L'SPACE MCA

Planetary Habitat Operations & ExploratioN Investigation eXpedition

SYSTEM REQUIREMENTS REVIEW

TEAM 1 - P.H.O.E.N.I.X

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Table of Acronyms

Abbreviation	Definition
AC	Alternating Current
ADV	Action/Decision Vector
AI	Artificial Intelligence
APR	Annual Percentage Rate
AZUR	AZUR Space Solar Power GmbH
BOL	Beginning of Life
CAD	Computer-Aided Design
CCB	Change Control Board
CCHP	Constant Conductance Heat Pipe
CDH	Command and Data Handling
CDR	Critical Design Review
CF	Carbon Fiber
CM	Configuration Management
ConOps	Concept of Operations
COTS	Commercial Off-The-Shelf
CP-MU	Critical Protection - Monitoring Unit
CRISM	Compact Reconnaissance Imaging Spectrometer for Mars
DC	Direct Current

Abbreviation	Definition
DMU	Data Management Unit
DRE	Dust Removal Efficiency
DSN	Deep Space Network
DSON	Deep Space Optical Network
ECR	Engineering Change Request
ECC	Error-Correcting Code
EDS	Electrodynamic Dust Shield
EMI	Electromagnetic Interference
ERE	Employee Related Expenses
ESDMD	Exploration Systems Development Mission Directorate
FMEA	Failure Modes and Effects Analysis
FPS	Fluid Protection System
GNC	Guidance, Navigation, and Control
GNSS	Global Navigation Satellite System
GPR	Ground Penetrating Radar
HBS	Human Biology System
HDO	Semi-heavy Water (Hydrogen-Deuterium Oxide)
IMU	Inertial Measurement Unit
ISRU	In-Situ Resource Utilization

Abbreviation	Definition
JMARS	Java Mission Planning and Analysis for Remote Sensing
JPL	Jet Propulsion Laboratory
LOX	Liquid Oxygen
MCCET	Mission Concept Cost Estimate Tool
MCR	Mission Concept Review
MG	Mission Goal
Mini-TLS	Miniature Tunable Laser Spectrometer
MLI	Multi-Layered Insulation
MRO	Mars Reconnaissance Orbiter
MDR	Mission Definition Review
PCM	Phase Change Material
PDR	Preliminary Design Review
PLM	Product Lifecycle Management
PDS	Power Distribution System
PWR	Power
RAD	Radiation Assessment Detector
RFI	Radio Frequency Interference
RIMFAX	Radar Imager for Mars' Subsurface Experiment
RLS	Raman Laser Spectrometer

Abbreviation	Definition
ROI	Region of Interest
ROSE	Roll-Out Solar Array
RSR	Remote Sensing Reflectance
RTG	Radioisotope Thermoelectric Generator
SDR	Software Defined Radio
SMD	Science Mission Directorate
SRB	Systems Review Board
SRR	System Requirements Review
STM	Science Traceability Matrix
SYS	System
TCS	Thermal Control System
TBD	To Be Determined
TBR	To Be Resolved
TLS	Tunable Laser Spectrometer
TRL	Technology Readiness Level
UHF	Ultra High Frequency
VCHP	Variable Conductance Heat Pipe

1.0 System Requirements Review

1.1 Mission Statement

P.H.O.E.N.I.X (Planetary Habitat Operations & Exploration InvestigatioN eXpedition) is a low-cost, unmanned Mars rover mission designed to investigate subsurface ice reservoirs and characterize environmental hazards to support future human exploration. As part of NASA's Discovery-class architecture, the mission will land in the northern mid-latitudes of Mars, a region with high potential for shallow subsurface ice, favorable solar access, and traversable terrain. The mission aligns with the priorities of NASA's Science Mission Directorate (SMD) and Exploration Systems Development Mission Directorate (ESDMD) by targeting in-situ resource utilization (ISRU), hazard mitigation, and habitability research.

Science goals are derived from NASA's HBS-1LM Moon to Mars Objective and the Origins, Worlds, and Life Decadal Strategy goal Q10.3b. Four key objectives guide the mission: (1) assess the effects of radiation on a pressurized Earth-fluid sample using a submersible gamma neutron probe, informing ISRU and life support safety; (2) analyze subsurface stratigraphy and dielectric properties with RIMFAX radar to evaluate ice accessibility; (3) determine the Deuterium-to-Hydrogen (D/H) ratio in hydrated volcanic rock with a Miniature Tunable Laser Spectrometer to trace water source evolution; and (4) identify the crystal structure of asteroid-impact minerals using a Raman Laser Spectrometer to study endogenic and exogenic processes affecting water distribution.

To accomplish these objectives, P.H.O.E.N.I.X is engineered for long-duration autonomous operation in harsh Martian conditions. Its mechanical subsystem utilizes a heritage rocker-bogie suspension, titanium fittings, and an aluminum chassis for resilient mobility across 10+ km of variable terrain. The power subsystem includes ROSA-style solar panels, a 500 Wh lithium-ion battery, and a smart distribution unit for efficient, fault-tolerant energy management. The CDH system features radiation-hardened processing and redundant communication pathways. Thermal control maintains internal temperatures from -120°C to +30°C using multilayer insulation and resistive heating. All subsystems meet planetary protection guidelines and mission constraints on mass, volume, and cost.

By locating accessible water ice, monitoring radiation exposure, and expanding understanding of Martian water cycles, P.H.O.E.N.I.X delivers critical data to guide astronaut landing site selection and surface system design. The mission represents a significant step toward enabling a sustainable human presence on Mars.

1.2 Science Traceability Matrix

The STM focuses on two main goals for the mission. The first goal is the Human Exploration goal, HBS-1LM, which is addressed by two objectives. The first objective shown in the STM aims to investigate the long duration Martian environmental impacts on a protected and pressurized earth-fluid sample, documenting unknown hazards that may threaten the integrity of future mission-critical life support, rocket propellant, and agricultural fluids in transportation, long-duration storage, and recycling. The CP-MU DMU-100 Submersible Gamma Neutron Probe was chosen to complete the objective, using a passive ionization chamber to monitor gamma radiation levels in μSv/h and recording data at weekly intervals over a one-year period for transmission back to Earth. The second science objective shown in the STM, aims to investigate how subsurface stratigraphy, dielectric properties, and dust layer thickness influence the accessibility and long-term stability of near-surface ice. The Radar Imager for Mars' Subsurface Experiment (RIMFAX) is the selected instrument for this objective, using ground-penetrating radar to analyze radar signal delays and reflection strength, allowing to identify subsurface layer boundaries, material transitions, and dielectric properties indicative of dust and possible ice-rich zones [18]. From these observables, dielectric permittivity and radar wave velocity can be estimated to derive subsurface material properties such as layer thickness, composition variation, and porosity across a 10 km traverse [18]. There is direct alignment with NASA's HBS-1LM science goal of understanding the environmental effects, risks, and hazards faced when exposing mission-critical fluids and their protective storage systems to the harsh Martian surface during long-duration missions, as safeguarding life support systems, rocket propellant, and recycled fluids from radiation hazards is vital to mission success, astronaut health, and prevention of physical injury from depressurization and stable access to subsurface ice is essential for life support, fuel production, and long-term habitat sustainability. In addition, these objectives support mission constraints by operating within the instrument's 15kg allocated mass, volume constraint, and resolution requirements while providing essential periodic data that will advance material science and engineering innovation for fluid storage systems that will be utilized during long-duration missions on Mars and environmental data to inform ISRU planning and reduce operational risk in future human missions to Mars [23, 31].

The second goal is the Science Exploration goal, Q10.3b, which is also addressed by two objectives. In the first objective the deuterium-to-hydrogen ratio in hydrated volcanic rock serves as a geochemical tracer in understanding the history of water on Mars by providing insights into the sources, losses, and recycling of water. Using the Miniature Tunable Laser Spectrometer (TLS) the objective will collect absorbance spectra in the 2500–25,000 nm range of H in selected hydrated volcanic rock samples at multiple surface sites in order to define the relative abundance of protium and deuterium within samples of hydrogen from hydrated volcanic rock. This will demonstrate an understanding of the long-term controls that have influenced the availability of liquid water on Mars through both endogenic, such as internal volcanic and geologic processes, and exogenic, such as surface-atmospheric interactions. The second objective aims to determine the crystal structure of minerals formed by asteroid impacts that interact with exposed subsurface ice within 0-1m depth as derived from the broader scientific goal of understanding the long-term endogenic and exogenic controls on the presence of liquid water on planet Mars. The selection of a Raman Laser Spectrometer with a predicted spectral resolution of 10 cm^{-1} and a peak separation capability of 6–8 cm^{-1} will allow for precise and accurate identification of hydroxyl groups and collect raman spectra in Olivine in selected asteroid rocks at multiple surface sites. This investigation explores the geological history of Mars and its evolution to the present state through the interaction between the dynamic forces on planet Mars that have formed and reshaped its surface through time as stated in the scientific goal. Furthermore, the goal is derived from multiple stakeholders and customer constraints. The stakeholder's experiment constraint reserves 185 kg total of mass for the system so perfectly falling within mass constraints, and an overall budget of $450 million to ensure that only compact, cost-effective science instruments can be used. Under the prohibited materials constraint, the Radioisotope Thermoelectric Generator or RTG is prohibited, and radioactive materials are limited to under 5 grams, which are not used or exceeded, concluding that these experiments help create the opportunity to carry out these objectives to serve the mission's purpose.

<table>
<tr><th rowspan="2">Science Goals</th><th rowspan="2">Science Objectives</th><th colspan="2">Science Measurement Requirements</th><th rowspan="2" colspan="2">Instrument Performance Requirements</th><th rowspan="2">Predicted Instrument Performance</th><th rowspan="2">Instrument</th><th rowspan="2">Mission Requirements</th></tr>
<tr><th>Physical Parameters</th><th>Observables</th></tr>
<tr><td rowspan="8">“HBS-1LM: Understand the effects of short- and long-duration exposure to the environments of the Moon, Mars, and deep space on biological systems and health, using humans, model organisms, systems of human physiology, and plants.” — Moon to Mars Objectives, NASA</td><td rowspan="4">Investigate the long duration Martian environmental impacts on a protected and pressurized earth-fluid sample for unknown hazards that may threaten the integrity of future mission-critical life support, rocket propellant, and agricultural fluids.</td><td rowspan="4">Periodically monitor the Earth fluid sample for risks, hazards, and contamination that may bypass the custom-engineered Fluid Protection System’s protective layers and document via data generation.</td><td rowspan="4">Use a passive ionization chamber to monitor gamma radiation levels in μSv/h, recording data at weekly intervals over a one-year period for transmission back to Earth.</td><td>Range</td><td>1 μSv/h to 10Sv/h</td><td>1 μSv/h to 10Sv/h</td><td rowspan="4">CP-MU DMU-100 Submersible Gamma Neutron Probe</td><td rowspan="3">The instrument must survive fluid submersion for a minimum of one year while measuring for potential radiation contamination within the fluid protection system.</td></tr>
<tr><td>Operating Temperature</td><td>30°C to +57°C</td><td>30°C to +57°C</td></tr>
<tr><td>Accuracy</td><td>±5%</td><td>±10%</td></tr>
<tr><td>Time Constant</td><td>12 seconds slow</td><td>2 seconds fast, 12 seconds slow</td><td rowspan="3">The instrument must study the difference in permittivity to identify insulating dust layers and potential ice-rich zones</td></tr>
<tr><td rowspan="4">Investigate how subsurface stratigraphy, dielectric properties, and dust layer thickness affect the accessibility and long-term stability of near-surface water ice, in support of in-situ resource utilization and environmental risk reduction for future human exploration.</td><td rowspan="4">Estimate dielectric permittivity and radar wave velocity to characterize subsurface material properties, including layer thickness, composition changes, and porosity variations across a 10 km traverse.</td><td rowspan="4">Analyze radar signal delay and reflection strength to determine layer boundaries, depth to subsurface features, and dielectric (Ɛ) contrasts indicative of dust deposits and possible ice-rich zones.</td><td>Penetration Depth</td><td>≥ 10 km</td><td>≥ 10 m</td><td rowspan="4">Radar Imager for Mars’ Subsurface Experiment (RIMFAX)</td></tr>
<tr><td>Frequency Range</td><td>100-1200 MHz</td><td>150-1200 MHz</td></tr>
<tr><td>Permittivity Range</td><td>ΔƐr ≤ 0.1</td><td>ΔƐr ≤ 2</td><td rowspan="2">The instrument must detect the subsurface layering to a depth of at least 10 m to assess ice stability underneath dust and regolith</td></tr>
<tr><td>Vertical Resolution</td><td>≥ 15 cm</td><td>15 cm - 30 cm</td></tr>
</table>

<table>
<tr><td rowspan="8">“Q10.3b: What are the long-term endogenic and exogenic controls on the presence of liquid water on terrestrial planets?”— Origins, Worlds, and Life: A Decadal Strategy for Planetary Science and Astrobiology 2023–2032</td><td rowspan="4">Determine the Deuterium to Hydrogen (D/H) ratio in hydrated volcanic rock on Mars' surface.</td><td rowspan="4">Define the relative abundance of protium and deuterium within samples of hydrogen from hydrated volcanic rock.</td><td rowspan="4">Collect absorbance spectra in the 2500–25,000 nm range of H in selected hydrated volcanic rock samples at multiple surface sites.</td><td>Wavenumber Range</td><td>3593.3-3594.3 cm^-1</td><td>3593.3-3594.3 cm^-1</td><td rowspan="4">Miniature Tunable Laser Spectrometer (Mini-TLS)</td><td rowspan="3">System must navigate to and collect samples of hydrated volcanic rock.</td></tr>
<tr><td>Spectral Resolution</td><td>0.0001 cm^-1</td><td>0.0001 cm^-1</td></tr>
<tr><td>Sensitivity</td><td><80 ppb</td><td>10 ppb</td></tr>
<tr><td>Integration Time</td><td>1 s</td><td>2.4 s</td><td rowspan="3">System must have the ability to heat volcanic rock to 935 K to study structural water released as gas.</td></tr>
<tr><td rowspan="4">Determine the crystal structure of minerals formed by asteroid impacts interacting with exposed subsurface ice.</td><td rowspan="4">Identify chemical structure, crystal structure, and bond structure of Olivine from asteroids.</td><td rowspan="4">Collect raman spectra in the 11,111–33,333 nm range of Olivine in selected asteroid rocks at multiple surface sites.</td><td>Mineral Identification Accuracy</td><td>±10%</td><td>≥ 90%</td><td rowspan="4">Raman Laser Spectrometer (RLS)</td></tr>
<tr><td>Detection Sensitivity</td><td>≤ ~100 ppm</td><td>6–8 cm^{-1} peak separation</td></tr>
<tr><td>Power Consumption</td><td>20 - 30W</td><td>Between 20 - 30 watts</td><td rowspan="2">System must operate in the Martian Temperature range of 293.15° K to 120.15° K</td></tr>
<tr><td>Spectral Resolution</td><td>10 cm^{-1}</td><td>10 cm^{-1}</td></tr>
</table>

Figure 1.2.1: Science Traceability Matrix

1.3 Summary of Mission Location

The location selected for the P.H.O.E.N.I.X mission is the Erebus Montes region of Arcadia Planitia. The two factors which play into selecting a mission location are: its adherence to the customer constraints and the location containing regions of interest for completion of the science objectives. Erebus Montes is located within 60° latitude North or South, the Potential High Priority Radar Targeting Zone (PHPRTZ), and a region containing excess subsurface ice within the upper meter of the regolith.

Erebus Montes has ample regions of scientific interest to complete the objectives located within the STM. The objectives necessitate the presence of hydrated volcanic rock and asteroid impacts which exposed subsurface ice. The region is situated between two Amazonian lava flows and contains exposed terrain from the older Noachian-Hesperian era [50]. This access to both new and old geological terrain with glacial processes offers the ability to categorize the way water ice has interacted with Martian geological and climate changes [51]. The TES dust index in **Figure 1.X** shows the region is moderately dusty, posing challenges to the systems operation and communication. However, this lends the advantage of the subsurface ice being more cold and therefore more stable [50]. There is a flat pocket of less dust in the Northwestern corner of the ROI where the rover will land to avoid communication issues during deployment.

The P.H.O.E.N.I.X ROI is a 60 km wide, 17 km long ellipse. The ROI sits on a region of Noachian-Hesperian transition terrain in the Northeastern region of Erebus Montes [50]. In the Northwest corner of the ROI there is a concentric crater fill (CCF). CCFs form through asteroids contacting ice-rich regions with subsequent glacial movement carving the crater walls [54]. This introduces the asteroid minerals to interactions with ice, interactions which are important to understand the way water can change crystal structures. Evidence points towards asteroids bringing life-sustaining elements (C, H, H, P) to Mars [68]; many studies hypothesize that organics may be preserved in ice-rich environments [66]. The possibility of organics within asteroid minerals is real and must be studied at the CCF of interest.

There are multiple ice rich lobate debris aprons (LDA) in the Southern region of the ROI [50]. LDAs are formed when rock debris piles up next to escarpments. The ROI LDAs consist of the Noachian-Hesperian crust, a period with mass volcanic activity [54] [65]. The Noachian-Hesperian terrains that feed into LDAs have been found to contain hydrated minerals suggesting the LDAs would make hydrated volcanic rock easily accessible [66]. This makes Erebus Montes' LDAs a compelling target for collecting preserved, hydrated volcanic material.

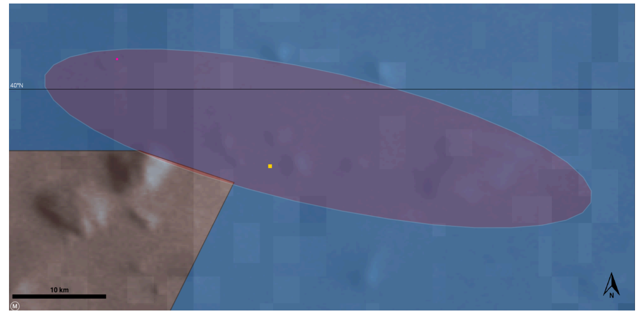
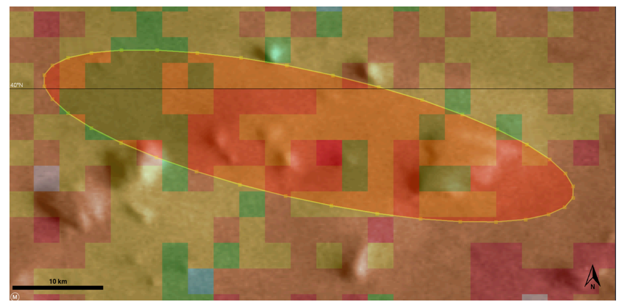
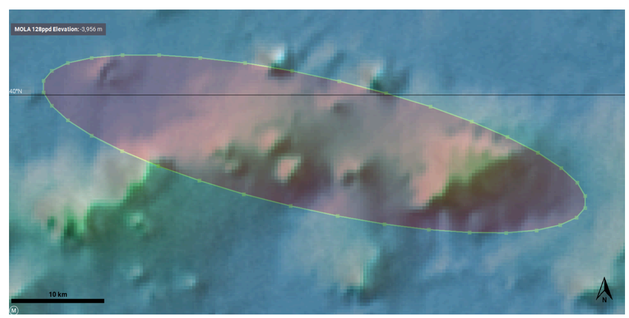
ROI Overlaid with PHPRTZ This displays the region of interest within the contractors requirements. Additionally the small pink and yellow squares show the locations of CCFs and LDAs respectively.	
ROI TES Dust Index As stated Erebus Montes is dusty, but in the ROI there are regions where the rover can avoid dust.	
ROI Elevation Map The region has large cliffs where the LDAs are present. These cliffs will need to be navigated to study hydrated volcanic rock.	

Figure 1.X: P.H.O.E.N.I.X ROI JMARS Maps

1.4 Mission Requirements

Customer constraints are a key driver of mission architecture, which determines the high-level requirements concerning mass, volume, and budget. Team P.H.O.E.N.I.X seeks to meet the system constraints presented by NASA, serving as the funding agency for the Mission Concept Academy's Discovery-class mission.

The spacecraft shall not exceed a mass of 200 kg. In a stored configuration, the spacecraft shall not exceed the dimensions of 2.5 m x 2.5 m x 2.5 m. This volume will house all the electronics, instruments, and payload suite. The spacecraft shall maintain the stored configuration for the entirety of the launch, transit, and entry into the Martian atmosphere. In an expanded form, there is no volume or mass constraint placed on the spacecraft. The spacecraft shall demonstrate resistance to temperatures consistent with atmospheric entry and descent. The spacecraft shall incorporate a landing attenuation system capable of withstanding surface impact.

After deployment on the landing site selected, the spacecraft shall traverse the terrain effectively to travel a minimum of 10 km. The spacecraft shall demonstrate an ability to traverse various Martian terrains, including sandy regions, icy regions, and small, medium, and large-sized rocks. The spacecraft shall demonstrate the ability to endure fluctuations in Martian atmospheric conditions, including dust storms, diurnal temperature variations, and reduced atmospheric pressure.

The spacecraft shall carry a scientific payload containing all instrumentation to complete science objectives. The volume of the scientific payload shall not exceed a cube of dimensions 0.5 m x 0.5 m x 0.5 m, nor a mass of 15 kg. This is to ensure the mission satisfies the human exploration goal and gets samples from the Martian surface that can be transmitted back to Earth for research. Furthermore, this research will contribute a great deal to the future of sustainability on Mars and future manned missions.

P.H.O.E.N.I.X is a discovery mission and not a flagship mission; hence, the budget allocated to this mission is 450 million USD and shall be used effectively for the manufacturing of the spacecraft, its components, employee-related expenses (ERE), and testing of the spacecraft. The Spacecraft system shall not have a Radioisotope Thermoelectric Generator (RTG) or any similar power generation system. Furthermore, any radioactive material is allowed for use on other spacecraft subsystems, but cannot exceed a cumulative mass of 5g of radioactive material on all subsystems. The spacecraft must be ready for integration with the other systems by October 1st, 2029, and must be ready for launch on December 1st, 2029. The launch site shall be in Cape Canaveral in Florida

Req #	Requirement	Rationale	Parent Req.	Child Req.	Verification Method	Req Met
MG 0.1	System shall survive the martian environment for 1 year	The system must be able to survive the martian environment to fulfill its purpose and send data back to earth ground station and potentially return martian samples		SYS.02	Demonstration	Met
MG 0.2	Shall investigate the presence of ice glaciers on Mars for future missions and sustainability	Foundational science driver for the mission:Human habitation requires large volumes of drinkable water, water for propellant and agricultural use for long term sustainability missions on Mars	Customer	SYS.01 SYS.03 SYS.05 SYS.06 SYS.07 SYS.08 CDH.01 CDH.02	Demonstration	Met

Figure 1.4.1: Requirements Table

The system-level requirements for the P.H.O.E.N.I.X mission are derived directly from NASA's customer constraints and ESDSD objectives. These top-level requirements form the foundation upon which all subsystem requirements are built and are essential to ensure the spacecraft can successfully complete its science operations and contribute valuable data toward future human exploration of Mars.

The first MG states that the spacecraft must survive the Martian environment for the full duration of its science operations. This includes enduring extreme temperatures, dust storms, and radiation conditions typical of the Martian surface. Demonstrating environmental survivability is critical for collecting and transmitting meaningful data back to Earth.

The second MG states that the system shall investigate subsurface ice on Mars, aligning with both the scientific and human exploration goals. This foundational objective drives the selection of instrumentation, rover mobility capabilities, and site selection criteria. It supports key downstream system requirements such as power (SYS.01), thermal regulation (SYS.02), mobility (SYS.03), communications (SYS.07), and planetary protection (SYS.08).

Each mission goal has been decomposed into a series of child system requirements (SYS), with clearly defined verification methods such as demonstration, test, analysis, and inspection. For instance, mobility requirements ensure the system can traverse at least 10 km over varied Martian terrain, while power and thermal systems must maintain subsystem functionality during solar cycles and cold nights.

1.5 System Definition

1.5.1 Spacecraft Overview

The P.H.O.E.N.I.X spacecraft is a low-cost Mars rover designed to investigate near-surface water ice and assess environmental risks to support future human exploration. The system architecture integrates six core subsystems: Mechanical, Power, Command & Data Handling (CDH), Thermal, Payload and Comms.

The Mechanical Subsystem comprises the chassis, rocker-bogie suspension, and wheels, this subsystem provides structural integrity and terrain adaptability. It must withstand static loads up to 1500 N, vibrational frequencies up to 2000 Hz, shock loads up to 6000 N, and maintain ≥95% actuation performance throughout the mission (MECH.01–MECH.05). The Power Subsystem comprises ROSA-based solar panels, a 500 Wh lithium-ion battery, and a redundant power distribution unit. The subsystem must generate at least 200 Wh per sol, handle 120 W peak loads, and maintain safe operation from -30°C to +50°C (PWR.01–PWR.05). The Command and Data Handling Subsystem comprises onboard processing and communication between systems and Earth. Requirements include a 1 GHz processor, 1 Mbps uplink, and 16 Kbps downlink, with sufficient bandwidth to handle telemetry and science data (CDH.01–CDH.02). The Thermal Subsystem regulates temperature-sensitive components using passive insulation and active heating. It maintains the system within 303–313 K and instrument-specific thermal ranges (TCS.01). The Payload Subsystem contains all instruments include RIMFAX (subsurface radar), a Gamma Neutron Probe (radiation monitoring), a Miniature Tunable Laser Spectrometer (D/H ratio), and a Raman Laser Spectrometer (mineralogy). Each must meet strict mass/volume limits and operate under mission-specific scientific thresholds.

