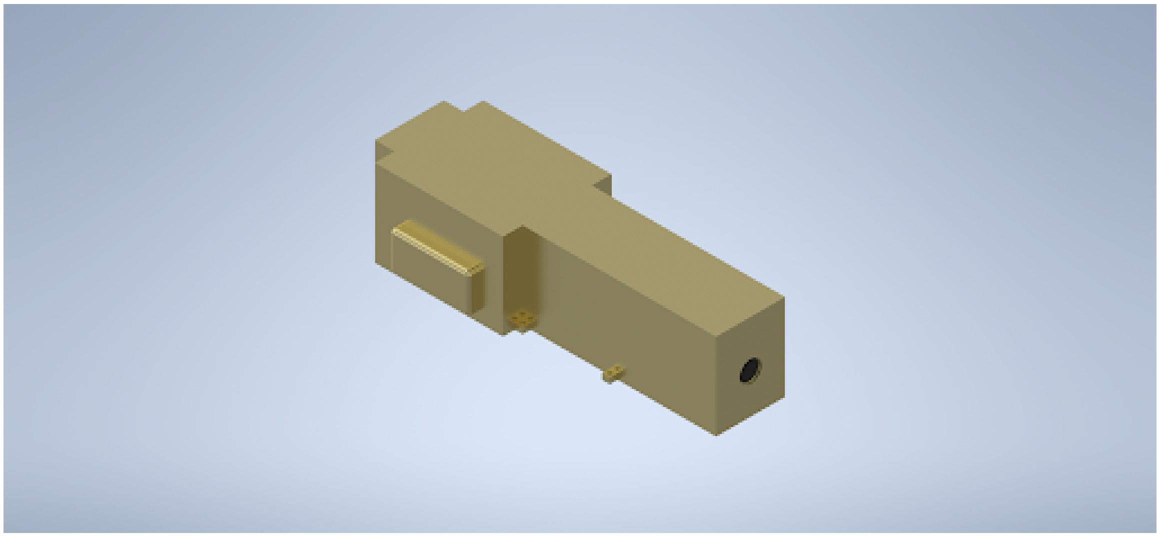


Figure 23: 3D model of the AOTF IR spectrometer

This spectrometer, taken from the prototype developed by Tawalbeh and others, was appealing because of its wavelength range. More specifically, it has a functional wavelength of 1.6-3.6 microns used to provide mineral signatures (Tawalbeh et al.). The wavelength range of this instrument allows the rover to characterize a variety of minerals and analyze the area's geology to understand how life could have arisen or if any remains persist. The associated IR spectrometer has a volume of 10.4 cm x 6 cm x 4 cm and a mass of 330 grams (Bain 122). It uses 2.4 W of power, and the operation of the acousto-optic filters was tested and shown to operate in the range of between -50°C and 40°C (Mantsevich et al.) (Tawalbeh et al.). This instrument was chosen because AOTF IR spectrometers are increasingly being used to analyze biological samples on planetary in-situ missions, and this instrument was designed with that purpose in mind (Tawalbeh et al.) (Korablev et al.).

The IR spectrometer will determine the molecular constituents of samples and soil, which includes carbonates, perchlorates, oxides, and ortho- and chain silicates (olivine) (Reess et al.). It emits infrared light with a wavelength in the range from 1.6 and 2.6 microns, with a spectral resolution of between 4 and 12 nm (or calculated wavenumbers of between 2.5 million - 833,333 wavenumbers) (Tawalbeh et al.). The bands that will be used will be 1.9 microns for carbonates (Harner and Gilmore), 3.4 microns for perchlorates ("Magnesium Perchlorate"), 2.7 microns for pyroxene (Clark), 1.7 microns for olivine ("Olivine-Group Minerals"), and about 2.91 microns for OH stretching from water in smectites (Djowe et al.). The wavelengths absorbed and their interactions are unique to the type of chemical bond (and therefore the molecule) present. The IR spectrometer is a compromise between the mission's original science objectives and chosen instruments, and what could be achieved realistically. The spectrometer was originally meant to satisfy Objective 1, but was realized to satisfy Objective 2 also. Given that this instrument is well-suited to identify minerals within the wavelength range it utilizes, it will identify Jezero's variety of rocks and soils, as well as whether those rocks and minerals could preserve signs of life, primarily through the detection of carbonates.

RAD 750 Processors (1 for redundancy)

The RAD 750 is an aerospace industry standard computer chosen for its high reliability on spacecraft. The processor has been used in previous NASA missions to Mars such as Perseverance and Curiosity, and is radiation-hardened for the vacuum of space and the Mars environment. It has a die-size of 130 mm x 130 mm, weighs 9 g , uses 5 W of power at 133 MHz, has an operating temperature of between -55°C and 125°C, a speed of 200 MHz, 2 gigabytes of flash memory, 256 megabytes of dynamic random access memory (DRAM), and 256 kilobytes of electrically erasable programmable read-only memory (EEPROM) (Cady) ("RAD750®Spacevpx Single Board Computer") (*Rover Brains.*) (Vis). It will be used to process orders from control on Earth and to store data to be transmitted to Earth. In case the primary processor fails, is unable to start, or is otherwise damaged, the second redundant processor will be used as a backup. This would especially prove to be useful in the event of a processor failure prior to needing to execute EDL maneuvers, and will ensure the longevity of the mission if the primary processor fails due to age or overuse.

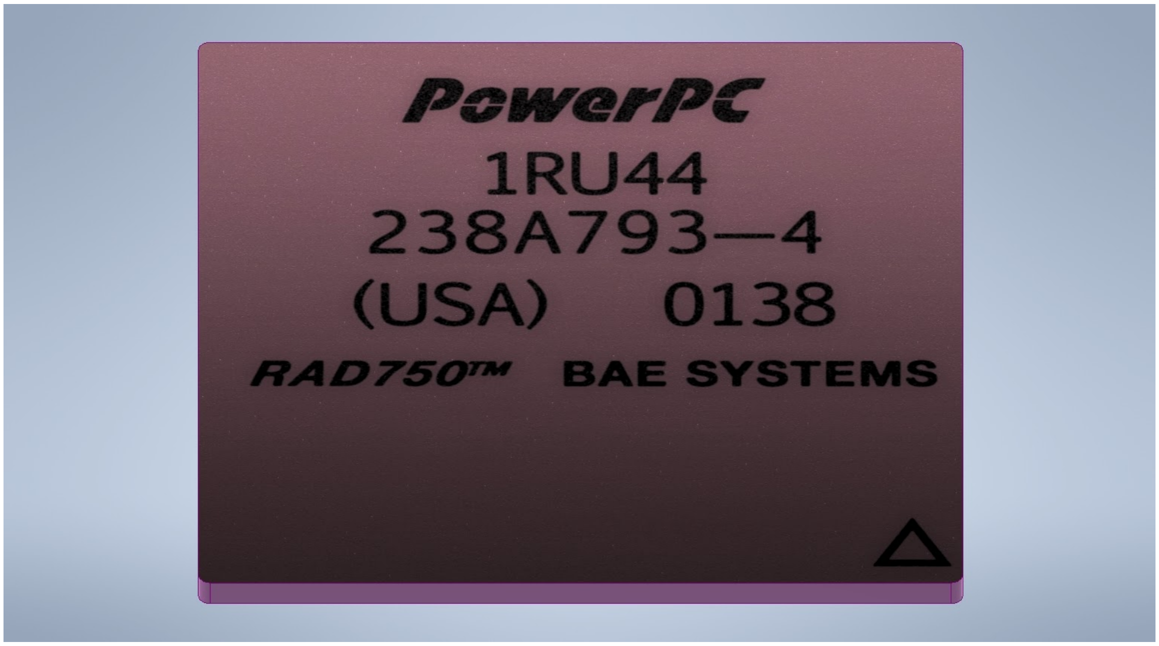


Figure 24: 3D model of the RAD 750 Processor

Navigation Camera (NavCam)

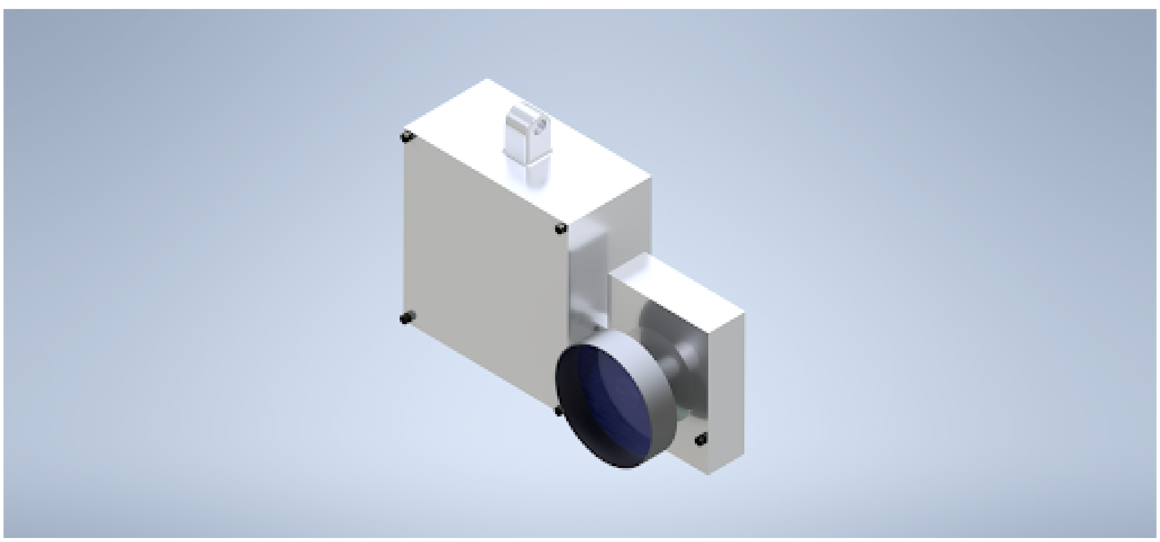


Figure 25: 3D model of NavCam

The NavCam, or navigation camera, has been used on the Curiosity mission, is soon to be used on Perseverance, and has been proposed and used on several other rover missions to aid in navigation (Hatakenaka et al. 5). NavCam consists of an electronics box about 6.7 cm x 6.9 cm x 3.4 cm, a detector head of 4.1 cm x 5.1 cm x 1.5 cm, and a 3.5 W heater (Maki et al. 4). NavCam weighs about 220 g, uses 2.15 W on its own, 5.65 W total with the heater on, and operates at a temperature range between -55°C and 5°C (Maki et al. 3).

NavCams are mounted on a mast and provide a general view of the area. The NavCam used in the mission will be attached below the IR spectrometer to provide distant and close up views of samples and the local environment before another instrument initiates sample analysis. It could be used to image geological landscapes or features of interest to help the rover navigate and provide additional data on the geology of Jezero.

This was chosen as a relatively inexpensive investment for the navigation system on the payload. Given that Jezero is one of the most hazardous landing sites on Mars, a well-considered navigation system is critical for the continuation of the mission past the landing phase. Given the volume and cost constraints, a NavCam was the bare-minimum necessity for adequate navigation into and through Jezero and for detecting hazardous landmarks.

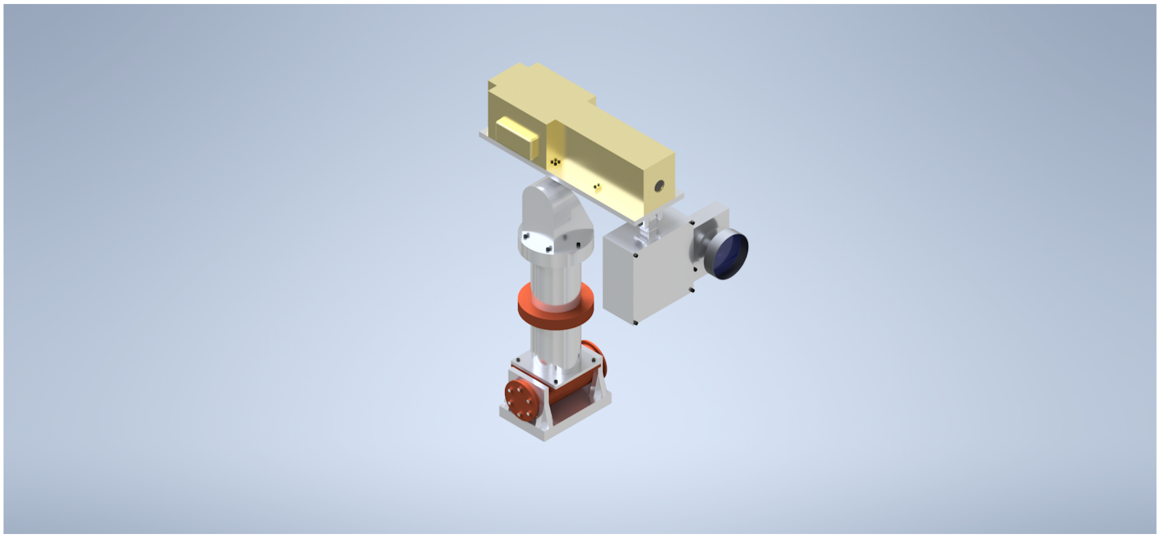


Figure 26: 3Dmodel of the IR spectrometer and NavCam on the mast

Helios Deployable Antenna

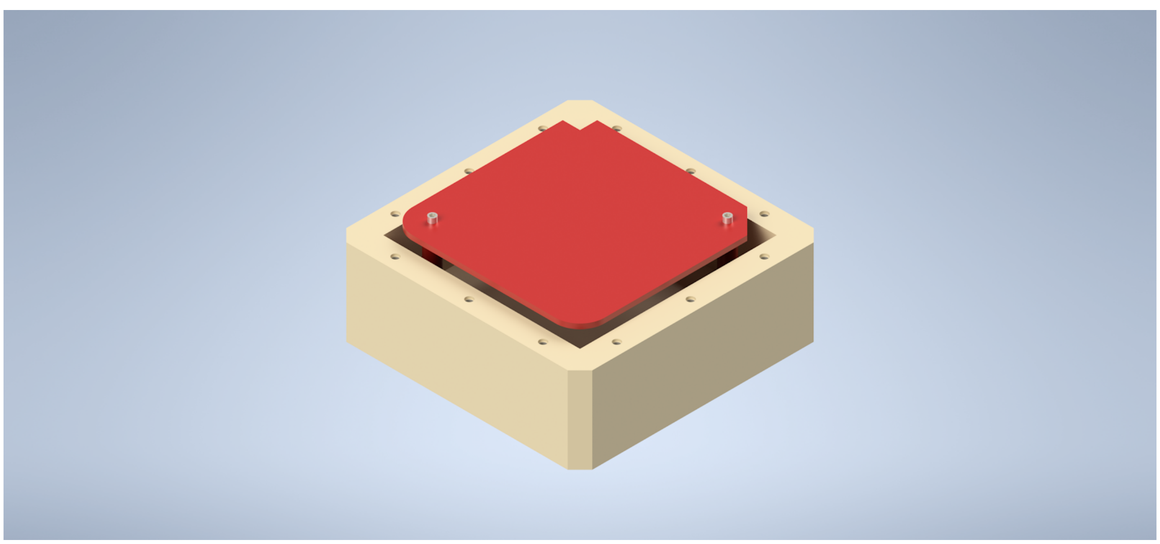


Figure 27: 3D model of the Helios Deployable Antenna

The Helios antenna has dimensions of a length of 100 mm, a width of 100 mm, and a stored height of 35 mm, with a deployed height of 330 mm. The antenna uses 1W of power to operate and operates between -40°C and 85°C (*Helios Deployable Antenna.*).

The antenna is able to support the communications system. It will be able to transmit signals up to 30 km from the landing site and communication package. It is space-rated by the United States Air Force, is heritage tested, and will be able to ensure communication with the communications package despite significant elevations to be experienced by the rover.

Flexible Silicone Thermofoil Heaters

Removing the RTG from the instruments list led the team to explore alternative power supplies, ultimately deciding on solar power. This was a great option for the constraints of the payload because the size and shape of the solar panels could be customized easily and made compact. Using solar panels required the use of batteries in order to power the payload at night. It was decided that a Lithium Ion battery would be the best option, due to its high energy density, high charging rate, and small footprint. A downside to this power source was that, unlike the RTG, the solar panels would not provide the payload with sufficient heat to keep the instruments and electronics inside at a safe and operable temperature. In order to combat this, a Flexible Thermofoil Heater Unit would be used.

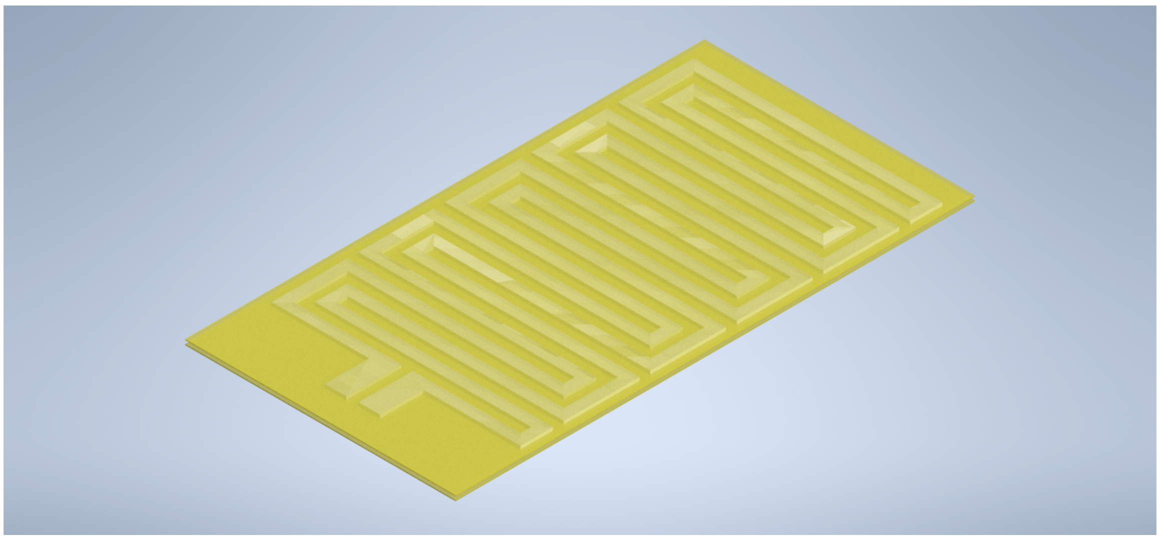


Figure 28: 3D model of the Silicon Rubber Thermofoil Heater

The Silicone Rubber Thermofoil Heater is a flexible heater that can provide heat to the rover and its components where necessary. The heater weighs about 0.04 g/cm^2, and 50 grams of the heater, covering an area of 12 m^2, would require 11.83 W to increase the temperature of Fortitude by 20°C in 60 seconds (”Flexible Heaters Design Guide”). They are thin, versatile, and require 300 W at 115 V.

The Thermofoil Heater is the main component of the Thermal Subsystem. They can be installed like a band-aid or double sided tape, and will be installed on the inside of the case housing all the components. The heaters will heat up the air inside the components, which will raise the temperature of the instruments over time.

This was chosen due to its versatility, low cost, efficiency, and thin flexible design. Given the thermal and heat demands of the rover, this was determined to be the best way to heat the rover and keep it and its instruments above minimum operating temperature.

2.3 Evolution of Mission Experiment Implementation Plan

Many things about science instruments and objectives changed. The team was very ambitious at first, rooted in the ambitious objectives of Viking and the AFL. However, given practicality and the size of the mission, many things were changed to result in the size and payload now decided upon, and Table 1 is used to illustrate the changes to the payload over time. As the team better understood the mission’s constraints, the goal shifted from directly finding signs of microbial life through atomic composition and imaging to indirectly detecting these signs of life through analyzing the geology with spectrometry. A major theme in the development of this mission was downscaling and downsizing. As the team developed the mission, many instruments, considerations, and even a science objective was cut as the team realized how small the mission actually was going to be. Much was learned in the capabilities of a small mission size and how decisions on projects such as these are made. The mission went from being independent, solitary, and broad in scope to being a scout/surveyor/precursor mission for future explorations of Jezero. The original Objective 1 was cut out, being: “Does the area show signs in the rock record that it once had the right environmental conditions to support past microbial life?”, further emphasizing a sense of downscaling.

Rejected Designs	**Accepted Design**
SHERLOC	Silicon Thermofoil Heaters
WATSON	AOTF-based IR spectrometer
Energy Dispersive X-Ray spectrometer	NavCam
APXS (Alpha Particle X-Ray spectrometer	RAD750 Processor
SuperCam	Helios Antenna
General Purpose Heat Source	
MHW-RTG (Multi-Hundred Watt Radioisotope Thermal Generator)	
Lightweight Radioisotope Heater Units	

Table 1: Instruments in the payload

The original idea was to directly observe or discover signs of life. However, given budget and size constraints, it was deemed beyond the scale of the mission to directly observe signs of past or present life, more suited for flagship missions such as Perseverance. However, life could be confirmed indirectly through the detection of molecules and minerals such as carbonates which are significant and related to the development and preservation of microbial life, giving cause for further investigation by future missions. The team can indirectly find and deduce microbial life by studying the geology of the area, determining the types of rocks and life-significant geology and minerals to determine past habitability. Atomic composition was determined to not be as important for this context as determining what molecules those atoms formed. In addition, the properties of those molecules and whether or not they could have preserved past life, or indicate signs of past life, can give us more information on Jezero in the context of the presence of life, than knowing the composition and distribution of the elements in Jezero. With that in mind, it was concluded that molecular analysis and identification can provide more significant and useful information than can atomic composition in finding signs of life and fulfilling the core ideas on which the mission was inspired by: determining if life arose and if signs of that life (past or present) still persist.

3 Descent and Lander Design

3.1 Selection, Design, and Verification

3.1.1 System Overview

Designing a lander to safely deliver a payload to the surface of Mars is not a trivial task. The martian atmosphere is very different from the atmosphere of Earth. Mars has an atmosphere that is composed of approximately 95% carbon dioxide, 2.59% nitrogen, 1.94% argon, and 0.058% carbon monoxide. The atmospheric density is approximately 1% of the atmospheric density of Earth. The average temperature on the surface of Mars is around -80°C while on Earth the average surface temperature is 15°C. It is clear that entry, descent, and landing (EDL) systems designed to work on Earth will not work on Mars. A completely different set of considerations and assumptions has to be made to safely deliver a payload to the surface of Mars. The largest factor of the EDL system is clearly the atmosphere. With the atmosphere being so thin, the lander system cannot rely on slowing down sufficiently, purely using atmospheric drag. The atmosphere is thick enough for aerothermodynamic heating to cause issues, and the different composition of the atmosphere makes the heating effects relatively unknown.

The process for designing an EDL system for Mars begins with being in a stable orbit around the planet. For our system, it was assumed that the orbit conditions around the planet would be at an altitude of 400 km above ground level and with a velocity of 3362 m/s. This payload is a secondary payload from a larger mission, and it is assumed that the two payloads will separate at these conditions above Mars. The criteria of our mission is to create an entire EDL system that must not exceed the volume constraint of 61 cm by 71 cm by 96 cm and the mass constraint of 72 kg, while the entire budget of the mission must not exceed 100 million US dollars. The cost of the EDL system is not included in the 100-million-dollar budget as the funding would be provided by another sponsor.

The landing site chosen for providing the best scientific value is Jezero Crater located at 18.23°N 77.35°E. It is believed to be a lake bed and could contain signs of life within the clay in the ground. Landing at this site is very difficult as there are many hazards, such as craters and sharp elevation changes located at the landing site. The main locations of scientific interest are very specific, so it is crucial for the mission to be successful to land near the areas of scientific interest. Decreasing the landing zone uncertainty was a top priority for the design of the EDL system. To accomplish the goals of the EDL system, the proper descent profile and lander design had to be determined.

The descent profile for Mars had to be chosen carefully. If the lander entered the martian atmosphere at too shallow of an angle, the lander runs the risk of skipping off the atmosphere and overshooting the landing site. If the lander enters the atmosphere at too steep of an angle, the aerothermodynamic heating effects and deceleration loading effects can exceed the limitations of the lander design and cause it to break up during descent. To help in choosing the proper descent profile, a Python program was written to approximate the conditions expected during descent. The program uses equations of motion (EOM) derived by NASA to predict the entry conditions of Apollo spacecraft in the upper atmosphere of earth. The chosen entry conditions for this mission were determined to be an angle of 14 degrees, a velocity of 3450 m/s, and an altitude of 350 km AGL.

The entry angle is defined as the angle between the tangent of the normal circular orbit path and the tangent of the parabolic path of the spacecraft. The angle of 14 degrees was chosen because this is a similar angle of entry used by previous Mars landers, such as the MSL spacecraft, the pathfinder spacecraft, the phoenix spacecraft, and the MER AB spacecraft. The lander has an entry velocity 3450 m/s because the spacecraft accelerates due to the gravity of Mars. The altitude is 50 km lower than the initial separation altitude because the spacecraft loses altitude during entry maneuvers. With these conditions, the expected maximum deceleration is expected to be around 8 g and the maximum temperature is expected to be around 2000°C.

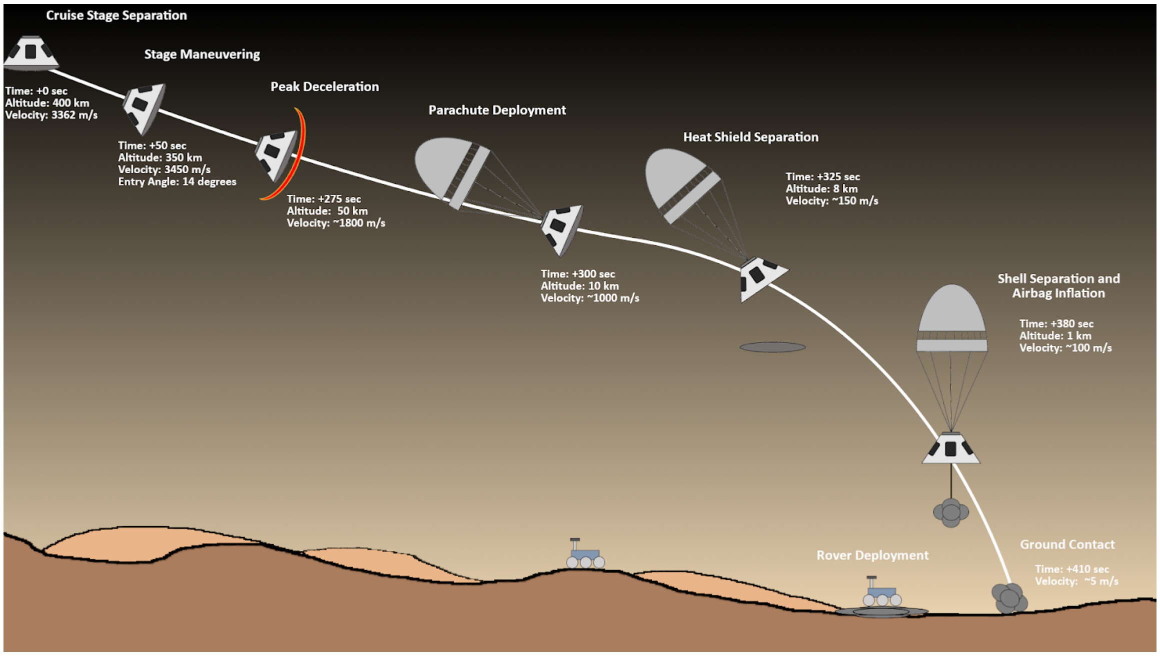


Figure 29: EDL graphic

Designing a lander with these entry conditions, the lander had to be split into multiple stages to ensure the payload will land safely. In Figure 29, the graphic gives a visual overview of all the stages of the lander. The EDL starts when the lander is separated from the primary payload. Here, the cold gas reaction control system (RCS) will activate, putting the spacecraft in the proper descent profile, and begin orientating the spacecraft for entry. At 50 seconds into the EDL process, it is expected that the lander is orientated properly and is on the proper descent profile of 14 degrees entry angle, 3450 m/s entry velocity, and at an altitude of 350 km AGL. Peak deceleration heating and deceleration of the spacecraft is expected soon afterwards. Here, the heat shield is expected to protect the spacecraft from the heating effect of the atmosphere. After peak deceleration and heating, the parachute deployment system (PDS) will deploy the 2-meter diameter disk-band-gap parachute. Deployment of the parachute will stabilize the spacecraft and decelerate the lander. After the deployment of the parachute, explosive bolts will fire, ejecting the heat shield from the lander stage. With the heat shield gone, the lander is lighter and decelerates faster. The impact stage, containing the payload, and the terrain relative navigation (TRN) system also become exposed.

