System Requirements Review - Team 27

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Table of Acronyms

Abbreviation	Definition
STM	Science Traceability Matrix (1.2)
MCR	Mission Concept Review (1.2)
SiPM	Silicon Photomultiplier (1.2)
O2	Oxygen (1.2)
LEAP	Lunar Exploration Analysis Group (1.2)
LiDAR	Light Detection and Radar (1.3)
CC	Customer Constraints (1.4)
MG	Mission Goals (1.4)
SCI	Science Constraints (1.4)
SC	Spacecraft Constraints (1.4)
TCS	Thermal Control System (1.5.1.1)
MLI	Multi-Layered Insulator
NASA	National Aeronautics and Space Administration
GNC	Guidance, Navigation, and Control
SRR	System Requirements Review
TRL	Technology Readiness Level
PWR	Power Subsystem Requirements (1.5.3.1)
PRG	Power Generation (1.5.3.1)
PRS	Power Storage (1.5.3.1)
PRD	Power Distribution (1.5.3.1)
PRD	Power Distribution (1.5.3.1)
RTG	Radioisotope Thermoelectric Generator (1.5.3.1)
NASA	National Aeronautics and Space Administration (1.5.3.2)
TRL	Technology Readiness Level (1.5.3.2)
MCCET	Mission Concept Cost Estimate Tool (1.5.3.3)
CDH	Command and Data Handling
PMU	Power Management Unit (1.5.4.2)
I2C	Inter-Integrated Circuit (1.5.4.2)
UART	Universal Asynchronous Receiver-Transmitter (1.5.4.2)
CAN	Controller Area Network (1.5.4.2)
RHM	Radiation-Hardened Memory (1.5.4.3)
CPU	Command Processing Unit (1.5.4.3)

SRAM	Static Random-Access Memory (1.5.4.2)
EEPROM	Electrically Erasable Programmable Read-Only Memory (1.5.4.2)
TCS	Thermal Control System (1.5.5.2)
MTM	Meyer Tools & MFG (1.5.5.3)
MLI	Multi-layer Insulation ((1.5.5.3)
QTG	Quest Thermal Group (1.5.5.3)
IMLI	Integrated MLI (1.5.5.3)
STCH	Spacecraft Thermal Control Handbook (1.5.5.3)
LM	Lockheed Martin (1.5.5.3)
SP	SunPower (1.5.5.3)
INT	Instrument (1.5.6.1)
Ah	Ampere hours (1.5.7)
SPF	Single Point Failure (1.5.7)
ARES	Active Radiation Environment Sensor (1.5.7)
SRR	System Requirements Review (1.7.1)
LSE	Lead Systems Engineer (1.7.1)
PM	Project Manager (1.7.1)
CS	Chief Scientist (1.7.1)
DPMR	Deputy Project Manager of Resources (1.7.1)
MDR	Mission Definition Review (1.7.2)
KDP	Key Decision Point (1.7.2)
NICM	NASA Instrument Cost Model (1.7.2)
LCR	Life Cycle Review (1.7.3)
PDR	Preliminary Design Review (1.7.3)
CDR	Critical Design Review (1.7.3)
PRR	Production Readiness Review (1.7.3)
SIR	System Integration Review (1.7.3)
SAR	System Acceptance Review (1.7.3)
MRR	Mission Readiness Review (1.7.3)
SMSR	Safety and Mission Success Review (1.7.3)
CERR	Critical Events Readiness Review (1.7.3)
DRR	Disposal Readiness Review (1.7.3)
CCB	Change Control Board (1.7.4)
RFA	Requests for Action (1.7.4)
ADV	Advisory (1.7.4)

1. System Requirements Review

1.1. Mission Statement

Through past and current lunar exploration, it has been found that the Earth's Moon had volcanic activity about 3.8 to 3 billion years ago.[1] During volcanic eruptions, magma would reach the surface and, while subjected to the lower surface temperatures, begin to harden into a crust layer. During this process, the magma continued flowing under the surface, and lava tubes were formed. As the eruptions subsided, the magma eventually exited the subsurface entirely and left hollowed lava tubes. Throughout the years the Moon's surface has experienced impacts from micrometeorites, which created a number of lunar pits and caves that can allow entry into the lava tubes. Current data suggests that lunar pits and caves can protect humans and equipment from extreme fluctuations in temperature and radiation, which would allow long-term habitation of humans for future exploration on the moon.[2]

This mission aims to determine if the Mare Ingenii lunar pit is a viable option for sustained human presence on the Moon, to develop precursor lunar robotic missions and to define those scientific activities that astronauts will conduct on the Moon.[3] This mission aims to measure the strength and direction of magnetic fields correlated with lunar swirls using a magnetometer. This mission also seeks to identify the amount of radiation present within and outside the Mare Ingenii pit to determine the radiation's effects on the microclimate of the pit via using a Silicon Photomultiplier measuring photon counts to determine radiation amount and type.[4] As previously stated, this mission aims to determine if lunar pits can provide safe and enduring habitation systems to protect individuals, equipment, and associated infrastructure.[5] Muon imaging will be used to collect data on the characteristics within the Mare Ingenii lunar pit in order to define the depth, height, variations in terrain, and entry access. The structural integrity will be determined by sourcing the thickness and structural data acquired from muon imaging as well. All of these objectives and goals shall be obtained via the use of a single rover, and will improve the understanding of the composition of the Moon as well as how a sustained presence on the Moon will impact human health.

1.2. Science Traceability Matrix

The science objectives for the mission are categorized into two science goals: "develop precursor lunar robotic missions and define those scientific activities that astronauts will conduct on the Moon" and "provide safe and enduring habitation systems to protect

[1]Heng-Ci Tian et al., "Surges in volcanic activity on the Moon about two billion years ago," Nature Communications 14 (2023).

[2]Raymond P. Martin and Haym Benaroya, "Pressurized lunar lava tubes for habitation," Acta Astronautica 204 (2023): 157-174.

[3]Committee on the Planetary Science and Astrobiology Decadal Survey et al., Origins, Worlds, and Life.

[4]Hamamatsu Photonics, "MPPCs (SiPMs) / MPPC Arrays."

[5]Lunar Exploration Analysis Group (LEAG), "The Lunar Exploration Roadmap: Exploring the Moon in the 21st Century: Themes, Goals, Objectives, Investigations, and Priorities."

individuals, equipment, and associated infrastructure."[6] [7] The first three science objectives fall within the first science goal, which include extracting 10% oxygen from regolith in a lunar condition, determining the location of magnetic fields within and around the Mare Ingenii pit, and discovering the effects of radiation on the pit's microclimate to gauge the sustainability of human life. The last two science objectives are beneath the second goal, which includes determining the characterization of the Mare Ingenii pit to determine the viability of human habitation and discovering the structural integrity of the pit to evaluate its capabilities of supporting human life.

The first science objective of extracting oxygen from regolith is highlighted in the Science Traceability Matrix (STM) table as it is outside the mission's scope since a carbothermal reactor is necessary for oxygen extraction from regolith (Table 1). Although carbothermal reactors have been on the moon, such an instrument's cost, dimensions, and power needs would be far too much to handle for this mission. This was discovered after the creation of the Mission Concept Review (MCR) and solidified through thorough research into alternative instruments and methods.

To determine the location of magnetic fields, a fluxgate magnetometer between 0 and 23 nT shall be used upon landing.[8] Since both instruments selected for this objective have very low noise and considerable measurement capabilities, they shall conduct the samples for this objective. The final science objective for the first science goal is to determine the effect of radiation on temperature, humidity, and pressure within the lunar pit, which will be measured every second. This will allow researchers to have extensive data for comparison between peak and minimum radiation levels. The selected instruments for this objective will be capable of withstanding the environment inside the Mare Ingenii pit and shall be active until comparative data has been collected.

When measuring the omnidirectional dimensions of the Mare Ingenii pit to determine the viability of human habitation for the first science objective of the second science goal, the measurements shall be taken every 45 m of the 180-diameter pit to ensure the entirety of the pit is surveyed. This will allow for comparisons in any overlapping areas and will allow for the possibility of errors in measurements. The last science objective is to determine the structural integrity within the Mare Ingenii lunar pit and better understand its capabilities to support human life. A silicon photomultiplier (SiPM) will examine the pit's structural integrity by observing the flux of muons beginning with the pit's surface and continuing after dropping into the pit.

[6] Tian, Heng-Ci et al.. "Surges in volcanic activity on the Moon about two billion years ago." Nature Communications 14 (2023).

[7] Committee on the Planetary Science and Astrobiology Decadal Survey, Space Studies Board, Division on Engineering and Physical Sciences, and National Academies of Sciences, Engineering, and Medicine. Origins, Worlds, and Life: A Decadal Strategy for Planetary Science and Astrobiology 2023-2032. Washington, D.C.: National Academies Press, 2023.

[8] Kramer, Georgiana Y., Jean-Philippe Combe, Erika M. Harnett, Bernard Ray Hawke, Sarah K. Noble, David T. Blewett, Thomas B. McCord, and Thomas A. Giguere. "Characterization of Lunar Swirls at Mare Ingenii: A Model for Space Weathering at Magnetic Anomalies." Journal of Geophysical Research: Planets 116, no. E4 (2011).

Science Goals	Science Objectives	Science Measurement Requirements: Physical Parameters	Science Measurement Requirements: Observables	Instrument Performance Requirements		Predicted Instrument Performance	Instrument	Mission Requirements
"Develop precursor lunar robotic missions and define those scientific activities that astronauts will conduct on the Moon" - Origins, Worlds, and Life: A Decadal Strategy for Planetary Science and Astrobiology 2023-2032	Determine if 10% of oxygen (O_2) can be pulled from regolith in lunar conditions.	Define the amount of heat needed on the lunar surface for regolith to be heated to the correct temperature to produce a 10% yield of oxygen.	Identify the minimum temperature range required for oxygen extraction from regolith samples in the lunar environment.	Temperature range:	1273-1373 K	228.15-2073.15 K	Carbothermal Reactor	The magnetometer must be capable of measuring in nanotesla.
	Determine the location of any potential surface magnetic fields within and around the Mare Ingenii lunar pit, specifically those correlated with lunar swirls.	Identify the magnitude and direction of magnetic fields present in a range of 15km.	Detect magnetic fields in the 0-500nT range over an area of 10km.	nanoTesla range:	0-23 nT	-20,000-20,000 nT	3-Axis Fluxgate Magnetometer Model AP235	
				Noise level:	-0.05-0.05 RMS nT/√Hz	≤150 pTrms/√Hz		
				nanoTesla range:	0-23 nT	-60,000-60,000 nT	Bartington Fluxgate Magnetometer, 3-axis, Aerospace/Space	Mission must measure the thickness of the Mare Ingenii pit's walls, floor, and ceiling to determine its structural integrity.
				Noise level:	-0.05-0.05 RMS nT/√Hz	<5 pTrms/√Hz		
	Determine the effect of radiation on temperature, humidity, and pressure within lunar pits to determine if they can sustain human life.	Identify the percentages of Alpha, Beta, and Gamma radiation and fluctuation in temperature, humidity, and pressure within the Mare Ingenii pit.	Detect temperature, humidity, and pressure changes during peak ranges of Alpha, Beta, and Gamma radiation percentages every second.	Radiation levels:	0-1,370 W/m^2	0-1,500 W/m^2	Active Radiation Environment Sensor	
				Temperature range:	16-18 °C	0-60 °C	Model HMP60 Humidity Sensor + Model DI-808	Rover will land within 1 meter of the Mare Ingenii pit in the Thomson M crater.
				Temperature range:	16-18 °C	-200-1,000 °C	PT1000 Class F0.3 + Pressure-Temperature	Mission must collect 5 data samples of magnetic field strength from the landing site to the opening of the Mare Ingenii pit.
				Pressure range:	10^{-12}-10^{-10} torr	0-30 torr		
"Provide safe and enduring habitation systems to protect individuals, equipment, and associated infrastructure" - (Lunar Exploration Analysis Group (LEAG) Habitation: HAB-SAT Report, Priority Objectives	Characterize the depth, height, terrain variation, and ease of access within lunar pits/caves to determine the viability of human habitation.	Define the characteristics of the Mare Ingenii pit within a vertical distance of 55m and a diameter of 180m.	Identify the omnidirectional dimensions of the Mare Ingenii pit every 45m.	Measurement Range:	55 m	70 m	MID-360 LiDAR	Mission must collect 5 data samples using the Silicon Photomultiplier to determine the structural integrity of the entire Mare Ingenii pit.
	Determine the structural integrity within the Mare Ingenii lunar pit to determine if it can support human life.	Define the thickness and structure of the Mare Ingenii pit utilizing the flux of muons.	Detect the thickness of the Mare Ingenii pit's walls, floor, and ceiling in meters every 40m.	Wavelength range:		425nm	Silicon Photomultiplier + EJ-200	Rover must drop into the Mare Ingenii pit to determine the microclimate, structural integrity, and viability of human habitation in the pit.
				Wavelength range:		425nm		
				Wavelength range:		425nm		
				Wavelength range:		425nm		

Table 1.The STM of the mission.

1.3. Summary of Mission Location

The mission's final selected landing site is located within the Mare Ingenii lunar pit, situated inside the Thomson M crater. This pit lies at the coordinates -35.9494°N, 166.095°E, chosen for its scientific value and optimal landing conditions.

Mare Ingenii is located on the Moon's outer edge of the South Pole-Aitken basin and is known for its distinctive lunar swirl patterns. These swirls are a key feature of interest for the mission, providing a unique opportunity to study the Moon's magnetic field. The Mare Ingenii lunar pit, positioned near the center of Thomson M crater, also offers the potential to contain lava tubes. The pit's depths range from 39 to 64 meters, with overhangs on the western and southern sides, offering access to exposed geological layers that will be explored during the mission.[9] Measurements of the microclimate, structural integrity, and radiation will be taken after dropping into the lunar pit. The measurements for radiation and pressure will be taken every second and temperature every 5-10 seconds. In the pit, the thickness of the pit walls will be assessed by the SiPMs every 40 meters along with

[9]"Pits | Lunar Reconnaissance Orbiter Camera." Accessed October 24, 2024.

Light Detection and Radar (LiDAR) mapping every 45 meters.

The Mare Ingenii location was selected for three primary reasons: the presence of lunar swirls, the potential for lava tubes, and viable landing terrain. The unique lunar swirls surrounding the pit provide a valuable research opportunity, allowing the mission to explore these features in greater detail.[10] The lunar pit also is known to have the potential for many lava tubes within the pit. Finally, the terrain surrounding the lunar pit is viable for landing due to satisfying the maximum 10-degree slope requirement.

The landing site is located 1 kilometer East of the Mare Ingenii pit fulfilling the 5-kilometer range for the mission landing location. The landing site coordinates are -35.9494°N, 166.095°E. This location was chosen for its access to a lunar swirl, large open area for landing, and weaker magnetic field strength in the area. Five measurements of the magnetic field strength will be taken en route to the pit every 200 meters. This landing site is situated on top of the lunar swirl, allowing direct access to investigate these features with lunar swirl measurements taken upon landing. This location was also chosen for its wide berth from the large craters in the area, allowing a greater area of viable landing. The landing site benefits from a lower magnetic field strength compared to areas west of the pit, which enhances the mission's ability to conduct sensitive magnetic field experiments.

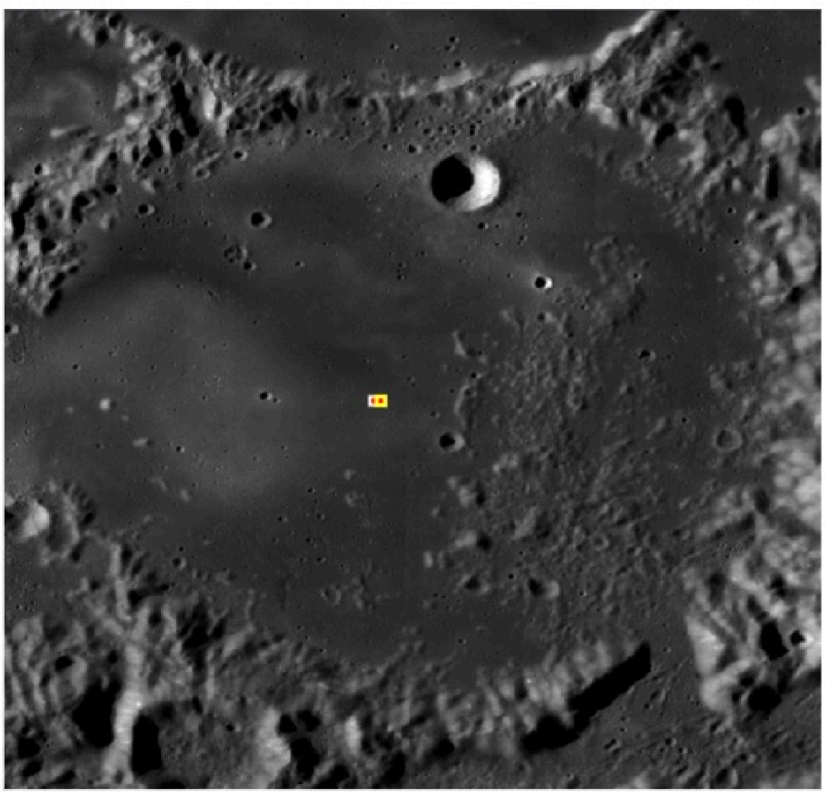

Figure 1.Thomson M crater within Mare Ingenii showing both the lunar pit and the landing locations.

[10]Kramer, Georgiana Y., Jean-Philippe Combe, Erika M. Harnett, Bernard Ray Hawke, Sarah K. Noble, David T. Blewett, Thomas B. McCord, and Thomas A. Giguere. "Characterization of Lunar Swirls at Mare Ingenii: A Model for Space Weathering at Magnetic Anomalies." Journal of Geophysical Research: Planets 116, no. E4 (2011).

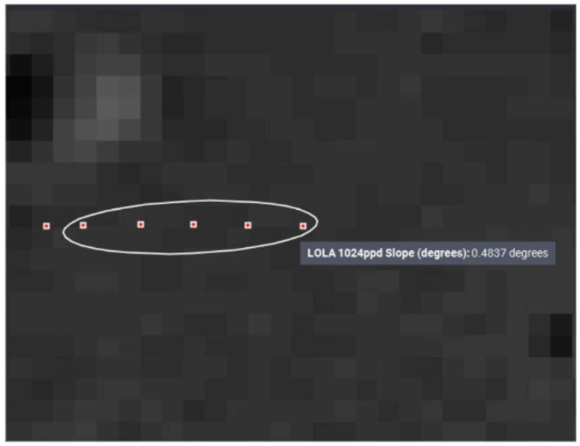

Figure 2. Mare Ingenii pit (very left dot) and the terrain slope of the landing site (very right dot) along with the evenly spaced 200 meter measurement points for the magnetic field.

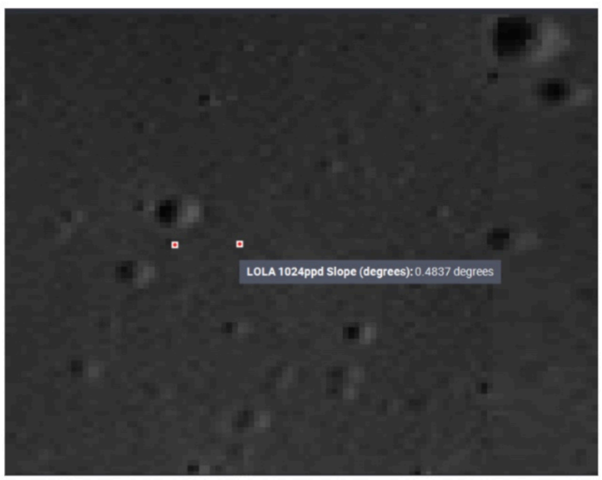

Figure 3. Mare Ingenii pit (left dot) and the terrain slope of the landing site (right dot).

1.4. Mission Requirements

The mission requirements are detailed in a 'flow-down' structure through tabular format (Table 2). This table has been altered slightly from the MCR to improve readability and present a more updated list of mission requirements. Each requirement is given a unique number, seen in the leftmost column, coupled with the acronym of the type of constraint it refers to. These include Customer Constraints (CC), outlined by the customer in the Mission Task document provided by L'SPACE, corresponding Mission Goals (MG), Science Constraints (SCI), and Spacecraft Constraints (SC). Constraints defined by the customer are fundamental and are listed at the top of the table, while constraints pertaining to science and the spacecraft are built off of them and listed underneath. The table depicts a sort of hierarchy represented by shades of blue, with darker shades indicating higher-level requirements–or higher level of flow–and progressively lighter shades representing lower-level requirements–or lower level of flow–that follow from higher-level ones.

Requirements highlighted in yellow have been put on hold for this mission. For more information on this decision, see section 1.7.4. *Change Control*.

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Relevant Subsystem	Req met?
CC - 0.0	The mission shall adhere to customer constraints.	Parameters outlined in document, in relation to the given constraints and requirements	N/A	CC 0.1 - CC 0.10 MG - 1.0	Analysis	All	Met
CC - 0.1	System must adhere to mass constraint of 350 kg	Limited by the mission task, and to ensure not to overexceed mass capacity during transport.	CC - 0.0		Inspection	All	Met
CC - 0.2	System must adhere to dimension constraint of 2 m x 1.25 m x 1.25 m during transit	Limited by spacecraft transport dimensions; cannot exceed due to constraint.	CC - 0.0		Inspection	All	Met
CC - 0.3	System must explore the lunar/pits caves and perform surface science operations	Outlined mission statement and goal for research.	CC - 0.0		Inspection	Payload	Met
CC - 0.4	System must not exceed cost constraint of $425M	Limited by set amount of alloated funding. Cannot exceed due to unavalible funds.	CC - 0.0		Inspection	All	Met
CC - 0.5	System must be ready to launch by March 1st, 2030, at Cape Canaveral.	Hard deadline, set by mission timeline.	CC - 0.0		Inspection	Payload	Met
CC - 0.6	System cannot have a Radioisotope Thermoelectric Generator (RTG) or any derivative thereof; System must not exceed radioactive material cumulative mass of 5g	The usage of the RTG is prohibited, in accordance with the mission.	CC - 0.0		Inspection	Electrical, Structural, Mechanical	Met
CC - 0.7	System will be transported on a separate, primary vehicle launch (rocket) with power sufficient during transport. No scientific data will be collected during transit	Transportation structurlization and unavalibality of research during transit.	CC - 0.0		Inspection	Structural	Met
CC - 0.8	System shall arrive at target destination adhering to landing site constraints: given a 100m diameter landing zone, it shall not exceed a slope of 10 degrees, and should traversal be necessary to the science site or lunar cave, it must no be no further than 5 km from the landing zone (only applicable for mobile vehicle)	Landing site parameters/criterias outlined in mission.	CC - 0.0		Analysis	Structural, Mechanical	Met
CC - 0.9	System shall communicate data to Earth via relay of spacecraft orbiting the Moon; spacecraft must be able to communicate with the orbiting spacecraft directly, and primary mission orbiter which shall remain in a circular polar orbit of 100km during the mission's lifespan	Communications necessary in order to establish control, navigation, and data collection.	CC - 0.0		Demonstration	Electrical, Communication, Mechanical	Met
CC - 0.10	System shall undergo analog testing in a comparable volcanic environment; drop testing is necessary should the landing site not have a traversable entrance, and tether or winch system	Necessary to evaluate mission success and analyze potential faults and feasibility.	CC - 0.0		Test	All	Met
MG - 1.0	The mission shall conduct scientific research of the Mare Ingenii Pit in relation to the sustainability of human infrastructure.	In conjunction with the Mission Goal: "Develop precursor lunar robotic missions and define those scientific activities that astronauts will conduct on the Moon" and ""Provide safe and enduring habitation systems to protect individuals, equipment, and associated infrastructure"	CC - 0.0	SCI - 1.1 SCI - 1.2 SCI - 1.3 SC - 1.4	Demonstration	All	Met

Table 2. A portion of the Team 27 Mission Requirements table, detailing mission requirements. The full table is available in the Appendix.

1.5. System Definition

1.5.1. Spacecraft Overview

MechE details

The power subsystem is the fundamental component that sustains and enables the entire spacecraft's functionality. The purpose of the power subsystem is to ensure the spacecraft has enough electrical energy to fulfill the mission's objective. Its three sub-assemblies: power generation, power storage, and power distribution, make up the power subsystem. The power generation, enables the spacecraft to extend its mission duration by generating power derived from solar energy, as opposed to solely using its power storage energy, from being discharged and depleted. Additionally, to maintain efficiency and correct power management, the power distribution sub-assembly is implemented for dispersion and smart power allocation.

The power subsystem for this spacecraft consists of two Li-Ion batteries (power storage) developed by Blue Canyon Technologies, connected in parallel, to supply a nominal voltage of 28V, and a 24V - 33.6V unregulated voltage. It has a current capacity of 13.6 Ah, and a power capacity of 380.8 Wh, with dimensions of 3.6 in x 7.2 in x 3.5 in (raw dimensions), converted to 4.1 in x 7.7 in x 3.5 in to account for space/designer margin.[11]

It also utilizes a custom solar cell array, consisting of 348 triple junction GaAs solar cells developed by AZUR SPACE Power GmbH, consisting of a total area of 1.05 m^2. The total area was calculated based on the bare cell efficiency value, and a baseline power draw based on over-draw and essential functionality draw (200 Wh). It has an average bare cell efficiency of 30% under a solar constant of 1367 W/ m^2 and can deliver 253.1 W/m^2 to the power distribution sub-assembly. The acceptance value or electrical output

[11] "6.8 Ah Flight Proven Spacecraft Battery," n.d.,

of the solar cell is 2350 mV for voltage at the operating point (Vop), 505 mA for average current at the operating point (under the Vop), and 475 mA for minimum current at the operating point.[12]

Furthermore, it has a radiation tolerance of 30 kRads (qualified for 35 kRads), enough to sustain the lunar terrain's lack of atmosphere.[13]

Top Level Requirements (Electrical) [Not Colored]

SC - 1.4.2	The spacecraft shall have sufficient power to maintain mission operations.	Spacecraft functionality is derived from its power.	SC - 1.1(Mission Requirements Table)	PWR - 0.0	Demonstration	All	Met
PWR - 0.0	The power sub-system shall have enough power to maintain mission operations.	All spacecraft functionality and systems are derived from power.	SC - 1.4.2	PRG - 1.0, PRS - 1.0, PRD - 1.0	Inspection	All	Met
PRG- 1.0	The power sub-assembly: "power generation," shall supply enough voltage, and amperage for instrument operations.	All spacecraft functionality and systems are derived from power.	PWR - 0.0	PRG -1.1	Inspection	All	Met
PRS - 1.0	The power sub-assembly: "power storage," shall store and have the capacity to carry the voltage and current required for the electrical system.	In order to derive and reference power, there must be a source or "pool" of power to pull from.	PWR - 0.0	PRS - 1.0	Inspection	All	Met
PRD -1.0	The power sub-assembly: "power distribution," shall distribute the amounts of required voltage and amperage per each sub-system power draw requirements.	For the system to operate, power must be distributed to each subsystem or instrument, with its corresponding power draw needs.	PWR - 0.0	PRD - 1.1	Inspection	All	Met

Table 3. The top level requirements of the power subsystem.

CDH section here

The thermal subsystem will be in charge of balancing the spacecraft's thermal energy. The main goal of the thermal subsystem is to create a thermal control system (TCS) in order to keep the rover in an ideal temperature range (from 253.15k - 328.15k). The temperature range was created by the operating range of the electronic components inside the spacecraft. Successfully keeping the rover in the ideal temperature range, will allow

[12] "30% Triple Junction GaAs Solar Cell Type: TJ Solar Cell 3G30C - Advanced Best in Class EOL-Values!" (AZUR SPACE, August 19, 2016),

[13] "SuperNova PCDU," n.d.,

all the instrumentation on board to gather accurate readings. Furthermore, regulating the spacecraft's thermal energy will reduce the thermal fatigue on the rover's components and on its chassis. The thermal subsystem consists of two sub-assemblies. The first one is the insulator sub-assembly and the second one is the cooling sub-assembly.

The insulator sub-assembly consist of a multi-layered insulator (MLI) manufactured by Quest Thermal Group. The number of layers that the insulator sub-assembly will consist of will be 28. The number of layers was chosen based off NASA's Spacecraft Thermal Control Handbook, page 166, figure 5.5 MLI blanket effective emittance chart.[14] The MLI will have an emittance of .005. The total area that the MLI will cover will be .6641 m^2. The area was calculated by adding the front, back, bottom, and sides area of the rover. The top of the rover will not be covered in MLI to ensure maximum radiated energy.

The cooling sub-assembly will consist of SunPower Ametek's CryoTel DS Mini cryocooler. The DS Mini will make sure that the scientific instruments run at their appropriate operating range for data gathering. This will be the only active system from the thermal assembly. In this case, an active system is an element that will draw power from the batteries to operate. The DS Mini will operate at 1.8W @ 77k.[15] Having a cryocooler for this mission is a requirement because, based off the thermal calculations, the rover will be running at surplus of thermal energy. Requiring the spacecraft to be cooled down to ensure proper instrumentation readings.

Top Level Requirements (Thermal)

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Relevant Subsystem	Req met?
SCT-1	The Spacecraft shall have 0w load of thermal energy during both cold and hot case condition	To ensure the spacecraft does not cool down or heat up. Causing thermal fatigue.		TH-1	Analysis	All	Met
SCT-2	The Spacecraft shall prioritize passive elements over active elements	Reducing the amount of active elements will decrease the risk of failure on the TCS	SCT-1		Inspection	Thermo	Met

Table 4. The top level requirements of the Thermal Subsystem.