All subsystems are designed within a maximum stored configuration of 2.5 m x 2.5 m x 2.5 m, a mass cap of 200 kg, and a cost ceiling of $450M, with readiness milestones targeted for integration by October 2029 and launch by December 2029.

Req #	Requirement	Rationale	Parent Req.	Child Req.	Verification Method	Req. met?
MG 0.1	System shall survive the martian environment for a minimum of one year.	The system must be able to survive the martian environment to fulfill its purpose and send data back to earth ground station and potentially return martian samples		SYS.02 CDH.04	Demonstration	Met
MG 0.2	Shall investigate the presence of ice glaciers on Mars for future missions and sustainability	Foundational science driver for the mission: Human habitation requires large volumes of drinkable water, water for propellant and agricultural use for long term sustainability missions on Mars	Customer	SYS.01 SYS.03 SYS.05 SYS.06 SYS.07 SYS.08 CDH.01 CDH.02	Demonstration	Met
SYS.01	The system shall have sufficient power to carry out the objectives for the duration of its mission	System needs power to operate, communicate back to earth and carry out its objectives	MG 0.1 MG 0.2	PWR.01 PWR.02	Test	Met
SYS.02	System shall maintain operating temperatures and survive the harsh thermal environment ranging from on the martian surface	The system and its scientific instrumentation must be kept in operating temperature ranges in order to function properly	MG 0.1	TCS.01	Test	Met
SYS.03	System shall traverse the martian surface smoothly and reach the required science points of interest	Points of interest are marked across potential high priority Radar targeting zones on Mars that are defined by the thickness of the atmosphere to allow for easy landing and research point.	MG 0.2	CDH.01 CDH.02 CDH.03 CDH.04 MECH.01 MECH.02 MECH.03 MECH.04	Test	Met
SYS.04	System shall not exceed a total mass of 200kg	Constraints provided by NASA for the mission	Customer		Inspection	Met
SYS.05	System shall have a backup that is always ready to take over	In the case of failure, if the main system fails, the backup can takeover and still carry out the	MG 0.1		Analysis	Met

		mission				
SYS.06	System must withstand the solar winds for the duration of its mission	All components on the rover must be strong enough to withstand the strong solar winds on mars	MG 0.1		Test	Met
SYS.07	System shall send and receive data collected with the science instrumentation back to the earth ground station	Data sent back to the earth ground station about Mars will be essential to future scientific research for sustainability on mars	MG 0.2	CDH.01 CDH.02 CDH.03 CDH.04	Analysis	Met
SYS.08	System shall comply with all applicable planetary protocol regulations	NPR 8020.12D *Planetary Protection Provisions for Robotic Extraterrestrial Missions*	MG 0.2		Analysis	Met
SYS.09	Radioactive material used for any subsystem excluding the power subsystem shall not exceed a total mass of 5g	Constraints provided by NASA for the mission	Customer		Inspection	Met
SYS.10	System shall not make use of a Radioisotope Thermoelectric Generator (RTG) or any derivative thereof for power generation	Constraints provided by NASA for the mission	Customer		Inspection	Met
SYS.11	System shall not exceed the dimensions of 2.5 m x 2.5 m x 2.5 m while in its stored configuration	Constraints provided by NASA for the mission	Customer		Inspection	Met
SYS.12	System shall not exceed a cost of $450M	Constraints provided by NASA for the mission	Customer		Inspection	Met
CDH.01	The Command and Data Handling (CDH) system shall have a minimum uplink rate of 1 Mbps and a minimum downlink rate of 16 Kbps.	The system must be able to send scientific and telemetric data to and from Earth.	SYS.03 SYS.07	CDH.02	Analysis	Met
CDH.02	The CDH system shall have a minimum processing rate of 1 GHz.	The system must be able to process the scientific and telemetric data it receives from both Earth and instrumentation/sensors.	SYS.03 SYS.07 CDH.01	CDH.03 CDH.04	Test	Met

CDH.03	The CDH system shall have a minimum memory of 8 GB of RAM.	Provides working memory for executing flight software, processing sensor data, running algorithms, and sending commands.	SYS.03 SYS.07 CDH.02		Inspection	Met
CDH.04	The CDH system shall have a minimum storage of 10 TB.	Due to incoming datastreams and intermittent opportunities for uplink, large onboard data storage helps limit data loss and data can be stored for the duration of the mission.	MG 0.1 SYS.03 SYS.07 CDH.02		Inspection	Met
MECH.01	The chassis shall tolerate a static load up to 1500 N.	The system must not risk fracture or fatigue that would result in complete structural failure and inability to carry out the mission.	MG.01		Analysis	Met
MECH.02	Mechanical subsystems shall tolerate vibrations up to 2000 Hz.	The system must be able to tolerate vibrations from travel.	SYS.03		Analysis	Met
MECH.03	Suspension sub assembly shall tolerate shock loads up to 6000 N.	The system must be able to tolerate shock loads from travel.	SYS.03		Analysis	Met
MECH.04	Suspension sub assembly shall withstand a 45 degree tilt in any direction.	The system must be able to tolerate tilts from travel.	SYS.03		Demonstration	Met
PWR.01	The system shall generate at least 200 Wh per sol under average Martian insolation and withstand peak power draws of 120 W for a minimum of 30 minutes up to 3 times per Sol.	Supports nominal rover operation, mobility, communication and science payloads	SYS.01		Analysis	Met
PWR.02	Solar panels shall deploy autonomously and tolerate up to 20 m/s wind.	Ensures survivability under common Martian conditions	SYS.06		Test	Met
PWR.03	The system shall maintain operation of critical components between -30°C to +50°C.	Ensures battery and electronics functionality	SYS.06		Test	Met

PWR.04	The system shall minimize integration risk using EMI shielding to absorb shockwaves and modular connectors to divert signals from critical components.	Reduces failure during integration and operations	SYS.05		Test	Met
PWR.05	Provide redundant power paths for critical systems via RCE's and power and analogue modules	Enhances fault tolerance by dividing power distribution and handling between two computers, helping to reduce uptime	SYS.01 SYS.05		Test	Met
TCS.01	The Thermal Control System (TCS) shall help maintain the system at the allowable temperature range of 303 K to 313 K.	This ensures the TCS keeps components within safe temperature limits to prevent failure from Mars' extreme thermal conditions.	SYS.02		Test	Met
PAYL.01	RIMFAX shall detect radar signal changes in subsurface layers down to 10 meters depth.	Fulfills Human Exploration science objective #2 by identifying ice-rich zones defined by a permittivity difference of less than or equal to 0.1.	MG.02		Test	Met
PAYL.02	CP-MU Submersible Gamma Neutron Probe shall measure and record radiation dosage measurements ranging from 1 μSv/h to 10 Sv/h.	Fulfills Human Exploration science objective #1 by recording Martian environmental radiation impact data in the transported Fluid Protection System over a minimum one year duration period.	MG 0.1		Test	Met
PAYL.03	Miniature Tunable Laser Spectrometer shall collect and receive data from hydrated volcanic rock within a 1-second integration time.	Fulfills Science Exploration science objective #2 by determining the Deuterium to Hydrogen (D/H) ratio.	SYS 0.7		Test	Met
PAYL.04	Raman Laser Spectrometer shall collect Olivine Raman spectra in the 11,111–33,333 nm range.	Fulfills Science Exploration science objective #2 by identifying the chemical structure, crystal structure, and bond structure of Olivine from asteroids.	TCS.05		Test	Met

Figure 1.5.1.2: System Requirements Table

1.5.2 Mechanical Subsystem

1.5.2.1 Mechanical Subsystem Requirements

The mechanical subsystem contains the chassis, suspension, and wheels, and must ensure structural integrity, terrain adaptability, and environmental resilience throughout the mission.

The chassis must withstand static loads up to 1500 N, shock loads up to 6000 N, and vibrational frequencies up to 2000 Hz, as verified by finite element analysis (FEA). The system shall withstand 45° tilts and maintain functionality across rugged terrain, minimizing risk of tip-over or mechanical failure. All moving components shall preserve at least 95% of range of motion over the mission duration, with dust-tolerant designs verified in CAD. Components must resist performance degradation from dust, radiation, and thermal cycling typical of the Martian environment. Verification methods include FEA simulations and CAD-based tolerance analysis for thermal, mechanical, and particulate stress cases.

These requirements are derived from system requirements and are designed to minimize mechanical risks over the intended mission lifetime.

1.5.2.2 Mechanical Subsystem Overview

Each subassembly within the mechanical subsystem has been designed with emphasis on safe traveling ability, reliable performance in Martian conditions, and integration of subassemblies. The challenges that influenced such decisions include traversing rugged terrain, exposure to extreme temperatures, and strict mass criterion [26]. The rover will encounter rocky obstacles, thermal cycling, and various loads, and the mechanical subsystem was designed with these challenges in mind.

The chassis is the structural backbone of the rover,holding all subsystems together. For a durable and lightweight frame, the chassis will be made of aluminum 6061. The chassis will offer protection to scientific instruments and other subassemblies from dust and radiation exposure. The chassis will feature thicker aluminum walls with a tantalum coating, which is proven to significantly reduce radiation [21]. The chassis will connect to the suspension system with a differential through titanium fittings, which will allow the chassis and its inner components to remain at the average position of the suspension system for increased stability.

The rover will implement a rocker-bogie suspension system. Featured on all NASA Martian rover missions, this suspension system has demonstrated its ability to consistently and safely traverse the ragged Martian surface and was therefore chosen for this mission [39]. This suspension system consists of six wheels connected through a pivoting linkage system that allows each wheel to move independently and climb over obstacles twice the diameter of the wheels.

The rocker refers to the longer pivoting arms, which connect to the front wheels and the bogies. The bogie refers to the ‘responding’ arms attached to the rockers that connect to the middle and rear wheels. As the rover encounters obstacles, the rockers and bogies pivot independently at the joints to maintain constant contact with the ground. The chassis remains at the average pitch angle of the two rockers, which are connected to each other and the chassis through a differential. The differential tilts as obstacles are encountered to maintain at least one rocker wheel on the ground at all times for stability.

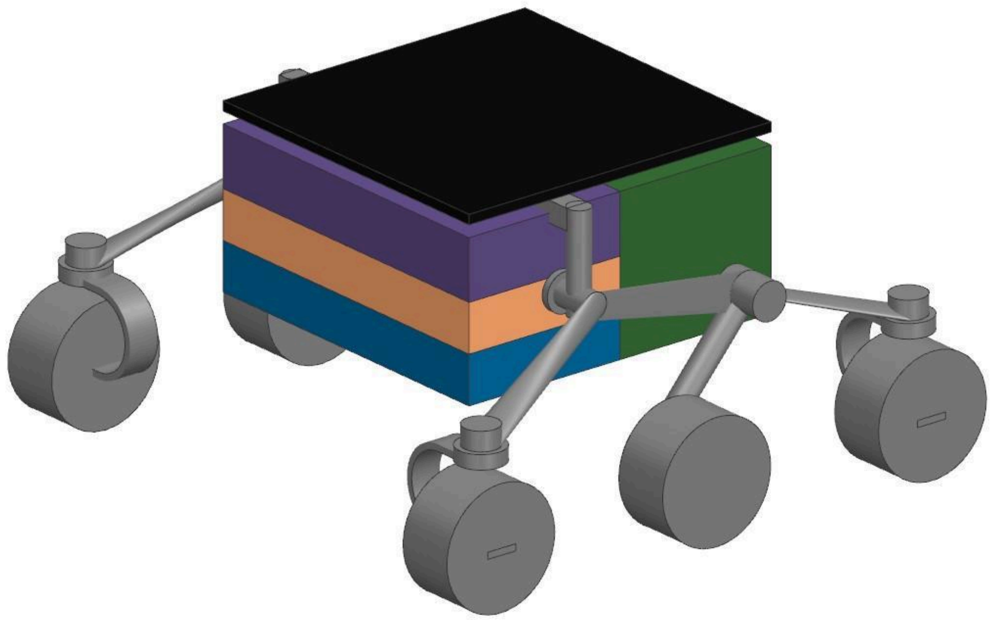


Figure 1.5.2.2.1: High-Level Rover Concept

The rocker and bogie bars will be made of titanium tubing, like past rovers. The rocker is L-shaped, with one end attached to a drive wheel and the other attached to the bogie through a pivot point. The bogie is similarly L-shaped, with one end attached to a drive wheel and the other attached to an idle wheel used for balance. The bars will connect to the wheel mounts through titanium fittings, allowing for each wheel to have independent motion. The rockers connect to the bogies through a pivot joint, and the rockers connect to each other and the chassis through a differential. The differential connects the rockers to the chassis through similar titanium fittings as those used on the wheel mounts, which allows for the rockers to pivot independently and the chassis to remain at its average position. This system distributes the rover’s weight equally among the six wheels, allowing for balanced loads. Furthermore, this will avoid one wheel or leg joint wearing faster than another, which is an added benefit of the rocker-bogie system along with excellent obstacle handling.

Subassembly	Mass (kg)	Dimensions (m^3)	Max Power Draw (W)
Chassis	65	3	0
Suspension	20	8.75	0
Wheels	3	0.02	200

Figure 1.5.2.2.2: Mass, Volume, and Power Subassembly Estimates

Similar aluminum chassis have been flown on past NASA Martian rovers, but not in the intended environment. Similarly, the rocker-bogie system has been implemented on every NASA Martian rover. The fundamental concept is flight proven, but hardware for PHOENIX will not be exactly similar to those flown on previous missions. Since geometries and specific component designs will be modified to fit PHOENIX's design, the chassis and suspension systems will have a TRL of 7.

Subassembly	TRL	Justification
Chassis	7	Similar chassis are flown on every rover mission, but the particular version designed for PHOENIX will be different from past rovers. At the highest, PHOENIX's chassis will demonstrate operation in a similar, but not actual, environment.
Suspension	7	Concept is flight proven, but the particular version designed for PHOENIX will be different from past rovers. At the highest, PHOENIX's suspension system will demonstrate operation in a similar, but not actual, environment.
Wheels	7	Aluminum wheels with symmetrical treads have substantial flight heritage (e.g., Sojourner, Spirit, Opportunity), and the performance characteristics are well-understood in Martian environments.

Figure 1.5.2.2.3: Mechanical Subassembly Requirements Table

1.5.2.3 Mechanical Subsystem Trade Studies

Mechanical Subsystem Trade Study on Leg Material

Figure 1.5.2.2.1: Leg Material Trade Study

The legs of the suspension system must support the weight of the rover during travel, handle shock loads experienced throughout travel, and avoid performance degradation throughout constant loads and thermal cycling. Four aspects were weighed equally to determine the optimal material for the legs: density, load-bearing ability, shock absorption, and thermal performance. Three materials were considered for the legs - aluminum 6061, titanium, and a carbon fiber composite - based on common aerospace materials and previously used rover leg materials.

The legs' material is favored to have a high density unlike other components of the mechanical subsystem. Having a center of mass closer to the ground is beneficial for stability, so the three materials were ranked by their densities. Titanium had the highest of 4.51 g/cm³. Aluminum 6061 had the second highest of 2.7 g/cm³, and CF composite had the lowest of 1.5-2 g/cm³.

High shear strength is an important criterion to ensure the legs could reliably tolerate all encountered loads, particularly static and shock loads. Having a high shear strength avoids risk of material fatigue and ensures the legs can comfortably tolerate all loads. Titanium had the highest shear strength of approximately 500+ MPa. Aluminum 6061 had the second highest of around 200 MPa, and CF composite ranged from 50-150 MPa.

High fracture toughness is another important characteristic for the leg material due to the suspension system requiring shock absorption ability. High fracture toughness reduces the risk of cracks propagating in the material, which is a significant risk due to frequent shock loads encountered during travel by the suspension system. Titanium had the highest fracture toughness of 100 MPa·m½. Aluminum 6061 had the second highest of 29 MPa·m½, and CF composite had the lowest possible fracture toughness, ranging from 5 to 50 MPa·m½.

Adequate thermal performance is another important factor as the rover must withstand extreme thermal cycling in the Martian environment. A material with a low coefficient of thermal expansion was favored since thermal cycling would have less of an impact. CF composite had the lowest thermal expansion coefficient of 1-5 μm/m·°C. Titanium had the second lowest of 8.6 μm/m·°C, and aluminum 6061 had the highest of 23.5 μm/m·°C.

Based on the scoring, titanium ranked as the highest overall leg material choice. This aligns with previous rover leg materials, and Perseverance implemented titanium tubing as its leg material [32].

Mechanical Subsystem Trade Study for Chassis Material

Figure 1.5.2.2.2: Chassis Material Trade Study

The chassis has similar expectations as the legs of the rover, and it must support the weight of the rover during travel and handle shock loads experienced throughout travel.Four aspects were considered to determine the optimal material for the chassis: density, load-bearing ability, shock absorption, and thermal performance. Four materials were considered for the legs - aluminum 6061, carbon fiber composite, and stainless steel - based on common aerospace materials and previously used rover chassis materials.

The chassis' material is favored to have a low density, and the density criterion was weighed the heaviest in order to adhere to mass requirements as the chassis is the largest component of the mechanical subsystem. The four materials were ranked by their densities, with the lowest densities ranked the highest. Aluminum 6061 and CF composite had satisfactorily low densities of 2.7 g/cm^3 and 1.5-2 g/cm^3 respectively, both ranking as a 1. Stainless steel's density of 7.5-8 g/cm^3 was far too high considering the mass budget and ranked a 5.

High shear strength is an important criterion to ensure the chassis could reliably tolerate all encountered loads, particularly static and shock loads. Having a high shear strength avoids risk of material fatigue throughout the mission lifetime and ensures the chassis can comfortably tolerate all loads. Stainless steel had the highest shear strength of 500+ MPa, and aluminum 6061 had the second highest of approximately 200 MPa. CF composite ranges due to its anisotropic properties, and its shear strength can be anywhere from 50-150 MPa, putting it in last place.

High fracture toughness is another important characteristic for the leg material due to the suspension system requiring shock absorption ability. High fracture toughness lessens the risk of cracks propagating in the material, which is a significant risk for the chassis as it experiences shock loads and is under constant static loading. Stainless steel and aluminum 6061 each had a fracture toughness above 25 MPa·m½ which ranked them both a 1. CF composite ranked a 4 with a fracture toughness ranging from 5 to 50 MPa·m½

Adequate thermal performance is another important factor as the rover must withstand extreme thermal cycling in the Martian environment. A material with a low coefficient of thermal expansion was favored since thermal cycling would have less of an impact. CF composite had the lowest thermal expansion coefficient of 1-5 µm/m·°C. Aluminum 6061 and stainless steel had the highest of 23.5 µm/m·°C and 17.3 respectively, putting them at a 5 and 4. Based on the scoring, aluminum 6061 was ranked as the highest overall chassis material choice.

Mechanical Subsystem Trade Study for Wheel Material

Figure 1.5.2.2.3: Wheel Material Trade Study

The wheel subsystem design emphasizes the need for a lightweight and durable material that ensures mobility and reliability throughout its function. To determine the most suitable material, 3 candidate materials were selected: aluminum, titanium, and fiberglass. The selection of these materials was based on 4 criteria for critical performance: density, load-bearing, shock absorption, and thermal performance. Each criterion was weighted equally at 25% each, reflecting the team's equal prioritization of mass, strength, durability, and performance.

Density is the criterion that impacts the overall system mass, affecting the maneuverability and energy efficiency of the rover. Load-bearing indicates the maximum pressure (MPa) that the wheels can withstand in varied terrain stresses before becoming permanently deformed and non-functional. Shock absorption measures the rover's ability to endure impacts from terrain obstacles. Finally, thermal performance measures the integrity of the wheels based on their material in varying temperatures.

For the criterion of density, Aluminum and fiberglass scored a 1, given that their densities are 2.7 g/cm^3 and 2.4~2.76 g/cm^3, respectively. Titanium scored a 4 due to its high density of 4.51 g/cm^3. In the load-bearing category, aluminum and titanium received the highest scores with shear strengths of 207 and 550, respectively. Fiberglass received the lowest score with a shear strength of roughly 30 MPa. When measuring the shock absorption of the materials, titanium came in first with a fracture toughness of 100 MPa-m½. This was followed by aluminum with a fracture toughness of 29 MPa-m½ and fiberglass (0.6~2 MPa-m½). Finally, in the thermal performance category, fiberglass and titanium received the highest scores with coefficients of thermal expansion of 5~10 μm/m-°C and 8.6 μm/m-°C, respectively. Aluminum slightly fell behind with a higher coefficient of thermal expansion of 23.5 μm/m-°C.

Aluminum consistently received top scores in most categories, scoring a 1 in the density and load-bearing criteria. The material had a slightly weaker performance in shock absorption and thermal performance, scoring a 2. Overall, aluminum demonstrated structural resilience and a low density (therefore a lower mass under constant volume), making it the highest-ranked option with a total score of 87.5%. Titanium offered strong load-bearing, shock absorption, and thermal performance, receiving top scores in all of the criteria. However, the high density of titanium resulted in a criteria score of 4. This averaged to a final composite score of 81.25%, making it the second-best candidate. Although fiberglass is a very light material with a low coefficient of thermal expansion, it severely underperformed in both load-bearing and shock absorption, with a score of 5 and 4, respectively. These structural characteristics resulted in the lowest composite score of 56.25%.

[Mechanical Subsystem Trade Study for Wheel Tread]

Figure 1.5.2.2.4: Wheel Tread Trade Study

The wheel tread design affects the rover’s ability to handle varying terrain conditions, resist wear over time, and remain operationally reliable through repeated usage. The criteria selected for the wheel tread were wear resistance, grip, and reliability, weighted at 30%, 30%, and 40% respectively. Wear represents how well the tread is able to resist abrasion over time. Grip measures how effectively the tread is able to maintain traction on different surfaces. Finally, reliability will measure the fatigue life of the tread pattern.

Because quantitative data (coefficient of kinetic friction or specific fatigue cycles) was not consistently available across all tread types, this trade study utilized a relative performance scoring system based on industry-standard design knowledge. Symmetrical treads are uniquely beneficial in that their symmetric patterns allow for even wear throughout the wheel, improving tread life. Directional treads typically wear faster, making it fall behind symmetrical and asymmetrical treads in wear performance [22]. Asymmetrical treads take advantage of engraving patterns that allow for stronger cornering grips [22]. Similarly, directional treads take advantage of lateral grooves that allow for the wheel to maintain contact with wet surfaces and resist hydroplaning at high speeds. After applying the weighted scores, symmetrical treads emerged as the optimal choice with a final cumulative score of 77.5%, followed by asymmetrical treads (60%) and directional treads (30%).

1.5.3 Power Subsystem

1.5.3.1 Power Subsystem Requirements

The following requirements were created to ensure that the rover's power subsystem shall operate in its full capacity for the entire duration of the mission. These requirements are grounded in the rover and its external experiment’s energy needs while traversing Martian terrain, operating science equipment, and communicating to and from Earth. Each requirement aligns with expected environmental extremes, mobility objectives, and mission safety margins.

The generation of 200 Wh per sol (PWR.01) ensures the rover can maintain baseline functionality and guarantees survivability during periods of low insolation or dust storms. Deployment and survivability constraints (PWR.02, PWR.04) were instituted to ensure the safety of the rover during landing procedures and regular operation. The operating temperature range (PWR.03) supports electrical and thermal compatibility across systems. Risk mitigation systems (PWR.05) safeguard against failure and simplify system-level design.

1.5.3.2 Power Subsystem Overview

The power subsystem was designed with a strong emphasis on power efficiency, system reliability, and modular integration. Each subassembly has been engineered to address Mars-related challenges such as extreme temperatures, accumulation of dust due to weather activity, and a restricted mass budget.

The Power Generation for the Mission is achieved by utilizing two high-efficiency 4G32C solar panels manufactured by AZUR Space. Together, the panels have a surface area of 1.2 square meters, and are equipped with a Maximum Power Point Tracking controller (MMPT). To mitigate risks from partial shading and potential hot spot formation, the system will integrate external bypass diode protection. Bypass diodes will be placed across solar cell strings (every 5–8 cells) to maintain power output integrity and protect against failure propagation in case of local damage or dust accumulation. This controller allows the panels to adjust dynamically in order to optimize electrical system output during weather events and varying solar conditions. The deployment mechanism of the panels is designed for self-deployment upon Martian touchdown. The mechanism is built to withstand wind gusts up to 20m/s. This subassembly, and Roll-Out Solar Array (ROSA) received a TRL of 5 for this application.