The TRN is a system composed of a RAD750 processor, a quadrifilar helical antenna, a real-time Terrain Relative Navigation Lander Vision System (TRN-LVS), a Miniature Inertial Measurement Unit (MIMU), and the cold gas RCS. The RAD750 processor is the same processor used on the rover. This processor will be used entirely for the EDL portion of the mission. It is tasked with getting real-time images from the TRN-LVS and comparing the images with a map on-board to determine the location of the lander. The processor is also responsible for sending data back to the communications satellite during EDL so the data can be processed back to Earth. The processor will transmit the data via the quadrifilar helical antenna located on-board the payload. The TRN-LVS is a 20-megapixel CMOS camera responsible for taking images of the terrain in real-time and sending them to the RAD750 computer system. It will be facing downwards and located right next to the computer system. The MIMU is a Honeywell LN200 and is an inertial measurement unit (IMU). The MIMU is responsible for determining the orientation of the lander and the forces acted upon it. This data will be continuously streamed to the computer system during the entire EDL process. The cold gas RCS is responsible for making any corrections to the trajectory of the lander. The cold gas system is a compressed nitrogen system. It is made up of two storage tanks located on the sides of the lander and a system of valves and nozzles all around the lander stage. The RAD750 computer system will continuously receive data for the MIMU and the TRN-LVS, make corrections with the RCS system, and continuously stream data back to a communications satellite via the quadrifilar helical antenna located on the payload.

Once the TRN system detects that the lander is 1 km AGL, the lander will lower the impact stage via a Kevlar rope, and the airbag system located on the impact stage will deploy. The impact stage is an aluminum 7075 container designed with an airbag system and an unfolding deployment system that houses the payload. The impact stage is designed to detach from the lander stage shortly before contacting the surface of Mars. It will also have airbags inflate to help cushion the landing, like the Spirit and Opportunity missions. The airbags will surround the impact stage and will be filled by the nitrogen gas generators located inside the containing walls of the impact stage. The airbags will inflate to create a sphere of a 1.5 m diameter and will have multiple air pocket containers to ensure the airbags can survive bouncing and rolling on the martian terrain. The impact stage is suspended from a Kevlar rope below the lander stage to provide room for the airbags to deploy and ensure that the rocket assisted deceleration (RAD) engines will not damage the airbags. The RAD engines assist with slowing down the payload to minimize the impact velocity and to steer the lander stage away from the impact stage, so they do not run into each other. Once the on-board computer of the payload detects that the impact stage is no longer moving, ventilation flaps will open in the airbag system allowing the airbags to deflate. Once the airbags are fully deflated, the impact stage will start to open using the unfolding deployment system. The unfolding deployment system is a combination of pneumatic pistons and nitrogen gas generators located in the walls of the impact stage. The impact stage opens in a configuration to right itself up if it is resting on the side or upside-down. Once the impact stage is fully open, the rover drives off the impact stage and starts its mission.

3.1.2 Subsystem Overview

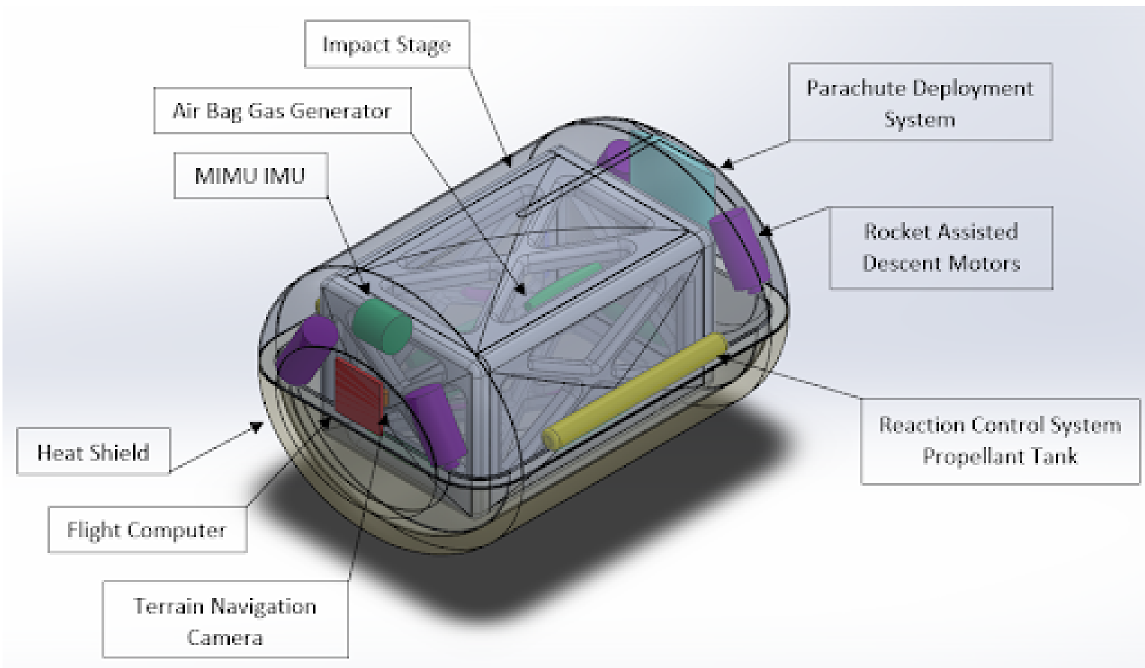


Figure 30: Complete descent and lander design

Lander Shell

The lander shell is the primary body where all the landing stage subsystems will be located. It is the transparent, grey upper half of the lander as shown in Figure 30. It is constructed of a 3 mm thick sheet of aluminum 7075 and has a mass of approximately 10 kg. The shape of the lander shell is designed to conform with the elliptical cylinder shape of the entire lander. The top of the shell has a grove embossed in the surface to allow the parachute lines from the parachute to connect to the shell and be in-line with the center of mass. This is done to prevent the lander stage from descending through the atmosphere in a crooked manner. Throughout the lander shell are holes for the nozzles of the RCS.

Reaction Control System

The RCS is used to control the position and orientation of the lander throughout the EDL. The system uses compressed nitrogen and expands nitrogen in a convergent-divergent nozzle system to produce thrust. Cold gas systems have weak performance but benefit from the simplicity of the design. Multiple pressure regulators and a nozzle will be placed around the lander shell to ensure the lander can be controlled. The pressurized nitrogen gas is contained in two separate containers located on the sides of the lander. In Figure 30, the RCS propellant tanks are indicated by the long yellow containers. The containers, in total, will be able to store 1750 cm^3 at a pressure of 4000 psig. Higher pressures allow for a high specific impulse (ISP) for the cold gas thrusters, but the primary limitations of the system are the pressure containers. Space is very cold, and cold temperatures make metals brittle. To keep the weight of the system down, the primary construction will be aluminum 7075. High strength steel would allow for the system to have higher pressure, but the system would be too heavy. The expected mass of the system is to be around 10 kg.

Nitrogen was chosen as the gas for the RCS system because it is inert, has a moderately high ISP of 80, and has a relatively high density of 0.28 g/cm^3. The RCS system will perform during entry maneuvering to prepare the lander for entry and adjust the lander during parachute descent with the remaining gas to make any corrections.

Heat Shield

The heat shield is the primary focus of the lander system. The heat shield is responsible for protecting the lander from the aerothermodynamic heating during descent. The heat shield material chosen for this lander is phenolic-impregnated carbon ablator (PICA). It is a low-density, insulative, ablative material well-known for its ability to withstand extremely high temperatures. PICA has been used by the MSL lander for the Curiosity mission and by SpaceX on the dragon capsule. PICA rejects heat by consuming the material of the heat shield and making an insulative char layer. Due to the material of the heat shield being consumed, the material must be sufficiently thick enough to protect the payload and lander system.

Without an in-depth study of the aerothermodynamics of the entry, it is difficult to determine how much heating is expected. Previous large-scale landers, like MSL, used a PICA heat shield with tiles that are 3.2 centimeters thick, allowing it to withstand thermal loads of up to 197 W/cm^2 and temperatures up to 2000°C. MSL was a very large lander with a mass of 1625 kg, had an entry velocity of 5600 m/s, and an entry angle of 14 degrees. The Fortitude lander has only a mass of 180 kg, an entry velocity of 3450 m/s and an entry angle of 14 degrees. To ensure that the heat shield would be able to protect the lander, a similar thickness was adapted and a thickness of 3.75 cm was chosen.

To support the heat shield through maximum deceleration, a 3 mm thick 7075 aluminum panel will be used. With the density of PICA only being 0.27 g/cm^3 and the thickness of the heat shield being 3.75 cm, the total mass of the heat shield comes out to be 12 kg with the aluminum support. After the lander passes the maximum heat and deceleration loading and the parachute has deployed, the heat shield will be jettisoned via explosive bolts. The heat shield is removed to decrease mass and uncover the TRN system.

Parachute Deployment System

The PDS is based on the Mars Exploration Rover parachute deceleration system. This system is simple and operates much like a cannon. The system is contained in a cylindrical container with one end closed and the other open and acts as a pressure vessel. The parachute is packed and placed near the open end of the container. On the other end is a charge or gas generator that can rapidly produce gas. When the parachute is ready to deploy, an electrical signal is sent to the charge and causes a reaction to occur. The reaction produces gas that propels the parachute out of the cylinder and causes it to unfold in the airstream.

The pressure vessel of the PDS is an elliptical shape to be able to fit in-between the impact stage and the wall of the lander stage. The PDS is located to the side of the lander stage to ensure that it does not interfere with the impact stage. This causes the problem with the parachute lines, which will not be attached properly above the center of mass. To address that issue, the lines of the parachute were attached to the center of the lander shell, and the lines were stowed away in a grove on the lander shell. The expected mass of the PDS is 5 kg.

Parachute

When traveling through the atmosphere at speeds multiple times the speed of sound, strange aerodynamic phenomena occur, and pressure waves begin to form. Traditional parachute designs that work at subsonic speeds on Earth cannot be used. They are no longer effective, and the forces experienced by the parachute would cause it to fail. The atmosphere on Mars is also very thin, creating the need for a very large parachute to slow down sufficiently. Creating a parachute for entry into Mars is a very difficult design challenge. The materials need to be very strong, the parachute needs to be very large, and the design of the parachute needs to work at supersonic speeds.

The parachute for the EDL system is a 2-meter diameter wide disk-band gap parachute made of nylon and Kevlar. This style of parachute was chosen because it is shown to work best in the Mach range of 2 to 3 which is the range that the parachute would operate. The diameter was determined by the descent profile program referenced in section 3.1.1. The proper diameter parachute was found by adjusting the size and seeing the estimated max deceleration. If the max deceleration was too large, the parachute would need to be smaller, and if the spacecraft was going too fast, the parachute would need to be larger. It was determined that a max loading of 8 g would be the upper limit of the spacecraft as it did not cause problems with the payload, and the lander would be able to decelerate enough for landing. With this method, a 2-meter diameter parachute was deemed appropriate for the lander. The material of the parachute was influenced by previous parachute designs, such as MSL for the Curiosity rover. Nylon is a strong and lightweight synthetic polymer that can be easily woven into the fabric of the canopy. Kevlar is an incredibly strong synthetic fiber and is heat resistant, which made it the perfect candidate for the parachute lines. The estimated mass of the parachute is 5 kg.

RAD750

The RAD750 was chosen for its high reliability on spacecraft. The processor has been used in previous NASA missions to Mars such as Perseverance and Curiosity and is radiation-hardened for the vacuum of space and martian environment. There will be three processors used in the mission. Two are in the rover and one is in the lander shell. The processor in the lander stage will be responsible for conducting all EDL-related calculations. This processor is the center of the TRN system. It will gather real-time images of the martian surface from the TRN-LVS and compare those images with a map located in the memory of the processor to determine the position of the lander during EDL. The processor will also be gathering data from the MIMU to determine the orientation of the lander and what forces the lander is experiencing. The RAD750 will also transmit diagnostic data during the EDL process to a communication satellite via the antenna located on the rover. If trajectory correction is required, the RAD750 will control the RCS system to make sure the lander is in the proper orientation and trajectory. The expected mass of the RAD750 computer system is 1 kg.

Terrain Relative Navigation Lander Vision System

The TRN-LVS is a CMOS camera system responsible for providing real-time imaging of the martian terrain to the EDL flight computer. The camera is a PhotonFocus DS1-D1024 CMOS visible imager with global shutter. The lens is a Kowa 6M6HC 6mm American Institute of Aeronautics and Astronautics 9 lens which, when paired with the camera, obtained an 85-degree FOV. The camera exposure times were on the order of 1 ms and the images were collected on demand at a rate of about 1 Hz with an image transfer time of around 30 ms. A similar system was used on the MSL and is being used on the 2020 Mars rover. The expected mass of the system is 0.3 kg.

Miniature Inertial Momentum Unit

The MIMU is a system developed by Honeywell to be able to determine the orientation of the spacecraft and the forces acting upon it. The MIMU is based off the LN200 IMU which contains fiber optic gyros and micro-machined (MEMS) accelerometers. Gyros can measure angular velocity, and the accelerometers can measure proper acceleration. The combination of these two sensors create the IMU and are able to determine the orientation of the spacecraft and the forces acting upon it. This sensor will allow the EDL flight computer to determine when to stage and when to use the RCS system. MIMU sensors have been used in many spacecraft including low-Earth-orbiting satellites, planetary missions like the Mars Reconnaissance Orbiter, and deep-space-probe missions like New Horizons. The expected mass of the system is 0.75 kg.

Rocket Assisted Deceleration

The RAD system consists of four solid rocket engines that will fire a few hundred meters AGL to help decelerate the payload. The RAD system has been used before on the Spirit and Opportunity rovers. They are responsible for slowing the payload down to zero velocity 10-20 meters AGL. The RAD engines are based on a scaled version of the Thiokol TX-58-4 solid rocket motors. The propellant of the engines used are C-1 polyurethane composite. The combined thrust of the four engines is 20 kN. This is enough thrust to decelerate a lander with a mass of 168 kg from 120 m/s to 0 m/s. The four engines are cantered to avoid the hot exhaust damaging the impact stage and the Kevlar bridal cable. The expected mass of the RAD system is 10 kg.

Airbag System

The airbag system is responsible for ensuring that the payload will contact the martian terrain safely. The atmosphere is not thick enough to decelerate enough with only a parachute, and the RAD engines are not enough to land propulsively. The airbags cushion the landing for the last few meters of descent. Airbag systems have been used by the Spirit and Opportunity rovers and have been proven to be successful. Each wall of the impact stage contains airbags on the outside. The airbags are multi-layer Kevlar material and have individual pockets to ensure that the sharp rocks of Mars will not puncture the bags. Contained in the walls of the impact stage are nitrogen gas generators, and they create the gas that will fill the airbags. The airbags will fully inflate with nitrogen gas in about 2 seconds, to a pressure of about 10.5 kPa. The generators will burn the propellant for a total of about 22 seconds. Once the lander has stopped moving, ventilation flaps will start to open for 1.5 hours to deflate the airbags and get them out of the way. The expected mass of the airbag system is 12 kg.

Unfolding Deployment System

Once the payload is resting safely on the surface of Mars and the airbags are fully deflated, the impact stage will now need to unfold to release Fortitude. The unfolding deployment system (UDS) is a pneumatic system located within the walls of the impact stage. Pneumatic pistons are used to actuate the walls to unfold them. Gas generators similar to the ones used for the airbag stage are used but are smaller and are used to power the pistons. The impact stage will fold out in a pattern that would right itself upwards in case it was not already upright. The expected mass of the UDS is 4 kg.

3.1.3 Dimensioned CAD Drawing of Entire Assembly

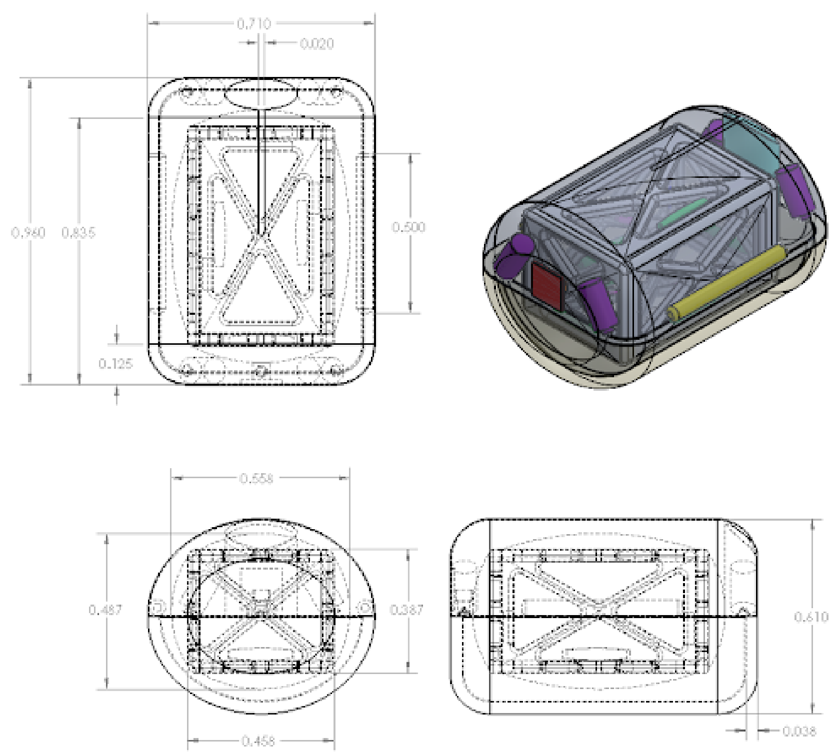


Figure 31: Lander assembly

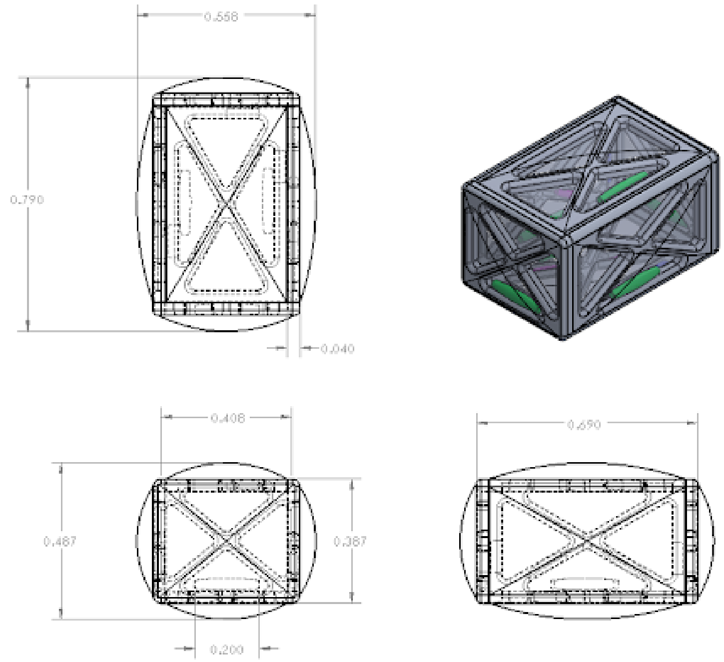


Figure 32: Impact stage assembly

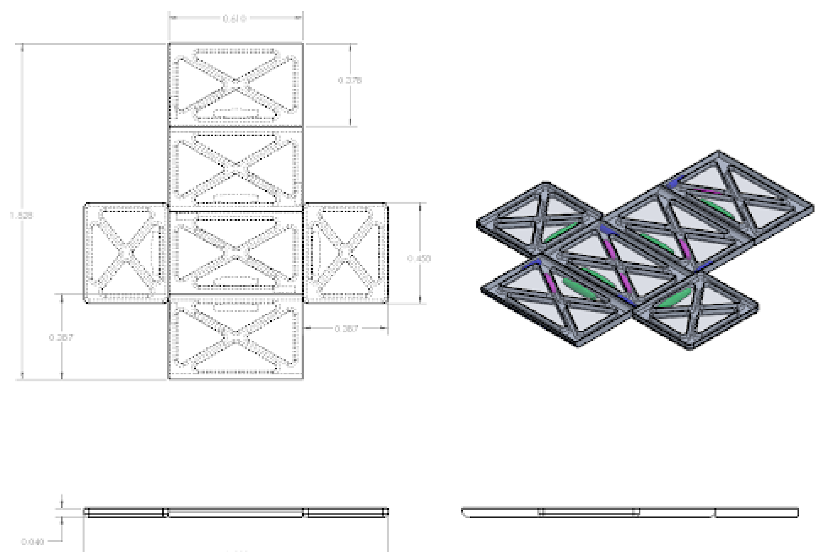


Figure 33: Unfolded impact stage assembly

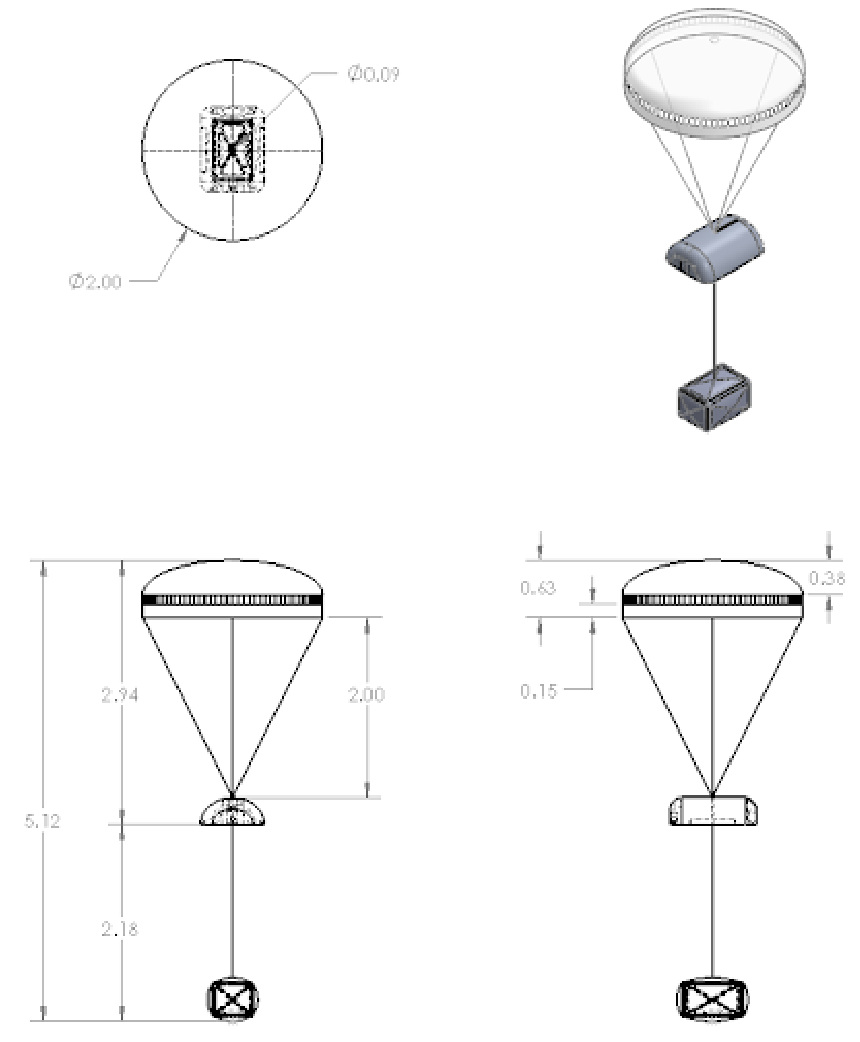


Figure 34: Lowered impact stage lander assembly

3.1.4 Manufacturing and Testing Plans

During the entry into Mars' atmosphere, the heatshield is the most vital instrument. Without it, the rest of the payload would burn, and the mission would be a failure. The team intends to solidify the mission by having Lockhead Martin manufacture and test a heat shield, as they regularly do for NASA. The structure needs to withstand 120% of the force that it is expected to face during the entry process. In order to imitate this strenuous amount of pressure, Lockheed Martin uses vacuum pumps to simulate the martian conditions. Manufacturing and testing will take a minimum of four months if no major difficulties arise.

The next phase of entry involves the parachute, responsible for slowing the payload to about 1/17 of its entry speed. The parachute will be manufactured by South Windsor Company and is required to withstand tens of thousands of pounds of drag force. Testing for the parachute will take place at Ames Research Center at NASA,. Here, everything from the deployment method to the expansion of the parachute will be tested, ensuring consistent performance under 125% of the drag expected on Mars. Manufacturing and testing will take 5-6 years, as testing requires the manufacturing of multiple rockets that will allow for proper testing.

The hydraulic engines, used to slow the lander down to about 4 meters per second, will be manufactured and tested by the NASA Stennis Space Center and will undergo the same six-series test that the previous E10002 model underwent. Manufacturing will take approximately six months, while testing will only take three months.

The airbag landing system is expected to take a relatively lighter impact, and bring the payload to a gentle stop. This vital part of the EDL stage has been previously manufactured by International Latex Corporation Dover Frederica and will be manufacturing the airbag landing system for this project. The airbags are made out of a light ventral fabric. The airbag testing includes a Mars-like environment that will be simulated in the Space Power Facility in NASA's Lewis Plum Brook Station, and multiple static drop tests will test the strength and effectiveness of the airbag system, as well as airbag deployment, inflation, and retraction for reliability. Manufacturing and testing of the airbag landing system will take approximately 18 to 24 months.

A cuboid lander will unfold and allow for the rover to make its way onto the Red planet. It will be manufactured by Lockheed Martin Space and will be tested in their facilities, meeting the same level of requirement that the older, tetrahedral-shaped lander met. The lander itself will only take 12 months to manufacture and test, however, it cannot begin to be manufactured until the scientific instruments and body of the rover undergo the manufacturing and testing processes.

The body of Fortitude and the wheels will be manufactured and tested by Boeing, Lockheed Martin, and NASA's Jet Propulsion Lab laboratory, in a combined effort. This testing will cover the sustainability and navigational capacity of the rover. The testing process will include having Fortitude navigate through courses that mimic the terrain on Mars. The body of the rover will take 36 months to manufacture and test, however, it cannot begin to be manufactured until the CAD designs of the scientific instruments are finalized.

The silicone thermofoil heater units are to be manufactured by TUTCO Heating Solutions group, as they specialize in working with these heating instruments that are able to operate under similar conditions like those on the face of Mars. The heater units will be tested in the NASA ARC Mars, where the atmosphere of Mars can be simulated. The heaters would be the most readily available component and would only take 2 months to test.

Finally, the scientific instruments which include the infrared spectrometer and the NavCam are to all be designed, manufactured, and tested by participating universities under the supervision of NASA's Jet Propulsion Lab. Testing of the instruments will be a dual effort between NASA and the universities. They will test the resilience of the materials, the reliability of the technology, and the quality of the data that the instruments are able to collect. Each instrument will take a specified amount of time to be designed, manufactured, and tested, however, they are all required to be ready for launch in a 48-month time frame.

3.1.5 Validation and Verification Plans

Validation

Validation plans serve to establish that all systems as properly functioning. Validation tests are crucial for interplanetary missions because without them, there is uncertainty in the systems' designs. For this mission, the EDL system will undergo validation tests, first with individual components, then as a whole. Below is a description of some of the tests chosen for validating.

Heat Shield Testing

The heat shield must be able to withstand the extreme temperatures it will experience. Therefore, it will be heated to an exaggerated temperature of about 2200°C ("NASA's Mars 2020 Mission Passes Critical Heat Shield Test"). The heat shield will also be tested for its aerodynamic competence. When departing from the entire system, the heat shield must not contact the lander, nor must it cover the camera system (Mitcheltree, et al.). Testing the release system in a wind tunnel will provide the aerodynamic analysis needed to qualify the heat shield design.

Parachute Testing

Validating the parachute system is essential to prevent the parachute from tearing or getting tangled, ultimately leading to a failed mission because the lander's speed was not reduced to a safe threshold. Three main components to parachute validation include testing for correct deployment, inflation testing, and drag and stability testing. Testing will be performed in the wind tunnel located at the National Full-Scale Aerodynamics Complex at the Ames Research Center.

The first tests do not need to be performed in the wind tunnel or in any special facility. The mortar firing tests serve to determine deployment speed and to observe parachute behavior during deployment. High speed cameras will be recording parachute deployment to collect speed data and to observe any possible entanglements. Once the parachute can deploy quickly and risk of entanglement is brought to a minimum, the following test can begin.

