L'SPACE Academy Team 37.5

Preliminary Design Review

April 18, 2019

MaRI-SIW

Martian Radar Imager for Subsurface Ice Water

A Close-Range Martian Application of Heritage Subsurface Water Stock Characterization Methodologies

I) Summary of PDR Report

1.1 Team Summary

Project Managers
Christi Cummings - Pikes Peak Community College, Colorado
John Mark Readle - University of Colorado Denver, Colorado

Deputy Project Managers
Ghanem Alatteili - Westminster College, Utah
Joshua Irwin - University of Colorado Boulder, Colorado

Documentation and Budgeting
Brieana Molinari - Colorado Mesa University, Colorado

Engineering and Design
Connor Catus - Casper College, Wyoming
Jynette Tigner - University of Utah, Utah

Entry, Descent, and Landing Design
Lowell Hanson - Metropolitan State University of Denver, Colorado

Landing Site Coordinator
Danielle Weldon - University of Colorado Colorado Springs, Colorado

Scientific Objectives and Instrumentation Design
Justin Tackett - Brigham Young University, Utah
Peyton Willis - Pikes Peak Community College, Colorado

1.1.1 List of Universities Represented by Team 37.5

Brigham Young University, Utah
Casper College, Wyoming
Colorado Mesa University, Colorado
Metropolitan State University of Denver, Colorado
Pikes Peak Community College, Colorado
University of Colorado at Boulder, Colorado
University of Denver at Denver, Colorado
University of Utah, Utah
Westminster College, Utah

1.1.2 List of Locations Represented by Team 37.5

Boulder, Colorado
Casper, Wyoming
Colorado Springs, Colorado
Denver, Colorado
Grand Junction, Colorado
Salt Lake City, Utah

1.1.3 List of Relevant Experience Brought by Team 37.5

Ghanem Alatteili: Payload, atmosphere research, landing, and landing.

Connor Catus: Excel, Matlab, Maple, Space Systems, Robotics

Christi Cummings: Project Design, Scheduling Coordination, Project Management, Documentation

Lowell Hanson: Orbital Mechanics and Astrodynamics, Computational Modeling.

Joshua Irwin: Systems Engineering, Solidworks, Mechanical Engineering, Design and Fabrications, Management, Technical Writing

Brieana Molinari: Software Design, Coordination

John Mark Readle: Software Development, Embedded Systems Engineering, Electrical Systems, Project Management, Documentation

Justin Tackett: Soldering, Mathematics, Eagle, Brown Dwarf and Traveling Exoplanet Research, Machine Shop

Jynette Tigner: Material Science Engineering, Chemistry, NCAS

Danielle Weldon: Biology, Environmental Science, Mechanical Engineering, NCAS.

Peyton Willis: Calculus, Chemistry, Lab Experience

1.2 Entry Summary

1.2.1 Entry, Descent, and Landing Sequence

Entry

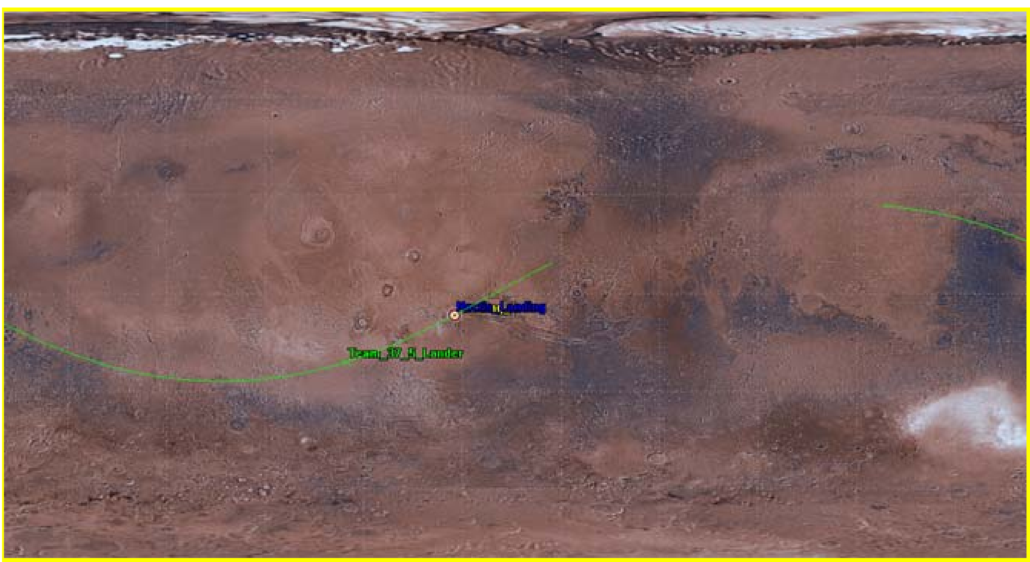
Initially MARI-SIW will be co-located to an orbiter at an altitude of 250 km, (semi-major axis equal to 3646.2 km). The orbiter will maneuver the lander into a ballistic trajectory to our landing site at Noctis Landing. According to our models with

Systems Tool Kit (STK) an initial orbit with the following elements would satisfy our entry requirements.

Semi-major Axis	3646.2 km
Eccentricity	0
Inclination	23.25 deg
RAAN	0 deg
Argument of Periapsis	0 deg
True Anomaly	0 deg

Figure 1. Initial Orbit

It should be stressed that these orbital elements have been found to be viable but are not meant to suggest the optimal approach trajectory and further modeling is necessary.

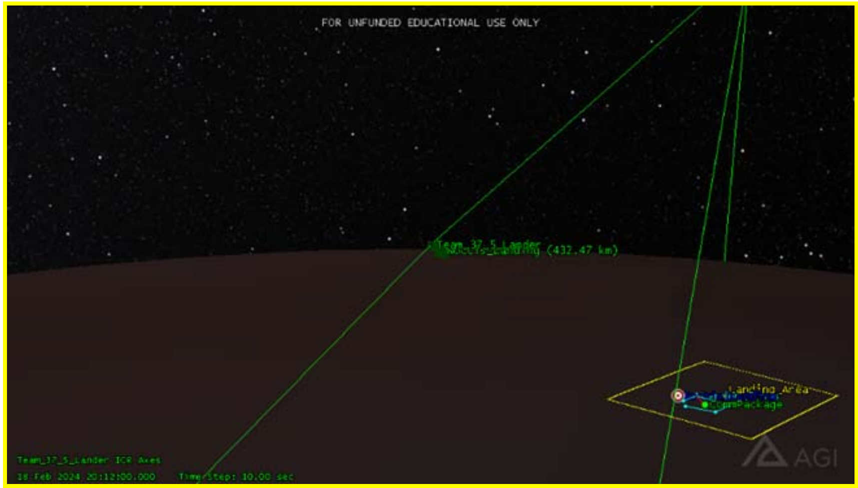


Figures 2 and 3 : Proposed insertion orbit towards Noctis Landing (2d and 3d views) (Generated using AGI STK)

Descent:

The probe will be on an unguided ballistic approach as it enters the Martian atmosphere. The atmosphere, although very thin relative to Earth's, will cause a significant decrease in velocity and require heat shielding on the probe. This descent was modeled from an entry elevation of 110 km and initial velocity from orbit of 5.6 Km/s and an entry degree of 13 degrees below horizontal. These numbers were determined by using analogous examples from Phoenix and other Martian missions. According to our trajectory computational model, this descent should begin 336.4 km prior to the landing site at Noctis Landing, measured along the surface.

Over the first 200 seconds of descent, under the frictional forces of the expanded heat shield, the probe will decelerate to a velocity of about 143 m/s before the parachute is deployed. The parachute should be timed to deploy at an elevation of no less than 8.6 km, similar to other missions. The small mass of our probe would have allowed us to deploy at a lower altitude and still have sufficient time for the parachute to slow the craft, and therefore subject the landing trajectory to less weather-related uncertainty, but the terrain surrounding Noctis Landing makes this a high-risk decision. While Noctis Landing's surface is near the mean Martian elevation, the surrounding terrain exceeds 8 km in some areas. A delay in opening the parachute could result in catastrophe if the initial trajectory is off by a slight margin.



Figures 4 and 5: The lander on its initial descent trajectory towards the landing location. (Generated using AGI STK)

Figures 6 and 7: Proposed insertion orbit towards Noctis Landing (2d and 3d views)
(Generated using AGI STK)

This is modeled in greater detail in section 4.2.3.

Landing:

Our model shows that a parachute with a surface area of 15 m^2 and a drag coefficient of 1.75 will reduce the velocity to approximately 9.7 m/s prior to landing. This impact speed would still pose too great of a risk to our lander and another method must be used to further slow the probe. Due to the thin Martian atmosphere, doubling or even tripling the size of the parachute would not result in an acceptable impact velocity. For this reason, we are including a secondary velocity arrest system consisting of a four-booster retropulsion system which fires shortly after disengaging the parachute to reduce the velocity from 9.7m/s to 2m/s for a soft landing.

While the optimal retro-rocket system would include stability/attitude control, we found that the implementation of this system is outside the mass and computational budgets of our lander. For this reason, we decided to instead implement a solid-fuel retropulsion system. We elected to use heritage Ammonium Perchlorate Composite Propellant (ACPC) solid fuel boosters as our benchmark for determining the **Δ**V output capabilities of our own ACPC retropulsion system. ACPC is widely used both in space industry and in model rocketry for its reliability and high thrust output. In order to accurately choose the correct Specific Impulse (ISP) value to use in our calculations, several considerations had to be made. Firstly, the shape of the nozzle drastically affects the performance of the rocket. For these early calculations, we made the assumption that our finalized thruster nozzle design will be able to mimic the efficiency of the large boosters from which this data is collected. Secondly, the ISP of a rocket changes significantly based on the pressure gradient outside the nozzle, meaning that the thrust output of the booster will vary depending on whether it is in Earth's atmosphere, the vacuum of space, or the surface of Mars. Since the atmosphere of Mars is 1% the density of Earth, it seems appropriate that the I_{sp} on Mars will be to within 1% accuracy of the expected vacuum I_{sp}. Given these considerations, we used the finalized value of 285.6s, based off the vacuum I_{sp} data of the Titan IVX booster (Northrop Grumman Catalog, 2018) for the standard expected I_{sp} of APCP rockets. Note that although this data comes from a rocket many orders of magnitude more massive than the scale of our own, I_{sp} should theoretically scale, and difficulties in finding data sheets for scaled down model rocketry applications led us to use this as the most readily available quantification of specific impulse for APCP propulsion systems. The calculations are outlined below.

$$Rocket\ Equation:\ V_f = V_e\ ln\left(\frac{M_0}{M_f}\right)$$

$$M0 = initial\ mass = 5kg$$

$$Mf = final\ mass = ?$$

$$Isp\ (Specific\ Impulse) = 285.6s$$

$$V_e = (Isp)(Gravitational\ Constant) = 2782m/s\ *$$

$$V_f = final\ change\ in\ velocity = 7.7\ m/s\ (9.7m/s - 2m/s)$$

Solving the rocket equation yields a M_f of 4.986 kg and a reactionary mass of APCP solid fuel of (M_0 - M_f) 13.81g of solid rocket fuel required to change the velocity of the rocket from 9.7m/s to 2.0 m/s. Even when combined with the necessary structural integration, this mass is well within our mass-budget allowance for the descent. The lack of control associated with this system, however, will bring additional complications such as ignition reliability, stability concerns, and timing requirements. We have considered these challenges and addressed them in our failure modes analysis (Section 4.5.2) and testing criteria explanation (Section 4.4)

Upon successful reduction of flight velocity to ~2m/s just above the surface of Mars, MARI-SIW will absorb the impact of landing on the diagonal underbelly wheels. As these wheels are already designed at a 45 degree angle from the ground (See Section 4), we plan to integrate shock absorption mechanics into the wheel attachment points as a way to mitigate impact. Alternatively, as the design develops, if we find the ability of the wheels insufficient to handle the material stress of impact to within a +-1m/s tolerance of 2m/s, we are prepared to alter the design to include retractable, shock-absorbing landing struts to ensure safe touch-down.

1.3 Lander and Payload Summary

1.3.1 Description of Each Mission Element (Probe and Science Payload)

The primary science payload aboard MaRI-SIW payload consists of a single instrument, ground penetrating radar. It will be an adapted version of the Radar Imager for Mars' Subsurface Experiment (RIMFAX), which is a ground-penetrating radar (GPR) that sees geologic features under surfaces, currently being developed for use in the MARS 2020 mission. This instrument will provide our team with the ability to detect underground water-ice at depths that can reach about 30ft (10 m) with vertical resolution from 15 to 30 cm. Depending on our geological land site, the RIMFAX will provide us wide ranges of geologic features and beneficial information about the surface history of water existence. While the official RIMFAX weighs approximately 3kg, given the mass constraints of our lander, we have made the decision to scale the instrument down to fit the scope of our mission. According to our payload criteria, the mass of the RIMFAX-GPR weighs 2kg including the antenna and occupies the volume of (19.6 × 12.0 × 0.66 cm). The RIMFAX System is contained within the protective shell of the MaRI-SIW rover. The radar operates by transmitting radio waves into the ground at a frequency range of 50-1200 MHz and then detect the reflected signals as a function of time to reveal structural signs of water-ice existence.

Lander

The team lander is composed of eight basic elements: the lander body, heatshield, 4 booster rockets, battery, CPU, solar panels, battery, and a parachute system.

List of Components

Elements	Mass	Size
Heat Shield	2kg	Retracted: 50cm x 50cm Expanded: 500 cm diameter
4 Rocket Boosters	0.05kg (each)	Diameter: 2cm Length: 5cm
Body	1.1 kg	(0.210*0.210*0.121 cm)
Drive Motors	0.160kg	2cm diameter
Solar Panel	0.035kg	N/A
parachute	0.65kg	Diameter: 100 cm
CPU	0.350kg	15.24cm x 11.23cm (internal)
Battery	0.090 kg	46.0mm x 71.0mm x 18.9mm (internal)
Transceiver	0.025kg	N /A

1.3.2 Description of Surface Experiment Deployment System

MARI-SIW's surface experiment deployment system is based on the team's payload criteria. Our payload criteria consist of only the lander. After the lander has successfully landed onto the Martian surface, the lander will incorporate a low-gain antenna to receive data signals that regards landing commands from our team. The Low-gain antenna provided high-altitude data for entry science, vehicle control and parachute deployment. The lander antenna will then switch into operation at separation of the heatshield and provided altitude data for guidance and control, terminal descent.

1.3.3 Summarize Experiments and How the Science Obtained will Map to the Overall Mission Objectives

MARI-SIW, short for Martian Radar Imager for Subsurface Ice Water, is a Mars lander designed to provide us with insight of the Martian subsurfaces. The MARI-SIW lander and payload is constrained within the mass of 5 kg and size dimensions of (50 x 50 x 50 cm). The sum of the masses of the landing equipment and payload exceeded about 4.61 kg. In addition to the mass limitation, our payload occupied the total volume of (42.47 x 47.9 x 47.9 cm). The MARI-SIW lander and payload is now designed to provide us with data of scientific evidence of water existence on the subsurfaces of Mars. MARI-SIW contributes to the main objectives of this mission regarding the exploration of mars.

Our goal is to identify the features that would be signs of water existence on the landing site. Our payload science instrument is inherited from the Mars Mission 2020, and orbital mechanics and landing operations were modeled similar to the Phoenix Mars Lander. Once the lander spacecraft lands safely on the landing site, the the modified version of the RIMFAX will be deployed on the payload segment of the mission which will fit within all the necessary elements of the lander for a successful mission goal. We planned the payload to be able to analyze subsurface material compositions from various depth using the RIMFAX-GPR technology in order to confirm water existence. Having a greater insight of the material's composition underground will help us understand the evolution of the Martian Atmosphere which will lead us to significant evidence of water existence. The detection of water volatiles means there is a possibility of biological life on Mars and capability to utilize this H20 stock as a resource for future missions.

II) Evolution of Report

2.0 Changes Made Since the Initial Concept Was Identified

Important Note: Due to falling rates of member participation, Team 37.5 (previously teams 37 and 38) experienced a team merger on March 7, 2018, less than one month before the design deadline. Both teams were tasked and rewarded with the opportunity to combine the best of both groups of work into one final design. The previous considerations from pre-merger teams 37 and 38 will be included in this section as documentation of evolution.

2.1.1 Changes Made to Entry Descent and Lander Criteria (EDL)

The initial concept from team 37 was a parachute-only arrest system. However, this idea was abandoned given that the atmospheric density of Mars is approximately 100 times thinner than that of Earth and does not provide enough drag to bring the lander to a soft stop.

Team 38's initial EDL concept also involved a parachute but was additionally supported by an airbag landing system like the ones used on Pathfinder or MERS. One concern with this proposed method was the decreased accuracy of landing, as the air bag system from a ballistic trajectory can lead to continuous high-speed bouncing on contact with the surface.

Mass constraints were ultimately the deciding factor on our final choice to combine two EDL systems: a parachute for atmospheric deceleration and a solid rocket motor system to reduce the final flight velocity before touchdown. The decision to switch from airbags to solid rockets was made because the airbag system turned out to be overly costly in terms of mass. Airbag material is dense, and the additional requirement of a gas generator put the system out of our mass allowance. Solid rocket motors, our final choice for near-Mars deceleration, are lighter and demand considerably less computerization and integration than controllable liquid fueled retropulsion systems. Initial investigations showed that for many Mars lander missions, attitude-controlled retropulsion systems often account for approximately 30% of the mass of the lander, which was outside our mass budget. For this reason we opted for the solid fuel booster system, which is considerably less complex, lower in hardware requirements, and lower in mass. However, development of the project using solid rocket motors will require allocation of resources into thoroughly testing the entry profile and ensuring that measurements are accurate and burns fire correctly. We have accounted for this in our budgeting and scheduling timeline.