To address dust accumulation on the equipment, which can lead to reduced power output over time, evaluation of three dust mitigation strategies occurred via a trade study: Electrodynamic Dust Shield (EDS), vibration-based cleaning, and passive anti-reflective coatings. The EDS method scored the highest in overall effectiveness and integration potential. It achieved high marks in dust removal efficiency, (Grade 1), low power consumption (Grade 1), high reliability due to lack of moving parts (Grade 1), and low integration mass and complexity (Grade 1), for a total weighted score of 100%. In contrast, passive coatings and vibration methods scored significantly lower, with final weighted scores of 50% and 82.5% respectively. Based on these results, EDS will be integrated into the solar panel design in order to preserve the functionality and lifespan of the solar arrays.

Power storage is provided by an EaglePicher SAR-10211-based lithium ion battery pack that is rated at 500Wh and a lifetime of 2000 cycles. The batteries will be configured in a 4S6P arrangement that delivers high efficiency. Each cell will be individually monitored by a Battery Monitoring System with overcharge, over-discharge, and thermal runaway protections in place. This battery pack supports a wide range of power delivery, with peak outputs of up to 120 W for 30-minute bursts, and continuous draw of up to 20 W [1]. To withstand the extreme temperature swings on Mars (–80 °C to +20 °C), the battery is enclosed within a thermally insulated sleeve incorporating Phase Change Material (PCM) that passively regulates temperature around 0 °C–30 °C [1]. Active resistive heaters embedded within the enclosure provide supplemental heating when PCM capacity is exceeded. Thermal vacuum chamber tests with EaglePicher SAR-10211 cells have demonstrated effective temperature stability from –40 °C to +60 °C, but with the PCM-integrated housing still under environmental validation, this subassembly is currently assigned TRL 5, while the bare cell chemistry stands at TRL 7 due to extensive flight heritage under NASA’s Orion Multi-Purpose Crew Vehicle.

In determining the most effective thermal regulation method for battery and subsystem protection, a thermal trade study was conducted. Three methods were evaluated: resistive heaters with MLI insulation, PCM thermal storage compartments, and a pipe grid with PCM thermal buffering. Each was graded across performance, mass, energy efficiency, and reliability. The pipe grid + PCM system scored highest (86.25%), with optimal energy efficiency and reliability. It offered passive thermal buffering and minimal risk from moving parts. PCM compartmental systems followed closely at 77.5%, while simple resistive heaters and MLI lagged (76.25%) due to higher energy usage and less precise thermal control. Based on this study, the hybrid PCM + heater configuration remains the baseline with future integration of pipe-grid enhancement under evaluation.

Power Distribution is implemented through a hybrid redundant bus architecture, which combines central routing with distributed load switching. The system incorporates current-limiting smart FETs, microcontroller-based load prioritization logic, and real-time health telemetry. Each power rail includes EMI filtering, overcurrent protection, and modular plug-in interfaces for subsystem compatibility. The distribution unit is based on a modified commercial-off-the-shelf (COTS) avionics design and has reached TRL 6.

The power subsystem incorporates space-qualified DC-DC converters with galvanic isolation to supply regulated, stable power to the externally mounted scientific experiment. These converters are critical for providing the required 28 V DC output at up to 45 W peak load, ensuring power quality and electrical isolation for sensitive payload electronics. DC-DC converters operate by receiving unregulated or semi-regulated DC voltage from the rover’s main power bus and converting it to a precisely regulated output voltage. Internally, these converters use high-frequency switching and transformer isolation to achieve both voltage transformation and galvanic isolation, which prevents ground loops and protects sensitive payloads from electrical noise, surges, and transients originating elsewhere in the system. The selected converters feature output voltage ripple below 2%, isolation withstand voltages greater than 1 kV, and efficiency typically above 90% [64]. Their compact form factor and robust EMI filtering make them ideal for integration in the rover’s constrained external payload bay, where space, mass, and environmental shielding are at a premium.

Integration-wise, the DC-DC converters are mounted within the payload’s thermally regulated housing, minimizing exposure to temperature extremes and mechanical vibration. The converter is a space-qualified, commercially available unit with extensive use in planetary and orbital missions. However, as it has not been flown in this exact configuration or integrated with this specific payload, it is assessed at TRL 7 (system prototype demonstrated in a space environment). This TRL reflects both the maturity of the underlying technology and the integration status within our subsystem.

As the battery thermal enclosure remains the least mature component, the entire power subsystem is currently assessed at TRL 5. Additional environmental testing is required to advance the system to full flight readiness.

Subassembly	TRL	Justification
Solar Panel + MPPT	6	ROSA-derived design validated in terrestrial analog and orbital environments
$LiFePO_4$ Battery with PCM	5	COTS cells qualified; integrated PCM in development
Power Distribution Unit	6	Based on COTS hardware with flight heritage
Overall Power Subsystem	**5**	Limited by battery thermal integration

Figure 1.5.3.2.1 Power Subassembly TRL Table

Subassembly	Mass (kg)	Volume (cm^3)	Max Power Draw (W)
Solar Array	3.5	11,000	N/A (Power generation only)
Battery Pack	4	9,000	120
Power Distribution	1.5	4,000	5
Cabling & Interfaces	1	3,000	<1
Total	**10**	**27,000**	**~120 W (peak)**

Figure 1.5.3.2.2 Power Subassembly Mass, Volume, and Power Estimate

1.5.3.3 Power Subsystem Trade Studies

Power Subsystem Trade Study for Dust Mitigation Technologies

Figure 1.5.3.3.1 Dust Mitigation Technologies Trade Study

Dust accumulation on solar panels is one of the most significant challenges to sustained power generation for Mars surface missions. Since the Pathfinder landing in 1997, Mars rovers such as Spirit and Curiosity have experienced a consistent daily average power loss of approximately 0.2% due to dust settling on their solar panels in the absence of active cleaning measures. This gradual decline, observed across multiple missions, highlights the need for effective dust mitigation strategies to maintain energy availability and extend rover operational lifetimes [20].

To address this, a trade study has been created to compare three leading dust mitigation technologies: Electrodynamic Dust Shield (EDS), Vibration-Based Cleaning (Chladni Patterns), and Passive Coatings (Hydrophobic/Anti-Static). The evaluation focused on four criteria: effectiveness, reliability/robustness, power consumption, and mass & integration. These criteria were weighted at 35%, 25%, 20%, and 20% respectively, reflecting the mission-critical importance of maintaining high solar array output and system reliability, while also considering the constraints of limited power and mass on the rover.

The Electrodynamic Dust Shield (EDS) technology, which uses transparent electrodes to generate electric fields that actively repel and remove dust, emerged as the top performer. EDS has demonstrated over 90% Dust Removal Efficiency (DRE) in both laboratory and Mars-analog field tests, with minimal power consumption (less than 1% of array output) and no moving parts, making it highly reliable and straightforward to integrate [7].

In contrast, Vibration-Based Cleaning employs actuators to create resonance patterns that dislodge dust from the panel surface. While this method can achieve moderate dust removal (typically 50–90%), it introduces additional moving parts, increasing the risk of mechanical failure and requiring more power for operation [7].

Passive Coatings, such as hydrophobic or anti-static layers, provide a lightweight and power-free solution by reducing dust adhesion, but they do not actively remove settled dust and their long-term durability under Martian conditions is still under study [75].

Scoring was assigned based on published performance data and the technology readiness of each option. EDS received the highest marks across all criteria, due to its proven effectiveness, robust design, and ease of integration. Vibration cleaning was penalized for its complexity and reliability concerns, while passive coatings, though excellent in reliability and integration, scored lower for effectiveness. As a result, EDS is recommended as the primary dust mitigation approach for the rover's solar arrays, with passive coatings considered as a potential supplementary measure. This recommendation is strongly supported by current research and NASA technology demonstrations, ensuring the greatest likelihood of sustained power generation throughout the mission.

Power Subsystem Trade Study for Solar Panel Deployment

Figure 1.5.3.3.2 Solar Panel Deployment Trade Study

The criteria taken into account was mass, stowed volume, reliability, and complexity with weights of 25%, 30%, 30%, and 15% respectively, reflecting the critical importance of minimizing launch mass and stowed volume, while ensuring reliable deployment and manageable integration risk.

Quantitatively, ROSA demonstrated the lowest mass and stowed volume among the options, scoring a 1 (optimal) in both categories. For example, ROSA's mass per unit area is typically less than 20 kg for a 10 kW array [51], compared to 30–35 kg for equivalent mechanical hinge systems, as demonstrated in recent NASA ISS deployments. Ultraflex arrays, as used on the InSight lander, offer a moderate compromise, with a typical mass of 6–10 kg per wing and a stowed volume of approximately 0.05-0.1 m^3 per array for Mars-class missions [35].

Reliability was assessed based on flight heritage and deployment success rates. Mechanical hinges, with extensive use on Spirit, Opportunity, and Curiosity, scored a 1 for reliability due to their proven track record in Martian conditions. ROSA, while newer, has demonstrated successful deployment on the International Space Station and is being adopted for upcoming planetary missions, earning a reliability score of 2. Ultraflex, with successful deployments on Mars Phoenix and InSight, scored a 3, reflecting some flight heritage but also highlighting increased complexity and moderate risk [36].

Complexity was evaluated by the number of moving parts and deployment steps. ROSA and Mechanical Hinge mechanisms both scored a 2, while Ultraflex scored a 3 due to its tensioned structure and additional deployment steps.

When total weighted scores were calculated, ROSA achieved the highest overall score (88.75%), followed by Ultraflex (63.75%) and Mechanical Hinge (47.5%). These results indicate that ROSA offers the best balance of mass, volume, reliability, and manageable complexity for Mars surface operations.

Power Subsystem Trade Study for V Cell Type

Figure 1.5.3.3.3 Photovoltaic Cell Type Trade Study

Selecting the optimal photovoltaic (PV) cell type is a critical aspect of the Mars rover power subsystem, as it directly influences power generation efficiency, mass, and long-term reliability under harsh planetary conditions. For this trade study, three leading space-qualified PV technologies were compared: Spectrolab XTE-SF (triple-junction), AZUR SPACE 3G30C (triple-junction), and AZUR SPACE 4G32C (quad-junction). The criteria and their weights were chosen to reflect the most significant factors affecting Mars surface missions: Beginning-of-Life (BOL) efficiency (15%), End-of-Life (EOL) efficiency (35%), radiation resistance (20%), and specific power (30%). EOL efficiency was given the highest weight because the ability of the solar array to maintain performance over the mission's duration is crucial for sustained science operations, especially given the harsh Martian environment and exposure to radiation and dust. Specific power was also heavily weighted, as minimizing array mass is essential for rover mobility and payload accommodation. Radiation resistance is vital due to the thin Martian atmosphere, which provides limited shielding from cosmic rays and solar energetic particles, while BOL efficiency remains important for maximizing initial energy conversion.

Scoring was based on manufacturer datasheets, published test results, and recent reviews of PV performance in space environments. The AZUR SPACE 4G32C quad-junction cell outperformed the other options, achieving a BOL efficiency of 32.5%, EOL efficiency above 28%, radiation-induced power loss below 5%, and a specific power exceeding 370 W/kg [9]. In comparison, both the Spectrolab XTE-SF and AZUR SPACE 3G30C triple-junction cells demonstrated lower EOL efficiencies (26–27%), higher radiation degradation (5–10% power loss), and specific powers in the 300–350 W/kg range [10][11].

Based on the weighted scoring, the AZUR SPACE 4G32C quad-junction cell has been chosen as the rover's solar array. Its combination of high efficiency, excellent radiation resistance, and superior specific power offers the best balance of performance and risk mitigation for Mars surface operations, supporting sustained science activities throughout the mission.

Power Subsystem Trade Study for Storage and Battery System

Figure 1.5.3.3.4 Power and Battery Trade Study

Lithium-ion batteries are a common energy storage solution in spacecraft and rovers due to their high energy density and above-average cycle life. Not all lithium ion batteries are equally suited for operation on the Martian surface or extended mission duration. Possible battery configurations were examined: commercial Li-ion packs, custom $LiFePO_4$ assemblies, and aerospace-certified lithium-ion battery systems.

The evaluation criteria included thermal stability (25%), cycle life (25%), flight heritage (30%), and integration complexity (20%). These weights reflect the critical need for reliability under Martian extremes, the need for long-term functionality over several hundred Martian sols, and the reduction of integration risk for flight-ready systems.

The EaglePicher SAR-10211 aerospace-certified lithium-ion battery was ultimately selected. This battery has strong heritage in space applications, including NASA's Orion MPCV and the Dream Chaser spacecraft. Its performance in high-reliability, high-radiation environments makes it particularly well-suited to a Mars surface mission. It scored highest in flight heritage (Grade 1) and thermal performance (Grade 1), while offering acceptable integration complexity and volume. While slightly heavier than commercial options, its safety profile and robust telemetry support provided a compelling margin of confidence for mission longevity. The final configuration includes PCM-based thermal housing, which further enhances survivability during cold Martian nights. This subassembly, although composed of TRL 7 components, is rated at TRL 5 overall due to ongoing validation of the integrated thermal housing.

Power System Trade Study for Isolated Power System

Figure 1.5.3.3.5 Isolated Power System Trade Study

The integration of an externally mounted scientific payload on the Mars rover requires an isolated power system to ensure reliable operation of sensitive instruments under strict environmental and operational constraints. The payload, with a mass of 15 kg and dimensions of 0.5 m x 0.5 m x 0.5 m, demands a stable, regulated 28 V DC supply capable of delivering 45 W peak during sampling and 5 W in standby mode. Due to its sensitivity to voltage fluctuations, electromagnetic interference (EMI), and thermal gradients, the power architecture must provide galvanic isolation, superior voltage regulation, and minimal EMI susceptibility, while integrating efficiently with the rover's limited mass and volume budgets.

Three technical options were evaluated: an isolated DC-DC converter, transformer-based AC-DC isolation, and a dedicated battery pack with isolated charger [12]. The trade study was evaluated using six criteria: voltage regulation, isolation quality, EMI susceptibility, integration complexity, reliability, and power efficiency. They were weighted to reflect mission-critical priorities: ensuring stable power for sensitive electronics, minimizing integration risk, and supporting long-duration operation in the harsh Martian environment. Voltage regulation and isolation quality were given the highest weights (25% each), recognizing the payload's susceptibility to noise and the need to prevent ground loops or leakage currents that could compromise scientific data integrity. EMI susceptibility (20%) was prioritized due to the proximity of high-power actuators and antennas, while integration complexity (15%), reliability (10%), and power efficiency (5%) rounded out the assessment, consistent with best practices in planetary payload design.

The isolated DC-DC converter emerged as the top choice, offering excellent voltage regulation (less than 2% ripple), full galvanic isolation (>1 kV withstand), and proven EMI filtering in a compact, space-qualified package [12]. This approach is widely adopted in NASA and ESA science payloads, providing high reliability and efficiency (>90%) with minimal added mass or integration burden. Best practice is to contact VPT or a similar supplier for a space-qualified, isolated DC-DC converter with a 28 V output [8]. Vendors such as VPT and Vicor routinely provide custom output voltages for space missions, ensuring optimal integration and performance for the payload. If a custom 28 V output module is not available, the next best approach is to use a standard model such as the SVRFL2815S (15 V, 100 W) in combination with a space-qualified, isolated boost converter to step up to 28 V, or to use two compatible modules in series if approved by the manufacturer [8]. While this introduces additional complexity and some efficiency loss, it remains a feasible backup if procurement or schedule constraints preclude a custom module.

Transformer-based AC-DC isolation, while capable of delivering high isolation quality, scored lower due to increased mass, volume, and integration complexity, as well as moderate efficiency and heritage for low/medium power applications. The dedicated battery pack option, though providing full isolation and ride-through capability, was penalized for added mass, thermal management complexity, and the need for periodic recharging and monitoring.

Given the external mounting requirement, the selected power system must also withstand temperature swings from -20°C to +40°C in operation (and -40°C to +60°C survival), with rapid changes mitigated by suitable thermal shielding or active heating [27]. The DC-DC converter solution is compatible with these needs, as it can be housed within the payload's thermally regulated enclosure and placed at a safe distance from vibration and EMI sources, as required by the integration constraints [7]. This ensures the integrity of precision atmospheric sensors and the mass spectrometer, both of which are highly sensitive to electrical and mechanical disturbances.

The trade study supports the implementation of a space-qualified isolated DC-DC converter as the optimal solution for powering the external experiment payload. This architecture delivers the required voltage stability, isolation, and EMI protection, while minimizing integration complexity and supporting the constraints of mass, volume, and environmental requirements of Mars surface operations.

1.5.4 Command and Data Handling (CDH) Subsystem

1.5.4.1 CDH Subsystem Requirements

The CDH (Command & Data Handling) Subsystem consists of the Onboard Computer (OBC) subassembly, Data Storage subassembly, Data Interfaces subassembly, and Telecommunications subassembly. In order to meet all of the mission requirements, the subsystem must be able to navigate and analyze its surroundings and goals for mission success.

The Command and Data Handling (CDH) Subsystem enables spacecraft coordination, autonomy, and communication. It integrates four key subassemblies Onboard Computer (OBC), Data Storage, Data Interfaces, and Telecommunications each of which supports critical mission functions. The OBC executes flight software and manages autonomy routines. Its performance, memory, and radiation tolerance are vital for maintaining control and system integrity in the harsh Martian environment. Data Storage ensures scientific and telemetry data are logged reliably and accessibly, necessitating sufficient capacity, fast write speeds, and resilience to environmental degradation. Data Interfaces serve as the internal communication backbone, linking the OBC to instruments, actuators, and sensors with reliable timing and low signal noise key for coordinated subsystem execution and fault response [5, 6]. Telecommunications handles the uplink and downlink of mission data between the rover and Earth via orbital relays. Efficient encoding, low power draw, and fault-tolerant transmission are essential to maintaining operational awareness and ensuring scientific return. Each subassembly is interdependent, and failure in any one of them risks total mission loss.

In order for the CDH Subsystem to meet all of its requirements, constant verification and testing of subcomponents is needed for maximum performance. Methods would include demonstrations of the technological capabilities in the CDH Subsystem in a safe space, using effective programming for debugging and early software testing.

1.5.4.2 CDH Subsystem Overview

The Command and Data Handling (CDH) Subsystem serves as the brains of the rover, managing data processing, storage, communication, and coordination across all subsystems. It includes four key subassemblies: the Onboard Computer (OBC), Data Storage, Data Interfaces, and Telecommunications.

The selected Onboard Computer is the RAD5545, a flight-proven, radiation-hardened processor with a Technology Readiness Level (TRL) of 9 [2, 4]. Its multi-core 1 GHz-class architecture and >2 GB of memory provide ample computing power for autonomy routines, sensor fusion, and real-time control tasks. The processor’s radiation tolerance and heritage from previous missions ensure reliability in the harsh Martian environment.

For data logging and instrument outputs, the rover employs a Non-Volatile Memory Express (NVMe) Solid State Drive. This device offers >512 GB of storage and high-speed data transfer rates exceeding 2000 MB/s [6]. Its high TRL, coupled with efficient power consumption and tolerance to environmental stress.

Subsystem-level communication is handled through RS-422, a space-proven serial interface with strong noise immunity, deterministic timing, and low integration complexity [5]. It connects all subsystems to the OBC, enabling command routing and telemetry gathering in real time.

For external communications, an Ultra High Frequency (UHF) radio was selected. This device operates at TRL 9 and offers low power consumption, omnidirectional coverage, and a long flight heritage on missions such as Curiosity and Perseverance [14]. It provides a communication relay between the rover and the Mars Reconnaissance Orbiter (MRO), which then transmits data to Earth via the Deep Space Network (DSN). The UHF system is particularly well-suited for the limited bandwidth and latency-tolerant nature of planetary missions.

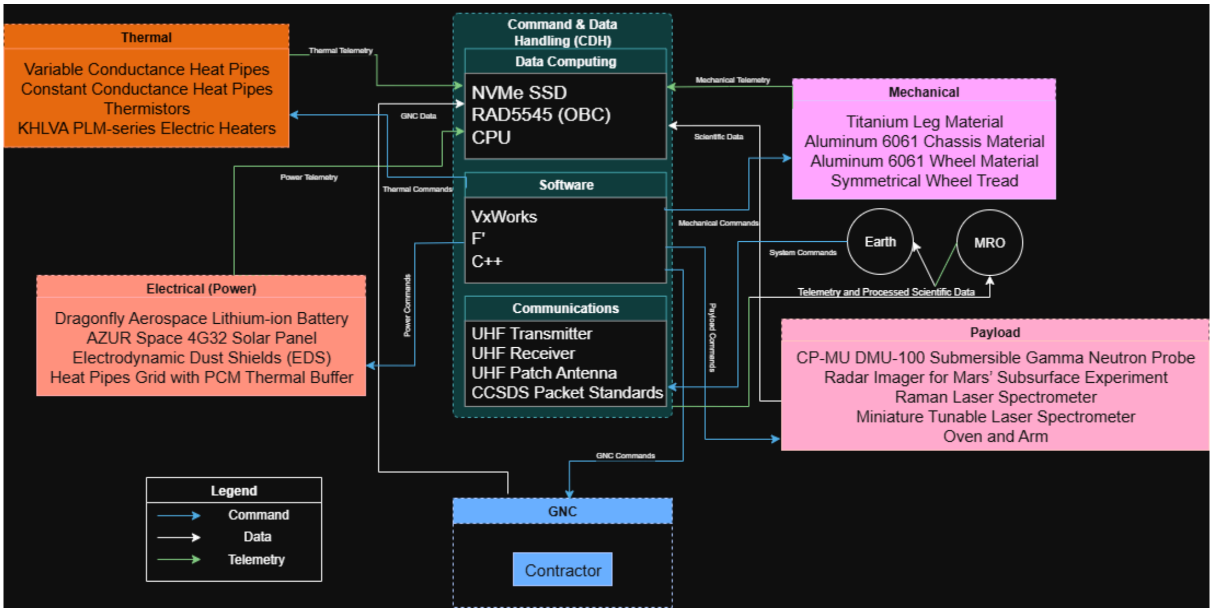


Figure 1.5.4.2.1: Software Architecture Flowchart

The CDH subsystem is structured around three functional blocks: Data Computing, Software Execution, and Communications. These blocks together manage the collection, processing, prioritization, and transmission of data. A software architecture diagram illustrates how the CDH interfaces with other rover systems, using RS-422 for physical connections and standardized software protocols for message handling and execution.The Electrical Power System (EPS), which includes lithium-ion batteries and deployable solar panels with electrodynamic dust shields, is tightly integrated with the CDH system. The CDH receives real-time telemetry from the EPS battery voltage, current draw, charge status, solar panel efficiency, and internal temperature and compares this data against expected operating profiles. When deviations are detected, CDH commands can modify EPS behavior: enabling/disabling charging, deploying solar arrays, or redistributing power across subsystems. Power modes such as science mode, sleep mode, and emergency mode are also managed via CDH logic based on power and mission conditions.

Thermal regulation is another key interface. The thermal control system includes passive elements such as multi-layer insulation (MLI) and active elements like electric heaters and variable conductance heat pipes [2, 27]. Thermistors throughout the rover provide temperature data to the CDH, which responds by activating or deactivating heaters or adjusting heat pipe flow [27]. For colder Martian nights, the CDH can initiate power-conserving overnight heating cycles and shut down non-essential systems to preserve critical functionality. All thermal telemetry is logged and used to refine thermal response strategies and plan future power availability.

Although the GNC subsystem is provided by an external contractor, the CDH subsystem is responsible for processing the GNC data onboard. The OBC receives measurements from the IMU, including accelerometer and gyroscope data, and fuses them with wheel odometry and actuator feedback through an Extended Kalman Filter [16]. This provides an accurate real-time state estimate of the rover's position, velocity, and orientation, which is essential for localization and path planning. CDH software then uses this information to calculate error between the current and target state, sending corrective commands to the mobility subsystem to close the loop [16].

The CDH software is written in C++ due to its performance efficiency and suitability for real-time embedded systems. Code development shall be done through Visual Studio Code inside Docker containers for embedded portability, and managed using Git version control [17]. The modular software architecture separates drivers, control routines, and communication layers to reduce risk and enhance reliability. Safety-critical logic is incorporated to maintain spacecraft operability even under adverse conditions.

Commands from Earth are prepared by mission control and sent during scheduled DSN communication windows. These messages are relayed via the MRO to the rover's UHF radio. Following CCSDS standards, these commands include instructions to activate instruments, update configurations, modify science plans, or adjust mobility parameters [15]. The CDH system checks message integrity and routes commands to the appropriate subsystems. Priority execution ensures that time-sensitive commands (e.g., fault recovery or power reallocation) are handled immediately, while others are queued and executed as bandwidth and system state allow [15].

All data generated on the rover telemetry, health logs, science measurements is collected and processed by the CDH subsystem before transmission. The data is compressed, timestamped, compressed into packets, and assigned priority. Critical telemetry such as power levels, internal faults, and system temperatures are prioritized over less urgent science data. These packets are transmitted via UHF to MRO during scheduled overpass windows and subsequently sent to Earth. The CDH system ensures redundant transmission of urgent data and manages lower-priority data buffering for delayed downlink opportunities.