Inflation tests will be conducted in a wind tunnel with very high wind speeds. The parachute will be deployed and inflation of the parachute will be studied to determine if the parachute will inflate to its full capacity.

Determining the drag and stability coefficients is the most difficult of the parachute tests. Exposing the parachute to high powered winds, at different angles of attack, and relating test data to heritage "flight and wind tunnel data", will be sufficient (Mitcheltree, et al.).

After the parachute has passed these tests, it will be ready to perform in the martian atmosphere.

Airbag Testing

Validating the airbag system will consist of three main tests: high speed inflation tests, drop tests, and retraction tests. These three test categories will cover the three objectives of the airbag system, which are to inflate completely, protect the payload upon landing, and to deflate and retract the airbags.

All three major tests will be performed at the Space Power Facility. Using the vacuum chamber, the pressure will be decreased to 600 Pa and the temperature will be dropped to -80°C using liquid nitrogen. These conditions will replicate conditions the lander is expected to encounter when landing. All tests will also incorporate black markings on the airbags to monitor their behavior using high speed cameras and imagers.

Once these conditions are set, inflation tests will begin. The three gas generators will be set off and the airbags will inflate from their packed storage formation. High speed cameras will be recording the inflation process in order to determine inflation speed and any problem areas that may pose a threat for improper deployment. In order to validate the airbag system, the inflation process must take less than 2 seconds, and the airbags must fully inflate without any leaks.

The inflated airbags will be carrying a substitute lander weighing the same amount as the real lander and will be dropped from 15 m AGL onto a flat surface. After several tests are performed on the flat surface, the platform will be set to a 60 degree angle, and covered with a variety of dry sand and rocks (NASA). The rocks will range in size from 0.25 m to 1 m (NASA).

All drop tests will be analyzed for drop speed, post-impact speeds, spin orientations, and spin speeds. These post impact test criteria will help to understand the behavior of the lander when it is bouncing to a stop. After each test, the lander will be inspected for any punctures, tears, or any other damage to the system as a whole, repairing any abrasions after each test in order to not contaminate the following tests' data.

The final stage of testing will include multiple retraction tests. These tests serve to ensure proper deflation and withdrawal of the airbags. The retraction tests will be performed on a surface simulating the martian terrain. This will consist of a surface covered in dry sand of different grain size, and sharp rocks 0.25 m to 1 m in size. This substrate will serve to reproduce martian regolith and rock conditions. The Airbag Retraction Actuators (ARA) will begin to pull the ventilation flaps and deflate the airbags, providing a clear path for the payload to exit once it positions itself upright. If upon deflating, the airbags do not deflate properly or become entangled among themselves or in rocks, the system cannot be validated. The entire deflation process must take 90 minutes. If the process takes a longer or shorter amount of time, the ARAs must be altered to perform at the correct retraction rate.

Propellant Tank Testing

The propellant tank is what contains the propellant used by the system. If this tank does not efficiently store the propellant for the entire duration of flight, there will not be enough propellant for the system to use. In order to validate the propellant tank, it must be stress tested and thermodynamically tested. Along with these tests include Bubble Point Testing and Hydrostatic Burst Testing, which test the pressure constraints of the system. Successful testing of the tank with these procedures will ensure the integrity of the design.

RAD Motor Testing

Without a properly validated RAD Motor system, the lander will not safely land on Mars. These motors play a huge part in reducing the speed of the lander. Practice tests can be performed by attaching the motors to a substitute lander and dropping it from multiple elevations from 5 m to 20 m. This range of elevation will give a good indication of how the motors will react when they are at the expected 10 m to 15 m above the martian surface.

Only after these conditions are met can the airbag lander system be validated for launch.

Once the individual EDL components can be validated for successful performance, the assembled system will undergo a series of tests. EMC testing will be performed to locate any interference between the electronics. Vibration tests will ensure the EDL system can withstand the intense turbulence of entry.

Verification

A verification plan is a plan that serves to ensure all systems are working when they are out in the martian atmosphere. This plan is a way of communicating confirmation that EDL systems have completed their goals. Verification will consist of a number of sensors that communicate signals to the computer and then to Earth.

Using the Honeywell MIMU LN200, a change of speed will be detected when the parachute deploys, which will be communicated with the computer, and a signal will be sent back to Earth. This detection of the change in speed will suffice in verifying that the parachute has deployed correctly by comparing the data with what is expected.

Once the parachute is deployed, the camera system will become activated. The camera will take an image before the heat shield is jettisoned. It will compare that image to images it has stored in its system and determine when the heat shield has been deployed and is no longer obstructing its field of view. If the images determine an obstruction past the allotted time for heat shield deployment, then it can be concluded that the heat shield did not deploy correctly.

Once the heat shield has been removed, images of the ground will be taken in order to determine its velocity and path. "For the Mars landing application, images are taken during parachute descent and processed to extract matches between descent images and the map" ("Real-Time Terrain Relative Navigation Test Results from a Relevant Environment for Mars Landing"). All images will be evaluated through the computer system to determine the amount of correlation between the images taken and the ones that were previously stored. If the correlation levels are low, it can be concluded that the path taken is not the path that was intended, or that there is a fault in the camera system, depending on how much of a difference there is in the two sets of images.

Sensors on the gas generators will communicate when all of the solid propellant has been used. This will verify that the airbags have been inflated correctly and can protect itself upon impact. Shortly after the bridle has been cut, the camera system will take one last image to verify that the lander has been released.

These steps ensure that the major systems have completed their objectives and communicate that back to Earth. These verification plans will give a better understanding of what has actually happened upon arrival and will help scientists and engineers to better understand current EDL strategies, ultimately leading to improvements for future missions.

3.1.6 FMEA and Risk Mitigation

- **Occurrence:** Occurrence Scale (1-10) with 1 being highly unlikely and 10 being almost certain.
- **Severity:** Severity Scale (1 -10) with 1 being not noticed by a customer and 10 being hazardous or life-threatening and could place the product survival at risk.
- **Detection:** Detection Scale (1-10) with 1 being almost certain to detect and 10 being almost impossible.
- **RPN:** Risk Priority Number is calculated by multiplying the Occurrence x Severity x Detection.

SEVERITY										
10	A B	C								
9			E F							
8										
7										
6										
5										
4										
3										
2										
1			D							
	1	2	3	4	5	6	7	8	9	10
	OCCURRENCE									

Table 2: FMEA Criticality Matrix - EDL

High
Med
Low

Table 3: Criticality key

Function(s)	Failure Mode(s)	Effects(s)	SEV	Cause(s)	OCC	Design Controls (Prevention)	Design Controls (Detection)
A. Heat shield The heat shield protects the lander and rover from the intense heat from entry into the martian atmosphere and aerodynamically acts as the first "brake" for the spacecraft	Fail to protect the lander and rover from the heat during entry into the martian atmosphere and fail to decelerate the aircraft while entering in the martian atmosphere	The rover would 'burn' in the martian atmosphere	10	Extreme friction and heat that occurs during entry into the martian atmosphere	1	Perform temperature testing at about 2200°C, testing the release system in a wind tunnel will provide the aerodynamic analysis needed to qualify the heat shield design	First part of test on startup

Table 4: FMEA Criticality Matrix descriptions (A) - EDL

Function(s)	Failure Mode(s)	Effects(s)	SEV	Cause(s)	OCC	Design Controls (Prevention)	Design Controls (Detection)
B. Parachute system Prevent the parachute from tearing or getting tangled	Fail to reduce the lander's speed to a safe threshold	Spacecraft would crash	10	Being incapable of hypersonic declaration	1	Perform testing for correct deployment, inflation testing, and drag and stability testing	First part of test on startup
C. Airbag system Inflate completely, protect the payload upon landing, and to deflate and retract the airbags	Fail to inflate completely, protect the payload upon landing, and to deflate and retract the airbags	Spacecraft would crash	10	Being incapable of protecting the payload upon landing and deflating and retracting the airbags	2	Perform high speed inflation tests, drop tests, and retraction tests	First part of test on startup
D. RAD750 Process orders from control on Earth and store data to be transmitted	Primary processor fails, is unable to start or is damaged	The second redundant processor would be used as a backup. The mission and the spacecraft would not be affected	1	The primary processor could fail due to age or overuse, or it could be damaged during EDL	3	No design control (prevention) needed	No design control (detection) needed
E. Propellant tank Stores the propellant used to accelerate the spacecraft	Fails to safely store the propellant for the length of the mission needed	The energy needed to accelerate the spacecraft will not be entirely produced	9	The tanks are not successfully protected from mechanical shocks or temperature changes that may crack the grain	3	Perform Bubble Point Testing and Hydrostatic Burst Testing	First part of test on startup

Table 5: FMEA Criticality Matrix descriptions (B-E) - EDL

Function(s)	Failure Mode(s)	Effects(s)	SEV	Cause(s)	OCC	Design Controls (Prevention)	Design Controls (Detection)
F. Rocket Assisted Descent Motors Essential for a safe landing on Mars. Bring the downward movement of the lander to a halt some 10-15 m above the surface	Fail to provide the force needed to further assist in reducing the speed of the lander	Rover would hit the ground roughly as fast as about 350 km/h	9	RAD being incapable of slowing down the spacecraft enough to provide a safe, low speed landing	3	Perform testing by dropping the "test stand" from a helicopter which would fly two miles above the ground	First part of test on startup

Table 6: FMEA Criticality Matrix descriptions (F) - EDL

DET	RPN	Recommended Action(s)
3	30	Carefully analyze the data after temperature testing and testing the release system in a wind tunnel have been performed
3	30	Carefully analyze the data after the testing for correct deployment, inflation testing, and drag and stability testing have been performed
3	60	Carefully analyze the data after high speed inflation tests, drop tests, and retraction tests. have been performed
1	3	No action recommended
3	81	Carefully analyze the data after the Bubble Point Testing and Hydrostatic Burst Testing have been performed
3	81	Carefully analyze the data after the "test stand" testing has been performed.

Table 7: Recommended Actions Chart - EDL

3.1.7 Performance Characteristics and Predictions

Having landed safely, the payload will begin its way to the investigation sites. The path taken is illustrated below in Figure 35. This path was chosen because it was deemed to involve the least amount of hazards.

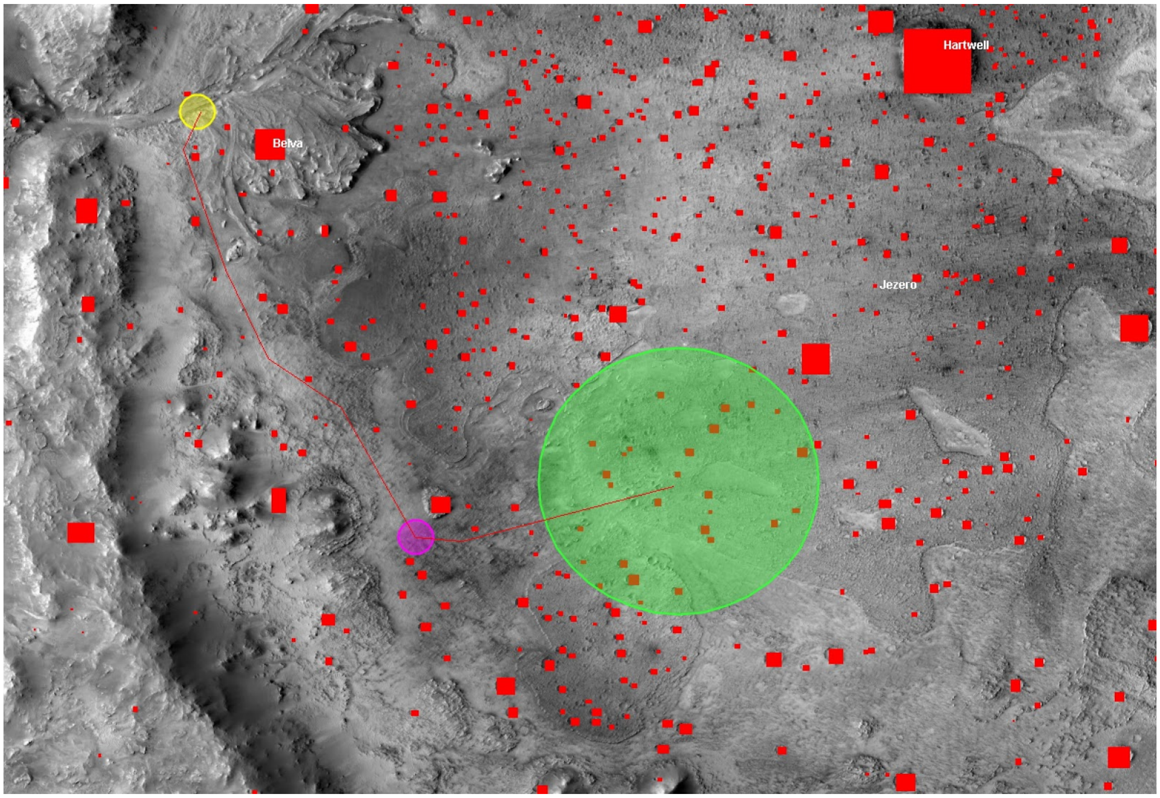


Figure 35: Payload path

The Mars atmosphere is only 1% as dense as Earth's. An atmosphere that is thick enough to create substantial heating, but without a sufficiently low terminal descent velocity, will create challenges. Since hypersonic deceleration occurs at much lower altitudes on Mars than on Earth, the time remaining for subsequent EDL events is a concern. By the time the velocity is low enough to deploy supersonic or subsonic decelerators, the vehicle may be near the ground with insufficient time to prepare for landing. In this case, the heatshield's aerodynamics will act as a 'brake' to decelerate the payload. The implementation of a parachute would also significantly contribute to the deceleration of the descent module during entry into the martian atmosphere.

The friction with the atmosphere is also to be considered when the spacecraft enters Mars' atmosphere. In order to protect the rover from extreme friction and heat that occurs during entry into the martian atmosphere, a heat shield made of PICA (Phenolic Impregnated Carbon Ablator), would ensure that the spacecraft does not 'burn' in the atmosphere.

The Mars' lower limit temperature is about 125°C. This temperature limit is considered to be a significant risk for the rover, especially during the night. The low temperatures could cause 'the freezing' of the rover. The implementation of 1-watt Heater Units, would prevent the heat from leaving Fortitude's body. The heat generated from Fortitude's electronics could also be considered a source of heat.

Dust storms, which are capable of blanketing the entire planet, could cause issues in the descending/landing stage. Dust can cover equipment and decrease the amount of sunlight hitting the panels. Less sunlight would mean less energy created and a decrease in the efficiency of solar panels. Dust could also delay certain phases of the mission. When global storms hit, surface equipment has to wait until the dust settles. This either to conserve battery or to protect more delicate hardware. Additionally, significant atmospheric dust content increases the temperature of the lower atmosphere, reducing density and requiring conservatism in the selection of landing site elevation.

3.1.8 Confidence and Maturity of Design

The speed and angle at which the payload is expected to enter Mars' atmosphere were adjusted as more accurate information was provided to ensure a safe landing. Critical and extreme conditions, such as entry burn, were not only considered but were also modeled to a high degree of accuracy using computer simulations. This allows for the proper preparation to be executed in a timely manner and, as a result, allows for there to be confidence in the trajectory of the payload upon entry.

Having drawn reference from other successful lander designs, such as Opportunity and Curiosity, allowed for early iterations of Fortitude to be extraordinary starting points. From there, phase strategic maneuvers were considered in every stage during the descent, including how the impact stage would release from the lander. The mission requirements and restrictions did influence the lander's geometric identity.

The conical shape of the lander was redesigned multiple times to maximize the probability of a successful mission. After deliberating on the trade-offs that come with changing a design, a cylindrical casing ultimately proved to be able to grant optimal cargo space for the rover and its instruments, as well as enough downward-facing surface area so the heat shield and lander won't burn upon entry.

The testing methods and their justification all exceed the conditions that systems will be expected to face on Mars. The heat shield has to be able to withstand 120 percent of the heat it will face upon entry. Additionally, every instrument that was elected for use, as well as the final assembly of the instruments will be tested in state-of-the-art facilities from prestigious and relevant corporations such as Boeing, Honeywell, and Lockheed Martin. To record the ongoing progress of the descent and lander upon entry, we rely on the MIMU and the TRN-LVS to send diagnostic data to mission control to appropriately respond in the unlikely situation that some aspect of the entry, descent, or landing process is compromised.

To summarize, the entry descent and landing design has gone through many iterations, and there will be an ample amount of testing programs that put the design through a sufficient amount of rigor that ensures the confidence of the design.

3.2 Recovery/Redundancy System

To ensure that the mission is successful, multiple recovery and redundancy systems have been put in the lander system. Due to size and weight constraints, it is difficult to have redundant systems in the EDL. If a redundancy could not be put in place, a safety margin of at least 20 percent was placed on that system and followed a rigorous testing phase to ensure the system will not fail during EDL.

The heatshield is a critical component of the lander that could not have redundancies put in place. The heatshield is designed with at least a 20 percent safety margin to ensure the lander will not overheat during descent. Multiple heatshields will be created and tested until failure to be certain the system will not fail during EDL.

The parachute system of the lander will not be able to have redundancy. This is due to the mass and size constraints of the lander. The parachute's materials will have a 20 percent safety margin and will be put under a rigorous quality control and testing program. The PDS system is very simple but will not be able to have redundancy. The charge used to create the gas, which will propel the parachute out of the PDS cylinder, will have an extra igniter to make sure the PDS will not fail.

The RCS will have two gas reservoirs. If one fails, then the other tank will still be able to operate. The EDL flight computer will adjust how often it will use the RCS system to make sure the system will not run out of propellant prematurely. The RCS also has multiple nozzles and regulators throughout the lander, so if one should fail, then the system will still be able to operate to a limited degree.

If the RAD750 EDL flight computer fails, then the lander will switch to the on-board computer system in the rover. The backup EDL flight computer system will run to a limited degree in a safe mode, which will involve a stripped down EDL program system. The sensors of the EDL systems will transmit data to the computer at a lower bit rate. This to ensure that the communications between the EDL systems and the backup computer do not become saturated and overload the system.

The MIMU and TRN-LVS subsystems will not be able to have redundancy due to size limitations. The MIMU has a great record of reliability with previous spacecraft like New Horizons and the Mars Reconnaissance Orbiter. If the MIMU system fails during descent and the parachute is deployed the parachute will stabilize the lander reducing the need of the MIMU system. The MIMU will follow a rigorous quality control and testing program to ensure it will not fail during EDL. The TRN-LVS will not have any substitutes to provide live images of the ground. It will determine the approximate location of the lander the flight computer can communicate with the nearby communications orbiter and use the Doppler effect to determine how far away they are from each other. Then, the deep space network can provide the communications orbiter its approximate location, and at that point, a computer program can be used to determine the approximate location of the lander.

The 7075-aluminum impact stage will not be able to have redundancy incorporated. The construction of the frame will have a 20 percent safety margin and be thoroughly tested. The airbag system will be under the most demanding conditions and therefore, must have redundancies. The airbag system has six nitrogen gas generators that can cross-feed if one fails. The airbags themselves have multiple layers of Kevlar material with a 20 percent safety margin to ensure the rocks will not puncture the airbags. If a rock punctures an airbag, there are nine pockets that will be able to seal themselves to ensure the entire airbag does not deflate. The entire airbag system will follow a rigorous testing cycle to ensure they will not fail. The UDS will have six gas generators that can cross-feed in the event one fails. If a pneumatic actuator fails, the system is still expected to work with only four of the walls only folding down. The pneumatic actuators and gas generators will be tested thoroughly during the testing phase.

Overall, Fortitude, in its entirety, has been thoroughly equipped and prepared for the unfortunate instance of a subsystem failing at any time throughout EDL process.

3.3 Payload Integration

NavCam and the IR spectrometer are mounted on a mast, as can be seen in Figure 36. The mast has several joints to allow the spectrometer to swivel and tilt up and down. The way the mast is designed allows the NavCam to take pictures of samples before the IR spectrometer performs analysis, and provides a flexible, versatile way to image and analyze samples.

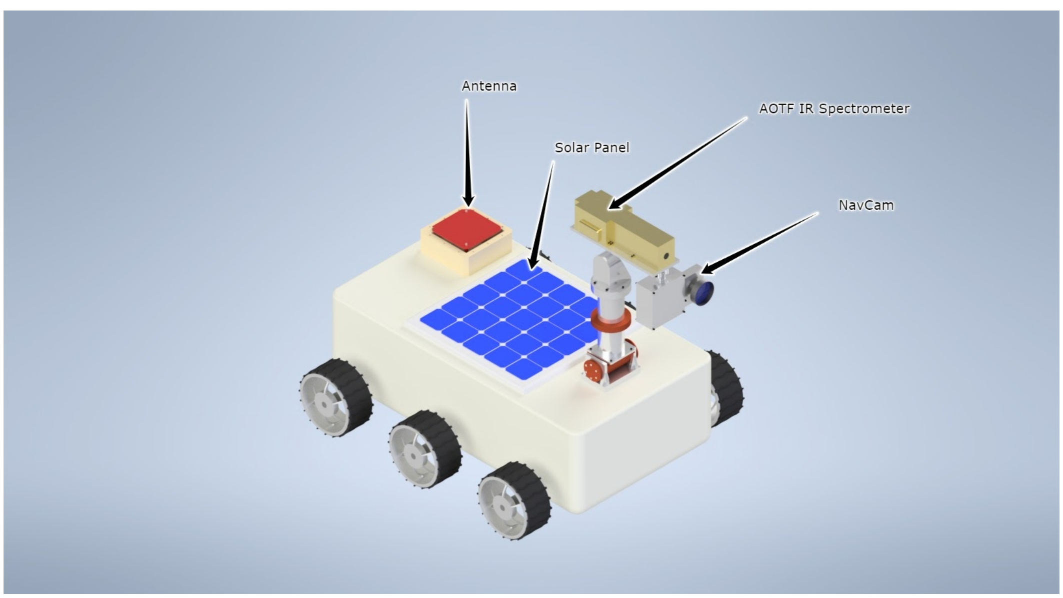


Figure 36: Estimation of the rover design

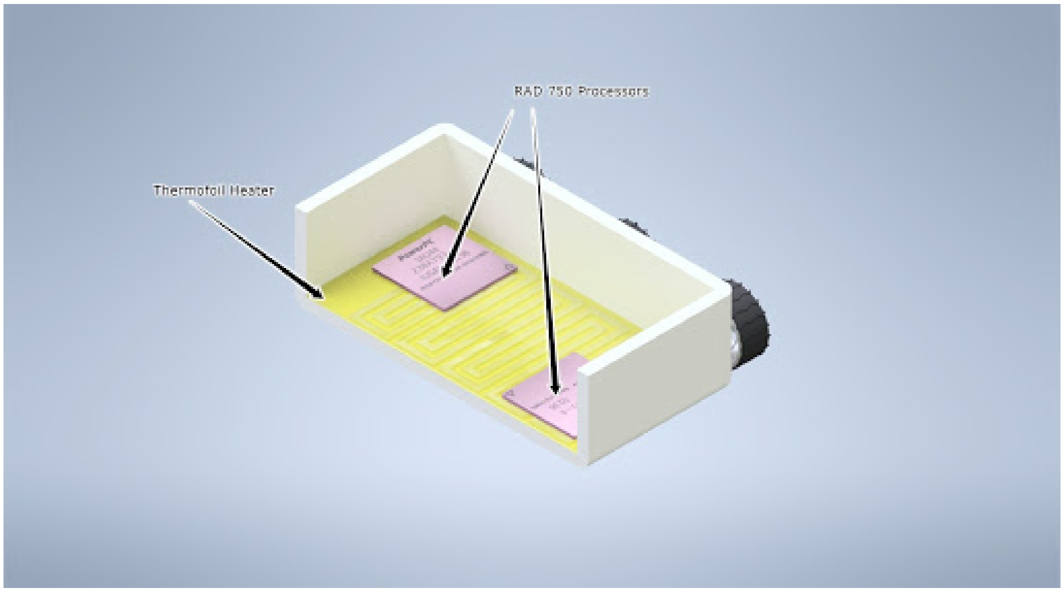


Figure 37: Inside the rover

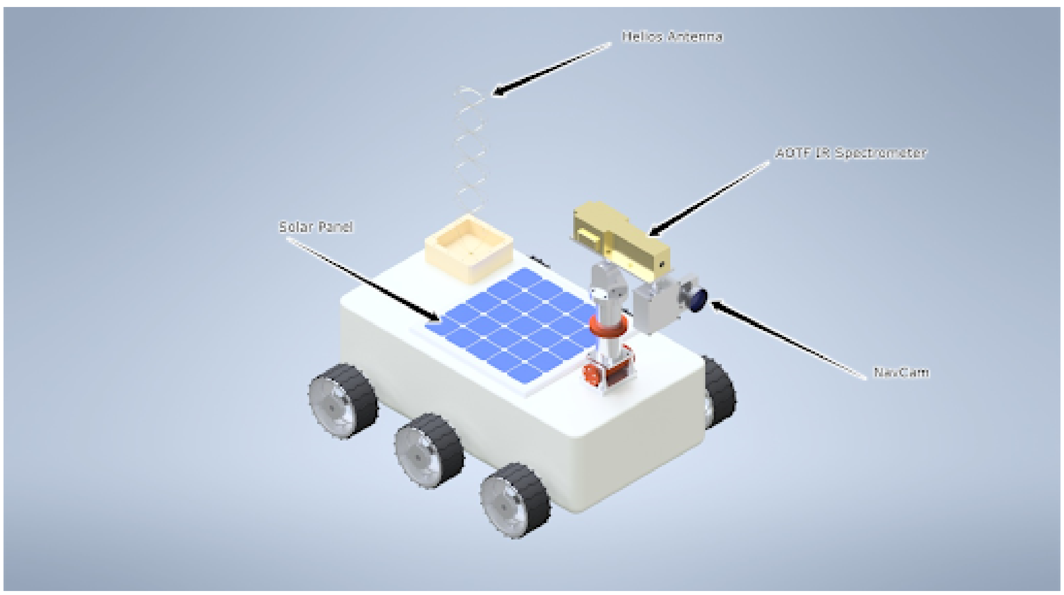


Figure 38: Rover with antenna deployed

As shown by Figures 36 and 37, the rover design is roughly simple. The mast containing the IR spectrometer and NavCam is at the front of the rover, with the deployable antenna at the back of the rover. In between the two would be an array of solar panels covering an area of about 6.2 square meters (scaled down for simplicity). One processor would be used to send and receive commands to the Fortitude rover, with the other acting as a redundancy.

The antenna will receive a deployment command from one of the RAD750 processors. During this command, the control system will tell the power system to allocate eight volts of direct current at seven amps to the antenna, which is the power required by the communications system to deploy the helical antenna. This power will be supplied for the entire duration of deployment spanning a rough time of 60 to 90 seconds. Once the antenna has been deployed, the power command will cease ("Helios Deployable Antenna").

The rover will be contained in the impact stage and will deploy once the lander settles down and unfolds.

4 Payload Design and Science Experiments

4.1 Selection, Design, and Verification

4.1.1 System Overview - N^2 Chart

The payload will have five main subsystems: Power, Thermal, Navigation, Data Acquisition, and Communications and Data Handling. Together, these subsystems allow the payload to explore the martian terrain, protect the instruments that carry out the mission's science objectives, and store and send data to the communications package, which transmits data back to Earth. The N^2 chart below summarizes how the subsystems interact with each other.

Thermal	- Heat to Maxon Motors		- Heat to Navcam - Heat to AOTF IR Spectrometer	- Heat to RAD750 - Heat to Helios Quadrifilar Antenna - Heat to Electronics
	Navigation			- Stereo Imaging
- 11.83 Watts for every 50g of Silicone Rubber Thermofoil Heaters	- 600 Watts (100 Watts for each motor)	**Power**	- 2.15 Watts to NavCam - 2.4 Watts to AOTF IR Spectrometer	- 1 W to Helios Quadrifilar Antenna
			Data Acquisition	- Raw Data from AOTF IR Spectrometer - Images from NavCam - Emergency Operation Commands
			- Image Acquisition and Processing Commands to NavCam - Data Acquisition Commands to AOTF IR Spectrometer -System Procedures	**Communication, Data Handling, and Control**

Table 8: N^2 chart

4.1.2 Subsystem Overview

The subsystems previously mentioned in section 4.1.1 are described in more detail below.