The payload subsystem includes all important instrumentation that will drive the mission's science objectives. The ultimate purpose of this subsystem is to contain every

[14] Gilmore, David G., ed. Spacecraft Thermal Control Handbook. 2nd ed. El Segundo, Calif: Aerospace Press, 2002.

[15] "7.0 Thermal Control - NASA." Accessed October 27, 2024.

instrument and their documented purpose and specifications in order to lead a successful mission. All research will be done through the use of the instrumentation as part of the payload subsystem.

The Carbothermal Reactor is the primary instrument planned to use for the extraction of oxygen from lunar regolith. However, the related science objective is beyond the mission's scope. Thus, the instrument will not be of use and is removed from the instrumentation.

The 3-Axis Fluxgate Magnetometer and the Barlington Fluxgate Magnetometer are the primary and secondary instruments, respectively, used to detect magnetic fields around the lunar pit. They are each required to have the ability to measure magnetic fields in the nanoTesla range of 0 nT to 23 nT.

The Active Radiation Environment Sensor is the primary instrument for radiation sensing. The instrument is required to have the ability to detect the radiation level range of 0 W/m² and 1,370 W/m².

The Model HMP60 Humidity Sensor + Model DI-808 is the primary instrument used to sense the temperatures of the lunar pit, and the PT1000 Class F0.3 is the secondary instrument. They each are required to detect a temperature range of 16 °C to 18 °C. The PT1000 Class F0.3 is also a secondary instrument for pressure sensing and is required to detect a pressure range of 10^{-12} torr to 10^{-10} torr.

The MID-360 LiDAR is the primary instrument used to measure the omnidirectional dimensions of the lunar pit. The instrument is required to measure its surroundings of up to 55 meters.

Lastly, the Silicon Photomultiplier is the primary instrument used to measure the thickness of the lunar pit. The instrument is required to make such measurements every 40 minutes.

Instrument	Mass	Dimensions	Max Power Draw
Model HMP60 Humidity Sensor + Model DI-808	99 g + 453 g	(36 x 36 x 9 mm) + (13.81 x 10.48 x 3.81 cm)	10W
Lunar Outpost Canary-S	2.27 kg	21.59 x 17.78 x 15.24 cm	20W
PT1000 Class F0.3 + Pressure-Temperature	6 kg	5.0 x 2 x 1.3 mm + 304.8 mm^3	250W
Silicon photomultiplier	TBD	TBD	TBD
Active Radiation Environment Monitors	2.01 kg	18.69 x 14.48 x 5.46 cm	3.6W
MID-360 LiDAR	265 g	65 x 65 x 60 mm	6.5 W
3-Axis Fluxgate Magnetometer Model AP235	150 g	10.52 x 4.06 x 2.86 cm	3.24 W
Bartington Fluxgate Magnetometer, 3-axis, Aerospace/Space	94 g	20 x 20 x 20 mm	0.175 W

Table 5. The instruments of the Payload Subsystem.

1.5.2. Mechanical Subsystem

The mechanical subsystem consists of subassemblies containing components categorized in the structural framework of the rover, mobility, or guidance, navigation, and control (GNC). Team 27 decided to focus on a few components in mobility and GNC. It must be noted, however, that due to the loss of the primary Mechanical Engineer in the week leading up to the SRR deadline, the mechanical subsystem did not have adequate time to be worked on, since team members from other areas and expertises had to cover knowledge not always well-versed in. This lack of time can be showcased in the Team's inability to complete trade studies for components such as the chassis, body, and motor controller for the rover, which were pre-selected with mass, power draw, and costs calculated. These will be mentioned in the mechanical subsystem overview.

The most substantial section within the mechanical subsystem portion of this System Requirements Review (SRR) will be the Mechanical Subsystem Trade Studies.

1.5.2.1. Mechanical Subsystem Requirements

The mechanical subsystem requirements will be completed fully in the future, when Team 27 has more time to work on the table.

1.5.2.2. Mechanical Subsystem Overview

Due to the loss of the primary Mechanical Engineer leading up to the SRR deadline and members needing to pick up the work of the mechanical subsystem, the primary member of whom does not have expertise in the field of engineering, it was difficult to put together a largely detailed and comprehensive account of the entire overview of the mechanical subsystem. Thus, in selecting the chassis, motor controller, and body of the rover, Team 27 decided it best to find costs for these three based on raw estimates, and no specific manufacturers or products were selected. $10,000,000 was abstractly estimated to be the cost of those three components combined, just less than half of the mechanical subsystem budget of $21,250,000. Additionally, the components of the rover not explained in more detail will fall into the weight constraint of 350 kg, and it is estimated that the total mechanical subsystem will fall within a weight range of just over 290 kg, giving the unknown components a weight restriction of 250 kg while taking into account the masses of the other subsystems, including the masses of the components which were selected in more detail. Thus, in total, the mechanical subsystem, including mechanical and GNC components, comes out to a cost of $16,700,000 and 291.44 kg specifically.

Technology readiness level (TRL) and further mechanical subsystem overview tables will be provided in the future when there is sufficient time.

1.5.2.3. Mechanical Subsystem Trade Studies

In evaluating options within the mechanical sub-assemblies of the subsystem, it was vital to compare amongst manufacturers and designs and to use strict criteria, including Mass, Performance, Reliability, and Risk. Thus, trade studies were completed for some of the components of the mechanical subsystem that time allowed for, including the motors (stepper and steering), tires, and cameras.

Of most significance, the motors of the rover were carefully selected keeping in mind the overall mass of 350 kg that the rover must not exceed, as well as the overall budget of $425 million and mechanical subsystem budget of $21,250,000. Team 27 sought to utilize two stepper motors, which will allow the rover to drive forwards and backwards carefully, and four steer motors, one for each of four wheels, that will allow the rover to make twists and turns around and on top of uneven terrain, as well as easily navigate to explore inside of pits and caves. In order to fit within the rover, the stepper and drive motors are small and compact, with dimensions set to be 8 inches with 3 inches in diameter maximum. Thus, two manufacturing companies were compared for their stepper and steer motors, and were graded on a strict criteria of Mass, Performance, Reliability, and Risk. In grading each company/motor design, Mass was weighted at 30% because the mass of the motors should not be so large that it contributes significantly to the overall weight limit. Performance was weighted at 40%, since the motors are what allow the rover to traverse and therefore complete the mission. Both Reliability and Risk were weighted at 15% each, since the Reliability and Risk of a motor are related to one another and risks based on reliability on the motor are similar amongst motors in general. The two manufacturing companies that were considered in selecting stepper and steer motors, known for their participation in providing motors for previous NASA spacecrafts, are Ducommun[16] and Moog.[17]

The Ducommun motors have been used in NASA satellites and the Mars rover, and have a mass of 0.02 kg each, for a total of 0.12 kg for six motors, and have a power draw of about 5 W each maximum, for a total of 30 W[18]. The Moog motors have also been used in NASA satellites in assistance with Boeing, and have an estimated mass of about 0.005 kg each, for a total of 0.03 kg, and a power draw of about 3 W each max, for a total of 15 W.[19] Otherwise, the two types are similar, and were carefully compared in a trade study based on the aforementioned criteria (Table 6).

In comparing the two motors, based on raw values for weight which are both so small, the Ducommun motors were given a (10) and the Moog motors were given a (10) for the Mass criteria. For Performance, based on both company's previous experiences with NASA exploration missions, Ducommun was given a (9) and Moog was given a (8), since they are both recently relevant and high-quality technology–but Ducommun was given a slightly higher score because of its current work with Mars 2020. For Risk and Reliability, which are related to one another, the scores were somewhat relevant to the Performance gradings because reliability is showcased through performance and recent relativity, and since Ducommun was given a slightly higher score for Performance, it was given respective Risk and Reliability scores of (8) and (8) while Moog was given a (7) and (7), which tipped Ducommun's overall grading higher to allow Team 27 to reasonably conclude that they were the correct manufacturer to select motors from. It should also be mentioned that when considering mass and power draw, the cost for six motors produced by Ducommon is higher than Moog's, at $1,500,000, compared to $600,000 (using the L'SPACE Mission Concept Cost Estimate Tool (MCCET)). This was also significant in considering the total mission budget of $425 million, as well as the mechanical subsystem budget of $21,250,000.

[16]"Space Qualified Motors, Actuators and Resolvers." Ducommun. Accessed October 31, 2024.
[17]Moog. "Moog." Moog to Steer NASA SLS Exploration Upper Stage for Boeing. Accessed October 31, 2024.
[18]Ducommun. Space-Qualified Stepper Motors. Accessed October 31, 2024.
[19]Moog. High Performance DC Motor. Accessed October 31, 2024.

Motors					
Criteria	**Explanation**	**Grade**	**Weight**	**Ducommun**	**Moog**
Mass	Refers to the mass of a single motor multiplied by 6. Lower mass (preferred) is significant for ensuring the total does not exceed the mission mass constraint of 350 kg.	10 = Mass within 0.15 kg 5 = Mass within 0.20 kg 0 = Mass within 0.25 kg	30%	10	10
Performance	Performance describes how the motor type is believed it will perform based off of facility performance/past/current mission performance. Important for achieving mission goals.	10 = Excellent performance 5 = Moderate performance 0 = Poor performance	40%	9	8
Reliability	Considers relevance and incorporation of recent technological advancements and potential for success. Also considers historical reliability (if applicable) and corresponding performance.	10 = Highly reliable 5 = Moderately reliable 0 = Unreliable	15%	8	7
Risk	Includes potential for damage to planetary surface, lack of knowledge of type never used before in space exploration, potential for malfunction, etc.	10 = No risk 5 = Moderate risk 0 = Catastrophic risk	15%	8	7
		TOTALS:	100%	90.00%	83.00%

Table 6.The trade study for the motor selection.

Implementing cameras in allowing the rover to navigate successfully and avoid hazards is essential. Team 27 looked to implement two navigation cameras, both on the front of the rover, and four hazard cameras, with two being on the front and two being on the back of the rover (6 cameras total). Engineering cameras from NASA missions were looked at, in particular, the cameras utilized in the previous Curiosity mission and the ongoing Perseverance mission. In both of these missions, NASA worked with Malin Space Science Systems (MSSS)[20] to develop them, so because of the lack of difference between manufactures, when comparing the two slightly varied camera types, they will be referred to as the Curiosity cameras and the Perseverance cameras. In all cases, the navigation cameras and the hazard cameras are similar to each other, so they were grouped together and instead the general question of which camera, Curiosity or Perseverance, should be selected, was asked. They were compared based on Mass, Performance, Reliability, and Risk. Mass was given a weight of 10% due to the insignificant difference of mass between the two types; Performance was given a weight of 40% due to the significance in ensuring the rover performs and that being contingent on its ability to navigate and avoid obstacles properly with the cameras; Reliability was given a weight of 25% to account for the importance of the camera's reliability in being functional and guiding the rover's navigation; Risk was given a weight of 25% relative to the risk each camera type may pose on the success of the rover's operations and the mission as a whole. Each criteria has a grading scale of (0) to (10), with (0) being the worst possible score to receive and (10) being the best.

[20] "All MSSS Projects." Space Camera/Systems, and Planetary Science Projects - Malin Space Science Systems. Accessed October 31, 2024.

The Curiosity cameras[21] assisted the rover in autonomous navigation on the planetary surface of Mars during this mission. They were successful in doing so and aided in the overall mission success of Curiosity finding that Mars could have supported life.[22] These cameras have a mass of about 220 g each, for a total of 1,320 g for six cameras, and a power draw of about 2.2 W each, for a total of 13.2 W.[23] Additionally, the Curiosity cameras have fisheye lenses, allowing them to view the surface before them at a 45 degree x 45 degree field of view both horizontally and vertically, and 67 degrees diagonally.[24] Similarly, the Perseverance cameras[25] are currently being used for navigation of the rover on Mars, and are proving to be quite successful as well. These cameras have a mass of about 71 g each, for a total of 426 g for six cameras, and a power draw of about 2 W each, for a total of 12 W total.[26] The navigation cameras have a view horizontally and vertically of 96 degrees x 73 degrees respectively, while the hazard cameras have a view of 136 degrees x 102 degrees.[27] Both camera types, based on the overall spacecraft size constraints of 2 m x 1.25 x 1.25 m and the other components of the mechanical subsystem and their respective dimensions that have been set/determined, are set to have dimensions of 0.3 m x 0.3 m x 0.3m maximum each–which is a large margin for these small devices, but ensures adequate room just in case.

For grading, the Curiosity cameras and Perseverance cameras were given a (9) and (10) respectively in the Mass category, based on their raw weight sums of six cameras each and because the first is heavier than the second. In Performance, both cameras were given (10), based on the current/past success and camera usages of the Curiosity mission and the Perseverance mission. For reliability, given that the camera's abilities are less advanced technologically and numerically in the Curiosity mission, they were given (7) while Perseverance cameras were given (8), not too high since the mission is still ongoing and total success/results have not been secured. Risk scores were based on reliability, related to age and quality of the cameras and their ability to potentially malfunction, giving the Curiosity cameras a (7) and the Perseverance cameras a (9). Conveyed in Table ?, it was determined that the cameras similar to the ones manufactured and currently being used in the Perseverance mission would be selected for Team 27's mission. It was also significant that utilizing mass and power draw numbers for each camera, the Perseverance cameras produced a lower overall cost of $1,500,000 compared to the Curiosity cameras, which produced an overall cost of $2,500,000 (using the L'SPACE Mission Concept Cost Estimate Tool (MCCET)). GNC components broadly fall under the mechanical subsystem, and so it is important to ensure meeting the overall $425 million budget and mechanical subsystem-defined 5% chunk of that budget, $21,250,000.

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[21] "Annotated Image of Curiosity Rover and All of Its Cameras - NASA Science." NASA. Accessed October 31, 2024.

[22] "Curiosity." Encyclopædia Britannica. Accessed October 31, 2024.

[23] PDS Geosciences Node, Washington University in St. Louis. "Analyst's Notebook Help." Analyst's Notebook. Accessed October 31, 2024.

[24] PDS Geosciences Node, Washington University in St. Louis. "Analyst's Notebook Help." Analyst's Notebook. Accessed October 31, 2024.

[25] "Perseverance Rover Components - NASA Science." NASA. Accessed October 31, 2024.

[26] "The Mars 2020 Rover Engineering Cameras." NASA. Accessed October 31, 2024. Printed in Appendix.

[27] "The Mars 2020 Rover Engineering Cameras." NASA. Accessed October 31, 2024. Printed in Appendix.

Cameras					
Criteria	**Explanation**	**Grade**	**Weight**	**Curiosity**	**Perseverance**
Mass	Refers to the mass of a single camera multiplied by 4. Lower mass (preferred) is significant for ensuring the total does not exceed the mission mass constraint of 350 kg.	10 = Mass within 2 lbs 5 = Mass within 7 lbs 0 = Mass within 12 lbs	10%	9	10
Performance	Performance describes how the camera type is believed it will perform based off of facility performance/past/current mission performance. Important for achieving mission goals.	10 = Excellent performance 5 = Moderate performance 0 = Poor performance	40%	10	10
Reliability	Considers relevance and incorporation of recent technological advancements and potential for success. Also considers historical reliability (if applicable) and corresponding performance.	10 = Highly reliable 5 = Moderately reliable 0 = Unreliable	25%	7	8
Risk	Includes potential for damage to planetary surface, lack of knowledge of type never used before in space exploration, potential for malfunction, etc.	10 = No risk 5 = Moderate risk 0 = Catastrophic risk	25%	7	9
		TOTALS:	100%	84.00%	92.50%

Table 7.The trade study for the cameras selection.

Looking to wheel and tire development and size of lunar-roving vehicles[28] across NASA missions and keeping in mind the defined mission and subsystem requirements, it was decided that Team 27 would opt for tires that implemented a flexible material to traverse the lunar surface and regolith with minimal track depth left behind. For both final tire choices, it was not necessary to directly compare across manufacturers. This is because these unique, flexible tire styles have been designed specifically for NASA space vehicles by the NASA Glenn Research Center in collaboration with The Smart Tire Company.[29] Both tire types are set to follow dimension sizes of 22 in x 8 in, based on the sizes of other components of the rover. However, some of the additional materials of each of the two types of tires vary, which impact criteria related to Mass, Performance, Reliability, and Risk. Mass was given a reduced weight of 15% because it was reasoned that heavier tires may be required in order to have tires capable of carrying such a large rover. Performance was given a weight of 30% because it utilizes past mission/facility performance to project future performance, and performance is ultimately what allows the mission to achieve its goals. Reliability was given a weight of 25% because the product's relativity to modern technology and historical performance is largely important. Lastly, Risk was given a weight of 30% because avoiding damage to the lunar surface and pits is one of Team 27's top priorities. Each criteria has a grading scale of (0) to (10), with (0) being the worst possible score to receive and (10) being the best.

The first tire type is made of shape memory alloy (SMA)[30] coils and is airless with a sturdy wheel frame, which is a flexible metal that has been tested within planetary-simulation facilities by NASA and The Smart Tire Company to be capable of reverting

[28]NASA. The development of wheels for the lunar roving vehicle. Accessed November 1, 2024.
[29]"About." The SMART Tire Company. Accessed October 31, 2024.
[30]"Superelastic Tire." NASA. Accessed October 31, 2024.

back to its original shape, traversing uneven terrain, leaving minimal track damage, and carrying amounts of weight that exceed the mission constraint of 350 kg. From NASA research[31] and development, it is estimated that the maximum mass of four of these SMA tires combined is about 40 kg.[32] The second type of tire, which will be referred to as the lunar-roving vehicle (LRV) tire due to its similarity to the type used for the LRV from Apollo 15, 16, and 17 missions, was also considered because it is made of flexible wire-mesh while also implementing an inner metal wheel frame and tread strips for structure and successful navigation across the rocky lunar surface.[33] The maximum mass of four of these LRV tires combined is about 22 kg.[34] Both tires mimic the Thus, merely based on their masses, the LRV tire received a higher score of (7) for the Mass criteria, while SMA received a (3). Additionally, due to incorporating a more rigid wheel frame for the tire, it was reasoned that the tread strips of the LRV tires would be more likely to leave deeper track marks on the regolith soil than the SMA tires, so for Risk LRV received a (6) and SMA a (8). For Reliability, SMA tires were believed to be quite reliable (8), solely because their development has occurred much more recently and with much newer technology; LRV's reliability was also considered quite highly (8), even considering its outdatedness. Lastly, for performance metrics based on past performance and facility performance, SMA received an (8) and LRV a (7) because Thus, in completing trade studies of the tire type, it was found that the SMA tires would be the best selection for the rover (Table ?).

It must not go without saying, however, that the utilization of the SMA tires is not believed to eradicate all risks; these tires are newly developed and have been extensively tested in high-end facilities that simulate planetary surface – but not the actual lunar surface. Therefore, along with other spacecrafts of upcoming Moon and Mars missions[35], Team 27's rover would be one of the first to test out SMA tires in space. Additionally, while SMA metals have the ability to 'remember' their previous shape should they be deformed by irregular terrain while traveling, the extreme cold of the lunar temperatures could lead to possible permanent deformation[36] of the SMA tires until they are exposed to warmer temperatures. Additionally, in considering mass, the SMA tires produced a slightly higher overall cost of $2,700,000 compared to the LRV tires, which produced an overall cost of $2,100,000 (using the L'SPACE Mission Concept Cost Estimate Tool (MCCET)). This is significant in keeping record of to ensure meeting the overall $425 million budget and mechanical subsystem-defined 5% chunk of that budget, $21,250,000.

[31] NASA. Development and implementation of large-scale numerical methods. Accessed November 1, 2024.
[32] "Reinventing the Wheel." NASA. Accessed October 31, 2024.
[33] "Reinventing the Wheel."
[34] "The development of wheels for the lunar roving vehicle."
[35] "The Little Tires That Could... Go to Mars." NASA, July 26, 2023.
[36] "Shape memory strains and temperatures in the extreme." Accessed November 1, 2024.

Tires					
Criteria	Explanation	Grade	Weight	SMA	LRV
Mass	Refers to the mass of a single tire multiplied by 4. Lower mass (preferred) is significant for ensuring the total does not exceed the mission mass constraint of 350 kg.	10 = Mass within 10kg 5 = Mass within 30kg 0 = Mass within 50kg	15%	3	7
Performance	Performance describes how the tire type is believed it will perform based off of facility performance/past mission performance. Important for achieving mission goals.	10 = Excellent performance 5 = Moderate performance 0 = Poor performance	30%	8	7
Reliability	Considers relevance and incorporation of recent technological advancements and potential for success. Also considers historical reliability (if applicable) and corresponding performance.	10 = Highly reliable 5 = Moderately reliable 0 = Unreliable	25%	8	8
Risk	Includes potential for damage to planetary surface, lack of knowledge of type never used before in space exploration, potential for malfunction, etc.	10 = No risk 5 = Moderate risk 0 = Catastrophic risk	30%	8	6
		TOTALS:	100%	72.50%	69.50%

Table 8.The trade study for the tires selection.

1.5.3. Power Subsystem

The power subsystem is the backbone of the spacecraft operations, providing "life" into the mechanical device, and allowing it to fulfill its mission duties and objectives. The following corresponding subsections outline the following: *Power Subsystem Requirements, Power Subsystem Overview,* and *Power Subsystem Trade Studies*, with their corresponding tables.

1.5.3.1. Power Subsystem Requirements

The following table outlines the Power Subsystem Requirements in a "flow-down" structure, beginning with the fundamental constraints and limits outlined in the CC (defined by L'SPACE) and then branching out to its power sub-system requirements (PWR), and its following child requirements such as its sub-assemblies: power generation (PRG), power storage (PRS), and power distribution (PRD). Each flow level corresponds to its color; a darker shade of color signifies a higher level of flow, while a lighter shade signifies a lower level of flow within a foundation.

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Relevant Subsystem	Req met?
CC - 0.6	System cannot have a Radioisotope Thermoelectric Generator (RTG) or any derivative thereof; the System must not exceed the radioactive material cumulative mass of 5g	The usage of the RTG is prohibited, in accordance with the mission.	CC - 0.0 (Mission Requirements Table)	SC - 1.4.2, PWR - 0.0, PRG 1.0,	Inspection	Electrical	Met
SC - 1.4.2	The spacecraft shall have sufficient power to maintain mission operations.	Spacecraft functionality is derived from its power.	SC - 1.1(Mission Requirements Table)	PWR - 0.0	Demonstration	All	Met
PWR - 0.0	The power sub-system shall have enough power to maintain mission operations.	All spacecraft functionality and systems are derived from power.	SC - 1.4.2	PRG - 1.0, PRS - 1.0, PRD - 1.0	Inspection	All	Met
PRG- 1.0	The power sub-assembly: "power generation," shall supply enough voltage, and amperage for instrument operations.	All spacecraft functionality and systems are derived from power.	PWR - 0.0	PRG -1.1	Inspection	All	Met

PRG-1.1	The power generation sub-assembly shall utilize a solar array system with a minimum area of 1.05m^2.	In order to maintain and generate power, the utilization of solar energy for power conversion is beneficial.	PRG - 1.0	—	Demonstration	All	Met
PRS - 1.0	The power sub-assembly: "power storage," shall store and have the capacity to carry the voltage and current required for the electrical system.	In order to derive and reference power, there must be a source or "pool" of power to pull from.	PWR - 0.0	PRS - 1.0	Inspection	All	Met
PRS - 1.1	The power storage sub-assembly shall utilize a storage system that supplies enough power for its subsystems.	In order for the system to operate, a nominal voltage, current, and power draw must be calculated and referenced from the power storage capabilities.	PRS - 1.0	—	Demonstration	All	Met
PRD -1.0	The power sub-assembly: "power distribution," shall distribute the amounts of required voltage and amperage per each sub-system power draw requirements.	In order for the system to operate, power must be distributed to each subsystem or instrument, with its corresponding power draw needs.	PWR - 0.0	PRD - 1.1	Inspection	All	Met
PRD -1.1	The power distribution sub-assembly shall be able to manage and regulate power derived from the power storage and power generator.	Regulation, management, distribution, and utilization of power are critical to ensure all systems are operational.	PRD - 1.1	—	Demonstration	All	Met

Table 9.The power subsystem requirements listed in tabular form.

1.5.3.2. Power Subsystem Overview

The Spacecraft Subsystem: "Power Systems," is the elementary element necessary to fulfill all mission goals and tasks, by supplementing electricity and power to the corresponding instruments and sub-assemblies/components. The power sub-system consists of three sub-assemblies: the power generation sub-assembly, which generates the power necessary to accomplish mission objectives, such as research, navigation, and surveillance; the power storage sub-assembly, which retains and stores power that is not being used, for later utilization; and the power distribution sub-assembly, which distributes the necessary amounts of power at critical periods of instrument utilization, and during periods of necessity, such as in sunlight, or in the dark. Additionally, the power distribution sub-assembly also manages the voltage regulation, by distributing the necessary amounts of voltage for the corresponding instrument's voltage needs.[37] The following table (Table ?.) in the following section reflects the components selected and their properties.

Electrical Engineering								
Component	**Mass**	**TRL**	**Power Draw**	**Voltage**	**Current**	**Dimension**	**Primary/Secondary Vendor**	**Comments / Location**
Lithium Battery Pack from Blue Canyon (2P8S) [x2]	2.4 kg	5	380.8 Wh (Amount Available)	24V - 33.6V (Nominal Voltage: 28V) (Amount Available)	13.6 Ah (Amount Available)	4.1 in x 7.7 in x 3.5 in	Blue Canyon Technologies	It will need to be located in the center or somewhere where it is well-shielded/ and protected from the lunar environment
Triple Junction GaAs Junction Solar Cell [x348]	~0.903 kg	5	1 Wh	2.4V	0.44 Ah	1.05m x 1.05m x 0.04m	AZUR SPACE Solar Power GmbH	The solar array is built based on a solar array surface area requirement of 1.05m²; hence the dimensions are adjusted to reflect that surface area, with an account for design error.
SuperNova PCDU	1.0 kg	5	1500 Wh (Theoretical)	22V -34V (Nominal Voltage: 28V) (Amount Distributed)	50 Ah (Theoretical)	60 mm x 135 mm x 120 mm	Bradford Space	It will need to be well insulated as its operating temperatures are from -30 °C to 60 °C, as well as protected from the lunar environment.

Table 10.The chosen electrical engineering components listed in tabular form.[38]

The following power generation sub-assembly selected a solar array, utilizing a "Triple Junction GaAs Junction Solar Cell". It is manufactured by AZUR SPACE Solar Power

[37] Karen Cunningham, John Carr, and John Lewis, "MSFC Electrical Power Systems for Cubesats" (NASA, November 9, 2018).

GmbH, with a cost of $2,600,000.00 (rounded). The solar array system has a dimension of 1.05m x 1.05m x 0.04m, covering a solar array surface area of 1.05m, and is accompanied by a mass of roughly 0.903kg. The solar array power generation was calculated using a basic array power generation formula, sourced from the National Aeronautics and Space Administration (NASA) template; with roughly 253.1 W/m^2 being delivered to EPS on the lunar surface. Located within the chassis of the rover, it is able to generate power throughout the lunar day, maintaining spacecraft operation power needs, without the need for the battery's power to be discharged during ideal conditions. A drawback to this generation is that due to its reliance on sunlight to generate power, during lunar night periods, the rover's performance is severely limited, with an average cycle of 327.5 sunlight and 327.5 darkness.[39] With all factors in consideration, the Technology Readiness Level (TRL) value given to this sub-assembly is (5), as it has been tested in a space environment before, albeit under different circumstances.[40] The aforementioned utilization of the triple junction cell was on the Spirit and Opportunity rovers[41]; a disadvantage though is because it was utilized on the Mars terrain rather than the Lunar terrain, the TRL had to be adjusted to (5) compensate for the difference in environment, and the difference in sunlight/darkness cycle. Furthermore, the determination of the solar array surface area needed was based on a rough estimate of power needed based on operational capacity; using component power draw as a reference point. Table ? reflects the solar array calculations.

[39] "Moon" (Glenn Research Center, November 22, 2023).

[40] "30% Triple Junction GaAs Solar Cell Type: TJ Solar Cell 3G30C - Advanced Best in Class EOL-Values!"

[41] Ewell, R., B. Ratnakumar, M. Smart, K Chin, L. Whitcanack, S. Narayanan, and S. Surampudi. "Battery Control Boards for Li-Ion Batteries on Mars Exploration Rovers," n.d.

Solar Constant from Environment:	1361.0 W/m²
Solar Cell Efficiency:	30%
Solar Cell Temperature Coefficient:	88%
Solar Cell EOL Environment	93%
Solar Panel Packing Density	90%
Solar Panel AOI	99%
MPPT Efficiency, Line Loss, Diode	85%
Power delivered to EPS	253.1 W/m²
Average power needed from PEL/Profile	200.0 Wh
Add in growth margin	20%
Solar array area needed	0.95m²
Designer margin	10%
Total array needed:	1.05m²

Table 11.Calculations related to electrical component selection.