1.5.4.3 CDH Subsystem Trade Studies

CDH Subsystem Trade Study for Onboard Computer

Figure 1.5.4.3.1: Onboard Computer Trade Study

The OBC serves as the core of the CDH subsystem, managing software execution, inter-subsystem coordination, sensor fusion, and autonomous tasking. It must reliably support navigation, data management, and communication protocols in a high-radiation, energy-limited environment. This trade study evaluates the RAD750, RAD5545, Cobham GR712RC, and Pumpkin OBC (CubeSat Kit) across five weighted criteria: Processing Performance (30%), Radiation Tolerance (20%), Memory Capacity (15%), Mass & Power Efficiency (25%), and Flight Heritage/Technology Readiness Level (10%).

Processing Performance (30%) received the highest weight due to the computational demands of autonomy, real-time sensor fusion, and data handling for science payloads. The OBC must handle task parallelism efficiently to avoid command delays or data loss. Radiation Tolerance (20%) is an essential safeguard against single-event effects (SEEs) and total ionizing dose (TID) risks are significant over long mission durations. Memory Capacity (15%) supports instrument buffering, log storage, and data pre-processing. Mass & Power Efficiency (25%) is a combined factor representing the importance of every watt and gram within the strict system constraints. Flight Heritage/TRL (10%) offers assurance that the system has performed reliably in similar missions, reducing integration risk and shortening testing timelines.

Figure A.1: RAD5545 SpaceVPX single-board computer

RAD5545 features a multi-core 1GHz-class processor, over 2GB of memory, and is radiation-hardened beyond 100 krad, making it ideal for executing autonomous algorithms, instrument control, and onboard data management [4]. Its moderately high TRL (8–9) reflects successful use in a variety of missions.RAD750 remains widely used and well-proven with high radiation tolerance and strong flight heritage, but its lower processing performance (200–500 MHz) and less memory make it less favorable for autonomous operations. Cobham GR712RC is based on a dual-core LEON3 processor and performs adequately in power and radiation terms, but its limited memory and weaker processing speed place it behind for high-autonomy scenarios [18]. Pumpkin OBC, while lightweight and efficient, lacks radiation protection and processing capability. It is more suited to CubeSat-class missions in low Earth orbit rather than long-term missions [19].

The RAD5545 was selected as the OBC. Its computational power, memory capacity, and robust radiation protection support high-throughput science, autonomy, and real-time system management [4]. The system’s growing flight record further supports its viability for deep space missions making it a great fit for the rover.

CDH Subsystem Trade Study for Data Storage

Figure 1.5.4.3.3: Data Storage Trade Study

The mission is expected to gather and transmit data at different times, and is therefore required to store the collected data efficiently to support scientific operations involving high-resolution imaging and complex sensor data. Selection of proper data storage is therefore significantly relevant to the performance of the CDH subsystem as a means to log and transmit data for analysis. The trade study evaluates Micro Secure Digital (SD) cards, Embedded MultiMediaCards (eMMCs), and Non-Volatile Memory Express Solid State Drives (NVMe SSDs) as potential storage solutions using four weighted criteria: Data Storage Capacity (35%), Data Transfer Speed (25%), Radiation Tolerance (20%), and Power Consumption (20%).

Storage Capacity (35%) was given the highest weight due to the intent to collect and store large volumes of multispectral and subsurface data from the instruments. Transfer Speed (25%) was prioritized next as several instruments will produce large datasets that must be quickly written to prevent scientific data loss. Radiation Tolerance (20%) is essential for safeguarding mission-critical data against noise and stray interference, primarily utilizing error-correcting codes (ECCs). Power Consumption (20%) directly affects system longevity and thermal load; however, since storage systems are not always in use and thermal load can be accounted for by the TCS subsystem, this criterion can be allocated less weight.

Micro SD cards score the lowest on capacity and radiation tolerance, which are mission-critical for long-term reliability and scientific throughput. Their largest strength is extremely low power consumption. eMMC offers a better balance with moderate capacity, acceptable speeds for sensor data capture, and manageable power usage. Though not radiation-hardened, its industrial design gives it an edge over consumer SD cards. NVMe SSDs provide excellent capacity and speed, which is vital for fast camera data dumps and buffering. However, they consume significantly more power and are heavier and bulkier, which is not ideal under mission constraints.

The NVMe SSD was selected as the primary data storage option for its performance in capacity and speed, which outweighs the tradeoff in power draw and moderate radiation robustness. The Phison 8TB M.2 PCIe Gen4 NVMe SSD passed NASA TRL 6 certification in December of 2022, and would be ideal for data storage.

CDH Subsystem Trade Study for Data Interfaces

Figure 1.5.4.3.4: Data Interfaces Trade Study

The CDH subsystem must support robust data communication between onboard sensors, actuators, and the Onboard Computer (OBC). The selected data interface must facilitate consistent throughput, low error rates, and compatibility with spaceflight hardware. This trade study evaluates RS-422, MIL-STD-1553, I2C, and Ethernet as candidate data interface protocols for P.H.O.E.N.I.X, using five weighted criteria: Data Rate (30%), Noise Tolerance (25%), Physical Complexity (20%), Time Accuracy

Preservation (15%), and Technology Readiness Level (10%).

Data Rate (30%) received the highest weight due to the need to stream continuous sensor data—including camera feeds and GPR logs—into the CDH system with minimal delay or buffering. Noise Tolerance (25%) is especially critical in the harsh Martian environment, where electromagnetic interference and radiation events may corrupt signals unless handled by fault-tolerant protocols. Physical Complexity (20%) was included to account for ease of integration and the routing burden on the spacecraft harness. Lower wire count and compact topology reduce mass and integration risk. Time Accuracy Preservation (15%) is needed to synchronize timestamps and ensure determinism in time-sensitive science sequences and actuator commands. Technology Readiness Level (10%) provides assurance of reliability based on mission heritage and space qualification.

Figure A.3: RS-422 Pinout Table

RS-422 offers modest data rates (10–50 Mbps), but provides incredible noise tolerance (rad-hard, fault-tolerant implementations exist), low physical complexity , and strong time determinism. Its spaceflight heritage gives it the highest TRL score. MIL-STD-1553 is the most robust and fault-tolerant protocol considered, with strong shielding and redundant architecture. However, it suffers from very low data rates (<1 Mbps) and high physical complexity, making it less favorable for large data real-time science streaming or camera feeds. I2C is extremely lightweight, modular, and simple to implement, making it ideal for short-range internal board-level communication. However, it is poorly suited for spacecraft-scale interfaces due to limited data rate, very poor noise tolerance, and lack of time determinism. Ethernet provides very high data rates (>100 Mbps) and is widely adopted in terrestrial systems. However, its minimal noise tolerance and bulkier shielded cable assemblies present integration and timing challenges, especially in spaceflight environments without specialized protocol layers.

RS-422 was selected as the optimal data interface.Its strengths in noise tolerance, time determinism, and space-proven reliability make it ideally suited for the Martian environment, where command accuracy and telemetry integrity are mission-critical.

CDH Subsystem Trade Study for Telecommunications

Figure 1.5.4.3.6: Telecommunications Trade Study

The telecommunications subsystem is responsible for sending scientific data and health telemetry from the Martian surface to Earth. For P.H.O.E.N.I.X, this requires a communications system that is low in power consumption, robust to environmental constraints, and capable of transmitting significant volumes of data while mitigating risk to transmission interference. This trade study evaluates Software Defined Radios (SDRs), Deep Space Networks (DSNs), Deep Space Optical Networks (DSONs), and Ultra High Frequency (UHF) Radios as candidate telecommunications architectures using the four weighted criteria: Power Consumption (35%), Data Rate (25%), Technology Readiness Level (25%), and Directionality (15%).

Power Consumption (35%) received the highest weight due to the scale of power cost in poorer telecommunications candidates, and the net cost that would have on mission lifespan. Data Rate (25%) was emphasized to ensure timely offloading of science data and prevent onboard data storage bottlenecks. Technology Readiness Level (25%) was prioritized to reduce implementation risk, as there are many heritage systems and newer developments.Directionality (15%) was included to assess risks related to poor alignment and aiming. Lower directionality systems simplify antenna design and improve robustness.

Ultra High Frequency (UHF) Radio offers extremely low power consumption (<5W), omnibearing directionality, and high technology readiness with consistent performance in NASA missions including Phoenix, Curiosity, and Perseverance [32]. While its data rate is relatively limited (1–10 Mbps), the benefits in power and technological reliability indicate favorable implementation. Software Defined Radio (SDR) performed moderately well across all categories. While consuming slightly more power (5–10W), SDRs allow flexible reconfiguration in software and can support a wide range of frequencies and protocols. This adaptability is valuable for redundancy, though it lacks the ultra-low power edge of UHF. Deep Space Networks (DSNs) are well-established and used for Earth return, but its integration onboard involves relay communications rather than direct-to-Earth links from a low-cost rover. Its moderate power draw and data rate (10–50 Mbps) make it viable, but its directional requirements and complexity limit onboard implementation. Deep Space Optical Network (DSON) scored lowest due to extreme power consumption (>40W), low TRL, and high directionality sensitivity.

Ultra High Frequency (UHF) Radio was selected as the optimal telecommunications method due to its superior balance of low power draw, high reliability, and simple, robust integration. Its low directional sensitivity and TRL 9 flight heritage on previous Mars missions make it ideal for surface operations with relay to orbiting assets. The data rate, while lower than other options, is generally sufficient and can be mitigated through integration and relay support from orbiting assets like the MRO.

1.5.5 Thermal Management Subsystem

1.5.5.1 Thermal Management Subsystem Requirements

The thermal subsystem includes MLI coatings, heat pipes, radiators, and optional radioisotope heater units. It must protect critical hardware from Martian hazards such as dust storms and radiation, while maintaining thermal stability throughout the mission.

The thermal components of the subsystem must protect the designated instrumentation and powering system to ensure no failure that will result in the mission being compromised. Such components include MLI coatings, which shall aid in thermally isolating the system to maintain its operational temperature range. The MLI design includes an outer cover layer, which is key to protecting the under layers and rover hardware from the outer environment, radiation, and micrometeoroids. It also consists of levels of reflective layers underneath the outer cover, as well as minimizing thermal conduction from the exterior to the interior. The MLI shall also have an innermost layer called the inner cover, which faces the interior hardware. This layer must generate a minimal amount of particulate contaminants, aid in reducing stress for the interior reflective layers, and not be aluminized to reduce the risk of an electrical short, which is a fire hazard. These layers shall be separated by a form of netting in the design to minimize thermal conductivity within the layers. For the separation of the inner cover and the reflective layers, the design must utilize adhesives, adhesive transfer tapes, and/or metalized tape to aid in the thermal conductivity properties. Emissivity and absorptivity values must be taken into consideration to thermally isolate the system. The design shall also implement a form of ventilation within the MLI with the inclusion of perforations in areas of the MLI to aid in outgassing to minimize pressurization within the MLI.

To ensure the rover system is able to function properly without freezing in the Martian environment in adherence to the system's requirements, an active TCS is necessary to tackle this challenge. One such active TCS is an RHU, the RHU will provide passive heating to the system using an isotopic radioactive material via radioactive decay. To control the heating, heat pipe designs shall be implemented and integrated into the system to equally distribute heat around the system. Radiators shall be used for excessive heat dissipation. These design measures shall help in meeting the system requirements.

In order for the TCS to meet the system requirements, methods of verification and validation must be conducted to ensure each component meets the requirements. These methods include CAD thermal modeling of the subassemblies.After acquiring results, the components shall be tested in a thermal vacuum chamber at one of NASA’s research facilities to validate the thermal modeling analysis on CAD. The components shall then be tested with other subsystems in a mission simulation testing area to ensure all thermal subsystems comply with the mission system requirements.

1.5.5.2 Thermal Management Subsystem Overview

The thermal subsystem comprises of MLI (multilayer insulation) coatings, heat pipes, thermal sensors, and electric heaters. With no thermal controls, the system was expelling abundant amounts of heat, losing 4000W of heat. In order to insulate the rover, double-sided mylar was decided to be the best option, as it is one of the most effective at preventing heat from radiating out, with an exceptionally low emissivity value.The entire rover will be coated with double-sided mylar MLI. The TRL of this MLI is 5 as it has heritage.For the inner layer of the MLI blanket, goldized Kapton has been selected for its high absorptivity and low emissivity. Since the system is so cold, it was decided that those were the traits to prioritize, taking in more warmth from the sun and preventing heat from leaving. The TRL of goldized kapton is 5 for the same reason as the double mylar MLI.

The majority of instrumentation on the rover can operate within the same temperature range of 303 K to 313 K, however the mini-TLS requires a lower temperature for operation. To solve this, the mini-TLS will be placed further from overly hot components such as the oven, and will have heat pipes removing excess heat. This is doubly necessary since the mini-TLS itself is prone to overheating. These heat pipes will remove a controlled amount of heat from the area dedicated to the Mini-TLS to the outside environment. Since the Mini-TLS has a more delicate operational temperature these heat pipes will be separate from the ones around the other instruments. The heat pipes here will be variable conductance heat pipes, so that the amount of heat expelled can be controlled as the outside temperature or heat sink temperature drops. The TRL of variable conductance heat pipes is 5, because it has successfully operated in space environments, but is not legacy technology nor has been on similar missions [2, 3].

The other system of heat pipes will move heat from the oven to the RLS, RIMFAX, gamma neutron probe,and OBC which need to be kept at higher temperatures. The oven itself will produce notably high temperatures while in use and the heat pipes will allow that heat to be dissipated to the rest of the system. The heat pipes used will be constant conductance heat pipes, so that this system is passive. These are easier to integrate and have a higher TRL than variable conductance heat pipes. The TRL of constant conductance heat pipes is 7, because very similar units have successfully operated in space before but not on identical missions.

For the hot end of expected daytime temperatures, ~100W of heating is still needed. In order to solve this, a selection of electric heaters was made to be connected to the computer system along with thermal sensors. A thermal sensor will be dedicated to each instrument, to monitor each instrument is within the operational temperature and to alert the system if they start to fall out of that range. Connected to the thermal sensors, there will be electric heaters to provide the necessary heat to maintain the operational temperatures. The thermal sensors will be a thermistor, and the electric heaters will be KHLVA, PLM-series. With these heaters Qnet would equal zero.

	Mass	Dimensions	Max power
VCHP	1.6kg	0.5 m long, 2 cm radius	70W
CCHP	3.2kg	1 m long, 2 cm radius	0W
Electric Heater	0.4kg	6 in^2	40W

Figure 1.5.5.2.1: Thermal Subassemblies

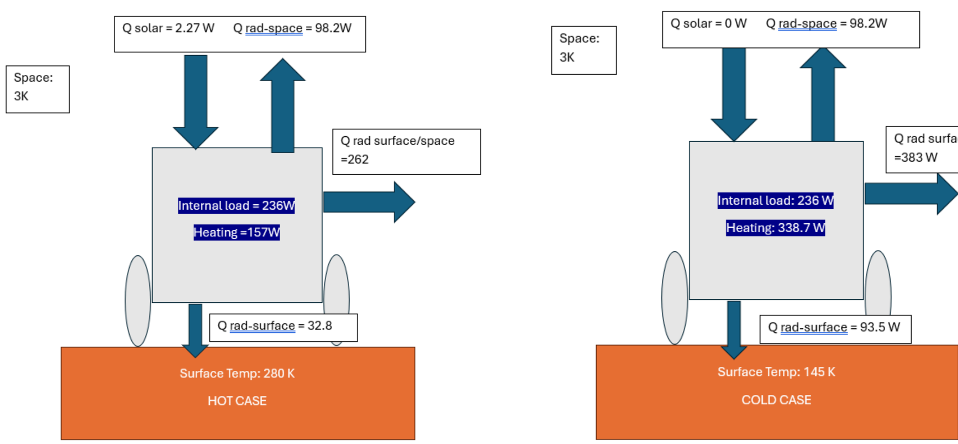


Figure 1.5.5.2.2: Heat Flow Maps

1.5.5.3 Thermal Management Subsystem Trade Studies

Thermal Management Subsystem Trade Study for MLI(Outer Layer)

Figure 1.5.5.3.1: MLI (Outer Layer) Trade Study

To find the ideal outer cover of the multilayer insulation(MLI) system that can help tackle the environmental, cosmic, and solar challenges P.H.O.E.N.I.X. will face on the mission, a trade study was conducted on 4 potential candidates of the system; beta cloth, beta cloth aluminized, double sided mylar, and kapton coated and backed.

These outer cover materials are evaluated on criteria set in stone based on mission requirements. One of the criteria set for the outer cover is the operational temperature range, weighted at 25%, as the material chosen for the outer cover must be able to withstand extreme day/night seasonal temperature swings in the Martian region of Erebus Montes. Another is the evaluation of tensile strength, which is weighted at 15%, as the outermost layer of the MLI it is crucial that the structural integrity of the outer cover must be able to withstand extreme weathering throughout the mission duration. Finally, two extremely vital criteria when it comes to the selection of material are the emissivity and absorptivity values of the chosen material, each weighing 30%. It is vital to keep the system thermally isolated from the environment around it; therefore, when choosing the outer cover, low absorptivity and low emissivity are favored.

The Beta cloth offers the capability of withstanding extreme temperatures ranging from 122.15K to 533.15K [13], which is more than enough to cover the Martian temperature swings. The material's tensile strength increased with the addition of PTFE Teflon coating to combat MMODs(micrometeoroids) and UV radiation. The Beta cloth however, has a high absorptivity, which is not ideal when it comes to an environment like Mars, where daytime temperatures can skyrocket. It also offers high emissivity values, which is not favorable when it comes to making sure the hardware doesn't overheat. Its application in past spacecraft, capsules, and rovers makes it an exemplary candidate. The Aluminized version of the Beta cloth, although having similar properties to its parent the Beta cloth, shares similar emissivity and absorptivity values which are not favored. The Kapton Coated and Backed offers a great temperature range in par with its Beta cloth variants but lacks the tensile strength when it comes to withstanding the weathering from MMODs. It also falters when it comes to ensuring low absorptivity with its relatively moderate range. In an environment where low emissivity is favored, the Kapton only provides a moderate level of emissivity, which will affect the thermal balancing of the entire system. The Double Sided Mylar offers an excellent temperature range of 22K to 394K [14] which is enough to cover the temperature swings of the Martian atmosphere. The material also offers a decent tensile strength which is important when it comes to the selection of the outer layer and withstanding the weathering of MMODs and dust storms. The Double Sided Mylar, unlike its other counterparts being compared in the trade study, has a low absorptivity value and low emissivity value. Meaning that in our calculations it conserved the most heat. Aluminized Beta Cloth for comparison was losing 2290W of heat compared to 154W lost with double sided mylar. Making it the best choice for, being the closest to reach a Qnet of zero.

The Double Sided Mylar was ultimately selected for the outer cover material due to not only its ability to withstand extreme temperature swings on the Martian surface but also its exceptionally low absorptivity value and low emissivity tops off the other candidates when it comes to the deployment in the Martian environment and ensuring the system doesn't freeze or overheat.

[Thermal Management Subsystem Trade Study for MLI(Interior Layer)]

Figure 1.5.5.3.2: MLI (Interior Layer) Trade Study

Throughout the mission duration, the interior layer will be required when it comes to aiding in minimizing conductance from the outside to the inside and vice versa. A trade study was conducted on 3 potential candidates when it comes to combating this challenge. These include: Aluminized Kapton, Aluminized Mylar, and Goldized Kapton. These requirements are evaluated per the mission systems requirements.

These materials are evaluated on criteria set for them: OTR (Operational Temperature Range), TRL, absorptivity, and emissivity. OTR is set to 25% to ensure that it meets on par with the operational temperature range of the outer cover, inner cover, and materials used to bind them together. TRL is set to 15% as these proposed materials are heritage and have been flown in past missions, but are included to reduce any lingering risk. Absorptivity is at one of the highest levels at 30%, due to ensuring that there is a minimal amount of conductivity throughout the layers when heat passes through. Emissivity value was also valued at 30% and was added to ensure emissivity values are in the low-mid ranges, which aid in minimizing conductivity as well.

Aluminized Kapton offers an excellent temperature range at 23.15K to 563.15K [15], which aids in not only working in the extreme Martian environment but also when the rover enters a hot case or a cold case scenario. The TRL was assessed as this type of interior layer was proven in both Mars missions and deep space missions as well making it TRL 5-6, which helps assess the technological risks. However, it has high values of absorptivity, thus when it comes to ensuring that the inner layers minimize conductance as much as possible to ensure thermal isolation of the system. Emissivity values were at the low-mid range when it comes to minimizing conductance, as well as for the Aluminized Kapton. The Aluminized Mylar, although having similar absorptivity and emissivity values as the Aluminized Kapton, does not meet temperature requirements, with it being 213.15K to 423.15K [51], and will at times shrink when the rover approaches its hot case, which will affect the structural integrity of the MLI system especially for the long-term duration. The Goldized Kapton possessed a greater temperature range and better absorptivity when compared to its two other components. Thus, it is favorable when it comes to wanting to achieve as minimal conductance as possible.

Goldized Kapton was ultimately chosen as the ideal interior layer when it comes to attempting to minimize as much thermal conductance as possible in order to aid in thermal balancing. Such perks include its exceptional temperature range, heritage levels of TRL proven in past space missions, as well as high absorptivity and low-mid emissivity values. These perks aid in solidifying its choice in being an ideal interior layer for the MLI when it comes to meeting systems requirements.

[Thermal Management Trade Study for Heat Pipes]

Figure 1.5.5.3.3: Heat Pipes Trade Study

Heat Pipes are necessary when it comes to distributing heat evenly throughout the rover system and maintaining thermal balance. This trade study evaluates the 3 heat pipe candidates, which include: Variable Conductance Heat Pipe(VCHP), Constant Conductance Heat Pipe(CCHP), and Thermosyphon. This is to ensure that these selections meet the mission system requirements.

These pipes are evaluated based on criteria set for them, such as operational temperature range(OTR), TRL, reliability, and complexity. The OTR is set at 30% because it is crucial that these heat pipes work under a desirable temperature to reduce the risk of cascading failures, which jeopardize the mission. The TRL is set at 15% to evaluate whether the technology is mature enough for this mission. The reliability is set to the highest at 35%, to ensure that the heat pipe can operate in extreme conditions, along with long durations which is crucial when it comes to thermal control. Complexity is set at 20% to ensure whether the heat pipe design can be integrated into the system without having to alter other subsystems.

The VCHP is a passive heater that contains a reservoir containing a non-condensable gas(NCG). The gas acts as a thermal spring, expanding and contracting as the pressure inside the heat pipe changes with varying temperatures [2]. The temperature range of the VCHP ranges from around 120K to 374K [2] , which makes it a great choice for a heat pipe. The TRL of the proposed heat pipe is space proven and has been flown before in space, thus making it a TRL-5. The VCHP is reliable as well and it can accommodate passive heating for long-duration missions. However, for the VCHP to work in the Martian environment, it must be designed with wicks for it to work properly in any orientation, mainly affected by gravity [3]. The CCHP works similarly to a VCHP, transferring heat from one location of the pipe to another via evaporation and condensation. It has a decent temperature range of 200K to 423K [4], which makes it work as well as its VCHP counterpart. The TRL for the CCHP is set at a TRL-5 as it has been used on past Mars rover TCS before. Like its VCHP counterpart, it has proven to be reliable when it is deployed for long-duration missions. The CCHP can work properly without any concern for the effects of the Martian gravity being applied to it. The Thermosyphon has a lower heat transfer capacity, which is a negative factor when it comes to being applied in the Martian environment. The TRL levels for the Thermosyphon are as low as there are no records that indicate it can withstand the Martian gravity, which is significantly less than that of Earth which it has relied for its

Earth-based applications. It has proven to be reliable when it comes to other heat pipe alternatives, but very complex when transitioning to a different gravitational orientation of Mars, which can hinder its efficiency.

Both the VCHP and CCHP were chosen following the rover design. This system shall work in unison to provide thermal stability of the system as a whole. It shall have two loops to control the heat sink which is the outer environment. This selection will aid in meeting the system requirements for P.H.O.E.N.I.X.

Thermal Subsystem Trade Study for Electrical Heaters

Figure 1.5.5.3.4: Electrical Heater Trade Study

An active heating subsystem must be implemented into the system itself as a whole to meet the mission system requirements. A trade study was conducted on 3 electrical heater candidates to fulfill the requirements; the KHLVA, PLM-Series Electrical Heater, SHK series Electrical Heater, and the Polyimide Thermofoil HK Series Electrical Heater.

Power is set to the highest at 30% since it is more favorable to be as power efficient as to thermal efficiency, due to the limited power supply. Mass is set at 25% as the compactness and its weight are important factors when it comes to meeting (SYS.04). TRL is set at 20% to ensure that the proposed electrical heater is mature for Martian applications. The temperature Range is at 25% to evaluate whether the proposed electrical heater has an acceptable temperature range that allows the system to operate properly.