Power/Electrical

Working through each specific instrument and subsystem, the total power required was identified and summed to have a specific power requirement for the entire payload. A 30% power budget was included on top of the required value to ensure that all of the power demands would be met in the case of an emergency, as well as for any future power requirements during manufacturing. An iterative process was used to derive the values of different parameters involved in the design calculation of the electrical power subsystem (EPS). During the daytime on Mars, the rover will be powered by a solar panel with an estimated area of 6.1 m^2 (a worse case scenario estimation), providing enough power to carry out daily operations. During the night, when there is no access to sunlight, the rover will be powered by a Lithium-Ion battery to carry on mission work. The calculation estimates that the payload will have the power to last one martian year (687 days), which dictates the charge and discharge cycle of the battery. The iterative process used for the EPS found the mass of the battery to be 86.21 kilograms. A Lithium-Ion battery was chosen due to that specific type of battery being very energy-dense and allowing the team to minimize the payload's battery mass.

Eclipse Time on Mars (night time) (Te)	12.59 hrs
Daylight Time on Mars (Ts)	12.03 hrs
Power Output of the Solar Array (PSA - Solar Flux)	2318.4 W
Power Density Beginning of Life (PBOL)	416.64 W
Power Density End of Life (PEOL)	380.26 W
Desired Solar Array Size (ASA)	6.1 m^2
Number of Cycles	687 Days (1 Martian Year)
Depth of Discharge (DoD)	90%
Capacity of One Battery (Cr)	12930 W.hr
Battery Mass	86.21 kg

Table 9: Table of the design variables

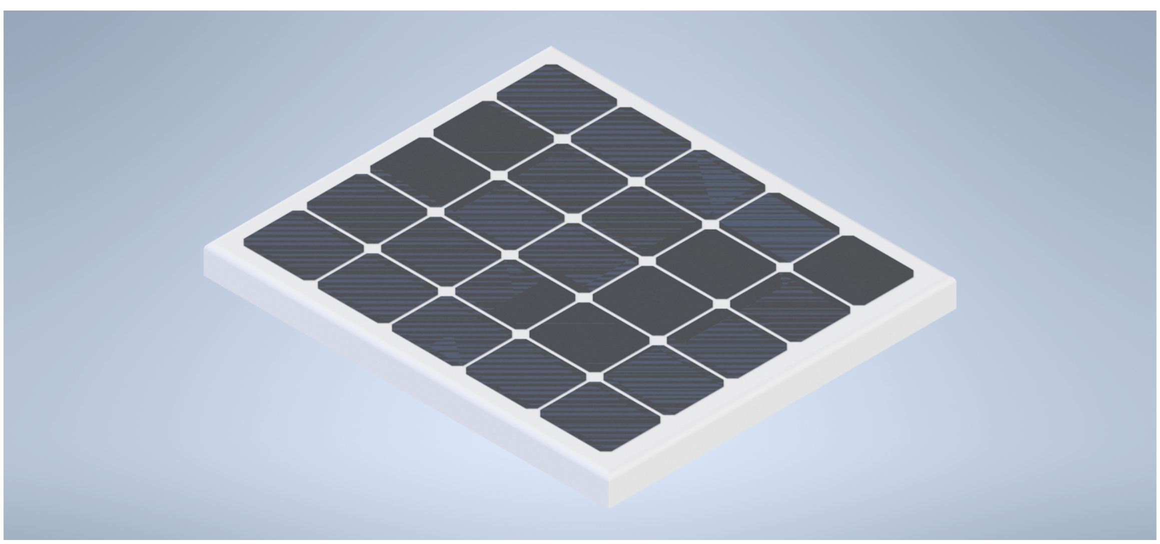


Figure 39: A sample of the solar panels used on the mission.

Thermal

The thermal system is important because it regulates the temperature of the different instruments and subsystems on-board the payload. The design process of the thermal subsystem starts by knowing the operating temperature limit of all the on-board subsystems, payload, and other structural elements. Then, the spacecraft attitude and orientation, relative to the sun, is predicted to set the boundary conditions for the mission.

The heater chosen for the thermal system will be a Silicone Rubber Thermofoil Heater, which is a type of resistive heater that will ensure that the temperature of the spacecraft does not reach its lower temperature limit. The heater is thin, versatile, and provides the necessary heating, while remaining operable at Mars temperatures. The heater is also very lightweight, typically weighing only 0.04 g/cm^2. The power required for the heater depends on the desired temperature change. The power requirement of these heaters was estimated to be 11.83 W to power 50 g of the silicone rubber covering an estimated area of 1,200 cm^2. This is enough power to raise the temperature of the spacecraft by 20°C in 60 seconds ("Flexible Heaters"). The heaters can be installed easily using Acrylic PSA, a NASA approved paint, requiring only applied pressure and the removal of a covering sticker, similar to a band-aid or double sided tape. They will be installed on the inside of the case that houses all of the components. The heaters will warm the air inside the components, which will raise the temperature of the instruments over time. Ideally, the heater would cover as much area as possible to generate the maximum heating effect.

To prevent the payload from reaching its maximum temperature limit, the team decided to use a thermal paint called AZ-93, which will reflect the solar rays and will help maintain the inner temperature of the spacecraft ("AZ Technology"). Another useful passive thermal control unit for heat rejection/circulation is CFC-11 or CFC-12 coolant loops ("Power for Mars"). By using loops similar to what is being used on the Perseverance rover, the payload will be able to manage excess heat more effectively. The heat rejection system works due to the CFC coolant that travels through the loop, decreasing the temperature of the container through contact/conduction. The system dissipates excess heat, maintaining the proper operating temperature of the payload. The operating temperatures of the instruments on-board the payload are summarized in the table below.

Instruments/Components	Minimum Operating Temp (°C)	Maximum Operating Temp (°C)
NavCam	-55	5
(AOTF) IR spectrometer	-50	40
RAD750 Processors	-55	125
Helios Quadrifilar Antenna	-40	85

Table 10: Operating temperature limits of the instruments

Navigation

The navigation system will allow the payload to traverse the unforgiving martian terrain. The payload must be able to navigate through river deltas, small crater impacts, nearby cliffs, and boulders present in the Jezero Crater landing site. The navigation system consists of wheels, which will travel over rough, rocky terrain as well as sandy terrain, suspension, which will support Fortitude over rough and uneven terrain, motors, to propel the rover, and a NavCam, to act as the eyes of the rover, viewing the martian landscape.

The wheels of the rover will be constructed of solid aluminum 7075 and will have cleats/treads for better traction. The wheels will have a diameter of 21 in and a width of 8 in. Six wheels are required to support the rocker-bogie suspension mechanism. Aluminum is tested and reliable and has been used in the past for rovers, such as Curiosity, and will be used for the Perseverance rover. Aluminum is chosen for the design because aluminum alloys are often used in spacecraft for structural and nonstructural applications because it can withstand degradation in a space environment (Muraca). This is important because the rover is expected to travel an estimated 15-30 km (depending on the landing zone and route of travel of the rover) and must survive the challenging terrain for a long period of time. Another justification for using aluminum 7075 is that it is cheaper than titanium with a cost of \$1.80/kg versus titanium's cost of \$16.25/kg (Props).

The suspension system design chosen for the payload will be a rocker-bogie suspension mechanism, which was also included on the Perseverance rover, and has been used in the past on the Curiosity rover, the Mars Exploration rovers, and the Pathfinder rover missions. This suspension mechanism is suitable for uneven and rough terrain due to its uniform distribution of weight over each of the rover's six wheels. This even weight distribution also reduces the rover tilt as it drives, making it less likely to tip over. The suspension system can support the rover as it travels over rocks as large as its wheels and the four wheel steering (back and front) allows for 360-degree rotation in place ("Wheels"). The rocker-bogie suspension mechanism is also ideal for traversing the martian surface because it supports the low speed travel of the payload (Wang).

The rocker-bogie suspension mechanism consists of six wheels. Four of those wheels connect to the bogie with two of those wheels acting as the middle wheels. The other two remaining wheels are attached to the rockers. The bogie system connects its four wheels to the rocker system. A bar connects the left and right rockers, known as the differential (Wang). The suspension system will be made out of aluminum 7075. The image below represents a typical rocker-bogie suspension mechanism.

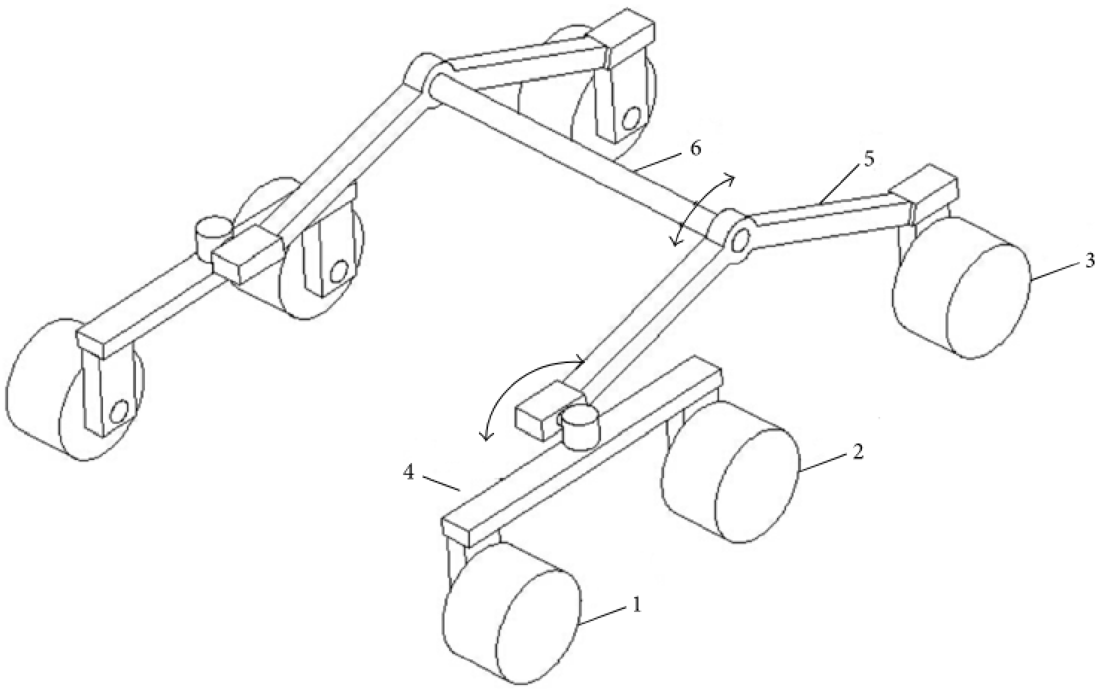


Figure 40: A skeleton of the typical rocker-bogie suspension mechanism

The motors selected for this mission are brushless direct current (BLDC) motors supplied by Maxon. NASA has used Maxon technology for the Mars Exploration Rovers and the Curiosity rover and will use it again for the Perseverance rover ("Once Again,"). Brushless motors were chosen due to their long life expectancy of 10,000 hours compared to brushed motors with a life expectancy of 2,000-5,000 hours ("The Emergence of Brushless Motors"). Brushless motors also have higher efficiency with Maxon motors claiming to have an efficiency of more than 90 percent ("5 Years"). Six motors will be required, one for each wheel and two motors for steering. Maxon modifies the motors to survive launch vibrations, as well as the harsh conditions on Mars, such as withstanding the atmosphere and temperatures between -120°C and 25°C. The specific motors used on the rover have a power rating of 100 W, a supply voltage of 24 V, a diameter of 68 mm, a width of 43 mm, with the ability to output a speed of 4250 rpm ("Maxon Brushless DC Motor"). A gearbox with an estimated 100:1 gear reduction ratio will propel the rover at a speed of roughly 4 cm/s, given a wheel diameter of 21 cm.

Data Acquisition

The data acquisition subsystem enables the payload to fulfil the mission duties by collecting data to be sent back to Earth. The two instruments that will be used to collect data on the payload are the Acoustic-Optical Tunable Filter Infrared (AOTF IR) spectrometer and the NavCam. NASA is using NavCams on the Perseverance rover, as well as the SuperCam, an instrument that contains a very similar IR spectrometer ("Rover Cameras").

The AOTF IR spectrometer was chosen to be the main data collecting instrument for the payload because it was determined to have the best collection capabilities to satisfy the mission objectives. The spectrometer has a functional wavelength of 1.6-3.6 microns and will be used to provide mineral signatures and collect data on pyroxene, olivine, carbonates, perchlorates, and smectites. The AOTF IR spectrometer is also ideal for this mission because it is very compact, rugged, qualified for space and very efficient and effective at low temperatures. The AOTF IR spectrometer will be mounted on the mast of the rover. The power consumption of the spectrometer is 2.4 W (Tawalbeh et al.).

The NavCam was chosen as the data collection instrument with the purpose of navigating Fortitude through the difficult terrain on Mars. The NavCam is a stereo navigation camera capable of providing panoramic, three-dimensional imagery with the use of visible light. The camera also has a field of view of 45 degrees and will be mounted on the mast of the rover, providing a detailed view of the terrain while acting as the rover's eyes ("The Rover's 'Eyes' and Other 'Senses'"). The NavCam includes a pixel scale of 0.82 microradians per pixel, it utilizes a 1024 by 1024 pixel detector with red and near IR bandpass filters centered around 650 nm, and it has stereo ranging out to 100 m (Maki). The camera's hyperfocal distance is 1.0 meters but is also capable of identifying an object as small as a golf ball from a distance of 25 m. The camera communicates with the processors on-board which calculate the number of wheel rotations given the desired distance travelled to allow for autonomous navigation ("Rover Cameras"). The NavCam acquires images using an image command for image acquisition and processing carried out by the RAD750 processor. The power consumption of the NavCam is 2.15 W (Maki).

Communications/Command and Data Handling

The communications subsystem allows for communication between the payload and a communications package which transmits data to and from the orbiter, which communicates with Earth. The communications package will be separately dropped to the surface of Mars in the local payload area and will bridge communication between the payload and the orbiter. The payload will include a quadrifilar helical antenna with an L band frequency range of 1290 to 1340 MHz. The antenna's main beam gain is 3dBi+, which is low enough to ensure transmission with the communications package despite the change in elevation that the payload will encounter on its journey. The antenna will be able to support transmission at a sufficient distance of roughly 30 km, a distance greater than the payload will be expected to travel ("Dive into Antenna Gain"). The antenna has a length of 100 mm, a width of 100 mm, and a stored height of 35 mm with a deployed height of 330 mm. The antenna uses 1 W of power to operate. In addition to the antenna, there is a transmitter package with a mass of 0.075 kg which requires 5 W of power. The chosen antenna will be sufficient for the transmission of data between the payload and communications package. It is also space rated by the United States Air Force and has undergone sufficient vibration and thermal testing conducted by the Air Force Institute of Technology at Wright-Patterson Air Force Base, Ohio ("Helios Deployable Antenna").

As the payload collects data, mainly from the AOTF IR spectrometer and the NavCam, it will transmit this data in real-time to the communications package to be transmitted to the orbiter which will be transmitted to Earth. The purpose of this is to reduce the risk of data loss if the payload becomes compromised. However, this may not always be possible because the payload may lose transmission with the communications package from time to time during its journey. When that happens, data will be stored on one of the payload's two computers, both of which are RAD750s. Once the payload can communicate with the communications package again, it will transmit the stored data.

In addition to transmitting data, the payload will also receive commands from Earth via the orbiter and communications package. The commands sent to the rover will be for the purpose of navigating through the rocky martian surface. The RAD750 is the brain of the rover and directs interfaces with all of the different components, including the instruments on the payload, to exchange the commands and science data. In addition to this, the command and data handling system monitors and reacts to any occurring problems on-board the payload, such as problems with the power, thermal, or other subsystems.

The RAD750 is an aerospace industry standard computer chosen for its high-reliability in spacecraft. The central processing unit is radiation-hardened and specifically designed to withstand the radiation environment of the Mars operation. In addition to this, the special hardware can withstand the vibrations experienced during launch, ascent and descent, and can survive in the vacuum of space. The processor also operates at a speed of 200 megahertz, 10 times faster than the RAD6000 used on the Spirit and Opportunity rovers. The computer is also equipped with 2 gigabytes of flash memory, 256 megabytes of dynamic random access memory (DRAM), and 256 kilobytes of electrically erasable programmable read-only memory (EEPROM) ("Rover Brains).

Flash memory is preferred over other memory types such as Hard Disk Drive memory because flash memory is non-volatile, memory dense, and has no moving parts, consuming less power and operating at a faster speed. The Flash memory will store data such as images and other scientific data until it can be transmitted back to Earth. DRAM is important because it is very fast and can read and write data to it, making it excellent for active rover operations. EEPROM memory is also very fast at reading data but unlike DRAM, it writes data very slow, and it is a non-volatile memory type, meaning that data stored will persist in the memory even when it receives no power. EEPROM will store and run the VxWorks real time operating system ("Types of Computer Memory").

VxWorks is a real-time operating system (RTOS) primarily used within space and defense/military applications and has been used by NASA for almost 25 years in dozens of missions, including the entry, descent, and landing operations of Curiosity ("Wind River's VxWorks"). VxWorks will be the operating system for the spacecraft control system performing mission critical tasks, ground operations control, data collection, as well as managing the antenna for the transmission of data. The operating system is extremely stable and can provide time-critical support, making it a good candidate for automatic spacecraft. VxWorks is a hard RTOS, meaning it must operate within the confines of a strict deadline and is specifically designed to prioritize performance and reliability. VxWorks is also a preemptive multitasking operating system, allocating a set time to a task before giving another task a turn to use the operating system. VxWorks is a reliable and secure RTOS and will be the operating system for the payload (Seo).

4.1.3 Precision of Instrumentation, Repeatability of Measurement, and Recovery System

The precision values for the AOTF IR spectrometer include a spectral resolution of 4-12 nm and a spatial resolution of 2 mm or greater (Korablev et al.). The precision values for the NavCam include a resolution of 1024 by 2048 pixels with each pixel being about the size of 144 square microns large. The NavCam also has an angular resolution of 0.82 microradians per pixel. Data collected within these parameters can be claimed as valid.

In order to increase the accuracy of the data collected by the AOTF IR spectrometer and NavCam, it is important to repeat measurements. For each sample of data collected by the spectrometer, three measurements will be taken. This is to ensure that the data collected is accurate and that enough data was collected to be studied. The NavCam will take at least three images of each chosen sample to ensure that there are multiple high quality images of the sample to be studied.

As data is acquired by the AOTF IR spectrometer and the NavCam, it will be automatically saved to the RAD750 computer and then transmitted. This is to ensure that data has been collected and captured. When it is able to, the payload will transmit the collected data to the communications package, which will then transmit it to the orbiter, which will transmit the data back to Earth to be analyzed and studied. The goal is to transmit the data as soon as possible in order to reduce the risk of data being lost if the payload becomes compromised. However, it is possible that sometimes the payload will lose the ability to communicate with the communications package from time to time. There will also be periods of time when the orbiter is out of range of the communications package. It is for these reasons that the data will also be saved and stored on-board the payload. Data will specifically be saved to the non-volatile flash memory on the RAD750 computer. This will ensure that the data is safe and secure, even when the payload powers down.

4.1.4 Validation and Verification Plan

In order to ensure the payload will work as expected on Mars, it is important to subject it to various testing in the validation and verification phase. During validation, the payload will endure tests that simulate the conditions that it will encounter during launch as well as the conditions of the space and the Mars environment. In order to verify that the instruments will work as intended on Mars, various checks for each instrument including calibration will be implemented and performed during testing in the laboratory and on Mars.

Validation

Validation begins by testing each instrument and subsystem prior to assembly on the payload body. Each subsystem is subjected to a functionality test and each instrument is calibrated before undergoing a comprehensive performance test. This is followed by a series of environmental qualification tests including electromagnetic interference and compatibility testing, vibration testing, and thermal vacuum testing. Preceding each test, instruments and equipment are aligned. After each test is completed, the instruments are again evaluated and measured to observe any changes that may have occurred during testing. When subsystem testing concludes, payload integration can begin ("Flight Systems and Integration Test").

The first step in integration is electrical harness integration during which wires, terminals, and connectors are assembled to link each component to the computer ("Wiring Harnesses"). This, along with the power electronic interface integration and command and data handling integration, will allow for power and information to be transferred throughout the vehicle. Finally, thermal components are integrated for the purpose of protecting the equipment from extreme temperatures ("Flight Systems and Integration Test"). Once instruments and subsystems are integrated, aligned, and calibrated, payload testing may begin.

The first test performed is functional testing during which instrumentation and subsystems undergo a series of comprehensive testing followed by deployment testing. The payload is subjected to another series of tests including electromagnetic interference compatibility testing, radio frequency testing, vibration testing, acoustics testing, and thermal balance and vacuum testing. Prior to each test, instruments will be aligned, and after each test, they will undergo a comprehensive performance test to ensure that everything will work when subjected to a Mars environment ("Flight Systems and Integration Test").

All testing will be completed in the NASA Jet Propulsion Laboratory's Environmental test facilities, specifically the Space Simulator Facility in Pasadena, California. The simulator is a stainless steel cylinder with a diameter of 8.23 m and a height of 26 m, large enough to test the fully assembled payload ("JPL'S Torture Chamber for Spacecraft"). The space simulator will produce true interplanetary conditions including extremely cold and hot temperatures, a high vacuum, and intense solar radiation. The tests experienced here by both the individual subsystems and instruments as well as the completely integrated payload are described in the sections that follow.

Dynamics Testing

In order to verify that the instruments on the payload can handle the conditions of launch, leaving Earth's atmosphere, and entering Mars' atmosphere, the payload must undergo acoustics and dynamics testing. During launch, it is possible that the spacecraft may be damaged by intense launch sounds, and for this reason, acoustics tests are conducted ("Mars Exploration Rover Mission"). In the acoustic noise chamber, the rover will experience 145 decibels of sound pressure to simulate launch conditions (Fisher). Next, the payload is prepared for the vibration test in the dynamics testing area.

The dynamics testing area includes four electro-dynamic vibration systems capable of shaking with forces anywhere from 90 kN to 265 kN. Dynamics testing for the payload will be conducted on a vibration table and on a large centrifuge. Individual subsystems, including the antenna, the suspension and wheels, the chassis and structural mechanisms, the electronics and processors, the NavCam, and the AOTF IR spectrometer, will be tested in random vibration environments. These tests will be performed on all three axes with different vibrations between 5.5 and 8.0 Grms (root-mean-square acceleration). Dynamics testing concludes with spin balance testing to determine the center of gravity and the moment of inertia of the payload (Fisher).

Thermal Testing

Thermal testing is conducted in the Space Simulator Facility to ensure that the payload can survive the atmospheric conditions of deep space and Mars. Component testing includes thermal vacuum tests performed on the electric motors, the chassis and structural mechanisms, the electronics and processors, the NavCam, the AOTF IR spectrometer. The first thermal vacuum test simulates Mars surface conditions with a pressure of 8 Torr and a CO_2 rich environment ("Red Planet Rover"). Next, the payload will endure a thermal blanket bake-out, remaining in the thermal chamber for a total of 50 hours, experiencing a temperature 110°C and high vacuum conditions. The third thermal test simulates deep space with high vacuum levels of 10^{-6} Torr and a temperature of -185°C achieved with liquid nitrogen. The final test simulates a day and night on Mars with a pressure of 8-10 Torr and temperature values maintained between -130°C and 20°C (Fisher).

Solar Testing

The solar simulation will be conducted in the Space Simulator and will mimic the sunlight and solar radiation conditions experienced on Mars. The sparse Mars sunlight will be simulated by multiple giant light panels, and the solar simulation will produce various beam sizes and intensities from an array of 37 xenon arc lamps with a power range of 20-30 kW ("JPL'S Torture Chamber for Spacecraft). The payload will experience the solar flux density on Mars at 590 W/m^2 over a 4 m wide hexagonal area (Novak, "Sunlight on Mars"). The solar simulator will imitate the

Mars sunlight to test the solar panels and verify that it can generate enough power for the rover.

Electromagnetic Interference/Electromagnetic Compatibility Testing

Electromagnetic Interference/Electromagnetic Compatibility (EMI/EMC) testing is conducted to ensure that the electronics don't interfere with each other. The three test objectives of EMI/EMC testing include testing for radiated emissions, radiated susceptibility, and self-compatibility. The radiated emissions test verifies that the electromagnetic energy emissions are within the specified limit of the spacecraft. The radiated susceptibility test is the response the payload has to external electric fields that it may experience. This test verifies that the spacecraft will respond correctly when exposed to the launch and space operations environments. The final test for self-compatibility verifies that there is no electrical interference within the spacecraft and ensures that the communications system does not experience degradation (Lauretta). This concludes EMI/EMI testing.

Mobility Assembly Testing

In order to validate successful payload mobility, the payload will undergo a series of tests on a tilt table known as the Variable Terrain Tilt Platform (VTTP). The VTTP, located at JPL, can simulate many different martian terrains that the payload may encounter.

The payload will drive across the 16-square-foot platform at angles between 0 degrees and 25 degrees, going up-slope, down-slope, at 45 degrees from up-slope, down-slope, and cross slope. These sloped-drive tests will be carried out with three different terrain substrates: high friction material bottom, 6-inch deep coarse gravel, and 6-inch deep dry sand. The three substrates help to simulate the variety of rock types found in Jezero.

After these tests, concrete obstacles will be placed in front of both sides, as well as either side, of the rover in similar configurations as what was described above. These obstacles simulate larger rock structures that the rover may encounter. The obstacles range in size from 10 to 21 inches in height. Being able to overcome these obstacles further validates that the rover will be able to withstand the atmosphere in Jezero Crater.

A final round of tests will be performed in conditions similar to the first round of tests (elevation angle, substrate, angle of attack), but the rover will perform a 360-degree turn. This final round of tests will validate that the rover can handle a full range of motion at various angles.

In these three rounds of tests, data will be collected at every degree change of the platform. The data collected will include wheel slippage, rover slip, rover side slip, currents of the drive motors, and wheel speeds. Collecting this data will provide information on whether the rover can provide enough torque, speed, friction, and power to traverse along its path through the difficult terrain.

<u>Verification</u>

In order to ensure that the NavCam and the AOTF IR spectrometer will work on Mars, each instrument must be subjected to their respective verification test. For the NavCam, a series of imaging tests will be carried out in order to ensure the camera is working properly. For the AOTF IR spectrometer, verification includes confirmation signals which will be sent back to Earth.

The first test carried out will be a command for the NavCam to take two panoramic images, one facing Fortitude and one facing the horizon in front of Fortitude. Once the commands are completed, the computer will compare the panoramic images to similar ones taken on Earth. This test will verify that the NavCam is capable of taking panoramic images on Mars and that the mast can rotate 360 degrees. Next, the payload will be given commands to travel a distance from the landing site of half of a meter, one meter, five meters, and eight meters, taking images of the landing site at each location. The resolution of these images will then be compared by the computer to similar images taken by the NavCam back on Earth. This test will verify that the NavCam can take images with the expected resolution.

Verification of the AOTF IR spectrometer will include signals sent from Mars back to Earth to let the scientists and engineers know that the instrument is working. When the spectrometer is in use, the mast will have to move into position for the instrument to take measurements. When this happens, the computer will be signalled that the spectrometer is in use and will send this information back to Earth. The NavCam will also take an image of the sample being analyzed by the spectrometer. This image along with the data collected by the spectrometer will also be sent back to Earth to verify that the spectrometer is being used.

This verification system will ensure that the instruments on-board the payload have completed their tasks and are communicating with Earth.

4.1.5 FMEA and Risk Mitigation

- **Occurrence:** Occurrence Scale (1-10) with 1 being highly unlikely and 10 being almost certain.
- **Severity:** Severity Scale (1 -10) with 1 being not noticed by a customer and 10 being hazardous or life-threatening and could place the product survival at risk.
- **Detection:** Detection Scale (1-10) with 1 being almost certain to detect and 10 being almost impossible.
- **RPN:** Risk Priority Number is calculated by multiplying the Occurrence x Severity x Detection.