The following power storage sub-assembly selected was a Lithium Battery Pack (2P8S) utilizing two battery packs, configured in a parallel system (2P). It is manufactured by Blue Canyon Technologies, with a cost estimate of $4,800,000.00 (rounded). The two battery packs have a combined dimension of 4.1 in x 7.7 in x 3.5 in, with .5 inches added in for designer margin, accompanied by a mass of 2.4 kg. The dual battery pack system has the specifications of 24V-33.6V, with a nominal voltage of 28V, and power capacity of 380.8Wh, and a current capacity of 13.6Ah. With all factors in consideration, the TRL value given to this sub-assembly is (5), as a similar Li-Ion battery has been previously tested in a space environment before, albeit under different environments and manufacturers. The aforementioned tested environments were in Opportunity and Spirit rovers.[42]

The following power distribution sub-assembly was selected: the SuperNova PCDU. It is manufactured by Bradford Space, with a cost estimate of $2,500,000.00 (rounded). The PCDU has the dimensions of 65mm x 140mm x 120mm with 5mm added in for designer margin, and is accompanied by a mass of 1.0kg; The PCDU is able to distribute the fol-

[42] Ewell, R., B. Ratnakumar, M. Smart, K Chin, L. Whitcanack, S. Narayanan, and S. Surampudi. "Battery Control Boards for Li-Ion Batteries on Mars Exploration Rovers."

lowing amounts of voltage necessary to each instrument/sub-system based on its voltage draw requirements; the SuperNova PDCU is able interfacing between 28V nominal Li-Ion batteries, a 22V-34V unregulated bus, making it highly compatible with the rover's Blue Canyon Technologies Li-Ion batteries. Additionally, it can charge/discharge a 50A battery current and has a radiation tolerance of 30 krad (qualified to 35krad). A big drawback with this PCDU is the operating temperature of -30°C — 60°C; however, it is manageable with proper insulation.[43] With all factors in consideration, the TRL value given to this sub-assembly is (5) as it has not been tested in a lunar environment, and based on the functionality of the PCDU is within specifications, and from a reputable company/manufacturer Bradford Space, in which some of their other products have been tested/used in space such as their product "Comet 1000" which has launched successfully on the HawkEye 360 Cluster Satellites, and additionally, the SuperNova PDCU has been selected to be the primary PDCU on the Brookhaven National Laboratory mission: LuSEE-Night lunar mission.[44]

1.5.3.3. Power Subsystem Trade Studies

When examining the potential candidates for the power sub-assemblies, the following criteria/factors were accounted for: risk, performance, mass, and dimensions. The reason why the following factors were selected was based on the mission's most limiting constraints as well as to provide lee-way for other contingent spacecraft subsystems, as not to over-exceed the outlined parameters/restrictions. The following table outlines the trade studies for the following sub-assemblies: power generation, power storage, and power distribution; each with 2 potential choices of sub-assemblies, accompanied by a raw score value in terms of unbiased favorability.

[43] "SuperNova PCDU."

[44] "Bradford Space Legacy," n.d., https://www.bradford-space.com/heritage; "Bradford Space's Post," LinkedIn, Bradford Space (blog), n.d.

Criteria	Explanation	Grade	Weight	Blue Canyon Technologies	Pumpkin Space
Cost	The cost estimate of the sub-assembly is crucial to ensure there are enough financial resources distributed for other subsystems; the total cost of the mission should not exceed $425M (CC - 0.4) The cost estimate is based on the MCCET. This is in accounting for two battery packs.	10 = negligible cost 5 = manageable cost 1 = impractical cost	20%	5	8
Risk	The risks include the following parameters involving a multitude of variables including failure threshold, unstable parameters, and any derivative (mass instability, voltage stability, impendence, etc.)	10 = low risk association 5 = medium risk association 1 = high risk association	20%	7	3
Performance	The performance is based on raw data evaluation based on voltage, current, and power in accordance with two battery packs.	10 = high performance 5 = optimal performance 1 = poor performance	20%	9	3
Mass	The mass is the raw value based on two battery packs.	10 = negligible mass limit 5 = manageable mass limit 1 = upper mass limit	20%	6	7
Dimensions	The dimension is based on the raw value in accordance with two battery packs.	10 = negligible dimension limit, 5 = manageable dimension limit 1 = upper dimension limit	20%	6	7
		TOTALS:	100%	68.00%	56.00%

Table 12.Trade study of the power storage component.[45]

The table above reflects the trade study matrix of the power storage component, comparing two similar Li-Ion batteries based on the aforementioned criteria of cost, risk, performance, mass, and dimensions. The cost is an estimate based on the Mission Concept Cost Estimate Tool (MCCET) formula [1516(Electronic Mass)0.74], with Blue Canyon

Technologies costing $4,800,000, and Pumpkin Space costing $1,965,135.84; resulting in a raw value of (5) for Blue Canyon Technologies and (8) for Pumpkin Space. The risks of each battery pack are roughly the same, however, due to the amount of voltage available on the Pumpkin Space Li-Ion battery being lower, it reduces the risk by lowering the insulation requirements, arcing risk, and size; however, the power draw and voltage draw is below specifications thus resulting in a value of (7) for Blue Canyon Technologies and (3) for Pumpkin Space. The performance is where the Blue Canyon Technologies battery shines, as it has a higher voltage capacity of 24V - 33.6V; compared to the Pumpkin Space battery voltage capacity of 10.4V - 16.8V; these variables were accounted for in determining the performance of the battery storage components with the value of Blue Canyon Technologies (9) being more favorable than Pumpkin Space (3). The following next variables are simple raw values, with each grade value being determined by which is better numerically. The mass of the Blue Canyon Technologies Li-Ion battery is 2.4kg while the Pumpkin Space Li-Ion battery is 1.42kg; resulting in a grade value of (6) for Blue Canyon Technologies and a (7) for Pumpkin Space. The dimensions (without accounting for designer margin) of the two batteries are as follows: Blue Canyon Technologies with a dimension of 3.6in x 7.2in x 3.5in and Pumpkin Space with a dimension of 96.8mm x 100mm x 100mm, resulting a value of (6) for Blue Canyon Technologies and (7) for Pumpkin Space.[46]

[46] "6.8 Ah Flight Proven Spacecraft Battery"; "28V Batteries"; "Intelligent Protected Lithium Battery Module with SoC Reporting (BM 2) [SATNow]"; "Intelligent Protected Lithium Battery Module with SoC Reporting (BM2)."

Criteria	Explanation	Grade	Weight	AZUR SPACE	Ecuadorian Space Agency (EXA)
Cost	The cost estimate of the sub-assembly is crucial to ensure there are enough financial resources distributed for other subsystems; the total cost of the mission should not exceed $425M (CC - 0.4) The cost estimate is based on the MCCET. This accounts for the cost of the total amount of cells required to allocate the surface area of $1.05m^2$.	10 = negligible cost 5 = manageable cost 1 = impractical cost	20%	8	6
Risk	The risks include the following parameters involving a multitude of variables including failure threshold, fragility of the solar cell, and operating temperature tolerance.	10 = low risk association 5 = medium risk association 1 = high risk association	20%	7	7
Performance	The performance is based on raw data evaluation based on power efficiency, tolerance, and surface area.	10 = high performance 5 = optimal performance 1 = poor performance	20%	7	7
Mass	The mass is the raw value based on the total amount of cells.	10 = negligible mass limit 5 = manageable mass limit 1 = upper mass limit	20%	7	6
Dimensions	The dimension is based on the raw value in accordance with the surface area of a singular cell. The smaller the cell area; the less power generation loss per area.	10 = compact cell area 5 = mid-size cell area 1 = larger cell area	20%	8	4
		TOTALS:	100%	74.00%	60.00%

Table 13.A second trade study of the power storage component.[47]

The table above reflects the trade study matrix of the power storage component, comparing two similar Triple Junction Solar Cells based on the aforementioned criteria of cost, risk, performance, mass, and dimensions. The cost is an estimate based on the MC-

CET formula [1516(Electronic Mass)0.74] of the combined amount of cells needed; to calculate this, the provided mass in mg/cm2 form, was converted to g utilizing the formula: cell area x area/mass, and then multiplied by the number of cells needed to fulfill the required surface area of 1.05m2. AZUR SPACE had a mass of 2.60g (calculation: 86mg/cm2 converted to 0.086g/cm^2; multiplied by the surface area of 30.18 cm^2) and requires a total of 348 cells (calculation: 1.05 m^2 converted to 10,500cm2; divided by the surface area of 30.18cm2 and then rounded to the largest integer) to fulfill the surface area requirement, resulting in a net mass of 0.903kg; this amounts to a cost of $1,406,011.69. The Ecuadorian Space Agency had a mass of 7.24g (calculation: 120mg/cm^2 converted 0.12g/cm^2; multiplied by the surface area of 60.30 cm2) and requires a total of 175 cells (calculation: 1.05 m^2 converted to 10,500 cm^2; divided by the surface area of 60.30cm2 and then rounded to the largest integer) to fulfill the surface area requirement, resulting in a net mass of 1.266kg; this amounts to a cost of $1,805,411.86 With a disparaging difference in cost values, the following grade values are given: (8) for AZUR SPACE and (6) for Ecuadorian Space Agency. The risks of each solar cell are roughly the same due to both of them being the same design; albeit the AZUR SPACE solar cell is smaller; the surface area required does not change due to their efficiency being the same at 30%; thus resulting in an equal value of (7) for both competitors. The performance of both solar cells is roughly the same, with an efficiency average of 30% for both solar cells, but, the Ecuadorian Space Agency solar cell is slightly better in terms of numerical values of Voc, Iis, and Vmp, however, because of the advantage is so small. It is negligible the corresponding grade value for both competitors is (7). The following next variables are simple raw values, with each grade value being determined by which is better numerically. The mass of the AZUR SPACE solar cell array is 0.903kg while the Ecuadorian Space Agency solar cell array is 1.266kg; resulting in a grade value of (7) for AZUR SPACE and a (6) for Ecuadorian Space Agency. The dimensions are based upon a singular surface area of a cell and do not account for designer margin: 30.18 cm^2 for AZUR SPACE and 60.30 cm^2, with each grading value corresponding to its area; however since the efficiency solar conversion is the same at 30%, they each amass the same total surface area. Because the AZUR SPACE cell's surface area is smaller, if a single cell fails, there is less significant impact due to less surface area loss; resulting in a grade value of (8) for AZUR SPACE and (5) for Ecuadorian Space Agency.[48]

[48]"30% Triple Junction GaAs Solar Cell Type: TJ Solar Cell 3G30C - Advanced Best in Class EOL-Values!"; "Solar Cells (80x80) - Triple-Junction Solar Cells for Space-Grade Panels."

Criteria	Explanation	Grade	Weight	Bradford Space	Ibeos
Cost	The cost estimate of the sub-assembly is crucial to ensure there are enough financial resources distributed for other subsystems; the total cost of the mission should not exceed $425M (CC - 0.4) The cost estimate is based on the MCCET. This is in accounting for a singular power distribution module.	10 = negligible cost 5 = manageable cost 1 = impractical cost	20%	6	8
Risk	The risks include the following parameters involving a multitude of variables including operating temperature threshold, rated voltage, and load capabilities.	10 = low risk association 5 = medium risk association 1 = high risk association	20%	7	5
Performance	The performance is based on raw data evaluation based on power distribution.	10 = high performance 5 = optimal performance 1 = poor performance	20%	9	4
Mass	The mass is the raw value based on a singular power distribution module.	10 = negligible mass limit 5 = manageable mass limit 1 = upper mass limit	20%	6	8
Dimensions	The dimension is based on the raw value in accordance with a single power distribution module.	10 = negligible dimension limit, 5 = manageable dimension limit 1 = upper dimension limit	20%	6	8
		TOTALS:	**100%**	68.00%	66.00%

Table 14.A third trade study of the power storage component.

The table above reflects the trade study matrix of the power storage component, comparing two similar power distribution modules based on the aforementioned criteria of cost, risk, performance, mass, and dimensions. The cost is an estimate based on the MCCET formula [1516(Electronic Mass)0.74] of a single power distribution module; Bradford Space's PDCU has a cost estimate of $2,500,000 while Ibeos has a cost estimate of $353,852.02; resulting in a grade value of (6) for Bradford Space, and (8) for Ibeos. The risk for both systems is similar, however, Bradford Space's triple-redundant battery charge controller gives it an advantage by making it more favorable in terms of robustness and capabilities, giving Bradford Space a grade value of (7) while Ibeos gets a (5). The performance is where Bradford Space's PDCU module really shines, as it can input/output at a higher capacity of 1.5kW compared to Ibeos 250W input capacity, resulting in a grading value of (7) for Bradford Space, and (4) for Ibeos. The following next variables are simple raw values, with each grade value being determined by which is better numerically. The mass for the Bradford Space PDCU is 1.0kg while Ibeos EPS is 1.410-1kg, giving the Ibeos EPS an advantage in terms of mass; the following grade value given is (6) for Bradford Space, and (8) for Ibeos. The dimensions (without accounting for designer margin) of each power distribution are as follows: 60mm x 135mm x 120mm for Bradford Space PDCU, and 96mm x 90mm x 14.5mm for Ibeos EPS; resulting in a grade value of (6) for Bradford Space and (8) for Ibeos.[49]

1.5.4. Command and Data Handling (CDH) Subsystem

The Command and Data Handling (CDH) subsystem integrates high-order hardware with software that enables the spacecraft to operate reliably in the harsh environment of space. The communication interfaces ensure proper effectuation of data transfer and execution of commands, while redundancy and fault tolerance exist as a safeguard for long-duration missions. Capabilities such as these make the CDH subsystem an integral part of the spacecraft that will perform all autonomous data processing, command execution, and operability for the duration of the mission. The following sections will explore this subsystem.

1.5.4.1. CDH Subsystem Requirements

This makes the CDH subsystem very important, as it will be involved in the ability of the spacecraft to manage operations, process commands, and communicate with mission control. The main drivers that will guide the design of the CDH subsystem include strict mass, volume, and power limitations. These are usually determined by how much the launch vehicle of a mission can carry and how much energy overall the spacecraft is budgeted for, where all the subsystems must not only operate within these constraints but also not steal power from other vital sections of the spacecraft. For the CDH subsystem, a limitation in power consumption at 20 watts keeps the system lean yet allows room for other systems, which will be propulsion, thermal management, and scientific instruments, to function correctly. It should also be compact enough to fit into the structural space allotted by the spacecraft. Most of the time, these are constraints driven by a customer or mission designers, who consider the particular mission profile and operational environment of the spacecraft.

The CDH subsystem provides for real-time command execution and transparent com-

[49] "SuperNova PCDU"; "E28-200 28V SmallSat Electric Power System (EPS)," n.d.

munication with other subsystems, necessary to operate the spacecraft effectively. This means it operates commands from the Earth with a maximum latency of one second, derived from the need for a response in the critical phases of the mission. To support industry-standard communication protocols, the CDH subsystem uses I2C, UART, and CAN to interface onboard sensors, actuators, and power systems.[50] These communication norms ensure that data interchange among subsystems is accurate and timely; the information is of great significance in controlling activities on the spacecraft. In addition, the CDH subsystem must safely store high volumes of mission-critical data, such as telemetry and scientific results, based on radiation-hardened memory.[51] Space environments expose the electronics to a high amount of radiations, which tend to corrupt data and destroy components. This sub-system utilizes memory resilient to such conditions; hence, it guarantees the survival of critical data or the possibility to send it back to Earth without corruption or loss. The basis for this requirement is that the standard in usual developments within space industries-NASA and ESA-implies that long-duration space missions require radiation-tolerant hardware.

One of the salient features of the CDH subsystem is redundancy: its fault tolerance upon events of hardware failures ensures mission reliability. In space, where no repairs are possible, the capability for redundancy in key areas such as processing, storage, and communication means that a mission can never be impeded, even upon the failure of a primary system. This high level of redundancy is necessary because this is a long-duration mission and a key factor in achieving high levels of Technology Readiness Levels that ensure the subsystem is able to meet the challenges of the mission over time.[52] Moreover, the CDH subsystem will play a vital role in maintaining communication with the lunar orbiters or relay satellites for the transmission of mission data back to Earth and receiving updated commands. This is a very critical communication functionality for either lunar or deep space missions, where immediate contact with Earth is not possible. The CDH subsystem, besides its communication duties, is responsible for onboard scientific instrument data acquisition, processing, and transmission back to Earth in support of the mission science objectives. Additionally, the system must provide for continuous health monitoring and diagnostics for continued operational integrity. That is very critical when in deep space and on lunar missions. In addition, that shows the extreme level at which a spacecraft is subjected to radiation. The CDH subsystem should be designed to withstand at least 100-krad TID, its operation being required for the entire mission duration. These requirements are inspired by the great experience of space agencies like NASA and ESA, ensuring that the subsystem can handle the harsh realities of space travel.

[50]"An Overview of the Current State of the Art on Small Spacecraft Avionics Systems." In AIAA SCITECH 2022 Forum, p. 0521. 2022.

[51]"RAD750 TM SPACEWIRE-ENABLED FLIGHT COMPUTER FOR LUNAR RECONNAISSANCE ORBITER." In International SpaceWire Conference. 2007.

[52]"Technology readiness levels." White Paper, April 6, no. 1995 (1995): 1995.

Req #	Requirement	Rationale	Parent Req	Child Req	Verification Method	Relevant Subsystem
CDH-00	The CDH subsystem shall adhere to system-wide mass, volume, and power constraints.	Ensures CDH remains within spacecraft resource limits.	N/A	TBR	Analysis, Inspection	All
CDH-01	The CDH subsystem shall process commands within 1 second and support I2C, UART, and CAN protocols.	Required for real-time spacecraft control and inter-subsystem communication.	CDH-0.0	TBR	Testing, Simulation	All
CDH-02	The CDH subsystem shall store at least 256 GB of data using radiation-hardened memory.	Ensures reliable data storage under space radiation conditions.	CDH-0.0	TBR	Testing, Inspection	All
CDH-03	The CDH subsystem shall provide redundancy in processing, storage, and communications.	Enables data relay to Earth and receipt of commands during the mission.	CDH-0.0	TBR	Testing, Analysis	Electrical, Structural, Mechanical
CDH-04	The CDH subsystem shall have a maximum power consumption of 20 W.	Ensures continued operation in case of failure, essential for long-duration missions.	CDH-0.0	TBR	Testing, Simulation	Electrical, Structural, Mechanical
CDH-05	The CDH subsystem shall maintain communication with lunar orbiters and transmit mission data.	Ensures power efficiency and compatibility with spacecraft power systems.	CDH-0.0	TBR	Demonstration, Testing	Payload
CDH-06	The CDH subsystem shall handle and process scientific data from onboard instruments.	Allows proper storage, processing, and transmission of scientific mission data.	CDH-0.0	TBR	Science, Communication	Science, Communication, Mechanical

CDH-07	The CDH subsystem shall provide real-time health monitoring and diagnostics for critical systems.	Protects system integrity in the space radiation environment.	CDH-0.0	TBR	Science, Communication, Mechanical	Science, Communication, Mechanical
CDH-08	The CDH system must withstand a total ionizing dose (TID) of 100 krad for the mission duration.	Required for continuous monitoring and fault detection during operations.	CDH-0.0	TBR	Processing Unit, Data Storage, Communication	Structural, Mechanical

Table 15. The CDH subsystem requirements.

1.5.4.2. CDH Subsystem Overview

The mission would be designed in such a way that it performs a scientific study on the surface of the Moon, gaining data from the lunar surface that is as valuable as possible for advancement of the knowledge about the Moon's composition and the environment around it. This is a mission operating with extreme autonomy, having continuous contact between the space vehicle and Earth Mission Control. All these operations' central entity is the CDH subsystem, which serves as the center for command management, data processing, and communication with other onboard systems and Earth. It supports mission-critical functions such as scientific data gathering, control of the spacecraft's subsystems, and transmission through this sub-system. Thus, the constitution of the CDH will be based on a radiation-hardened processor, data storage systems, communications interfaces, and power management units integrated in such a way as to maintain the autonomy of the spacecraft and to preserve in a kept-safe environment the data processed in space. The paper summarizes the status of the currently designed status of the CDH subsystem; subsequently, it breaks down its subassemblies by looking into their functionality and an assessment in terms of TRL for each component. It also covers a breakdown in terms of mass, dimensions, and power requirements of the CDH subsystem, accompanied by an associated flow chart detailing the subsystem's data-handling architecture.

The Command and Data Handling subsystem is designed with several major subassemblies that integrate together to ensure that the spacecraft can receive commands, manage critical data, and interface effectively with mission control. The processing unit, as the brain of the system, executes commands from Earth and coordinates the spacecraft's real-time operations. It must be radiation-hardened due to the extreme conditions it will face during its space mission, especially from high levels of radiation. It is powered by the RAD750 processor, which is widely used for radiation-hardened microprocessors, operating onboard computation up to 200 MHz and previously being quite reliable during missions such as NASA's Mars Curiosity rover.[53] The Field Programmable Gate Array complements the processor, offering hardware configurations that, given the mission requirements, the spacecraft could adapt to.[54] The flexibility will be very handy regarding

[53]"RAD750 TM SPACEWIRE-ENABLED FLIGHT COMPUTER FOR LUNAR RECONNAISSANCE ORBITER." In International SpaceWire Conference. 2007.

[54]Wang, J. J., R. B. Katz, J. S. Sun, B. E. Cronquist, J. L. McCollum, T. M. Speers, and W. C. Plants. "SRAM

response to changes in the mission environment or objective, since the FPGA provides for changes in hardware functions even post-launch.

One more critical component of the CDH subsystem is data storage, which stores all telemetry, scientific measurements, and command logs securely. It consists of a subsystem that provides at least 256 GB storage capacity; this is provided by radiation-hardened SSDs. The capacity allows the spacecraft to store a volume of mission data accumulated throughout its journey. The SSDs are designed such that they can resist space radiation, retaining data intact without corruption during the mission.[55] Complementing the SSDs, the non-volatile memory ensures that critical data is retained should power be disrupted, and even during system resets.[56] This is significant in the case of deep space missions, where partial loss of data would amount to jeopardizing mission objectives, hence resulting in a chain of drawbacks. The Power Management Unit (PMU) does this through the distribution of power from the primary spacecraft power system to the different components of the CDH for efficient operation. This has to be well regulated because each component would require a stable supply of electricity for its optimal performance. The DC-DC converters in the PMU ensure proper conversion of power provided from the main systems of the spacecraft to the proper voltage levels for each component of the CDH.[57] On the contrary, it is the battery management system within this section that perceives the health and efficiency of the batteries within the spacecraft to guarantee continuation of the power supply without any disruption, even on occasions when the spacecraft itself is not generating any energy-for example, during an eclipse. This system is crucial in keeping the functions running on the CDH, taking into consideration that any stop in power would directly affect performance related to major subsystems. Communication interfaces of the CDH subsystem are crucial in supporting information flow between all the spacecraft subsystems and also in ensuring that communication to Earth remains unbroken. Internal and external communication falls under the logic of this subassembly, which sends out crucial information on telemetry, scientific data, and system health reports. These include the internal communication interfaces connecting the CDH subsystem with other onboard systems like power and propulsion, scientific instruments, etc. In general, external interfaces provide links with mission control on Earth for continuous contact; therefore, data and commands can be exchanged. That necessitates the reliability and low latency of communication interfaces for the large volume of data exchange over huge distances during space missions. Among others, some communication protocols used within the CDH subsystem include Inter-Integrated Circuit (I2C), Universal Asynchronous Receiver-Trasmitter (UART), and Controller Area Network (CAN) bus protocols-which are follow-on large and well-established protocols for their reliability and efficiency. These protocols allow communications between CDH and several sensors and actuators located within every part of the spacecraft to enable smooth coordination of operations regarding the

based re-programmable FPGA for space applications." IEEE Transactions on Nuclear Science 46, no. 6 (1999): 1728-1735.

[55]"Technology readiness levels." White Paper, April 6, no. 1995 (1995): 1995.

[56]"Design of modular power management and attitude control subsystems for a microsatellite." International Journal of Aerospace Engineering 2018, no. 1 (2018): 2515036.

[57]"Design of a mission network system using SpaceWire for scientific payloads onboard the Arase spacecraft." Earth, Planets and Space 70 (2018): 1-9.

spacecraft.[58] I2C is intended for short-distance communication of low-speed devices and therefore is suited best for inter-subsystem data transfers. More complex communicational needs are handled by UART and CAN bus to make sure all subsystems keep up with the critical operations in sync. Moreover, SpaceWire-a high-speed data bus specifically designed for space missions-will enable fast data transfers with negligible latency. The SpaceWire is also supposed to handle the volume of data that emanates from scientific instruments onboard so that the processed information will be forwarded with much success by the CDH subsystem.

The selection of an antenna in the CDH subsystem is a high-gain antenna that operates either at X-band or at Ka-band, which would provide enough bandwidth and range for workable communications over such large distances involved with space missions. High-gain antennas take advantage of the focus in signal strength to enable clear and dependable communications with Earth mission control and efficiently transmit scientific data and telemetry. Optimal placement would be on a stable part of the outside of the spacecraft, positioned for maximum flexibility in its alignment and minimum blockages. This ensures that its line of sight is free, an important factor for maintaining links in a continuous manner. To be more compatible with the subsystem's power requirements, a nominal 28V supply will shift to an unregulated bus range of 24V-33.6V. In doing this, there is less over-voltage concern, hence bringing in stability on the CDH components for continuous operations, especially at peaks of power demand periods or variable conditions of the power supply.

Data integrity assurance with minimal delay in transport over huge distances is indeed one of the critical problems of space communication. SpaceWire responds to this challenge by providing a high level of fault tolerance and data integrity, both critical in ensuring continuous, reliable communication between the spacecraft and mission control. Particularly, this aspect is critical when scientific data is being downloaded from onboard instruments: any loss or corruption of data will undermine the scientific objectives of the mission. Apart from that, the communication interfaces will work together with the redundancy implemented at the spacecraft level in such a way that, even in the event of an failure on any of the subsystems, the communication will be maintained continuously without amber. In addition, the redundancy and fault tolerance contributory features are innate to the CDH subsystem, which allow the continuity of operation of the spacecraft even in the event of any occurrence that results in hardware failure. In deep space flight missions, where repairs cannot be affected, the factors of redundancy become of essence for mission success. The CDH subsystem has duplicate processors, data buses, and power lines to preclude a single-point failure taking out an entire system. Besides having a standby processor, in case of failure of the primary one, a backup processor would automatically take over within a fraction of a second without affecting the operations of the spacecraft. This switch-over would be done automatically, so that the spacecraft can continue its mission uninterruptedly by manual intervention from mission control. The redundant data buses act like alternative paths for subsystems to communicate. In case one of the data buses experiences failure, the CDH subsystem can forward data using

[58]"Evaluation of I2C communication protocol in development of modular controller boards." ARPN Journal of Engineering and Applied Science 11, no. 8 (2016): 4991-4996.

another bus, hence ensuring that the integrity of the communication network is retained.

This is particularly important at mission-critical phases such as during the landing of data collection, where continuous communications between the subsystems are necessary for successful operation. In addition to mechanisms for fault tolerance, the subsystem will continuously monitor and diagnose health, locate potential problems that may become critical, and take preventive action. Diagnostic tools may enable the spacecraft to detect anomalies at an early stage and take corrective actions such as switching over to backup systems or rerouting data, for instance, to prevent mission failure. Such redundancy does not relate to only processing and communication components but further extends to the power management system. The PMU is designed with backup power lines, such that whenever there is failure in a primary power line feeding the CDH subsystem, there is an assurance of a continuous power supply in the CDH system. This ensures that the CDH system is always on, even when there's an unexpected disruption in power distribution. One interesting aspect of the design in the CDH subsystem is that it needs to function in less-than-ideal power environments by scaling back non-essential functions in case the spacecraft operates on a limited power supply. This would keep those few most important components, like the processor and communication interfaces, running even at reduced power availability. Integration with these subsystems empowers the CDH to execute not only complex operations of spacecraft but also to recover from malfunctioned hardware and continue further operations autonomously. Finally, the CDH subsystem is among the most important segments in a spacecraft since it is entrusted with the processing of all onboard data, executing commands from Earth, and maintaining communication between the spacecraft and mission control. Be it a processor, data storage, communication interfaces, or power management unit, each module has its contribution to enable the spacecraft to carry out its tasks for which it was launched into space.

The CDH subsystem integrates high-order hardware with software that enables the spacecraft to operate reliably in the harsh environment of space. The communication interfaces ensure proper effectuation of data transfer and execution of commands, while redundancy and fault tolerance exist as a safeguard for long-duration missions. Capabilities such as these make the CDH subsystem an integral part of the spacecraft that will perform all autonomous data processing, command execution, and operability for the duration of the mission.

Component Name	**Subsystem**	**Function**	**TRL**	**Remarks**
RAD750 Processor	CDH (Command Processing)	Executes spacecraft commands from Earth	TRL 5	Flight-proven in space missions like NASA's Mars Curiosity rover
Field-Programmable Gate Array	CDH (Command Processing)	Customizes hardware functions post-launch	TRL 5	Allows dynamic reconfiguration of the spacecraft's hardware
Radiation-Hardened SSD	Data Storage	Stores mission data, telemetry, and command logs	TRL 5	Tested in space, withstands radiation
Non-Volatile Memory	Data Storage	Retains critical data during power	TRL 5	Common in deep space missions to

		disruptions		ensure data safety
I2C, UART, CAN Bus Protocols	Communication Interface	Manages internal spacecraft data communication	TRL 5	Reliable, industry-standar d protocols for subsystem communication
SpaceWire Data Bus	Communication Interface	High-speed communication between subsystems	TRL 5	Proven for handling large data volumes and low-latency transfers
DC-DC Converters	Power Management Unit	Regulates power supply to CDH components	TRL 5	Flight-proven in spacecraft for power efficiency and stability
Battery Management System	Power Management Unit	Monitors battery health and ensures power flow	TRL 5	Ensures continuous operation during power fluctuations
Backup Processor	Redundancy System	Takes over if the primary processor fails	TRL 5	Standard in long-duration missions for ensuring redundancy and reliability
Redundant Data Buses	Redundancy System	Provides alternate data paths for communication	TRL 5	Ensures uninterrupted communication in case of failure
Diagnostic and Monitoring Tools	Health Monitoring	Continuously checks subsystem health	TRL 5	Detects anomalies and initiates corrective actions

Table 16. The CDH subsystem overview.