The KHLVA, PLM-Series Electrical Heater offers a low power draw at 0.775W/cm^2, which is favored when power is limited [17]. This heater is also very compact being 2x1 inch and 0.01 in thick, making it small and easy to integrate. The heater was deployed through launches of NASA CubeSat satellites, marking it at a high level of TRL. This series offers an exceptional temperature limit of 216K to 505K, which is sufficient to keep the rover system operational. The SHK Series has a similar power draw as its KHLVA, PLM-series counterpart, also numbering at 0.775W/cm^2. The heater also offers compactness when it comes to dimensional constraint considerations. The SHK was used in CubeSats as well, demonstrating a higher degree of TRL. However, the SHK temperature ranging window is not up to par with its KHLVA counterpart, it still offers a great range at 238K to 423K [47] with the inclusion of Aluminum foil. The Polyimide Thermofoil HK Series electrical heater draws a higher power per cm^2 at 3W/cm^2, which is significantly higher than the two other electrical heaters [34]. This specific heater, like its other counterpart, is extremely lightweight and easy to integrate into any component of the spacecraft/rover. The TRL of this specific electrical heater is also high as well as it was also used in CubeSats. The Polyimide also offers an impressive range of 73K to 473K [47], which, like the KHLVA series and SHK series, will be able to keep the rover at operating temperatures.

The KHLVA, PLM Series Electrical heater was chosen as it meets the power draw efficiency, along with an exceptional temperature range. The KHLVA also demonstrated that it was flight-proven via CubeSats along with its compactness, making it an ideal candidate. This selection aids in meeting the system requirements for P.H.O.E.N.I.X.

Thermal Subsystem Trade Study for Thermal Sensors

Figure 1.5.5.3.5: Thermal Sensor Trade Study

Thermal sensors are an absolute component when it comes to designing of the TCS subsystem. A trade study was conducted on 3 thermal sensors: the Thermocouple, Thermistor, and RTD sensors, to ensure these work per the system requirements.

Sensitivity is set at 25% to ensure that the sensors can read any small temperature changes; high sensitivity is preferred. Accuracy is set at 20% for the measurement of how accurate the reading is to the actual temperature. Reliability is set at 25% which is to ensure that the sensor can withstand long-term missions. Response time is also set at 30% is crucial to detect a temperature change as fast as possible without any delay.

The Thermocouple has a low and small sensitivity, which is not ideal when trying to obtain a change in temperature data. The accuracy of the thermocouple is in the medium range, hence the grading. This particular is durable and resistant to corrosion due to a protective metal sheath, hence why it has an excellent reliability value. However, a sensor encased in a protective metal sheath is more resistant to wear and corrosion over time, but it costs more and offers less sensitivity. Its response time, like its sensitivity, lacks overall, making it not that great and not that bad in general. The Thermistor, unlike the thermocouple, has a high sensitivity making it ideal when measuring any minor flux in temperature. The accuracy of this sensor is also in the medium range when it comes to measuring results. It lacks durability when it comes to being deployed for long-duration missions, thus a redundant system needs to be implemented . Its response time is also similar to the thermocouple when it comes to detecting change. The RTD has a moderate level of sensitivity, which is not as impressive as a thermistor. However, the accuracy levels of the RTD are not that good and thus not recommended when it comes to implementing it into the system. The RTD also has good durability, but not as impressive as the thermocouple when it comes to long-duration missions. The RTD has a better response time overall than the thermocouple and the thermistor, but it will not help when it comes to this trade study. The grading for three of these instruments is based on National Instrument data [25].

Ultimately, the thermistor was chosen as the heat sensor for the system as a whole, as its grading is better off from the other candidates. Its sensitivity and reliability make it an ideal candidate when it comes to being utilized on P.H.O.E.N.I.X.

1.5.6 Payload Subsystem

To meet the Human Exploration Goal, *HBS-1LM*, the mission will conduct scientific experiments to achieve the 2 objectives on this focus. The detailed breakdown for the following payload selected is provided in the sections 1.5.6.2, 1.5.6.3. The selected instrumentation to achieve the objective of investigating the long duration Martian environmental impacts on a protected and pressurized earth-fluid sample for unknown hazards that may threaten the integrity of future mission-critical life support, rocket propellant, and agricultural fluids, is the CP-MU DMU-100 Submersible Gamma Neutron Probe [28]. This instrumentation is the best selection to conduct the experiment, and meet the selected scientific measurement requirements to periodically monitor the Earth fluid sample for risks, hazards, and contamination that may bypass the custom-engineered Fluid Protection System's protective layers and document via data generation. To further meet the Human Exploration Goal, the second objective sets to investigate how subsurface stratigraphy, dielectric properties, and dust layer thickness affect the accessibility and long-term stability of near-surface water ice, in support of in-situ resource utilization and environmental risk reduction for future human exploration, with the Radar Imager for Mars' Subsurface Experiment (RIMFAX) to be the ground penetrating radar instrumentation in use [18]. This instrument met the highest criteria to conduct the experiment and use GPR reflection delays, estimating dielectric permittivity and radar wave velocity to characterize subsurface material properties, including layer thickness, composition changes, and porosity variations across a 10 m traverse. The mission will also meet a Science Exploration Goal, Q10.3b, by conducting scientific experiments to achieve the 2 objectives on this focus. The selected instrumentation to achieve the objective of determining the Deuterium to Hydrogen (D/H) ratio in hydrated volcanic rock on Mars' surface, is the Miniature Tunable Laser Spectrometer (Mini-TLS) as it met the best criteria to be used to define the relative abundance of protium and deuterium within samples of hydrogen from hydrated volcanic rock. Concluding with the last objective, which is to determine the crystal structure of minerals formed by asteroid impacts interacting with exposed subsurface ice, the instrumentation meeting the highest criteria was the Raman Laser Spectrometer (RLS). This instrument will identify chemical structure, crystal structure, and bond structure of Olivine from asteroids.

1.5.6.1 Payload Subsystem Requirements

The payload subsystem consists of four science instruments selected to achieve the mission's primary objectives as defined in the Science Traceability Matrix (STM).

Each instrument requirement is derived from its scientific function, expected performance under Martian conditions, and its interface with other subsystems such as thermal control, command and data handling, and mechanical mounting.

The Radar Imager for Mars' Subsurface Experiment (RIMFAX) supports the goal of identifying accessible water ice deposits. It must detect dielectric differences down to a depth of 10 meters with a permittivity sensitivity ≤ 0.1. To meet this requirement, the instrument must be mounted on a stable platform with minimal electromagnetic interference and sufficient power and data bandwidth [18]. It must also interface with the mobility subsystem to maintain ground coupling while traversing varied terrain.

The Submersible Gamma Neutron Probe (SGNP) addresses the objective of monitoring radiation hazards to mission-critical fluids. It shall operate continuously for at least one Martian year, detecting gamma and neutron radiation levels between 1 μSv/h and 10 Sv/h. To support long-duration performance, the instrument requires thermal stability within operational bounds and mechanical integration within a sealed fluid testbed. The probe must be electrically and thermally isolated while still interfacing with CDH for periodic data collection and transmission.

The Miniature Tunable Laser Spectrometer (Mini-TLS) enables analysis of the Deuterium-to-Hydrogen (D/H) ratio in hydrated volcanic rock. It must collect absorbance spectra with a sensitivity better than 80 ppb and an integration time of ≤1 second. The spectrometer must be mounted on a mobility-accessible arm or stable surface to acquire accurate samples, with localized thermal control and vibration damping to maintain optical calibration. Integration with CDH is required for both real-time command and high-resolution data logging.

The Raman Laser Spectrometer (RLS) is used to analyze the crystal structure of Olivine and other minerals formed via ice-impact interactions. It must collect spectra in the 11,111–33,333 nm range and resolve features with at least 10 cm^{-1} spectral resolution. This instrument operates effectively only within a thermal envelope of 228 K to 313 K, requiring close coordination with the thermal control system (TCS). In addition, the RLS must be positioned to access mineral targets and avoid shadowing or dust contamination.

Each instrument's performance is subject to verification through functional testing under simulated Martian conditions (temperature, radiation, and mechanical stresses). Requirements also include placement, power draw, signal conditioning, and structural mounting interfaces, which will be fully specified in the Interface Control Document (ICD).Together, these instruments ensure the P.H.O.E.N.I.X payload can deliver high-resolution, high-value science within the constraints of volume, mass (≤15 kg), and energy budget allocated to the payload subsystem.

1.5.6.2 Payload Subsystem Overview

The chosen instrument to achieve the first objective is the CP-MU DMU-100 Submersible Gamma Neutron Probe. Waterproof and made to withstand high heat applications, it is durable, and built to last as a tried-and-true TRL level-6 scientific safety instrumentation actively utilized by first responders, nuclear facilities, and hazmat teams in extreme environments, proving its worth through a history of reading Gamma radiation dosage in multiple sources such as water, nuclear reactor cores, and fuels. It will now extend its earth legacy to a multi-planetary one, operating inside of the team's custom-engineered and rover-attached Fluid Protection System (FPS) for monitoring a protected earth fluid sample for Martian environmental impacts and contamination in weekly intervals over a one year duration. This recorded data will provide a strong contribution to materials science and engineering breakthroughs, uncover unknown threats to mission-critical fluids, and aid in the engineering of the next generation fluid storage systems that are necessary for astronaut safety and sustaining of long-duration missions on Mars. With a dosage recording range of 1 µSv/h to 10 Sv/h and 12 seconds slow time constant (response time), the neutron probe is capable of precise, accurate, and stable contamination detection across subtle to extreme Gamma radiation level increases. The instrument was originally designed by the manufacturer for mobile operation using batteries, but the team has implemented a hardwire power method via rover where it will use an estimated maximum power of 270 mW. The instrument is designed to be compact, maintaining a lightweight mass of 1.3 kg and a total volume of 3362 cm^3, fitting well within the 15kg total mass and 0.5 m x 0.5m x 0.5m total volume constraints for the human exploration mission goal.

The main instrument selected to support the second objective is the Radar Imager for Mars' Subsurface Experiment (RIMFAX). This ground-penetrating radar system is designed to characterize subsurface stratigraphy and identify dielectric contrasts that can indicate dust layers, material transitions, and potential ice-rich zones. RIMFAX operates between 150 and 1200 MHz, allowing it to achieve vertical resolutions down to 15 cm in the shallow subsurface and detect features as deep as 10 meters under ideal conditions [18]. It works by transmitting radar pulses into the ground and measuring the delay and strength of reflected signals, which can then be used to estimate dielectric permittivity and radar wave velocity [18]. From these values, the instrument enables indirect assessments of porosity, composition changes, and regolith layering, all critical to evaluating the accessibility and stability of near-surface ice. The instrument includes a folded dipole antenna mounted on the rear of the rover chassis, oriented downward to allow continuous ground-facing data collection during the traverse. This antenna transmits and receives radar signals across a wide bandwidth, helping to resolve both shallow and deeper subsurface structures with high fidelity. It collects data continuously along the rover's traverse, generating radargrams that provide a 2D view of subsurface structure. The system is compact around 3 kg, requires relatively low power (~5–10 W), and involves no moving parts, reducing the risk of mechanical failure. It also fits within the mission's experiment mass and volume constraints. With a Technology Readiness Level (TRL) of 9, it is flight-proven, having successfully operated on NASA's Perseverance rover. Its performance, reliability, and data return make it the most effective and mission-aligned instrument for achieving the subsurface science objectives.

The instrument chosen to fulfill the mission's third objective (Determine the Deuterium to Hydrogen (D/H) ratio in hydrated volcanic rock on Mars' surface) was the Miniature Tunable Laser Spectrometer. Specifically, the Mini-TLS is included in the mission to perform the task of defining the relative abundance of protium and deuterium within samples of hydrogen from hydrated volcanic rock. To carry out its task, the instrument needed to collect absorbance spectra in the 3593.3-3594.3 cm^-1 range of H in selected hydrated volcanic rock samples at multiple surface sites. It collects its data through a process very similar to its predecessor, the Tunable Laser Spectrometer (TLS). This process begins with sample collection. Once a sample has been acquired, it is put into an oven where it is then heated up. During this process, the water molecules inside of the rock are released as water vapor (both the HDO and H_2O), and this evolved gas is redirected to a different chamber with a tunable infrared laser. For this mission, the laser will be tuned to detect the absorption of HDO and H_2O. Since deuterium absorbs a different wavelength of light than protium, the Mini-TLS is able to distinguish between the two and perform the D/H ratio calculation (D/H Ratio = $[HDO]/[H_2O]$). At only 1 kg and 23.5 × 16.3 × 15.5 cm, the instrument is low power taking less than 8 watts [17]. The instrument has an approximate technology readiness level of 6, as it has been tested in analog Mars conditions but has not been utilized in a mission.

The selected instrumentation to achieve the science exploration goal and its last objective includes the Raman Laser Spectrometer (RLS) which is designed to analyze and detect what materials from Mars are made of based on its reactive reflection to laser light. It is designed to function by analyzing the crystal, chemical, and bond structure on a sample of Martian rock fragments and asteroid olivine that has been altered by geophysical and environmental forces to determine what elements may be within those samples. The scientific goal of investigating how the dynamic forces on planet Mars have altered the existence of water in its liquid form is pursued through the direct analysis of the surface and subsurface chemical composition. The instrument conforms to the mission's rover with a mass of 2.4 kg and a volume of approximately 81 × 98 × 125 mm^3 [24]. It has a power consumption of 20 to 30 watts that varies with the temperature and operation mode along with performance metrics that ensure evident mineral detection as well as detection of trace components. This instrument will collect Raman spectra with a range between 11,111 to 33,333 nanometers of olivine within the selected location of Erebus Montes. The TRL of the RLS instrument is at a 9 based on environmental testing and system level integration [24]. It begins to function when a laser beam is directed into a held sample in its analysis chamber. As the laser interacts with the molecules in the sample, it provokes the Raman effect which is where a small portion of the light is inelastically scattered. This scattering results from changes in the vibrational energy levels of the molecules, producing unique shifts in light energy. These shifts correspond to specific vibrational modes, which it detects and converts into spectral peaks. This collection of data creates a Raman spectrum that reveals the sample's molecular and mineral composition to better understand Martian geology [14].

Overall Payload Subsystem TRL is a 6, determined by our lowest TRL payload.

Instrument	Mass	Dimensions/Volume	Max Power
CP-MU DMU-100 Submersible Gamma Neutron Probe (Probe) + External Box	1.3kg	3362 cm^3	0.25 Watts
Radar Imager for Mars' Subsurface Experiment (**RIMFAX**)	3 kg	1552.32 cm^3	10 Watts
Miniature Tunable Laser Spectrometer (**Mini-TLS**)	1 kg	5937.28 cm^3	8 Watts
Raman Laser Spectrometer (**RLS**)	2.4 kg	992.25 cm^3	30 Watts
Totals	7.7 kg	11843.85 cm^3	48.25 Watts

Figure 1.5.6.2.1: Instrumentation Table

1.5.6.3 Payload Subsystem Trade Studies

[Payload Subsystem Trade Study for Fluid Protection & Testing System]

Figure 1.5.6.3.1: Thermal Management Requirements Table

Selecting the optimal instrument for monitoring gamma radiation in the Earth sample proposed three options for comparison within the trade study: *CP-MU DMU-100 Submersible Gamma Neutron Probe*, *Mirion TN-15*, and *CT007-T Thermal Neutron Detector*. Instrument evaluation was initiated based upon the following criteria and associated weight (in order of priority that is most critical for science objective success): Size(40%), Submersed Durability(25%), Data Transmission Method(25%), Mass(10%).

The first and most important criteria to consider is the scientific instrument's size as it must internally reside within the custom Fluid Protection System(FPS), requiring a length of no more than 25cm x 25cm. Excessive instrument size would result in requiring a larger scale system, potentially increasing overall mass, fluid capacity, material requirements, costs, and involving additional hazards with internal pressurization, rover balance and terrain-induced physical damage. All three instruments (among others found outside of the trade study) easily met this requirement as most compact Neutron Detectors fall within the measurements.

The second criteria Submersed Durability is critical to the life expectancy of the instrument and fulfilling the scientific goal by its ability to generate periodic data of gamma radiation dosage for more than one year while submerged in multiple variations of fluid for data transmission back to earth. Neither the CT007-T Thermal Neutron Detector nor the Mirion TN-15 are able to withstand being submerged in fluid without the requirement of a custom housing, which could induce additional risks of instrument failure as it would be submerged for at least one year. The CP-MU DMU-100 Submersible Gamma Neutron Probe is a perfect fit to not only meet, but exceed expectations in this criteria as it is not only intended to be submerged in water, but can also be used in fuels as it was designed for use in reactor cores, components, and spent fuel rods to generate dose rate measurements of gamma radiation. The durability, versatility of fluids, and tried and true usage in the nuclear industry sparks confidence of reliable data retrieval over a fully submerged one year period, if not longer.

The third criteria Data Transmission Method is critical to allow radiation dose data generated by the instrument to be transmitted directly to the rover’s command processing unit. Many compact scientific instruments use wireless transmission for ease of use, but would not be feasible due to potential signal interference, the FPS’s protective housing preventing outgoing signal transmission, or a sub-optimal transfer speed rate of data. To mitigate as many risks as possible a wired ethernet connection is required for reliable, interference-free, and timely transfer speeds without the risk of signal loss when handling data vital to the objective. Out of all three instruments, the CP-MU DMU-100 Submersible Gamma Neutron Probe wins first place due to a unique feature not commonly seen in compact instrumentation, allowing ethernet connection.

The fourth criteria, Mass, plays an important role in instrument selection as the FPS encompasses the scientific instrument, protective housing, internal fluids, wiring, and mount that all must total under 15 kg mass to fit within the HBS-1LM Human Exploration Goal constraint. To allocate enough mass for the heaviest parts of the system without risking exceeding constraint, the instrument is required to be compact with mass not exceeding 2kg. Due to being compact instrumentation, all three instruments evaluated met this requirement, with most averaging 60-70% of the maximum kg.

Based upon the conducted trade study results, requirements, logic, and quality of data, the scientific instrument chosen to fully achieve the Human Exploration Goal is the CP-MU DMU-100 Submersible Gamma Neutron Probe. Scoring in at 3362 cm^3, under 1.3 kg in total mass (for both the external detector box and probe combined), engineered for long-duration submersal, a wide range fluid type versatility, proven track record in the nuclear industry, and an efficient Ethernet-based data transmission.

Payload Subsystem Trade Study for Ice Accessibility Instrumentation

Figure 1.5.6.3.2: Ice Accessibility Instrumentation Trade Study

To determine the best instrument for measuring near-surface ice accessibility and understanding regolith effects, the following three options were compared: RIMFAX, HP^3, and a combined RIMFAX + HP^3 system. A weighted scoring system was used, where a score of 1 is the best and 5 is the worst, and the highest percentile total score wins.

The highest weight of 22% was given to depth and quality factors as subsurface layers and possible areas of ice will need to be detected. RIMFAX got the best score here because it is able to reach ~10 meters with high resolution [18], while HP^3 was more limited and had issues deploying on the InSight Mars station in 2018 [76], so received the higher (worse) score. Mass and volume were weighted highest at 25% because of the customer's constraints where the total system must stay within 15 kg for mass and 0.5 m^3 for volume. The combined system (RIMFAX + HP^3) scored worse here since it would likely break both limits.[18, 76] Instruments that are already flight-proven were prioritized hence why the TRL was weighted at 18%. RIMFAX and HP^3 both flew on Mars, but RIMFAX got a slightly better score due to smoother integration on the Perseverance rover. Thermal Stability criteria is about how well the instrument can contribute to measuring temperature weighted at a 10% for data purposes. HP^3 scored better here since it was designed to measure heat flow. RIMFAX doesn’t directly do that but still contributes through context. Data Volume was weighted at an (15%) as this about how much data is produced and whether it can be handled easily. RIMFAX scored better since its radargrams are compact. The combined system would create too much data for the current bandwidth.

The end results showed RIMFAX to come out on top. It gave the best performance for depth and resolution, stayed within mission limits, and carried less risk. The hybrid system was too heavy and complex, and HP^3 on its own didn’t give enough subsurface clarity. Based on this trade study, RIMFAX is the most balanced and reliable option for the science mission goals.

Payload Subsystem Trade Study for Deuterium:Hydrogen Ratio Trade Study

Figure 1.5.6.3.3: Deuterium:Hydrogen Ratio Trade Study

In order to select the best instrument to measure the deuterium to hydrogen (D/H) ratio in hydrated volcanic rock on Mars' surface, a trade study was conducted to analyze the scientific potential of 3 instruments (and/or instrument combinations): The Mini Tunable Laser Spectrometer (Mini-TLS), Orbitrap, and the Tunable Laser Spectrometer (TLS). These instruments were evaluated based on the following: mass, measurement precision, volume, power consumption, support system, and reliability. Each criteria was given a certain weight in alignment with the mission, with the lowest being weighted at 10% and the highest 3 criterias weighted at 20%.

Beginning with one of the highest rated criterias, measurement precision is weighted at 20%. Measurement precision is extremely important to the mission because correct data is imperative to good science. To fulfill this objective in particular, a very high resolution is required. An instrument must be able to fully distinguish between Hydrogen and Deuterium to minimize spectral overlap and measurement uncertainty.

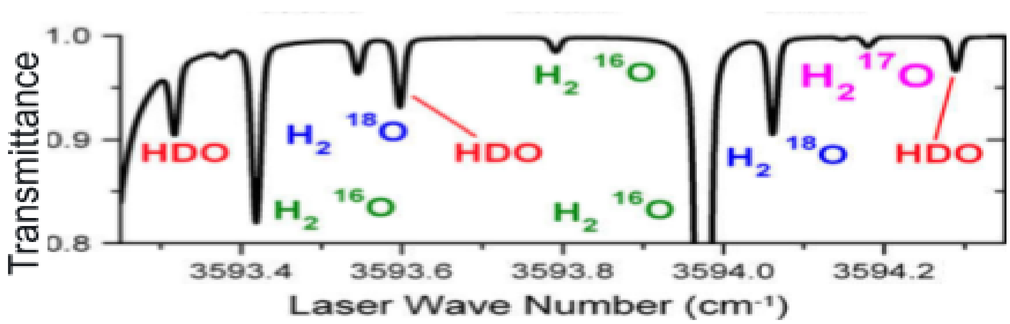


Figure 1.5.6.3.4: HDO and H_2O Transmittance Spectrum from Curiosity [54]

The second criteria that is weighted at 20% is reliability. This is a less quantifiable, but just as important metric. The system that is chosen will go through tough conditions including dust, radiation, and frequent temperature changes. Additionally, the system must last long enough to deliver important scientific data back to Earth. This criteria is judged based on how exposed the instrument is to failure under Martian conditions as well as past successes under similar conditions.

The third with a 20% weight is the support system. Due to the requirements of the scientific objective to measure D/H ratio in volcanic rocks, many spectrometers need a system to obtain the evolved gas from these samples. This could include pyrolysis, laser ionization, chemical extraction, vacuum sublimation, etc. To account for the power draw, mass, volume, and overall needs of this support system, this criteria is given a heavy weight to ensure the system is relatively free standing.

The next criteria, power draw, is weighted at 15%. Power, especially when RTGs are forbidden by the customer, is a huge consideration for a rover. Since power is limited, instruments that have low power requirements help to reduce risk of overloading the system and increase the amount of science which can be achieved.

Mass and volume are weighted at 10% and 15% respectively. The rover must fit within strict mass and size constraints imposed by the customer. Either of these criteria in excess reduces the allocation of mass and volume to the rest of the rover. This places critical systems to operate the rover at risk of failure if they are forced into trade-offs. Volume is weighed slightly more due to the abundance of lightweight spectrometers that were possible instruments, but volume often being too large to justify for the rover.

The conclusion of this trade study was that the Mini-TLS was the best fit for this mission with the highest score of 80.63%. Its lightweight and compact volume make it the ideal choice to optimize the mass and volume constraints. Its development from the TLS led to its very high level of precision, 10 times more sensitive than the TLS is. Additionally it does require an oven like the TLS, but a smaller one because it requires 1/100th of the gas. However, it is given a lower score in reliability because the TLS has proved itself on Mars in similar conditions to where P.H.O.E.N.I.X will be deployed.

Orbitrap represents the very best in terms of precision, but this comes at a cost. Its mass and volume both exceed that of Mini-TLS and partially of TLS. It also faces the need for a large support system including a vacuum and laser ionization system to obtain gas. This is scored roughly the same as TLS because at max temperature the oven on TLS maxes out its power connection power and requires a lot of thermal insulation [38]. As a whole the vacuum and laser would require a similar power level, and although the thermal insulation would be less, a vacuum requires reinforced materials.

Payload Subsystem Trade Study for Crystal Structure Instrumentation

Figure 1.5.6.3.5: Crystal Structure Instrumentation **Trade Study**

To evaluate the best instrument for determining crystal structure on Mars, a trade study was conducted to compare these three candidates: the Raman Laser Spectrometer (RLS), SuperCam, and CheMin (XRD/XRF). The goal for the trade study was to assess which system best aligns with the science objective to determine the crystal structure of minerals formed by asteroid impacts interacting with exposed subsurface ice.