SEVERITY											
	10	D		C							
	9										
	8			B F							
	7			A E							
	6										
	5										
	4										
	3										
	2										
	1										
		1	2	3	4	5	6	7	8	9	10
	OCCURRENCE										

Table 11: FMEA Criticality Matrix - payload

High
Med
Low

Table 12: Criticality key

Function(s)	Failure Mode(s)	Effects(s)	SEV	Cause(s)	OCC	Design Controls (Prevention)	Design Controls (Detection)
A. NavCam Safe navigation of rover, particularly when the rover operates autonomously, making its own navigation decisions without consulting controllers on Earth	Total or partial loss of functionality to aid in navigation of the rover safely on Mars	Unsuccessfully performs aid in autonomous navigation	7	Atmospheric conditions of deep space and harsh martian environment	3	Perform thermal vacuum test on NavCam	First part of test on startup
B. AOTF IR spectrometer Study of martian surface composition in the vicinity of a rover	Loss of functionality to perform the study of martian surface composition	The surface of Mars will not be successfully studied	8	Atmospheric conditions of deep space and harsh martian environment	3	Perform thermal vacuum test on AOTF IR spectrometer	First part of test on startup
C. Electronics and Processors Ensure the functionality of the rover	Total or partial loss of rover's functionality	The mission might fail or the objectives will be partly accomplished	10	Atmospheric conditions of deep space and harsh martian environment, Interference of Electronics with one another	3	Perform thermal vacuum test, Electromagnetic Interference/ Electromagnetic Compatibility Testing on rover's electronics and processors	First part of test on startup
D. Solar Panels Production of power for the rover	Failure to power the rover for the entire mission.	The mission would fail	10	Not enough energy would be produced in the solar flux density of martian environment	1	Perform solar testing by simulating the sunlight and solar radiation conditions experienced on Mars	First part of test on startup

Table 13: FMEA Criticality Matrix descriptions (A-D) - payload

Function(s)	Failure Mode(s)	Effects(s)	SEV	Cause(s)	OCC	Design Controls (Prevention)	Design Controls (Detection)
E. Payload Mobility Ensure that the rover can handle a full range of motion at various angles on the martian surface	The rover would not be able to provide enough torque, speed, friction, and power to traverse along its path through the difficult terrain	Lacking a full range of motion on the surface of Mars could lead to limitations when performing certain maneuvers to collect samples or put the rover at risk when changing its path because of atmospheric conditions	7	Rover would not be able to withstand the harsh martian surface	3	Perform Mobility Assembly Testing simulating many martian terrains that rover may encounter	First part of test on startup
F. Payload NavCams, AOTF IR spectrometer, Electronics and processors, Solar panels	Not being able to handle the conditions of launch, leaving Earth's atmosphere and entering Mars' atmosphere successfully	Not being able to overcome the dynamics testing could affect the success of the mission in a critical way	8	The payload and the listed subsystems/ components would not be able to withstand the vibration due to forces from 90 kN to 265 kN	3	Perform dynamics testing on the payload and the listed subsystems/ components	First part of test on startup

Table 14: FMEA Criticality Matrix descriptions (E & F) - payload

D E T	R P N	Recommended Action(s)
3	63	Carefully analyze the data after thermal vacuum test has been performed
3	72	Carefully analyze the data after thermal vacuum test has been performed
4	120	Carefully analyze the data after thermal vacuum test and Electromagnetic Interference / Electromagnetic Compatibility Testing have been performed
1	10	Carefully analyze the data after solar testing has been performed
3	63	Carefully analyze the data after Mobility Assembly Testing has been performed
2	48	Carefully analyze the data after the dynamics testing has been performed

Table 15: Recommended actions chart - payload

4.1.6 Performance Characteristics

The average temperature is about -60°C, although it can vary from -125°C, near the poles during the winter, to as much as a comfortable 20°C, at midday near the equator. While most of the equipment on the rover is already capable of withstanding temperatures as low as -50°C, we rely on Silicone Rubber Thermofoil Heaters from TUTCO Heating Solutions to maintain the rover at a temperature where all of the instruments can operate efficiently. Additionally, our intended landing site, Jezero, is north of the martian equator, to avoid the extreme, unforgiving conditions that await the rover at either one of its poles or in the southern hemisphere.

One problematic weather condition would be the martian sand storms. Dust is a permanent feature of Mars' atmosphere. Martian dust storms are the largest in the solar system and are capable of blanketing the entire planet. During winter, the temperatures in the polar regions are cold enough for the CO_2 in the atmosphere to condense into ice on the surface. Then, CO_2 sublimates off the ice cap in the spring and summer, returning to the atmosphere. One of the Mars Exploration Rovers fell victim to a storm in 2004. Despite noticing small signs, such as sudden temperature drops, NASA scientists are still unable to accurately predict martian dust storms. To battle light gusts of wind carrying dust, the solar panels on Fortitude track the sun. They are aligned at steep angles twice daily and allow for dust to fall off of the panels. A possible way to combat this weather would be to use nuclear power instead of solar power while on the martian planet.

A second problematic weather condition would be martian snow. The martian snowflakes are made of CO_2, instead of water, and are small particles that create a fog effect rather than appearing as snow. A relatively simple and agreeable tactic to tackle martian snow is to stay on martian soil, since Mars' atmosphere is not thick enough to allow martian snow to hit the ground.

The AOTF IR spectrometer is deemed as especially trustworthy because it can operate under the cold and sandy conditions because it operates using infrared light to detect minerals in the soil. Even in a worst-case scenario, where Fortitude gets caught in a sandstorm and has a fogged lens, the IR spectrometer will still be able to detect what minerals are rendering the rover immobile.

4.2 Science Value

4.2.1 Science Payload Objectives

The science objectives are:

- Objective 1: Does the area have a variety of rocks and "soils", including those from an ancient time when Mars could have supported life?
- Objective 2: Are the rock types at the site able to preserve physical, chemical, mineral, or molecular signs of past life?

These are supported by the team's instrument choice of the IR spectrometer. Its ability to detect carbonates and life-significant minerals apply to these objectives and the team's focus within those objectives of finding carbonates. The data the team gathers which answering these objectives will help determine indirectly whether there are or have been signs of life. These include determining if there are a variety of rocks and soils, especially from when Mars could have supported life, and if that geology could have preserved signs of past life.

These objectives were chosen because they best fit the inspiration from the Viking and Astrobiology Field Laboratory missions, which this mission is inspired by. The goals of Viking and AFL were to find signs of existing life on Mars and classifying areas that are of astrobiological interest and potential on Mars, assessing habitability through analyzing the possibility of prebiotic chemistry and finding biomarkers and organic compounds. Given the constraints of the mission, the team was not able to go to the same extent as these previous missions. However, the data gathered could build the foundation of knowledge for future missions to continue exploring and building upon.

The data expected from Jezero includes readings on carbonate signatures, perchlorate signatures, OH bond stretching, pyroxene signatures, and olivine signatures. The AOTF IR spectrometer is expected to find these signatures in the areas of scientific investigation.

This data gathered from Jezero could determine whether life could have been preserved in the area and whether water is or was present in the area. Considering Earth, where the presence of water suggests life, if signs of water are found on Mars, the same implication can be made. Future missions would have to investigate data that this mission finds more closely to confirm or deny these implications.

4.2.2 Creativity/Originality and Uniqueness/Significance

When comparing the four possible landing sites, Jezero Crater (18.408N, 77.687E) stood out amongst the others because of its high scientific value. It was also deemed to be the most dangerous when considering the rough terrain within its approximate 1770 km^2 area. In the end, the rewards outweigh the risks of the mission, and Jezero Crater was selected as the location of investigation.

Jezero Crater is believed to be rich in organic molecules, clay minerals, and carbonates, all of which would help better understand the martian surface and its potential to preserve life in the past or present. It is believed that the crater was "once home to an ancient river delta" that "could have collected and preserved ancient organic molecules and other potential signs of microbial life from the water and sediments that flowed into the crater billions of years ago" (Brown and Wendel). This concentrated collection of materials collected from other nearby areas can prove to be high in potential for ground breaking discoveries, "the material carried into the delta from a large watershed may contain a wide variety of minerals from inside and outside the crater" (Brown and Wendel).

There are two fan deposits that show a potential for carbonate analysis. These fan deposits are located in the northern and western parts of Jezero, "the northern fan is spectrally characterized by a mixture of Mg-rich carbonate and olivine, while the western fan is characterized by Fe/Mg-smectite" containing "variable amounts of Mg-rich carbonate and olivine in isolated exposures" (Goudge et al.). In order to pinpoint the location of scientific interest and analysis, it was decided that the northwestern "corner" and southern rim of Jezero were the most promising to carry out the mission objectives.

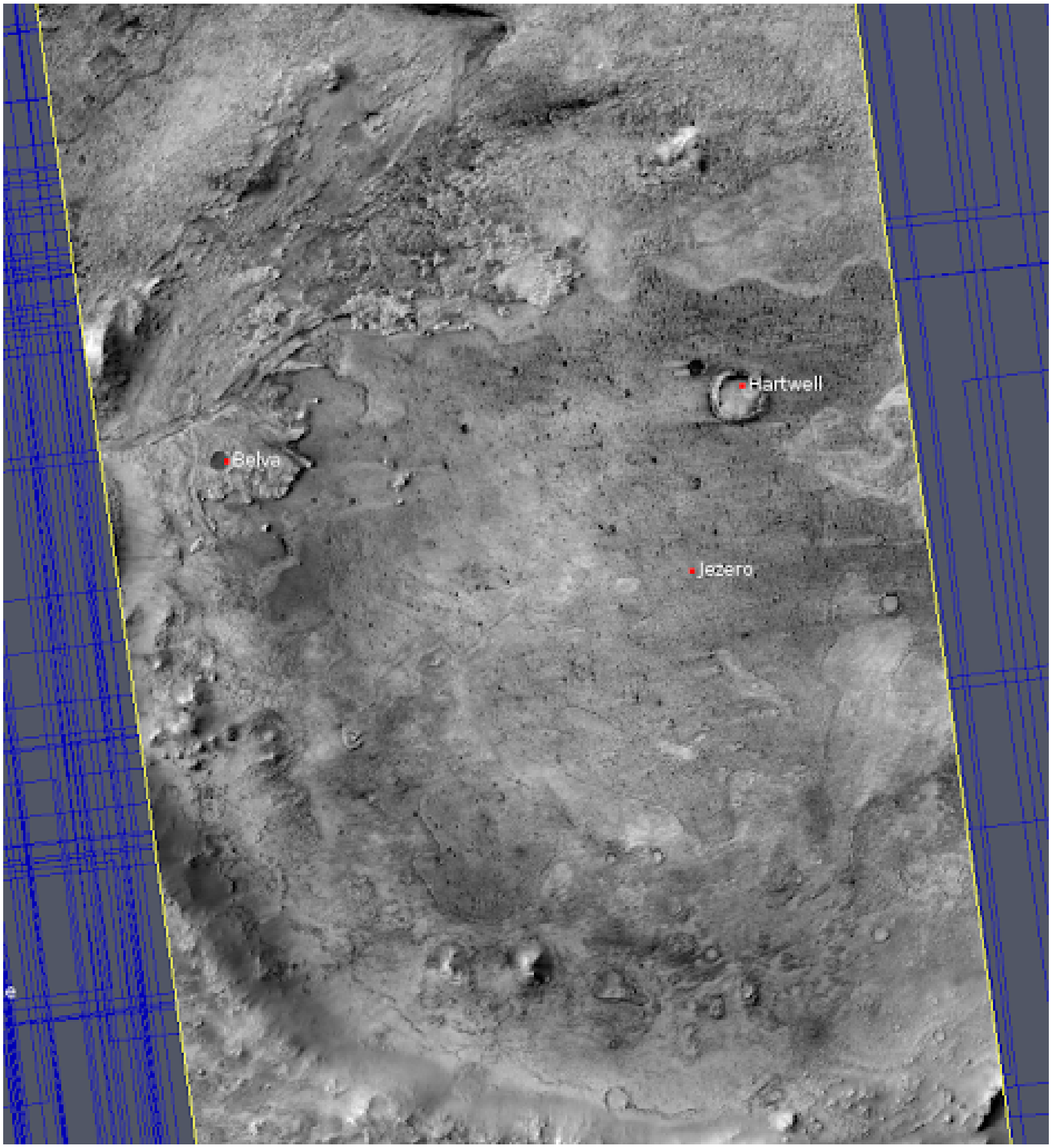


Figure 41: Macroscopic view of mission area

Figure 41 displays the proposed mission area. The following images were taken using the HiRISE DTM Stamps (Christensen, P.R.et al.). The area of scientific interest is the northwestern delta (18.486N, 77.36E) shown near Belva Crater. Figure 42 shown below details the primary mission area, including the proposed 8 km landing site (in green), and the two sites of investigation (in purple and yellow).

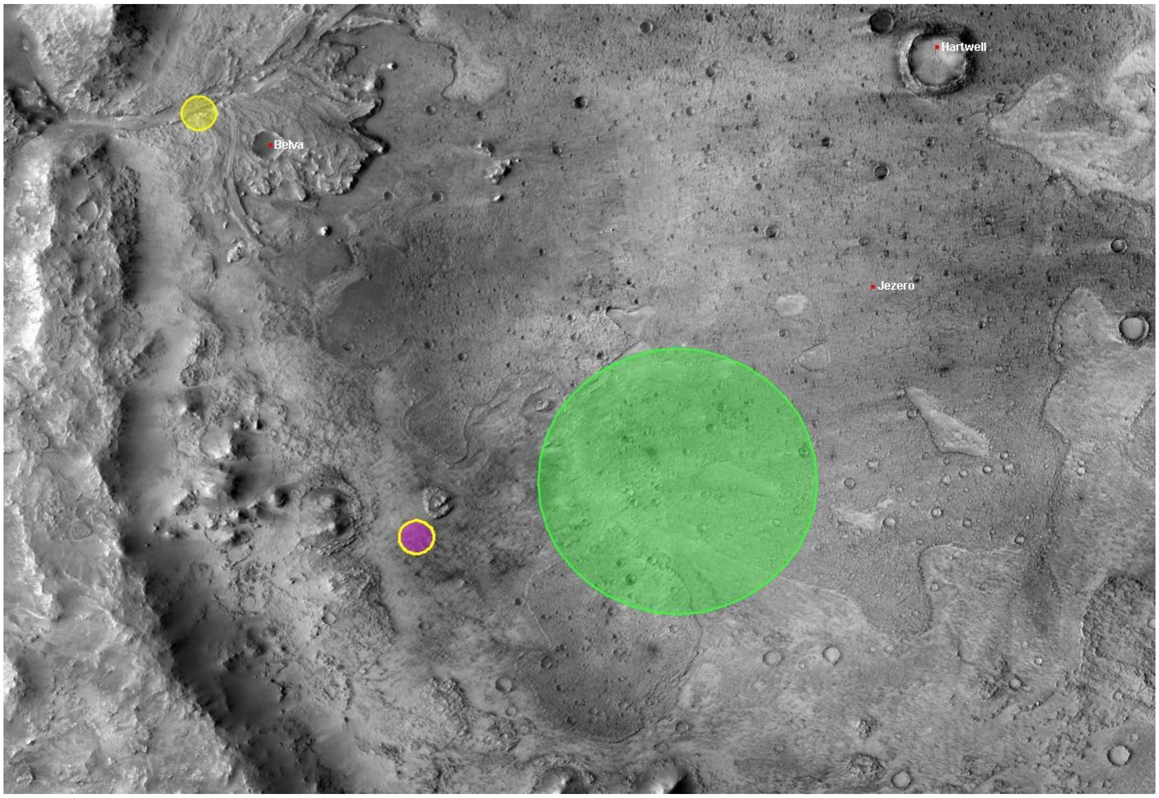


Figure 42: Proposed landing and investigation locations

The first area of investigation (in purple), is located at (18.28N, 77.45E) at an altitude of about -2,630 m. This investigation area is 1 km in diameter and is believed to have a 0.02041 abundance level of high and low calcium pyroxene, using the OMEGA Band depth at 2000 nm layer (Christensen, P.R.et al.). The areas in the crater that are believed to contain this mineral appear in the blue squares in Figure 43 below. The lighter the shade of blue, the more abundant the mineral is in that area. This first investigation site also shows signs of olivine. The JMARS layer used (OMEGA Olivine 1360nm Band, Ody et al 2012) described the data in Figure 44 as having a preference towards "olivine with a large Fe content and/or with large grain size and/or with high abundance"(Christensen, P.R.et al.). The data in this figure indicates a level of about 0.97305 abundance. There also appears to be a 1.04043 abundance of olivine with a high iron content or large grain size displayed in Figure 45, further solidifying the belief that high iron content olivine can be found in this first investigation site. This was found using the OMEGA Olivine, Fe or Coarse - Ody et al 2012 JMARS layer (Christensen, P.R.et al.).

This first investigation site is also home to a low abundance level of ferric oxide, "mineral phase of iron", found using the JMARS OMEGA Ferric Oxide Map layer, as well as 0.06532 areal fraction abundance of "carbonate mineral endmembers" - using the TES Carbonate Abundance (Bandfield, 2002) + 1 plots JMARS layer. (Christensen, P.R.et al.). It is theorized that the abundance levels of pyroxene, olivine, ferric oxide, and carbonates located here will be able to provide a substantial amount of information that will ultimately lead to completing the two mission objectives.

The second investigation site (in yellow), also 1 km in diameter, is centered at (18.49N, 77.34E) at an altitude of about -2,395 m. This investigation site was chosen to be centered on the delta because the belief that this area was once a river delta that flooded the crater with many clay minerals and organic matter from areas outside of the crater. Focusing the second investigation site to this delta will help to prove or disprove this theory. It can also be seen from Figure 43 that there is a section of the investigation area that has pyroxene minerals in abundance of 0.02042 using the OMEGA Band depth at 2000 nm, Ody et al 2012 JMARS layer (Christensen, P.R.et al.). Figure 44 indicates a section of the investigation area contains olivine with 0.97912 abundance. This area also contains a low level abundance of ferric oxide, which can be seen using the OMEGA Ferric Oxide Map JMARS layer, as well as a similar abundance level of carbonates as the first site of 0.06532 areal fraction abundance (Christensen, P.R.et al.).

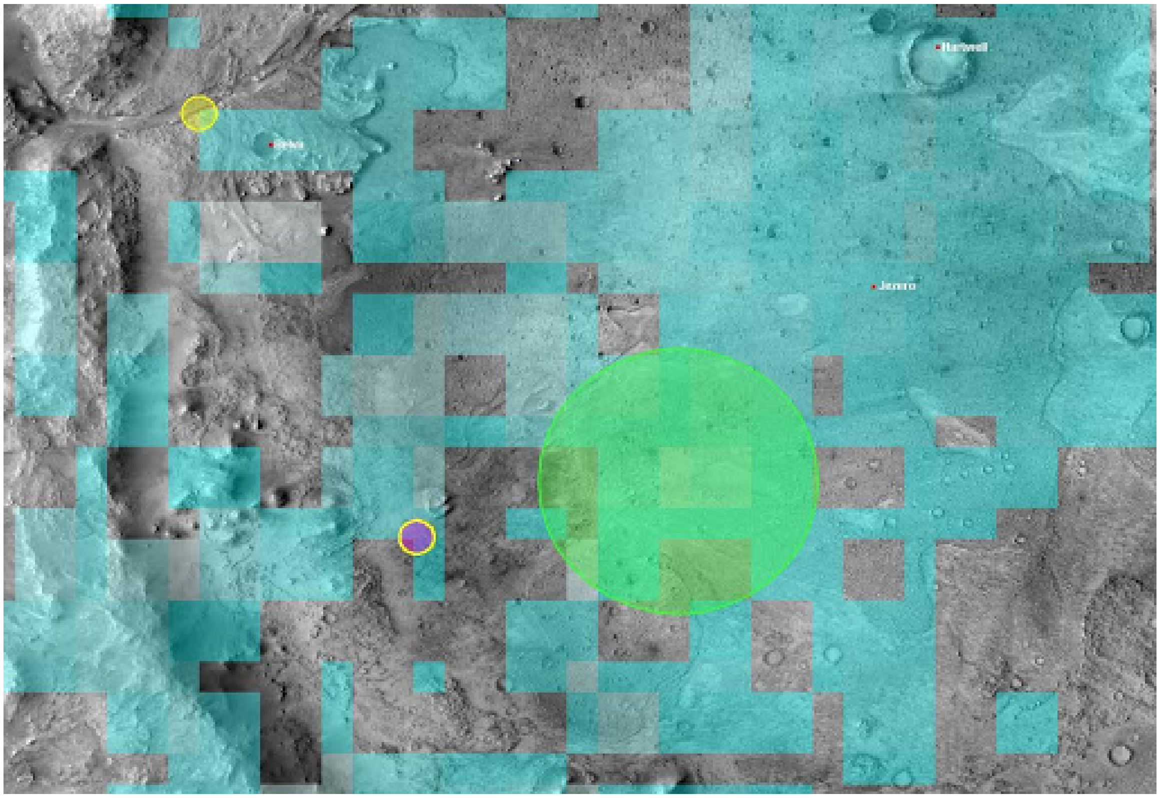


Figure 43: High and low level pyroxene abundance

This mission stands apart from previous missions by going straight to the areas with the greatest possibilities of providing evidence of past/present life, and efficiently determining their molecular composition with the IR spectrometer analysis.

When considering Jezero as the landing site, the team was well aware of the potential hazards in the terrain. Jezero Crater is home to many obstacles, including river deltas, small crater impacts, nearby cliffs, and boulders (Grossman). If Fortitude cannot maneuver its way through the terrain, the science cannot be completed. Figure 46 below demonstrates the many craters within Jezero Crater.

The individual craters are demonstrated by the red squares (Christensen, P.R.et al.). It is clear that navigating through this terrain poses many threats. The northern and eastern parts of Jezero Crater proved to be too dangerous for landing or navigating. The areas of interest are located in the northwestern and southwestern parts of the crater. The landing site was chosen to be towards the southern part of Jezero because it contained the least amount of craters. The landing location is at an altitude of about -2672 m.

In order to land within this proposed area, reference features (blue) will be identified by the lander to calculate its trajectory. Using the Custom Shape Layer in JMARS, Figure 47 shows the reference features to be used by the lander when descending to its landing location.