The CDH system, which is critical to the success of space missions, is that form of a spacecraft system responsible for command execution, data processing, power regulation, and communications. It characterizes the maturity and operational readiness for every subsystem within the CDH using NASA's Technology Readiness Level, from TRL

1-basic principles observed to TRL 9-proven in operational environments. High TRLs are important features that subsystems operating in outer space should possess because they can assure that the components will be able to sustain extreme conditions in space, including high-level radiation and power fluctuations. Processor RAD750 normally is considered the brain of the CDH and, through the performance on numerous NASA missions, among them Mars rovers, realizes TRL 9. This very high rating reflects its strong performance in commanding and handling critical data under unfriendly conditions. Field Programmable Gate Array: Supporting the processor, FPGA realizes post-launch hardware functionality. Rated at TRL 8, it can be reprogrammed mid-mission, thus adaptive to changes. These processing components underline the high level of readiness required for space operations.

Another critical function of the CDH system is data management, with storage subsystems designed to protect precious mission data. The radiation-hardened SSDs-tested at TRL 7-8-are intended to withstand radiation corruption of data and maintain scientific data integrity. Complementing this, non-volatile memory at TRL 8-9 protects the data in case of power interruption and prevents losses that may occur on deep space missions where power is not as reliable. The strong capability for data retention, along with High-TRL power management, forms the backbone for systems at CDH. Distribution and regulation of power within the spacecraft are done by PMU and BMS. These are mature systems, as the TRLs of PMU stand at 8, and that of BMS is around 7-8; hence, these provide stable power flows even at times of unexpected fluctuation, ensuring a supply of required energy to each subsystem for continuous operability. Mission-critical processes maintain functionality with high-TRL data storage and power management systems.

Communication subsystems aboard CDH ensure seamless data exchange within the spacecraft and with Earth. That means standard I2C, UART, and CAN Bus are at TRL 8, and SpaceWire, necessary for the high-speed, low-latency data transfer, is at TRL 9. These communication interfaces enable the on-board instruments to transmit mission data back to mission control quickly and reliably, an absolute necessity for both scientific and operational success. In the CDH, there is a redundant building, with backup processors and data buses ready to take over should primary systems fail, hence preserving mission integrity. Diagnostic tools monitor system health constantly; early detection of issues allows the spacecraft to perform corrective actions necessary for continued operations. These high-TRL components enhance mission reliability by supporting autonomous functionality and resilience. Thus, TRLs provide a roadmap of maturity for CDH subsystems and reflect rigorous testing and validation necessary for operation in the hostile environment of space.

Component	Function	TRL	Dimensions (cm)	Mass (kg)	Power Requirement (W)
Processor (RAD750)	Command Processing	5	10 x 10 x 2	0.5	10
Data Storage (Radiation-Hardened SSDs)	Data Storage	5	10 x 5 x 2 per SSD	0.6 (0.3 kg per SSD)	2-3
Power Management Unit (PMU)	Power Management	5	15 x 10 x 5	1.5	5
Communication Interfaces (I2C, UART, SpaceWire)	Data Communication	5	-	-	5
Overall CDH Subsystem	Subsystem Integration	5	30 x 25 x 20	5-Apr	20-25

Table 17. Required components for the CDH subsystem.

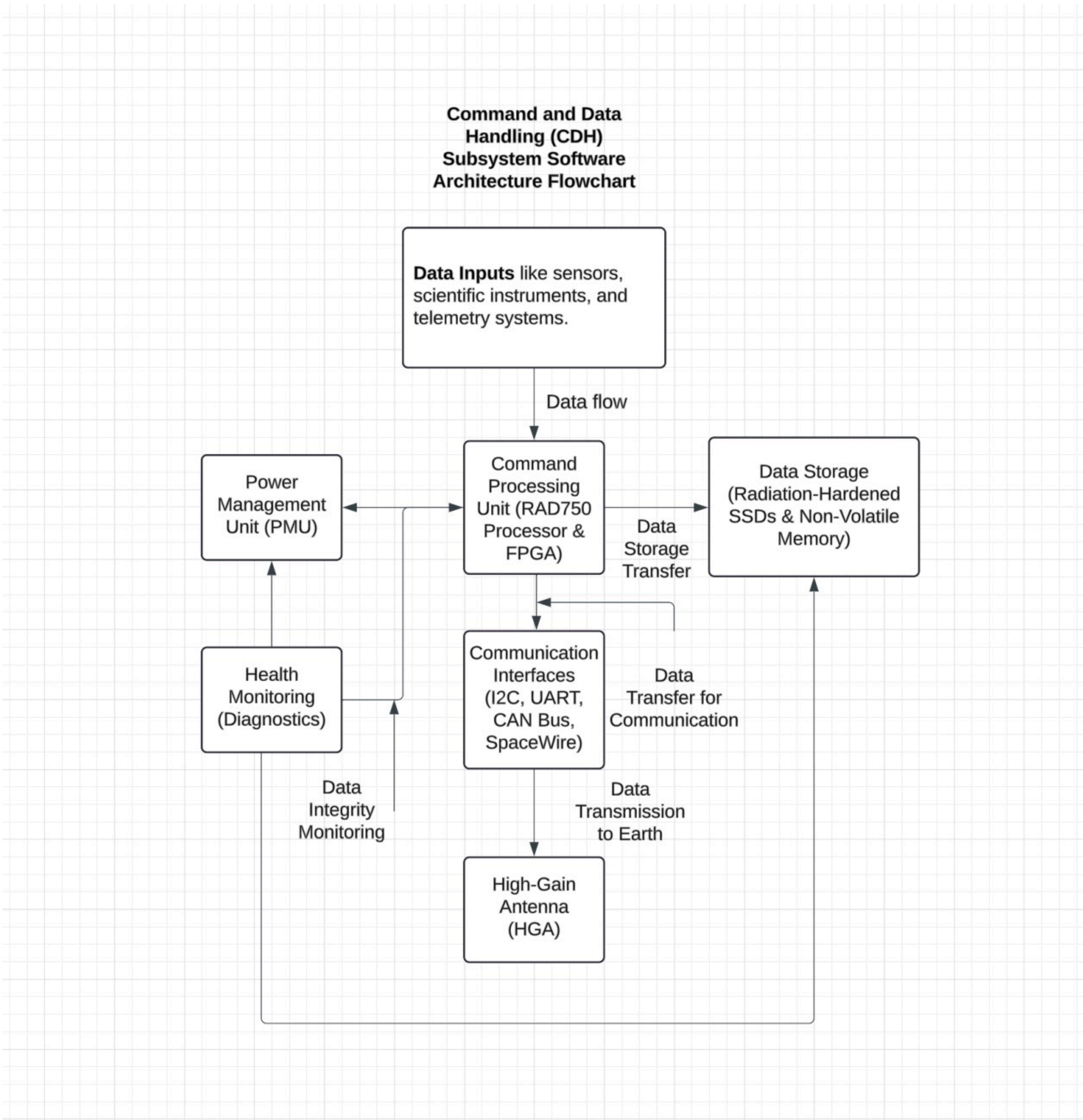

Figure 4. The CDH subsystem software architecture flowchart.

1.5.4.3. CDH Subsystem Trade Studies

The CDH subsystem serves as the backbone of any spacecraft, ensuring the ability to manage operations, process commands, and communicate effectively with mission control. For this design, four key components were evaluated: Command Processing Unit, Communication Interfaces (I2C, UART, CAN), Data Storage (Radiation-Hardened Memory (RHM)), and Redundant Systems. Each component underwent a thorough assessment based on criteria such as cost, risk, performance, and complexity, enabling informed decisions for a reliable and efficient subsystem.

The Command Processing Unit (CPU) is responsible for executing spacecraft commands and ensuring synchronization between subsystems. The three options considered were the RAD750, LEON-3, and Space Cube processors. The trade study evaluated these CPUs based on four criteria: cost, risk, performance, and complexity.

Criteria	Explanation	Grade	Weight	RAD750	LEON-3	Space Cube
Cost	Overall cost of the command processing unit. The total cost should not exceed $425M.Total cost of the CPU unit. Lower cost is preferred to remain within budget.	10 = high, 5 = medium 1 = low 0 = Fail	20%	6	8	4
Risk	Risk associated with potential failures and impact on mission success.	10 = high, 5 = medium 1 = low 0 = Fail	35%	9	7	6
Performance	Ability to meet processing requirements (1 second command execution).Processing speed and ability to handle complex commands efficiently.	10 = high, 5 = medium 1 = low 0 = Fail	30%	8	7	9
Complexity	Complexity of integration and operational usage within the spacecraft.Ease of integration with other subsystems.	10 = high, 5 = medium 1 = low 0 = Fail	15%	7	6	8
		TOTALS:	**100%**	77.5%	70.5%	65.5%

Table 18. A trade study of the command processing unit.

The RAD750 scored the highest overall, primarily due to its strong performance in terms of risk and reliability. Developed[59] by BAE Systems, it is a radiation-hardened processor and has been used in over 200 spacecraft, including NASA's Mars rovers . Its

[59]Mankins, "Technology Readiness Levels."

proven track record made it the top choice for missions requiring durability in harsh space environments. While the LEON-3 offers a cost-effective solution and is commonly used in ESA missions, its lower radiation tolerance and higher risk score placed it behind the RAD750. The Space Cube was the most advanced in terms of performance, offering superior data processing speeds, but its higher cost and unproven flight history made it less desirable. *The RAD750, a radiation-hardened processor developed by BAE Systems, has been used in over 200 spacecraft, including NASA's Mars rovers (Good 2020)*[60]. This connects the claim about the processor to the source.

The weighting was influenced by mission priorities: risk (35%) was the most critical, given the harsh radiation conditions of space, followed by performance (30%) to ensure mission objectives could be met in real time. Cost (20%) and complexity (15%) were secondary considerations but still influenced the decision to select the RAD750, which offered the best balance of reliability and performance.

Communication interfaces enable data transfer between different subsystems within the spacecraft. The protocols evaluated were I2C, UART, and CAN.

[60] Good, "The Eyes and Ears."

Criteria	Explanation	Grade	Weight	I2C	UART	CAN
Cost	Overall cost of the communication interface.Total cost of the communication protocol implementation.	10 = high, 5 = medium 1 = low 0 = Fail	20%	8	7	5
Risk	Risk associated with potential communication failures and data loss.	10 = high, 5 = medium 1 = low 0 = Fail	35%	6	7	9
Performance	Ability to meet data transfer requirements and latency constraints.Data transfer rate and the ability to meet latency requirements for spacecraft operations.	10 = high, 5 = medium 1 = low 0 = Fail	30%	7	6	8
Complexity	Complexity of integration and operational usage within the spacecraft. Ease of implementation and compatibility with other subsystems.	10 = high, 5 = medium 1 = low 0 = Fail	15%	8	7	6
		TOTALS:	**100%**	71%	66%	**72.5%**

Table 19. A trade study of communication interfaces (I2C, UART, CAN).

The CAN protocol emerged as the top choice due to its robustness and high fault tolerance, scoring 9 in risk. CAN is widely used in automotive and aerospace applications, where reliability and fault tolerance are paramount. Its ability to handle multiple devices while ensuring error detection and correction made it an ideal choice for space missions where communication must be fail-safe.

I2C, while more cost-effective and simpler to integrate (rated 8 in cost and complexity), scored lower in terms of risk and performance. It is typically used for shorter-range communication between multiple devices on a single bus but may struggle in high-speed or long-range communications . UART, known for its simplicity and moderate performance, lacked the robustness needed for fault-tolerant communication in space, placing it third.

The weighting for risk (35%) and performance (30%) was prioritized, given the need for reliable data transfer in critical spacecraft operations. Cost (20%) and complexity (15%) were considered secondary, as the robustness and reliability of the communication protocol were deemed more important for mission success.

Reliable data storage is vital for spacecraft to handle mission-critical data. The memory options evaluated included Flash Memory, Static Random-Access Memory (SRAM), and Electrically Erasable Programmable Read-Only Memory (EEPROM).

Criteria	Explanation	Grade	Weight	Flash Memory	SRAM	EEPROM
Cost	Overall cost of the radiation-hardened memory solution.	10 = high, 5 = medium 1 = low 0 = Fail	20%	7	6	8
Risk	Risk associated with data loss and memory corruption due to radiation.	10 = high, 5 = medium 1 = low 0 = Fail	35%	5	9	7
Performance	Ability to store and retrieve data reliably in harsh conditions.	10 = high, 5 = medium 1 = low 0 = Fail	30%	8	7	6
Complexity	Complexity of integration and usage in the spacecraft system. Ease of integration with the overall system.	10 = high, 5 = medium 1 = low 0 = Fail	15%	8	7	6
		TOTALS:	**100%**	66.5%	76%	68.5%

Table 20. A trade study of data storage (radiation-hardened memory).

SRAM was selected due to its high radiation tolerance and ability to store data in real-time applications . It scored 9 in risk, reflecting its resilience in space environments, making it ideal for long-duration missions. Although it is more expensive than other options, its performance (7) and complexity (7) were acceptable, and its reliability outweighed the cost concerns.

Flash Memory, while offering high storage capacity at a lower cost (7 in cost), posed significant risks in terms of radiation vulnerability (5 in risk). EEPROM offered a compromise between cost and risk but lacked the speed required for real-time data processing (scored 6 in performance). Ultimately, SRAM's superior radiation resistance and real-time capabilities made it the preferred choice for the mission.

Given the harsh conditions of space, risk (35%) was weighted heavily, followed by performance (30%) to ensure the storage solution could handle mission-critical data efficiently. Cost (20%) and complexity (15%) were secondary considerations, as reliability and data integrity were the primary concerns.

Redundant systems are essential to ensure mission success in the event of hardware failures. Three options were analyzed: Active Redundancy, Standby Redundancy, and No Redundancy.

Criteria	Explanation	Grade	Weight	Active Redundancy	Standby Redundancy	No Redundancy
Cost	Overall cost of implementing redundancy systems.	10 = high, 5 = medium 1 = low 0 = Fail	20%	4	7	9
Risk	Risk associated with potential failures of redundant systems.	10 = high, 5 = medium 1 = low 0 = Fail	40%	9	7	3
Performance	Effectiveness in maintaining operation during failures of primary systems.	10 = high, 5 = medium 1 = low 0 = Fail	30%	9	6	2
Complexity	Complexity of integrating redundancy systems into existing spacecraft architecture.	10 = high, 5 = medium 1 = low 0 = Fail	10%	6	7	9
		TOTALS:	100%	79%	68%	47%

Table 21. A trade study of redundant systems.

Active Redundancy was selected as it offers the highest reliability (scoring 9 in both risk and performance). Active redundancy means that all systems run simultaneously, ensuring that a backup system can immediately take over in the event of failure, which is crucial for long-term space missions where maintenance and repairs are not feasible. Although it comes with higher costs and complexity (scoring 4 in both), the ability to avoid mission-critical failures justifies the investment. For example, in NASA missions, redundancy is often prioritized to ensure uninterrupted operation of essential systems. According[61] to the NASA Systems Engineering Handbook, redundancy is critical for mission success, particularly for long-duration or high-risk missions (Jackson 2018). Given the mission's high stakes, active redundancy was deemed the best approach to safeguard the spacecraft against hardware failures and ensure operational longevity. This explains why active redundancy was selected in the trade study, emphasizing its importance despite its higher costs and complexity.

The trade studies conducted for the Command and Data Handling subsystem revealed a thoughtful and balanced approach to meeting the mission's stringent requirements. Each selected component was carefully evaluated to ensure optimal performance, reliability, and cost-effectiveness. The RAD750 was chosen as the Command Processing Unit due to its proven reliability in demanding space environments. For communication interfaces,

[61] Jackson, "NASA Systems Engineering Handbook."

I2C emerged as the best option, offering a practical balance of simplicity and cost savings while maintaining effective data transfer. In terms of data storage, SRAM was favored for its capability to handle real-time data processing, ensuring the system can respond swiftly to mission demands. Lastly, the decision to implement Active Redundancy underscores a commitment to mission reliability, ensuring that critical operations can continue seamlessly even in the event of hardware failures. Together, these selections provide a robust foundation for the CDH subsystem, capable of withstanding the challenges of space exploration.

1.5.5. Thermal Management Subsystem

The thermal subsystem is responsible for the proper functioning of all instrumentation within the rover and influences the operation of every sub-assembly. The following subsections provide details on the following topics: Thermal Management Subsystem Requirements, Thermal Management Subsystem Overview, and Thermal Management Subsystem Trade Studies.

1.5.5.1. Thermal Management Subsystem Requirements

The following table outlines the thermal management subsystem requirements in a "flow-down" structure derived from the customer's constraints outlined in the L'SPACE Mission Task Document. These requirements will guide the thermal engineers in selecting elements for the rover.

Criteria	Explanation	Grade	Weight	Active Redundancy	Standby Redundancy	No Redundancy
Cost	Overall cost of implementing redundancy systems.	10 = high, 5 = medium 1 = low 0 = Fail	20%	4	7	9
Risk	Risk associated with potential failures of redundant systems.	10 = high, 5 = medium 1 = low 0 = Fail	40%	9	7	3
Performance	Effectiveness in maintaining operation during failures of primary systems.	10 = high, 5 = medium 1 = low 0 = Fail	30%	9	6	2
Complexity	Complexity of integrating redundancy systems into existing spacecraft architecture.	10 = high, 5 = medium 1 = low 0 = Fail	10%	6	7	9
		TOTALS:	100%	79%	68%	47%

Table 22. The thermal management subsystem requirements.

1.5.5.2. Thermal Management Subsystem Overview

The Spacecraft Thermal Management Subsystem is an essential component required to maintain the rover at the appropriate operating temperature as specified in the thermal subsystem requirements section. This subsystem supports systems such as the Electrical and Scientific sub-assemblies by ensuring that their instruments operate within the correct temperature range, facilitating accurate readings and enhancing the longevity of the sub-assemblies.

According to the heat flow map (Figure ?), the rover will consistently experience heating if it relies solely on a passive Thermal Control System (TCS). Therefore, an active TCS will be required. Specifically, an active system must provide a cooling power of 300 K to maintain a 0 W load on the system at any given time. Below are the devices selected for the construction of the rover that will study the lunar surfaces.

Component	Mass	TRL	Power Draw	Vendor	Comments
CryoTel DS Mini	1.2Kg	5	1.8W	SunPower – Ametek	This will be the main active cooler.
MLI	18.03Kg	5	0	Quest Thermal Group	N/A

Table 23. The devices selected for the thermal management subsystem.

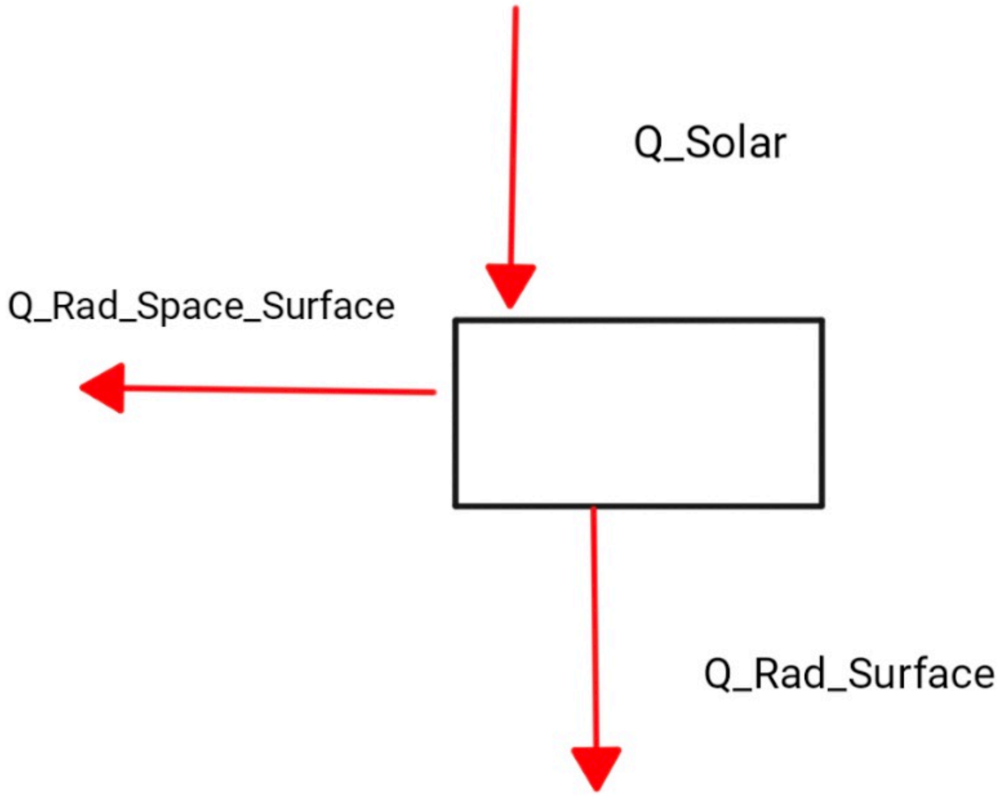

Figure 5. Heat map.

1.5.5.3. Thermal Management Subsystem Trade Studies

When comparing various instrumentation, radiators and heaters, for the TCS the following criteria were followed to choose the most adept solution. Mass, Cost, Reliability, Performance. The reasons these criteria were chosen was based on the customer's constraints. The following trade studies tables outline the following subsystems of the cooling apparatus.

Criteria	Explanation	Grade	Weight	Meyer Tool & MFG	Quest Thermal Group
Mass	The less mass the better	10 = high, 5 = medium 1 = low 0 = Fail	25%	0	6
Cost	The mission is constrained by cost	10 = high, 5 = medium 1 = low 0 = Fail	40%	6	6
Reliability	TCS is a crucial system that keeps all other subsystem s operational	10 = high, 5 = medium 1 = low 0 = Fail	15%	10	10
Complexity	The complexity of the Element will impact the risk of the mission	10 = high, 5 = medium 1 = low 0 = Fail	20%	10	10
		TOTALS:	100%	59%	74%

Table 24. Trade study of the multi-layered insulator.

Meyer Tools & MFG (MTM) did not include any information about their multilayer insulation (MLI) weight in the data sheet[62]. As participants in the L'SPACE program, requesting additional details was not permitted, which limited the available approach. Due to this lack of information, MTM received a score of zero. In contrast, Quest Thermal Group (QTG) provided a detailed data sheet for their various MLI product lines, making it straightforward to calculate the total weight of the MLI[63]. QTG will receive a score of 8.

To calculate the cost of the MLI, the formula from the MCCET will be utilized, requiring the total Thermal Mass. The calculation is as follows: 642 X (ThermMass)0.62=Total

[62] Meyer Tool & Mfg. "Multi Layer Insulation (MLI) Vacuum Insulation." Accessed October 26, 2024.

[63] 1. Alan Kopelove and Scott Dye, "Integrated MLI (IMLI) Product Sheet," Quest Thermal Group, accessed October 26, 2024.

Cost of MLI. To determine the total Thermal Mass, the formula provided by the Thermal Engineers in the second meeting is used: Area X Layers. Here, “Area” refers to the rover’s surface area that the MLI will cover, currently 0.6641 m^2. The number of layers is based on the chart on page 166 of the Spacecraft Thermal Control Handbook (STCH), which specifies the effective emittance needed for operation as 0.005. According to the STCH, achieving this emittance requires 28-30 layers. Selecting 28 layers allows for a lighter material without sacrificing emissivity. With an area of 0.6641 m^2 and 28 layers, both materials have the same weight of 19.9 kg. Substituting this Thermal Mass into the MCCET formula yields a total cost of $6,071,808.60. Since both companies cost the same, a score of 6 was assigned to each.

For complexity, both MTM and QTG received a score of 10. From a materials standpoint, neither requires active control by the rover to operate, as they are passive materials without moving or electrical components. This characteristic significantly reduces the risk of failure. The primary concern is ensuring that the material does not catch on any surfaces that could damage or detach it from the rover’s chassis.

In terms of reliability, both materials are passive elements that require no input from the rover to function. Once installed, the primary concern is to protect the MLI from any physical damage. Therefore, both MTM and QTG received a score of 10 for reliability.

For TRL, this MLI technology scores a 5. MLI has been a standard tool in the aerospace industry since NASA’s inception and is widely used across various spacefaring vehicles, not limited to NASA. It’s a well-established, mature technology. However, it has yet to be tested on a lunar rover designed for lunar exploration, so it will face this specific environment for the first time, which contributes to its moderate TRL score.

The rover developed by Team 27 will utilize QTG’s MLI for the exploration of lunar caves.

Criteria	Explanation	Grade	Weight	Lockheed Martin Space - MICRO1-2	SunPower – Ametek: CryoTel DS Mini
Mass	The less mass the better	10 = high, 5 = medium 1 = low 0 = Fail	25%	7	6
Cost	The mission is constrained by cost	10 = high, 5 = medium 1 = low 0 = Fail	40%	0	9
Reliability	TCS is a crucial system that keeps all other subsystems operational	10 = high, 5 = medium 1 = low 0 = Fail	15%	3	0
Performance	It should consume the least amount of wattage	10 = high, 5 = medium 1 = low 0 = Fail	20%	5	2
		TOTALS:	100%	32%	55%

Table 25. Trade study of the cooling element.

The cryocooler will be required to cool the rover, as it will operate at elevated temperatures. This trade study will compare the Lockheed Martin Space MICRO1-2 cryocooler and the SunPower Ametek CryoTel DS Mini cryocooler.

For the mass criteria, Lockheed Martin (LM) provides a detailed data sheet; however, the specific mass information is not included. NASA's website indicates that the Lockheed Micro 1-2 weighs 0.475 kg, resulting in a score of 7 for mass. SunPower (SP) includes the mass of its cooler, which is 1.2 kg, in its data sheet, earning SP a score of 6 in the mass category.

In terms of cost, the same MCCET equation will be applied. The total cost for LM's device is $624,956.90, resulting in a score of 8 for this category. Applying the MCCET formula to SP's cooler yields a cost of $1,110,161.05, earning SP a score of 7 in the cost category.

Reliability is a crucial criteria for an active system, as inherent risks are associated with their operation. The scores for the two companies are as follows: LM receives a score of 3 for reliability, while SP receives a score of 0. SP does not provide information regarding the reliability of their cryocooler[64]. In contrast, LM's data sheet indicates that reliability considerations were taken into account during the design of their cryocooler[65]. However, these claims will need to be validated through testing to ensure their accuracy.

The performance category will be evaluated using the formula: Cooling Power % Wattage. For this criterion, Lockheed Martin will receive a score of 5, as their cooler consumes 25 W while providing a cooling effect of 105 K[66], resulting in a ratio of 4.2. In contrast, SunPower receives a score of 2, as its device consumes 45 W while achieving a cooling effect of 77 K, yielding a ratio of 1.71.

In terms of TRL, both devices will be the first to undertake a mission on the Moon, exploring caves and gathering samples for study. According to NASA's website, LM's device is rated as TRL 7, while SP's device has a TRL of 6. However, since both will operate in different missions and environments, the TRL for both technologies will be adjusted downward to 5 and 4, respectively.

The rover developed by Team 27 will utilize SP's Cryocooler for the exploration of lunar caves.

For more of the math worked out to complete the thermal management subsystem, please see the appendix.

1.5.6. Payload Subsystem

1.5.6.1. Payload Subsystem Requirements

The payload subsystem requirements are presented in a table, with darker blue requirements representing the higher-level requirements and the lighter blue requirements representing lower-level requirements related to those at higher levels. The requirement highlighted in yellow showcases the instrument (INT) and the science objective that is related to the explanations in section 1.7.4. *Change Control*.

[64] CryoTel DS Mini. Accessed October 26, 2024.
[65] "General Design Standards." Lockheed Martin Space Systems Company. Accessed October 26, 2024.
[66] "7.0 Thermal Control - NASA." Accessed October 26, 2024.

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Relevant Subsystem	Req met?
INT-1	The Carbothermal Reactor shall provide heat in the temperature range from 1273 Kelvin to 1373 Kelvin.	In order to extract 10% of oxygen from the lunar regolith, this instrument must provide the necessary temperature of heat for the objective to be successful.	-	-	Demonstration	Thermal, Mechanical	Met
INT-2	The 3-Axis Fluxgate Magnetometer and the Bartington Fluxgate Magnetometer shall measure magnetic fields in the nanoTesla range of 0 nT to 23 nT.	The magnetometers must be able to detect the magnetic fields occurring near the lunar pit which have a specific detection range of 0 nT to 500 nT.	-	INT-2.1	Demonstration	Mechanical	Met
INT-2.1	The 3-Axis Fluxgate Magnetometer Model AP235 shall operate with a low noise level range of -0.05 RMS $nT/\sqrt{Hz}$ to 0.05 RMS $nT/\sqrt{Hz}$.	The magnetometers themselves can transmit noise through their mechanical and electrical subsystems in operation. Thus, their noise levels must be sufficiently low to mitigate data corruption.	INT-2	-	Demonstration	Electrical, Mechanical	Met
INT-3	The Active Radiation Environment Sensor shall detect radiation levels between 0 W/m^2 and 1,370 W/m^2.	In order to measure the radiation levels of the lunar pit, the instrument must be able to detect a sufficient range of radiation.	-	INT-3.1	Demonstration	Mechanical	Met
INT-3.1	The Active Radiation Environment Sensor shall detect Alpha, Beta, and Gamma radiation.	The radiation sensor must be able to detect different types of radiation in order to discover the radiation environment of the lunar pit.	INT-3		Analysis	Mechanical	Met
INT-4	The Model HMP60 Humidity Sensor + Model DI-808 shall detect a temperature range of 16 °C to 18 °C.	In order to measure the humidity levels of the lunar pit, the instrument must be able to detect the specific temperature range of the pit.	-	-	Demonstration	Mechanical	Met
INT-5	The PT1000 Class F0.3 shall detect a temperature range of 16 °C to 18 °C.	In order to measure the humidity levels of the lunar pit, the instrument must be able to detect the specific temperature range of the pit.	-	-	Demonstration	Mechanical	Met
INT-6	The PT1000 Class F0.3 shall detect a pressure range of 10^{-12} torr to 10^{-10} torr.	The pressure sensor must be able to measure the specific pressure range of the lunar pit.	-	-	Demonstration	Mechanical	Met
INT-7	The MID-360 LiDAR shall detect a measurement range of up to 55 meters.	In order to measure the dimensions of the lunar pit, this instrument must have a sufficient detection range.	-	INT-7.1	Demonstration	Mechanical	Met
INT-7.1	The MID-360 LiDAR shall take measurements of the lunar pit's dimensions every 45 meters.	The instrument should measure the lunar pit's dimensions frequently so as to not miss any areas of the pit.	INT-7	-	Test	CDH, Mechanical	Met
INT-8	The Silicon Photomultiplier shall detect the thickness of the lunar pit every 40 minutes.	In order to accurately measure the structure of the lunar pit, the instrument must take measurements frequently.	-	-	Test	CDH, Mechanical	Met

Table 26. The payload subsystem requirements for the mission.