2.2.1 Changes made to Payload Criteria

Several aspects of the Payload Criteria where the design experienced considerable evolution are mobility vs. stability, materials, and functional operation requirements. Before the merger, each team had different ideas about what the payload was going to accomplish and how to pursue the mission success criteria of quantifying 100 cubic meters of Martian water stock.

The original concept from team 37 was to chemically prove the presence of water with a mass spectrometer. However, it became apparent that without an additional underground sample apparatus, the analysis of surface samples would not be capable of characterizing subsurface water stock. Additionally, the weight limitations of the mass spectrometer were unrealistic for the scope of the design. The second payload design idea revolved around using a Ground Penetrating Radar (GPR) for scanning subsurface material permittivity. Team 38 conceptualized a drilling mechanism to collect potential ice samples, but also found challenges with characterization of water stock beyond the sample available directly beneath the lander.

After the team merger, GPR was finalized over the drill and the mass spectrometer for its lighter-weight profile and ability to scan at depth for subsurface material characteristics and properties.

Initially, the team considered using a stationary GPR instrument. We hypothesized that if we could leave the GPR stationary and tilt or spin it, by correctly interpreting the data we could develop a cone-shaped map of subsurface composition However, we encountered difficulty finding documentation proving that this could be done and preferred to stick to heritage methodologies. Ultimately, we realized that the rover needs to be mobile over several meters in order to collect data at intervals along the surface, so we began researching mobility options, which ultimately lead to the diagonal-wheeled final design. A multitude of brainstormed ideas were developed and scrapped as part of the development process.

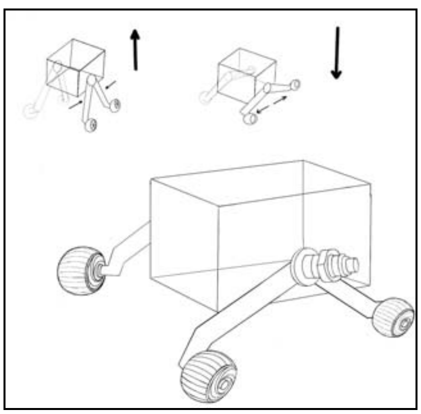


Figure 8. Modular Clearance Drive System

Figure 9. Spring-Force Drill-End

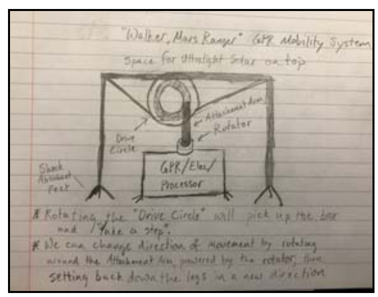


Figure 10. “Walker” mobility System

In addition to the mobility requirement, another design change was implemented when we discovered that the antenna for our selected GPR model needed to be 60 cm above the ground to acquire accurate data. As we had previously planned on incorporating the antenna into the body, we considered giving the payload extendable legs to lift the entire rover up to 60cm, but ultimately decided to use an antenna that unfolds out of the top of the lander when sampling data.

As with every aspect of the design, in order to meet our mass constraints, we had to readjust our original instrumentation design from 3kg allowance to 2kg to leave enough room for landing and operations hardware. Because we wanted to use a heritage GPR system, we instead devoted a development budget to scaling down the original system rather than starting from scratch with a new version of the instrument.

2.3.1 Changes Made to Mission Experiment Plan

Since the finalization of the GPR as our instrument of choice, the Mission Experiment Plan has adapted to meet the challenges of data collection. As discussed in section 2.2.1, the original plan was to rotate the instrument to conically scan the area under the lander for evidence of subsurface water stock. However, several issues became apparent with this methodology. Firstly, our research revealed a significant lack of documentation about the ability to retrieve interpretable data from using GPR at an angle. Secondly, even if we were to successfully quantify the water in a cone under the landing site, the mission would end after one set of scans, whether we determine that water can be found under the surface or not.

For this reason, we elected to supplement our rover with a lightweight mobility system to add versatility and reusability in the spirit of the Evolvable Mars Campaign. Once the rover completes a vertical linear scan in its landing spot, it can move a short distance and repeat the process. As the rover collects a greater number of data points, the numerical extrapolation of the subsurface water stock will become a richer and more reliable model upon which to plan for future human visitation.

III) Science Value

3.0 Highlight the Science Payload and Value to Mission Objectives

The purpose of our payload is to characterize Subsurface water ice as either H_2O or CO_2 and establish a wider knowledge base of the resources available at our landing site for future research.

Using a transmitting and receiving antenna MARI-SIW will send send radio frequency electromagnetic waves into the ground and capture a numerical image of the Martian subsurface at a depth of up to 10 meters.

Using Ground Penetrating Radar (GPR) technology, our payload will identify Martian subsurface material composition and geological formations, allowing for the categorization of subsurface ice in terms of abundance, chemical state, and composition. By extrapolating data from a variety of sample locations we can begin to piece together a clear picture of the Martian subsurface.

3.1.1 Describe Science Payload Objectives

The science payload objectives are to quantify the ice quality, type, depth, and accessibility within at least 1 square meter of the landing site. This will be for the

purpose of identifying practical locations where excavation of water could take place for a variety of purposes, including water to sustain human life on Mars as well as for a variety of other scientific and engineering purposes that will be key to the colonization of Mars.
Ice Quality: Ice quality will be measured by the ratio of regolith to ice for a cubic area.
Ice Type: Using the differences of dielectric constants of CO2 and H2O, ice will be distinguished by type.
Ice Depth: The ground penetrating radar that we will be using will be able to determine how deep pockets of liquid or solid H2O and CO2 are located.
Accessibility: Based off of the other criteria, namely, quality, type, and depth, the team will determine how accessible the target materials are to future missions.

3.1.2 Payload Success Criteria

The success of this mission depends on adequate measurements that find the presence of solid or liquid H2O or solid CO2. Adequacy will be determined by four criteria: Signal-to-clutter ratio, signal-to-noise ratio, spatial resolution of the area, and depth resolution of the area (Daniels).

Signal-to-Clutter Ratio

Within the signal the receiver gets, there will be a fair amount of signal received from objects that are not within our search parameters, some obvious examples being the regolith and other anomalies present therein that are not liquid or solid H2O or CO2. Typical reduction methods in other radar systems involve using MTI (Moving target induction) as a way to reduce the clutter. This method essentially uses the Doppler effect to separate the two, which is not applicable to stationary targets, such as is this case. One advantage, however, that typical radar applications do not have is the difference of the dielectric constant between the regolith and the targets of H2O and CO2. Because of these differences, the signal received will behave differently than that of the signal of the regolith. Dielectric constants change with temperature, however, and so depending on temperatures, this difference could potentially be more subtle than at other temperatures (Daniels).

Signal-to-Noise Ratio

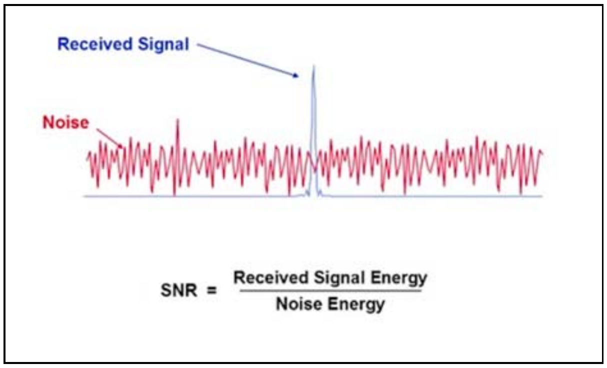


Figure 11. Signal to Noise Ratio. Surface Penetrating Radar. By D.J.Daniels, 1996

When receiving wavelets in a radar system, there is noise that is produced from a large variety of sources. This differs from clutter in its inconsistency. Where clutter is in some ways predictable, noise is produced by things such as galactic cosmic rays, the sun, the atmosphere and a variety of other sources. This noise is erratic and inconsistent, as shown in the figure above. A threshold must be defined, above which a signal can be identified. Sometimes, however, the noise may rise above this threshold creating a false alarm. Other times, for a variety of reasons, the signal may be lower than the threshold and may be missed. To mitigate that, one can integrate several measurements of the same area, and in doing so the targets signal will increase consistently larger than the erratic noise, which will increase but only slightly (Daniels, 1996)

Spatial Resolution

Spatial resolution depends essentially on pixel density of a produced image. If the image has a greater amount of pixels in an image, it has greater resolution. This is important in order to find exactly where ice and liquids begin and where regolith stops, and will be key in producing data more useful than that which we currently have. The data which we currently have has very low spatial resolution due to the nature of GPR technology in space, and is hard to use in practically determining the location of where to dig to obtain potential ice and liquids.

Depth Resolution

Depth resolution depends on the minimum axial distance that two points in data can be resolved. In other words, it depends on how close two objects can be to each other in depth and still be distinguished as two separate objects. This will be important in determining exactly how deep one would have to go in order to obtain the materials they are looking for.

If all of these criteria are met within the bound set, then the data will be considered adequate. At this point, tables of the data will be graphed and be interpreted in a tool such as JMARS (Daniels,1996).

3.1.3 Landing Site and Method of Investigation

The GPR system and movement capabilities of MARI-SIW will allow us to confirm and quantify suspected subsurface liquid water in the region. “Noctis Landing” is the lowest-altitude location on Mars that straddles both the Tharsis region (above average geothermal gradients) and Valles Marineris (minimal crustal thickness from surface (valley floor) to a subsurface liquid water table. Noctis Landing has the potential for being an ideal site for eventual deep drilling on Mars to access deep subsurface liquid water and potentially encountering extant life” (Lee et al., 2015). MARI-SIW’s mobile capabilities will allow it to transverse this location and retrieve multiple readings in order to gain a superior data-set than a stationary lander would provide. This data-set would give us insight to better be able to: quantify subsurface water ice in the Noctis Landing area, characterize ice as either H_2O or CO_2 , Measure depth of the ice layer, determine the trafficability in the region, and establish a wider knowledge base in a desirable area for future research.

After reviewing many sites researched by NASA as possible habitable zones, Noctis Landing became our chosen site to research. There were several reasons for this choice:

- This area is close to many major scientific regions of interest for future study such as Valles Marineris, Olympus Mons, and ancient crust sites;
- Despite being an area surrounded by craters and canyons, this site is an ideal future landing site as it’s level, stable, and close to the equator. Being close to the equator is a somewhat of a small gamble though, as a lot of glacier activity is found at mid-latitudes but being on the equator would facilitate future human missions, more specifically, a more temperate climate to exist in during their mission and an easier launch off of Mars;
- Despite its closeness to the equator, it’s had promising activity pointing to the presence of H_2O such as recurring fog and polyhydrated sulfates (Visaya, Day, & Lee, 2019) and there’s also a rich supply of hematite (see Figure 2). “Neutron spectrometry also suggests hydrogen is present within the topmost 0.3 m or so of 4 to 10 wt% WEH (Water Equivalent Hydrogen)” (Lee et al., 2015);
- This site was also investigated and presented during the "First Landing Site/ Exploration Zone Workshop for Human Missions to the Surface of Mars" in 2015, making it a well researched region (Lee et al., 2015). The image and description following directly after this list was presented at the “First Landing Site/ Exploration Zone Workshop for Human Missions to the Surface of Mars” and provides a good scope of the area in question;
- There’s a wealth of HiRISE, THEMIS, and CRISM images of this area accessible through JMARS which is incredibly useful as JMARS provided free access to the data collected by the orbiters around Mars. This GIS data is critical to mission planning and beginning our data analysis. One such image is shown below in figure 2;

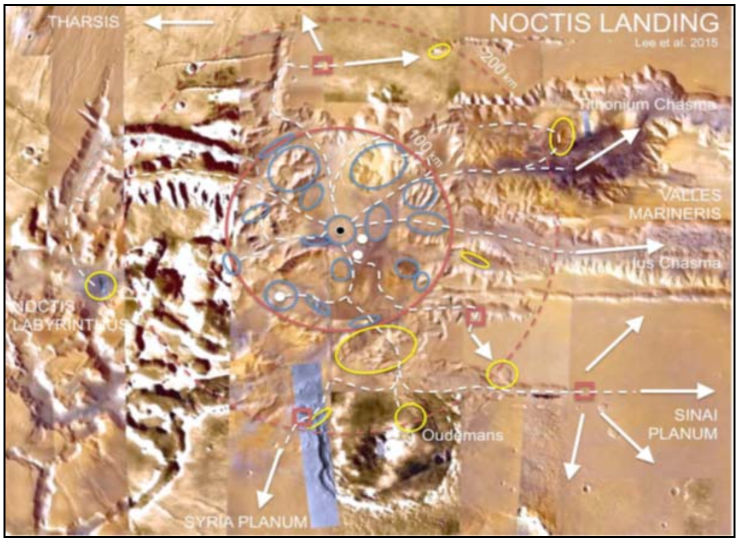


Figure 12. Map of the Noctis Landing LS/EZ
The solid red circle marks the distance of 100 km radial range from the Landing Site (LS), defining the primary Exploration Zone (EZ). The dotted red circle marks 200 km radial range from the LS. Areas circled (or ellipsed) in blue are high value science targets located within the primary EZ. Areas outlined in yellow are high-value science targets located outside the EZ, but within 200 km radial range from the LS. White dotted lines represent potential paths for pressurized rover traverses. White solid circles mark locations offering potential resources (hydrated minerals, iron and sulfur-bearing minerals, loose regolith). Red square boxes mark potentially trafficable access points to [surrounding] plateau tops. White arrows point to general directions for further regional exploration beyond 200 km radial range [from] the LS. (Background images [from] NASA and ESA). (Lee et al., 2015).

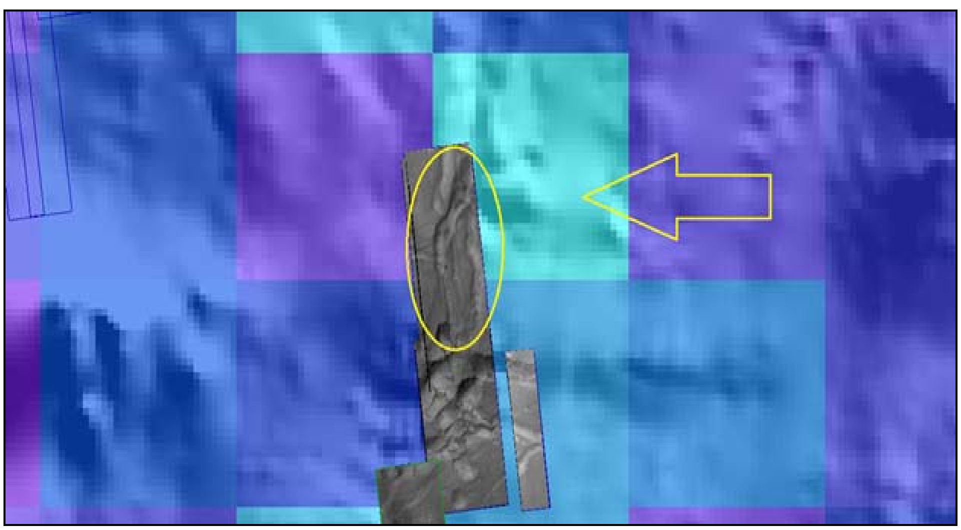


Figure 13. JMARS Image of Noctis Landing (Christensen, n.d.)
The circled area shows possible water flow evidence (photo is a HiRISE image). The arrow the block is pointing to shows some of the richest supplies of hematite on Mars. The presence of hematite indicates water either is or was present in this region. Most of Mars is covered in blocks that are that light purple color you see, averaging 0.02 areal fraction, but this area indicated measures 0.07882 areal fraction.

When teams 37 and 38 merged to Team 37.5 we were faced with choosing between the original two teams' landing site choices: Noctis Landing, our final choice, and Ismenius Cavus, which used to be a huge lake. Ismenius Cavus was also looked into during the "First Landing Site/ Exploration Zone Workshop for Human Missions to the Surface of Mars" in 2015 and has been heavily investigated and documented, including HiRISE images of the region. It has multiple glacier-like formations in its surroundings; the description from JMARS says: "Martian glacier-like forms (GLFs) indicate that water ice has undergone deformation on the planet within its recent geological past" (Christensen, n.d.). The H_2O concentration weight was also high at ~5.1% with Mars' highest being ~7.4%. Ultimately, Noctis Landing was chosen as our final landing site due to its close proximity to many scientific points of interest, combined with strong data suggesting subsurface water ice.

3.1.4 Describe test and measurement, variables and controls;

Testing of our ground penetrating radar will include taking measurements on a patch of imitation regolith. The control for the experiment will be H_2O ice under up to 10m of regolith. For our variables we will take measurements with the H_2O ice, CO_2 ice, D_2O ice, and H_2O ice with impurities. We will then experiment with the depth of the ice under the regolith. We will take measurements at 1m below, 5m below, and 10m below. These experiments will give us a better idea on the limitations of our equipment and give us data to compare to the data we receive from Mars.

Test

The original RIMFAX has undergone testing, using arctic conditions as an analog (Hamran,2015). This instrument will undergo similar testing in such an analog, using the detection of brine as an analog for distinguishing ice types. A variety of different situations, using the following variables and controls, will be used to test the effectiveness of the instrument and see in what circumstances adequate data is obtained, and what can be done to maximize the production thereof.

Measurement

Ground Penetrating Radar data is typically stored and sorted in a format involving three types of scan that are called A, B, and C scans. Each has a different approach in measuring and individual benefits in data analysis. In this section we will strictly address how the data is stored. First we will define the different kind of scans.

The A scan is a single wavelength scan. It considered the range of *z* depths, while keeping *x* and *y* constant.