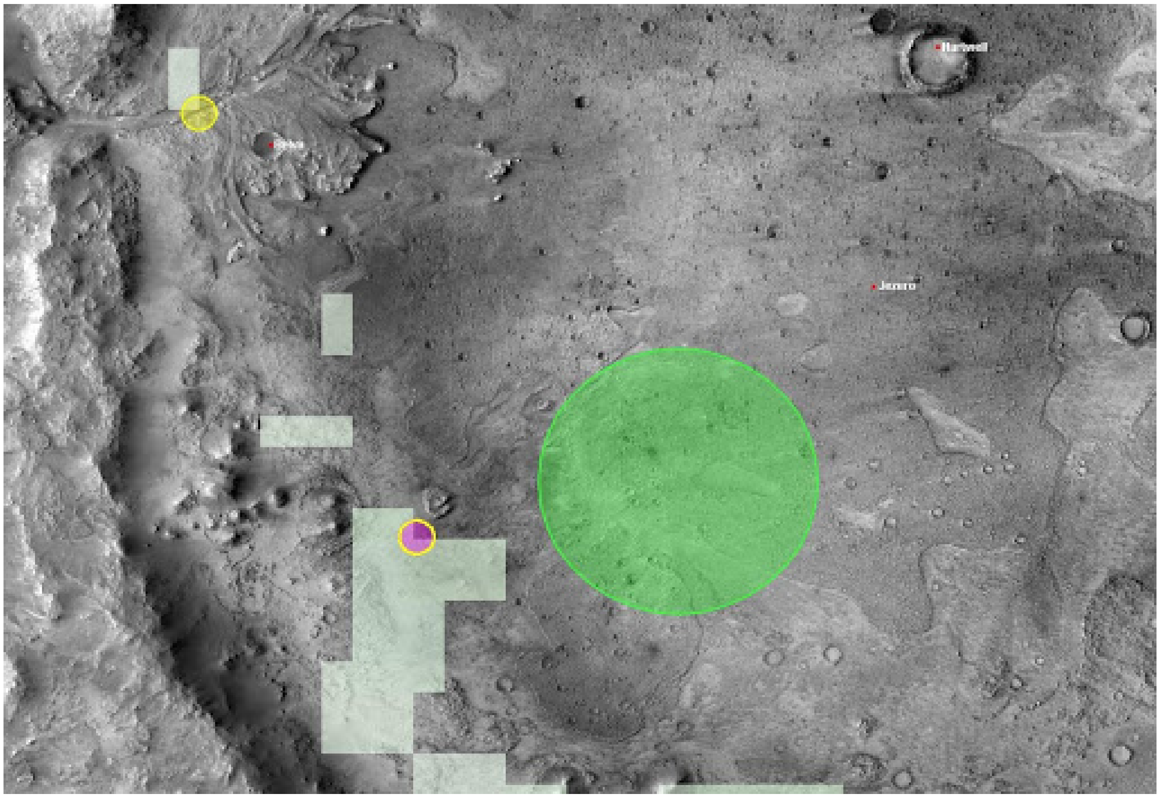


Figure 44: High iron content olivine abundance layer 1

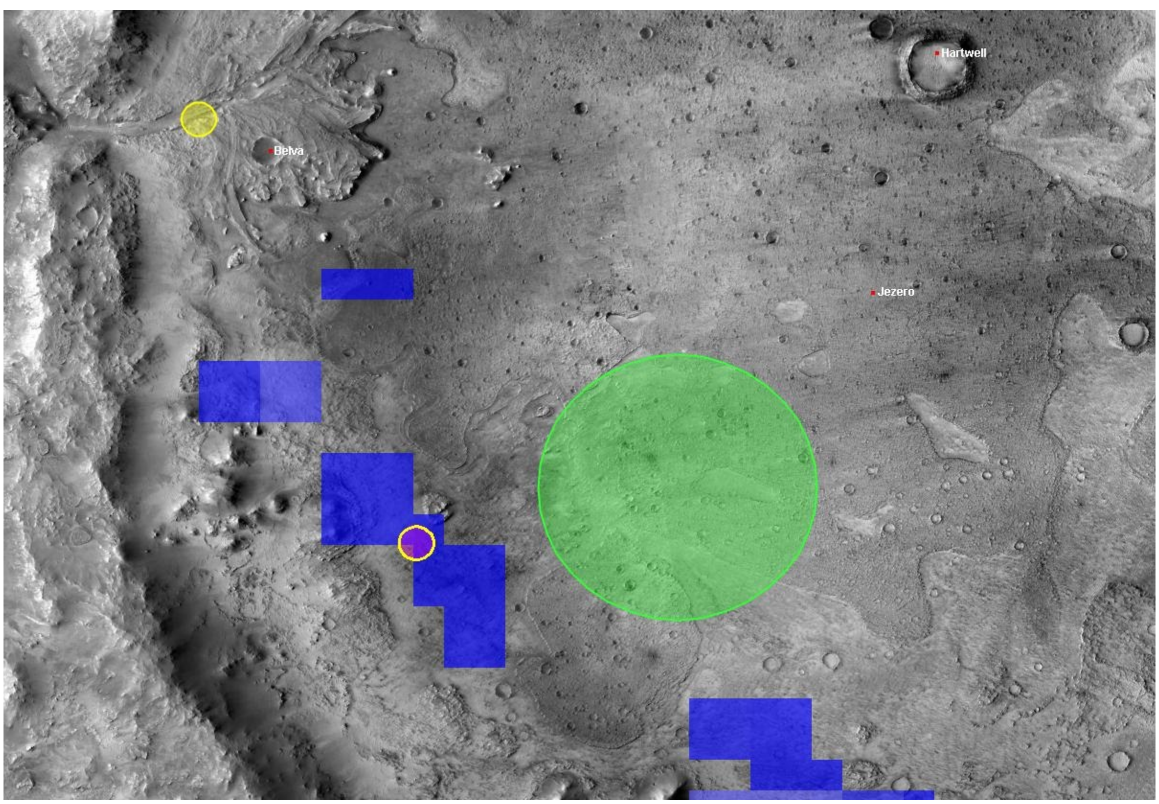


Figure 45: High iron content olivine abundance layer 2

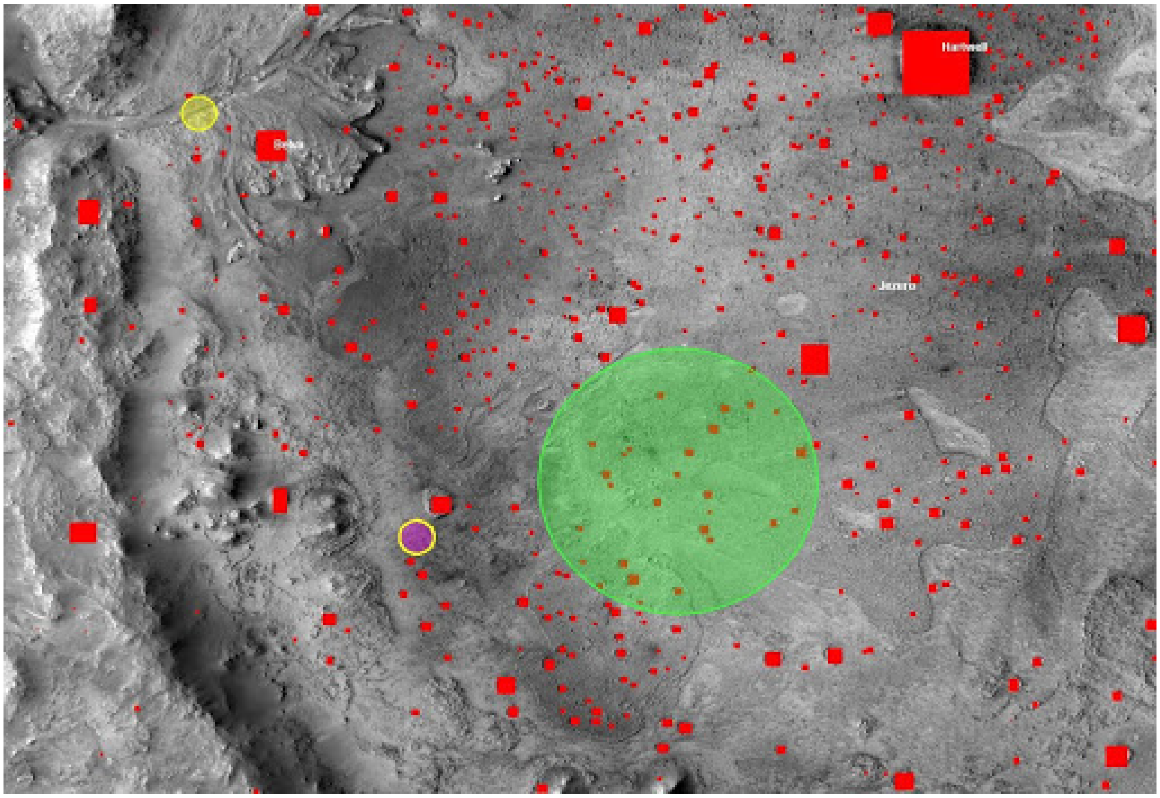


Figure 46: Small craters within Jezero Crater

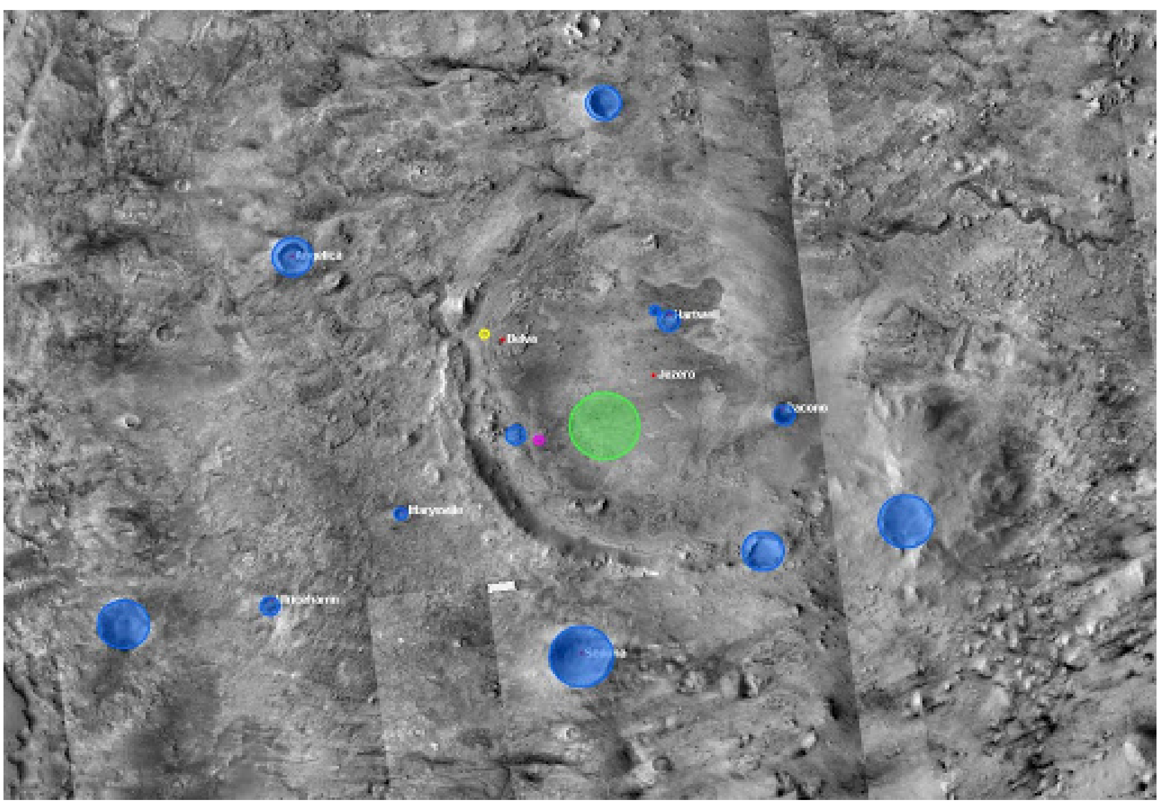


Figure 47: Landing area (green), investigation areas (pink & yellow), reference features (blue)

This design is unique in that it will be actively looking for biosignatures of past or present life. The inspiration behind this mission is the Viking mission, which was looking for signs of extant life on Mars with various biological experiments. The experiments done were gas chromatograph mass spectrometry to determine molecular weight, gas exchange to determine if microbial metabolism occurs in the soil when exposed to nutrient solution, the labelled release experiment (inoculating a sample of the martian soil with water and a nutrient solution containing glucose with carbon 14 to detect if the solution was metabolized into carbon dioxide) and the pyrolytic release experiment used to test for photosynthetic organisms (Williams). It also took atmospheric data and substantially imaged the surface (about 16,000 images were returned) ("Historical Log — Missions – NASA'S Mars Exploration Program"). There is also some inspiration taken from the cut Astrobiology Field Laboratory mission, which had the goal of classifying areas that are of astrobiological interest and potential on Mars, assessing habitability through analyzing available energy sources, the possibility of prebiotic chemistry, and classifying/finding biomarkers and organic compounds by "following the water" and "finding the carbon" (Beegle et al.). Although the team's goals align with the goals of previous and current NASA missions and although the team's resources are not as expansive as that used by Viking and other missions, the instrumentation and implementation of these goals is unique. Taking the components from Perseverance and Curiosity that have proven to be successful and the ideas from Viking and AFL to make something new, the mission is to find microbial life on Mars and test for its presence, to allow other present and future NASA missions to investigate further into the data the mission has collected. In a way, it's a sort of "survey" mission or precursor/junior mission to Perseverance. It gathers preliminary data before more sophisticated, larger, and capable missions can further investigate the data found at Jezero Crater.

Having some of the components Perseverance has gives the team an opportunity to measure samples and image features Perseverance might not be able to or cannot encounter. Fortitude's compact size makes the rover better able to navigate through Jezero Crater and get into hard-to reach locations of interest that might not be accessible to Perseverance. The choice of mass spectroscopy gives the team thorough detail of the samples collected on a molecular level. Detections of carbonates and other minerals could give an idea as to how long and to what extent water may have existed in Jezero, whether it was warm enough to be kept in a liquid state, and if this environment fostered the evolution of primitive microbial life.

Given that life tends to leave biomarkers and biosignatures in the geology of the area but that these may be destroyed by environmental processes, detecting these biomarkers and giving context to the data received from the samples will prove crucial in answering the science objectives (Beegle et al.). The IR mass spectrometer will be key to providing context and insight into the geology of Jezero Crater for further investigation by more sophisticated Mars missions. In characterizing the environment from which organic molecules and minerals are discovered, the team can learn more about the evolution and preservation of past potential life on Mars, as these sights would stand out against the harsh and barren martian environment and (as typically conceived) sterilized surface, and will benefit future investigations on Mars because the preliminary data the mission will gather on Jezero will help future and concurrent missions in their goals to determine if Mars had or has life.

4.2.3 Payload Success Criteria

Analyze at least 2 samples: one of an area within Jezero Crater that might have potentially preserved signs of life (such as by the river delta, where there are smectites and minerals that are important for organic preservation), and one of an area with less likelihood of preserving life (outside the river delta). If the IR spectrometer detects at least carbonates and perchlorates in the mission, it will have achieved payload success. Based on known hazards in the area, the areas that can be explored are the northwest area of Jezero (particularly the river delta, which the main navigation goal will be to get there), the western part of Jezero, and possibly the southwest of Jezero. For the significance of perchlorates, though they are normally toxic to plants and Earth life, there is evidence that some bacterial lifeforms are able to overcome perchlorates and even live off them (Acevedo-Barrios, Rosa et al.). Although Mars' conditions are harsh, particularly in that UV exposure on the surface tends to destroy organic molecules and form even more dangerous chemicals which in lab tests on Earth were shown to be more lethal to bacteria than the perchlorates alone, the possibility of microbes being resistant to percholates or even being dependent on them to survive warrants some investigation. In addition, perchlorates can absorb atmospheric water and allow for the existence of stable liquid water brines (through lowering the freezing point of water and absorbing water from the atmosphere), which could prove to have been an important development in the evolution of life on Mars (Martín-Torres et al.).

Smectite clays are notable for their ability to trap organic matter in the interlayer sites of the mineral structure. The kinds of clay deposits in Jezero Crater record two periods of martian history when the surface or near-surface was probably habitable, and thus has high potential for preserved life. In sedimentary basins on Earth, smectite clays are associated with many of the most organic-rich units. Olivine is a magnesium-iron silicate mineral that could prove important in the history of water in the area (Rice et al). These silicate minerals might have played a role in absorbing water, and where there is water, it is generally assumed that there is potential for life. Investigating these silicates, then, would be important in understanding the context of potential life in the past on Mars (i.e. high olivine concentrations with absorbed water could imply a large presence of liquid water).

The AOTF IR spectrometer will be able to determine the molecular constituents of samples and soil such as carbonates, oxides, and ortho- and chain silicates (olivine) (J. -M. Reess et al.). Using its functional short wave infrared wavelength range of 1.6-3.6 m, it will be able to detect the signatures of these molecules as well as perchlorates, making the instrument capable of answering if Jezero has a variety of soils/minerals, especially from when it might have had life. These organic minerals are critical to the functioning and preservation of life, so confirmation of these will help to answer whether Jezero could have been a site where Mars could have supported life in the distant past. This could then allow the team to determine if this life exists presently, whether on the surface, underneath the soil, or within rocks. This instrument determines the molecular makeup of many significant life-associated minerals, allowing it to be used in solving science Objectives 1 and 2.

The spectrometer will be put on the mast to provide it with the greatest distance to see and measure samples, as well as being able to angle itself down to the ground to scan samples. This kind of spectrometer was placed on the bottom of the Mast Unit of SuperCam/ChemCam in previous missions, so placing it on a mast is probably the best location choice for the instrument. This position would provide the best way for the spectrometer to collect data on its samples, especially given the ability to change its angle relative to the ground (pointing up vs down).

If the NavCam takes at least 1 photo and sends it back to Earth, it will achieve payload success. The NavCam, in addition to assisting in navigating the terrain of Jezero, will be used to take pictures of geological landscapes and features of interest as well as locations where samples will be analyzed. Having this instrument will be important to knowing what the surrounding area of analyzed samples looks like to give context for findings from samples, as well as providing data for future missions to investigate more thoroughly in the case of images of geological features and landscapes.

4.2.4 Describe Experimental Logic, Approach, and Method of Investigation

At least 2.4 W of power will be used from the power subsystem when performing experiments, with 4.55 W used when the NavCam is online. The payload will perform science experiments by analyzing the molecular composition of samples and detecting the presence of various molecules and minerals through infrared spectroscopy.

Figure 48 illustrates the spectrum of light being utilized on the mission. Particularly, the range to be used will be from the near infrared to the mid-infrared (JNH Lifestyles). Table 16 also shows the infrared regions being used by the AOTF IR spectrometer to characterize biologically significant molecules (Malwathumullage).

Figure 49 shows the transmittance spectrum of magnesium perchlorate, a mineral thought to be responsible for the existence of liquid water brines on Mars and assumed to be found at Jezero Crater (Wadsworth and Cockell) (Clark and Kounaves). The peak absorption of the compound is found where the transmittance is lowest, as the relationship between absorbance and transmittance (from Beer's Law) ("Beer's Law - Theoretical Principles") is defined as

$$A = -log(T),$$

where A is absorbance and T is transmittance ("Transmittance And Absorbance Table Chart"). The transmittance at around 3.4 microns is about 0.284 ("Magnesium Perchlorate"). Using the above equations, the absorption is calculated to be about 0.547 absorbance units (Au). 1.0 Au is ~10% transmittance and 2.0 Au is ~1% transmittance, meaning that absorbance is inversely related to transmittance on a logarithmic scale (Absorbance). The unit can also be conceptualized as the ratio of light absorbed to overall light striking the sample. The absorbance indicates how strongly the given compound absorbs certain wavelengths of light, reflectance indicates the strength of reflection of that light, and transmittance indicates how much light passes through the sample. If the amount of absorbance in a sample is observed to be the same as what is expected of that molecule, that molecule is most likely present. For example, if an absorbance of 0.547 is detected in a sample, perchlorates are likely present in the area. Absorbance would be the best metric to use given that the molecule and its chemical bonds uniquely absorb the energy from the infrared light emitted by the AOTF IR spectrometer, and absorbance peaks are typically used in spectroscopy to identify the presence of compounds.

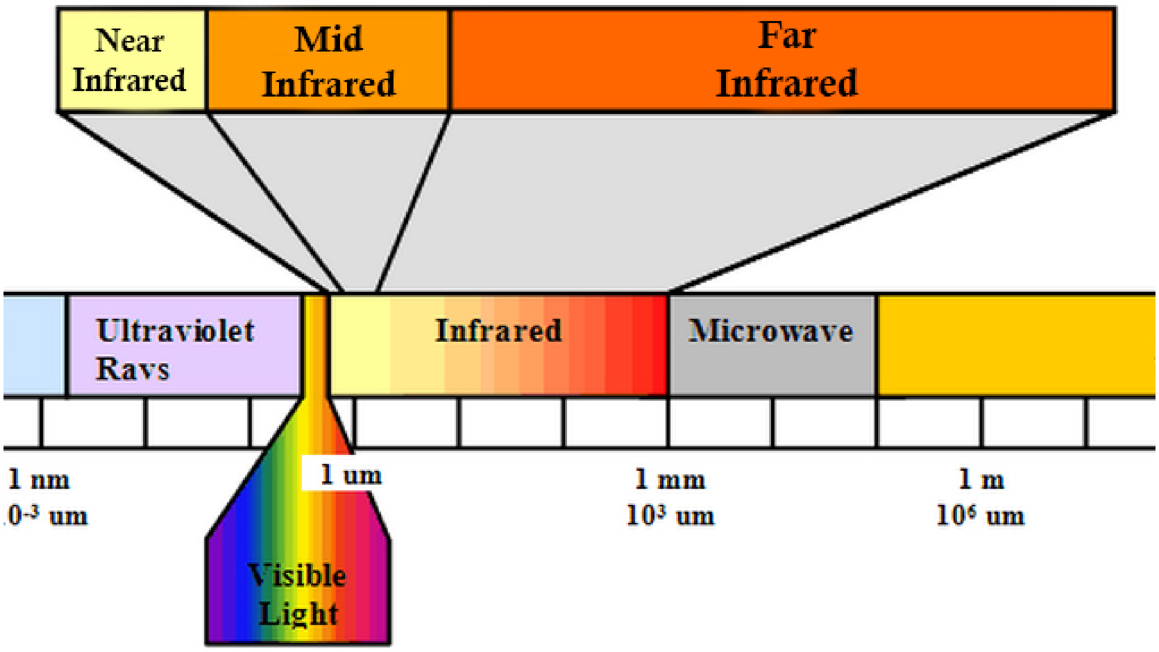


Figure 48: IR spectrum visualization

Region	Wavelength(μm)	Wavenumber (cm^{-1})	Main Applications
Near-IR	0.78 - 2.5	12,800 - 4,000	Quantitative determination of solid, liquid and gaseous samples
Mid-IR	2.5 - 50	4,000 - 200	Qualitative determination of complex solids, liquids and gaseous mixtures (specifically organic molecules)

Table 16: Visualization of the wavelength range being used, along with their main applications (Malwathumullage)

The wavelength of 3.4 microns is at the end of the range of the spectrometer, and is a peak absorption band for magnesium perchlorate, so 3.4 microns will be used for detecting perchlorates. A similar peak absorption percentage at 3.4 microns is found for a similar compound, sodium perchlorate ("Sodium Perchlorate"). This indicates that 3.4 microns is a reliable band to search for perchlorates in the context of Jezero Crater and that using these compounds as representations of perchlorates is reliable, as sodium and magnesium are expected to be found at the site in significant quantities, and it can be assumed that these would form compounds with perchlorates.

Olivine can be detected at around 1.7 microns ("Olivine-Group Minerals").

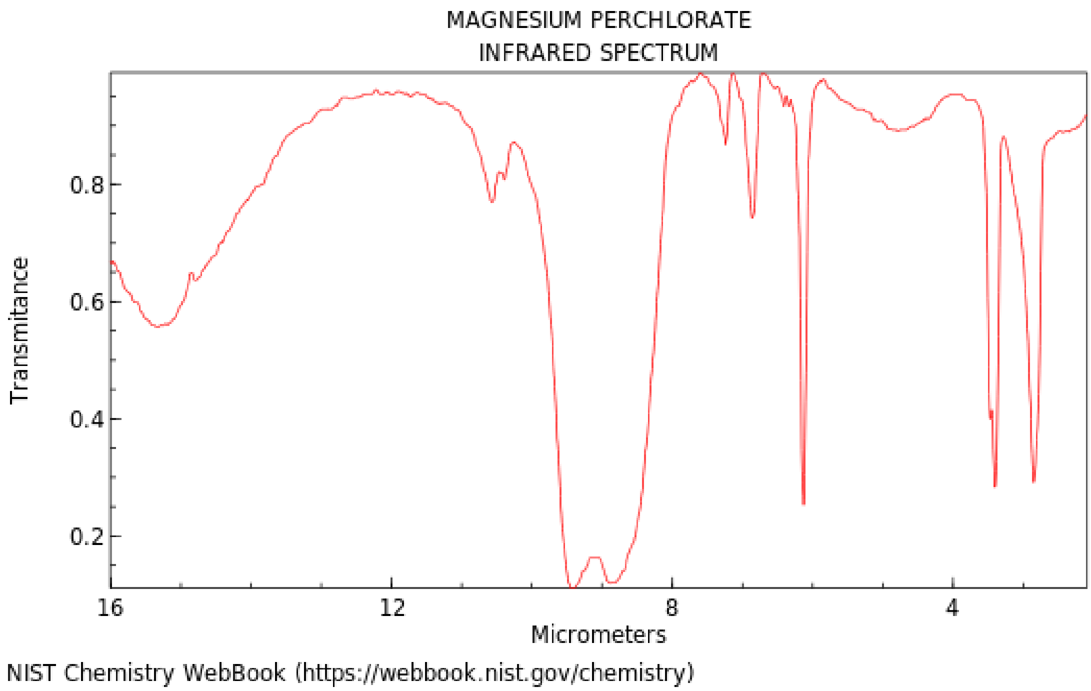


Figure 49: Transmittance spectrum of magnesium perchlorate (“Magnesium Perchlorate”)

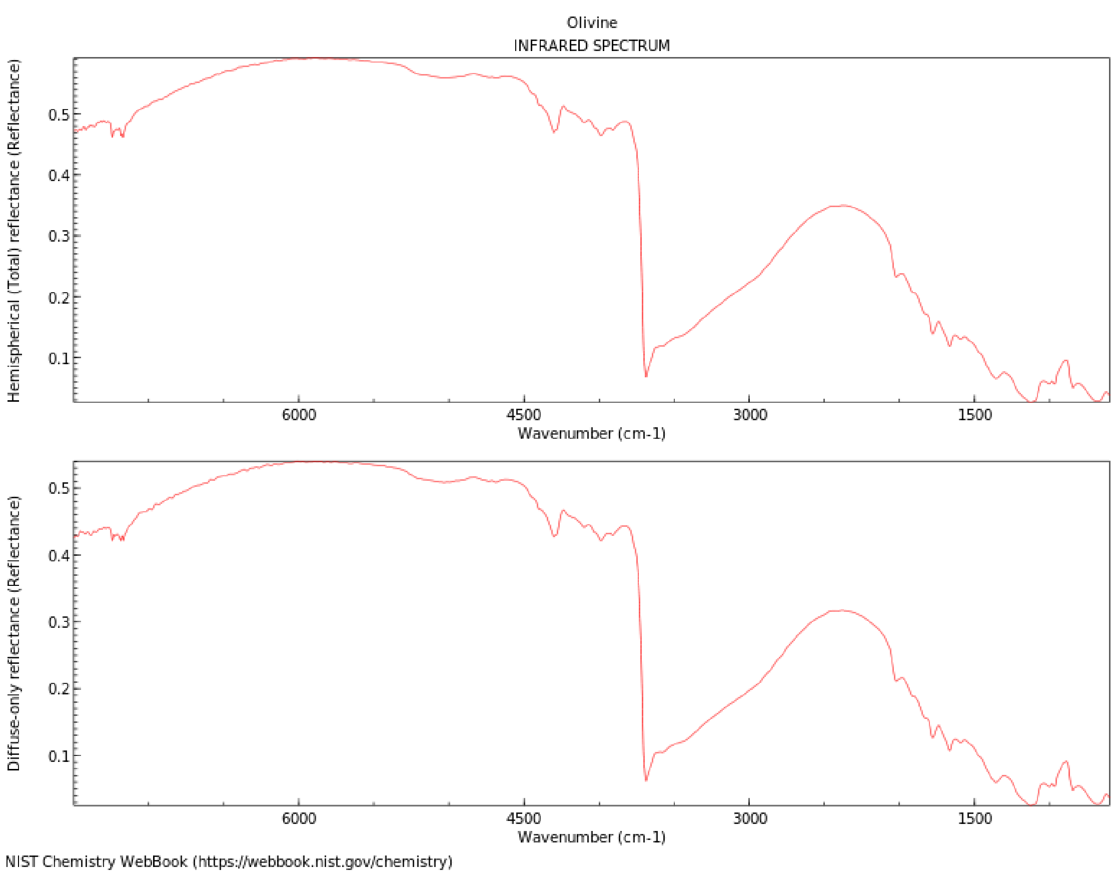


Figure 50: Transmittance graph of olivine

As Figure 50 shows, at a wavenumber of around 5880 (the corresponding wavenumber for 1.7 microns), reflectance is high. Given reflectance, it can be assumed that

$$A = -log(1 - R),$$

where R is reflectance, A is absorption, and transmittance, T, is defined as 1-R (Bisson).

At 1.7 microns, reflectance is around 0.6, making absorbance to be around 0.4. Given the context of the instrument, a wavelength of about 1.7 microns is a good band to use for the detection of olivine.

Carbonate - 1.9 microns (Harner and Gilmore).

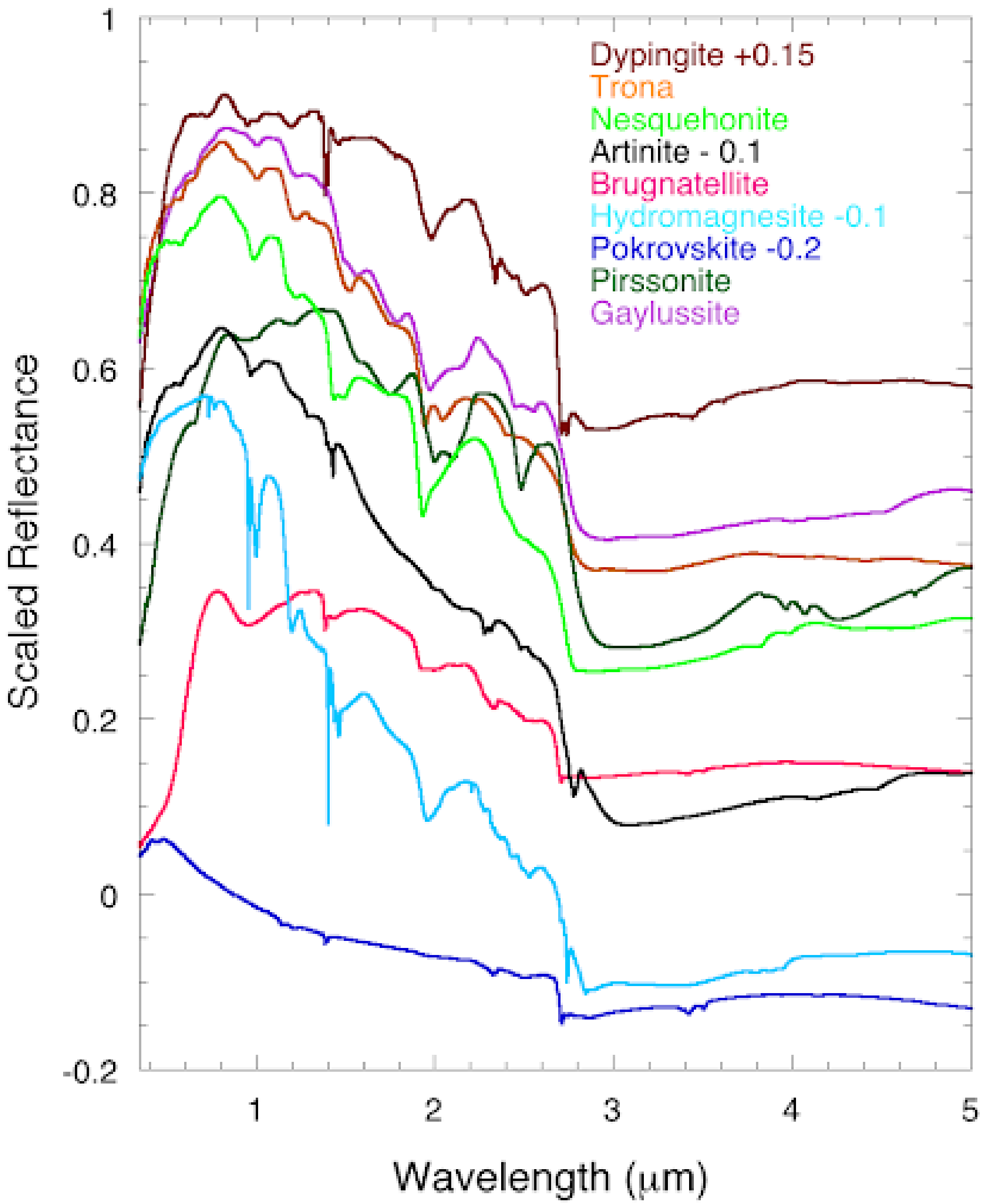


Figure 51: VNIR Spectra of various hydrous carbonates

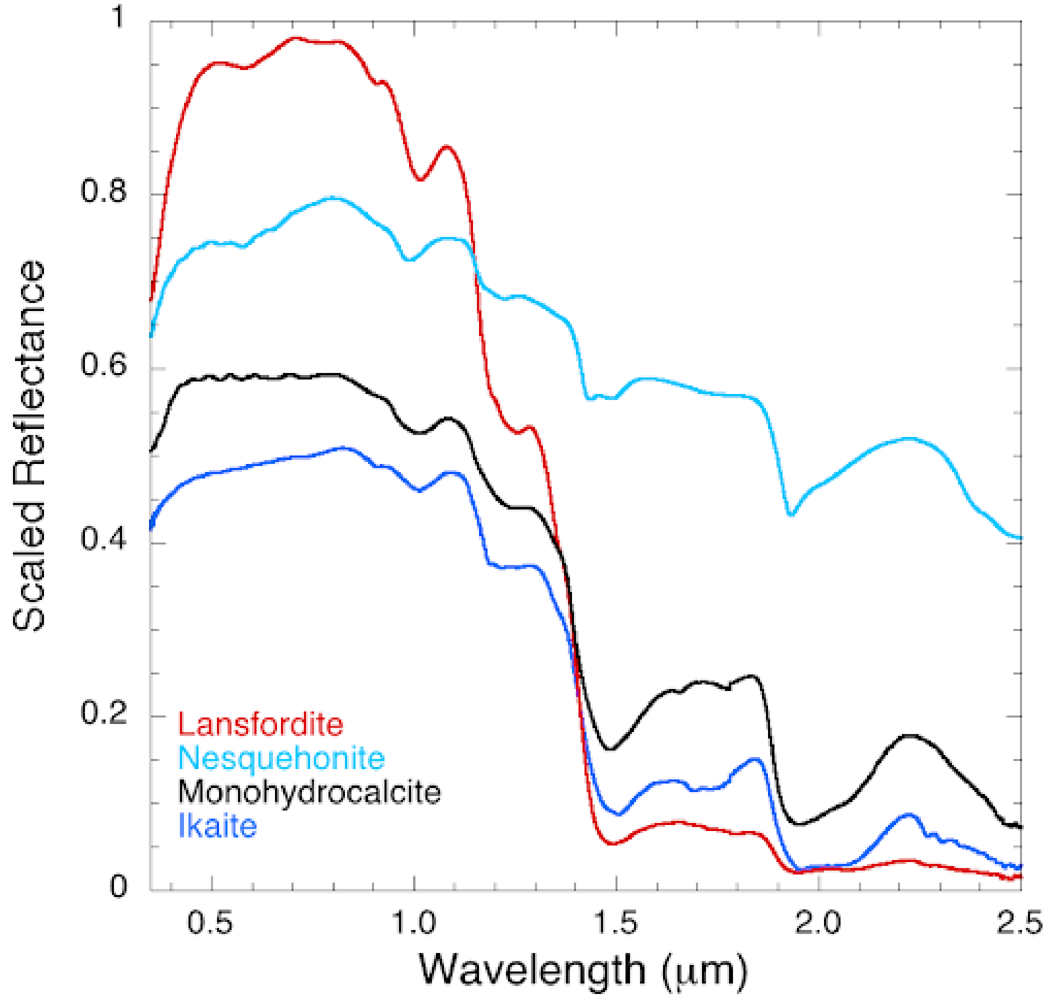


Figure 52: Spectra of 2 synthesized carbonate samples (monohydrocalcite and ikaite) compared with 2 natural carbonates

Figures 51 and 52 illustrate the visible and near-infrared reflectance of 13 carbonate samples, including 4 synthesized carbonates. There is significant absorption at around 1.9 microns, especially for the hydrous carbonates (Harner and Gilmore). At 1.9 microns, given a reflectance of around 0.05, the absorbance is calculated to be 1.30. This peculiarly strong absorption signature makes 1.9 microns a good band to observe carbonates with, even Mg-OH carbonates, which might have a significant presence at Jezero (Harner and Gilmore).