1.5.6.2. Payload Subsystem Overview

The instruments selected are derived from the mission's five scientific objectives:

1. To extract 10% of oxygen from regolith in lunar conditions.
2. Find the location of magnetic fields on the lunar surface, specifically on/near lunar swirls.
3. Measure temperature, humidity, and pressure fluctuations in the Mare Ingenii pit during radiation level changes.
4. Characterize the terrain variation, depth, height, and ease of access within the Mare Ingenii pit.
5. Examine the structural integrity of the inside of the pit.

The first science objective has been determined to be outside the mission's scope, and as a result, the carbothermal reactor has been omitted from the instruments list. This instrument isn't included on the list due to its mass, dimensions, and max power draw being too large compared to the restraints of the mission.[67] The second science objective will be investigated using a 3-Axis Fluxgate Magnetometer Model AP235 as a primary instrument and a Barington Fluxgate Magnetometer, 3-axis, Aerospace/Space as a secondary instrument. Both instruments will be used to measure the magnetic field strength of the lunar surface, including lunar swirls surrounding the Mare Ingenii pit.

The instruments selected for the third science objective include a Model HMP60 attached to a Model DI-808, a Lunar Outpost Canary-S, and a PT1000 Class F0.3 joined with a Pressure-Temperature instrument. These instruments will measure the pit's temperature, humidity, and pressure throughout the mission. A silicon photomultiplier was selected as a primary instrument to measure the radiation found in the pit, and an Active Radiation Environment sensor was selected as a secondary instrument. The final two science objectives will be tested using a MID-360 LiDAR and a Silicon Photomultiplier respectively. These two instruments will use flux photons to measure the pit's dimensions and structural integrity and are used as secondary instruments for each other.

The 3-Axis Fluxgate Magnetometer Model AP235 is an instrument used to measure the magnetic field in nanoteslas.[68] This fluxgate magnetometer has a very low noise level, which is imperative to the accuracy of the information relayed back to researchers. Similarly, this instrument boasts the highest sensitivity possible, which ensures a very accurate reading down to the slightest magnetic field strength change. Although the dimensions, weight, and power draw aren't the most ideal, this instrument's accuracy makes it the best choice to research magnetic field strength on the lunar surface (Table ?). This instrument has a TRL rating of 7 due to being used on previous lunar missions, such as the Lunar Surface Magnetometer (LSM) during Apollo 16.[69] Although the dimensions, weight, and technological advancedness differ, the concept of a 3-axis magnetometer is the same. The 3-Axis Fluxgate Magnetometer Model AP235 is mounted on the vehicle using a CubeSat for the most accurate readings. This is similar to the LSM, which astronauts placed onto the lunar surface on a 3-legged mount.

The Bartington Fluxgate Magnetometer was selected as a secondary instrument since the sensitivity and noise levels aren't specified.[70] Although the low noise is highlighted in the description, it doesn't seem to have the same level of accuracy as the primary magnetometer selected for this scientific objective. This instrument is less expensive, smaller, lighter, and functions at a lower temperature range, which are very important factors when choosing an instrument for a low-cost mission. This instrument is a TRL 7 for very similar reasons to the 3-Axis Fluxgate Magnetometer Model AP235. The Bartington Fluxgate Magnetometer is a 3-axis fluxgate magnetometer as well and can be compared to the LSM previously mentioned. A significant difference between the two is that this instrument isn't mounted, which plays a factor in having a higher noise level than the primary instrument.

As the primary instrument for the third science objective, the Model HMP60 attached to a Model DI-808 has proven its value due to its small size and weight, extensive temperature range, and low price point. This instrument measures the relative humidity and

[67] "Carbothermal Reduction Demonstration (CaRD)." Accessed October 28, 2024.
[68] "Model AP235 - Applied Physics Systems," December 6, 2023.
[69] NASA Space Science Data Coordinated Archive. "Lunar Surface Magnetometer." Catalog.
[70] "Model AP235 - Applied Physics Systems," December 6, 2023.

temperature of the environment using a sensor attached to a data logger. Although other options exist for measuring the environment's weather, this instrument's cost is unbeatable since it is also an accurate tool. A TRL 4 has been given to this instrument since this type of sensor has never been tested in a lunar-like environment. The Model HMP60 would likely withstand the environment since it has operating temperature ranges similar to other instruments selected for this mission. Still, it's unknown whether it could withstand other elements found in the lunar environment.

A Lunar Outpost Canary-S has been selected as a secondary instrument for measuring the temperature of the lunar environment because of the various tools it can utilize. This instrument measures the temperature, relative humidity, and pressure while also having space for up to three extra sensors.[71] Although this instrument is heavier, weighs and costs more, and has a higher power draw, it acts as a great secondary instrument in the event that the Model HMP60 fails. The Lunar Outpost Canary-S is a TRL 4 because the instrument has never been tested in a similar environment to the moon. This instrument would be an outstanding primary instrument for testing multiple characteristics of the environment if it were undoubtedly capable of withstanding the harsh lunar environment.

The PT1000 Class F0.3 joined with a Pressure-Temperature (PT) instrument, has been chosen as a secondary instrument for measuring the pressure of the lunar environment. This instrument is the most expensive, heaviest, and largest dimensions on the instrument list and has few capabilities. Although it has many downsides, it would still assist the Model HMP60 if it is no longer functioning and unable to provide the data for this science objective. Since this instrument is primarily used on airplanes rather than lunar rovers, it is a TRL 4. The PT and sensor are functional and notable instruments for harsh environments in the Earth's atmosphere, but the moon's high radiation levels and low temperatures could prove detrimental.

A Silicon Photomultiplier (SiPM) has been selected as a primary instrument to test for radiation and the structural integrity of the Mare Ingenii pit.[72] These are two of the most important scientific objectives of research on the moon, as they're the main factors in determining the possibility of human habitation and sustainability on the lunar surface. For this mission, the SiPM will collect muon data throughout the pit, creating a 3D map to show the structural integrity and dimensions. The dimensions, mass, and max power draw have yet to be determined as a scintillator using multiple SiPMs is planned to be created. Upon completing the mechanical build of the rover, these factors will be addressed and updated. This instrument is a TRL 5 since it has been used on similar space missions, such as MoonBEAM, to investigate the creation of black holes.[73] Although the relevant environment isn't exactly the same as that of lunar rover exploration, the SiPM has shown its capabilities to withstand harsh space temperatures.

Acting as a secondary radiation detection device, an Active Radiation Environment Monitors (REM) instrument has been selected because of its small weight and size and its capability to monitor the flux of muons similar to the SiPM.[74] This instrument has a

[71] "Lunar Outpost Canary-S." Accessed October 30, 2024.
[72] "AFBR-S4K33C0135L." Accessed October 30, 2024.
[73] "MoonBEAM: A Beyond Earth-Orbit Gamma-Ray Burst Detector for Gravitational-Wave Astronomy." March 18, 2018.
[74] "Tools Used in Space Radiation Operations." Accessed October 30, 2024.

low power consumption, which allows the rover to measure the radiation constantly while simultaneously running temperature, pressure, and humidity checks. The REM instrument is a TRL 5 because it detects radiation levels in Commercial Lunar Payload Services (CLPS) missions. The REM has been used on Artemis 1 and is planned to be used on Artemis 2, showing its capabilities of handling the lunar environment.

Lastly, the MID-360 LiDAR will be used to conduct the fourth science objective by creating a 3D model of the Mare Ingenii pit's dimensions, similar to the SiPM.[75] The capabilities of this instrument are a large reason it was selected as a primary for this science objective, as it has a field of view of 360 x 59°. This allows the MID-360 LiDAR to create a complete map of the pit after taking less than 5 samples, which can ensure accurate data between this instrument and the SiPM. This instrument is a TRL 4 because this model has never been used on the moon. Many lunar rovers have used LiDAR for navigational purposes because it helps them detect obstacles to avoid early on. This is solid evidence that this instrument would serve as an excellent primary instrument for the last science objective after being tested in a relevant environment.

The overall payload system is currently a TRL 4 because of the PT, Lunar Outpost Canary-S, and Model HMP60's low TRL. Even though half of the instruments selected for this mission are a TRL 5 or 7, since four of the mission's instruments have never been tested in a relevant environment, it brings down the overall payload system's TRL. Many of the instruments that have been selected must be thoroughly tested to reach a higher overall TRL before being implemented on this mission's rover, which will allow for a more scientifically accurate mission.

Instrument	Mass	Dimensions	Max Power Draw
Model HMP60 Humidity Sensor + Model DI-808	99 g + 453 g	(36 x 36 x 9 mm) + (13.81 x 10.48 x 3.81 cm)	10W
Lunar Outpost Canary-S	2.27 kg	21.59 x 17.78 x 15.24 cm	20W
PT1000 Class F0.3 + Pressure-Temperature	6 kg	5.0 x 2 x 1.3 mm + 304.8 mm^3	250W
Silicon photomultiplier	TBD	TBD	TBD
Active Radiation Environment Monitors	2.01 kg	18.69 x 14.48 x 5.46 cm	3.6W
MID-360 LiDAR	265 g	65 x 65 x 60 mm	6.5 W
3-Axis Fluxgate Magnetometer Model AP235	150 g	10.52 x 4.06 x 2.86 cm	3.24 W
Bartington Fluxgate Magnetometer, 3-axis, Aerospace/Space	94 g	20 x 20 x 20 mm	0.175 W

Table 27. Table of instruments selected for the mission along with their mass, dimensions, and power draw.

[75] "Livox Mid-360." Accessed October 30, 2024.

1.5.6.3. Payload Subsystem Trade Studies

Choosing between primary and secondary instruments for each science objective is important as they rely on multiple factors. For the mission's structural integrity science objective, it was decided after considering five criteria, which include reliability, manufacturing information, mapping capabilities, temperature control, and heritage(Table x). The reliability has been given the largest weight since the failure of the instrument would prove the mission to be purposeless. The instrument's manufacturing information is weighted second highest due to a need for reproducibility. If the mass, dimensions, and max power draw aren't known, replicating the instrument is nearly impossible. The mapping capabilities criteria have been chosen to represent the ideal goal of the science objective: to create a 3D map. A temperature control criteria were established due to the lunar environment's very low temperature at night. Lastly, heritage was decided to be the lowest weighted criterion of 10% because of the ability to raise an instrument's TRL higher after going through testing despite it not being a heritage instrument.

Despite its low heritage score, the Silicon Photomultiplier became the primary instrument because all other scores are close to their max value.[76] The MID-360 LiDAR was established as the secondary due to it not being a heritage instrument and mid-to-low reliability and temperature control scores. These scores were decided since the MID-360 LiDAR has never been used on a lunar mission, which could prove to be an issue later.[77] The Lunar Penetrating Radar is an instrument with a perfect score in the heritage and reliability criteria, but it was chosen not to be used on the mission due to the difficulties of finding manufacturing information and less than ideal temperature range.[78]

The weights for the radiation trade studies table criterion were established similarly to that of the structural integrity other than replacing the mapping capabilities with radiation detection because of the change in overall objective. The same reasoning stands for the importance of the previous criterion, but the radiation detection has a weight of 15% because of the importance of gathering this data for the mission. Without the radiation level data, the safety of lunar human habitation is unknown, and the mission would be nearly purposeless. A silicon photomultiplier was selected as the primary instrument for this science objective due to its perfect score in all categories other than reliability and heritage. The instrument's heritage score carries over from the last trade study, but the reliability score was lowered when compared to the Lunar Lander Neutron and Dosimetry instrument's reliability.[79] The Active Radiation Environment Sensor has been selected as a secondary instrument due to its higher manufacturing information and temperature

[76] "AFBR-S4K33C0135L." Accessed October 30, 2024. https://www.broadcom.com/products/optical-sensors/silicon-photomultiplier-sipm/afbr-s4k33c0135l.

[77] "Livox Mid-360." Accessed October 30, 2024. https://www.livoxtech.com/mid-360.

[78] "Pattern Analyses of Lunar Penetrating Radar Images at Chang'E□3, 4, and 5 Landing Sites: A New Insight Into the Evolution of Lunar Regolith - Liu - 2023 - Journal of Geophysical Research: Planets - Wiley Online Library." Accessed November 1, 2024. https://agupubs.onlinelibrary.wiley.com/doi/full/10.1029/2022JE007675?saml_referrer=.

[79] Wimmer-Schweingruber, Robert F., Jia Yu, Stephan I. Böttcher, Shenyi Zhang, Sönke Burmeister, Henning Lohf, Jingnan Guo, et al. "The Lunar Lander Neutron and Dosimetry (LND) Experiment on Chang'E 4." Space Science Reviews 216, no. 6 (August 18, 2020): 104. https://doi.org/10.1007/s11214-020-00725-3.

control scores compared to the Lunar Lander Neutron and Dosimetry instrument.[80]

The microclimate measurements trade studies table's weight assignments are the same as the previous table other than replacing the radiation detection with accuracy. The accuracy of temperature and pressure changes is very important because a noticeable temperature variation can be smaller than 1°C. The choice to select the Model HMP60 Humidity Sensor + Model DI-808 as the primary despite its low-temperature control and heritage scores and because of its higher overall score. The secondary instrument for this science objective is the Lunar Outpost Canary-S because of it's overall above-average scores in all criterion.[81] Lastly, the magnetic fields trade study table's criterion weight was carried over from the previous trade studies table, as accuracy is just as important as the microclimate's accuracy.

Structural Integrity						
Criteria	**Explanation**	**Grade**	**Weight**	**Silicon Photomultiplier**	**MID-360 LiDAR**	**Lunar Penetrating Radar**
Reliability	The likelihood of each instrument's ability to successfully measure the dimensions and structural integrity of the Mare Ingenii pit.	10 = Lowest possibility of failuare, 5 = Somewhat high chance of failiure, 1 = Very high likelihood of failure, 0 = Impossible	40%	9	7	10
Manufacturing Information	Information such as mass, dimensions, and power draw are very useful in determining the manufacutring of the instrument.	10 = All information known, 5 = Some information, 1 = Little information, 0 = No information	25%	9	10	4
Mapping Capabilities	This is determined based on how well the instrument can create a 3D map to best understand the structural integrity.	10 = Create a full 3D map, 5 = Measures the dimensions accurately, 1 = Can measure distance, 0 = Isn't a measuring tool	15%	10	9	9
Temperature Control	Due to the harsh lunar temperatures, the instrument's capability to withstand a broad range of temperatures is essential.	10 = Can withstand near lunar temperatures, 5 = Moderately close to lunar temperatures, 1 = Temperatures aren't close to lunar temperatures, 0 = Nonoperational	10%	10	6	7
Heritage	Instruments previously used on the lunar surface are considered the highest level of heritage because they're known to be successful in the environment.	10 = Used on a previous lunar rover, 5 = Used in a slightly similar space mission, 1 = Uncessfully used/mission was cancelled, 0 = No heritage	10%	7	0	10
		TOTALS:	100%	90.50%	72.50%	70.50%

Table 28. The trade studies table used to decide the primary and secondary instruments to test the Mare Ingenii pit's structural integrity.

80 Whitman, Kathryn, Phil Quinn, Ramona Gaza, and Shaowen Hu. "Tools Used in Space Radiation Operations." Accessed October 30, 2024. `https://ntrs.nasa.gov/citations/20230004380`.

81 "Lunar Outpost Canary-S." Accessed November 1, 2024. `https://www.aqmd.gov/aq-spec/sensordetail/lunar-outpost-canary-s`.

Radiation						
Criteria	Explanation	Grade	Weight	Silicon Photomultiplier	Active Radiation Environment Sensor	Lunar Lander Neutron and Dosimetry
Reliability	The likelihood of each instrument's ability to successfully measure the radiation levels of the Mare Ingenii pit.	10 = Lowest possibility of failure, 5 = Somewhat high chance of failure, 1 = Very high likelihood of failure, 0 = Impossible	40%	8	8	9
Manufacturing Information	Information such as mass, dimensions, and power draw are very useful in determining the manufacturing of the instrument.	10 = All information known, 5 = Some information, 1 = Little information, 0 = No information	25%	10	9	7
Radiation Detection	As this is a crucial factor in determining how possible it is for human sustainability on the moon, the instrument's capability of measuring the radiation dose is detrimental to the mission.	10 = Accurately detects radiation, 5 = General idea of radiation levels, 0 = Can't measure radiation	15%	10	10	10
Temperature Control	Due to the harsh lunar temperatures, the instrument's capability to withstand a broad range of temperatures is essential.	10 = Can withstand near lunar temperatures, 5 = Moderately close to lunar temperatures, 1 = Temperatures aren't close to lunar temperatures, 0 = Nonoperational	10%	10	7	6
Heritage	Instruments previously used on the lunar surface are considered the highest level of heritage because they're known to be successful in the environment.	10 = Used on a previous lunar rover, 5 = Used in a slightly similar space mission, 1 = Unsuccessfully used/ mission was cancelled, 0 = No heritage	10%	7	7	10
		TOTALS:	100%	89.00%	87.50%	84.50%

Table 29. The trade studies used to decide the instruments for testing the radiation levels of the environment inside the Mare Ingenii pit.

Microclimate Measurements						
Criteria	Explanation	Grade	Weight	Model HMP60 Humidity Sensor + Model DI-808	Lunar Outpost Canary-S	PT1000 Class F0.3 + Pressure-Temperature
Reliability	The likelihood of the instrument's ability to successfully measure the environmental factors of the Mare Ingenii pit.	10 = Lowest possibility of failure, 5 = Somewhat high chance of failure, 1 = Very high likelihood of failure, 0 = Impossible	40%	9	7	9
Manufacturing Information	Information such as mass, dimensions, and power draw are very useful in determining how the instrument will be manufactured.	10 = All information known, 5 = Some information, 1 = Little information, 0 = No information	25%	10	10	10
Accuracy	As the temperature and pressure vary instantaneously, the accuracy of a tool to measure them should be the highest possible.	10 = Extremely accurate, 5 = Not very accurate, 1 = Slightly accurate, 0 = Not accurate	15%	10	8	5
Temperature Control	Due to the harsh lunar temperatures, the instrument's capability to withstand a broad range of temperatures is essential.	10 = Can withstand near lunar temperatures, 5 = Moderately close to lunar temperatures, 1 = Temperatures aren't close to lunar temperatures, 0 = Nonoperational	10%	5	8	10
Heritage	Instruments previously used on the lunar surface are considered the highest level of heritage because they're known to be successful in the environment.	10 = Used on a previous lunar rover, 5 = Used in a slightly similar space mission, 1 = Unsuccessfully used/ mission was cancelled, 0 = No heritage	10%	0	7	0
		TOTALS:	100%	81.00%	80.00%	78.50%

Table 30. The trade studies table helped determine which instrument should be the primary for this science objective.

Magnetic Fields					
Criteria	Explanation	Grade	Weight	3-Axis Fluxgate Magnetometer Model AP235	Bartington Fluxgate Magnetometer, 3-axis, Aerospace/Space
Reliability	The likelihood of the instrument's ability to successfully measure the Mare Ingenii's magnetic fields.	10 = Lowest possibility of failure, 5 = Somewhat high chance of failure, 1 = Very high likelihood of failure, 0 = Impossible	40%	10	10
Manufacturing Information	Information such as mass, dimensions, and power draw are very useful in determining how the instrument will be manufactured.	10 = All information known, 5 = Some information, 1 = Little information, 0 = No information	25%	10	10
Accuracy	Since the magnetic field strength varies by nanotesla throughout the lunar surface, it's important that the selected instrument has an extremely high accuracy.	10 = Extremely accurate, 5 = Not very accurate, 1 = Slightly accurate, 0 = Not accurate	15%	10	8
Temperature Control	Due to the harsh lunar temperatures, the instrument's capability to withstand a broad range of temperatures is essential.	10 = Can withstand near lunar temperatures, 5 = Moderately close to lunar temperatures, 1 = Temperatures aren't close to lunar temperatures, 0 = Nonoperational	10%	7	8
Heritage	Instruments previously used on the lunar surface are considered the highest level of heritage because they're known to be successful in the environment.	10 = Used on a previous lunar rover, 5 = Used in a slightly similar space mission, 1 = Unsuccessfully used/ mission was cancelled, 0 = No heritage	10%	7	7
		TOTAL $:	100%	94.00%	92.00%

Table 31. The trade studies table used to choose between two different 3-axis fluxgate magnetometers as the primary instrument for measuring the field strength of the magnetic fields.

1.5.7. Recovery and Redundancy

Recovery and redundancy are critical parts of any mission. Redundancy involves the backups for hardware throughout all subsystems (i.e. Mechanical, Power, CDH, Thermal, and Payload). The following section details redundant subsystems within the rover.

i) Mechanical Subsystem

The Mechanical Subsystem was not developed enough to begin researching redundancy in the system.

ii) Power Subsystem

The electrical subsystem consists primarily of two 2P8S batteries in 2P configuration. As previously mentioned, this battery configuration allows for a margin of 3.02 Ampere hours (Ah), which gives low risk of failure as well. Due to the high cost of the batteries, in order to not exceed the budget a voltage regulator is used as redundancy for this subsystem. The voltage regulator trips a backup circuit when voltage draw of the battery exceeds the maximum amount set. A signal will be sent within the rover to the voltage regulator, which then terminates the charge of the battery and prevents it from being overdrawn. This allows the Computer Hardware system to correct errors of excessive voltage draw before the battery is then usable again.[82]

The electrical subsystem also utilizes a Triple Junction Solar Cell. Due to cost and size constraints of $40 million and 2m x 1.25m x 1.25m, respectively, the solar cells shall be an SPF. It shall be safe to proceed with the mission despite not having a secondary system for the solar cells because they are highly reliable, as proved in the trade studies in Section 1.5.3.3.

iii) CDH Subsystem

The CDH Subsystem is equipped with duplicate processors, data buses, and power

[82] CalTech JPL, "Phase 1A Study Report: Voyager Spacecraft, Volume 3: Voyager Program Plan."

lines as mentioned in Section 1.5.4 to prevent the failure of the subsystem. In the event that the main data bus fails, data will be programmed to be forwarded to the backup to prevent loss of science. The PMU is also equipped with backup power lines to ensure minimal failure of the power system. The power draw of all rover components has also been considered, and with this in mind Team 27 has decided to create reduced operating systems options in the event that one or more scientific instruments draw too much power.

iv) Thermal Subsystem

The Thermal Subsystem is equipped with a CryoTel DS Mini as the main active cooler for the subsystem, alongside the MLI as the main insulator. Due to high success rate and even higher costs, these will be recognized as Single Point Failures (SPF).

v) Payload Subsystem

The science conducted during the mission is the most critical part of the operation, as the science was the motive to go to the Moon in the first place. Thus, there are massive amounts of redundancy within the payload system. To begin, the primary method of measuring structural integrity is the use of a SiPM. Team 27's secondary for this instrument is MID-360 LiDAR, which measures distances to surfaces via using laser pulses and detecting the light sent back.[83] This is a low-cost and condensed version of the SiPM, and makes the perfect backup in case the SiPM fails.

As the SiPM is also the primary instrument for measuring radiation type and levels, the secondary instrument is an Active Radiation Environment Sensor (ARES). ARES will utilize measurements of flux to also measure radiation levels in the event that the SiPM cannot. Lastly, Team 27 also has redundant humidity sensors and magnetometers in order to ensure the success of the mission.

This concludes the overview of redundant subsystems. Next, the recovery subsystems shall be analyzed. Recovery deals with all software-related techniques that will ensure the self-fixing of Team 27's rover.

i) Mechanical Subsystem

The Mechanical Subsystem was not developed enough to begin researching recovery in the system.

ii) Power Subsystem

As mentioned in the previous section, the use of a coded voltage regulator shall be the Power Subsystem's primary method of avoiding failure due to voltage overdraw. Team 27 has implemented the ”floating bus” idea from NASA's Curiosity rover to measure and help diagnose sources of voltage shorts.[84] As most of the components of this subsystem are purely hardware-related, and expensive, the remainder of the power subsystem shall be recognized as a SPF.

iii) CDH Subsystem

The CDH subsystem software component is arguably one of the most important subsystems of the entire rover due to the fact that it is what will transfer data taken on the

[83] “Livox Mid-360.” Accessed October 25, 2024.

[84] Webster, “Curiosity Resumes Science After Analysis of Voltage Issue.”

Moon to Earth. Thus, no corners can be cut when ensuring the success of the subsystem even in the face of failure. The first step to ensure this success is to equip the rover with a redundant computer system to ensure that if the primary system fails, it has a duplicate secondary to fall back on and proceed as normal. This idea of two computers was utilized in the Curiosity Rover Mars Mission, and proved to allow for the lengthened success of the rover even after memory failure of the primary computer.[85]

In addition to the secondary computer that is complete with redundant coding, the rover shall also be equipped with an extensive system of memory allocation. The reason for this focus is because overburdened memory can cause the failure of the entire rover, as the rover will not function unless there is space within the memory of the computer. This was a failure that was seen in a Mars Exploration Rover, and was corrected by manual allocation of files.[86] Team 27 would like to correct for this issue in advance to ensure the success of the rover.

iv) Thermal Subsystem

The Thermal Subsystem notably has a Thermal Control System composed of the CryoTel DS Mini and MLI. Both of these components together allow for constant temperature readings across the entire rover. These components also ensure proper temperature is maintained by sending commands to either heat or cool certain components of the rover, as previously mentioned. Insulation is also used to ensure the system is protected. As the cost is quite high for the system as a whole, and testing must be done to ensure all components would survive in a lunar environment, the Thermal Subsystem must be recognized as a SPF. All coding to ensure temperature stability has been included in the primary components of the subsystem. Further improvements would result in a steep price increase due to component price and testing, alongside the fact that the rover simply doesn't have additional space to accommodate a larger Thermal Subsystem.

v) Payload Subsystem

It is very clear that some instruments of the Payload Subsystem have the highest voltage draws of the entire rover. For example, the large draw of the SiPM single handedly led to the specific battery configuration of the rover to allow for enough voltage to be available for it. As a prevention technique for voltage overdraw of the instrument in the payload system, it has been decided that during peak voltage-draw times only a single instrument must be ran at a time. This is the particular case when the SiPM itself is in use. While inconvenient for the mission, this allows for safe margins of voltage draw from all payload instruments.

1.5.8. Interface Control

In lunar rover design, each part—each subsystem—must be carefully combined to create a complete and functional product for a successful lunar mission. The N∧2 and system block diagrams detail the critical interconnections between its subsystems: Payload, Command and Data Handling (CDH), Communication, Navigation, Mechanical, Thermal Control System (TCS). The science team has focused on the specific types of instruments, research objectives, and workload, which directly are tied into subsystem requirements

[85] Agle, "Curiosity Rover's Recovery Moving Forward."
[86] "MER Spirit Flash Memory Anomaly."

and the overall rover design aswell. More details will be provided in this Interface Control section, expanding on the interfacing relationships between the different subsystems and also defining them.

The Payload subsystem comprises all scientific instruments and sensors necessary for mission objectives. The payload will be managed by the science team under the leadership of Chief Scientist as they will be the ones commanding specific activities and adjusting parameters in real-time or via pre-programmed sequences helping the team push forward the boundaries of lunar exploration. It relies heavily on the Mechanical Subsystem for structural support of its subsystem and mobility, along with CDH Subsystem for managing the data collection and analyzing the data sets for research.

The CDH Subsystem, acting as the central hub, processes all onboard data and transmits instructions across the rover. It interfaces with the mechanical subsystem to relay movement commands and receive feedback on positional adjustments. It also collects and processes data from the payload subsystem's scientific instruments, and the efficiency of the processes of data is determined by the rate at which samples are processed, the time required for data collection, and the amount of sets of samples, ensuring the mission's scientific goals are met efficiently and effectively. Data collected during these operations is securely stored in the SSD. For such data handling and a standardized way to connect everything to CDH, the rover will be using a Spacewire high-speed, reliable, and fault-tolerant communication network.

An Integrated communication and navigation system is used for the rover. Communication systems on the lunar rover are designed to handle long-distance data transmission between the rover and Earth using a high-gain antenna for direct communication. All the needed data will be sent by the high-gain antenna for this mission. For this the rover must optimize its use of available bandwidth, prioritizing essential data for transmission during optimal communication windows to maximize the efficiency of data sent to Earth. Communication systems are designed with redundancy to safeguard against the failure of any single component.

The navigation system uses cameras to allow the rover to detect obstacles and plan safe paths across the terrain. The team also has decided to use LiDAR, which emits laser pulses and measures the time for them to bounce back. This data from LiDAR will provide much more depth perception and topographical mapping, crucial for navigating complex environments done by using two stepper motors for forward and backwards movement and four steer motors for controlling the direction of the rover. Real-time data from navigation systems are transmitted back to Earth, where mission planners can adjust commands based on the rover's progress and any detected anomalies. This integrated approach ensures that the rover can not only navigate autonomously, but also adjust to new commands from Earth promptly and efficiently. Therefore, communication and navigation are intertwined systems for the lunar rover.