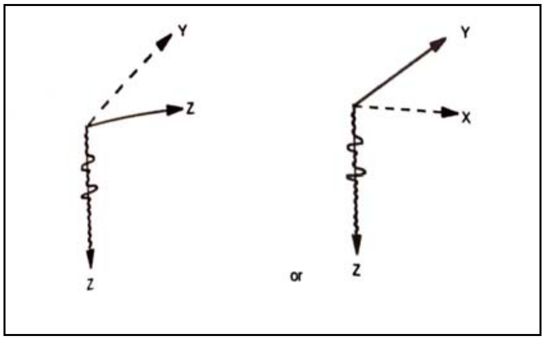


Figure 14. A-Scan. Surface Penetrating Radar. By D.J.Daniels, 1996

The B scan is an ensemble waveform set.This can be thought of most easily as a collection of A scans. In these scans, the range of *z* values are examined as well within a range of *x* or *y* values. As shown in this figure, this can be conventionally thought of to represent a plane of values perpendicular to the *x-y* plane.

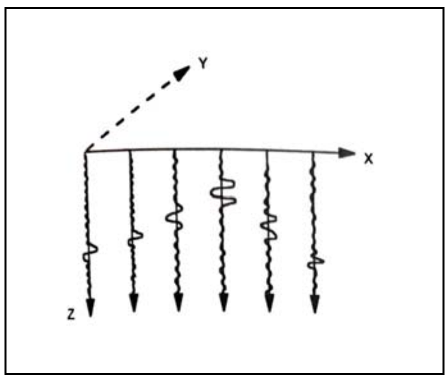


Figure 15. B-Scan. Surface Penetrating Radar. By D.J.Daniels, 1996

In a C scan, *z* is always held constant and *x* and *y* are considered as variables. This can be thought of as taking an arbitrary layer of depth and considering all the data points at that depth.

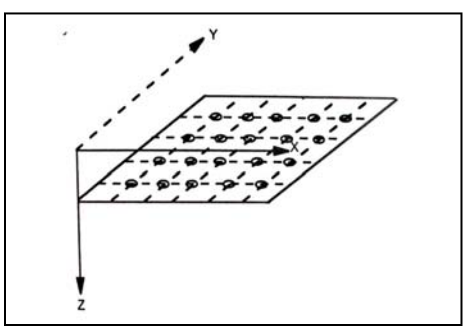


Figure 16. C-Scan. Surface Penetrating Radar. By D.J.Daniels, 1996

Variables

There are many variables that will play important roles in determining the success of the mission:

- First, temperature. Dielectric constants change with temperature (Dielectric Constants, n.d.). Studies will be done beforehand to determine the range of dielectric constants of CO_2 and H_2O in both solid and liquid states, as well for the regolith to define margins and be able to mitigate the risks of the variables.
- Second, presence of ground clutter and anomalies in the regolith. The effect clutter has on a signal is inversely related to the fourth power of the distance the clutter is from the antenna (Daniels,1996). This means that the closest clutter will have a significantly greater effect than those which are deeper in the ground. The amount of clutter will contribute, and as mentioned earlier, typical MTI methods will not be helpful in signal processing. Fortunately, our targets are towards the deeper end of our range, and so clutter will not have as significant of an effect as it could have.
- Third, total path loss. As the signal goes through the material, there are many factors that can contribute to the total loss of the signal in between transmission and reception. Among these factors include antenna efficiency loss, antenna mismatch losses, transmission loss from air to material, retransmission loss from material to air, antenna spreading losses, attenuation loss of material, and target scattering loss. Each of these present unique, complex problems that would need to be worked through in a lab, although they are not new problems, and have been faced by former missions and have been successfully dealt with (Daniels,1996).
- Fourth, Martian climate. Things such as a dust warm and the change in weather are variables that one must consider. A variety of scenarios could lead to several negative effects on the probe, ranging from inadequate data to damaging the instrument. Testing will be done in the analog to help ensure the success of the probe in even the most extreme conditions.

Controls

First, air composition. The air composition in the analog will be consistent, in a similar way that the Martian air composition will be consistent. With this control, we will be able to see how effectiveness changes with the variables, and be able to make similar extrapolations when the instrument is tested in settings that will better reflect Martian atmosphere, like vacuum chambers.

- Second, the materials of the instrument. The instrument materials, such as the composition of the antenna and the electronics, will be constant, and will be measured to see how they react under the different variables and the extreme conditions.
- The material being detected will be unchanging in the variety of tests, and will be measured to see how it changes under the variety of variables.

3.1.5 Relevance of Expected Aata & Accuracy/Error Analysis

Adequate data must meet criteria that was previously defined, and in order to maximize the amount of adequate data, several methods of signal processing will take place to reduce error and reduce data into something that can be interpreted.

Signal Processing

Signal processing is a complex subject, on which books can and have been written. An entire treatment on the subject will not be made here, rather we will touch lightly and generally on some of the techniques used in processing the data and making it adequate and ready for interpretation.

Zero-Offset Removal

Due to the wave nature of the signals, the average mean value of any A scan should be near zero, assuming that the amplitude probability distribution of it is symmetric around the mean value and is not skewed (Daniels, 1996). If a variety of circumstances (such as a DC offset or a RF time-varying gain being used) line up, it may result in a skewed data set, which can be taken care of using a variety of algorithms (Daniels,1996).

Noise Reduction

There are two simple ways that noise reduction can be done: either by averaging individual A-scans, or by storing repeated A-scans and averaging those (Daniels, 1996). This process will have no effect on clutter, but will reduce randomly produced noised (Daniels,1996).

Clutter Reduction

Clutter reduction can be done through “subtracting from each A-scan an averaged value of an ensemble of A-scans or B-scans taken over the area of interest” (Daniels,1996). Through doing this, and especially with the relatively small area of our mission, great caution must be taken in this process as to not accidentally remove targets from our data set. This caution depends on carefully picking the number of samples and the number of A-scan waveforms that we will be using for the process (Daniels,1996).

Frequency Filtering

High Pass filtering will be useful in this circumstance, and will help improve the signal-to-clutter ratio (Daniels,1996). Due to the stark difference of the target materials chemical composition and that of the regolith, this process will be particularly useful. This can be done by taking a Fourier transform of the signal, excluding the lower frequencies produced by clutter, and reconstructing the series to produce a signal with a better signal-to-clutter ratio.

3.1.6 Preliminary Experiment Process Procedures

There are many preliminary experiments that will need to be taken place before hand. We must determine the dielectric constants of solid CO_2 at Martian temperatures and pressures. This is key in distinguishing the difference between solid H_2O and solid CO_2. As the GPR transmits wavelets, the incoming wavelets received will reflect composition and location of a variety of materials. The thing that determines the differences is each materials' relative permittivity, which is quantified and sometimes referred to as a dielectric constant, a vacuum being 1 and other materials scaling up, with higher numbers correlating to higher levels of conductivity (Dielectric Constant, n.d.). Between regolith and substances like H_2O and CO_2 the differences are quite noticeable (ElShafie et al., 2012), making a difference more distinguishable in running tests. However, dielectric constants change with variables such as temperature (Dielectric Constant, n.d.). Because of this, experiments at a range of pressures and temperatures will need to be conducted before hand in order to minimize errors and maximize our capacity to distinguish between solid CO_2 and solid H_2O. Liquid H_2O has a significantly different dielectric constant in the range of ~70 at ~20 degrees Celsius (Dielectric Constant, n.d.) while that of solid H_2O and CO_2 is about 3.17 and 1.7, respectively (Cambridge, Simpson et al, 1980). Once a range of dielectric constants (or if, possible, even a function) is determined with tests using small vacuum chambers and appropriate equipment to measure dielectric constants. This experiment will be the most important to the success of the scientific objectives.

Other experiments will be conducted to ensure that the changes made to the RIMFAX do not compromise the scientific competency of the instrument. This will be done largely through the analog test, where the largest variety of environments within that analog will be used to maximize our understanding of how the adjusted instrument reacts.

3.1.7 Steps and Procedures to Integrate Communication with the Engineering Team to Optimize Science Return.

In order to fit within our prescribed 5 kg limit and dimensional constraints, optimization and integration will be key steps in assuring the success of the mission. There are several aspects that will be addressed in this process, and they will be communicated through email and regular meetings between engineers and scientists. The RIMFAX itself is more massive than both what is necessary and what is possible in order to achieve our scientific objectives, and so over the course of the development process, careful assessments will be made to determine what can be forgone in order to maximize the amount of mass that will be of use in the development of the lander. This will be done to ensure the safety of the arrival of the payload. Furthermore, restraints will be placed as to keep these reductions of the instrument from dropping below that quality which will yield adequate data that will fulfill our scientific objectives.

We have multiple weekly video chat meetings that all parts of the team can join. During these meetings the science team and engineering team can brainstorm different ideas and share concerns we have with each other and come to solutions together. On top of that, Justin, John Mark, and Connor shared an email thread where they discussed and tackled different issues concerning our probe.

IV) Decent and Lander Criteria

4.1. Selection, Design, and Verification of Descent and Lander Mechanism

4.1.1 Mission Statement, Requirements, and Mission Success Criteria

Mission statement
Our mission is to quantify Martian subsurface water stock and material composition of the Noctis Landing site; we will accomplish this by safely landing and deploying MaRI-SIW's ground penetrating radar instrument which takes samples from a variety of locations to quantify the presence and depth of water deposits within that area.

Mission Success Criteria
- Successful communication between Earth and the delivery system
- Timely release of ballistic probe/ lander
- Successful deorbiting of the lander
- Landing procedures followed
- Heat shield durability
- Deployment of parachute
- Initiation of rockets
- Soft landing of 2 meters/second
- Wheels remain sturdy and intact after landing
- Detachment of thrusters
- First commands of daily operations initiated to test payload functionality

In case of an emergency of system failure, troubleshooting and additional commands are necessary. If one rocket is not working per protocol, adjustments will be made to compensate. The lander must touch down with minimal damage to the payload for success. The payload must be fully functional after landing. Each criterion is vital in the landing process and actions to reverse component failures must be initiated in the "six minutes of terror" to ensure mission success.

4.1.2 Major Milestone Schedule

Milestone	Due Date
Project Initiation	February 2019
Preliminary Design Review	April 2019

Design Stage	June 2019
Design Verification	December 12, 2019
Critical Design Review	February 2020
Manufacturing	August 2021
Operations Review	December 2021
Test Readiness Review	January 2022
All Systems Assembly Testing	February 2022
Design Corrections / Retesting	May 2022
Launch Readiness Review	August 2022

4.1.3 Landing System Review

The landing system contains a heat shield to greatly reduce speed on entry and descent. It unfolds and expands accordingly. The heat shield detaches as the lander approaches the earth. The parachute deploys to further decelerate the lander. The parachute was chosen because it is lightweight and compact. The parachute alone is not enough to slow the lander down to two meters per second. Therefore, the simple rockets initiate to prepare the lander for a soft touch down. Simple rockets are light and doable for the remaining mass available. The thrusters are detachable to improve battery life and mobility of the rover. The wheels touch down first. They are situated at a 45-degree angle to help mitigate the force felt during the landing. The wheels are chosen over legs to reduce the overall weight. Lastly, the low-gain antenna is vital in the system to communicate the trajectory of the lander and to receive signals for course corrections and landing commands. The antenna was chosen for its miniaturized size and signal strength

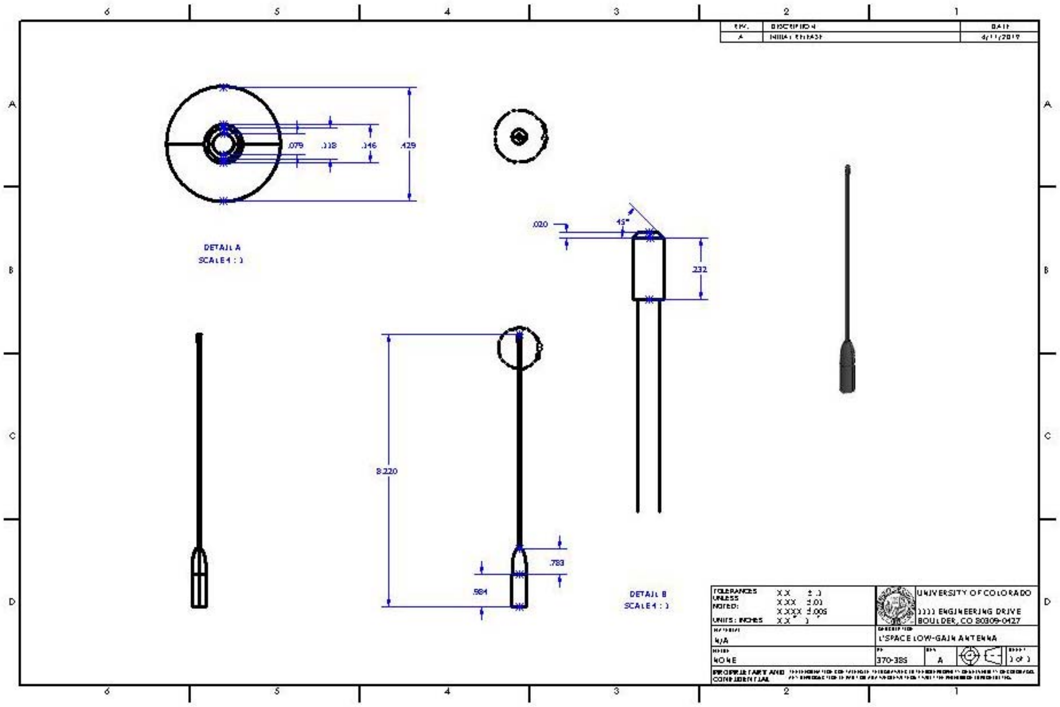
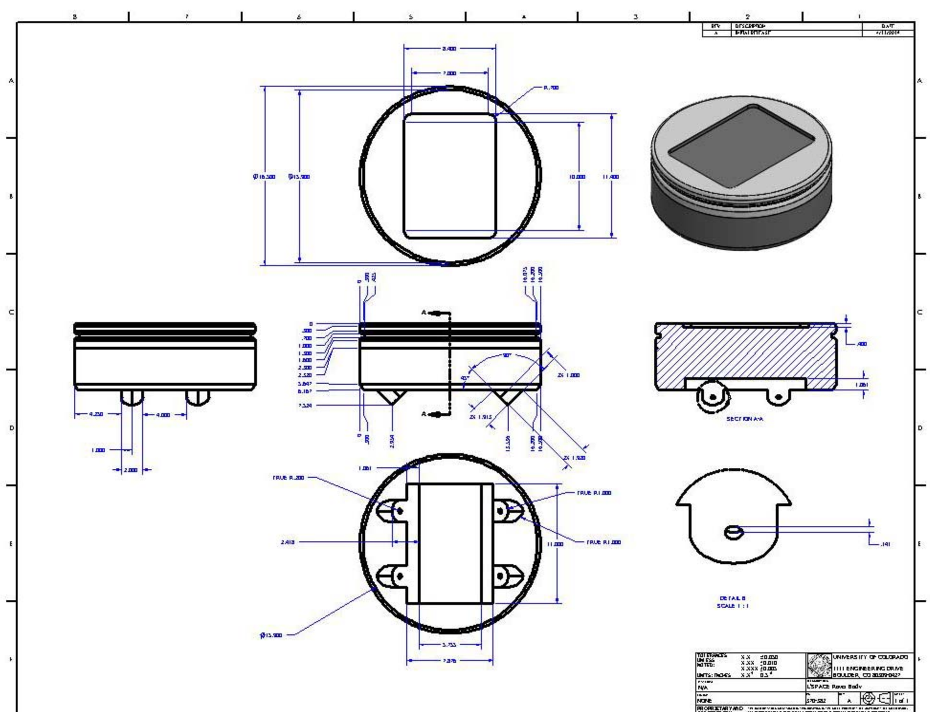
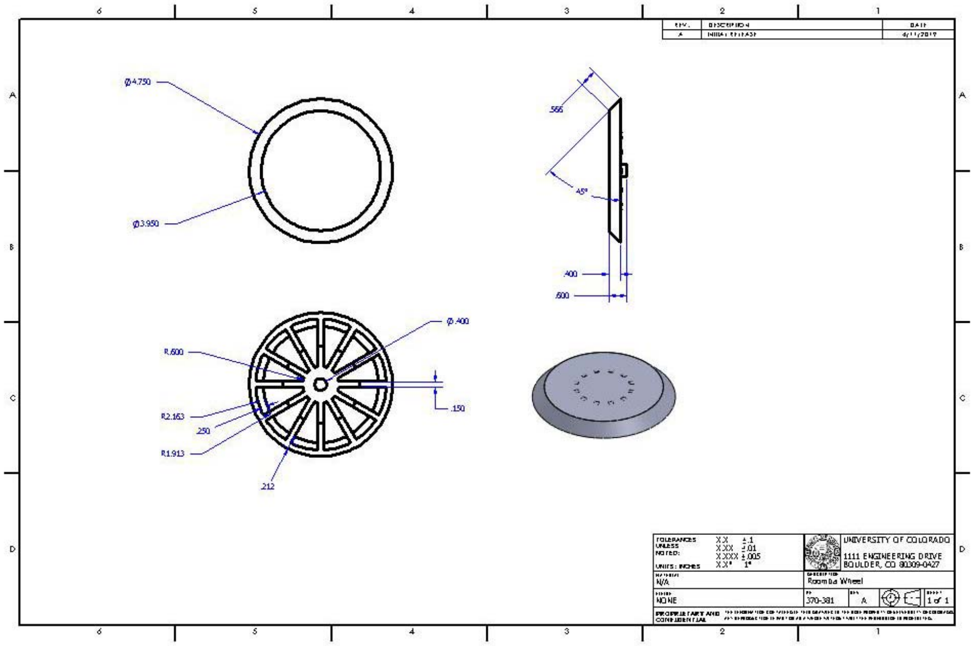