Pyroxene has been found to be detected within the range of 1 and 2 microns. The band that will be used will be about 2.7 microns (Clark).

Figure 53 shows the reflectance of various grain sizes of pyroxene. With all grain sizes, pyroxene reflectance decreases for all grain sizes around 1 micron, 2 microns, and 2.7 microns, and the larger the grain, the more light is absorbed. The band which has the lowest reflectance (and therefore, the highest absorbance) seems to be apparent at the sharp drop around 2.7 microns. The difference in magnitude of reflectance between 2.0 and 2.7 microns is most apparent when looking at the 5-10 micron sized grains. At around 1.9-2.0 microns, the reflectance is about 0.5. At 2.7-2.71 microns, the reflectance is about 0.275. Given the relationship between absorption and reflectance, absorption is then found to be about 0.561. Given this value, 2.7 microns is a good band to use to determine the presence of pyroxenes.

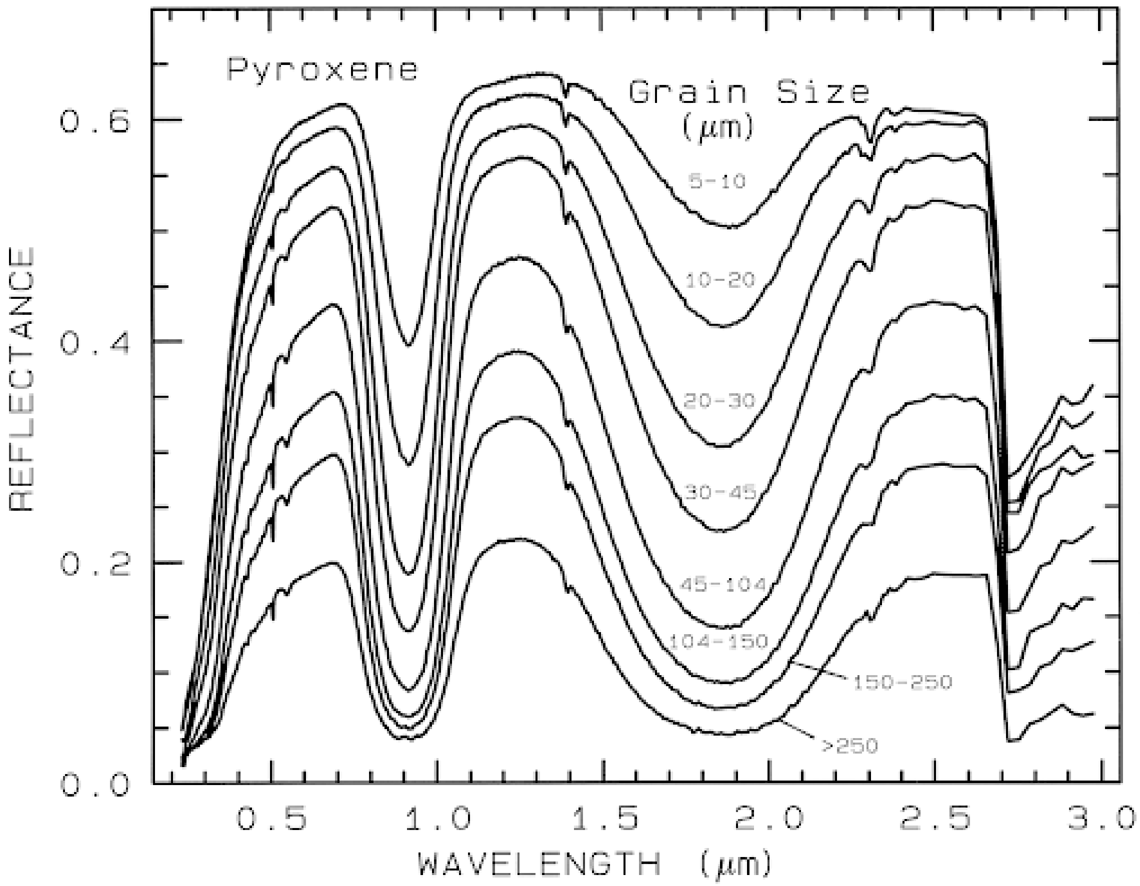


Figure 53: Reflectance of different grain sizes of pyroxene

Maxima (cm^{-1})		Assignment
P	A	
3623	3620	OH stretching
3442	3431	OH stretching from H_2O
1641	1637	OH bending from H_2O
1088	/	Si–O stretching vibration (out-of-plane)
1036	1037	Si–O stretching vibration (in-plane)
915	916	Al–Al–OH bending
874	/	Al–Fe–OH bending
841	843	Al–Mg–OH bending
794	793	Cristobalite
692	695	Quartz
623	625	R–O–Si with R = Al, Mg, Fe (Li)
519	521	Si–O–Al vibration (Al octahedral cation)
467	467	Si–O–Si bending vibration

Table 17: Important smectite spectral ranges for smectite and other molecules in the infrared, visible, and ultraviolet spectrum (Tomic et al.). Given the limitations of the instrument, only OH stretching will be detected, but this is significant.

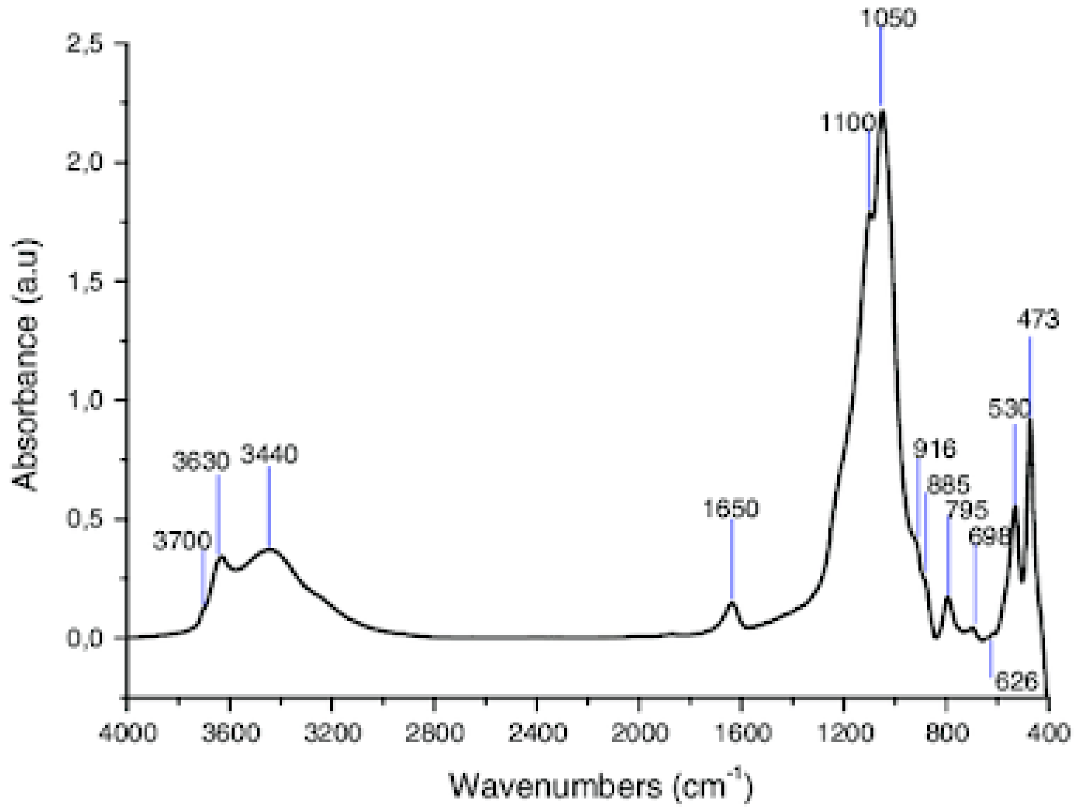


Figure 54: Smectite absorbance vs. wavenumers

Smectites and other clays are thought to be present at Jezero Crater and are worthy of some investigation. The above graph shows the absorbance of a sample of smectite clay from west Cameroon. The bands at 3,630 cm^{-1} and 3,700 cm^{-1} correspond to stretching of O-H bonds. The one at 1,650 cm^{-1} is related to stretching vibrations of O-H vibrations in water and deformation of adsorbed water, being very close to the wavenumber for OH bond bending in water (Tomic et al.). Though the instrument does not have the ability to go to 1050 cm^{-1} , where the absorption is at a maximum (the minimum wavenumber is 2778 cm^{-1}), it will be able to detect the stretching of bonds within water, which could act as an indirect indicator of the presence of smectites and prove to help in finding signs of past life by "following the water". Thus, the instrument will use a band of 3440 cm^{-1} (about 2.91 microns) to detect OH bond stretching and determine a context for the area being studied (How Do You Convert From Microns To Wavenumbers And Vice Versa?). Whether it is wet/aqueous or dry will be of interest when examining the area more closely, as where there is water, there is likely to be life. Finding water through this method might not only be helpful in determining potential smectites, but also in determining a more thorough context for potential perchlorates detected in the sample. Given that perchlorates can lower the freezing point of water to a point where liquid brines can exist, if they are detected in the area alongside the OH bond stretching in water, this could be something that a later, more sophisticated mission can investigate closely to see if life may have arisen, what the role of perchlorates were, what the history of the area could have looked like, what this data could tell about what Jezero looked like in the past, and its role in supporting potential life.

When reviewing the payload's path, the elevation levels needed to be taken into consideration. Figure 55 displays the elevation changes that will be encountered by the payload on the selected path. The estimated path will be around 21.32 km with the steepest climb at around the 18.55 km mark. This steepest climb has a slope of about 13.36^{o}, which was determined to be within the rover's limits.

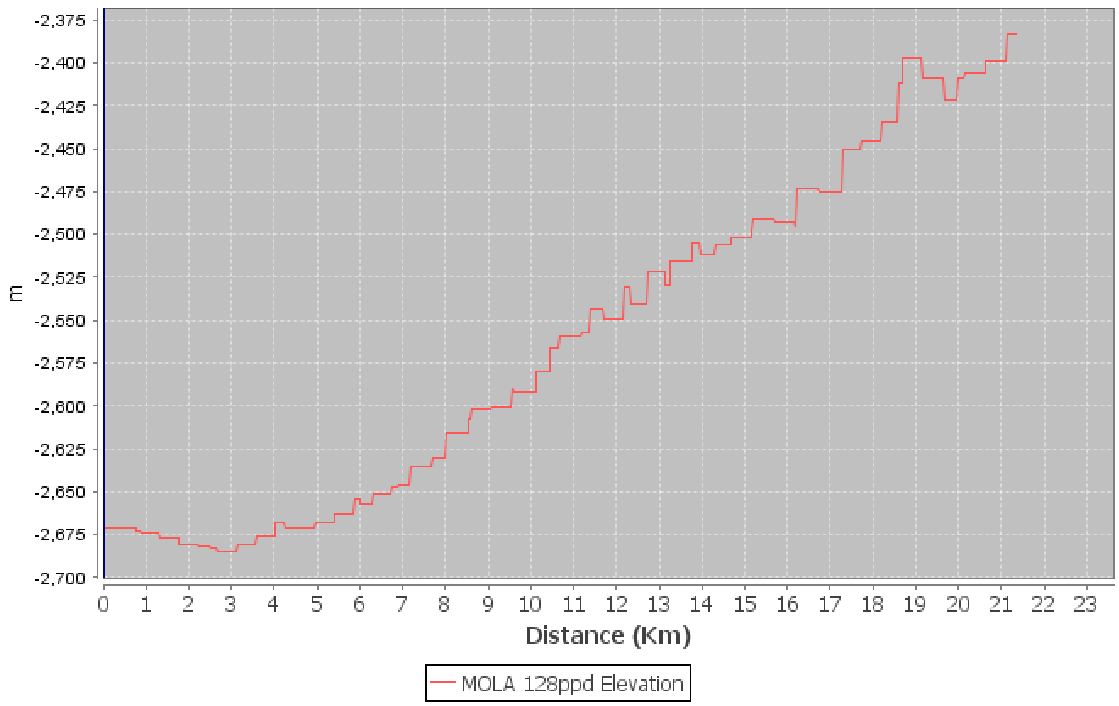


Figure 55: Elevation changes encountered by Fortitude on selected path

4.2.5 Describe Testing and Measurements, Including Variables and Controls

While AOTF IR spectrometers are suitable for interplanetary exploration, significant changes in the environment, specifically the temperature, can cause variation in the instrument's outputted spectral characteristics and response curves. This debilitates the accuracy of data and the instrument's ability to give continuous, steady data. To combat the effects of temperature on AOTF infrared instruments, it is necessary to calibrate the system. The calibration process for the AOTF IR spectrometer includes subjecting it to two independent temperature tests.

In the first test, the radio frequency (RF) power amplifier of the spectrometer will be placed in a thermal vacuum chamber and will be held at 20°C while the detector remains in a room temperature environment. Next, the temperature of the chamber will be lowered to -50°C and will eventually climb to 40°C. While this happens, the digital number reported by the spectrometer will be recorded over different temperatures and wavelengths. After normalizing and analyzing the data, an influence model of the temperature on the RF power amplifier will be obtained.

The second test is quite similar to the first test, except the spectrometer detector will be placed in the thermal vacuum chamber while the RF power amplifier will remain at room temperature. The chamber will be subjected to the same temperature range of -50°C to 40°C. Again, the digital numbers for various working wavelengths of the spectrometer at varying temperatures will be recorded. Once the data has been normalized and analyzed, an influence model of the temperature on the detector will be obtained.

Once the temperature influence models on both the RF power amplifier and the detector of the spectrometer have been obtained, a measurement correction model can be derived. The model will allow the spectrometer to operate as it is expected to in its operating temperature range of -50°C to 40°C while it is onboard the payload on Mars. This model will be applied by the RAD750 to all of the data collected by the AOTF IR spectrometer in order to acquire the most accurate measurements (He).

Figure 56 shows the wavelength range the team is using, from 1.6 microns to 3.6 microns, showing a typical 256-scan average of the gold reflectance standard, which has a constant reflectivity throughout the wavelength range. The fine structure, near 2.7 µm, is caused by water vapor in the measurement path. These discrepancies are generally stable and are removed from a measurement by computing the ratio of a raw measurement scan to a raw reference standard scan (Tawalbeh et al.).

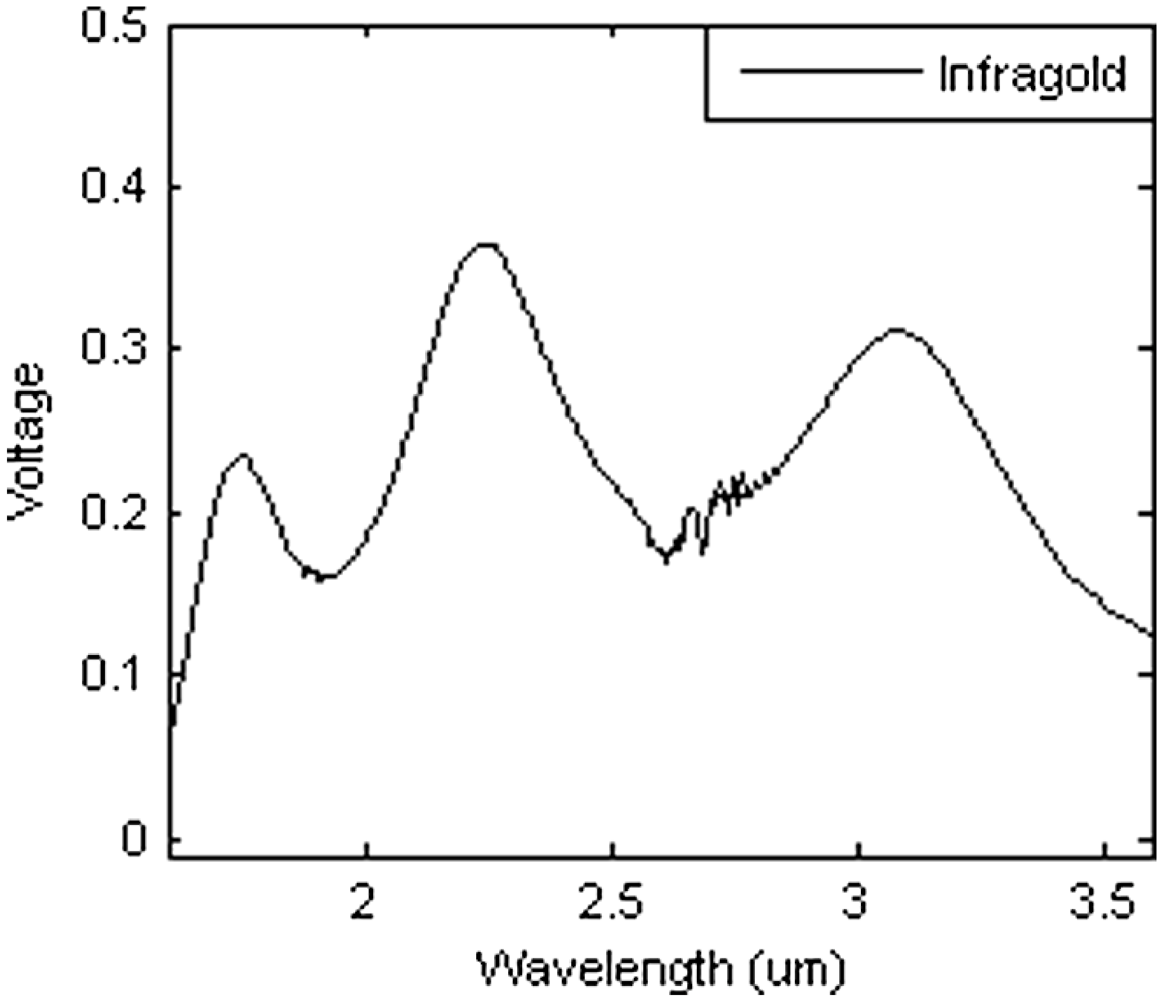


Figure 56: Wavelength range being used

The efficiency of the AOTF IR spectrometer may be negatively impacted by a radio frequency mismatch between source and AOTF transducer, which is dependent on the radio frequency cable length. Water vapor in the measurement path can cause a fine structure near 2.7 µm. This is a consistent, but stable issue which can be neglected by computing the ratio of a raw measurement scan to a raw reference standard scan.

The control variable for the AOTF IR spectrometer will be the data taken from previous experiments.

Since typical preflight temperature changes of a NavCam component, charge-coupled device (CCD), ranges from 1°C - 5°C, it is necessary to calibrate the NavCam prior to the launch and during spaceflight operations.

A flight software has the ability to perform a considerable amount of image processing on board the rovers prior to downlink. This software is primarily designed to reduce the volume of data to be downlinked to the Earth. These processes include a shutter smear correction, a flat-field correction, a bad pixel correction, image down-sampling, image subframing, pixel summing, 12-to-8 bit scaling through lookup tables, and image compression. The shutter smear correction, if performed on board, removes the bias offset, the masked area dark current, and the shutter smear.

To do so, the flight software acquires a zero-second exposure image immediately after the subject image is acquired and subtracts it from the subject image. If the on board shutter smear correction has been applied to an image, it is indicated with a "TRUE" value in the image's data file header. The flight software has the ability to scale the 12 bits per pixel data to 8 bits per pixel using either a lookup table (LUT) or bit shifting. Noise is proportional to the square root of the number of electrons detected. Employing a roughly square root LUT can thus allow the image size (in bits) to be reduced without losing statistically significant information. An inverse LUT can be used to scale the images back to 12-bits.

Additionally, bias, added to the signal by a voltage offset in direct relation to very low temperatures, are recorded. A shutter smear correction is performed for each image to neglect bias, regardless.

Because there is no way for the NavCam to measure the amount of dark current that accumulates in a pixel during daytime operations on Mars, and nighttime temperatures are too low to generate measurable dark current, a model should be employed. The rate at which charge accumulates in a pixel is strongly dependent on the temperature of the CCD and can be approximated by an exponential function. It takes about 5.4 seconds to read out a full-frame NavCam image, during which time dark current accumulates and is added to the signal. Rows farther from the readout register take longer to read out, and therefore, accumulate additional dark current than the rows closer to the readout register. A shutter smear correction also allows for the masked area dark current to be negligible.

The received images will be compared to legacy data, particularly the information stored in JMARS.

4.2.6 Show Expected Data & Analyze (Error/Accuracy, Data Analysis)

The Compact Reconnaissance Imaging Spectrometer (CRISM) for Mars, which uses detectors that see in visible, infrared and near-infrared wavelengths to map the kind of mineral residue that appears where water once existed. Data taken by CRISM will mirror the expected data from the AOTF IR spectrometer, and therefore, can be analyzed for some context of the mission's anticipated data. CRISM covers an area in close proximity to a crater in the northern plains of Mars. Figure 57 shows strong Fe-rich olivine (yellow) and high calcium pyroxene (HCP, blue) signatures in the ejecta near the crater rim. This is indicative of mafic minerals that have been excavated by this impact crater from a depth of up to 4 km below the current surface, possibly representing previously buried ancient crust. Extensive sedimentary and volcanic deposits cover the northern lowlands of Mars dating back to the Hesperian era. Craters like this one can offer a glimpse of the composition of the underlying older crust.

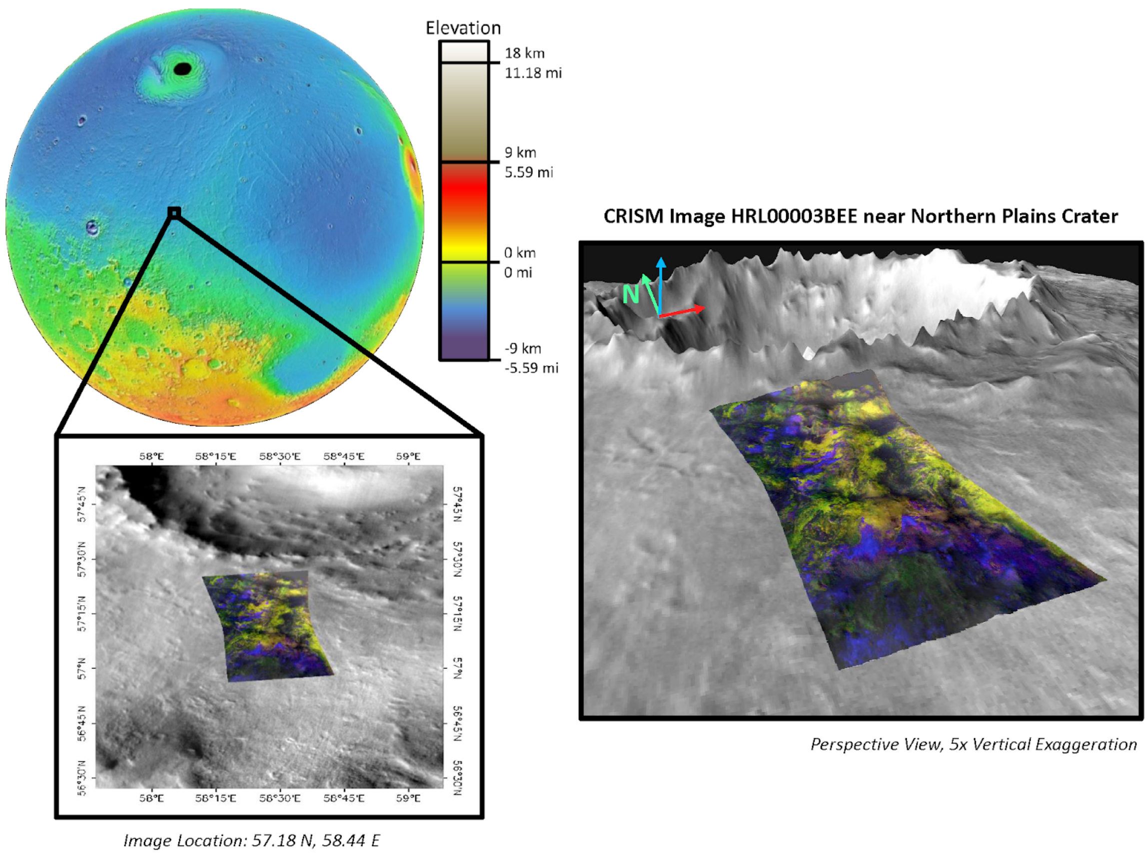
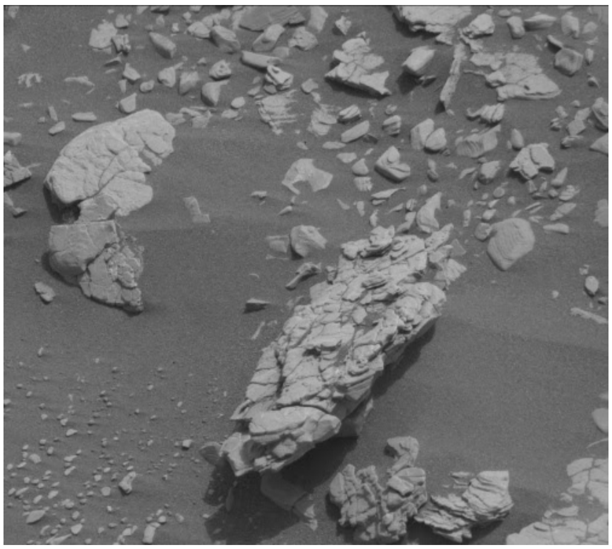


Figure 57: Signatuers of Fe-rich Olivine and HCP

The data will be memorized by the RAD 750 processor and will be transmitted to locations on Earth. Evidence of carbonates, perchlorates, olivine, or pyroxene in Jezero Crater would suggest the existence of water on Mars. These biosignatures would help the team determine if there was life on Mars in the past and whether or not it has been sustained until today. Mars even holds the best record of the prebiotic conditions leading to life, even if life does not or has never existed there. This will benefit the study of the origins of life because of its similarity to the early Earth. Due to Mars' proximity and similarities to Earth, this information will dictate future habitability for humans.