The criteria chosen for this trade study were based on their relevance to mission success and the constraints mentioned by the stakeholders. Power consumption was weighted at 15% because limited power availability due to the system's mass and reliance on solar energy makes low power instruments more favorable for prolonged operations. Durability against solar winds was assigned a 10% weight, as instruments must withstand the charged particle exposure to ensure the data is protected and not lost in the harsh Martian environment. Heat required carried a 30% weight, recognizing that instruments reduce the risk of mission complications due to Mars' extreme temperature range. Finally, operational lifetime was given the highest weight at 45%, since instrument success would fulfill the mission’s objectives across its planned duration. Each instrument was graded on a 1 to 5 scale, where 1 indicates the best performance and 5 indicates the worst. These scores were then multiplied by the assigned weight percentage and then all the scores across all criteria were summed to generate a total weighted score for each instrument.Raman Laser Spectrometer (RLS) scored 100%, SuperCam scored 71.25%, and CheMin (XRD/XRF) scored 85.00%

The Raman Laser Spectrometer had the winning total score with 100% which shows its best performance across all critical mission constraints. It demonstrated to meet every weighted criteria. While SuperCam and CheMin have strengths in specific areas, their higher heat requirements, shorter lifespans, and increased power draw made them less suitable under the mission requirements. In conclusion, the Raman Laser Spectrometer was selected as the optimal instrument for determining the crystal structure of minerals formed by asteroid impacts interacting with exposed subsurface ice.

1.5.7 Recovery and Redundancy

Recovery and redundancy are key risk mitigation tools for preventing system loss of various kinds. Each subsystem is responsible for identifying possible risks and various recoveries and redundancies that can be implemented to prevent catastrophic mission failure.

Mechanical Subsystem

The rover chassis is too large to feasibly implement redundancies. Individual attachment points shall incorporate redundancies through a FOS in attachments, however, these methods allow for cascading failure under large loads and alone are not satisfactory. Recoveries are possible for wheel assembly failure in isolation through robust autonomous locomotion controllers utilizing partial functioning systems.

Power Subsystem

The rover contains multiple redundant electrical systems that shall work alongside the primary power systems in the event that an issue renders the primary systems unusable or inactive. The rover’s design includes primary and secondary power sources. In the event of solar panel degradation, the power subsystem will revert to Lithium ion batteries in order to continue operating instead of carrying a backup set of panels in the payload, as they would be too massive. There is a second, redundant battery included in the mission payload in the event that the primary backup battery does not operate as expected. If the power system goes offline, the included power buffer shall prevent power loss or surges that could harm the mission operation.

Command & Data Handling (CDH)

The rover will store its own local world map, and is capable of using its elevation relative to its surroundings as a heuristic for signal propagation effectiveness, and therefore if the rover loses contact with the MRO, or earth ground station it will be able to autonomously search for acceptable command signals. Due to the lightweight nature and incredible importance of the OBC and data storage devices, failure mitigation can be implemented through duplicate OBCs and data storage such that [there will be two of each]. Larger subassemblies including the antenna, however, do not justify complete redundancy due to their large mass and volumetric constraints. Antenna failure follows grateful degradation and therefore is unlikely to become completely inoperative, but rather to

Thermal Control System

The Thermal Control System subassembly's redundancy shall focus on the heat pipes, thermal sensors, and electrical heaters. There will be two systems of heat pipes, so that every instrument does not rely on the same system. A failure in one should not affect the other instruments.There shall also be multiple thermal sensors per instrument to ensure cross-verification of the measuring of the instrumentation temperature. This also ensures that in the event of failure, the other will be able to monitor the instrumentation and keep the TCS intact. Redundant heaters will be implemented. For the Thermal Recovery, there shall be a recovery software implementation where if the TCS fails, a fail-switch sensor sends the signal to command the TCS, primarily the electrical heater to commence a re-startup sequence.

Payload Subsystem

The payload subsystem will address recovery and redundancy for each instrumentation in the event of issues arising with their function or performance. CP-MU DMU-100 has a freeze risk that will be addressed by running testing during the night to heat up the internal housing fluid environment as the instrument operates at a minimum of 30°C which is warm. On Mars, clays and ice-dust layers can tamper RIMFAX's radar pulses and disrupt its depth, to confront this, the instrument will stick to the low end of its 150–1200 MHz band, limit the soundings to 2–4 m, and stack multiple passes for a clear return. For the Raman spectrometer, a quick calibration check will be made before and after every run and only trust fully quantitative data when the sample sits at the controlled 935 K, outside those windows spectra is treated as qualitative data. With the Mini-TLS, rather than investing in extra cooling gear, measurements will be conducted at night, when Martian temperatures reliably stay between –10 °C and 20 °C, keeping the laser and optics within requirements.

1.5.8 Interface Control

Spacecraft System Interfaces						
Spacecraft Structure	Mechanical Structure	Mechanical Structure	Mechanical Structure	Mechanical Structure	Mechanical Structure	Mechanical Structure
	Payload Instrument module	Science data output Instrument status				
	Experiment start/stop Instrument Activation Command	**Command and Data Handling**	Science data Transmission System telemetry System Health monitoring report Autonomous event log	Navigation command Mobility path update Route execution signal	Thermal data report Environmental Status Update	Commands (Power distribution to systems)
		Received data and Terrain Info Surface Map	**Comm**			
		Location update System Health data		**Navigation**		
Radiative Heat Dissipation	Active Heating and Cooling	Temperature Data Readings	Active Heating	Active Heating	**Thermal**	Active Heating
Power	Power	Power	Power	Power	Power	**Power**

Figure 1.5.8.1: N^2 System Interfaces Chart

The system interface diagram shows the relationship between each subsystem and how each subsystem interfaces with the other. These subsystems include the following: structures, payload, data handling, comms, navigation, thermal and power. Each of the subsystems play a distinct role and are crucial to the success of this mission. By mapping out the interfaces, failure modes can be identified and mitigated early on in the design process of the rover.

Each subsystem is configured to have its own specific function but it all comes down to system integration, hence the N^2 chart is a visual representation of how all subsystems will have to work together in sync, ultimately supporting the spacecraft's goals and ensuring operational integrity.

The spacecraft structure is the foundational subsystem, it consists of the chassis, wheels, motors, actuators and all mechanical components that are physically present on the rover. This subsystem is positioned at the highest level on the interface chart because it serves as the primary housing and structural support for all the other subsystems. The key functions of this subsystem include providing support and physical protection to all internal and external subsystems ensuring structural solidity of the spacecraft.

The main system interfaces the structure subsystem receives outputs from is the power and thermal control subsystem. The structure is reliant on the power subsystem to supply electrical energy to the mechanical components, such as the motors on the wheels or any actuators the rover will have. This reinforces the rover's mobility and the operation of any deployable or moving parts, such as a deployable radiator used by the thermal control subsystem or deployable solar cells used for generating electrical power.Adequate power supply is essential for maintaining the rover's navigation, scientific experimentation and data collection capabilities.

Furthermore, the chart also illustrates the most vital output to the spacecraft structure which is heat dissipation through the structure from the TCS subsystem. TCS is a critical system that keeps the spacecraft structure, any components and electronics within nominal temperature ranges considering the harsh thermal environment on Mars with temperatures that go as low as XX degree celsius and as high as XX degrees celsius. Despite these extreme temperature fluctuations, it is important to keep the structural elements of the spacecraft and any mounted subsystems operational.

The spacecraft structure is essential to the integration of all subsystems which highlight the unidirectional flow of power and thermal management into the spacecraft structure, emphasizing its dependence on these subsystems for flight readiness.

The payload subsystem, which is the instrument module, houses all of the scientific instrumentation that is designed to achieve both the scientific and human exploration goals provided by NASA that is the customer. This suite of instrumentation is an integral part of the mission, as it enables collection and transmission of valuable scientific data that pertains to future sustainability and manned missions on Mars.

The instrument module consists of distinct and unique instruments, each having its own functionality but working towards achieving the desired science objectives.

This subsystem solely sends inputs to the CDH subsystem but receives outputs from CDH, TCS and the power subsystem. The payload sends an instrument status signal to the CDH subsystem which includes an onboard computer that processes that signal and decides if the instrumentation is currently offline, if that is true, the CDH subsystem outputs an experiment activation command signal to the payload subsystem, where the instrumentation begins data collection. Once the data has been collected and the instruments have performed their operation at the research point, the scientific data is then inputted into the CDH subsystem which then sends an experiment stop signal to the payload subsystems turning all of the instruments offline to conserve power. This closed loop interaction ensures that scientific experiments are conducted efficiently and that payload operates only when required, saving power and conserving the instrument's operational lifetime.

The outputs that the payload subsystems receive from TCS is in the form of active heating and cooling. Each instrument in the payload has its own nominal operating temperature range and going above or below it would compromise on the effectiveness of data collection and instrument redundancy. To counteract this, the TCS provides heating through the means of a heat pipe or cooling through cryogenic cooling systems, depending on the time of the day and solar loads the structure receives.

Furthermore, it also receives an output in the form of electrical power from the power subsystem as all of the instrumentation will require power to function and start experimenting.

The Command and Data Handling (CDH) subsystem serves as the central data repository for all of the scientific data that is transmitted to and from the instrumentation to the onboard computer. This subsystem is responsible for smooth data flow throughout the duration of the mission. The CDH subsystem interacts with each of the subsystems in a certain order, first interacting with the payload subsystem where it interacts in a closed loop, with inputs from the payload and output from the CDH subsystem back into the payload.

This subsystem also inputs an autonomous event log and an overall system health report to the communications subsystem which then relays this information to the MRO that relays that same scientific data back to the earth ground station.

The CDH subsystem also inputs different signals and commands into the navigation, TCS and power subsystem. The inputs into the navigation include path and mobility data based on the surface terrain data that it receives from the communications subsystem as well as the navigation command to direct the rover on where to find the next science research point of interest. It also transmits a thermal data report to the TCS subsystem that includes the temperature at the surface of the mars so that the TCS can heat or cool all of the subsystems accordingly. Finally, it inputs commands and signal feeds into the power subsystem for how much power is required for the system at the moment.

This is all done simultaneously as the CDH subsystem receives outputs from the communications, TCS and power subsystems. The outputs it receives from communications include terrain mapping data information. TCS sends a thermal status update to the CDH subsystem to ensure it is operating as required and is allocating the heat based on the thermal data report of the environment and lastly, it receives electrical power from the power subsystem as the onboard computer does need power to function.

The comm subsystem is a very important interface as it is responsible for sending and receiving important data from and to the MRO. This subsystem outputs data and commands received from MRO back to the onboard computer that is part of the CDH subsystem and also sends terrain and surface mapping data back into CDH so the onboard computer can come up with various efficient pathfinding routes and nearest research points for the rover to reach.

It also receives outputs from the TCS as it includes electronics that must be kept under operating temperatures as well as electrical power to operate.

The TCS subsystem is a critical subsystem that will guarantee success of the mission given the extreme thermal environment on Mars. Every subsystem requires either active heating or cooling. For instance, the payload would need both active heating and cooling, this is because the instruments operate nominally under specific operating temperature ranges and hence the instrument would either need cooling or heating.

This subsystem gets inputs from the CDH subsystem and outputs from the power as the TCS does require electrical power to operate.

Finally, the power subsystem lays all the groundwork to ensure all of the other subsystems and components operate and have the required voltage to function. Considering the fact that the rover will be semi-autonomous, everything will be connected to the onboard computer through spacewire that will then be connected to the power distribution unit which will be connected throughout the rover’s main chassis to transfer power to each component.

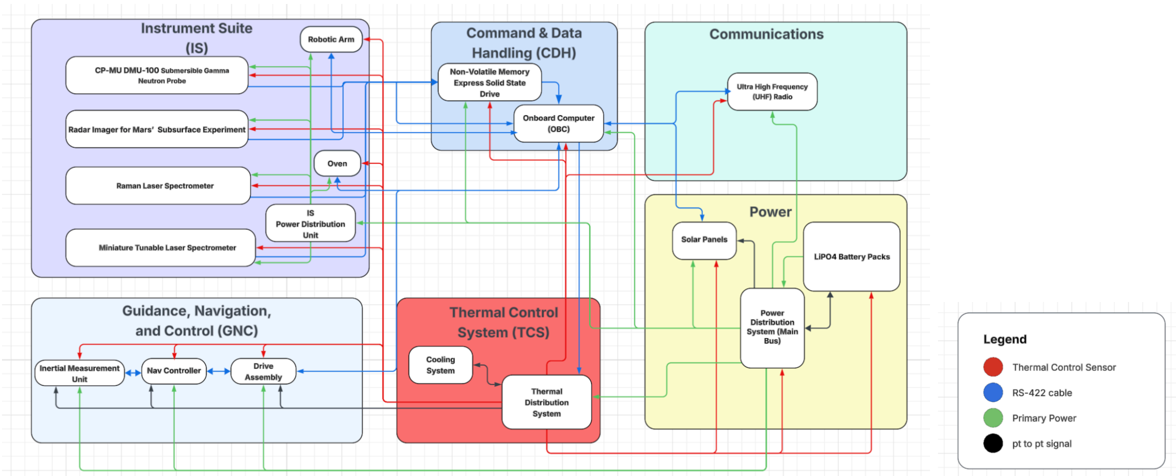


Figure 1.5.8.2: System Block Diagram

The block diagram is a more detailed version of the N^2 chart as it gives a better understanding of how each subsystem is interfacing with each other and what sort of connections they have with each other. At the top left, the instrument suite (IS) is located which contains all of the scientific instruments such as a spectrometer, a neutron probe, a robotic arm, an oven and a power distribution unit. The function of the robotic arm is to collect rock samples from the martian surface which will then be heated up in the oven to collect gas that will then be measured by the spectrophotometer to gather valuable scientific information. These instruments, however, receive primary power from the power distribution unit that is located in the instrument suite. The IS power distribution unit receives primary power from the main power distribution System that receives electrical power from the LiPO4 battery packs which it then sends to the solar panels and every single subsystem component in the system. The power distribution system is also connected to both the solar panels and the LiPO4 battery packs through a pt to pt signal connection, this connection is bi-directional to the battery packs as the signals ensure the PDS knows the status of the battery pack and it then feeds the amount of power available to the solar panels. The TCS subsystem includes a cooling system and a TDS, the TDS is connected to every single subsystem component through thermal sensors that ensure all of the components in the spacecraft are operating nominally. The cooling system is responsible for decreasing the temperature of the spacecraft components, in the case that it gets extremely hot, especially during the daylight hours and that's where the radiators dissipate the heat and cool the systems down. The Cooling system interfaces with the TDS through a pt to pt signal which sends each other the status of both systems ensuring they are maintaining their operational integrity and so that in the case of a failure, one of the systems could be shut down or backup heaters/coolers can take over.

The GNC subsystem houses all of the mechanical subsystems and these consist of the inertial measurement unit, a navigation controller and a drive assembly. All of these subsystems receive their primary power from the PDS and have thermal control sensors on them to ensure they are under nominal temperature and do not excessively heat up to a point where they lose their navigation capability. They are also connected to the TCS through pt to pt signals, sending status and health reports to ensure each of the subsystems are operating properly. They are also connected with each other through RS-422 cables to ensure streamlined data and command execution with each other.

The navigation controller determines and executes the rover's pathfinding capability. It processes navigation commands, plans routes between waypoints, and calculates the necessary steering and speed adjustments to follow the calculated path. The controller uses algorithms to ensure the rover avoids any obstacles, follows smooth paths and maintains stability; it does that by generating target velocities and steering angles that are then sent to the drive system. In autonomous mode, it can also adapt to changing terrain to update the route in real time.

The drive assembly is the mechanical and electrical subsystem that physically moves the rover. It includes motors, wheels, gearboxes and any associated control components. The way this subsystem works is it receives commands from the navigation controller through the RS-422 cable connection on how fast the rover should move and in what direction and then translates these into actual movements the rover performs on the surface. It allows the rover to perform complex maneuvers on challenging terrain for which the boogie rover system was integrated.

The IMU is a sensor package that measures the rover's orientation, acceleration and orientation. This system contains a gyroscope, an accelerometer and a magnetometer. The IMU consists of an accelerometer and gyroscope, which provide real time data on the rover's lateral accelerations and angular velocities. This data is essential for navigation and stability, This allows the rover to utilize dead reckoning techniques and localization algorithms to track its position and orientation even when GNSS systems are unavailable.

1.6 Risk Analysis

Every space mission carries unique risks due to the complexity of spacecraft design and operations. The team is committed to identifying and mitigating these risks wherever possible.

By consensus, the team determined that the most effective methods for identifying risks include thorough failure modes and effect analysis (FMEA), in-depth fault tree analysis (FTA), and expert reviews. These approaches ensure that potential issues are uncovered early and addressed proactively.

To evaluate each risk, amatrix was created that carefully considers both the likelihood of occurrence and the potential consequences. This structured process allows efficient prioritization of risks to allocate resources for the areas of greatest concern.

As the spacecraft advances through the early stages of development and conceptual design, the team has established a robust plan to mitigate risks. This plan is centered around three strategies which include contingency planning, redundancy and weekly risk tracking.

Contingency planning is key to ensuring mission success and dealing with unexpected failures. Each subsystem will be equipped with backup protocols to address both major and minor failures. These protocols will define immediate actions, escalation procedures and recovery steps to minimize the impact of each failure.

Redundancy is embedded throughout the spacecraft’s design to enhance reliability and protect critical functions.This involves incorporating backup components, parallel systems and alternative algorithms for continuous operation of the system. Should a component fail, redundant systems are ready to take over instantly maintaining the component and system’s operational integrity. This approach not only mitigates the risk of the mission due to single failure modes but also provides flexibility for the spacecraft to adapt to any unforeseen challenges it might encounter.

Each subsystem comes with its own risks and they need to be assessed and mitigated early on in the development phase of the spacecraft.

Due to the presence of sharp, embedded rocks on Mars, there is a risk of wheel skin puncture or grouser breakage, resulting in compromised wheel integrity. This could reduce rover mobility or cause immobilization. The mitigation strategy the team will use for this risk is to use thicker wheel treads, adaptive driving algorithms, and regular wheel imaging to monitor and avoid high-risk terrain.

Furthermore, the rover is also at a risk of suspension fatigue due to its large mass and frequent traverses across the Martian surface leading to loss of shock absorption thus increasing risk of mechanical failure during rough terrain navigation. To mitigate this risk, robust materials will be used for suspension components, do periodic load analysis on the OBC, and design redundancy in suspension components.

The thermal control subsystem is also at a risk of failure and as this is a critical subsystem, there are certain risks associated with it. Bearing in mind the extreme temperature swings on Mars, the TCS may fail to maintain optimal temperature exposing electronics to extreme temperature conditions which could cause permanent damage to the electronics and compromise the mission. The way this risk is to be mitigated is to make use of MLI, redundant heaters, and real-time thermal monitoring. To mitigate this risk, backup heaters need to be included in the rover in case one of the heaters malfunction, the other can take over.

The instrumentation subsystem also comes with its risks and as a result of the abundance of the Martian dust, it may infiltrate the instruments and hinder their ability to function and record accurate data. The method to mitigating this risk is to make use of dust seals, protective covers, and periodic instrument cleaning routines.

The other issue is the fact that the instrumentation is at risk of getting misaligned or damaged due to mechanical shock during landing which will lead to inaccurate scientific data or instrument failure. The method to mitigate this risk is to employ shock absorbing mounts and post-landing calibration of the instrumentation to ensure accurate data readings.

LLIKELIHOOD	**5**		8 →	7 → , 11 →		
	4			10 → , 16 →		
	3		9 → , 19 →	15 →	13 →	
	2		15 →	1 → , 2 → , 4 →	14 → , 18 → , 20 → , 3 →	17→ , 6 →
	1				5 →	
		1	**2**	**3**	**4**	**5**
		CONSEQUENCES				

Figure 1.6.1: Risk Matrix

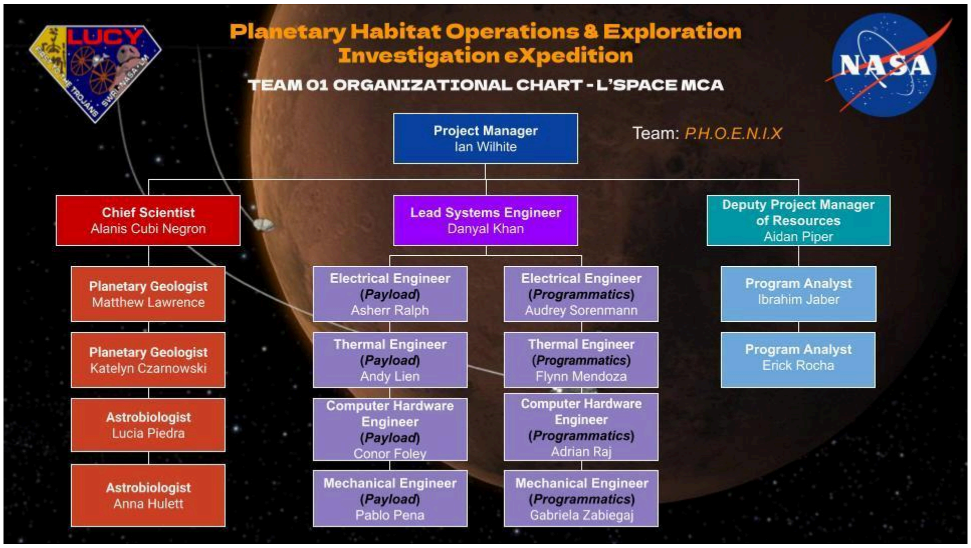
ID	Function	Failure Mode	Effects	Sev	Cause	Occ	Prevention	Det	RPN	Actions	Approach
1	Mechanical Subsystem	Wheel skin puncture or grouser breakage	Reduced mobility or immobilization	3	Sharp, embedded rocks on Mars	2	Thicker wheel treads, adaptive driving algorithms, regular wheel imaging	6	36	The main action is to monitor the damage and adjust operations to avoid hazardous terrain.	Mitigate
2		Suspension fatigue	Loss of shock absorption, increased risk of mechanical failure	3	High mass and frequent traverses	2	Robust suspension materials, periodic load analysis, redundancy	7	42	Reducing speed and avoiding rough terrain can help slow further fatigue.	Research
3		Loss of drive or steering calculator	Reduced mobility, inability to steer or drive one or more wheels	7	Hardware or cable failure, wear over time.	0	Redundant actuators, regular actuator health checks.	4	0	Switching to backup or manual control is the most logical step.	Mitigate
4		Loss of shock absorption capability	Increased transmission of shocks to chassis and instruments	3	Material fatigue, extreme temperature cycling.	0	Use of advanced materials (e.g., shape memory alloys), regular health monitoring.	5	0	Rough terrain should be avoided if this failure persists	Mitigate
5	Thermal Subsystem	Heat Pipe failure	Loss of thermal regulation, risk of overheating or freezing	3	Faulty manufacturing, extreme pressure changes	1	Redundant heat pipes, redundant thermal control systems in the form of heaters	3	9	Rely on electric heaters, plan for alternate cooling systems	Mitigate
6		Multi-layer insulation (MLI) degrades over time.	Increased heat loss, reduced thermal protection.	7	Micrometeoroid impacts, material aging, radiation	3	Use of high-durability MLI, periodic thermal performance checks.	3	63	Operate rover primarily during warm periods away from areas of high radiation	Research
7		Electric Heater Failure	Loss of active thermal regulation, risk of overheating or freezing	8	Material aging, overheating, electric failure	1	Redundant heaters, separate circuitry	1	8	Rely on redundant heaters and heat pipes	Research
8		Thermal sensor Failure	Loss of thermal monitoring, risk a system failure is not reported	7	Material aging, overheating, electric failure	1	Redundant sensors, separate circuitry	7	49	Rely on redundant sensors, develop a software recovery plan	Mitigate

9		RIMFAX signal attenuation	reduce the depth it penetrates and degrade the signal-to-noise ratio	2	Materials like clays or ice-dust mixture which absorb and scatter radar energy	0	Running at its low end frequency of 150-1200 MHz band as low frequencies penetrate better	0	0	Reduce depth objective to 2 m - 4m for clear data and schedule multiple low frequency soundings	Research
10	Payload Subsystem	RLS calibration drift	calibration change throughout image collection	3	temperature swings disrupt calibration due to temperature sensitivity	0	calibration verification before and after each science run	0	0	heating samples to 935 K and cooling them down to 120 K to reserve fully calibrated quantitative measurements for high temperature, and qualitative data for lower temperature	Mitigate
11		Mini TLS overheating risk	overheating, wavelength drift, or shutdown	0	Running over max operating temperature of 20 C	0	only run at temperature between -10 C up to 20 C	0	0	scheduling measurements strictly during nighttime	Mitigate
12		CP-MU DMU-100 freeze risk	instrument shutdown ending data transmission before meeting the one year minimum planned operation	6	Temperature below 0°C	3	Thermal material layer(s) built into instrument housing	3	54	Running testing during the night to heat up internal housing fluid environment as the instrument operates at a minimum of 30°C which is warm	Mitigate
13		Loss of battery capacity or failure to recharge	Reduced operational time, possible mission loss.	9	Repeated charge/discharge cycles, extreme temperatures.	5	Battery health monitoring, thermal management, redundant batteries.	5	225	Utilize a second battery. If second battery is also inoperable, begin powering down noncritical systems to reduce power load	Mitigate
14	Power Subsystem	Electrical short in power distribution	Loss of power to subsystems, reduced redundancy.	8	Dust, material degradation, component failure.	4	Robust insulation, regular voltage monitoring.	6	192	Shut down the system and reboot, if the issue persists then continue to monitor. Perform analysis on systems to ensure they remain within operational capacity.	Mitigate

15		Reduced power generation from solar panels	Insufficient power for operations	7	Dust accumulation, mechanical damage.	7	Dust removal systems.	3	147	Reduce subsystem power usage to redirect remaining power to critical systems. If transmission is impossible, store data for potential future collection	Mitigate
16		Drift in transistors not allowing them to turn on/off	Loss of signals for power distribution	5	Radiation ionizing deposits within technological components	7	Shielding systems, radiation-hardened materials in manufacturing	3	105	Perform maneuvers to shield components from further damage, reduce operation of damaged systems, continue to monitor	Mitigate
17	CDH Subsystem	Processor failure	Loss of command/control, mission halt	10	Hardware defect, radiation, overheating	4	Use radiation-hardened processors, implement thermal control, redundant processors	8	320	Physical Redundancy by having two OBCs	Mitigate
18		Memory corruption	Loss of stored data, erratic behavior	2	Radiation, aging	8	Use ECC memory, regular memory scrubbing, software validation	3	48	Offboard system cleansing processes, rad-hard memory units	Research
19		Data uplink/downlink loss	Loss of communication with ground, inability to send/receive commands	2	Antenna failure, RF interference, ground station issue	8	Antenna redundancy, RF shielding, multiple ground stations	4	64	Autonomous source finding recovery behavior	Research
20		Command errors	Incorrect commands executed, potential for unsafe actions	8	Software bug, memory corruption	4	Command validation, memory error detection, queue integrity checks	2	64	Increase detection using verification codes	Research

Figure 1.6.2: Risk Analysis Matrix

1.7 Programmatics



1.7.1 Organizational Chart

Figure 1.7.1: Organizational Chart

Current workload distribution follows a similar structure to past deliverables where the deliverable is split into broad sections which are then assigned to subteams by the project manager. Subteam leads then assign individual tasks and sections to individuals or small groups on their subteam. By tracking progress with a shared spreadsheet, the team reinforces mutual accountability while setting a clear pace for the completion of the deliverable. An updated organizational chart, seen in Figure 1.7.1, reflects the departure of dedicated mission assurance specialists. In response to changes in personnel, all team members are now expected to document and mitigate risks as they perform their expressly assigned tasks while the DPMR coordinates high-level tracking and communication regarding risks.