Figure 17. Antenna.
Design by Steven Minichiello. Retrieved from grabcad.com





Figures 18 & 19. MaRISIW Chassis Design
Created by Joshua Irwin based on Tertill Design by Franklin Industries

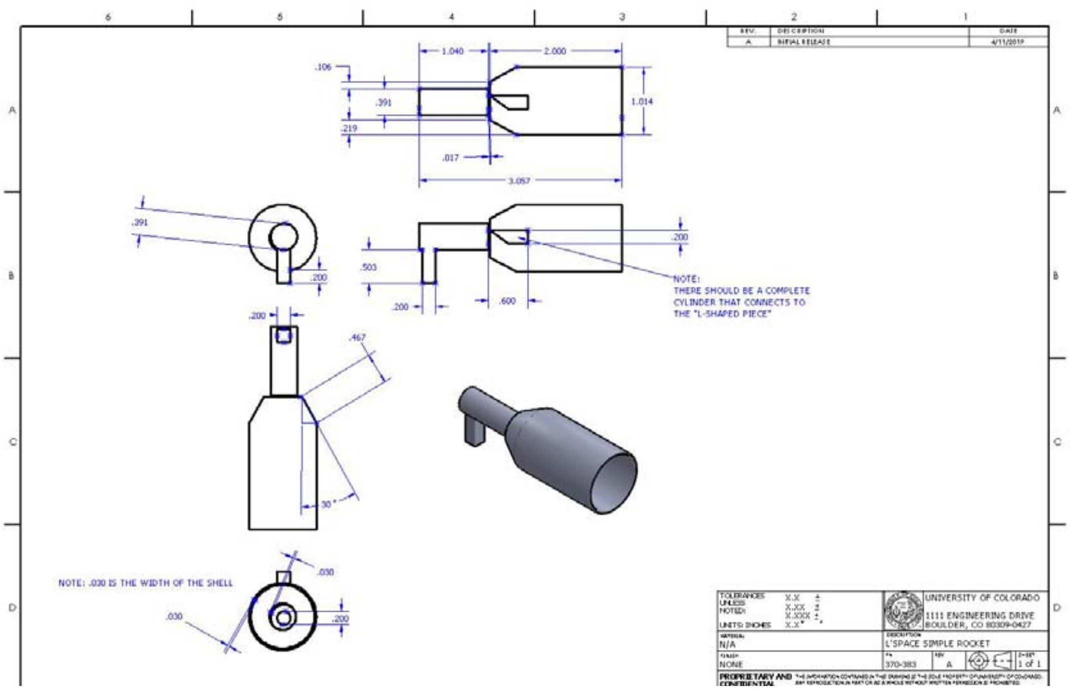


Figure 20. Thruster
Design by Abel Gomez. Retrieved from grabcad.com

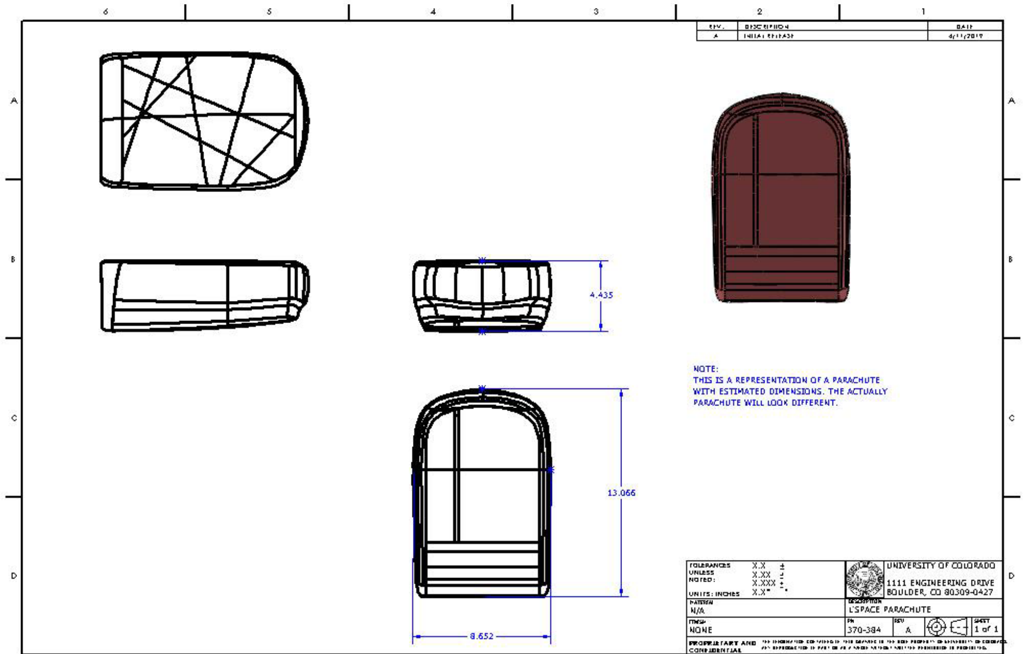


Figure 21. Parachute
Design by Daan Van Tuulder. Retrieved from grabcad.com

4.1.4 Subsystems Required to Accomplish the Mission

The subsystems required to accomplish the overall mission are organized and listed in the table in 4.1.5. Starting from the goal of the mission, the science package is the subsystem that will allow the rover to accomplish the mission objectives (refer to section 3.0). The subsystems that support that function are the electrical power system, consisting of rechargeable lithium-ion batteries in combination with high efficiency solar cells that have been developed at Glenn Research Center; the communication package, to send the required data back to Earth; and the command subsystem, which includes the on-board radiation hardened computer; and the mobility subsystem, allowing the rover motion.

Moving back from the rover operating safely on Mars are the subsystems that will be required to get it there. The temperature protection system and the descent & targeting are the subsystems that will allow for a safe landing of the rover. The temperature protection system is a deployable decelerator that will act as the heatshield. This assembly will be the current Inflatable Reentry Vehicle Experiment (IRVE-3) and will protect the structure when entering the atmosphere.The descent & targeting subsystem includes a parachute and retropropulsion thrusters that will slow descent and allow for a stable landing.

4.1.5 Subsystems Performance Characteristics and Verification Metrics

Subsystem		Assembly	Performance Characteristics	Verification Metrics
Electrical Power	x2	Rechargeable Lithium-Ion Cell Batteries (Saft 174565)	Provides energy storage for lander and rover functions	Working temperature -40°C to + 85°C Cycle life 2700 Max. continuous discharge current 8A Max. pulse discharge rate 16A
		High Efficiency Solar cells	Recharges the lithium-ion batteries for further function	Minimum efficiency of 30%
Temperature Protection System		Deployable Decelerator	Inflatable Reentry Vehicle Experiment (IRVE-3) heat shield system that protects structure when entering atmosphere	Areal mass of 5 kg/m^2 Maximum heat flux of 50-100 W/cm^2 Heat load capability of 15 kJ/cm^2
Communication		Low-gain antenna	Communicates with local communication package	Range of 75 meters
Command		On-board computer	3U cPCI Radiation Tolerant PowerPC SBC Processes data from science package	512KB boot EEPROM 1024MB fast DDR2 1GB (Expandable to 8GB) User Flash Memory Redundant 512k boot memory banks
Descent & Targeting		Parachute	Surface area of 15 m^2 and a drag coefficient of 1.75 will reduce the velocity to approximately 9.7 m/s	Deploy by 8.6 km Slow to approx 2 m/s
	x4	Retropropulsion	Polyurethane-bound aluminium-APCP solid fuel thruster	Specific impulse of 242 seconds (2.37 km/s) at sea level

Science Package		Ground Penetrating Radar	Retrieves data from below the surface of Mars	*(see section 3.0)*
Motion		Motor/ Drive	Receives direction from command system to drive rover	4 Wheel Drive

Figure 22. Subsystem Performance Characteristics Table

4.1.6 Verification Plan

The verification plan will have multiple tasks working in the stages listed below.

Stage 1: Simulation

The first stage will be used to verify that the chosen systems will be viable options for use on Mars. A computer simulation should be used to assemble a nonphysical system implementing the observed data from previous Mars missions as well as the base specifications of our chosen systems to confirm their operability for this mission.Using computer simulations we will be able to generate models to predict how well the payload will work as well as determine the overall limits that may be encountered. This stage will also let our team know if any system changes will be needed before procuring the physical equipment and if any changes to the design are necessary.

Stage 2: Performance

All the subsystems will have the attributes that are essential to the mission tested in individual ways to insure that the validation metrics listed in the table in 4.1.5 are met. Each attribute may be testing to failure, testing for function, or testing efficiencies of function depending on the listed attribute.The subsystems will be tested separately and before any assembly of the rover begins as an action of due diligence

Stage 3: Prototype

The final stage will be used to verify that the entire payload systems interface with each other and work as expected. The prototype will include all of the subsystems. The first step will be systems testing to ensuring that the systems put together can satisfy the requirements of the mission. This testing will involve onsite testing on Mars terrain detailed later in this PDR.
The current status of this verification plan is in the pre-simulation stage.

4.1.7 Risk Analysis and Minimization

The risks associated with the mission are identified and characterized into two main categories: Physical and Mission Related Risks, and Developmental Risks. The Physical and Mission Related Risks consist of post-launch operational issues which could jeopardize the mission through equipment/hardware failure, environmental hazards, or inability to complete mission objectives due to physical limitations. Developmental Risks pose a threat to the maturation of the project, and can include logistics, finance issues, and design complications. Both types of risks have been separately addressed and labeled below to clearly demonstrate thorough analysis.

Physical and Mission Related Risk Factors

Legend (Scale: 1-3)

Impact	*3 High*	*Mission success unlikely or unachievable. Workarounds unlikely or do not exist.*
	2 Medium	*Moderate impact to mission success criteria. Minimum mission success may still be met, and workarounds may exist to recover mission.*
	1 Low	*Minor impact to mission success criteria. Workaround may or may not be needed, and minimum mission success can still be met.*
Probability	*3 High*	*Event is very likely to occur. (Greater than 50%)*
	2 Medium	*Event is moderately likely to occur. (10-50%)*
	1 Low	*Event is unlikely to occur. (0-10%)*

Figure 23. Physical and Mission Related Risk Factors Legend

Unadjusted Risk(left), Adjustment (below), and Adjusted Risk (right)

#	Mission Risk Factors	Unadjusted		Adjusted	
		Impact	Probability	Impact	Probability
1	Separation / Deorbit Failure	3	1	2.5	0.5
2	EDL Malfunction/ Overspeed Impact	3	2	2.5	1
3	Deployment / Mechanical Failure	2.5	2	2	1
4	Power Systems Loss	2	1	1	1
5	CPU Corruption/SEU	2	2	1.5	1
6	Data Retrieval Errors	1	2	0.5	2
7	Calculation/Design Error	1.5	1.5	1.5	1
8	Software Error	0.5	0.5	0.5	0.5
9	Misdiagnosis of Data	1.5	2.5	1.5	1
10	Martian Weather	1.5	3	0.5	0.75
11	Extreme Terrain on Landing Site	2	2.5	1	2
12	Temperature Extremes	2	3	1	2

Figure 24. Physical Risk Adjustment Table

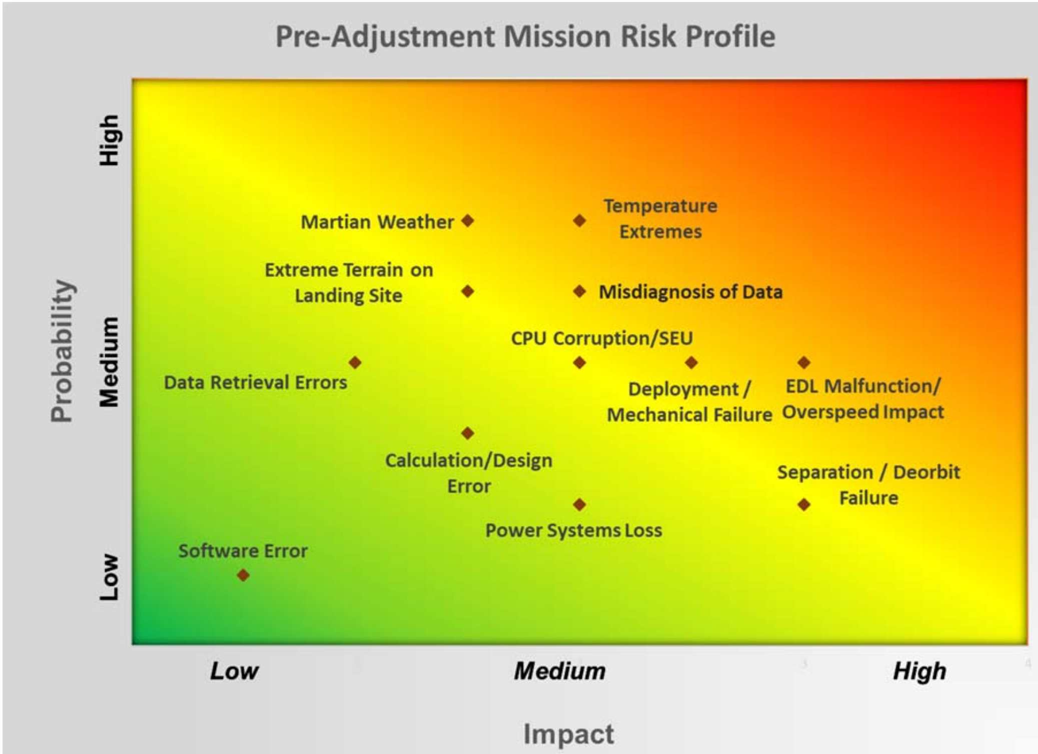


Figure 25. Pre-Adjustment Mission Risk Graphical Analysis

# Physical Risk	*Adjustment*
1) Separation / Deorbit Failure	Ensure compatibility of separation and ballistic trajectory design with transfer shuttle design engineers.
2) Entry, Descent, & Landing Malfunction / Overspeed Impact	Run multiple computerized simulations of Entry and Descent to ensure descent capabilities. Earth Test all EDL hardware for material tolerances.
3) Deployment / Mechanical Failure	Perform rigorous Earth-based testing of all mechanical systems in Mars-analog situations.
4) Power Systems Loss	Design to include redundant/split battery bank in order to ensure sufficient supply
5) CPU or Memory Corruption / Single Event Upset	Crosscheck separately partitioned process and memory banks on Radiation hardened processor for radiation-caused data storage faults
6) Data Retrieval Errors	All instruments will be calibrated on landing, and all retrieved data will be compared against analogous earth test data to ensure accuracy
7) Calculation / Design Error	All possible systems will be extensively tested first in simulation,

	and then in Earth-based systems which as much as possible replicate Martian systems
8) Software Error	Use rewritable EEPROM for in-situ software changes Test all software integration before flight.
9) Misdiagnosis of Data	Establish pre-defined data requirements for readouts
10) Martian Weather	Include dust protection mechanism for vital instrumentation.
11) Uneven landing site terrain	Include self-righting mechanism for lander with 6DoF Orientation Sensor.
12) Temperature extremes	Adjustment: Extensively test all hardware for functionality within 10% variance of expected mission temperatures

Figure 26. Mission Risk Mitigation Adjustments Table

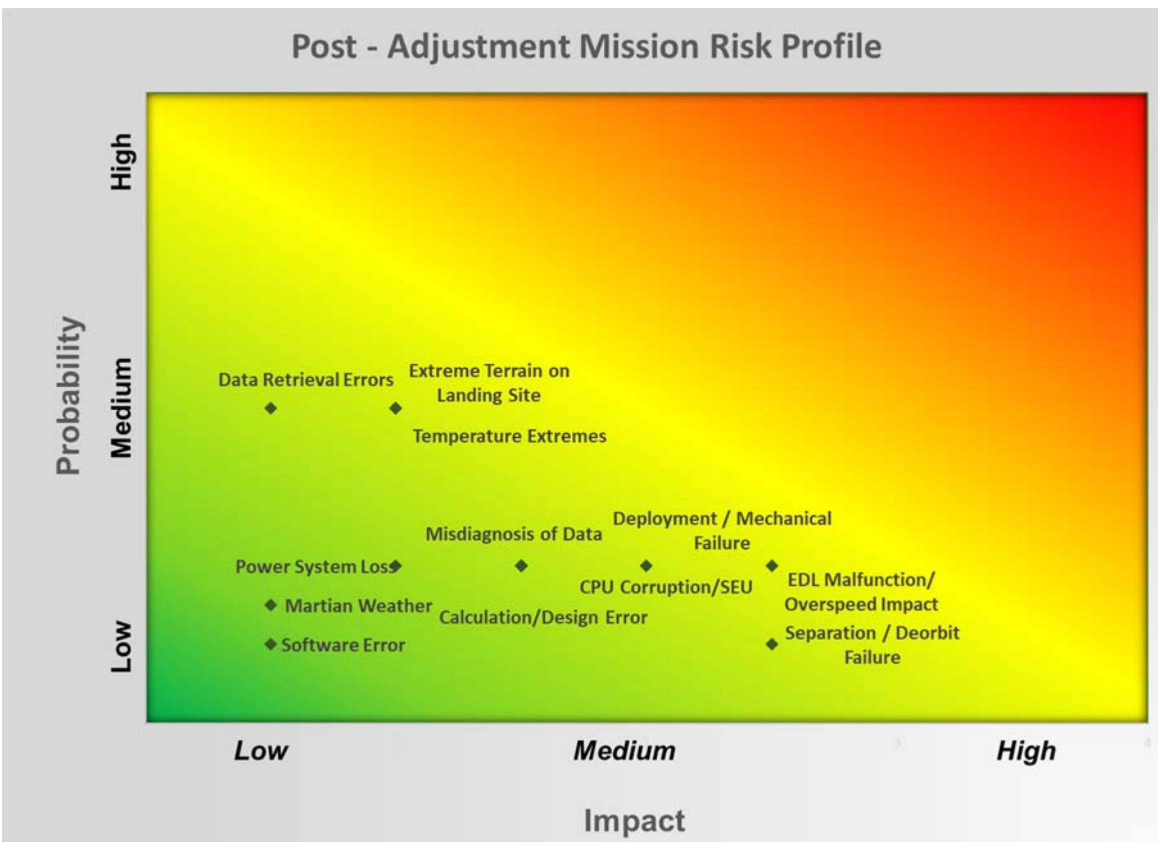


Figure 27. Post-Adjustment Mission Risk Graphical Analysis

Project Development Related Risk Factors

Unadjusted Risk(left), Adjustment (below), and Adjusted Risk (right)