Power Subsystem comes into play for operation of the whole rover and serves as the rover's lifeline, distributing vital electricity to all other subsystems. This subsystem is robust, capable of withstanding the temperature extremes and radiation levels of space. It features advanced solar cells harnessing solar energy, which will help the batteries store power as much as possible to help keep the instruments on and other subsystems on. Managing the distribution of power within the rover is critical, particularly given the fluctuations in energy production and the various operational demands of different subsystems. The power management system includes regulators to combat such demands.

Lastly, Thermal Control Subsystem ensures that all components across the spacecraft operate within their required temperature ranges. This system works closely with all subsystems and instruments and implements thermal insulation (MLI) solutions and manages heat generated by them using cryocoolers.TCS also uses power to run heaters also, such that the rover maintains operational temperatures for all electronic and mechanical parts within safe limits.

The Effective interface control diagrams below are showcased showing seamless system integration.

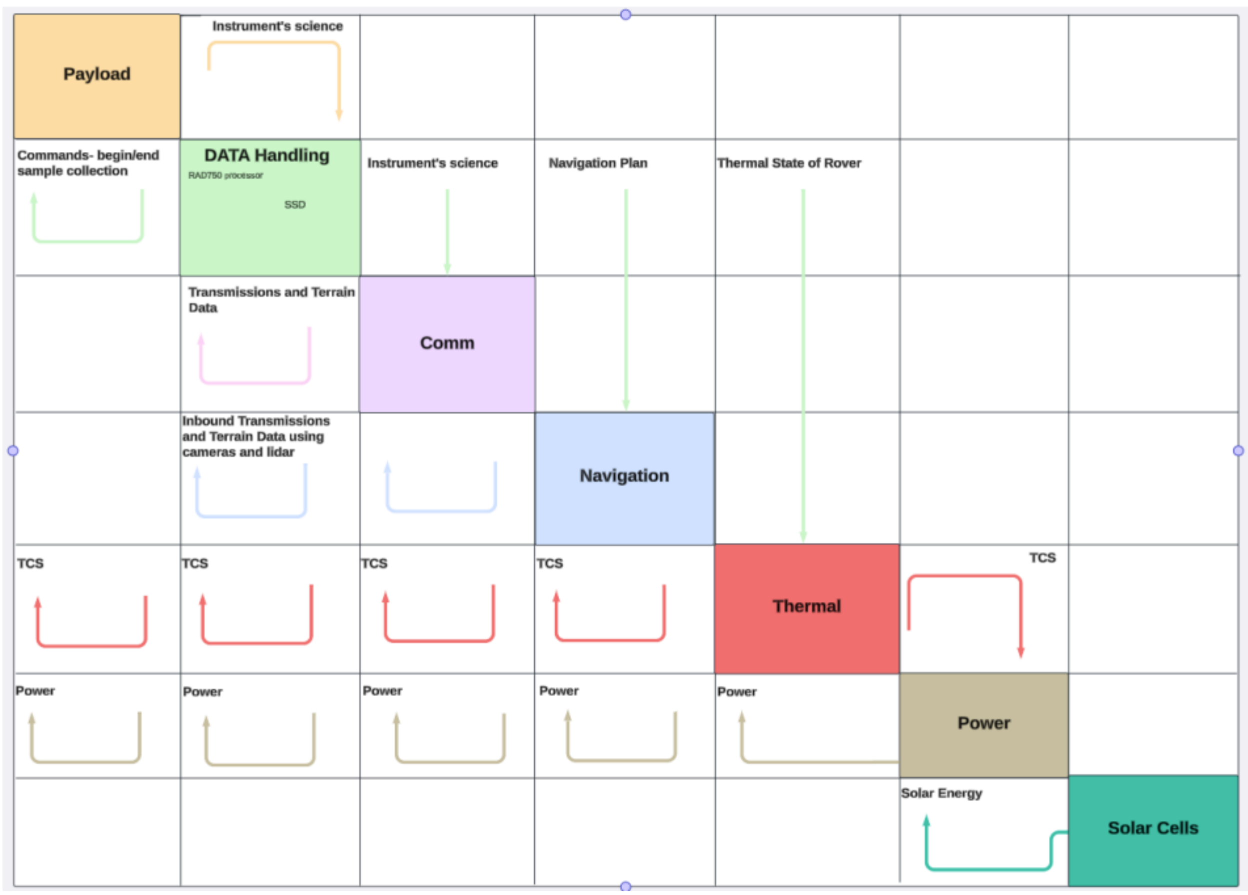

Figure 6. The block diagram for Team 27's subsystems.

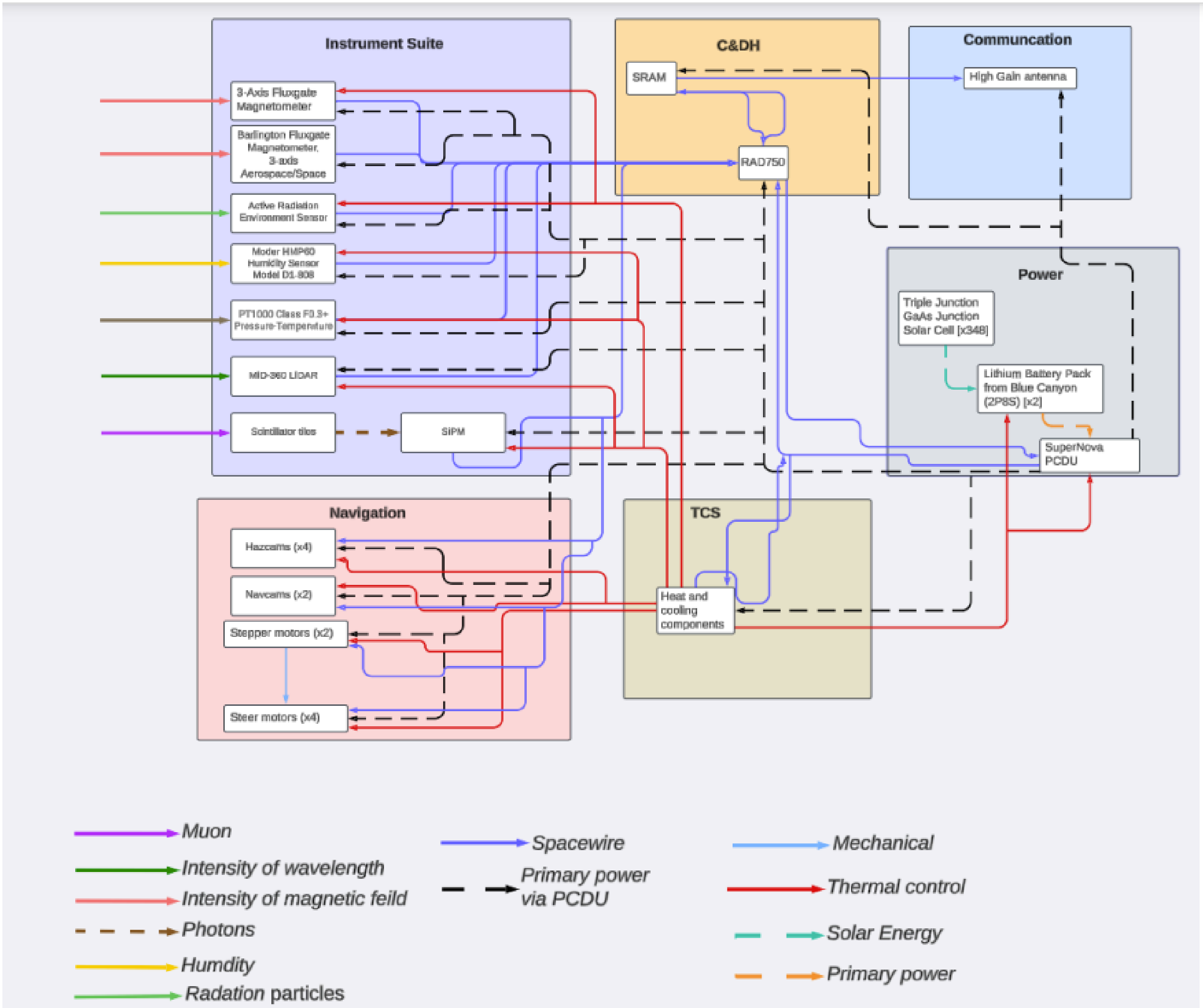

Figure 7. The N∧2 of Team 27's Subsystems.

1.6. Risk Analysis

This section will detail Team 27's efforts to minimize contamination during the mission, the decommissioning plan, and analyze further the risks within each engineering subsystem.

The exploration of the Mare Ingenii lunar pit poses the risk of contaminating the environment on the Moon through microorganisms and organic molecules. Mitigation of contamination can be implemented in order to protect the Moon's environment and ensure the science done is verifiable. Mitigation efforts are defined in accordance with the policies and guidance given by NASA's Office of Planetary Protection and relevance to the mission. In order to reduce the contamination of the mission's spacecraft equipment, all building, assembly, and installation will be completed in an ISO Class 5 cleanroom due to the sensitive nature of the mission's optical instruments.[87] All personnel who enter this

[87]Thasshwin Mathanlal, Maria-Paz Zorzano, and Javier Martine-Torres. "Design, development, and operation of an ISO class 5 cleanroom for planetary instrumentation and planetary protection protocols," Heliyon 10,

cleanroom are required to enter and leave through the antechamber where they will perform hand and forearm hygiene and dress in sterile protective garments including gowns, shoe covers, headgear, and facial hair covering (if applicable), and any person directly handling the spacecraft and its parts are required to wear powder-free gloves to prevent any fibers, hair, and skin particles from entering the cleanroom. Since the majority of the instruments that are going to be used on this mission are sensitive to temperatures above 50°C to 70°C, they will all be wiped with 70% Isopropyl Alcohol regularly from before final assembly to installation, instead of the frequently used Dry Heat Microbial Reduction method.[88]

Decommissioning the mission's spacecraft is the intentional act of terminating all operations and communications and the proper disposal of the vehicle(s). When the mission reaches the end of the timeline, or in the event of significant spacecraft damage or failure, the Mission Control Center will position the vehicle in an area on the Moon that is well-defined and flat to prevent any unintentional movement. As with the Apollo 15, 16, and 17 missions, the spacecraft will be disposed of in this location.[89] They will then send a series of shutdown commands, which will stop all of the spacecraft's systems and communication and will prevent any unintentional operations. The thermal systems that will be put in place to protect the vehicle from the harsh lunar conditions will be terminated, which will expose the spacecraft and the sensitive electronics to these conditions and will damage them.

In order to identify the risks associated with this mission, the subsystem requirements and trade studies done for mechanical, power, CDH, thermal, and instrumentation were thoroughly looked over. The identified risks were tracked in the risk summary table, with at least 2 risks for each subsystem, along with the likelihood of the risks occurring and the criticality of the consequences should they occur. The likelihood and consequences of the risks were inserted into the risk matrix, which is color-coded with the critical nature of each risk labeled. The following paragraphs will analyze the risks associated with the mission and propose plans on how these risks will be approached, detailing the matrix and table shwon together below.

no. 17 (2024): 2.

[88]Handbook for the Microbial Examination of Space Hardware, Edition b, s.v. "Considerations for Assay Performance."

[89]"50th Anniversary of Apollo 15 and the Lunar Roving Vehicle," National Archives, accessed October 25, 2024.

Risk Summary

ID	Summary	L	C	Trend	Approach	Risk Statement	Status
1	Mechanical	3	4	**NEW**	M	Mechanical subsystem fails to remain operational for required mission timeline	Active
2	Power	2	5	**NEW**	W	Power system fails to generate 200Wh of power	Active
3	Power	3	3	**NEW**	W	Power system fails due to lunar regolith debris	Active
4	Power	2	5	**NEW**	R	Sharp object penetrates electrical system	Active
5	CDH	2	4	**NEW**	W	Communication interface malfunctions	Active
6	Thermal	2	3	**NEW**	R	TCS fails to maintain ideal operating temperature	Active
7	Thermal	2	4	**NEW**	W	TCS fails to minimize thermal fatigue of the chassis	Active
8	CDH	3	5	**NEW**	W	CPU fails to execute commands	Active
9	CDH	1	4	**NEW**	W	SRAM fails to endure radiation	Active
10	Instrumentation	1	3	**NEW**	W	Instruments fail to endure exposure to high-radiation	Active
11	Instrumentation	3	5	**NEW**	R	Instruments fail to withstand physical hazards from the lunar environment	Active
12	Mechanical	4	5	**NEW**	M	Mechanical subsystem fails to endure the natural space environment	Active

L = Likelyhood (1-5)
1 = Not likely
5 = Extremely likely
C = Consequence (1-5)
1 = Low consequence
5 = High consequence

LxC Trend
↓ - Decreasing (improving)
↑ - Increasing (worsening)
→ - Unchanged
NEW - Added this month

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

Criticality
HIGH
MED
LOW

Figure 8. The risk matrix and summary table.

Risk ID 1 and 12 are associated with the mechanical subsystem. To preface, these risks are general in potential issues that any mechanical system of a spacecraft might encounter due to a Mechanical Engineer dropping the academy. The most common risks found for mechanical systems are failing to remain operational for the required mission timeline and failing to endure the natural space environment. For risk ID 1, given that the quality of the materials used is crucial in how long the system can function for, there is a possibility of improper materials being used which can result in damage to the system, thus the system could deteriorate and fail over time. For risk ID 12, since the space and lunar environment exposes the mechanical system to extreme temperatures, solar radiation, micrometeorites, and debris, there is a possibility that the mechanical system cannot endure these interactions, resulting in either reduced or no functionality and ultimately leading to mission failure. These risks can be mitigated through being familiar with the properties of the materials being used on the mission and how they interact with the environment and physical hazards, as well as testing in order to predict the limits of the system.[90]

Risk ID 2, 3, and 4 are associated with the power subsystem. The risks associated with this mission for the power system are failing to generate 200Wh of power, power system failing due to debris from the lunar regolith, and a sharp object penetrating the electrical system. For risk ID 2, given that 200Wh of power is needed to maintain base-line operations, there is a possibility that this amount of power can fluctuate and not be met, resulting

[90]Theodore Swanson, “Wear and Tear - Mechanical.”

in the battery power being discharged and depleting overall power capacity. For risk ID 3, since the lunar surface has regolith and debris present, there is a possibility of the regolith and debris obstructing the system, resulting in the vehicle either undergenerating or not generating the power needed due to the decreased collection of solar energy. For risk ID 4, given that there are sharp objects present on the lunar surface, there is a possibility of a sharp object penetrating the electrical system, resulting in a short or open in the circuit which can ultimately terminate operations prematurely. The approach for risk ID 2 and 3 is to watch them, they will be analyzed but not actively mitigated for the moment. The approach for risk ID 4 is to research it, it cannot be further analyzed until more is understood.

Risk ID 5, 8, and 9 are associated with the CDH subsystem. The risks associated with this mission for the CDH system are the communication interface malfunctioning, the CPU failing to execute commands, and the SRAM failing to endure radiation. For risk ID 5, since the spacecraft will have multiple components that must work together and depend on each other reliably, there is a possibility that the communication interface will malfunction, resulting in communication failure and data loss through failure to reliably transfer data between different subsystems. For risk ID 8, given that the CPU must be able to handle complex commands and allow collaboration with all systems of the vehicle, there is a possibility that the CPU fails to execute these commands, resulting in the subsystems not being synchronized and the vehicle not being able to perform the required operations for the mission. For risk ID 9, since space has high-radiation activity, there is a possibility of the SRAM failing to endure this radiation, resulting in damage to the equipment and the loss of ability to store and handle data collected during the mission. The approach for risk ID 5, 8, and 9 is to watch them.

Risk ID 6 and 7 are associated with the thermal subsystem. The risks associated with this mission for the thermal system are the TCS failing to maintain ideal operating temperatures and the TCS failing to minimize the thermal fatigue of the chassis. For risk ID 6, given that the lunar environment presents extreme temperature changes to the vehicle, there is a possibility of a passive TCS causing consistent heating which can lead to the components being less effective due to operating outside of the ideal temperature range of 253.15K to 328.15K. For risk ID 7, since the chassis is exposed to the extreme temperatures on the Moon, there is a possibility of the TCS failing to minimize the thermal fatigue of the chassis that is caused by significant temperature changes and high temperatures, resulting in cracks in the material and shortening the vehicle's lifespan. The approach for risk ID 6 is to research and the approach for risk ID 7 is to watch.

Risk ID 10 and 11 are associated with the instrumentation (payload) subsystem. The risks associated with this mission for the instrumentation system are the instruments failing to endure the exposure to the high-radiation conditions and the instruments failing to withstand the physical hazards from the lunar environment. For risk ID 10, given that the lunar environment has presence of high-radiation, it is possible for the instruments' operations to become compromised, resulting in the instruments becoming damaged or less effective at their intended purpose. For risk ID 11, since the lunar surface contains regolith and debris, there is a possibility that the instruments can become physically damaged, resulting in issues with completing their assigned tasks and communication with other subsystems. The approach for risk ID 10 is to watch and the approach for risk ID 11 is to research.

1.7. Programmatics

The upcoming sections will discuss the organization of Team 27, the cost and schedule estimate of the mission, and change control.

1.7.1. Team Organization

In the weeks leading up to the submission of the System Requirements Review (SRR), Team 27 faced unforeseen challenges that led to several re-arrangements of the team's organization of roles, which indicate the members' primary and secondary roles (Figure ? and ?, respectively).

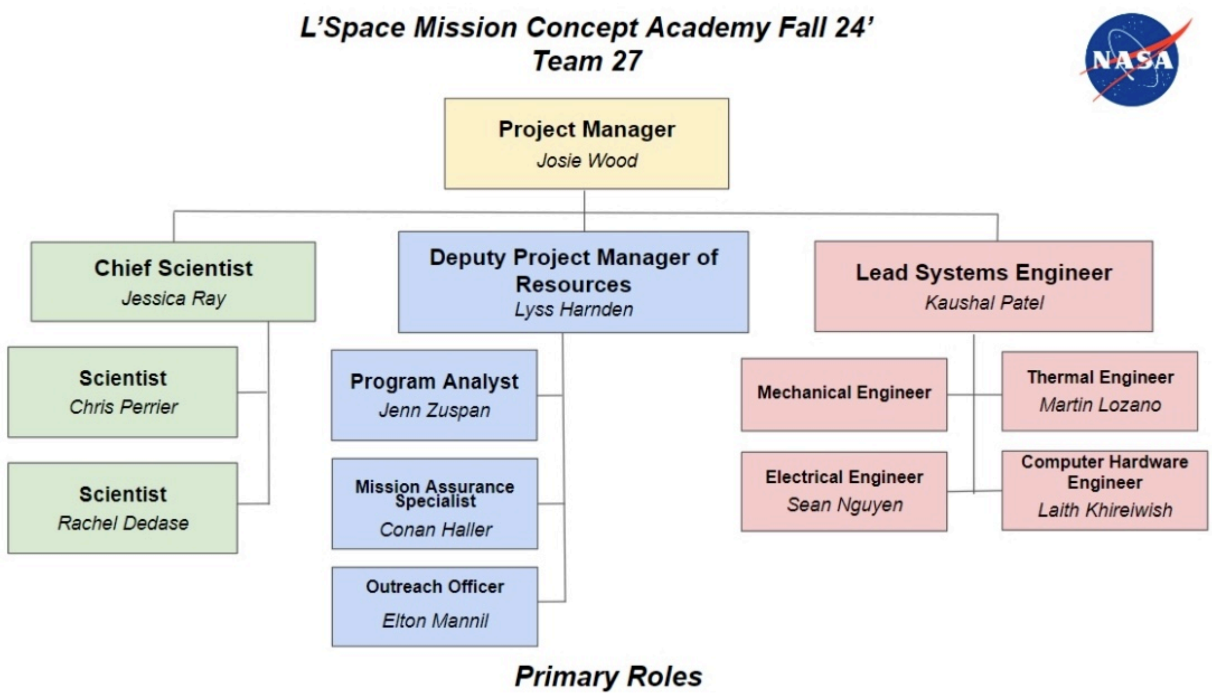

Figure 9. Organization of the primary roles of the members of Team 27.

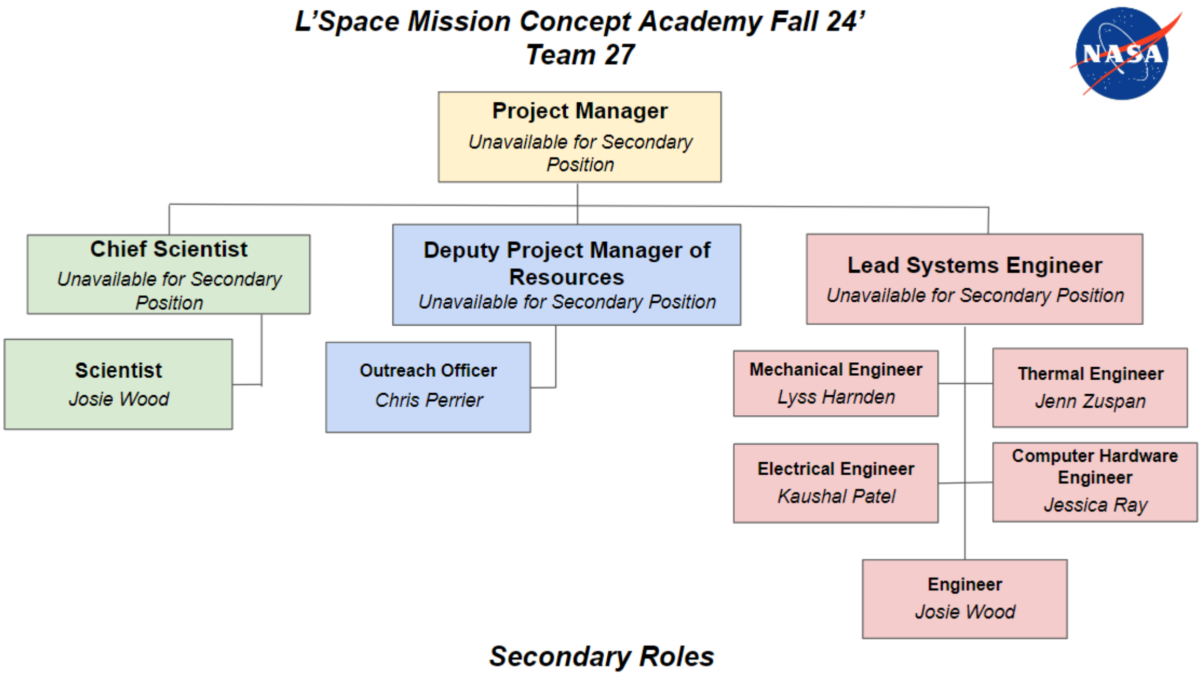

Figure 10. Organization of the secondary roles of the members of Team 27.

Team leads were originally voted for collectively by the team after listening to the candidates' elevator pitches during a group meeting. Additionally, leads for primary roles within each sub-team were arranged based on academic and related experience, as well as time availability to commit to the academy. For example, the Outreach Officer position, a role that has very little assigned tasks for the SRR, was given to a team member with a heavy schedule and limited time to give to this deliverable. In contrast, the lead Mechanical Engineer, who is substantially depended on for a significant portion of the SRR pertaining to the spacecraft's mechanical subsystem, was initially given to a team member currently studying and completing research in the field of mechanical engineering. However, Team 27 was informed a week before the SRR deadline that it had lost its lead Mechanical Engineer, challenging team leads to promptly respond in re-assigning Mechanical Engineer tasks to ensure they would be covered as best as possible. Team members–who also happen to be team leads–in the secondary Mechanical Engineer and Engineer roles were assigned these tasks with guidance from the Lead Systems Engineer (LSE), challenging them to balance their own SRR tasks and leadership along with covering the mechanical engineering tasks. Additionally, the Project Manager (PM) submitted an extension request, which was approved on the date the SRR was intended to be due. This allowed Team 27 to give some more time to take care of unfulfilled tasks caused by unforeseen challenges.

Being that the SRR is heavily engineering-focused, it was crucial to ensure that the engineering sub-team had utmost support to guarantee the success and completion of engineering tasks. Thus, not only do each of the engineer roles (excluding Mechanical) have primary leaders, but they have secondary leaders as well to provide additional assistance to the engineering team when they've finished the work for their primary roles. This especially applied to those whose primary roles were in sub-teams whose prevalence in the SRR was significantly smaller and therefore had less sub-team-specific work required of them, such as those in programmatics who took on secondary engineering roles.

Methods for organization, workload, and decision-making have been decided on, established, and altered as the team has gained more experience working with one another. Firstly, expectations on both sub-team and Team 27 levels were established to set requirements and objectives regarding topics such as communication, meeting attendance, task completion, and violation of guidelines. These are detailed in the Team Expectation Document (Figure ?), which was reviewed by the PM, Chief Scientist (CS), Deputy Project Manager of Resources (DPMR), and LSE and subsequently shared with the whole team during a meeting together. As was the case with the MCR, current workload amongst the team is determined through organized sheets and documents managed by the PM, with the ability to be edited and viewed by all members in order to ensure all are aware of and able to keep each other accountable of their respective tasks. Some of these documents include the Team SRR Responsibility Sheet, which is organized by role-task assignment and tracks the progress of each task, and the Baselined Task Schedule for the SRR, which lists the proposed and desired progress to be made on the SRR per week leading up to its due date. Decisions are made after thorough discussion during group meetings, shared voting amongst members with hard deadlines for votes to be cast, further communication amongst team leaders in a team leads-specific Discord channel, and ultimately declared by the PM (and sent out as an announcement to share with the entire team).

Collaborative Team Expectations
Team 27 - L'SPACE MCA

Overview

This document has been reviewed by the Project Manager, Chief Scientist, Deputy Project Manager of Resources, and Lead Systems Engineer, and contains information about expectations for the team. Additionally, each discipline (science, engineering, programmatics) has their own expectations for smaller group work and contributions. Any questions regarding expectations should be directed to the Project Manager.

Expectations for Each Team Member

1.1 Team members will complete all work assigned by the due date.

- If team members are struggling with their assigned work and/or are not able to complete it by the assigned due date, reach out to your Team Lead or the Project Manager immediately. Communication is critical in any team.

1.2 Team members should read the work of others to hold them to a high standard of work, and to ensure full team understanding of the material (i.e. do not just read the section you complete for a document, read the *entire* document).

1.3 Team members will treat each other with respect. In the event that a disagreement arises, it should be handled professionally and calmly.

1.4 All ideas will be explained so that *all* team members, regardless of education level, can understand them.

1.5 Team members should make the majority of the whole group and subteam meetings. If meetings are not attending, you must reach out to team leads or the PM for information on what you missed, and consult the meeting notes document for the meeting.

Figure 11. A portion of the Team Expectations document, detailing team requirements. The full document is available in the Appendix.

Some other challenges that Team 27 has faced include carefully assigning primary roles to members with the ability to commit extensive time and learning in roles outside their expertise and maintaining active and timely communication with one another to keep each other informed and ensure progress is being made on deliverables. For the first situation, this was reconciled by following some of the aforementioned methods, including the PM communicating with the engineering sub-team and individuals being considered for role re-arrangement, with the final reconstruction of the team organization being announced by the Project Manager to the rest of the team. In the second circumstance, the group collectively discussed and proposed improved communication methods and habits to practice in order to succeed, including required check-ins, public sharing of their task progress, and more. For future deliverables, Team 27 will continue to utilize methods in order to ensure timely progress on tasks, active and effective communication across sub-teams/the team as a whole, and take on any unexpected obstacles.

1.7.2. Cost Estimate

The overall budget for the mission task is $425 million dollars, which includes four main categories: personnel salary, travel, outreach, and direct costs. The total mission cost of $311,735,749 includes margins in four categories, consisting of facilities and administration (10%), manufacturing (50%), employee related finances (28%), and total cost (10%). This cost will be further refined for the Mission Definition Review (MDR). NASA's fiscal year begins the first of October every year. The fiscal years are aligned with each phase of the mission based on the schedule in section 1.7.3. The complete table for the entire mission budget based on these four categories can be found in Appendix A.

There are four types of personnel that are necessary to hire for the mission. Science personnel work on the mission experiment and analysis of collected mission data. Engineering personnel consists of a diverse team with expertise for each of the rovers subsystems including systems, mechanical, electrical, and thermal. They are responsible for designing, building, and testing the mission concept, as well as ensuring mission operations run smoothly. Technicians are responsible for assisting the engineers and scientists with manufacturing, assembly, and testing of instruments and systems for the mission concept. Administration personnel work on administrative and human resources tasks, and tracking the budget, schedule, and outreach for the mission. Project management organizes the mission personnel, budgets, and schedule. The salaries for each type of personnel is listed in Table X.

Science Personnel	$80,000 / year
Engineering Personnel	$80,000 / year
Technicians	$60,000 / year
Administration Personnel	$60,000 / year
Project Management	$120,000 / year

Table 32. Necessary hires along with corresponding salaries.

Each phase of the mission has different needs of personnel, leading to varying salary costs as shown in Figure X. Administrators and managers are the backbone of the mission team, therefore stay constant throughout. There will be 9 administration personnel and 5 management personnel throughout the mission. The management team consists of the PM, DPMR, CS, and LSE. The number of administrative personnel is based on the percentage of the programmatics sub-team for a 13 person team, along with accounting for HR, and other administrative tasks. During phases c and d, the final design, fabrication, system assembly, testing, launch and checkout take place, requiring engineers and technicians more than scientists. For these phases, there will be 5 engineers in each engineering field: Mechanical, Electrical, Command data and handling, and Thermal, leading to a total of 20. There will be 3-4 technicians working to fabricate the various subsystems, and assemble the total system, leading to a total of 15. There will be 2 scientists during this time to ensure the engineering design is compatible with the necessary science for the mission. Phases E and F consist of operations, sustainment, and closeout of the mission, therefore requiring more scientists and engineers than technicians. During these last phases, there will be 10 scientists to analyze the collected data from the mission. There will only be 15 engineers during this time, as they still have an important role for decommissioning efforts, but less are needed. There are no technicians for these phases because there is no longer any part of the system that is being built or tested.