The decision-making methodology of the team remains largely unchanged from its original iteration, focusing on input from the entire team before the leadership makes a decision. This process proved effective following the addition of a separate science experiment payload and the subsequently associated descope. As the decision was not time-sensitive, the team was able to discuss the options regarding how to incorporate the addon and the benefits and drawbacks of each approach. By taking in feedback from all members, this approach led to success in a real-world environment. After considering the feedback, team leads and the project manager selected an external mounting option.

Moving forward, there are a few areas where the team can improve. Firstly, by implementing a lightweight system of informal progress check-ins, whether it be at the beginning of each subteam's weekly meeting or through some asynchronous upload of a research summary, trade study, or section drafts, the team reduces procrastination without overburdening members. Secondly, reinforcing active usage of the task tracker will ensure everyone can get a status update on the team's progress at just a glance.

In recent weeks, the team has faced some challenges ranging from personnel turnover to the sudden announcement of a third-party spacecraft being integrated into the mission. The team distributed the responsibility of mission assurance across the entire team. Key impacts were identified across all three subteams and a cohesive plan to move forward within the new design constraints and considerations was devised.

1.7.2 Schedule Basis of Estimate

The Schedule Basis of Estimate centers around the non-negotiable ground rules, crucial assumptions, and primary drivers regarding the overall scheduling of the mission. Spanning project phases C to F, beginning with the submission of the PDR and subsequent KDP C and ending with the completion of the mission lifecycle, this basis aims to convey the surrounding context and scope for the following estimation, highlighting potential risks and justifying crucial assumptions in regards to each phase of the mission in regards to the schedule [29]. The team is interested in high-level system objectives and ensuring that all subsystems are designed, integrated, and tested to meet mission requirements for science return, reliability, and launch readiness.

Ground Rules

The scope of this estimation begins with PDR submission (August 18, 2025) before continuing along the standard NASA mission life cycle sectioned by a few notable hard dates. In order to leverage the 2029 Mars launch window, full system integration, corresponding to Phase D just prior to launch, must be completed by the System Integration Date of October 1, 2029 to be closely followed by the successful completion of KDP E prior to the Launch Readiness Date of December 1, 2029 [10]. These constraints reflect the narrow launch window to Mars and the necessary integration period before launch.

Assumptions

Fiscal Year (Oct 1–Sept 30) is used instead of calendar year to align with NASA and federal budget cycles. Phase C is projected to begin alongside FY 2026.

Fabrication begins following the completion of the CDR Schedule margin to address any unknown issues that may arise, particularly with phases C and D prior to launch, is going to vary based on each of the specific tasks slated for completion during the phase. With the addition of a science experiment designed by a third party, additional time is likely to be needed for testing and successful full-scale integration via external mounting to the PHOENIX platform. Entry Descent Landing system, and transit will be handled by external contractors. Operating time on the Martian surface is a baseline floor and not accounting for any mission extensions dependent on system integrity [10].

Drivers

Major drivers for this mission will be the time allotment for phases C and D due to the scope and intensity of the work undertaken in those phases [10]. Phase C encompasses the maturation of preliminary designs in regards to both hardware and software, flushing out strategies for implementation, integration, verification, and validation, and fabrication or acquisition of hardware. Identification and assessment of mission risks continues to check any unplanned increases in cost, scope creep, etc in the leadup to launch [29]. Due to the complexity of the tasks and major deliverables such as the CDR and SIR coupled with the planned onboarding of new technicians at the initialization of the phase, the phase is projected to span all of FY 2026 and 2027 and part of FY 2028 [10]. Phase D, accounting for full system integration, must now consider additional challenges brought about by the third-party science payload attached to PHOENIX while ensuring the mission remains on task to meet key deadlines as defined above.

1.7.3 Schedule Estimate

This schedule estimation is derived from previous NASA endeavors to the Martian surface. Pathfinder and its Sojourner rover, despite being a microrover with a budget around 330-340 million dollars adjusted for inflation, show an example of a discovery-class mission with a similarly aggressive timeline as PHOENIX [17, 47]. Twin rovers Spirit and Opportunity represent individual payloads more akin to the scale of this mission's rover with masses of 185 kilograms and smaller in volume 1.5x2.3x1.6m [47, 32]. InSight serves as the most recent example of a Martian mission that best aligns with the budget of this mission and a good basis for determining the longevity of more modern flight-proven hardware– heritage sourced from the 2008 Phoenix Lander [33]. Each phase, delineated by the KDP at its closure, will further be broken down based deliverables requiring a presentation to the SRB, launch, and arrival.

Phase C will consist of a period of 28 months dedicated to finalizing system documentation and begin fabrication of hardware and coding any necessary software [33]. Technicians, overseen on-site and on boarded by the LSE, will assist the team in fabricating engineering and flight units. Phase C starts with the completion of KDP C tentatively set for the beginning of FY 2026 on October 1st, 2025 and lasting for the next 30 months, terminating with the end of the first quarter of FY 2028. Key tasks are expected to last 27 months with a 3 month margin of roughly 11 percent when rounded to the nearest day. Successful completion of the CDR, projected to be 14 months into the phase, based around a margin of one month to address any issues with finalization of documentation alongside personnel onboarding challenges that may arise, will allow the team to start fabrication. Similarly, InSight passed its CDR in mid-May 2014 before entering integration and testing at the end of May 2015 [35, 36]. Occurring in March of 2028, the System Integration Review (SIR) serves as the culmination of Phase C overseen by the SRB just prior to the start of Phase D [33].

Phase D consists of full system-level integration and testing to be successfully completed by the system integration date (SID) of October 1, 2029 alongside the start of FY 2029 [10]. Lasting from April 1, 2028 to the SID, Phase D starts with clearing KDP D before transitioning into full system-wide testing and integration [33]. Compared to the typical timeline for Phase D, this mission has additional months allotted to crucial testing. InSight serves as a prime example of a risk that this mission has to address. InSight missed its 2016 launch window due to a French-made seismometer being delayed due to a technical failure, leading to an additional 150 million dollars being spent on the mission [37]. Due to the third-party science experiment, PHOENIX faces similar risk to InSight but without the flexibility of delay, requiring more conservative testing and integration margins. The mission-critical risk necessitates a more cautious approach to testing and integration with a higher margin than likely needed out of an abundance of caution. A period of 18 months is planned for Phase D’s integration and testing, with realistic expectations of completion in 15 months but a reserve of three months, which would align it closer to the testing timeline for Spirit and Opportunity alongside Pathfinder [17, 39]. Following the FRR on October 1, 2029, the mission will be cleared for the December 1 launch assuming approval and satisfaction of KDP E.

Phase E consists of the mission cruise, entry, descent, and landing (EDL), and surface operations [33]. Assuming a Homann transfer is used, an average travel time is around 9 months [78]. The science instrumentation used on the rover is slated to at a minimum deliver trustworthy data from the surface of Mars for at least a year based on preliminary analysis from the team’s scientists. During this time, the rover would complete the driving scientific procedures and acquire data to further the construction of an answer regarding its driving questions regarding the potential groundwork for human habitation with subsurface ice. At this stage, mission personnel are reduced to a bare minimum to monitor the rover’s progress and address any issues that may jeopardize the completion of the mission. Mission extensions may be granted based on the overall integrity of the system as a whole to maximize the extracted information. A plan for decommissioning through the decommissioning review would be formulated to be implemented in closeout regardless of any extensions.

Phase F consists of the decommissioning of the system alongside the analysis and storage of data returned from the mission for future usage [33]. As with other rover missions, the exact timeline for this phase is unknown and can develop rapidly if critical mission systems fail. Lastly, the final report is drafted as crucial lessons learned are documented for future reference [33].

1.7.4 Cost Basis of Estimate

The Budget Basis of Estimate (BoE) for P.H.O.E.N.I.X defines the ground rules, assumptions, and cost drivers used to develop the preliminary cost estimate for phases C through F of the mission's life cycle. The purpose of the BoE is to clearly define how cost estimates were developed from the rules, assumptions, and drivers.

Ground Rules

A $450M cost limit is established specifically for the Rover System, encompassing all expected mission costs including personnel, travel, outreach, hardware, testing, direct costs, as well as cost margins of safety [10],. The BoE only targets Phase C through F. Costs will be estimated primarily using parametric models. It is assumed that these tools provide an accurate reflection of the anticipated cost. These estimates are then aggregated into a budget template that is derived from the Lucy Mission Budget.

Assumptions

A constant 2.6% yearly compounding inflation rate is assumed to estimate the budget across the entire mission's lifecycle [61]. It is assumed that personnel turnover will be minimal. In cases where turnover does occur, replacement costs are expected to be negligible and are covered by the total cost margin. Outreach costs relate to the team's effort in increasing public awareness of P.H.O.E.N.I.X and the impact that it will have on the scientific community and the end science goals. A graphic designer and a social media specialist will be hired full time for the duration of the mission to develop, publish, and distribute outreach content. Personnel travel costs will be estimated through the City Pair Program for airfare. FedRoom for lodging, and per diem reimbursement for meals and rentals. Tests are conducted at relevant NASA centers across the country, and launch takes place at Cape Canaveral, Florida. Key personnel will be flown in to oversee and conduct in-person testing of relevant components and subsystems with rental cars, lodging, and meals priced out using the aforementioned resources [61] [62].

Drivers

The primary cost drivers for P.H.O.E.N.I.X include items such as scope changes or descopes, which can shift the required designs and greatly impact system costs. External government policies such as changes in import tariffs may introduce some budget uncertainty, especially for foreign-sourced hardware. Lastly, any unforeseen engineering testing failures may lead to cost inflation due to vendor lead times and/or potential redesigns. A comprehensive budget for P.H.O.E.N.I.X will be developed as it advances towards the PDR. The full budget will include breakdowns of costs for each phase of the mission as well as a per-item cost breakdown. To account for any delays, uncertainties, and unexpected problems, a 30% total cost margin will be applied to the budget totals. This margin aligns with the standard 70% confidence level in lifecycle cost estimates at the PDR stage [29].

1.7.5 Cost Estimate

The total estimated cost for P.H.O.E.N.I.X is approximately 405 million dollars, as detailed in the Figure 1.7.5.1. This cost encompasses the full scope of the mission and includes personnel, travel, outreach, facilities, and other direct costs. The budget is organized into four primary categories: Personnel, Travel, Outreach, and Direct Costs. Personnel is broken down further into six roles: science, engineering, technical, administrative, management, and outreach. Staffing allocations were planned per mission phase and are summarized in the top portion of Figure 1.7.5.1 in blue. The baseline team assumes a minimal configuration with additional support staff only during peak build and integration periods. 30 additional support staff were added during early phases, split ⅔ technicians and ⅓ engineers or scientists. Salaries are assumed fixed except for inflation rates with a 28% benefits rate applied [10].

Costs for travel were estimated using federal resources including the City Pair Program for airfare, FedRooms for lodging, and standard GSA per diem rates for meals and rentals. Outreach costs include the full-time employment of a graphic designer and a social media specialist, whose salaries were derived using U.S. Bureau of Labor Statistics data [61]. The dominant costs for P.H.O.E.N.I.X are the direct mission costs. These direct costs include items such as the Mechanical Subsystem and Science Instrumentation, calculated using Cost Estimating Relationships (CERs) from the MCCET. These CERs are from the NCIM Version 9c. Inputs to the calculations are mass, power, and design life where applicable [29].

The final budget incorporates a 30% Total Cost Margin to accommodate unexpected costs, ensuring a 70% confidence level in budget sufficiency at the Preliminary Design Review stage. An estimated budget can be seen in Figure 1.7.5.1, covering personnel, outreach, travel, and spacecraft related expenses.

Active Microwave Instruments	$1,244 * TotalMass^{0.36} * TotalMaxPwr^{0.50}$
RIMFAX - Mass: 3kg - Power: 5 to 10 watts	$1,244 * 3^{0.36} * 10^{0.50} = 5842.302394$ Using this CER value, plugged into MCCET to calculate cost breakdown per phase.

Personnel Phase Allotment Table

	Phase C	Phase C	Phase C-D	Phase D	Phase E	Phase F
# People on Team	FY 1	FY 2	FY 3	FY 4	FY 5	FY 6
Science Personnel:	10	10	5	5	15	15
Engineering Personnel:	15	15	15	15	10	10
Technicians:	30	30	30	30	0	0
Administration Personnel:	5	5	5	5	5	5
Management Personnel:	4	4	4	4	4	4
Total Personnel	64	64	59	59	34	34

P.H.O.E.N.I.X Preliminary Budget

Mission Phase	Phase C	Phase C	Phase C-D	Phase D	Phase E	Phase F	
Year	**Year 1**	**Year 2**	**Year 3**	**Year 4**	**Year 5**	**Year 6**	**Cumulative Total**
PERSONNEL							
Science Personnel	$ 800,000	$ 820,800	$ 420,800	$ 431,200	$ 1,324,800	$ 1,356,000	$ 5,153,600
Engineering Personnel	$ 1,200,000	$ 1,231,200	$ 1,262,400	$ 1,293,600	$ 883,200	$ 904,000	$ 6,774,400
Technicians	$ 1,800,000	$ 1,846,800	$ 1,893,600	$ 1,940,400	$ -	$ -	$ 7,480,800
Administration Personnel	$ 300,000	$ 307,800	$ 315,600	$ 323,400	$ 331,200	$ 339,000	$ 1,917,000
Project Management	$ 480,000	$ 492,480	$ 504,960	$ 517,440	$ 529,920	$ 542,400	$ 3,067,200
Total Salaries	$ 4,580,000	$ 4,699,080	$ 4,397,360	$ 4,506,040	$ 3,069,120	$ 3,141,400	**$ 24,393,000**
Total ERE	$ 1,278,278	$ 1,311,513	$ 1,227,303	$ 1,257,636	$ 856,591	$ 876,765	**$ 6,808,086**
Personnel Margin	$ 1,464,570	$ 1,502,648	$ 1,406,166	$ 1,440,919	$ 981,428	$ 1,004,541	**$ 7,800,272**
TOTAL PERSONNEL	$ 7,322,848	$ 7,708,586	$ 7,396,432	$ 7,766,553	$ 5,417,482	$ 5,675,658	$ 41,287,558
TRAVEL							
Total Flights Cost	$ 4,000	$ 3,000	$ 5,500	$ 6,000	$ 5,000	$ 17,000	**$ 40,500**
Total Hotel Cost	$ 4,960	$ 3,720	$ 6,820	$ 7,440	$ 6,200	$ 21,080	**$ 50,220**
Total Transportation Cost	$ 667	$ 500	$ 917	$ 1,000	$ 833	$ 3,000	**$ 6,917**
Total Per Diem Cost	$ 2,196	$ 1,647	$ 3,020	$ 3,294	$ 2,745	$ 9,333	**$ 22,235**
Travel Margin	$ 5,000	$ 5,000	$ 5,000	$ 5,000	$ 5,000	$ 5,000	**$ 30,000**
Total Travel Costs	$ 16,823	$ 14,228	$ 22,362	$ 24,507	$ 21,835	$ 62,617	**$ 162,372**
OUTREACH							
Total Outreach Materials	$ 10,000	$ 12,000	$ 15,000	$ 18,000	$ 20,000	$ 25,000	**$ 100,000**
Total Outreach Venue Costs	$ 5,000	$ 6,000	$ 8,000	$ 10,000	$ 12,000	$ 15,000	**$ 56,000**
Total Outreach Travel Costs	$ 7,000	$ 8,000	$ 10,000	$ 12,000	$ 15,000	$ 18,000	**$ 70,000**
Total Outreach Services Costs	$ 10,000	$ 12,000	$ 15,000	$ 20,000	$ 25,000	$ 30,000	**$ 112,000**
Total Outreach Personnel Costs	$ 143,000	$ 143,000	$ 143,000	$ 143,000	$ 143,000	$ 143,000	**$ 858,000**
Outreach Margin	$ 17,500	$ 18,100	$ 19,100	$ 20,300	$ 21,500	$ 23,100	**$ 119,600**
Total Outreach Costs	$ 192,500	$ 204,277	$ 221,025	$ 240,717	$ 261,096	$ 287,133	**$ 1,406,748**
DIRECT COSTS							
Mechanical Subsystem	$ 19,106,400	$ 19,106,400	$ 30,000,400	$ 21,788,000	$ 2,180,000	$ 2,180,000	**$ 94,361,200**
Power Subsystem	$ 12,403,200	$ 12,403,200	$ 10,199,200	$ 10,868,000	$ 1,088,000	$ 1,088,000	**$ 48,049,600**
Thermal Control Subsystem	$ 1,003,200	$ 1,003,200	$ 6,226,933	$ 1,144,000	$ 228,000	$ 228,000	**$ 9,833,333**
Comms & Data Handling Subsystem	$ 5,449,200	$ 5,449,200	$ 5,831,600	$ 6,406,000	$ 622,000	$ 622,000	**$ 24,380,000**
Guidance, Nav, & Control Subsystem	$ 5,449,200	$ 5,449,200	$ 5,831,600	$ 6,406,000	$ 622,000	$ 622,000	**$ 24,380,000**
Science Instrumentation	$ 5,586,000	$ 5,586,000	$ 5,978,000	$ 6,370,000	$ 490,000	$ 490,000	**$ 24,500,000**
Spacecraft Cost Margin	$ 9,799,440	$ 9,799,440	$ 12,813,547	$ 10,596,400	$ 1,046,000	$ 1,046,000	**$ 45,100,827**
Total Spacecraft Direct Costs	$ 58,796,640	$ 60,325,353	$ 80,879,107	$ 68,537,515	$ 6,928,704	$ 7,091,880	**$ 282,559,198**
Manufacturing Facility Cost	$ 500,000	$ 500,000	$ 1,000,000	$ 1,500,000	$ 500,000	$ 250,000	**$ 4,250,000**
Test Facility Cost	$ 7,250,400	$ 7,250,400	$ 12,954,000	$ 11,544,000	$ 296,000	$ 296,000	**$ 39,590,800**
Facility Cost Margin	$ 1,550,080	$ 1,550,080	$ 2,790,800	$ 2,608,800	$ 159,200	$ 109,200	**$ 8,768,160**
Total Facilities Costs	$ 9,300,480	$ 9,542,292	$ 17,615,530	$ 16,873,718	$ 1,054,541	$ 740,376	**$ 55,126,937**
Total Direct Costs	$ 68,097,120	$ 69,867,645	$ 98,494,636	$ 85,411,234	$ 7,983,245	$ 7,832,256	**$ 337,686,136**
Total MTDC	$ 58,796,640	$ 60,325,353	$ 80,879,107	$ 68,537,515	$ 6,928,704	$ 7,091,880	**$ 282,559,198**
FINAL COST CALCULATIONS							
Total F&A	$ 4,899,720	$ 5,052,591	$ 6,806,556	$ 5,794,112	$ 588,270	$ 604,588	**$ 23,745,837**
Total Projected Cost	$ 80,529,011	$ 82,847,326	$ 112,941,012	$ 99,237,123	$ 14,271,928	$ 14,462,251	**$ 404,288,651**
Total Cost Margin	$ 12,836,590 15.9%	$ 12,875,268 15.5%	$ 17,034,612 15.1%	$ 14,671,419 14.8%	$ 2,213,128 15.5%	$ 2,187,841 15.1%	**$ 61,818,858**
Total Project Cost	$ 80,529,011	$ 82,847,326	$ 112,941,012	$ 99,237,123	$ 14,271,928	$ 14,462,251	**$ 404,288,651**

F&A %	10%	10%	10%	10%	10%	10%
ERE - Staff	28%	28%	28%	28%	28%	28%
Inflation Rate	0.0%	2.6%	5.2%	7.8%	10.4%	13.0%
Science Personnel Salary	$ 80,000					
Engineering Personnel Salary	$ 80,000					
Technicians Salary	$ 60,000					
Administration Personnel Salary	$ 60,000					
Project Management Salary	$ 120,000		Graphic Designer	Social Media Specialist		
Outreach Personnel Combined Salary	$ 143,000		65000	78000		

P.H.O.E.N.I.X Budget Breakdown

Personnel	$ 41,287,558
Travel	$ 162,372
Outreach	$ 1,406,748
Spacecraft Direct	$ 282,559,198
Total Facilities Co	$ 55,126,937

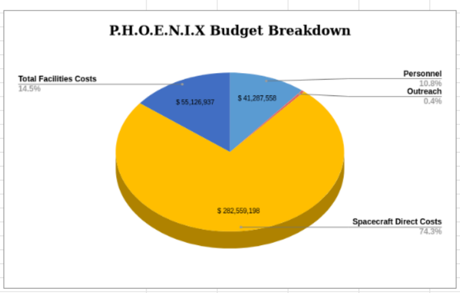


Figure 1.7.5.1: P.H.O.E.N.I.X Preliminary Budget and Pie Chart

1.7.6 Change Control

Once the mission has reached production, any engineering changes made will need to be formulated into an ECR (Engineering Change Request) and reviewed by a CCB (Change Control Board) [30]. The CCB will be comprised of leaders from the engineering team, science team, and programmatics team to ensure that all mission aspects of the change are considered.The team will be using a PLM package such as Siemens Teamcenter or PTC Windchill to retain version control for any CAD changes. Any engineering CAD changes will be drafted in the PLM, and attached to the ECR for review.

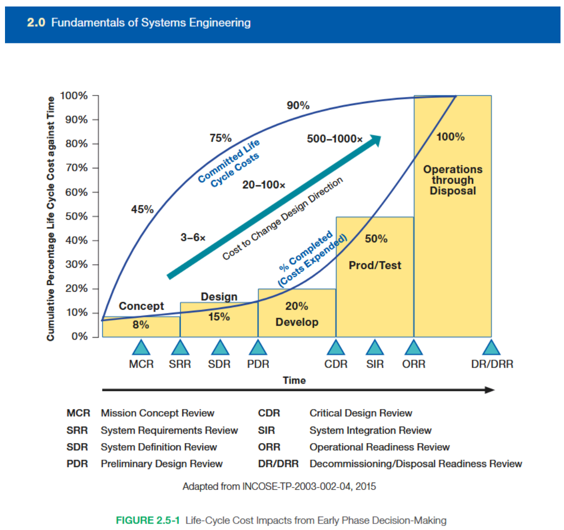


Figure 1.7.6.1: Life Cycle Cost Model

Note that as the mission reaches production, the cost to change the design direction is increased massively [29]. Hence avoiding engineering design changes at production is essential. Any design scope changes will need to go through a CCB as well as a cost estimate of the design change will need to be provided.Major changes suggested by SRBs will be tracked and their implementation progress documented on a team-wide tracker with each one being given its own specific reference number, allowing for feedback to be directly incorporated into further mission deliverables.