#	Project Development Risk Factors	Unadjusted		Adjusted	
		Impact	Probability	Impact	Probability
1	Instrumentation Data Functionality	3	3	2	0.5
2	Parts & Material Acquisition Holdups	2	3	1	2
3	Availability of Trained Personell	1	2	1	1
4	Access to Development Labs	1	1	0.5	0.5
5	Biological Contaminant Risk	2.5	1.5	1.5	1

Figure 28. Physical Risk Adjustment Table

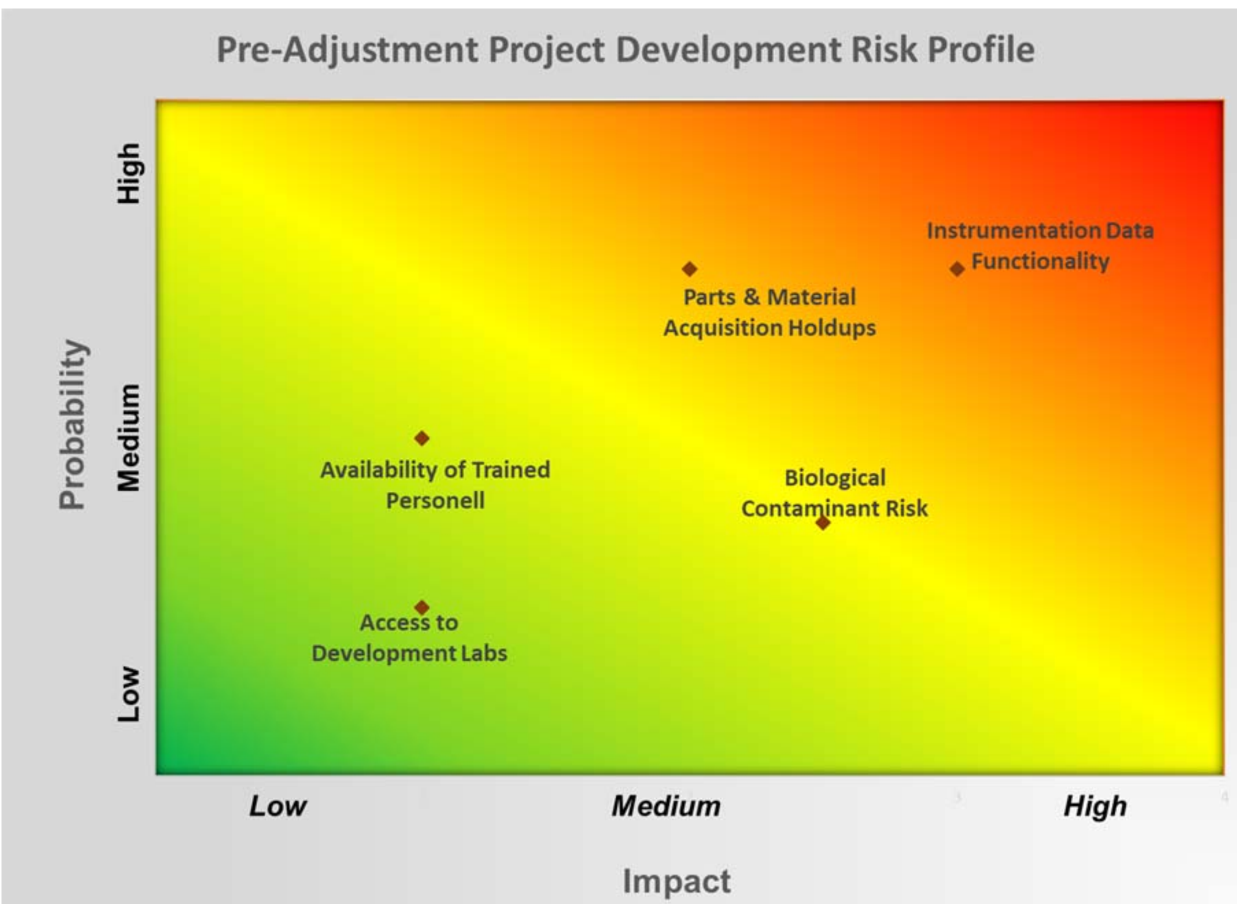


Figure 29. Pre-Adjustment Development Process Risk Graphical Analysis

# *Developmental Risk*	*Adjustment*
1) Instrumentation/Data Functionality	Budget time and money for extensive testing, first in a laboratory environment, then in sufficiently similar Mars analog environment
2) Parts & Material Acquisition Holdups	Research secondary vendors and order high lead time materials with marginal allowances
3) Availability of Trained Personnel	Set up recruiting campaign among network of engineers.
4) Access to Development Laboratories	Before start of assembly, research availability of development locations
5) Biological Contaminant Risk	Develop Sanitation / Bio-Removal Strategy for prevention of biological contaminants on Mars

Figure 30. Developmental Risk Mitigation Adjustments Table

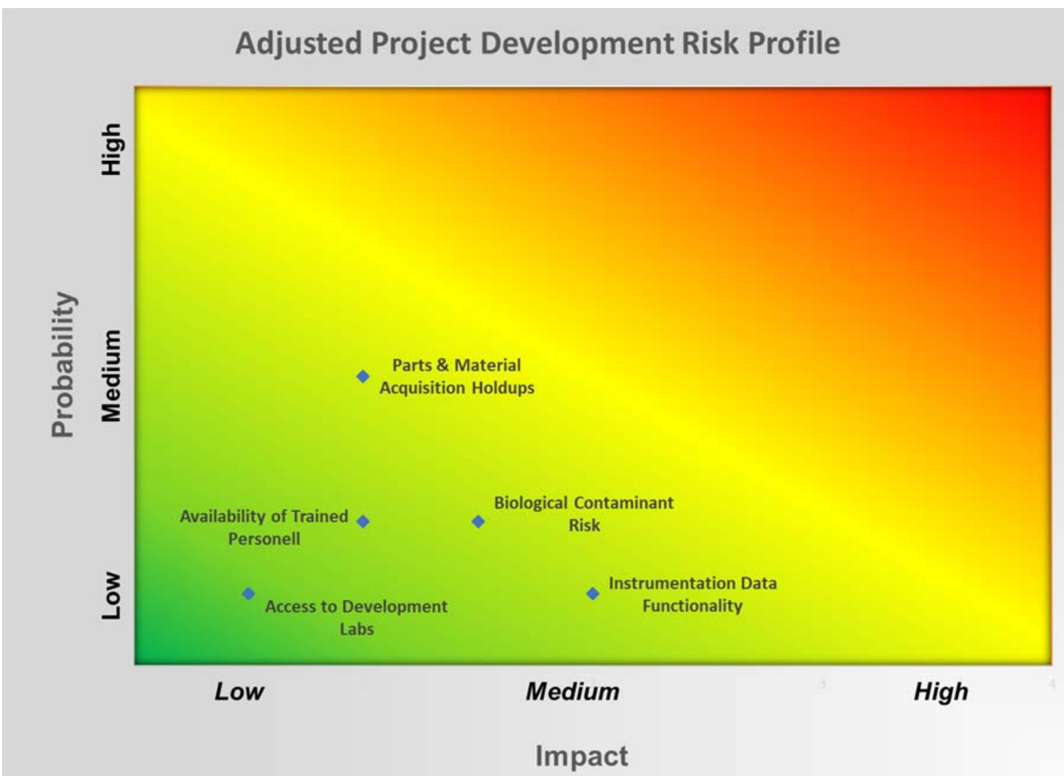


Figure 31. Post-Adjustment Development Process Risk Graphical Analysis

Risk Conclusion

The primary unadjusted risks associated with deployment of the lander were the reliability of data sent from the GPR as well as the physical challenges of environmental extremes. Through proper planning, including budgeting for development and extensive testing (both simulation and real world) of instrumentation, operations and materials, we believe the inherent risks associated with the mission and the design can be sufficiently mitigated through the measures listed above to maximize likelihood of achieving mission success criteria.

4.1.8 Demonstration of Components and How Risks/Delays Impact the Project

All of the components are necessary to complete the project. The heat shield is vital in repelling excess heat and slowing down the lander. The parachute is necessary to further slow down the lander. The thrusters are there to ensure a soft landing. The wheels are needed to take the impact. The antenna is needed to relay information back and forth between the communication package and the system. If one of the components malfunctions it affects the system.

The risks of a component malfunction, due to external influences on the system, are apparent. To prevent these issues, proper testing is required prior to launch. Delays in the project are unavoidable if components are malfunctioning during these tests. All components must be tested thoroughly in advance. If external factors increase their likelihood to negatively impact the system, then the project must be delayed in order to wait for ideal to adequate conditions. Vigilance and caution are required to, reliably, reduce the risks.

4.1.9 Demonstration of the Planning of Manufacturing, Verification, Integration

To manufacture the essential components for the project, access to a 3d printer, mill, CNC machine, and lathe should suffice to make the parts. Many of these parts will be manufactured by other companies or purchased separately. To verify that the products are up to professional standards, GD&T may be used for all manufactured parts. Integration is important in checking to see that each component operates in unison in the system. Component testing is the first step to check this. Going through to inspect the thrusters' functionality is noteworthy. Testing for overheating by monitoring temperatures over time of use will shed light on the viability of the component. Static testing is an additional step to ensure the software is running properly. Software engineers and computer scientists will search for mistakes in the launch code to ensure it is written properly.

4.1.10 Confidence and Maturity of Design

The uniqueness of this mission proposal includes a great deal of newer technology that have yet to be used in space (such as the puck-like design of the rover body, the inflatable heat shield, and the ground penetrating radar) but all aspects of this mission have been developed in collaboration with information gathered from past missions. These known factors were utilized in making designs that will be computer simulated for confidence.

All technologies exhibited in this report have been developed and tested by NASA or partners of NASA; establishing a consistent consideration for quality assurance.

4.1.11 Dimensional CAD Drawing of Entire Assembly

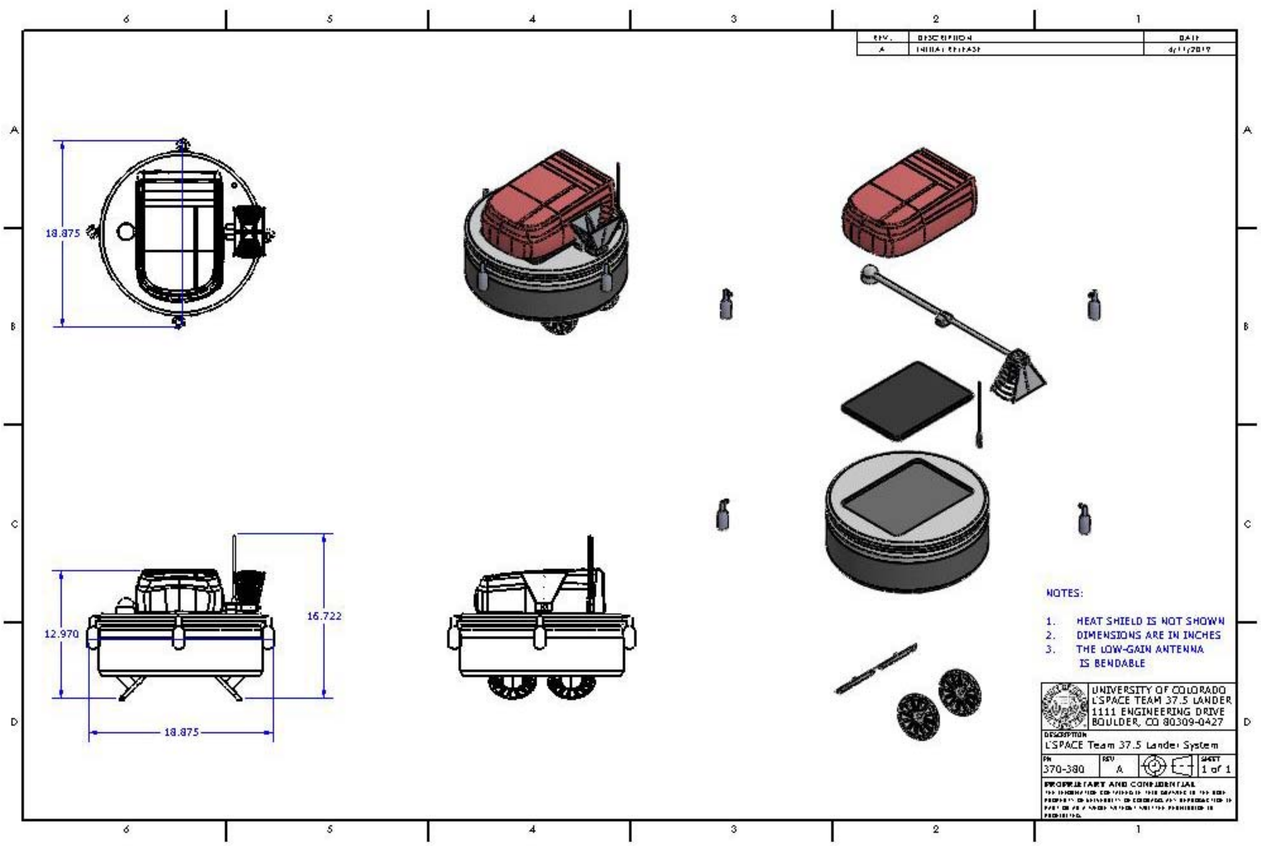


Figure 31. Dimensional CAD Drawing (in inches) of Entire Assembly

Note: Red "backpack" represents undeployed flight parachute attached. Parachute will detach before touchdown.

4.2 Mission Performance Predictions

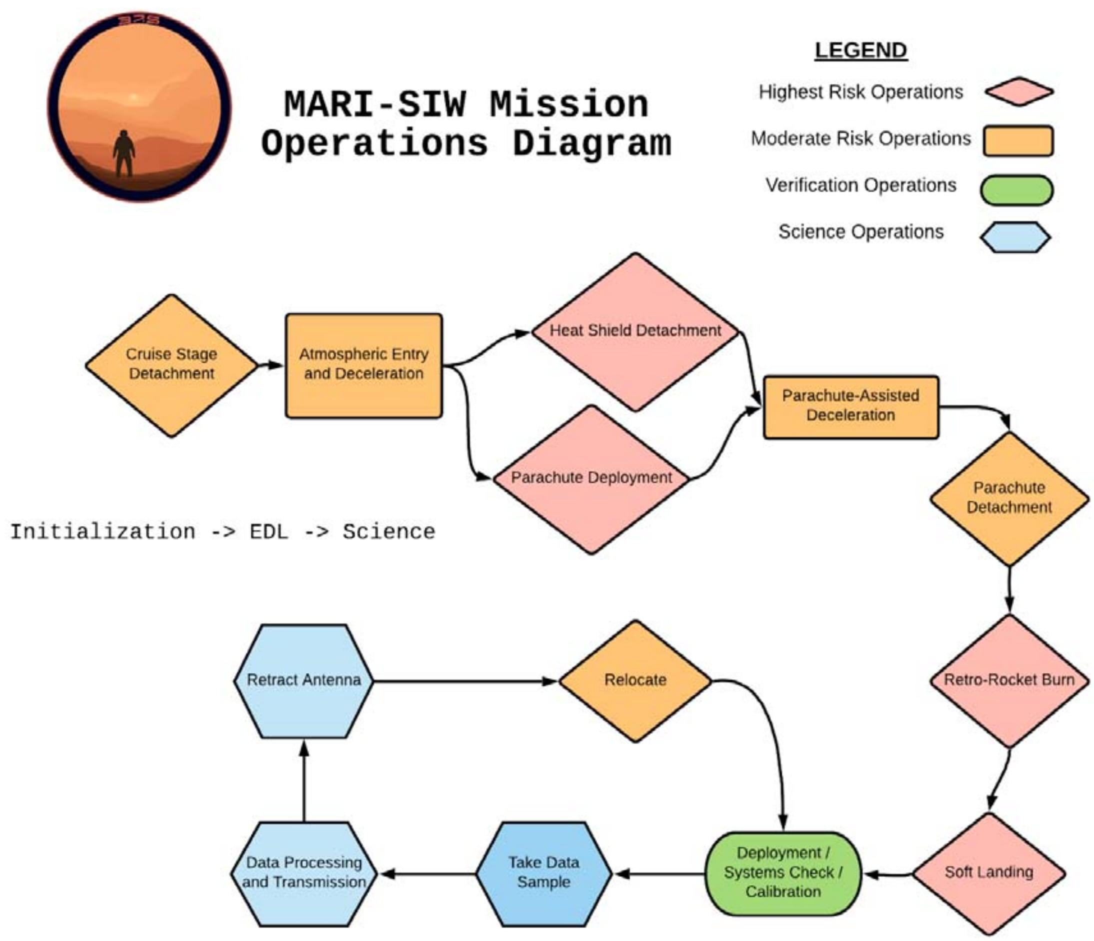


Figure 33. MARI-SIW End to End Mission Operations Diagram

4.2.1 Mission Performance Criteria

The probe must deorbit into the precise trajectory calculated to ensure landing success as the payload is only equipped with a basic landing system to accomplish an unguided, ballistic landing. Once this takes place, the heat shield will deploy during descent to protect the lander from burning up in the Martian atmosphere. Once the proper velocity is obtained, the parachute must deploy no less than 8.6 km, after which the heat shield must deploy from the lander. A parachute alone will not be enough to slow the velocity sufficiently, so there will be a small rocket system on board to slow the vehicle down to about 2 m/s prior to touchdown which should be sufficient to safely touchdown. (As discussed in section 1.2.1) Testing will be accomplished between project approval and launch to determine whether our probe will need a self-righting mechanism or if the position of the rockets used for landing will be enough to land upright and level but either way--it is imperative so that the GPR to works properly. Once landed, the probe will deploy its horn antenna to perform its first read and test to make sure the GPR system arrived safely. Once the data is transmitted back, the payload will either remain there to charge if needed or if it's day time, as the GPR does not need sunlight to work properly. After this period, the probe will test its mobility features to make sure they're intact by taking its first drive. First, the horn antenna must retract back into its initial position. Then the probe will go forward a set amount of rotations to be determined in future testing but only equaling a couple meters or so. The probe will then perform a new test of that area by using its GPR system. This pattern will continue as long as the probe continues to function properly.