Additional Information

	Phase C	Phase C	Phase C-D	Phase D	Phase D	Phase E-F
# People on Team	FY 1	FY 2	FY 3	FY 4	FY 5	FY 6
Science Personnel:	2	2	2	2	2	10
Engineering Personnel:	20	20	20	20	20	15
Technicians:	15	15	15	15	15	0
Administration Personnel:	9	9	9	9	9	9
Management Personnel:	5	5	5	5	5	5

NASA L'SPACE Mission Concept Academy Budget - *Team 27*

Mission Phase	Phase C	Phase C	Phase C-D	Phase D	Phase D	Phase E-F	
Year	**Year 1**	**Year 2**	**Year 3**	**Year 4**	**Year 5**	**Year 6**	**Cumulative Total**
PERSONNEL							
Science Personnel	$ 160,000	$ 164,160	$ 168,320	$ 172,480	$ 176,640	$ 904,000	$ 1,745,600
Engineering Personnel	$ 1,600,000	$ 1,641,600	$ 1,683,200	$ 1,724,800	$ 1,766,400	$ 1,356,000	$ 9,772,000
Technicians	$ 900,000	$ 923,400	$ 946,800	$ 970,200	$ 993,600	$ -	$ 4,734,000
Administration Personnel	$ 540,000	$ 554,040	$ 568,080	$ 582,120	$ 596,160	$ 610,200	$ 3,450,600
Project Management	$ 600,000	$ 615,600	$ 631,200	$ 646,800	$ 662,400	$ 678,000	$ 3,834,000
Total Salaries	$ 3,800,000	$ 3,898,800	$ 3,997,600	$ 4,096,400	$ 4,195,200	$ 3,548,200	**$ 23,536,200**
Total ERE	$ 1,060,580	$ 1,088,155	$ 1,115,730	$ 1,143,305	$ 1,170,880	$ 990,303	**$ 6,568,953**
Personnel Margin	$ 1,389,614	$ 1,462,814	$ 1,537,892	$ 1,614,848	$ 1,693,684	$ 1,435,982	**$ 9,134,834**
TOTAL PERSONNEL	$ 6,250,194	$ 6,617,463	$ 6,997,085	$ 7,389,209	$ 7,793,980	$ 6,751,168	$ 41,799,099

Table 33. Personnel allocation and cost per fiscal year.

There are multiple events throughout the duration of the mission's lifecycle that will require some or all of the team to travel to various locations, including key decision points (KDP), system testing, and mission launch. Travel costs are estimated based on commercial coach flight tickets[91], rooms from partnered hotels[92] and meals per diem[93] for each person along with car rentals for the team[94], that are based on the current average weekly per location, accounting for at least one SUV and one sedan.

Since this mission is a category 2 project, the KDPs must be conducted in Washington D.C. at NASA Headquarters, so the Mission Directorate Associate Administrator can sign off. There will be one trip in each of the phases C, D, and E. For each of these meetings, the total stay for the trip will be 5 days .The key personnel for these meetings is the project management team (5 people). Based on the average commercial coach flight ticket price from various locations around the US, each ticket will be $500. The per diem costs for lodging and food in Washington D.C. are $300 per night and $70 on the first and last days of travel and $92 during the stay respectively. Two cars will be used for the team for each trip, with a price of $400/week per car.

[91] U.S. General Services Administration. "City Pair Program (CPP)." Accessed October 23, 2024.
[92] "Home." Accessed October 23, 2024.
[93] U.S. General Services Administration. "Per Diem Rates." Accessed October 23, 2024.
[94] Enterprise. "Reserve." Accessed October 25, 2024.

A portion of the team will need to travel to the NASA Glenn Research Center in Cleveland, Ohio to conduct drop testing for our system. This will occur during phase D of the mission. The total stay for this trip will be 1 week. The flight cost based on the average commercial coach flight ticket price from various locations is $350. The lodging per person is $160 per night. The meals per diem are $60 on the first and last day of travel, and $80 during the stay. Three cars will be used for the team for the trip, with a price of $300/week per car.

A portion of the team will also need to travel to the black point lava flow in Flagstaff, Arizona. The total stay for this trip will be 1 week. The flight cost based on the average commercial coach flight ticket price from various locations is $350. The lodging per person is $150 per night. The meals per diem are $60 on the first and last day of travel, and $80 during the stay. Three cars will be used for the team for the trip, with a price of $450/week per car.

The key personnel for these meetings are the management team, along with 2 engineers per focus, and 2 scientists, leading to a total of 14 people.

All team members (59) who have worked on the mission will travel to Cape Canaveral, Florida for launch. This will be a total stay of 5 days, arriving two days prior and departing two days after the launch date. The flight cost based on the average commercial coach flight ticket price from various locations is $380. The lodging per person is $208 per night. The meals per diem are $55 on the first and last day of travel, and $75 during the stay. Eleven cars will be used for the team for the trip, with a price of $400/week per car.

The travel costs for each phase of the mission, along with the total of $226,741 are shown in Table X.

TRAVEL							
Total Flights Cost			$ 2,500		$ 34,720	$ 2,500	$ 39,720
Total Hotel Cost			$ 7,500		$ 81,128	$ 7,500	$ 96,128
Total Transportation Cost			$ 800		$ 7,450	$ 800	$ 9,050
Total Per Diem Cost			$ 2,080		$ 36,405	$ 2,080	$ 40,565
Travel Margin			$ 1,271		$ 17,613	$ 1,365	$ 20,249
Total Travel Costs	$ -	$ -	$ 14,887	$ -	$ 195,757	$ 16,097	$ 226,741

Table 34. Travel costs per fiscal year.

Based on the percentage of the total budget that is typically allocated to outreach, about $4,250,000 will be used to inform the public about the mission of Team 27 and its importance. These details will be discussed further in the MDR.

The direct costs for the mission consist of the costs for each subsystem, science instrumentation, and the manufacturing and test facility costs.[95] The power subsystem and science instrumentation costs are calculated based on the NASA Instrument Cost Model[96] (NICM) if prices aren't readily available for components. The rest of the subsystem costs are calculated based on the estimated percentage of the budget from the cost estimating handbook. A 50% margin has been added to the subsystem cost due to the estimation;

[95]NASA Software Catalog. "NASA Instrument Cost Model (NICM) Version 10(HQN-11886-1)." Accessed October 25, 2024.

[96]"NASA Cost Estimating Handbook Version 4.0," February 2015.

these costs will be further refined in the MDR. These are calculated in the second year of Phase C, because that is when fabrication is likely to begin. Table X provides the breakdown of the subsystems and facility costs that lead to the total amount of $186,656,535.

DIRECT COSTS							
Mechanical Subsystem		$ 21,250,000					$ 21,250,000
Power Subsystem		$ 45,378					$ 45,378
Thermal Control Subsystem		$ 21,250,000					$ 21,250,000
Comms & Data Handling Subsystem		$ 34,000,000					$ 34,000,000
Guidance, Nav, & Control Subsystem		$ 42,500,000					$ 42,500,000
Science Instrumentation		$ 394,032					$ 394,032
Spacecraft Cost Margin		$ 125,816,052					$ 125,816,052
Total Spacecraft Direct Costs	$ -	$ 251,632,104	$ -	$ -	$ -	$ -	$ 251,632,104
Manufacturing Facility Cost		$ 800,000					$ 800,000
Test Facility Cost		$ 290,000					$ 290,000
Facility Cost Margin		$ 124,620					$ 124,620
Total Facilities Costs	$ -	$ 1,246,200	$ -	$ -	$ -	$ -	$ 1,246,200

Table 35. Direct costs.

1.7.3. Schedule Estimate

There are four major phases that remain in the mission: final robotic system design and fabrication, system assembly and launch, operations and decommissioning.[97] To move to a new stage in the mission, KDPs must be completed.[98] These have different series of life cycle reviews (LCR) associated with them, which often require standing review boards (SRB). The KDP occurs at least 30 days after the SRB's snapshot report is complete.[99] As mentioned in Section 1.7.2, these are conducted at NASA Headquarters in Washington D.C., so it's important to schedule travel time for needed personnel, which is the project management team.

Final robotic system design and fabrication (phase C): This first phase involves the creation of the mechanical and electrical subcomponents, propulsion, payload systems and thermal systems to create an overall working system. This phase will begin after the completion and review of the Preliminary Design Review (PDR), which is KDP E. Since the PDR will be complete on December 8th, 2024, phase C is set to begin on December 23rd, 2024 to allow time for review. The key decision points of this phase include the critical design review (CDR), production readiness review (PRR), and system integration review. Developing a design with appropriate maturity to support full scale fabrication for the CDR and PRR will take slightly over a year. The verification that the systems segments, components, and subsystems are on schedule to be ready for integration is confirmed in the system integration review (SIR).[100] To account for communication time with several different vendors and fabrication time, this will take about 9 months.

[97] "SEH 3.0 NASA Program/Project Life Cycle." NASA, July 26, 2023.
[98] Chapter2. "NPR 7120.5E ." Accessed October 24, 2024.
[99] Congress, Nita. "NASA Standing Review Board Handbook."
[100] Administration, National Aeronautics and Space. "NASA Systems Engineering Handbook," December 2007.

Phase C Beginning	12/23/2024
Critical Design Review	3/1/2026
Production Readiness Review	3/1/2026
System Integration Review	12/20/2026
KDP D:	1/5/2027

Table 36. Phase C schedule.

System assembly and launch (phase D): Afterwards, phase D consists of completing rigorous testing that mimics lunar conditions is needed to fix or change anything within the system that might hinder the mission's data collection and overall functionality.Testing that is needed includes thermal vacuum, vibrational,[101] overall environmental testing,[102] and drop testing based on the mission's plan. To integrate the system with the launch vehicle, multiple checks and launch tests will need to be done. These tests are necessary to ensure all systems are robust enough to handle the intensity of the launch, space travel, and landing conditions of the lunar surface. All components must function perfectly in harsh conditions, as even a slight malfunction could lead to mission failure. The integration process will also involve coordination with the launch vehicle provider to ensure smooth harmony. The completion of this testing is confirmed with the System Acceptance Review (SAR). Based on the nature of the process to ensure proper assembly of the system, and thoroughly test it, this process is projected to take 2 years. The Mission Readiness Review (MRR) and Safety and Mission Success Review (SMSR) are the final reviews before launch, and occur four weeks before the launch date.[103] Once testing is finished, the system is launched on the set date, March 1st, 2030.It will take approximately four days for the spacecraft to land on the moon.[104] This phase ends with the post launch assessment review, which often takes about 30 days after the launch date to complete.[105]

[101] Crystal Instruments. "Satellite Vibration Testing." Accessed October 24, 2024.
[102] "Space Environmental Testing at the NASA Space Power Facility," Accessed October 24, 2024.
[103] Oberhettinger, David. "Review." NASA Preferred Reliability Practices, January 1997.
[104] "Apollo 17." Accessed October 24, 2024.
[105] NASA Science. "Chapter 7: Mission Inception." Accessed October 23, 2024.

Phase D Begins	**1/1/2027**
System Acceptance Review	1/5/2029
Operational Readiness Review	1/17/2030
Mission Readiness Review	1/28/2030
Safety and Mission Success Review	1/28/2030
KDP E: Launch	3/1/2030
Lunar Arrival	3/5/2030
Post Launch Assessment Review	3/30/2030

Table 37. Phase D schedule.

Operations (phase E): This phase focuses on data collection and analysis from the lunar robotic system. Through the entire mission, it will be necessary to monitor and control the robot's functions, and fix any obstacles it may face from ground system controls. After this, collecting data is crucial to achieve the scientific research objectives. The data for this phase will be transmitted continuously back to Earth for the scientists to analyze, uncovering new insights about the lunar surface. Any anomalies encountered during operations must be addressed swiftly to avoid any misinterpretations in the data transmission. The Critical Events Readiness Review (CERR) is estimated to take two weeks after phase E begins. It is estimated that the data collection will take about five months, which is when the decommissioning review will be completed. Then, KDP F will take place shortly after, due to preparation and travel for the necessary personnel.

Phase E Begins	3/30/2030
Critical Events Readiness Review	4/13/2030
Decommissioning Review	9/6/2030
KDP F:	9/13/2030

Table 38. Phase E schedule.

Decommissioning (phase F): The final phase includes deactivating the robotic system once it is confirmed that the data collection and preservation is finished. The mission data will also be archived for future use during this phase. The final reports will be conducted as a comprehensive record of the current missions achievements and challenges. It is estimated that this will take one week to finalize the Disposal Readiness Review (DPR).

Phase F Begins	9/13/2030
Disposal Readiness Review	9/20/2030

Table 39. Phase F schedule.

1.7.4. Change Control

Team 27 has made substantial changes since the initial MCR. This involved a lengthy system of requesting changes, which is detailed in the following paragraphs.

The team first identified changes needed based on research conducted by team members. For example, Team 27 has notably eliminated the previous science objective "Determine if 10% of oxygen can be pulled from regolith in lunar conditions" from the STM. The PM and CS both continued research of this objective from the MCR, and ultimately found that costs were too pricey and dimensions were too large to include the instruments needed for the science to happen on the rover. Thus associating major amounts of risk to this objective. This information was then presented to the LSE and DPMR for consideration. Once the proposition of elimination of the objective was approved by all team leads, it was then presented before the entire team. A request for change form was then submitted by the PM. A copy of the change request form can be seen below, with a full version in the appendix.

L'SPACE Change Request Form

Deliverable Maturity Matrix

Deliverable Maturity	MCR	SRR	MDR	PDR
Mission Concept	**Baseline**	Updated	Updated	Updated
Vehicle Systems	Preliminary	**Baseline**	Updated	Updated
Cost and Schedule	Preliminary	Preliminary	**Baseline**	Updated
Risks	Preliminary	Preliminary	Preliminary	**Baseline**

Maturity Key	
Preliminary	Major architectural aspects of design are initially conceptualized
Baseline	Major architectural aspects of design are complete
Updated	Major architectural aspects of design are updated

At Baseline, TBD/TBR for these should be resolved

Team Number	**27**
Date of Request	**10/21/2024**
Area of Change	Science
Change Level*	Minor

*Minor changes include low risk, high certainty, and are well understood. Does not need a CCB.

Moderate changes include medium risk, some degree of uncertainty, and may have some SMEs questioning its validity. May need a CCB.

Extreme changes include high risk, high uncertainty, and will require a CCB before being accepted.

Description of requested change:
What is the change your team is requesting to make on the mission?

Figure 12. The change request form as filled out by the PM of Team 27.

The change request form required the PM of Team 27 to identify in a written document what the area of the change was (i.e. science) and what level the change was. The level of the change depends on the amount of risk associated with the change. High risk changes are listed as level Extreme, and must be reviewed before a Change Control Board (CCB) in order to get approval for the change. The PM was required to describe the change requested to be made, provide reasoning for the change and how it would impact the mission, and provide an updated timeline if the change were to hinder the progress of the mission. The PM began a folder to organize all change requests made, and numbered all requests for changes starting with #01. This concludes the system of requesting changes of Team 27.

Feedback from the MCR was presented in the form of mandatory Requests for Action (RFA) and suggested Advisories (ADV). The PM first went over all documented suggestions as listed from team mentors, and recorded which changes needed to be made per each sub-team in a shared spreadsheet. The PM then presented this spreadsheet of changes per sub-team to the CS, LSE, and DPMR for approval. After approval, each sub-team leader presented the changes necessary to their teams.

The PM created an extensive spreadsheet recording all changes made on a time scale of the deliverables required of Team 27. Thus far, all documented changes have been made prior to the completion of the SRR. Included in this process of addressing changes is required independent reading by each team member of the MCR feedback. All team members must report that they've completed the reading by method of reacting to messages on Discord once the task has been completed. This method has proven to be quick and reliable to see which team members have actually completed the task.

Team leads record all additional changes made by team members in their sub-teams, which are then reported to the PM. Once reported to the PM, said changes are recorded on the changes spreadsheet.

1.8. Conclusion

Despite having many setbacks, Team 27 has successfully established all components necessary to create working Mechanical, Power, Thermal, CDH, and Payload subsystems. Team 27 has conducted several trade studies to ensure the best success of the mission, which were highlighted in detail in their corresponding sections. The Team has also established the coordinates of the landing site for the mission, and the locations of the sampling sites as well. All possible alternatives for redundant systems have been analyzed and addressed if needed. All possible risks of the mission were discovered, and plans of action to address them have been made according to the level of risk they pose to the mission; as this is a low-cost mission, Team 27 has slim margin to address all risks proposed. The organization of the Team was finalized in this document, with detailed descriptions of decision-making and change processes. The decommissioning process was also reviewed, in the aspect that commands will be sent to the rover to terminate its subsystems and allow for the lunar environment to damage the rover.

It should be noted that due to the unique undermanned situation of Team 27 that there was an extension granted for the completion of this document. The Team would like to note the originally incomplete sections of the SRR were due to the Mechanical Engineer of the Team leaving the Academy within a week of the original deadline of this deliverable. Several team members also experienced illness, such as Covid-19, which resulted in an adjusted timeline of completion due to needed recovery time. As can be seen from the gaps within the organization of the team, there are not enough members to fill all primary positions. This means that anyone with available time outside of the completion of their own sections must pick up sections that were not theirs to complete. If the Team had additional time, the mechanical subsystem would have been created in greater detail, outreach plans would have been solidified, and direct costs would be more approximate. This will be remedied for the completion of the MDR.

Team 27 will next be focusing on the MDR, which will require the Programmatics sub-team to go into extreme detail about the scheduling of the mission and the risks. The Engineering sub-team will determine the manufacturing and procurement plans, including which suppliers to proceed with and which to have as backups. The Programmatics sub-team will also refine the budget via determining the entire amount of personnel needed for the remaining phases of the mission and analyzing direct, outreach, and travel costs. The Science sub-team shall aid in risk identification and mitigation techniques. The Team looks to foster better communication between members for the MDR, and to improve on the scheduling of tasks to be at more manageable times in accordance with the schedules of members.

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Declaration of AI in the Writing Process

Team 27 did utilize AI for part of the System Requirements Review. In particular, ChatGPT, developed and available for free by OpenAI, was utilized by the primary Thermal Engineer of the team to professionalize writing and remove pronouns of the sections *Thermal Management Subsystem, Thermal Management Subsystem Requirements, Thermal Management Subsystem Overview,* and *Thermal Management Subsystem Trade Studies*. The Thermal Engineer provided a document to the entire team detailing the reasoning for seeking assistance from ChatGPT, the original texts written along with the prompt provided to correct them, and the produced results.

I don't like to lie but I'm also not good at writing English in quick manner. So I brain vomited and had Chat GPT correct and remove pronouns. Green is what I wrote and what is not green is corrected egnlish

Green = self written
Not Green corrected by Chat GPT with the command "Don't transform it too drastically but make it professional and better please"

The Meyer Tools & MFG (MTM) did not provide any information about their multilayered insulations (MLI) in their data sheet. As part of the L'SPACE participant, we were not allowed to order. So, to work within the constraints, I decided to give them a zero in lieu of any information. Quest Thermal Group (QTG) has given a rather detail data sheet for their MLI for various product lines (datasheet). It's easy to figure out the total weight of the MLI. So they get a 8 for such specific information.

Meyer Tools & MFG (MTM) did not include any information about their multilayer insulation (MLI) in the data sheet. As participants in the L'SPACE program, requesting additional details was not permitted, which limited the available approach. Due to this lack of information, MTM received a score of zero. In contrast, Quest Thermal Group (QTG) provided a detailed data sheet for their various MLI product lines, making it straightforward to calculate the total weight of the MLI. For this level of specificity, QTG received a score of 8.

Figure ?. A portion of the document listed AI usage within the SRR, written by the Thermal Engineer of Team 27 (the user). The full document is available in the Appendix

For future deliverables, Team 27 will strive not to seek the assistance of AI, as originally and collectively decided.

Appendix A

Table 1. Full version of the Team 27 Mission Requirements table.

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Relevant Subsystem	Req met?
CC - 0.0	The mission shall adhere to customer constraints.	Parameters outlined in document, in relation to the given constraints and requirements	N/A	CC 0.1 - CC 0.10 MG - 1.0	Analysis	All	Met
CC - 0.1	System must adhere to mass constraint of 350 kg	Limited by the mission task, and to ensure not to overexceed mass capacity during transport.	CC - 0.0		Inspection	All	Met
CC - 0.2	System must adhere to dimension constraint of 2 m x 1.25 m x 1.25 m during transit	Limited by spacecraft transport dimensions; cannot exceed due to constraint.	CC - 0.0		Inspection	All	Met
CC - 0.3	System must explore the lunar/pits caves and perform surface science operations	Outlined mission statement and goal for research.	CC - 0.0		Inspection	Payload	Met
CC - 0.4	System must not exceed cost constraint of $425M	Limited by set amount of alloated funding. Cannot exceed due to unavalible funds.	CC - 0.0		Inspection	All	Met
CC - 0.5	System must be ready to launch by March 1st, 2030, at Cape Canaveral.	Hard deadline, set by mission timeline.	CC - 0.0		Inspection	Payload	Met
CC - 0.6	System cannot have a Radioisotope Thermoelectric Generator (RTG) or any derivative thereof; System must not exceed radioactive material cumulative mass of 5g	The usage of the RTG is prohibited, in accordance with the mission.	CC - 0.0		Inspection	Electrical, Structural, Mechanical	Met
CC - 0.7	System will be transported on a separate, primary vehicle launch (rocket) with power sufficient during transport. No scientific data will be collected during transit	Transportation structurlization and unavalibality of research during transit.	CC - 0.0		Inspection	Structural	Met
CC - 0.8	System shall arrive at target destination adhering to landing site constraints: given a 100m diameter landing zone, it shall not exceed a slope of 10 degrees, and should traversal be necessary to the science site or lunar cave, it must no be no further than 5 km from the landing zone (only applicable for mobile vehicle)	Landing site parameters/criterias outlined in mission.	CC - 0.0		Analysis	Structural, Mechanical	Met
CC - 0.9	System shall communicate data to Earth via relay of spacecraft orbiting the Moon; spacecraft must be able to communicate with the orbiting spacecraft directly, and primary mission orbiter which shall remain in a circular polar orbit of 100km during the mission's lifespan	Communications necessary in order to establish control, navigation, and data collection.	CC - 0.0		Demonstration	Electrical, Communication, Mechanical	Met
CC - 0.10	System shall undergo analog testing in a comparable volcanic environment; drop testing is necessary should the landing site not have a traversable entrance, and tether or winch system	Necessary to evaluate mission success and analyze potential faults and feasibility.	CC - 0.0		Test	All	Met
MG - 1.0	The mission shall conduct scientific research of the Mare Ingenii Pit in relation to the sustainability of human infrastructure.	In conjunction with the Mission Goal: "Develop precursor lunar robotic missions and define those scientific activities that astronauts will conduct on the Moon" and ""Provide safe and enduring habitation systems to protect individuals, equipment, and associated infrastructure"	CC - 0.0	SCI - 1.1 SCI - 1.2 SCI - 1.3 SC - 1.4	Demonstration	All	Met
SCI - 1.1	The mission shall conduct analysis and research of lunar regolith.	Analysis of lunar regolith for utilization of matter in terms of lunar mission sustainability.	MG - 1.0	SCI - 1.1.1	Demonstration	Science, Communication, Mechanical	Met
SCI - 1.1.1	The mission shall determine if 10% of oxygen (O2) can be pulled from the regolith in lunar conditions.	To determine the regolith's sustainability in terms of colonial/mission infrastructure.	SCI - 1.1		Analysis	Science, Communication,	Met
SCI - 1.2	The mission shall investigate magnetic fields within and around the Mare Ingenii Pit	To observe and analyze the abnormal magnetic fields	MG - 1.0		Analysis	Science, Communication	Met
SCI - 1.3	The mission shall analyze the viability of human habitation within the lunar pit.	To identify the plausibility of lunar colonial/mission infrastructure.	MG - 1.0	SCI - 1.3.1 SCI - 1.3.2 SCI - 1.3.3	Analysis	Science	Met
SCI - 1.3.1	The mission shall determine the properties of radiation on temperature, humidity, and pressure within the pit.	To determine the pit's sustainability in terms of colonial/mission infrastructure.	MG - 1.0		Analysis	Science, Communication, Mechanical	Met
SCI - 1.3.2	The mission shall identify and characterize the depth, height, terrain variation, and ease of access within the lunar pit.	To determine the lunar pit's infrastructure and identify its strengths and weaknesses for colonial/mission infrastructure.	SCI - 1.3.1		Analysis	Science, Navigation, Communication	Met
SCI - 1.3.3	The mission shall analyze the lunar pit's structural integrity.	To determine the lunar pit's infrastructure and identify its strengths and weaknesses for colonial/mission infrastructure.	SCI - 1.3.1		Analysis	Science	Met
SC - 1.4	The mission shall utilize a spacecraft for the duration of the mission and to fulfill mission research.	Mission requires a method of conducting research in a controlled and measurable manner.	MG-1.0 SCI - 1.1	SC - 1.4.1 SC - 1.4.2 SC - 1.4.3	Demonstration	All	Met
SC - 1.4.1	The spacecraft shall be able to traverse the lunar terrain in regards to its mission.	The Mare Ingenii Pit has a diameter of 130m; the spacecraft must be able to traverse the pit to fulfill its mission.	SC - 1.1	SC - 1.4.1.1 SC - 1.4.1.2	Testing	Structural, Mechanical, Navigation	Blank
SC - 1.4.1.1	The spacecraft will be remotely piloted to navigate the lunar terrain via cameras.	In order to navigate the spacecraft, a visual representation is needed to guide the device forward.	SC - 1.1		Demonstration	Structural, Mechanical, Navigation	Met
SC - 1.4.1.2	The spacecraft mobility parameters must be able to withstand the traversal conditions of the lunar plane.	The terrain of the lunar surface is very harsh with its sharp rocks, and micron-sized regolith dust. Its mobility features must be able to withstand these conditions.	SC - 1.1		Demonstration	Structural, Mechanical, Navigation	Met
SC - 1.4.2	The spacecraft shall have sufficient power to maintain mission operations.	Spacecraft functionality is derived off its power.	SC - 1.1		Demonstration	Electrical	Met
SC - 1.4.3	The spacecraft shall transmit and receive data to the corresponding entities for research and data collection.	Data obtained by the spacecraft must be transmitted to the research team (orbiter, space station, etc.) to be analyzed for scientific research. Must adhere to the CC - 0.9 restriction.	SC - 1.1		Demonstration	Communications	Met
SC - 1.4.4	The spacecraft will be equipped with scientific instruments to accomplish its mission tasks.	Data collection of scientific tools/instruments is crucial composing it into data that can be researched/analyzed.	SC - 1.1	SC - 1.4.4.1 SC - 1.4.4.2 SC - 1.4.4.3 SC - 1.4.4.4 SC - 1.4.4.5	Demonstration	Science, Mechanical, Electrical	Met
SC - 1.4.4.1	The scientific instrument shall be able to manipulate and measure lunar surface temperatures.	In order to accomplish the requirements outlined in SCI-1.1.1, the spacecraft must be able to measure surface temperature (or even manipulate) to identify the amount of heat needed for lunar regolith to produce 10% yield of oxygen.	SC - 1.4.4		Demonstration	Mechanical, Electrical, Communications	Met
SC - 1.4.4.2	The scientific instrument shall detect magnetic fields in the 0-5000nT range over an area of 10km.	In order to accomplish the requirements outlined in SCI-1.2, the spacecraft must be able to measure the magnetic fields within the area.	SC - 1.4.4		Demonstration	Mechanical, Electrical, Communications	Met
SC - 1.4.4.3	The scientific instrument shall detect radiation percentages of Alpha, Beta, and Gamma radiation.	In order to accomplish the requirements outlined in SCI-1.3.1, the spacecraft must be able to measure the radiation within the area.	SC - 1.4.4		Demonstration	Mechanical, Electrical, Communications	Met
SC - 1.4.4.4	The scientific instrument shall be able to determine terrain characteristics.	In order to accomplish the requirements outlined in SCI 1.3.2, the spacecraft must be able to identify the depth, height, and ease of access within lunar pits.	SC - 1.4.4		Demonstration	Mechanical, Electrical, Communications, Navigation	Met
SC - 1.4.4.5	The scientific instrument shall be able measure muon flux variations.	In order to accomplish the requirements outlined in 1.3.3, the spacecraft must be able to penetrate and measure the amount of muon flux variation.	SC - 1.4.4		Demonstration	Mechanical, Electrical, Communications	Met

Figure 2. Full version of the Team Expectations document (previously shown as a portion in **Figure ?**.).

Collaborative Team Expectations
Team 27 - L'SPACE MCA

Overview

This document has been reviewed by the Project Manager, Chief Scientist, Deputy Project Manager of Resources, and Lead Systems Engineer, and contains information about expectations for the team. Additionally, each discipline (science, engineering, programmatics) has their own expectations for smaller group work and contributions. Any questions regarding expectations should be directed to the Project Manager.

Expectations for Each Team Member

1.1 Team members will complete all work assigned by the due date.

- If team members are struggling with their assigned work and/or are not able to complete it by the assigned due date, reach out to your Team Lead or the Project Manager immediately. Communication is critical in any team.

1.2 Team members should read the work of others to hold them to a high standard of work, and to ensure full team understanding of the material (i.e. do not just read the section you complete for a document, read the *entire* document).

1.3 Team members will treat each other with respect. In the event that a disagreement arises, it should be handled professionally and calmly.

1.4 All ideas will be explained so that *all* team members, regardless of education level, can understand them.

1.5 Team members should make the majority of the whole group and sub-team meetings. If meetings are not attending, you must reach out to team leads or the PM for information on what you missed, and consult the meeting notes document for the meeting.