1.8 Conclusion

Through the SRR, P.H.O.E.N.I.X has matured the mission concept by updating all science objectives, finalizing scientific instrumentation, baselining all engineering components through trade studies, identifying potential risks to the mission at a subsystem level, providing an updated basis for schedule and overall mission budget based on NASA best practices. As the team transitions toward the Mission Definition Review (MDR) and the closeout of Phase A, the focus will shift to completing remaining tasks including resolving outstanding TBD/TBR items, expanding risk mitigation strategies, and improving the fidelity of the cost model and integrated schedule. This upcoming phase will solidify system definitions, interfaces, and integration strategies, laying the groundwork for a robust Preliminary Design Review.

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Declaration of Generative AI and AI-Assisted Technologies in the Writing Process

During the preparation of this document, the team used OpenAI's ChatGPT to assist with re-phrasing and content refinement. The tool was used to rephrase content for conciseness and completeness.

During literature reviews, the team utilized Google's AI in Search to identify cited resources. The team thoroughly [read all sources]

During the preparation of this document, the team utilized online translation tools including Google Translate to aid in writing and editing.

After using various tools, the team reviewed and edited all content to ensure accuracy, original contribution, and technical fidelity. Team 01 takes full responsibility for the content of this deliverable.

Appendix

Changes	Description
MCR-RFA-1, Section 1.2	The measurement observables must directly relate to the physical parameters with which the STM was addressed and a CRF was filled due to changes in both our human exploration goal objectives to meet this RFA.
MCR-RFA-2, Section 1.3	A thorough description of each region of interest and how they fulfill the team's science objectives must be provided, and was also addressed by defining the region of interest.
CRF - Science objective 1# of HBS-1LM (waiting on approval)	Science objective 1# of HBS-1LM: The team is requesting a minor alteration to the science objective and add clarification within the STM from “various samples” (exact quantity was originally unspecified) to one single sample. Science objective 1# of HBS-1LM - STM observable: Change data transmission interval from “monthly” to “weekly”.
CRF - Science objective 2# of HBS-1LM (waiting on approval)	The second objective of HBS-1LM’s main purpose still remains the same, but the measurement approach to provide quantitative data had to change.

Table A.1 (Mission Change Log)

Leg Material						
Criteria	**Explanation**	**Grade**	**Weight**	**Aluminum 6061**	**Titanium**	**Carbon Fiber (CF) Composite**
Density	Density (g/cm^3 or kg/m^3)	1 = > 4.5 g/cm^3 3 = < 3.5 g/cm^3 5 = < 1.5 g/cm^3	25%	3	1	4
Load-Bearing	Shear Strength (MPa) – Point at which permanent deformation begins	1 = > 300 MPa 3 = > 200 MPa 5 = < 100 MPa	25%	3	1	4
Shock Absorption	Fracture Toughness ($MPa/m^{1/2}$) – Total energy absorbed before failure	1 = > 50 $MPa/m^{1/2}$ 3 = > 20 $MPa/m^{1/2}$ 5 = < 10 $MPa/m^{1/2}$	25%	2	1	3
Thermal Performance	Coefficient of Thermal Expansion (μm/m·K)	1 = < 5 μm/m-°C 3 = < 10 μm/m-°C 5 = > 20 μm/m-°C	25%	5	3	1
		TOTALS:	**100%**	43.75%	87.50%	50.00%

Table A.2 (Leg Material Trade Study)

Chassis Material						
Criteria	**Explanation**	**Grade**	**Weight**	**Aluminum 6061**	**CF Composite**	**Stainless Steel**
Density	Density (g/cm³ or kg/m³)	1 = < 3 g/cm³ 3 = < 4.5 g/cm³ 5 = > 6 g/cm³	30%	1	1	5
Load-Bearing	Shear Strength (MPa) – Point at which permanent deformation begins	1 = > 300 MPa 3 = > 200 MPa 5 = < 100 MPa	20%	3	4	1
Shock Absorption	Fracture Toughness (MPa/m½) – Total energy absorbed before failure	1 = > 25 MPa/m½ 3 = > 15 MPa/m½ 5 = < 10 MPa/m½	30%	1	4	1
Thermal Performance	Coefficient of Thermal Expansion (μm/m·K)	1 = < 5 μm/m-°C 3 = < 10 μm/m-°C 5 = > 20 μm/m-°C	20%	5	1	4
		TOTALS:	**100%**	70.00%	62.50%	55.00%

Table A.3 (Chassis Material Trade Study)

Wheel Material						
Criteria	**Explanation**	**Grade**	**Weight**	**Aluminum**	**Titanium**	**Fiberglass**
Density	Density (g/cm³ or kg/m³)	1 = < 3 g/cm³ 3 = < 4.5 g/cm³ 5 = > 6 g/cm³	25%	1	4	1
Load-Bearing	Shear Strength (MPa) – Point at which permanent deformation begins	1 = > 150 MPa 3 = > 100 MPa 5 = < 50 MPa	25%	1	1	5
Shock Absorption	Fracture Toughness (MPa/m½) – Total energy absorbed before failure	1 = > 220 MPa 3 = > 120 5 = < 20 MPa	25%	2	1	4
Thermal Performance	Coefficient of Thermal Expansion (μm/m·K)	1 = < 10 μm/m-°C 3 = > 30 μm/m-°C 5 = > 50 μm/m-°C	25%	2	1	1
		TOTALS:	**100%**	87.50%	81.25%	56.25%

Table A.4 (Wheel Material Trade Study)

Wheel Tread						
Criteria	**Explanation**	**Grade**	**Weight**	**Symmetrical**	**Asymmetrical**	**Directional**
Wear	Abrasion resistance of the tread	1 = high, 3 = medium 5 = Fail	30%	2	3	5
Grip	Shows efficiency of wheel-ground interaction	1 = high, 3 = medium 5 = Fail	30%	3	1	1
Reliability	Fatigue Life (Cycles to crack initiation or tread breakoff)	1 = high, 3 = medium 5 = Fail	40%	1	3	5
		TOTALS:	**100%**	77.50%	65.00%	30.00%

Table A.5 (Wheel Tread Trade Study)

Dust Mitigation Methods						
Criteria	**Explanation**	**Grade**	**Weight**	**Electrodynamic Dust Shield (EDS)**	**Vibration-Based Cleaning (Chladni Patterns)**	**Passive Coatings (Hydrophobic/Anti-Static)**
Effectiveness	Ability to remove or prevent dust, maintain panel output over time	1 = high percentage dust removal; minimal power loss 3 = medium percentage dust removal; moderate power loss 5 = low percentage dust removal; significant power loss	35%	1	3	3
Power Consumption	Additional power required for operation (if any)	1 = low percentage of array output 3 = moderate percentage of array output 5 = high percentage of array output	20%	1	3	1
Reliability/ Robustness	Performance over mission duration, resistance to failure or degradation	1 = proven in Mars-like conditions; no moving parts 3 = some moving parts or limited Mars testing 5 = high failure risk; unproven or complex	25%	1	3	1
Mass & Integration	Added mass and complexity to the rover system	1 = minimal integration impact 3 = moderate integration effort 5 = major integration or design changes	20%	1	3	1
		TOTALS:	**100%**	100.00%	50.00%	82.50%

Table A.6 (Dust Mitigation Methods Trade Study)

Solar Panel Deployment						
Criteria	**Explanation**	**Grade**	**Weight**	**Mechanical Hinge**	**Ultraflex**	**ROSA (Roll-Out Solar Array)**
Mass	Minimize launch mass (kg)	1 = optimal for launch constraints 3 = acceptable for mission parameters 5 = significantly impacts payload margin	25%	4	2	1
Stowed Volume	Minimize stowed volume (m^3)	1 = minimal impact on rover design 3 = moderate accommodation required 5 = substantial accommodation required	30%	5	2	1
Reliability	Must deploy successfully on Mars	1 = extensive flight heritage 3 = some flight heritage 5 = limited testing or heritage	30%	1	3	2
Complexity	Lower complexity = less risk, easier integration	1 = simple mechanism; few moving parts; minimal deployment steps 3 = moderate complexity; manageable number of failure points 5 = high complexity; numerous moving parts; multiple failure points	15%	2	3	2
		TOTALS:	**100%**	47.50%	63.75%	88.75%

Table A.7 (Solar Panel Deployment Trade Study)

PV Cell Type						
Criteria	**Explanation**	**Grade**	**Weight**	**Spectrolab XTE-SF**	**AZUR SPACE 3G30C**	**AZUR SPACE 4G32C (Quad-Junction)**
BOL Efficiency	Measures the initial energy conversion efficiency of the PV cell	1 = ≥32% 3 = 29%-31.9% 5 = <29%	15%	1	3	1
EOL Efficiency	Efficiency after expected mission degradation	1 = ≥28% 3 = 26−27.9% 5 = <26%	35%	3	3	1
Radiation Resistance	Ability to withstand Mars radiation with minimal performance loss	1 = <5% power loss 3 = 5−10% power loss 5 = >10% power loss	20%	3	3	1
Specific Power	Power output per unit mass (W/g)	1 = >370 W/g 3 = 300-369 W/g 5 = <300 W/g	30%	3	3	1
		TOTALS:	**100%**	57.50%	50.00%	100.00%

Table A.8 (PV Cell Type Trade Study)

Power Storage and Battery System						
Criteria	**Explanation**	**Grade**	**Weight**	**Li-ion NMC 18650 (COTS)**	**LiFePO4**	**LTO (Low Temp)**
Energy Density	Energy produced per kg	1= >200 Wh/kg 3= 200-100 Wh/kg 5= <100 Wh/kg	30%	1	2	4
Temperature Tolerance	Function capacity during Martian weather events and Sol cycles	1= -30°C to +60°C 3= -10°C to +60 5= >0°C only	30%	4	2	1
Cycle Life	Number of charges available before battery capacity depletes	1= >2000 cycles 3= >1000 cycles 5= >500 cycles	25%	3	1	1
Mass	Amount of mass utilized by the individual battery	1= compact 3 = less compact 5= bulky	15%	1	2	3
		TOTALS:	**100%**	65.00%	81.25%	70.00%

Table A.9 (Power Storage and Battery System Trade Study)

Isolated Power System						
Criteria	**Explanation**	**Grade**	**Weight**	**DC-DC Converter with Galvanic Isolation**	**Transformer-Based AC-DC Isolation**	**Battery Pack with Isolated Charger**
Voltage Regulation	Ability to keep output voltage stable under all operating conditions	1 = <2% ripple, fast response, high stability 3 = 2–5% ripple, some fluctuation 5 = >5% ripple, unstable under load changes	25%	1	2	1
Isolation Quality	Degree of electrical separation from the main rover power bus	1 = full isolation, >1 kV withstand, minimal leakage 3 = partial isolation, moderate leakage 5 = no isolation, high leakage	25%	1	1	1
EMI Risk	Resistance to interference from other rover systems	1 = excellent filtering, minimal EMI/EMC risk 3 = moderate filtering 5 = poor filtering, high risk	20%	1	3	2
Integration Complexity	Effort and resources needed to install and connect the power system	1 = minimal added mass & volume, easy to integrate 3 = moderate mass/volume, some integration effort 5 = high mass/volume, complex integration	15%	1	3	2
Reliability	Likelihood of consistent, failure-free operation over the mission	1 = proven in spaceflight, minimal failure modes 3 = some heritage, moderate risk 5 = low heritage	10%	1	3	2
Power Efficiency	Power percentage delivered to the instrument	1 = >90% 3 = 80–90% 5 = <80%	5%	1	3	2
		TOTALS:	**100%**	100.00%	79.17%	91.67%

Table A.10 (Isolated Power System Trade Study)

Onboard Computer (OBC)							
Criteria	**Explanation**	**Grade**	**Weight**	**RAD750**	**RAD5545**	**Cobham GR712RC**	**Pumpkin OBC (CubeSat Kit)**
Processing Performance	Determines capacity to run flight software, autonomy, and instrument control.	1 = >1GHz, multi-core 2 = 500–1000MHz 3 = 200–500MHz 4 = 100–200MHz 5 = <100MHz	30%	3	1	4	5
Radiation Tolerance	Critical for deep space and Mars missions to avoid data corruption or total system failure.	1 = >100krad, SEE hardened 2 = 50–100krad 3 = 20–50krad 4 = 5–20krad 5 = <5krad	25%	1	1	2	5
Memory Capacity	Larger memory supports buffering of science data and autonomous ops.	1 = >2GB 2 = 1–2GB 3 = 512MB–1GB 4 = 128–512MB 5 = <128MB	20%	2	1	4	5
Mass & Power Efficiency	Important for small spacecraft; lower weight and power use extends mission life.	1 = <2W and <200g 2 = 2–5W and <500g 3 = 5–10W or ~1kg 4 = 10–20W or 1–2kg 5 = >20W or >2kg	15%	4	3	3	1
Flight Heritage / TRL	Indicates reliability and ease of integration; flight-proven systems reduce risk.	1 = TRL 9, >10 missions 2 = TRL 8–9, 3–10 missions 3 = TRL 7–8, 1–2 missions 4 = TRL 5–6, tested 5 = TRL <5, lab only	10%	1	2	3	3
		TOTALS:	**100%**	68.75%	90.00%	43.75%	20.00%

Table A.11 (Onboard Computer Trade Study)

Data Storage						
Criteria	**Explanation**	**Grade**	**Weight**	**Micro Secure Digital (SD) Cards**	**Embedded MultiMediaCard (eMMC)**	**Non-Volatile Memory Express Solid State Drive (NVMe SSD)**
Data Storage Capacity	Higher capacity allows storing more science data, and system logs.	1 = >512GB 2 = 256–512GB 3 = 128–256GB 4 = 64–128GB 5 = <64GB	35%	3	2	1
Data Transfer Speed	Faster read/write speeds allow faster instrument data dumps and reduce lag during operation.	1 = >2000MB/s 2 = 1000–2000MB/s 3 = 500–1000MB/s 4 = 100–500MB/s 5 = <100MB/s	25%	4	3	1
Radiation Tolerance	Radiation-hardened systems are necessary to prevent data corruption or permanent damage from cosmic rays.	1 = Radiation-hardened 2 = Industrial with ECC 3 = Industrial without ECC 4 = Consumer ECC 5 = Consumer, no protection	20%	5	3	2
Power Consumption	Low power consumption reduces heat production and power consumption, important for energy rationing on spacecraft.	1 = <0.5W 2 = 0.5–1W 3 = 1–1.5W 4 = 1.5–2.5W 5 = >2.5W	20%	2	1	4
		TOTALS:	**100%**	38.75%	68.75%	80.00%

Table A.12 (Data Storage Trade Study)

Data Interfaces							
Criteria	**Explanation**	**Grade**	**Weight**	**RS-422**	**MIL-STD-1553**	**Inter-Integrated Circuit (I2C)**	**Ethernet**
Data Rate	Higher throughput supports high-bandwidth sensors, cameras, and data logs.	1 = >100 Mbps 2 = 50–100 Mbps 3 = 10–50 Mbps 4 = 1–10 Mbps 5 = <1 Mbps	35%	4	5	5	1
Noise Tolerance	Throughput robustness to noise and packet loss to mitigate data loss.	1 = Rad-hard & fault tolerant 2 = Rad-hard 3 = Some fault tolerance 4 = minimal 5 = none	25%	2	1	4	5
Physical Complexity	Connection simplicity for ease of assembly and system integration.	1 = 1–2 wires 2 = 3–5 wires 3 = 6–10 wires 4 = Ribbon/bulk 5 = Shielded bundle	20%	2	4	1	5
Time Accuracy Preservation	Necessary for priority management and time-sensitive processes.	1 = Fully deterministic 2 = Hard real-time 3 = Prioritized messages 4 = Best-effort 5 = Non-deterministic	10%	1	4	5	5
Technology Readiness	Space-qualified systems with successful mission histories.	1 = Flown >5x 2 = Flown 2–5x 3 = CubeSat/LEO only 4 = Tested only 5 = Never flown	10%	1	1	2	3
		TOTALS:	**100%**	62.50%	42.50%	33.75%	40.00%

Table A.13 (Data Interfaces Trade Study)

Telecommunications							
Criteria	**Explanation**	**Grade**	**Weight**	**Software Defined Radio (SDR)**	**Deep Space Network (DSN)**	**Deep Space Optical Network (DSON)**	**Ultra High Frequency (UHF) Radio**
Power Consumption	Low power systems are essential for energy-constrained environments like Mars.	1 = <5W 2 = 5–10W 3 = 10–20W 4 = 20–40W 5 = >40W	35%	2	4	5	1
Data Rate	High data rate enables faster transmission of science data, reducing risk of memory overflow.	1 = >100 Mbps 2 = 50–100 Mbps 3 = 10–50 Mbps 4 = 1–10 Mbps 5 = <1 Mbps	25%	3	3	1	4
Technology Readiness	Systems with high TRL have been flight-proven and carry lower implementation risk.	1 = TRL 9 2 = TRL 8 3 = TRL 6–7 4 = TRL 4–5 5 = TRL <4	25%	2	1	4	1
Directionality	Systems with low pointing or alignment needs are easier to implement and more fault-tolerant.	1 = Omnibearing 2 = Wide-angle 3 = Narrow-angle 4 = Requires fine tracking 5 = Requires nanoradian precision	15%	2	3	5	1
		TOTALS:	**100%**	68.75%	53.75%	31.25%	81.25%

Table A.14 (Wheel Tread Trade Study)

MLI (Outer Cover)							
Criteria	**Explanation**	**Grade**	**Weight**	**Beta Cloth**	**Beta Cloth Aluminized**	**Kapton Coated and Backed**	**Double Sided Mylar**
Operational Temperature Range	Kelvin(K) (SYS.02)	1= exceptional temperature range 3 = moderate temperature range 5= does not meet the temperature range	25%	1	1	1	1
Tensile Strength	Evaluation of the tensile strength of the Beta cloth(kg/cm)	1= Good tensile strength value 3= Moderate tensile strength value 5= Low tensile strength value/no value	15%	1	1	4	2
Absorptivity	Evaluation of how much solar radiation is absorbed compared to how much it reflects. (α)	1= favorably low absorptivity 3= moderate absorptivity 5= high absorptivity	30%	5	5	3	1
Emissivity	Evaluation of the rate at which heat is radiated. Lower is favored for outer covers. (ε)	1= low emissivity 3=moderative emissivity 5=high emissivity	30%	4	3	2	1
		TOTALS:	**100%**	22.50%	30.00%	41.25%	71.25%

Table A.15 (MLI Outer Cover Trade Study)

MLI (Interior Layer)						
Criteria	**Explanation**	**Grade**	**Weight**	**Aluminized Kapton**	**Aluminized Mylar**	**Goldized Kapton**
Operational Temperature Range	Kelvin(K) (SYS.02)	1= exceptional temperature range 3 = moderate temperature range 5 = does not meet the temperature range	25%	1	2	1
TRL	Evaluation of the maturity of the proposed MLI material.	1 = Satisfactory TRL (TRL 5-6>>) 3 = Moderate TRL (TRL 3-4) 5 = Novice TRL (TRL 1-2)	15%	1	1	1
Absorptivity	Evaluation of how much solar radiation is absorbed compared to how much it reflects. (α)	1= high absorptivity 3= moderate absorptivity 5= low absorptivity	30%	5	5	2
Emissivity	The rate at which heat is radiated. Mid-low is favored for interior layers. (ε)	1= exceptionally low-mid emissivity 3= high-mid emissivity 5= high emissivity	30%	2	2	1
		TOTALS:	**100%**	62.50%	56.25%	92.50%

Table A.16 (MLI Inner Cover Trade Study)

Heat Pipes						
Criteria	**Explanation**	**Grade**	**Weight**	**Variable Conductance Heat Pipe (Ammonia-Cold)**	**Constant Conductance Heat Pipe**	**Thermosyphon**
Operational Temperature Range	Kelvin(K) (SYS.02)	1= can operate at low temperatures with an exceptional temperature range. 3= can operate at low temperatures within an acceptable temperature range. 5= small and limited temperature range.	30%	2	2	4
TRL	Evaluation of the maturity of the proposed heat pipe design.	1 = Satisfactory TRL (TRL 5-6>>) 3 = Moderate TRL (TRL 3-4) 5 = Low TRL (TRL 1-2)	15%	1	1	2
Reliability	Evaluation of whether the heat pipe design can perform in any condition.	1=Can operate in extremely harsh conditions 3= Can operate in relatively harsh conditions 5= cannot operate in harsh condition/can operate in normal conditions	35%	1	2	2
Complexity	Evaluation of whether the heat pipe design can be integrated into the system.	1=Integration is not complex 3= Requires effort to integrate. 5=Very hard to put into the system	20%	3	2	4
		TOTALS:	**100%**	82.50%	78.75%	50.00%

Table A.17 (Heat Pipes Trade Study)

Electrical Heaters						
Criteria	**Explanation**	**Grade**	**Weight**	**KHLVA, PLM-Series**	**SHK Series**	**Polyimide Thermofoil HK Series**
Power	PWR.01(W/cm^2)	1= low power consumption 3= moderate power consumption 5= high power consumption	30%	1	1	3
Mass	SYS.04(kg) Must be lightweight and easy to integrate.	1= lightweight and easy to integrate 3= moderate weight and requires some effort for integration 5= heavy and requires changing the system for integration.	25%	1	1	1
TRL	Evaluation of the technology-readiness level of the proposed electrical heater.	1 = Satisfactory TRL (TRL 5-6>>) 3 = Moderate TRL (TRL 3-4) 5 = Low TRL (TRL 1-2)	20%	1	1	1
Temperature Range	(SYS.02).	1= satisfactory temperature limit 3= smaller margin of the temperature limit 5= not a great temperature limit/does not fit the operating temperature range.	25%	1	2	1
		TOTALS:	**100%**	75.00%	68.75%	60.00%

Table A.18 (Electrical Heaters Trade Study)

Thermal Sensors						
Criteria	**Explanation**	**Grade**	**Weight**	**Thermocouple**	**Thermistor**	**RTD sensors**
Sensitivity	A measure of how sensitive the sensor is to small temperature changes	1: Exceptionally sensitive 3: Moderately sensitive 5. Low sensitivity	25%	5	1	3
Accuracy	The expected error range for the product	1: Exceptionally accurate to decimal places 3: Acceptably accurate with a margin of error of 1 or 2 degrees 5. The margin of error is exceptionally high	20%	3	3	5
Reliability	Evaluation of its ability to withstand long term missions	1: Very reliable 5: Minimally reliable	25%	1	4	3
Response Time	Measure of how fast the sensor can detect a temperature change	1: High 3: Medium 5: Low	30%	4	4	3
		TOTALS:	**100%**	34.00%	39.00%	32.00%

Table A.19 (Thermal Sensors Trade Study)

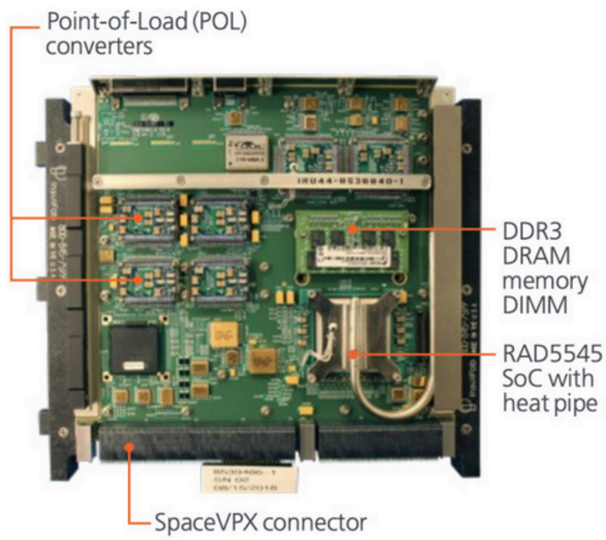

Figure A.20 (RAD5545 SpaceVPX single-board computer)

Pin 1	TXD-
Pin 2	TXD+
Pin 3	RTS-
Pin 4	RTS+
Pin 5	GND
Pin 6	RXD-
Pin 7	RXD+
Pin 8	CTS
Pin 9	CTS+

RS422/485 Pinout (9 Pin)

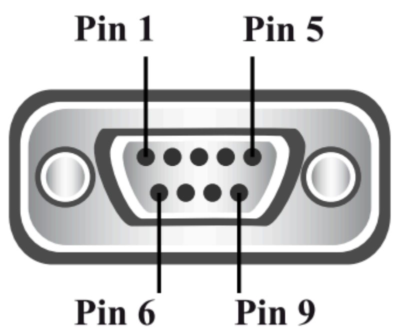

Figure A.21 (RS-422 Pinout Table)