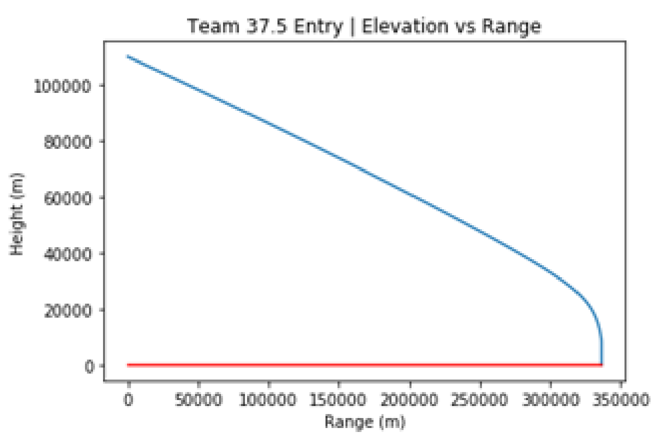
4.2.3 Flight Profile Simulations, Altitude Predictions, Component Weights, and Descent Profiles

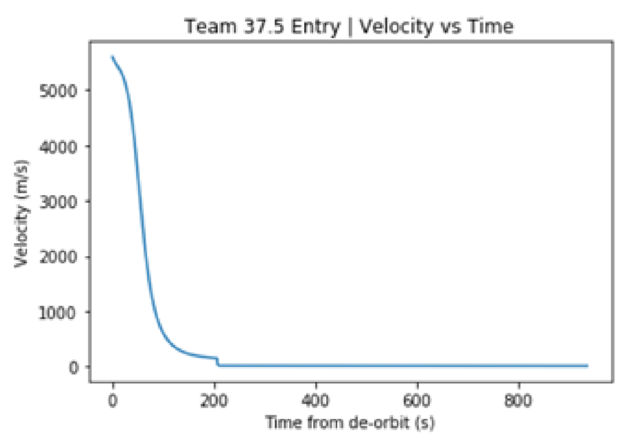
The model to determine landing trajectories was calculated using a Python program written by Lowell Hanson which utilized the Euler-Cromer Midpoint method to model projectile. This program was modeled from a program for ballistic motion found in Alejandro Garcia's "Numerical Methods for Physics".

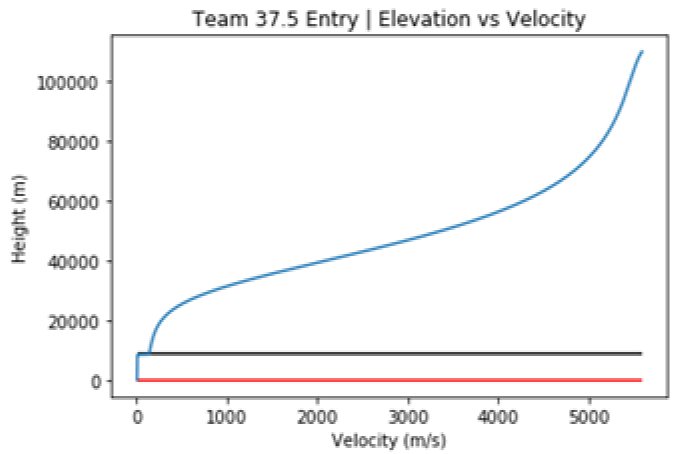
Using NASA resources, we determined atmospheric pressure as a function of elevation and used this information to model drag throughout the descent, both prior to and after parachute deployment. We modeled gravity as a constant 3.7 m/s^2, but this model could be further improved by also calculating gravity as a function of altitude.

The Python script can be found in the Team 37.5 Google Drive as "Team_37_5_Mars Descent.py" we made our calculations using a time step of 0.1 seconds per step.

The descent profile is described in section 1.2.1.







Figures 34, 35, 36. (Clockwise from Top Left): Descent Profile Elevation vs Range, Velocity vs Time, and Elevation vs Velocity.

Variations in weather, primarily wind, were modeled using a Python program developed by Lowell Hanson. This modeled the force of the wind on the probe and the resulting acceleration and change in trajectory from that wind. This model used a Monte-Carlo simulation to predict displacement caused by wind.
Average wind speed for Noctis Landing was found using JMARS and the standard deviation was calculated with a simple estimate using annual average, minimum, and maximum. This distribution could be further improved with more accurate wind data for this region of Mars as well as accounting for the seasonal changes in wind for our desired landing window. The wind for each run of the simulation was determined using a normalized random distribution for wind magnitude and direction. Current resources show that wind at Noctis Landing, which lies in the southern hemisphere, is predominantly from the southeast, but more accurate details of this distribution would improve the model.

Our model shows an average displacement from wind to be about 4.8 km to the northwest, but with a median value well below that. Further modeling is needed to find the optimal offset in or initial trajectory to account for displacement, as well as to determine our margin of uncertainty in landing location. It should be noted that the current values for cross-sectional area and coefficient of drag are very rough estimates and should be improved as more detailed designs become available.

We further modified the code to find the uncertainty in landing location. Using the median displacement as the accepted landing location we computed the percentage of simulations that fell within an acceptable error tolerance.

The following was calculated using 0.1 seconds per step and 10,000 iterations and an acceptable tolerance of 1 km:

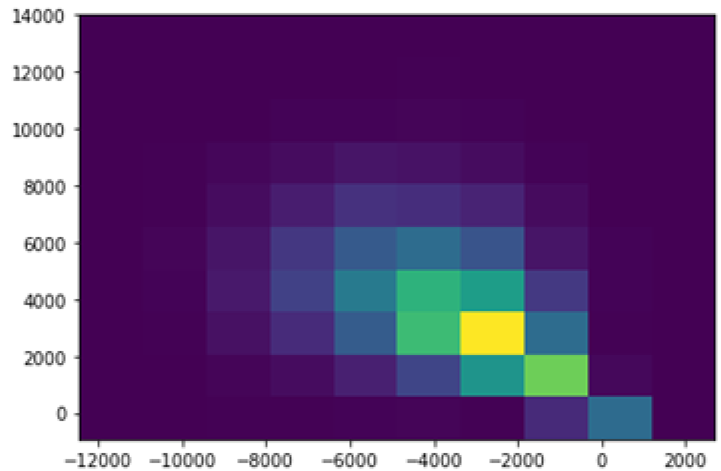


Figure 37. Landing Site Accuracy Distribution Map

- The average displacement in x due to wind is -3576.08093532 meters
- The average displacement in y due to wind is 3578.30474904 meters
- The percentage of simulations falling within 1000 meters of median landing location is 13.73 %
- This shows what is likely an unacceptable level of uncertainty in the landing location, especially given the terrain complications at Noctis Landing. It is likely that we would need to modify the flight plan to spend less time descending with a deployed parachute, or include some kind of guidance system for the probe. This model could be improved by accounting for seasonal wind speeds at the landing location rather than using annual averages.

The following was calculated using 0.1 seconds per step and 100,000 iterations:

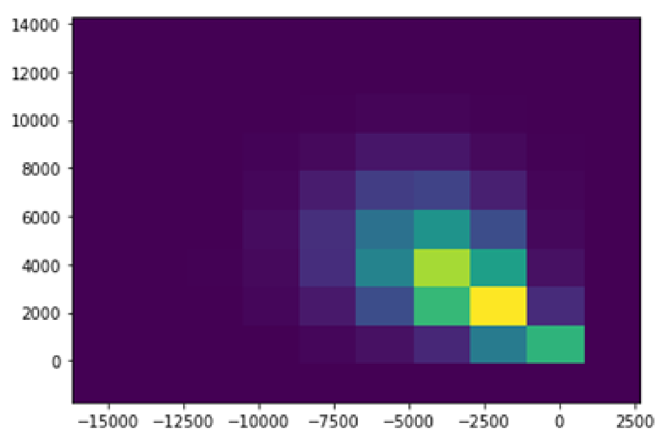


Figure 38. Wind-Inclusive Landing Site Accuracy Distribution Map

The Python script can be found in the Team 37 Google Drive as "Team_37_5_Wind_Monte_Carlo"

4.2.4 Stability Margin, Simulated CP: Center of Pressure/ CG: Center of Gravity Relationship and Locations.

This level of calculation is beyond the scope of our PDR, and moving forward, we will devote additional staff and engineering hours to developing accurate assessment of correct CP and CG calculations in order to ensure stability of the MARI-SIW payload during EDL.

Figure 39. Center of Mass Graphical Representation of Deployed System

4.3 Payload Integration

4.3.1 1 Integration Plan with Mars Orbiter and Probe Payload.

Our payload will be attached to the orbiter via hydraulic clamps, covered by the heat shield to protect the instruments during the flight. By taking this approach, on top of protecting our lander, it also leaves the lander ready for descent without any additional complications. In addition to the hydraulic clamps, the payload will also be connected to the orbiter with an umbilical connection to provide power and data to the lander, similar to the Viking mission. This gives a secondary tether to the orbiter as well as enables the ability to perform routine system checks during the cruising stage to monitor the payload systems prior to the final systems test after landing. Once the orbiter is at the designated height above our landing zone, the hydraulic clamps and umbilical connection will release, initiating the ballistic drop of the lander.

4.4 Earth Testing Operation Procedures

Testing Procedures: All testing will be conducted per NASA's defined testing standards and as outlined by the team safety officer (per section 4.5.1) to ensure environmental and human safety expectations are met.

4.4.1 Testing System Used for the Earth Analog Testing of a Component of the Science Experiment

Instrumentation/GPR Testing:

The first and most critical system to begin testing upon project acceptance will be the Ground Penetrating Radar array. As discussed in section 3, the data return from the GPR is highly dependent upon the pressure, temperature, and dielectric constant of the material it is scanning. The retrieval of reliable, consistent data is, in turn, highly dependent upon an exact standard of expected readouts for different materials under different conditions. In order to ensure the most accurate system of determining subsurface material composition, we will not only calculate the theoretical data return of the GPR, but also test our exact setup under a variety of Mars-analog conditions, including laboratory simulations of regolith-ice mixtures at various temperatures and pressures at varying distances from the instrument. These laboratory experiments will also be confirmed using field simulations of subsurface ice layer detection in Antarctica. These simulations will be used to create a comprehensive data sheet of expected GPR readings for both varying mixtures of CO2 and H2O ice at a range of temperatures and depths which can then be used to accurately characterize the instrument's Martian data return into information about the subsurface material composition of Mars.

Mobility / Drive System Testing:

The drive system will be tested using the Mars analog testing environment. With proper permissions this will likely involve either time at NASA's Mars Yard at JPL, or at Colorado's Great Sand dunes, which have also been previously used for rover testing. Because of the difference in gravitational constant between Earth and Mars, the rover will be tested as a partial assembly, adjusting the mass by removing non-drive-related systems to simulate Martian gravity and the traction the rover will experience upon deployment. Thorough testing of the drive system will ensure the following criteria: a) the ability to safely move across terrain without getting stuck or damaged b) the ability to circumvent obstacles c) the ability to correctly determine relative location and displacement based on triangulation with the established communication relay.

Fig 40. Mars Yard II, NASA JPL, Retrieved from https://www-robotics.jpl.nasa.gov

Computer System / Electronics Testing:

The computer and electronics system will undergo a series of laboratory tests to ensure proper assembly and execution of all functions, as well as simulation of the demanding environments of deep space, atmospheric entry, and the Martian surface. These testing requirements will include radiation-testing, vacuum testing, temperature extreme testing, and material and connection stress testing involving drop tests, vibration tests, and acceleration tests. Each test will run a full iteration of the probe's function, including execution of movement functions, instrument scanning functions, and data transmission functions.

Entry / Descent / Landing Hardware Testing:

Although it is currently impossible to perform a physical end to end EDL test in the Martian atmosphere, we plan to use NASA's advanced simulation software to extensively test a the physics of our system on its EDL profile. According to David W Way of Langley Research center, "the Program to Optimize Simulated Trajectories II (POST2) is a general Six Degree-of-Freedom (6-DoF) trajectory simulation tool, that solves both the translational and rotational equations of motions for up to 20 independent rigid bodies." (Way, 2013 , para. 1) By running a variety of simulations within the POST2 program, accounting for varying environmental conditions and entry angles, we will be able ensure that our descent system will function the same in flight as it does on paper.

In addition to the simulated entry profiles, we will perform separate physical tests of solid-fuel retropulsion system to ensure successful scaling of booster nozzle and thrust vectoring. These tests will occur Glenn Research Center's Space Power Facility, where we will use the Space Simulation Vacuum Chamber to simulate the Mars atmosphere and test the solid fuel boosters, tracking physical output and combined utility of the four-booster system to ensure that actual specific impulse of the fuel inside our scaled-down boosters (including nozzle functionality) accurately reflects theoretical specific impulse of the system in previously-used full-scale designs. This will guarantee that our design is capable of providing enough ΔV to decelerate the lander to less than 2m/s before impact.

4.4.2 Outline of Final Assembly for Earth Testing

While the majority of MARI-SIW's subsystems will require independent testing to ensure suitability for the mission, there are several key tests that will include the final assembly, including a test for structural integrity and verification of aerodynamic profile. These two final tests will be conducted with a functional replica of the finished product.

As the payload is deorbited from space, the materials and construction mechanisms will be exposed to extreme physical demands, and we would like to ensure that all that the design is sturdy, all connections are secure, and all materials are suitably strong for the physical extremes they will endure. For this reason, we will perform a structural integrity test in the Mechanical Vibration Facility at Glenn Research Center, which can apply a variety of vibrational forces to the payload, meant to replicate the extreme conditions of atmospheric entry. Prior to testing, we will develop a series of minimum physical requirements and allowable tolerances for the payload to pass the test, and compare our pre-test requirements against the post-test analytical observations from the trial. All systems must maintain complete integrity before successfully passing this test.

The second EDL system to be tested will be the aerodynamic profile of the craft during entry. Since we will be using the non-heritage system of a deployable foldable heat shield, and are not including an aerodynamic cover for retrograde turbulence mitigation, computerized simulations of entry will be less reliable of a method for ensuring stability during entry. For this reason, we will use the Arcjet Testing Facility at Johnson Space Center (which offers Martian atmosphere entry simulation) to verify stability of the craft while experiencing the extreme aerodynamic effects of entry. If the payload does not maintain stability to within our required tolerances, we must be willing to modify the design to increase stability.

The final and most important test will be a full deployment test of the assembled payload. This test will involve dropping the payload in a freefall which is meant to mimic in real physics the final stage of landing as well as successful deployment of all systems and subsystems. The payload will be released from altitude with a parachute designed to reduce the Earth-atmospheric descent velocity to 9.7m/s (Calculations TBD). The rockets will fire at the appropriate altitude upon release from the parachute in order to reduce the impact velocity to within tolerance limits. The rover will then take a GPR data sample and navigate to at least two other scanning locations, demonstrating its functional mobility and instrumentation systems. The last requirement of this test is that the rover successfully transfer the data it collects from the test to a nearby mockup of the provided communications array.

The following table is an outline of the test procedures in chronological order of development, referencing the tests mentioned in 4.4.1 and 4.4.2:

Testing Procedures

System	Integration	Min Requirements
Instrumentation / GPR	Separate	Accurately and repeatedly identify characteristics of regolith analog at varying depth and pressure
Mobility / Drive System	Separate / Weight Adjusted	Demonstrate capability to successfully navigate Martian terrain without becoming stuck or damaged.
Computer System / Electronics	First Separate, then as assembled payload	Successfully run all systems/subsystems including exposure to extreme temperature /pressure situations
EDL Profile Simulation	Separate	Mathematically demonstrate ability to safely and accurately deorbit via ballistic trajectory under a variety of atmospheric (wind) conditions.
Rocket Testing	First Separate, then as Assembled Payload	Verify theoretical thrust output and vectoring of miniaturized booster design and nozzle configuration
EDL Aero Testing	Fully Assembled	Demonstrate stability and even distribution of aero forces on the landing stage of the rover.
Drop Test / Deployment	Fully Assembled	Ensure that the lander can withstand impact of up to 200% of the intended landing velocity, and successfully deploy, scan, and return data.