Expectations for Science Team Members

- Science team members will attend sub-team meetings unless previous notice has been given. Meetings are such a vital part to allowing everyone to be informed about important decisions.
- If you don't understand something, please ask! An issue will likely arise if you don't feel crystal clear on a topic.

Expectations for Engineering Team Members

- Engineering sub-team will attend the meetings that are scheduled by me (LSE). In the meetings, we will cover a lot of engineering problems and analysis, so I expect everyone to pay attention and have their phones put aside.
- Have a questioning attitude throughout the team and during meetings, please ask as many questions as possible and do not be shy speaking up if you do not understand something.

- Draw out any ideas you have for anything and show it to the team, a picture is worth a thousand words!

Expectations for Programmatics Team Member

- General expectations for each team member will be followed within the programmatics sub-team.
- All messages sent in the programmatics channel on Discord will be required to be read and reacted to to ensure all teammates are up-to-date and actively understanding and processing information sent out.
- Members in the programmatics sub-team will maintain communication through private messages with their team lead (DPMR) and public channel conversation.
- Progress on tasks by each team member will be regularly sent into the public programmatics Discord channel, so that all are aware of what is being worked on and can review each other's work to ensure high quality.
- Programmatics sub-team meetings have required attendance for all of its members.

Resources Available for Team Members

- MCA Resources Folder – [link]
    - This google drive made by MCA Staff contains several templates as well as directions for what to do for deliverables.
- Weekly Role Meetings
    - Role meetings can be attended for questions that are unable to be answered by the team. We ask you to take notes during the meetings you attend and share these with the team to ensure total team understanding.
    - Weekly Role Specific Meeting Schedule - FA24

      10/14 - 12/6

Time (PT)	MONDAY	TUESDAY	WEDNESDAY	THURSDAY	FRIDAY
12:00 PM					
1:00 PM					
2:00 PM		Electrical Engineer Meeting Click to Join Zoom			CDH Engineer Meeting Click to Join Zoom
3:00 PM			Mechanical Engineer Meeting Click to Join Zoom		
4:00 PM			Thermal Engineer Meeting Click to Join Zoom	Science Meeting Click to Join Zoom	
5:00 PM	Program Analyst Meeting Click to Join Zoom		Mission Assurance Meeting Click to Join Zoom		
6:00 PM					

- Project Manager Availability – Josie Wood
    - I have my Discord notifications on, so feel free to send me a message whenever and I will get back to you when I can. I typically am working/have class 9-6 Monday through Friday, but I typically respond within two hours (unless it's past 11pm and I'm sleeping).
- Chief Scientist Availability – Jessica Ray
    - My availability is ever changing since I don't have a set schedule as an online student. If a question or concern arises, don't hesitate to message me; I should respond within 4 hours max.
- Lead Systems Engineer Availability – Kaushal Patel
    - ➢ My availability varies, but I'm free mostly in the evenings starting from 6:30pm Monday through Friday until 9pm. For weekends I'm mostly free throughout the whole day. I prefer to be on call rather than texting so we do not waste time texting and going back and forth. Discord works for me, so if you have any questions or want to talk, DM me on Discord and we can go from there.
- Deputy Project Manager of Resources Availability – Lyss Harnden
    - Please feel free to send me questions via Discord DM at any time, I will get back to you within 1-2hrs. I am free Monday through Friday 7am-11pm CT; on weekends I have less availability due to making time for loved ones and hobbies! However, you can still expect a response by me during the weekend within a few hours.

Figure 3. Math worked out for the thermal management subsystem.

Constants				
Rover Surface Temp	240	k	Value chosen by avg hot and cold range for instruments	
Boltzmann constant	0.0000000567			
Solar Flux	1440	w	Given by skill module	
White Paint (Alpha)	0.19		Given by skill module	
Top Surface of Rover	0.625	m^2	Dummy Value	Dummy values should be the only ones being changed
Side of Rover (Long)	0.625	m^2	Dummy Value	
Side of Rover (Short)	0.391	m^2	Dummy Value	

COLD SCENERIO		
Moon Surface Temperature	140	
Aluminuim Covered in MLI	0.005	Epsilon
Rover-Space	3317759919	
Rover-Surface	2933600000	
Q-Solar	0	
Q-Rad-Space	22.33889225	
Q-Rad-Space-Surface	-5.462452985	
Q-Rad-Surface	0.51979725	
Q-Internal	100	
Q-In	122.3388923	
Q-Out	-4.942655735	
Q-Total	117.3962365	
Given the results, we will need to have TCS to radiate extra energy		

HOT SCENERIO		
Moon Surface Temperature	400	
Aluminuim Covered in MLI	0.005	Epsilon
Rover-Space	3317759919	
Rover-Surface	-22282240000	
Q-Solar	171	
Q-Rad-Space	22.33889225	
Q-Rad-Space-Surface	-5.462452985	
Q-Rad-Surface	-3.9481344	
Q-Internal	100	
Q-In	293.3388923	
Q-Out	-9.410587385	
Q-Total	283.9283049	

Figure 4. The complete budget for the mission, including unchanged charts by L'SPACE.

Additional Information						
	Phase C	Phase C	Phase C-D	Phase D	Phase D	Phase E-F
# People on Team	FY 1	FY 2	FY 3	FY 4	FY 5	FY 6
Science Personnel:	2	2	2	2	2	10
Engineering Personnel:	20	20	20	20	20	15
Technicians:	15	15	15	15	15	0
Administration Personnel:	9	9	9	9	9	9
Management Personnel:	5	5	5	5	5	5

NASA L'SPACE Mission Concept Academy Budget - *Team 27*

Mission Phase	Phase C	Phase C	Phase C-D	Phase D	Phase D	Phase E-F	
Year	**Year 1**	**Year 2**	**Year 3**	**Year 4**	**Year 5**	**Year 6**	**Cumulative Total**
PERSONNEL							
Science Personnel	$ 160,000	$ 164,160	$ 168,320	$ 172,480	$ 176,640	$ 904,000	$ 1,745,600
Engineering Personnel	$ 1,600,000	$ 1,641,600	$ 1,683,200	$ 1,724,800	$ 1,766,400	$ 1,356,000	$ 9,772,000
Technicians	$ 900,000	$ 923,400	$ 946,800	$ 970,200	$ 993,600	$ -	$ 4,734,000
Administration Personnel	$ 540,000	$ 554,040	$ 568,080	$ 582,120	$ 596,160	$ 610,200	$ 3,450,600
Project Management	$ 600,000	$ 615,600	$ 631,200	$ 646,800	$ 662,400	$ 678,000	$ 3,834,000
Total Salaries	$ 3,800,000	$ 3,898,800	$ 3,997,600	$ 4,096,400	$ 4,195,200	$ 3,548,200	**$ 23,536,200**
Total ERE	$ 1,060,580	$ 1,088,155	$ 1,115,730	$ 1,143,305	$ 1,170,880	$ 990,303	**$ 6,568,953**
Personnel Margin	$ 1,389,614	$ 1,462,814	$ 1,537,892	$ 1,614,848	$ 1,693,684	$ 1,435,982	**$ 9,134,834**
TOTAL PERSONNEL	$ 6,250,194	$ 6,617,463	$ 6,997,085	$ 7,389,209	$ 7,793,980	$ 6,751,168	$ 41,799,099
TRAVEL							
Total Flights Cost			$ 2,500		$ 34,720	$ 2,500	**$ 39,720**
Total Hotel Cost			$ 7,500		$ 81,128	$ 7,500	**$ 96,128**
Total Transportation Cost			$ 800		$ 7,450	$ 800	**$ 9,050**
Total Per Diem Cost			$ 2,080		$ 36,405	$ 2,080	**$ 40,565**
Travel Margin			$ 1,271		$ 17,613	$ 1,365	**$ 20,249**
Total Travel Costs	$ -	$ -	$ 14,887	$ -	$ 195,757	$ 16,097	**$ 226,741**

OUTREACH							
Total Outreach Materials	$ 4,250,000						**$ 4,250,000**
Total Outreach Venue Costs							**$ -**
Total Outreach Travel Costs							**$ -**
Total Outreach Services Costs							**$ -**
Total Outreach Personnel Costs							**$ -**
Outreach Margin							**$ -**
Total Outreach Costs	$ 4,250,000	$ -	$ -	$ -	$ -	$ -	**$ 4,250,000**
DIRECT COSTS							
Mechanical Subsystem		$ 21,250,000					**$ 21,250,000**
Power Subsystem		$ 45,378					**$ 45,378**
Thermal Control Subsystem		$ 21,250,000					**$ 21,250,000**
Comms & Data Handling Subsystem		$ 34,000,000					**$ 34,000,000**
Guidance, Nav, & Control Subsystem		$ 42,500,000					**$ 42,500,000**
Science Instrumentation		$ 394,032					**$ 394,032**
Spacecraft Cost Margin		$ 125,816,052					**$ 125,816,052**
Total Spacecraft Direct Costs	$ -	$ 251,632,104	$ -	$ -	$ -	$ -	**$ 251,632,104**
Manufacturing Facility Cost		$ 800,000					**$ 800,000**
Test Facility Cost		$ 290,000					**$ 290,000**
Facility Cost Margin		$ 124,620					**$ 124,620**
Total Facilities Costs	$ -	$ 1,246,200	$ -	$ -	$ -	$ -	**$ 1,246,200**
Total Direct Costs	$ -	$ 252,878,304	$ -	$ -	$ -	$ -	**$ 252,878,304**
Total MTDC	$ -	$ 251,632,104	$ -	$ -	$ -	$ -	**$ 251,632,104**
FINAL COST CALCULATIONS							
Total F&A	$ -	$ 12,581,605	$ -	$ -	$ -	$ -	**$ 12,581,605**
Total Projected Cost	$ 10,500,194	$ 272,077,372	$ 7,011,972	$ 7,389,209	$ 7,989,737	$ 6,767,265	**$ 311,735,749**
Total Cost Margin	$ 1,389,614	$ 127,403,486	$ 1,539,162	$ 1,614,848	$ 1,711,297	$ 1,437,347	**$ 135,095,755**
	13.2%	46.8%	22.0%	21.9%	21.4%	21.2%	
Total Project Cost	$ 10,500,194	$ 272,077,372	$ 7,011,972	$ 7,389,209	$ 7,989,737	$ 6,767,265	**$ 311,735,749**

********** Do not change the numbers in the cells below unless mission concept instructions specify otherwise.						
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					

Figure 5. The complete change request form used for requesting to change the lunar regolith science objective from the STM.

L'SPACE Change Request Form

Deliverable Maturity Matrix

Deliverable Maturity	MCR	SRR	MDR	PDR
Mission Concept	**Baseline**	Updated	Updated	Updated
Vehicle Systems	Preliminary	**Baseline**	Updated	Updated
Cost and Schedule	Preliminary	Preliminary	**Baseline**	Updated
Risks	Preliminary	Preliminary	Preliminary	**Baseline**

Maturity Key	
Preliminary	Major architectural aspects of design are initially conceptualized
Baseline	Major architectural aspects of design are complete
Updated	Major architectural aspects of design are updated

At Baseline, TBD/TBR for these should be resolved

Team Number	27
Date of Request	10/21/2024
Area of Change	Science
Change Level*	Minor

*Minor changes include low risk, high certainty, and are well understood. Does not need a CCB.

Moderate changes include medium risk, some degree of uncertainty, and may have some SMEs questioning its validity. May need a CCB.

Extreme changes include high risk, high uncertainty, and will require a CCB before being accepted.

Description of requested change:
What is the change your team is requesting to make on the mission?

Team 27 would like to eliminate the science objective previously stated on the MCR STM as follows: determine if 10% of oxygen can be pulled from regolith in lunar conditions.
Reason for change and its impact: What is the reason your team wanted to make this change? How will this impact your mission? Think in terms of cost, schedule, and scope*.
Team 27 would like to eliminate this science objective, and NOT replace it with a new one, because it is out of the constraints of the mission. Achieving this objective would involve a carbothermal reactor, which far exceeds the cost and dimension constraints of the mission. The team is unable to incorporate feasible instruments on the rover to be able to conduct the science for this objective. Removing this objective effectively reduces the risk of the mission, as the carbothermal reactor would have had a high level of risk in regards to malfunctioning. While the carbothermal reactor method for extracting oxygen was a TRL 6, the massive size reduction, and new lunar environment, would have taken the reactor down to a TRL of 1. Eliminating this objective allows for more funding to be allocated to the subsystems of Team 27's rover, namely the mechanical subsystem to ensure minimal failure of the chassis.
Timeline and action plan: When can this change be expected to be implemented by? Use academy deliverables as key milestones (MCR, SRR, MDR, PDR) that signify a particular date. How will this change be implemented? What key personnel will be responsible for its completion?
This change will be implemented immediately. It has already been proposed to all members of Team 27. The team merely needs approval in order to begin the plan drafted to be implemented once this change is made. Namely, this change will be implemented starting in the SRR and will continue to be used throughout the MDR and PDR. All members of the science subteam will be in charge of ensuring these changes are made and followed through with.
Trade Study if relevant: If your change consists of swapping one thing for another (system related changes), you must complete a trade study. Include a screenshot of it below. If your change does not fall into this category, leave this blank.

*Scope would be defined as the relevant work, such as tasks performed by the mission, or the deliverables of these tasks and their quality / quantity.

Mentor Response	Select an option
Date of Response	
When2meet Link (If CCB is needed)	
Reason for Response	

Figure 6. The complete document describing the usage of AI within the SRR by Team 27.

I don't like to lie but I'm also not good at writing English in quick manner. So I brain vomited and had Chat GPT correct and remove pronouns. Green is what I wrote and what is not green is corrected egnlish

Green = self written	
Not Green corrected by Chat GPT with the command "Don't transform it too drastically but make it professional and better please"	

The Meyer Tools & MFG (MTM) did not provide any information about their multilayered insulations (MLI) in their data sheet. As part of the L'SPACE participant, we were not allowed to order. So, to work within the constraints, I decided to give them a zero in lieu of any information. Quest Thermal Group (QTG) has given a rather detail data sheet for their MLI for various product lines (datasheet). It's easy to figure out the total weight of the MLI. So they get a 8 for such specific information.

Meyer Tools & MFG (MTM) did not include any information about their multilayer insulation (MLI) in the data sheet. As participants in the L'SPACE program, requesting additional details was not permitted, which limited the available approach. Due to this lack of information, MTM received a score of zero. In contrast, Quest Thermal Group (QTG) provided a detailed data sheet for their various MLI product lines, making it straightforward to calculate the total weight of the MLI. For this level of specificity, QTG received a score of 8.

To calculate the cost of the MLI I will be using the formula used in the Mission Concept Cost Estimate Tool (MCCET). With that formula we must have the total Thermal Mass. The we do the following calculations: 642*ThermMass^.62 = Total Cost of MLI. To find the total ThermMass, I was given a formula by the Thermal Engineers in the second meeting for thermo engineers. The formula is the following Area * Layers. The area part of the equation is the total area of the rover where the MLI will cover the rover. The current working surface area of the rover is .6641 m^2. While the layers is calculated by the chart given on page 166 of the Spacecraft Thermal Control Handbook (STCH)(Source book). The effective emittance required for our operation is .005. By looking at the chart in the STCH, we can see that between 28-30 layers is required to reach that emittance. For both MTM and QTG will have to use the same amount of layers used, of 30. And cover the same amout of surface area (.6641 m^2). So both materials will weigh the same amount (19.9kg). When plugging the Thermal Mass in the formula, we get a total cost of $6,332,611.43. Since both cost the same they receive the same score, 6.

To calculate the cost of the MLI, the formula from the Mission Concept Cost Estimate Tool (MCCET) will be utilized, requiring the total Thermal Mass. The calculation is as follows: 642×(ThermMass)0.62=Total Cost of MLI. To determine the total Thermal Mass, the formula provided by the Thermal Engineers in the second meeting is used: Area×LayersArea×Layers. Here, "Area" refers to the rover's surface area that the MLI will cover, currently 0.6641 m^2. The number of layers is based on the chart on page 166 of the

Spacecraft Thermal Control Handbook (STCH), which specifies the effective emittance needed for operation as 0.005. According to the STCH, achieving this emittance requires 28-30 layers. Selecting 28 layers allows for a lighter material without sacrificing emissivity. With an area of 0.6641 m² and 28 layers, both materials have the same weight of 19.9 kg. Substituting this Thermal Mass into the MCCET formula yields a total cost of $6,071,808.60. Since both companies cost the same, a score of 6 was assigned to each.

For the complexity, they both the MTM and QTG receive a of score of 10. From a materials point of view, they do not need to be controlled by the rover to be operated. Since it's just a material with no moving or electrical parts, then there is low risk for the material to fail. The only risk that will need to be calculated is to make sure that the material does not get caught on a surface that will damage it or remove it from the chassis of the rover.

For complexity, both MTM and QTG received a score of 10. From a materials standpoint, neither requires active control by the rover to operate, as they are passive materials with no moving or electrical parts. This significantly reduces the risk of failure. The main risk to consider is ensuring that the material does not catch on any surfaces that could damage or dislodge it from the rover's chassis

In terms of reliability, again they are passive elements that require no input from the rover to operate. Once installed, the main focused is to keep the MLI from getting any type of physical damage. So both companies receive a score of 10 aswell for their reliability

In terms of reliability, both materials are passive elements that require no input from the rover to function. Once installed, the primary concern is to protect the MLI from any physical damage. Therefore, both MTM and QTG received a score of 10 for reliability.

In terms of Technology Readiness Level, the score for this technology would be a 6. MLI is a standard tool used in the aerospace industry since back from NASA inception. It's a well-tested technology used in many spacefaring vehicles (Not just NASA). So it's a matured technology. However, it has not been tested in a lunar rover that will explore a lunar car. So it will be tested in that environment for the first time. Thus it receives a low TRL.

For Technology Readiness Level (TRL), this MLI technology scores a 6. MLI has been a standard tool in the aerospace industry since NASA's inception and is widely used across various spacefaring vehicles, not limited to NASA. It's a well-established, mature technology. However, it has yet to be tested on a lunar rover designed for lunar exploration, so it will face this specific environment for the first time, which contributes to its moderate TRL score.

The rover created by team 27 will be using Quest Thermal Group's multi-layered insulation for our exploration of lunar caves

Cryocoolers (Trade Studies)

The cryocooler will be require to cool down the rover since it will be running warm. For this trade studies, the Lockheed Martin Space - MICRO1-2 cryocooler and the SunPower – Ametek: CryoTel DS Mini cryocooler will be compared.

The cryocooler will be required to cool the rover, as it will operate at elevated temperatures. This trade study will compare the Lockheed Martin Space MICRO1-2 cryocooler and the SunPower Ametek CryoTel DS Mini cryocooler.

For the mass criteria, Lockheed Martin (LM) provides a lengthy data sheet, but the information about its mass is not found on the data sheet but NASA provides the weight on their webpage. The Lockheed Micro 1-2 weighs 0.475kg. So, it will receive a score of 7 for its mass. SunPower (SP) has a data sheet that includes a mass for their cooler (1.2kg). For the mass category SP will receive a 6.

For the mass criteria, Lockheed Martin (LM) provides a detailed data sheet; however, the specific mass information is not included. NASA's website indicates that the Lockheed Micro 1-2 weighs 0.475 kg, resulting in a score of 7 for mass. SunPower (SP) includes the mass of its cooler, which is 1.2 kg, in its data sheet, earning SP a score of 6 in the mass category.

In terms of cost, the same MCCET equation will be used. The Total cost for Lockheed's device will be $624,956.90. Lockheed will be receiving a 8 in this category. Applying the MCCET formula to SunPower's cooler gives us a cost of $1,110,161.05. Earning SP a cost number of 7

In terms of cost, the same MCCET equation will be applied. The total cost for Lockheed Martin's device is $624,956.90, resulting in a score of 8 for this category. Applying the MCCET formula to SunPower's cooler yields a cost of $1,110,161.05, earning SP a score of 7 in the cost category.

Reliability is a crucial criteria for an active system. Since they will be active there are more risk inherently associated with it. The scores for the following companies is as follows. Lockheed gets a score of 3 for reliability. While SunPower receives a score of 0. SunPower does not describe how reliable their cryocooler is. While Lockheed shows in their data sheet that they have reliability when designing their cryocooler. However, their claims will still need to be tested to make sure they are acurate

Reliability is a crucial criteria for an active system, as inherent risks are associated with their operation. The scores for the two companies are as follows: Lockheed Martin receives a score of 3 for reliability, while SunPower receives a score of 0. SunPower does not provide information regarding the reliability of their cryocooler. In contrast, Lockheed Martin's data sheet indicates that reliability considerations were taken into account during the design of their cryocooler. However, these claims will need to be validated through testing to ensure their accuracy.

The performance category will be determined by the following formula: Cooling power % wattage. For this criteria, Lockheed will receive a score of 5 since their cooler uses up 25 W while cooling down by 105K. Which results in a ratio f 4.2. SunPower receives a score of 2 because their device consumes 45 watts while cooling down 77k. Granting them a ratio of 1.71.

The performance category will be evaluated using the formula: Cooling Power % Wattage. For this criterion, Lockheed Martin will receive a score of 5, as their cooler consumes 25 W while providing a cooling effect of 105 K, resulting in a ratio of 4.2. In contrast, SunPower receives a score of 2, as its device consumes 45 W while achieving a cooling effect of 77 K, yielding a ratio of 1.71.

In terms of TRL, both devices will be the first ones to go to a mission that on the moon that will be exploring the caves and gathering samples to be studied. On NASA's webpage, Lockheed's device is rated as a TRL of 7 and the SunPower has a TRL of 6, but as stated before they will be in different mission and environment. So that will be dropping the TRL of both technologies down. 5 and 4 respectively.

In terms of Technology Readiness Level (TRL), both devices will be the first to undertake a mission on the Moon, exploring caves and gathering samples for study. According to NASA's website, Lockheed Martin's device is rated as TRL 7, while SunPower's device has a TRL of 6. However, since both will operate in different missions and environments, the TRL for both technologies will be adjusted downward to 5 and 4, respectively.

The rover developed by Team 27 will utilize SunPower's Cryocooler for the exploration of lunar caves.

Figure 7. “The Mars 2020 Rover Engineering Cameras.” NASA. Accessed October 31, 2024..

THE MARS 2020 ROVER ENGINEERING CAMERAS. J. N. Maki[1], C. M. McKinney[1], R. G. Willson[1], R. G. Sellar[1], D. S. Copley-Woods[1], M. Valvo[1], T. Goodsall[1], J. McGuire[1], K. Singh[1], T. E. Litwin[1], R. G. Deen[1], A. Culver[1], N. Ruoff[1], D. Petrizzo[1], [1]Jet Propulsion Laboratory, California Institute of Technology (4800 Oak Grove Drive, Pasadena, CA 91109, Justin.N.Maki@jpl.nasa.gov).

Introduction: The Mars 2020 Rover is equipped with a next-generation engineering camera imaging system that represents a significant upgrade over the previous Navcam/Hazcam cameras flown on MER and MSL [1,2]. The Mars 2020 engineering cameras acquire color images with wider fields of view and higher angular/spatial resolution than previous rover engineering cameras. Additionally, the Mars 2020 rover will carry a new camera type dedicated to sample operations: the Cachecam.

History: The previous generation of Navcams and Hazcams, known collectively as the engineering cameras, were designed in the early 2000s as part of the Mars Exploration Rover (MER) program. A total of 36 individual MER-style cameras have flown to Mars on five separate NASA spacecraft (3 rovers and 2 landers) [1-6]. The MER/MSL cameras were built in two separate production runs: the original MER run (2003) and a second, build-to-print run for the Mars Science Laboratory (MSL) mission in 2008. Newer technologies and electronics parts obsolescence have brought the MER/MSL camera production to a close, with no additional production runs planned. The Mars 2020 MEDA SkyCam [7] will likely be the last MER/MSL camera to fly to Mars (the SkyCam utilizes a flight spare Hazcam from MSL).

Instrument Functional Description: The Navcams provide stereo image data for rover traverse planning, targeting of remote sensing instruments, operation of the robotic arm, and the acquisition of panoramas with a 360° field of regard around the rover. The Hazcams acquire context images immediately forward and aft of the rover (in particular the areas not viewable by the Navcams) for traverse planning, robotic arm operations, support for rover fine positioning, and imaging of the front and rear wheels. Both the Navcam and Hazcam cameras provide stereo image data to the rover autonomous navigation system. The Cachecam, a new type of rover camera, will document sample handling operations inside the rover sample caching system (SCS).

Instrument Description

The Mars 2020 engineering cameras incorporate heritage design principles from MER/MSL: 1) small, ruggedized camera bodies with fixed-focus lenses, 2) a simple camera head with limited processing and storage capabilities (image storage and compression functions are performed separately, by the spacecraft/rover computer) and 3) cameras mounted together as stereo pairs (the Cachecam, a monoscopic camera, is an exception). The Mars 2020 engineering cameras are packaged into a single, compact camera head (see figure 1).

Figure 1. Flight Navcam (left), Flight Hazcam (middle), and flight Cachecam (right). The Cachecam optical assembly includes an illuminator and fold mirror.

Each of the Mars 2020 engineering cameras utilize a 20 megapixel OnSemi (CMOSIS) CMOS sensor equipped with a Bayer pattern RGB color filter array. The Mars 2020 Navcams, mounted on the pan/tilt Remote Sensing Mast (RSM), will acquire color stereo images (figure 2) and panoramas from a height of approximately 2 meters above the Martian surface (figure 3). The Mars 2020 Hazcams are hard-mounted to the rover body at a height of approximately 0.7 meters above the surface (figure 3) and will acquire color stereo images of the areas immediately in the front and rear of the rover. The Cachecam will acquire color images of sample materials as they are being processed by the Mars 2020 Sample Caching System (figure 4).

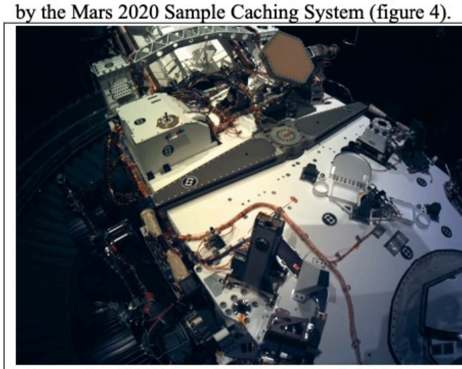

Figure 2. Single frame Navcam image of the Mars 2020 flight rover deck, acquired during system thermal vacuum testing in October 2019.

Table 1. Mars 2020 Engineering Camera Summary

Sensor Capabilities	
Type	20 MegaPixel CMOS
Optical Format	35 mm
Array Size	5120 x 3840 pixels
Pixel Size and Pitch	6.4 microns
Full well charge	15,000 e-
Pixel Dark Noise	8 e- RMS
Pixel Dark Current	125 e-/s @ 25C
Windowing	Region of Interest (ROI)
Shutter	Global
Pixel Quantization	12 bits
Electrical Interface	
Commanding & Data	LVDS
Protocol	MER/MSL/Mars2020 NVMCAM
Power Input	+5 V (+/- 0.5 V)
Power	< 10 W
Memory	1 Gbit SDRAM
Non-Volatile FPGA	MicroSemi Rad-Tolerant ProASIC3
Camera System Specifications	
Mass (CBE, no optics)	[< 425 g]
Volume (CBE, no optics)	[75 mm x 85 mm x 55 mm]
Operating Temperature	-55C to +50C
Survival Temperature	-135C to +70C
Mars2020 Optics Configurations	
Navigation Camera (Color)	FOV = 96° X 73°(H x V), f/12, IFOV = ~ 0.33 mrad 42.4 cm stereo baseline
Hazard Camera (Color)	FOV = 136° X 102°(H x V), f/12, IFOV = ~ 0.45 mrad
Sample Camera (Color)	EFL = ~ 30mm, Focus Distance = 270mm, f/8

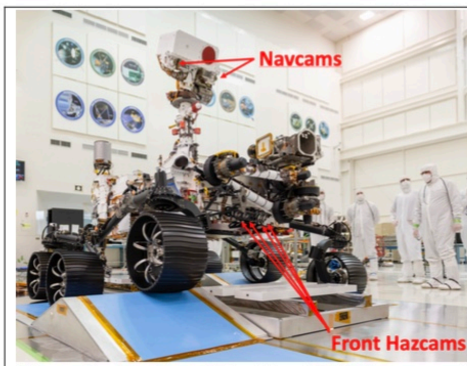

Figure 3. Location of the Mars 2020 Navcams and Front Hazcams. The Cachecam camera is located insided the rover body.

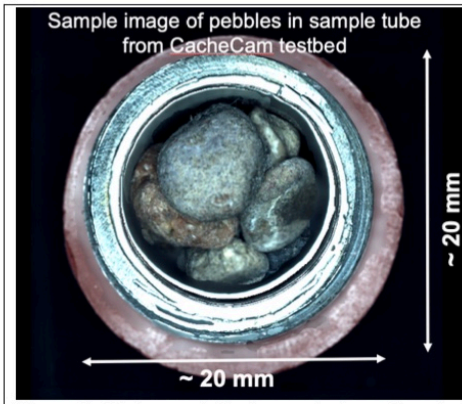

Figure 4. Cachecam testbed example image (acquired from a prototype camera).

Current Status: The Mars 2020 engineering cameras were delivered to the Mars 2020 project in the mid-2019 and integrated onto the rover shortly thereafter. The cameras have completed a variety of system tests in Mars 2020 ATLO, including thermal vacuum testing and geometric camera calibration. Performance of the cameras has been nominal during this period. Final prelaunch closeout imaging testing is scheduled for March of 2020. For a full description of the Mars 2020 engineering cameras, see [8]. The Mars 2020 launch window opens on July 17th, 2020, with landing on Mars scheduled for February 18th, 2021.

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