Images taken by NavCam during the mission will be similar to the images captured by the Mars Curiosity rover. Figure 58 shows example images.



(a) NavCam example image 1

(b) NavCam example image 2

Figure 58: Two example images taken by NavCam

The data will be memorized by the RAD 750 processor and will be transmitted to locations on Earth. The information will be compared to previously developed databases, particularly JMARS. The information can be used to update our older idea of the Mars surface, and determine if there have been any changes since the previous exploration of Jezero Crater. NavCam capability enables the acquisition of azimuthal 360-degree panoramas of the martian surface, images of the rover deck, and the martian sky. This will provide a clearer understanding of the terrain in Jezero Crater, which can be used to benefit future missions during their landing and rover transit processes.

5 Safety

5.1 Personnel Safety

5.1.1 Designated Safety Officer

Safety Officer, Kejsi Bishaj

In every mission, the safety officer has the responsibility to ensure the safety of the mission. He/she is responsible for the development and execution of the risk management, mitigation policies and procedures. When needed, he/she creates safety plans that include suggested improvements to existing infrastructure and proposes those to the science and engineering team. Among the mentioned duties, the safety officer is responsible for:

- Ensuring the safety of fabrication, assembly and test operations.
- Assessing risk and possible safety hazards of all aspects of the operation.
- Rating the risks and deciding on the approach towards mitigating them.
- Categorizing risks in order to relate the risks to the systems.
- Utilizing safety analysis to influence the system's development and design.
- Analyzing environmental factors and ensuring that the design will withstand all environmental variables.
- Reporting safety concerns and addressing those with team members.
- Ensuring the safety of the team members in the testing and working environment.
- Predicting/anticipating human error and ensuring human error management.
- Promoting a safety culture which encourages and enforces adherence to approved maintenance and quality assurance procedures.

After the potential risks have been identified, a certain approach is shown towards them. Each identified risk is evaluated, and the progress of the selected risk responses is tracked. Tracking applies to the Mitigate, Watch, and Research risk response types. The option types Accept, Close, and Elevate do not have tracking requirements related to them. Response types considered for this mission after the risks have been identified are:

- Watch - Watch option type does not involve changes to the project plan, as a result would not require monitoring of implementation.
- Research - Research option type requires a research plan whose implementation should be tracked.
- Mitigate - Mitigate option type involves mitigation options integrated into the project plan.
- Accept - No tracking requirements.

In order to identify the risks associated with this mission, two risk matrices were made to describe the rating that each individual risk may possess. The first risk matrix below of Likelihood Vs. Consequences illustrates the ratings that will be used to measure the risks related to the personnel safety and testing environment. For the likelihood metric, it will range from very unlikely to very likely, and for the consequence metric, the range will be negligible to severe. The second risk matrix below of Likelihood Vs. Consequences illustrates the ratings that will be used to measure the risks identified related to the martian environment. The likelihood metric and the consequence metric for the hazards related to the martian environment will be the same as for the personnel safety and testing environment. Upon completion of the way in which the risk will be analyzed, two risk tables were created to identify the risk, rank, category, position in risk matrix, approach, type, and mitigation methods.

5.1.2 List of Personnel Hazards & Personnel Hazards mitigation

Approach
A - Accept
M - Mitigate
R - Research
W - Watch

LxC Trend
$\downarrow$ - Decreasing (improving)
$\rightarrow$ + Increasing (worsening)
$\rightarrow$ - Unchanged

Rank	Risk Category	LxC	Approach	Risk Type	Mitigation
1	Physical hazards and others (Non Ionizing Radiation)	4x3 $\rightarrow$	M	Laser Hazards	Usage of appropriate eye protection and prevention of unprotected personnel from entering the area.
2	Physical hazards and others (Non Ionizing Radiation)	4x3 $\rightarrow$	M	Radiofrequency and Microwave Radiation	Usage of engineering controls by enclosing or effectively shielding the radiation emitting equipment. When engineering controls do not find usage, workers must be provided with protection, such as protective clothing and eyewear.
3	Physical hazards	1x5 $\rightarrow$	M	Radiation	Areas in which radioactive materials are used or stored must display the symbol for radiation hazards and access should be restricted to authorized personnel. Precautionary measures and personnel monitoring must be established for workers who are exposed to radiation. Personnel monitoring must be used when required to measure a worker's radiation exposure.
4	Physical hazards	3x2 $\rightarrow$	M	Ergonomic hazards	Becoming familiar with ways to control laboratory ergonomics-related risk factors to reduce the chances for occupational injuries.

Table 18: Personnel risks and mitigation (1-4)

Rank	Risk Category	LxC	Approach	Risk Type	Mitigation
5	Safety hazards	1x4 $\rightarrow$	M	Cryogenic substances	Protective clothes, face shields or safety goggles, safety gloves and long sleeved shirts, and lab coats must be worn during transfers of cryogenic fluids which might result in exposure to the liquid.
6	Safety hazards	2x4 $\rightarrow$	M	Electrical hazards, electric shock, electrocutions	Ensure that all electrical services near sources of water are properly grounded. Sufficient working space must be maintained around electrical equipment operating at $\leq$ 600 V to enable safe operation of electrical equipment, only qualified workers are allowed to work on electrical circuits and systems, and they must know the locations of circuit breaker panels in their working area.
7	Safety hazards	2x4 $\rightarrow$	M	Fire and explosions	Minimization of chances of accidental fire by proper procedures and training of the workers. Preparation of all workers to deal with fire emergencies by placing all containers of infectious materials in freezers and incubators.
8	Safety hazards	2x4 $\rightarrow$	M	Injuries related to service and maintenance operations	Establishment of basic requirements related to locking and tagging equipment during installation, maintenance, testing, repairing, and construction operations.
9	Safety hazards	1x4 $\rightarrow$	M	Compressed gases	All cylinders containing compressed gases must be stored upright, should never be dropped or allowed to be striked with force, be transported with protective caps in place and never dragged.
10	General Safety	1x3 $\rightarrow$	M	Non-mentioned research	Disallowing non-mentioned research or development operations at working facilities.
11	Public Safety	1x4 $\rightarrow$	M	Crowd-gatherings	Exclusion during crowd-gatherings events (launch).
12	General Safety	2x4	M	Unknown risks	Establishment of NRSR reports of interest to workers safety.
13	Safety hazards	3x3	M	Manufacturing risks	Training of employees with skills to avoid workplace accidents related to the use of heavy machinery or hand tools.

Table 19: Personnel risks and mitigation (5-13)

Likelihood	**5** Very Likely					
	4 Likely			1 2		
	3 Possible		4	13		
	2 Unlikely				6 7 8 12	
	1 Very unlikely			10	5 9 11	3
Table of Likelihood Vs. Consequences		**1** Negligible	**2** Minor	**3** Moderate	**4** Significant	**5** Severe
		Consequences				

Table 20: Likelihood vs. Consequences (1-13)

High
Med
Low

Table 21: Criticality key

5.2 Lander/Payload Safety

5.2.1 Environmental Hazards & Environmental Hazards Mitigation

Rank	Risk Category	LxC	Approach	Risk Type	Mitigation
14	Structure and configuration	5x4 ↓	R	Dust accumulation on exterior and internal parts of components	Implementation of electrostatic precipitators that will be able to effectively remove dust from the environment of Mars.
15	Structure and configuration	1x2 →	M	Instability and collision	Designation of systems on the assumption that rocks on Mars have a worst case hardness, similar to basalt. Testing of systems using rocks with a variety of worst-case surface roughnesses.

Table 22: Lander risks and mitigation (14-15)

Rank	Risk Category	LxC	Approach	Risk Type	Mitigation
16	Structure and configuration	3x3 →	R	Mechanical failure	Characterization of the bulk physical properties of the martian regolith that will interact with the rover vehicles.
17	Structure and configuration	4x3 →	M	Triboelectric charge by wheel movement	Addition of needles, half an inch long made of ultrathin (0.0001-inch diameter) tungsten wire sharpened to a point, at the base of antennas.
18	EPS (Electrical power system)	3x4 →	M	Loss of communication with satellite, transmission of data failure	Link budget & margin calculated for worst case scenario.
19	Thermal Control	1x5 →	M	Very low temperatures	Prevention of heat escape through Aeroglaze Z306 coating. Keeping the rover warm through heaters.
20	Thermal Control	2x3 →	M	Excess heat from the over electronics.	Development of Heat Rejection System (HRS).

Table 23: Lander risks and mitigation (16-20)

Further establishment on each environmental risk type & mitigation methods:

Rank 14 - The hazard presented by dust intrusion and adhesion is crucial for long missions on Mars, where the systems with which the rover operates, are affected by Mars' environment. Dust accumulation on external surfaces, driven by a wind process which is further stimulated by magnetic attraction, electrostatic adhesion, or intermolecular forces, could create significant changes in the efficiency of communication and power systems. The communication system could be disrupted, and the power systems could become less efficient. The solution could be electrostatic precipitators, which are already used in other industries. However there is still research needed related to how this technology could be adapted for use on Mars.

Rank 15 - Instability and collision is related to unexpected accidents, such as collision with rocks, which could lead to damage of critical systems. In order to ensure that the terrain the rover is navigating is safe for the successful completion of the objectives, it would be assumed that the rocks on Mars have the worst case hardness, similar to basalt. With this assumption, the successful testing of systems on sites composed of rocks with a high coefficient of hardness in the Mohr's scale, would ensure the proper functioning of critical systems, even in case of accidents.

Rank 16 - Mechanical failure by abrading surfaces that contact the regolith should be considered when deciding on the properties of rover's wheels. In order to avoid the possibility of grooves in the wheels, the following properties of martian regolith are characterized:

- Rock shape
- Regolith sinkage properties such as internal friction angle and shear strength
- Rock abrasive properties
- Rock distribution

These properties are determined by testing in a testing site similar to that of Mars.

Likelihood		1 Negligible	2 Minor	3 Moderate	4 Significant	5 Severe
	5 Very Likely				14	
	4 Likely			17		
	3 Possible			16	18	
	2 Unlikely			20		
	1 Very unlikely		15			19
Table of Likelihood Vs. Consequences		Consequences				

Table 24: Likelihood vs. Consequences (14-20)

High
Med
Low

Table 25: Criticality key

Rank 17 - Mars' soil is dry and electrically isolating. The lack of an electrical ground, similar to that of the Earth, would build a large potential difference between the rover's operation on the surface and its surrounding. Because the soil is insulating, a rover can build up triboelectric charge. If an arc between the rover and the surrounding happens, sensitive devices on the rover could be damaged. The solution to this would be the implementation of needles, half an inch long made of ultrathin (0.0001-inch diameter) tungsten wire sharpened to a point, at the base of antennas. The tungsten needles would ensure that any electrical charge which builds on the rover, passes to the atmosphere.

Rank 18 - A loss of communication with the satellite and the failure to transmit data are to be taken into consideration during long missions. The chances of occurrence are low and would remain the same through a careful calculation of the budget to remain prepared for the worst case scenario.

Rank 19 - The first way of protecting Fortitude from very low temperatures would be through Aeroglaze Z306 coating, which is an absorptive polyurethane coating. It provides thermal absorptivity for applications where superior heat absorption is required. Another way of protecting Fortitude from very low martian temperatures would be through heaters. Excess heat coming from the electronic devices on rover could be a good source of heat.

Radioisotope Heater Units (HRS) which are one-watt heaters could generate heat through the decay of a low grade isotope. Radioisotope Heater Units would also contribute to conserving nighttime battery power.

Rank 20 - An excess of heat from rover's electronics is a risk that should be considered as well . The solution to this would be a Heat Rejection System (HRS) which is composed of a pump in the cruise stage and tubing which releases heat into the atmosphere. The pump is capable of shuttling 150 watts of rover waste heat. The working fluid is CFC-12. This fluid is maintained between -7°C and 0°C (19°F and 32°F) throughout the cruise stage.

Rank	Risk Category	LxC	Approach	Risk Type	Mitigation
21	ADCS (Altitude Control System)	2x4 →	M	Regional or global dust storms or localized dust devils	The monitoring of martian weather through ground operations historical data would ensure that the dust storms, dust devils and even global storms are avoided during the landing stage.
22	ADCS (Altitude Control System)	2x5 →	M	Space debris affects the spacecraft to get out of orbit	Orbit maintenance ensured by a propulsion system with a large enough ΔV to ensure proper ability to maintain orbit.
23	Structure and Configuration	2x3 →	A	Vibration during takeoff, deformation	Material selection of Al 7075
24	Thermal Control	1x4 →	M	High temperature friction with the Mars atmosphere	Heat Shield establishment.

Table 26: EDL risks & mitigation

Rank 21 - The continuous monitoring of winds and dust storms through ground operations historical data would ensure that the rover lands safely on Mars. Based on the data provided by ground operations historical data, scientists and engineers could take advantage of the period of time between rover's entry into the martian atmosphere and rover's landing on the surface to select a safe landing territory within the designated area.

The monitoring of the atmosphere would also help to ensure the safe navigation of the rover on the surface of Mars. This is by avoiding strong winds and dust storms when possible.

Rank 22 - A propulsion system with a large ΔV would ensure that the spacecraft has enough fuel to perform the needed maneuvers in order to avoid space debris.

Rank 23 - Al 7075 exhibits very good mechanical properties, high strength, toughness, and good resistance to fatigue. While it is less corrosion resistant than other alloys (such as 5052 aluminium alloy), its strength more than justifies the downsides. These properties will limit deformation and damage caused by vibrations.

Rank 24 - The Mars atmosphere is thin enough to not contribute much in decelerating the spacecraft once it enters the atmosphere, but the friction force remains high enough to be considered a risk on a mission to Mars. The conventional solution would be a heat shield made of Phenolic Impregnated Carbon Ablator (PICA) which is resistant enough to protect the spacecraft from the entry into the martian atmosphere.

Likelihood	**5** Very Likely					
	4 Likely					
	3 Possible					
	2 Unlikely			23	21	22
	1 Very unlikely				24	
Table of Likelihood Vs. Consequences		**1** Negligible	**2** Minor	**3** Moderate	**4** Significant	**5** Severe
		Consequences				

Table 27: Likelihood vs. Consequences (21-24)

High
Med
Low

Table 28: Criticality key

6 Activity Plan

6.1 Budget

The budget for this mission is capped at one hundred million dollars. At first, we "busted" the budget a few times. After that we narrowed our focus to more specific science goals and smaller teams. Ultimately, our smaller teams consisted of five engineers, five scientists, and two administrators. The odd number of teammates for the engineer and science teams was intentional to try and keep disagreements to a minimum. The administration team was an exception because it is unlikely they would have a disagreement since most of the costs are fixed. However, there must be at least two administration personnel for dual control and accountability reasons. Ultimately, over a five year period, our salaries were $5,525,712 of the budget. This saved a lot of purchasing power for our equipment. Our equipment ended up being $25,674,280. Even after purchasing our equipment, we were left with a gross margin of $42,016,898. This gives a lot of financial "cushion" for run-ups or unforeseen costs.

Additional Information		FTE should be left as =1 for full time (40hrs/week), =0.5 for half time (20hrs/week), or =0 for no work					
	# People on Team	FTE of Team (Year 1)	FTE of Team (Year 2)	FTE of Team (Year 3)	FTE of Team (Year 4)	FTE of Team (Year 5)	FTE of Team (Year 6)
Science Team:	5	0.5	1	1	1	1	0
Engineering Team:	5	0.5	1	1	1	1	0
Administrative Team:	2	0.5	1	1	1	1	0

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Year	Yr 1 Total	Yr 2 Total	Yr 3 Total	Yr 4 Total	Yr 5 Total	Yr 6 Total	Cumulative Total
PERSONNEL							
Science Team	$ 200,000.00	$ 400,000.00	$ 400,000.00	$ 400,000.00	$ 400,000.00	$ -	$ 1,800,000.00
Engineering Team	$ 200,000.00	$ 400,000.00	$ 400,000.00	$ 400,000.00	$ 400,000.00	$ -	$ 1,800,000.00
Administrative Team	$ 80,000.00	$ 160,000.00	$ 160,000.00	$ 160,000.00	$ 160,000.00	$ -	$ 720,000.00
Total Salaries	$ 480,000.00	$ 960,000.00	$ 960,000.00	$ 960,000.00	$ 960,000.00	$ -	$ 4,320,000.00
Total ERE	$ 133,968.00	$ 267,936.00	$ 267,936.00	$ 267,936.00	$ 267,936.00	$ -	$ 1,205,712.00
TOTAL PERSONNEL	$ 613,968.00	$ 1,227,936.00	$ 1,227,936.00	$ 1,227,936.00	$ 1,227,936.00	$ -	$ 5,525,712.00
TRAVEL							
Total Flights Cost	$ -	$ -	$ -	$ -	$ -	$ 4,296.00	$ 4,296.00
Total Hotel Cost	$ -	$ -	$ -	$ -	$ -	$ 1,620.00	$ 1,620.00
Total Transportation Cost	$ -	$ -	$ -	$ -	$ -	$ 1,780.00	$ 1,780.00
Total Per Diem Cost	$ -	$ -	$ -	$ -	$ -	$ 3,834.00	$ 3,834.00
Total Travel Costs	$ -	$ -	$ -	$ -	$ -	$ 11,530.00	$ 11,530.00
OTHER DIRECT COSTS							
Total Outsourced Instrument Cost	$ -	$ -	$ -	$ -	$ -	$ -	$ -
Total Materials and Supplies Cost	$ -	$ -		$ -	$ -	$ -	$ -
Total Equipment Cost	$ -	$ -	$ 25,674,280.00	$ -	$ -	$ -	$ 25,674,280.00
Manufacturing Margin	$ -	$ -	$ 12,837,140.00	$ -	$ -	$ -	$ 12,837,140.00
Total Direct Costs	$ 613,968.00	$ 1,227,936.00	$ 39,739,356.00	$ 1,227,936.00	$ 1,227,936.00	$ 11,530.00	#NAME?
Total MTDC	$ 613,968.00	$ 1,227,936.00	$ 1,227,936.00	$ 1,227,936.00	$ 1,227,936.00	$ 11,530.00	#NAME?
FINAL COST CALCULATIONS							
Total F&A	$ 61,396.80	$ 122,793.60	$ 122,793.60	$ 122,793.60	$ 122,793.60	$ 1,153.00	$ 553,724.20
Total Projected Cost	$ 675,364.80	$ 1,350,729.60	$ 39,862,149.60	$ 1,350,729.60	$ 1,350,729.60	$ 12,683.00	$ 44,602,386.20
Total Cost Margin	$ 202,609.44	$ 405,218.88	$ 11,958,644.88	$ 405,218.88	$ 405,218.88	$ 3,804.90	
Total Project Cost	$ 877,974.24	$ 1,755,948.48	$ 51,820,794.48	$ 1,755,948.48	$ 1,755,948.48	$ 16,487.90	$ 57,983,102.06

********** Do not change percentages in the boxes below unless mission concept instructions specify otherwise.

F&A %	10%	10%	10%	10%	10%	10%
Manufacturing Margin	50%	50%	50%	50%	50%	50%
Total Cost Margin	30%	30%	30%	30%	30%	30%
ERE - Staff	28%	28%	28%	28%	28%	28%

Table 29: Mission Budget Analysis

6.2 Mission Schedule

Project Start:	Mon, 10/8/2018
Display Week:	1

TASK	COMMENT	PROGRESS	START	END
Make progress Gantt chart				
Phase C				
Select preliminary designs		0%	10/8/18	12/28/18
Add lower-level design specifications to system		0%	10/8/18	12/28/18
Perform development testing for subsystems		0%	10/8/18	12/28/18
Document final design		0%	12/14/18	12/28/18
Develop data package		0%	5/28/19	6/28/19
Develop integration plans		0%	12/28/18	5/28/19
Develop verification and validation plans		0%	12/28/18	5/28/19
Develop mission operations plans		0%	12/28/18	5/28/19
Develop decommissioning plans		0%	12/28/18	5/28/19
Develop plan for spares		0%	5/28/19	6/28/19
Develop communications plans	including commands and telemetry lists	0%	5/28/19	6/28/19
Refine procedure for integration		0%	6/28/19	10/28/19
Refine procedure for manufacturing and assembly		0%	6/28/19	10/28/19
Refine procedure for verification and validation		0%	6/28/19	10/28/19
Fabricate all components		0%	10/28/19	6/28/20
Identify risks		0%	10/8/18	6/28/20
Prepare launch site and post-launch activiation		0%	4/28/20	6/28/20
Finalize safety data package		0%	4/28/20	5/28/20
Complete Preliminary Design Review		0%	4/28/20	5/28/20
Complete Critical Design Review		0%	5/28/20	6/28/20
Phase D				
Assemble and integrate components	based on integration plans	0%	6/28/20	9/28/20
Perform system qualification verifications	based on validation and verification plans; includes environmental verfications	0%	6/28/20	7/28/20
Perform system acceptance verfications and validations	tests on all elements	0%	7/28/20	8/28/20
Assess and approve all verification and validation outcomes		0%	8/28/20	9/14/20

Table 30: Mission Schedule (Oct 2018 - Sep 2020)

TASK	COMMENT	PROGRESS	START	END
Resolve verification and validation plan and result discrepancies		0%	9/14/20	9/28/20
Develop verification and validation report		0%	9/14/20	9/28/20
Prepare manuals	includes operator's manual, maintenance manuals, operations handbook	0%	6/28/20	9/28/20
Prepare launch, operations and ground support sites		0%	9/14/20	9/28/20
Coordinate personnel trainings	includes operators and maintainers for initial system and contingency planning	0%	7/28/20	9/28/20
Confirm telemetry validations and ground data processing		0%	9/14/20	9/28/20
Confirm systems and systems' supporting elements are ready		0%	9/21/20	9/28/20
Prepare for launch and checkout of system		0%	9/21/20	9/28/20
Identify risks		0%	6/28/20	9/28/20
Phase E				
Conduct launch vehicle performance assessment		0%	9/28/20	10/8/20
Activate science instruments		0%	9/28/20	10/8/20
Implement spares plan		0%	9/28/20	10/8/20
Collect engineering and science data		0%	9/28/20	10/8/20
Train replacement personnel	including operators and maintainers	0%	9/28/20	10/8/20
Launch!		0%	10/8/20	10/9/20
Identify risks		0%	10/9/20	10/1/21
Address problems	develop failure reports	0%	10/9/20	10/1/21
Process and analze mission's data		0%	10/9/20	10/1/21
Prepare for deactivation		0%	10/9/20	10/1/21
Review lessons learned from mission data		0%	10/9/20	10/1/21
Develop post-flight evaluation reports		0%	10/9/20	10/1/21
Develop final mission report		0%	10/9/20	10/1/21

Table 31: Mission Schedule (Sep 2020 - Oct 2021)

Link to exact document: Click here

6.3 Outreach Summary

The future of Space Exploration is dependent upon the youth of the world. In order to ensure that NASA will be well staffed in its future endeavors, the team has considered two approaches to encourage students to engage with STEM related topics and spur the next generation of scientists, engineers, and mathematicians.

The team's first approach is to target middle school students to educate them about stem related topics and let them engage with their creativity. The goal is to make the idea of heading into a STEM-related field of study more attractive to students at an earlier age. Many schools already hold science fairs that allow students to be creative and work on something they're interested in by themselves or with some guidance. The team would like to encourage schools to incorporate a CAD class using relatively easy-to-use software where kids are told to design something that they need or something that would be useful to them in their science fair projects. The intent is for the students to have fun with the project and present their work at the science fair. Some high schools and college introductory classes follow similar structures, so having this experience presented to students at a young age, with minimal regulations and an emphasis on "fun" and "creativity," could make kids more interested in STEM fields. The team would need to pitch it to school boards to incorporate the idea into the curriculum. Then, the team would need to either train current teachers with a STEM background to teach the class or get newly-graduated teachers to focus on this particular class. This class is not meant to pressure the kids into making something amazing. It should let students test the technological waters and have an open mind about it. To do this, the teachers would need to be trained or informed of this and practice encouraging kids to be creative, rather than teaching them how to follow directions, as they all currently do.

Another program that Team 16, the L'Space Invaders, has chosen to support is the NASA Community College Aerospace Scholars (NCAS) program. NCAS is a program that helps motivate community college students to pursue STEM degrees from four year universities. Two of the L'Space Invaders have attended community college and subsequently transferred to four year universities. For those team members, NCAS was their first exposure to STEM.

NCAS is a five week course with a series of tests that culminate in a research project that can range from several space related topics. The curriculum over these weeks includes learning about current and previous NASA missions. Team 16 would volunteer data found on Mars' surface to NCAS students in order to engage their creativity and problem solving skills. By sharing the raw data found by the rover, the goal would be to allow community college students to learn and experience what it is like to analyze data from a current Mars exploration mission. This would simultaneously motivate and empower community college students to seek higher education and careers in STEM.

6.4 Program Management Approach

Given the four team structure, the team conducted a survey to assign team leadership positions. From this, the following roles were assigned: Anusha Mody as Project manager; Riley Ellis as Deputy Project Manager; Yogi Patel as Lead Engineer; Patrick Kozyra as Lead Scientist; and Will Yarbrough as Lead Administrator. Then, the Project Manager designated the remaining team members to a specific sub-team. Certain team members were placed into multiple sub-teams based on their skill-set and interests.

The Lead Engineer, Lead Scientist, and Lead Administrator collaborated and communicated with each other, and the Deputy Project Manager collected information and progress levels from team leads to share with the Project Manager. Weekly meetings occurred with sub-teams, the entire team, the team mentor, and occasionally, with a combination of multiple sub-teams. The Project Manager and Deputy Project Manager worked together to determine weekly tasks and share assignments with team leads and sub-teams. Tasks were delegated to sub-teams, and then assigned to specific people within each sub-team. It was the priority of team leads to ensure their sub-teams completed the assigned tasks.

There were various issues that arose during the course of the project. Firstly, and most prominently, there was a lack of communication and participation. To combat this, the Project Manager and Deputy Project Manager established weekly meetings, occasional check-ins with individual team members, and sent a survey to gauge comfortability, concerns, and successes. The Project Manager also implemented team-bonding sessions with online games to play to promote openness and comfortability within the team. All leads also encouraged the continual use of the team's Trello board. The Lead Administrator also struggled to get responses from sub-team members. The Project Manager reached out to discuss the issue and suggested that the Deputy Project Manager assist the Lead Administrator with assembling the team for a meeting. Lastly, as the project progressed, team members became stressed or overwhelmed with deadlines and workloads. The Project Manager and Deputy Project Manager provided emotional support, encouraged a supportive and understanding environment, and offered additional help with task completion.

7 Conclusion

L'Space Invader's 2020 mission aims to determine if rock types in Jezero Crater preserve past or present signs of life through the use of a mobile rover. The rover will analyze soil samples to find evidence of biosignature preservation in an effort to develop humanity's understanding of exobiology and the possibility of martian life. The lander system includes Rocket Assisted Descent (RAD) motors, a RAD 750 Processor, TRN-LVS for terrain navigation, a Mars Exploration Rover (MER) Parachute Deployment System, a Disk Band Gap (DBG) parachute, N_2 gas generators to assist the impact stage, cold gas Reaction Control System (RCS), a Miniature Inertial Measurement Unit (MIMU), and a Phenolic-Impregnated Carbon Ablator (PICA) heat shield. The payload consists of one Acousto-optic Tunable Filter (AOTF) InfraRed (IR) Spectrometer, two RAD 750 Processors, one NavCam, one Helios Deployable Antenna, and a Silicone Thermofoil Heater System, each of which support the various subsystems of the rover. The main subsystems are power, thermal, navigation, data acquisition, and communications and data handling.

The team stands confident that the mission has sufficient funding, is steady with an achievable schedule for launch in October 2020, and will meet the technical and scientific requirements of the project. The team has expressed concerns about the achievability of the two objectives and the design stability of the Entry, Descent, and Landing system, specifically the slow maturation of ideas and slow release of technical drawings. This may affect the project's ability to meet technical requirements, stay consistent with the optimistic project schedule, and other possible development obstacles that may higher the overall risk of the mission. L'Space Invader's management team will need to address these issues to satisfy the mission's scientific objectives, meet project milestones, and stay within the budget capacity.

The project entered Phase C, the implementation phase, in October 2018, has concluded the Preliminary Design Review in May 2020, and is moving forward to the Critical Design Review to ensure that the entire system can proceed into fabrication and testing. The CDR includes the finalization of designs through the determination of system and subsystem requirements, peer reviews, developed plans for test, evaluation, and integration, and completed analyses.

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