Fig 41. Testing Procedures Table with Integration Stage and Minimum Success Requirements

4.5 Safety and Environment for Protocols for Earth-based Testing of a Component from Science Experiment

4.5.1 Safety Officer

The authority in the hazards and safety of Earth testing will be the Safety Officer. This authority will be present in-person for all Earth testing. The chosen Safety Officer for this project is Jynette Tigner.

4.5.2 Failure Mode Effect Analysis (FMEA)

Note: This FMEA is a living document and is not meant to be a comprehensive or final determination of all mission risks. During the continued development of the project it will be mandatory that all considered risk factors be added to this FMEA and addressed to the satisfaction of lead scientists, engineers and managers.

Process Step / Input	Potential Failure Mode	Potential Failure Effects	SEV	Potential Causes	OCC	Current Controls	DET	RPN	Actions Recommended	Resp.
Parachute Deployment	Does not Deploy	Terminal- Velocity Impact/ Destruction of Craft	10	Electronics Malfunction	2	CPU initialized deployment Mechanism	2	40	Redundant Deployment Mechanism	Design Engineer
	Parachute lands on top of craft	Entanglement of Rover	8	Insufficient horizontal movement	6	None	3	144	Design Parachute Removal Mechanism	Design Engineer
	Deploys at Incorrect Altitude	Overspeed Impact / Decreased landing site accuracy	3	Incorrect Timing Functionality	6	Timer-based Deployment	3	54	Include Altitude Sensor	Design Engineer
Solid Rocket Deceleration	Failure to Fire	Overspeed Impact	8	Electronics Malfunction	3	CPU-controlled Electrical Ignition	4	96	Extensive Testing	Test Engineers
	Fires at Incorrect altitude	Overspeed Impact	8	Incorrect Timing	9	Timer-based Deployment	5	360	Include Altitude Sensor	Design Engineer/ Electronics Engineer
Payload Integration	Decouple from Cruise Stage	Failure to Deorbit / Inaccuracy in landing	7	Cruise Stage Attachment Failure	5	Extensive Simulated Software	3	105	Collaborate with Cruise Stage Engineers	Project Leads / Design Engineer

	Absorb Landing Impact	Damage to Drive System	6	Insufficient Wheel attachment Strength	4	Soft landing from Deceleration hardware	5	120	Implement shock absorption mechanism in diagonal wheels	Design Engineer
Drive System	Engage Drive System	Inability to move	9	Traction Issues /	5	Diagonal Wheel design	4	180	Experiment with wheel traction design for Martian Surface Application	Design Engineer
	Navigate between Testing Sites	Rover gets stuck on obstacle	4	Insufficient spatial awareness	5	IR Sensor / Communica- tions Beacon Triangulation	6	120	Include additional Spatial Awareness Tech	Design Engineer / Electronics Engineer
Instrumentation	Horn Antenna Fails to Deploy	Inability to do Science!	10	Mechanical Malfunction	5	Motorized Deployment Mechanism	2	100	Verify integrity of deployment mechanism	Test Engineers
	Inconclusive Data Return	Inability to interpret Science	2	Incorrect Reading Angle	7	6DOF Orientation Comparison	1	14	Develop Methodologies for predicting and interpreting data from non-perfect reading angles	Instrumentation Scientists

Fig 42. FMEA for major systems and subsystems

4.5.3 Safety and Hazards

Each Earth testing and manufacturing method comes with its own hazards and safety concerns but by recognizing these hazards and mitigating them this mission strives to be as safe as reasonably practicable. All tests should be done with the least risk to personnel and bystander as possible by adhering to safety parameters specific for the test.

Special attention will be paid to the solid fuel in the landing thrusters which are made up of Polyurethane-bound aluminium-APCP which contain ammonium perchlorate and are explosive. The Material Data Sheet listed below lists the safety parameter and handling in detail.

Material Safety Data Sheet May be used to comply with OSHA's Hazard Communication Standard, 29 CFR 1910 1200. Standard must be consulted for specific requirements.	**U.S. Department of Labor** Occupational Safety and Health Administration (Non-Mandatory Form) Form Approved OMB No. 1218-0072
IDENTITY: Polyurethane-bound Aluminum-APCP Solid Fuel	*Note: Blank spaces are not permitted. If any item is not applicable or no information is available, the space must be marked to indicate that.*

Section I Chose Manufacturer's Information Goes here

Section II—Hazardous Ingredients/Identity Information

Hazardous Components (Specific Chemical Identity, Common Name(s))	CAS Number
Ammonium Perchlorate	7790-98-9
Aluminum Powder	7429-90-5
Polyurethane	9009-54-5

Note: Ammonium Perchlorate is the active substance and what contributes to the hazard level of the lab.

Section III—Physical/Chemical Characteristics of Ammonium Perchlorate

Boiling Point	NA	Specific Gravity (H_20 = 1)	1.95
Vapor Pressure (mm Hg)	NA	Melting Point	NA
Vapor Density (AIR = 1)	NA	Evaporation Rate (Butyl Acetate = 1)	NA

Solubility in Water Soluble in Water

Appearance and Odor NA

Section IV—Fire and Explosion Hazard Data

Extinguishing Media

Suitable extinguishing media
Use water spray, alcohol-resistant foam, dry chemical or carbon dioxide.

Special Fire Fighting Procedures:

Wear self-contained breathing apparatus for firefighting if necessary

Unusual Fire and Explosion Hazards

Explosive; mass explosion hazard
Nitrogen oxides (NOx), Hydrogen chloride gas

Section V—Reactivity Data

NFPA Reactivity Rating 4

Section VI—Health Hazard Data

Health Hazards *(Acute and Chronic)*

May cause damage to organs through prolonged or repeated exposure
Causes serious eye irritation

Signs and Symptoms of Exposure

Irritating to skin, eyes, and respiratory system.

Medical Conditions
Generally Aggravated by Exposure

Emergency and First Aid Procedures

General advice
Consult a physician. Show this safety data sheet to the doctor in attendance.
If inhaled
If breathed in, move person into fresh air. If not breathing, give artificial respiration. Consult a physician.
In case of skin contact
Wash off with soap and plenty of water. Consult a physician.
In case of eye contact
Rinse thoroughly with plenty of water for at least 15 minutes and consult a physician.
If swallowed
Never give anything by mouth to an unconscious person. Rinse mouth with water. Consult a physician.

Section VII—Precautions for Safe Handling and Use

Steps to Be Taken in Case Material Is Released or Spilled

As an immediate precautionary measure, isolate spill or leak area in all directions for at least 50 meters (150 feet) for liquids and at least 25 meters (75 feet) for solids.

SPILL: Increase, in the downwind direction, as necessary, the isolation distance shown above.

Section VII—Control Measures

Personal Protection Equipment

Wear positive pressure self-contained breathing apparatus (SCBA). Wear chemical protective clothing that is specifically recommended by the manufacturer. It may provide little or no thermal protection. Structural firefighters' protective clothing provides limited protection in fire situations ONLY; it is not effective in spill situations where direct contact with the substance is possible

Fig 43. Ammonium Perchlorate Material Safety Data Sheet. Retrieved from pubchem.com (2019)

4.5.4 Environmental Concerns – Optimal Testing Environment Conditions

The majority of our testing schedule for mobility and EDL systems involves laboratory tests (as discussed in section 4.4.2). These tests, conducted both at our development facility and at various space centers, will depend on controlled environments meant to test specific systems under specific conditions in order to ensure that the most extreme of environments that MARI-SIW will encounter have been tested for. These laboratory-based control tests will test for tolerances in deep-space cosmic radiation, solar radiation, temperature extremes, vibration, acceleration, and impact.

The final test will be the drop and deployment test (also described in 4.4.2) of the test version of the probe, which is meant to test end-to-end performance of the final stage of landing as well as successful deployment of all systems and subsystems. Optimal environmental conditions for this test involve minimal wind, a clear day for visibility, observation, and recording of all testing information. Passing mission criteria for this test will include complete, autonomous functional operation of the rover from deployment of (physically analogous) parachute to completion of a set of scans and a battery recharge cycle. The surface onto which we will perform the drop test will be as similar as possible to Martian terrain, and include sand, uneven inclinations, and debris. Further investigation and coordination of this test will begin on project acceptance. A successful iteration of this test will ensure that all probe systems are functional, durable, and ready for launch.

V) Payload Criteria

5.1 Selection, Design, and Verification of Payload Experiment

5.1.1 System Design Review

The system chosen for the MARI-SIW mission has a unique design. The system, as a whole, is small and compact. It is comprised of a circular shape to decrease the chances of getting stuck or wedged into a ditch. The circular design also adds symmetry that helps to keep the center of mass near the center. This model is based on the Tertill Weeding Rover. The Tertill rover was designed to be a self-driving system that maneuvers around an environment with obstacles to locate weeds to be shredded. The Tertill is a lightweight rover that avoids steep inclines and large rocks through nanotechnology and transverses slopes of up to 22-degrees. Using a similar design, MARI-SIW will be able to move around in its environment, avoiding pitfalls, to locate ice under the soil, using RIMFAX GPR technology. To enhance mobility, the wheels are tilted to 45-degrees and are given spokes which aid in climbing and maneuvering. The system charges itself through the solar panel which is located at the top of the system. The power generated is stored in two rechargeable lithium batteries. Every aspect of the system previously listed are utilized to improve the mobility and functionality of the rover in the Martian environment.

5.1.2 Payload Subsystems Review

The payload subsystems are designed to aid the rover in accomplishing the payload objectives. There are two main objectives. First is to locate ice deposits near the landing site that are accessible near the surface for retrieval. Consequently, the rover needs to be mobile in order to scan multiple areas to accomplish this goal of discovering ice deposits. The four wheels work in unison to transport the rover. The motors inside the rover need to provide enough torque to push the rover over inclines and small rocks. There will be tiny touch sensors located at the front of the rover to inform the rover if there is an object in the way. The computer will utilize this information to send a command to the rover to change directions at the slightest touch. The second objective is to collect data about the quantity of ice and the chemical qualities of the ice at the site. This is done through the RIFMAX GPR technology. There will be an internal mechanism near the bottom of the system that communicates with an antenna at the top of the system that focuses, transmits, and receives signals to obtain data. A separate smaller antenna will transfer the data collected to the communication package. The GPR will pull approximately 10W of power at peak performance and requires a robust electronics subsystem to power it and successfully record and transmit the numerical data it will record. A single high-efficiency solar panel mounted onto the top of the chassis will absorb solar energy, while two high cycle-life Saft 174565 Rechargeable Lithium-Ion Cell Batteries will hold the charge for the system. Hence, the subsystems all work in unison to complete the payload objectives.

The GPR mechanical arm has specialized characteristics. It is able to extend to a set length. There are two support structures. The support structure furthest from the cone is stationary and allows the arm to swivel up and down, while balancing the weight to prevent the rover from tipping. A support structure would be added to the opposite end of the system and would extend up and down, acting like a second mechanical arm. This support arm would extend the cone up to the desired height of 60cm. There is an intermediate between the first mechanical arm and the GPR cone unit. This intermediate element rotates to position the cone to face up or down, perpendicular to the arm. It also rotates in a second direction to allow the GPR sensor to point directly downward, facing the ground. In summary, these components allow the GPR cone-shaped antenna to remain retracted and compact for portability or shifted and prolonged to a set height and angle to efficiently record data.

5.1.3 Performance Characteristics for System and Subsystems & Evaluation and Verification Metrics

The performance characteristics of the Mars rover allows it to travel short distances at a time in order to to collect data to be analyzed back to Earth that maps and verifies the location of liquid ice near the landing site. The system avoids obstacles and steep slopes through simple body-integrated touch sensors and ultrasonic sensors. The rover is programmed to cover as much area per day as solar recharging allows, to gather and transmit a collection of data points. A full charge of both batteries will support a total of 2 hours of run time of all onboard systems at once, although a full cycle of data sampling and movement will use only the drive train or the instrument at once. (Sampling will always be done while stationary) The GPR and antenna gather and transmit the data, respectively. The rover uses a system of orientation sensors, triangulation between itself, the nearby communication package, and orbiting satellites such as MAVEN to determine its displacement and relative location. Therefore, the performance characteristics of the rover enable the system to gather multiple data points around the landing site to record the local topography.

The following is a list of the evaluation and verifications metrics we will use to ensure that our payload experiment will fulfill the requirements laid out in the mission concept:

- Can the payload confirm the presence of ice at the chosen landing sight?
- Can the payload differentiate between H_2O and CO_2 ice?
- Can the payload determine the depth of the ice from the surface?
- Can the payload determine the total amount of ice at the chosen landing sight, confirming that it meets the $100m^3$ requirement?

These metrics will be the baseline for evaluation throughout the verification plan as outlined in section 5.1.4.

5.1.4 Verification Plan

The current status of our verification plan is in the pre-simulation stage. The verification plan will be comprised of three stages: a simulation using the system specifications from the payload and the conditions on the Martian surface at the landing site to verify the GPR will work as expected or better, developing an engineering unit to ensure the CPU will function properly with GPR and will operate as expected, and developing a full prototype to ensure all systems work as expected. By using these verification methods measured against the metrics established in section 5.1.3, our scientific payload should be more than able to fulfill the mission concept. Each stage is explained in more detail below:

Stage 1: Mission Simulation

This first stage will be used to verify that the chosen systems will be viable options for use on Mars. To this end, we will create a computer simulation using the observed data from previous Mars missions as well as the base specifications of our chosen systems to confirm their operability for this mission. By using computer simulations initially, we will be able to generate models to predict how well the payload will work as well as determine the overall limits that may be encountered. This stage will also let our team know if any system changes will be needed before procuring the physical equipment and if any changes to the design are necessary.

Stage 2: Engineering Unit

The second stage will be used to verify that the critical systems properly interface with each other and work as expected. The engineering unit will comprise the power system (solar cell and batteries), the CPU and the GPR. The first step of the testing will be systems testing, making sure the components interface properly with each other. After ensuring those systems work correctly, the second step of the testing will be to make sure the CPU is able to process the data received from the GPR correctly.

Stage 3: Prototype Unit

The final stage will be used to verify that the entire payload systems interface with each other and work as expected. The prototype will comprise of all the systems in the engineering unit as well as the drivetrain and communications array. The first step will be systems testing again, ensuring the rest of the systems interface properly with the engineering unit. After making sure that no issues come up in the first step, the second step will be checking to make sure that mobility does not interfere with the operations of the GPR. The final step will be to determine the optimal speed that the final product will move to get the most efficiency out of the power supply as well as being both time and cost efficient for the duration of the mission.

5.1.5 Preliminary Integration Plan

The initial plan for the integration begins with stage 1 of the verification plan. The simulations will assist in how to plan for operating in the Martian environment, taking in account the temperature range, radiation to determine the operating parameters of our mission. Using the data collected from the simulated models, we will be able to prepare the engineering unit. The parameters established will guide how we integrate the components of the engineering unit. The first step in integrating the systems is to ensure the power supply and CPU interface properly before adding the GPR to the assembly and performing a systems test again. This step will provide the basis for how the final design will be developed. Then during the prototype assembly process, another systems test will be performed to ensure that the communication array, drivetrain and CPU all integrate correctly. During this step, the communications system will be tested to make sure it interfaces with the separate communications package in preparation for Mars. The final step is after the mission has landed on the surface of Mars, all the systems will be remotely tested to ensure they still operate correctly after flight and descent.

5.1.6 Precision of Instrumentation and Repeatability of Measurements

The precision of the instrument depends largely on a couple key factors, discussed more in depth in section 3.1.2. These are spatial resolution and depth resolution, as well as to a lesser extent signal-to-noise ratio and signal-to-clutter ratio. In the strictest sense, these latter two will not affect the precision of the instrument but will certain effect how useful precision will be, for if we have precise measurements that are bombarded with noise and clutter, they will not be of much use. For more information about the details concerning these four factors, see section 3.1.2.

The nature of the instrument allows it to take measurements repeatedly, within the bounds of having enough power to do so. Within the given time period to recharge, it can continue to take measurements, store them, and transmit them limited only by the redundancy of excessive measurements in one place, and in frequency by the availability of power from the power source.

5.2 Payload Concept Features and Definition

Many of the design choices for our rover are based on heritage design or currently in-development systems, as these systems are generally more reliable than untested equipment, and in an endeavor as risky and expensive as sending a rover to mars, we valued dependable hardware over more creative, yet unproven theoretical options for our design.

5.2.1 Creativity and Originality

Our payload is called Martian Radar Imager for Subsurface Ice Water, or MARI-SIW. The inspiration for our ground-penetrating radar system comes from the instrumentation currently being developed for the Mars 2020 rover: the Radar Imager for Mars' Subsurface Experiment (RIMFAX). This type of ground penetrating radar (GPR) technology used by RIMFAX is also currently used on Earth to classify subsurface materials such as ground water.

Like RIMFAX, MARI-SIW will send radio frequency electromagnetic waves several meters deep into the ground; using the reflected radio signals we will be able to capture a numerical image of the Martian subsurface which will provide further insight into material layer structures and composition of Mars. This technology can identify subsurface material composition and geological formations, allowing for the categorization of subsurface ice in both abundance and chemical state and composition. Also, similar to RIMFAX, our GPR will use a single antenna as both the radio signal transmitter and receiver to provide imaging of subsurface features up to a depth of 10 meters. An additional benefit of this system is that the transmitter is very versatile and can output a variety of wavelength signals in order to provide a greater variety of testing parameters.

Significant differences between MARI-SIW and RIMFAX will include size, weight, and maneuverability. A large portion of our design budget will be devoted to developing a light weight GPR to fit our 5kg mass requirements. A distinction between the 2020 RIMFAX is that MARI-SIW will only include its electronics box, antennae, and a communications package, and no other instrumentation, thereby isolating the GPR from any other system interference. Our single-instrument payload will be run by heritage space grade radiation-hardened single-board computer (SBC). These computer systems are specifically developed and tested to withstand the extreme radiation and temperature environments of spaceflight and the Martian Surface. The computer will additionally have several sensors (6 degree of freedom, radio transmitter, ultrasonic sensor, and temperature sensor) mounted to it to provide orientation and localization capabilities. The horn antenna (GPR transmitter/receiver) will be externally mounted at the top of the payload and will be retractable during Entry/Descent Landing and severe weather conditions. As a single system payload, MARI-SIW will be mobile and able to travel across the Martian terrain.

Ideas for the structural design of our payload were centered around maneuverability. In order for our GPR to collect data, it must be mobile to allow for multiple samples across a specific surface area. Therefore, our GPR as a stand-alone unit, must be able to maneuver itself across the Martian surface. Designing a simple mechanism for maneuverability must ultimately include wheels.

A new system currently being developed by Franklin Robotics, called Tertill, is a light weight, four-wheel-drive, solar powered robot designed to hunt and cut weeds in a garden, and has been our biggest inspiration for mobility. The Tertill's diagonal four-wheel-drive design allows for maneuverability and stability in any direction on slopes under 40 degrees and over rocks. Tertill uses basic sensor functions such as a gyroscope for guidance to detect its surroundings so that it can be completely self-reliable. Franklin Robotics is currently still working to finish the design and a portion of our budget will go to securing the correct patent requirements for MARI-SIW's design. Our design for MARI-SIW will be based off of the Tertill's diagonal-four-wheel-drive mobility. MARI-SIW will feature the same wheel functionality to allow for it to travel short distances and in any direction. Its communication package and basic sensors for orientation will allow MARI-SIW to be able to detect obstacles as well as successfully navigate to its daily destination and stay independent from the need for manual guidance.

5.2.2 Uniqueness and Significance

The significance MARI-SIW has to offer is a new approach to quantifying subsurface Martian water ice as a light-weight, fully mobile, single-unit payload, unattached to any other equipment or systems. With the data collected by MARI-SIW, we will be able to detect water ice by simply sending radio waves into the ground. Because our GPR is modeled after the RIMFAX, its variable transmission strength will allow for ultra-detailed subsurface scanning.

As discussed in section 4.4.1 (Testing) we will use our GPR not only to map subsurface structuring, but also develop detailed tables of expected readouts and definitions for a large number of possible subsurface material compositions and characteristics. Since these tables will be created from real data contrived from Earth-based Mars analog environments, they should give us greater accuracy in our ability to interpret and accurately characterize the data specific to frozen H20 and C02 stock.

Our GPR will lead to a better understanding of the Martian subsurface. The surface of Mars can only tell us limited information about Mars' past and ultimately its future. The more we understand about Mars' subsurface geology and materials, the closer we will get to human integration. MARI-SIW will be designed to travel in increments of 1 meter, take a data sample, rinse, and repeat several times per day, pausing to recharge as needed until minimum mission timeline is met. In the future, if multiple MARI-SIW GPR payloads can be sent to various locations on Mars, we can begin to piece together a clear picture of the Martian subsurface at a variety of potential future exploration sites, gathering and analyzing its geological history and subsurface materials, including water ice. Once our system detects subsurface water ice and quantifies its presence and abundance, human habitation may finally become a reality on Mars.

5.2.3 Suitable Level of Challenge

MARI-SIW will assume the difficult task of identifying and quantifying subsurface water ice on Mars. If the ice can be identified and quantified using our GPR, further exploration of this site could eventually lead to human exploration of this area. The development of this design has met several significant challenges. One of the most significant challenges has been investigating the capabilities of our chosen GPR instrument to accurately differentiate between H20 and C02 at varying depths. There are some inherent material property differences between the permittivities of the two substances, but it remains to be determined, through extensive testing, how accurately we can differentiate the two at pressure and depth. Another challenge has been the objective of scaling down existing instrumentation and systems for use on a smaller rover application while maintaining validity and significance of science retrieved and integrity and functionality of hardware. These challenges, however, have been addressed both in our design and in the risk analysis and testing sections of this PDR, and we believe that with continued effort, our craft will be able to further our understanding of Mars' subsurface environment. Our payload will travel across millions of miles of space to reach its destination, safely deploy its landing systems, and explore the Martian terrain, quantifying water stock and paving the way toward the future human exploration and habitation of Mars.

VI) Activity Plan

6.1 Status of Activities and Schedule

All activities are currently up to date and on schedule. The PDR will be submitted for review and all further planning, coordination, engineering, and design will proceed its acceptance.

6.1.1 Budget Plan

Note: If further detail is required, the excel document for this schedule will be provided to necessary parties upon request to project management.

L'SPACE Budget	2019	2020	2021	2022	
Year	Yr 1 Total	Yr 2 Total	Yr 3 Total	Yr 4 Total	Cumulative Total
PERSONNEL					
Project Manager	$26,667	$80,000	$80,000	$2,849	**$189,516**
Duputy Project Manger	$26,667	$80,000	$80,000	$2,849	**$189,516**
Project Schedule Analyst	$26,667	$80,000	$80,000	$2,849	**$189,516**
Safety and Mission Assurance Manager	$26,667	$80,000	$80,000	$2,849	**$189,516**
Duputy Safety and Mission Assurance Manager	$26,667	$80,000	$80,000	$2,849	**$189,516**
Principal Scientist	$26,667	$80,000	$80,000	$2,849	**$189,516**
Radar Scientist	$26,667	$80,000	$80,000	$2,849	**$189,516**
Landing Site Operating Lead	$26,667	$80,000	$80,000	$2,849	**$189,516**
Navigation Engineering Lead	$26,667	$80,000	$80,000	$2,849	**$189,516**
Flight Systems Engineering Lead	$26,667	$80,000	$80,000	$2,849	**$189,516**
Ground Data Systems Analyst	$26,667	$80,000	$80,000	$2,849	**$189,516**
Radar Engineering Systems Analyst	$26,667	$80,000	$80,000	$2,849	**$189,516**
GPR Scientific Data Analyst	$26,667	$80,000	$80,000	$2,849	**$189,516**
Mission Operations Systems Engineer	$26,667	$80,000	$80,000	$2,849	**$189,516**
Navigation Engineer	$26,667	$80,000	$80,000	$2,849	**$189,516**
Lead Project Systems Engineer	$26,667	$80,000	$80,000	$2,849	**$189,516**
Deputy Systems Engineer	$26,667	$80,000	$80,000	$2,849	**$189,516**
Payload Systems Engineer	$26,667	$80,000	$80,000	$2,849	**$189,516**
GPR Mission Manager	$26,667	$80,000	$80,000	$2,849	**$189,516**
GPR Deputy Mission Manager	$26,667	$80,000	$80,000	$2,849	**$189,516**
Total Salaries	$533,340	$1,600,000	$1,600,000	$56,980	**$3,790,320**
ERE percentage	28.00%	28.00%	28.00%	28.00%	28.00%
Total ERE	$149,335	$448,000	$448,000	$15,954	$1,061,290
TOTAL PERSONNEL	$682,675	$2,048,000	$2,048,000	$72,935	**$4,851,610**
Total Personell (Post-Inflation)	$682,675	$2,109,440.29	$2,172,723	$80,565	**$5,045,404**
OTHER DIRECT COSTS					
Total Materials and Supplies	$0	$6,606,000	$0	$0	
Publications	$1,000,000	$0	$0	$0	**$1,000,000**
Total Travel	$0	$0	$18,575	$43,597	**$62,172**
Total Services	$0	$0	$0	$0	**$0**
Total Equipment	$0	$0	$0	$0	**$0**
Total Subcontracts (if you have contractors)	$0	$0	$0	$0	**$0**
Total Participant Support	$100,000	$300,000	$300,000	$10,800	**$710,800**
Tuition Remission (not applicable)	$0	$0	$0	$0	**$0**
Total Direct Costs	$1,782,675	$9,015,440	$2,491,298	$134,961	**$13,424,375**
Total MTDC	$1,682,675	$8,715,440	$2,191,298	$124,161	**$12,713,575**
Total Subcontract F&A	$0	$0	$0	$0	**$0**
College or University F&A	$168,268	$871,544	$219,130	$12,416	**$1,271,358**
Total F&A	$168,268	$871,544	$219,130	$12,416	**$1,271,358**
Total Project Cost	$1,950,943	$9,886,984	$2,710,428	$147,377	**$14,695,733**
FED FLOW THROUGH (JPL, ARC, etc.)	$0	$0	$0	$0	**$0**
Total Project Cost	$1,950,943	$9,886,984	$5,201,727	$282,338	**$17,321,992**

F&A %	10%	10%	10%	10%	
Inflation Factor	0%	3%	3%	3%	
Wages					
Scientist	80,000	82400	84872	87418	
Engineer	80,000				
ERE - Staff Rates	0.28	0.28	0.28	0.28	
EXAMPLE of Travel Worksheet - Use your own numbers!					
Travel Detail: ASU - Coco Beach, FL					
Duration (5 days; n-4 nights)					
Air-fare (roundtrip)		584	584	584	584
Per diem - Hotel ($121 per night x 4 nights)		484	484	484	484
Per diem - Food = $64 (day 1 and 5 @ 75%)		48	64	64	48
x 5 days (n-4 nights)		240	240	240	240
Car rental (compact)		50	50	50	50
x 5 days (weekly rate+ taxes)		327	327	327	327
Airport Parking, tolls, gas for rental vehicle		120	120	120	120
Total		1,755	1,755	1,755	1,755

Travel Detail: Research Excursions	Cost	# of Days	# of People	Total
Car Rental	$300.00		5	$1,500.00
Air-face(roundtrip)	$200.00		5	$1,000.00
Per diem - Hotel	$128.00	7	5	$4,480.00
Travel Days Stipend	$53.25	2	5	$532.50
Non-travel Days Stipend	$71.00	5	5	$1,775.00
Total				$9,287.50
Total (x3)	Travel expenses multiplied by 3 for: two trips to Gleen Research Center; and one trip to Johnson Space Center			$27,862.50
Travel Detail: Launch at Cape Canaveral	**Cost**	**# of Days**	**# of People**	**Total**
Car Rental	$300.00		22	$6,600.00
Air-fare(roundtrip)	$300.00		22	$6,600.00
Per diem - Hotel	$128.00	5	22	$14,080.00
Travel Days Stipend	$53.25	2	22	$2,343.00
Non-travel Days Stipend	$71.00	3	22	$4,686.00
Total				$34,309.00
Grand Travel Total				$62,171.50

Note: Component Cost includes 2x materials for 2 Functional Rovers, one for testing and one for flight.

Component	Cost
Rockets	*200000*
Computer	*900000*
Drive Motors	*6000*
Solar Panels	*20000*
Materials and Manufacturing of Structure	*1000000*
Heat Shield	*800000*
Parachute	*80000*
RIMFAX Redevelopment	*2000000*
Rimfax Equipment	*1000000*
Patent Licensing Allowance	*600000*
TOTAL	*$6,606,000.00*

Figure 44-48. Budget Summary

6.1.2 Schedule – L'SPACE Academy 1

The Following Gantt Chart outlines all major activity from the beginning of the project to the PDR:

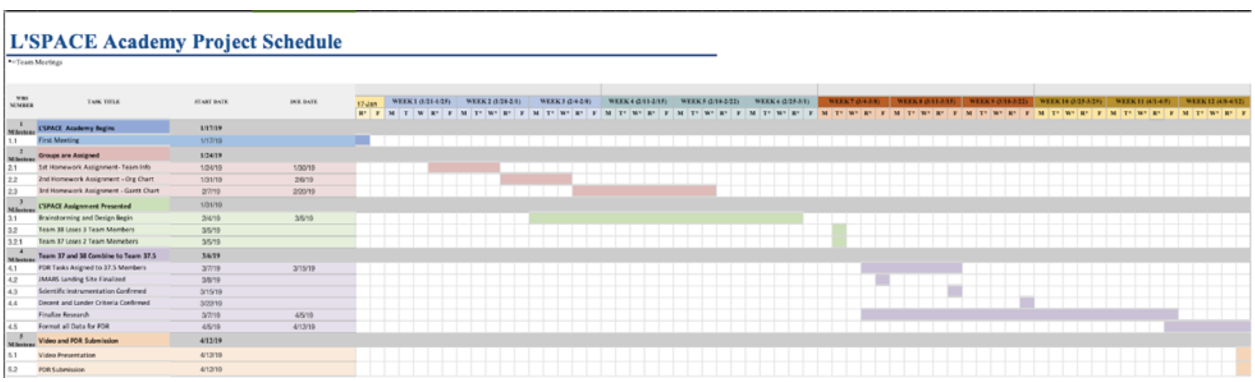


Fig 45. Gantt Chart Identifying Important Activities from Beginning of L'SPACE Academy to PDR

Note: If further detail is required, the excel document for this schedule will be provided to necessary parties upon request to project management.

6.1.3 Mission Education and Public Outreach Summary

To help raise awareness for the mission and involve the public, there will be three main approaches: social media, Highschool competitions, internships and presentations in local communities by ambassadors for the mission.

Social Media
Twitter, Facebook, and Instagram accounts will be created to represent the mission. They will be a group effort, managed generally by the ambassadors of the mission and supplemented with a variety of videos explaining in a clear and entertaining way the purpose of the mission and the implications it has for future human exploration of Mars. Special emphasis will be made of the human exploration part, to maximize the effect it has on mainstream media. Pictures of progress will be regularly posted within reason and updates on the progress will occasionally be posted to social media. To keep people interested over a long period of time, these posts of progress will be accompanied with questions to involve people, such as asking their opinions upon general things about space exploration.

Highschool Competitions
There will be competitions inspired by the mission that will be heavily advertised in select locations of particularly high population density, as well as locations near universities known for their engineering programs. Teams will be made up of 4-5 people, and there will be a prize for the winning teams, namely the opportunity to be an intern that will function as an ambassador for the mission. They will be lightly compensated to encourage the competitions and the awareness thereof. There will also be a way for students outside of the heavily targeted areas to be part of the competition through submitted their projects online. The projects will vary from year to year, and will be designed to provide unique engineering challenges. Less emphasis will be placed on the scientific component of the mission due to the level of education of the competitors, but there will still be a component thereof in the competition. They will be similar in many ways to the L'SPACE program, but less challenging to accommodate the different audience.

Internships
There will be internships, of both scientific and engineering positions for the development part of the mission. These positions will have a light stipend, and will be available to undergraduates. These will also be heavily advertised in universities in a large variety of locations.

Ambassadors
The winning teams of each highschool competition will become ambassadors for the mission, and will function as public representatives that will be given a variety of materials and resources to help them in the effort to raise awareness among people who are outside of the target audience of social media. Winners will be selected based off of regions, so that there is a number of ambassadors spread throughout the country that can raise awareness. They will be required, during a summer long internship that they will be involved in, to hold at least three informative events of their own. Criteria will be given, and resources to arrange these events will be provided, but they will be largely in charge of seeing that the events take place and will be accountable each

week in a form of a written report as to their goals, plans, and progress.

Presentations

These aforementioned presentations will be done in public places, with preference on having them take place in universities, schools, or planetariums when available. One formal presentation will be required per summer per group of ambassadors, and two others of any type will be required as well. For these latter two, this will be an opportunity for the interns to consider what would work best for their community, and what would reach the most amount of people. Reports of each event will be required directly following it.

VII) Conclusion

7.0 Summary of Mission

MARI-SIW's mission criteria will be to deorbit, survive atmospheric entry, complete a soft landing, and deploy the ground penetrating radar in order to retrieve detailed data about the subsurface material composition of its landing site and the immediate vicinity. By verifying the existence of accessible water stock, our mission will pave the way for significant scientific exploration of the Noctis Landing site, one of the most scientifically rich locations on Mars.

7.1.1 Progress on Mission Formulation and Design up to CDR

The following Gantt chart outlines our consideration of important deadlines and action items for the continuation of the MARI-SIW project, beginning at the submission of this preliminary design review, and continuing through presentation of the Critical Design Review.

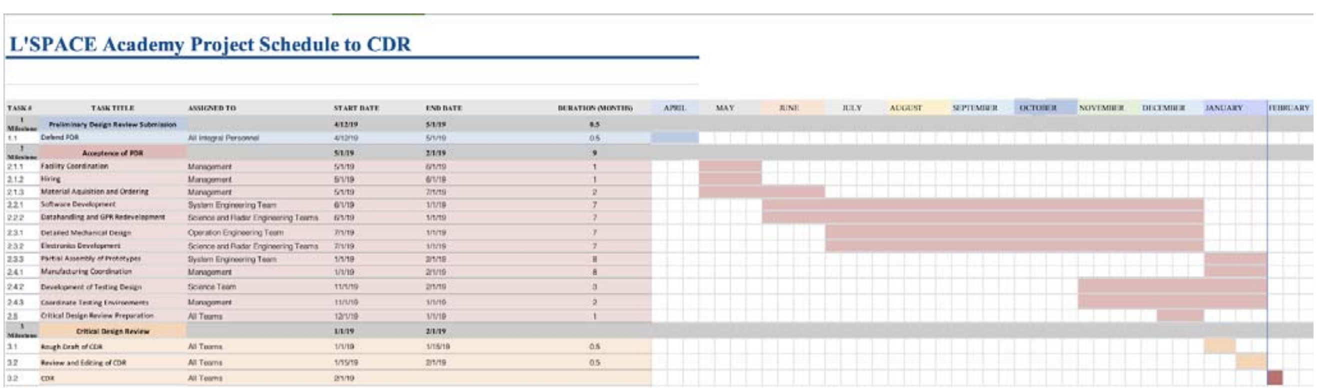


Fig 46. Gantt Chart Identifying Important Activities from presentation of PDR to finalization of CDR

Note: If further detail is required, the excel document for this schedule will be provided to necessary parties upon request to project management.

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