

Preliminary Design Review

Team #24

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Table of Acronyms

Abbreviation	Definition
ADC	Analog-to-Digital Converter
ADCS	Attitude Determination and Control System
ALARA	As Low As Reasonably Achievable
APU	Application Processing Unit
BS	Base Station
CAD	Computer-Aided Design
CAN-FD	Controller Area Network Flexible Data-Rate
CCSDS	Consultative Committee for Space Data Systems
CDH	Command and Data Handling
CDR	Critical Design Review
CER	Cost Estimating Relationship
CFS	Core Flight System
CLPS	Commercial Lunar Payload Services
CMOS	Complementary Metal-Oxide-Semiconductor
COTS	Commercial Off-The-Shelf
Det	Detection (FMEA)

DOE	Diffractive Optical Element
DR	Drone
DSN	Deep Space Network
DTE	Direct-to-Earth
DVR	Digital Video Recorder
ECSS	European Cooperation for Space Standardization
EDAC	Error Detection and Correction
EDL	Entry, Descent, and Landing
EM	Electromagnetic
EMI	Electromagnetic Interference
EPS	Electrical Power System
ESDMD	Exploration Systems Development Mission Directorate
FDIR	Fault Detection, Isolation, and Recovery
FMEA	Failure Mode and Effects Analysis
FOV	Field of View
FPGA	Field-Programmable Gate Array
fps	Frames Per Second
GaN	Gallium Nitride
GDS	Ground Data System

GNC	Guidance, Navigation, and Control
GNSS	Global Navigation Satellite System
HDR	High Dynamic Range
HEMT	High Electron Mobility Transistor
IMU	Inertial Measurement Unit
ISS	International Space Station
JPL	Jet Propulsion Laboratory
KDP	Key Decision Point
LADEE	Lunar Atmosphere and Dust Environment Explorer
LCL	Latching Current Limiter
LDEX	Lunar Dust Experiment
LED	Light Emitting Diode
LETS	Linear Energy Transfer Spectrometer
LVDS	Low-Voltage Differential Signaling
MEMS	Micro-Electro-Mechanical Systems
MIMU	Miniature Inertial Measurement Unit
MLI	Multi-Layer Insulation
MOS	Mission Operations System
MSE	Mini-Suite Electronics

NFT	Natural Feature Tracking
NICM	NASA Instrument Cost Model
NPD	NASA Policy Directive
NRE	Non-Recurring Engineering
NVMe	Non-Volatile Memory Express
OBC(s)	Onboard Computer(s)
Occ	Occurrence (FMEA)
ORR	Operational Readiness Review
PCB	Printed Circuit Board
PDR	Preliminary Design Review
PMAD	Power Management and Distribution
PSR	Permanently Shadowed Region
RAID	Redundant Array of Independent Disks
RAM	Random Access Memory
RF	Radio Frequency
RPN	Risk Priority Number
RSSI	Received Signal Strength Indicator
RTEMS	Real Time Executive for Multiprocessor Systems
SCALPSS	Stereo Cameras for Lunar Plume-Surface Studies

SDST	Small Deep Space Transponder
SEFI	Single Event Functional Interrupt
SEL	Single Event Latch-up
SEU	Single Event Upset
Sev	Severity (FMEA)
SGM	Semi-Global Matching
SIR	System Integration Review
SMD	Science Mission Directorate
SoC	System on a Chip
SPI	Serial Peripheral Interface
SQCOTS	Space-Qualified Commercial-Off-The-Shelf
SSD	Solid State Drive
SSPA	Solid State Power Amplifier
STM	Science Traceability Matrix
SUDA	Surface Dust Analyzer
TAGCAMS	Touch-and-Go Camera System
TCS	Thermal Control Subsystem
TID	Total Ionizing Dose
ToF	Time-of-Flight
TRL	Technology Readiness Level
TRN	Terrain Relative Navigation

TVAC	Thermal Vacuum
UHF	Ultra-High Frequency
VIBE	Vibration Testing
VIPER	Volatiles Investigating Polar Exploration Rover

1. Mission Overview

1.1. Mission Statement

Sustained lunar operations will require high-fidelity terrain characterization, environmental hazard mapping, and surface mobility assessments that do not yet exist at the lunar south pole. This mission performs foundational environmental surveying and aerial maneuverability to ensure safe, long period surface exploration. Operating from the Shackleton-de Gerlache Ridge, the mission characterizes how extreme terrain gradients, thermal transitions, radiation sensitivity, and dust mobilization constrain surface navigation across the surrounding exploration zone, including nearby permanently shadowed regions (PSRs).

The mission employs a dual-platform architecture: a stationary Base Station that serves as the power, communications, environmental sensing, and data-handling hub, and an autonomous scout drone that provides mobility into terrain the Base Station cannot reach. The drone carries imaging, depth-sensing, and surface-mapping instruments for aerial and in-PSR data gathering, while the Base Station hosts environmental monitoring, primary science-processing, data storage, and Direct-to-Earth downlink functions. Measurements include 3D topographical elevation and slope from the TAGCAMS Optical Navigation Suite and Stereoscopic Terrain Camera, surface albedo and thermal emissions from the same optical/terrain-imaging suite, airborne dust charging and flux behavior from the Dust Impact Mass Spectrometer and Radiation Spectrometer, and real-time flight hazard detection from the Time-of-Flight Depth Camera across the exploration zone.

These measurements directly address the science question of how the south-pole environment constrains reliable surface navigation, and instrument performance. The resulting dataset establishes a topographical, thermal, and environmental hazard baseline which provides critical data on PSR slope accessibility, regolith mobilization thresholds, and optical limits that future Artemis-era crewed surface missions can directly build upon.

1.2. Science Traceability Matrix

The mission architecture utilizes a stationary Base Station alongside an autonomous Scout Drone to provide coverage into rugged terrain that ground-based platforms cannot safely reach. This design directly aligns with the NASA Science Mission Directorate (SMD) goal of evaluating interactions between the environment and exploration activities, including dust, radiation, and impact hazards, and the Exploration Systems Development Mission

Directorate (HSMD) goal of enabling safe, expandable surface mobility and hazard avoidance using a fully autonomous mobility platform. The mission addresses the overarching science question of how south-pole illumination extremes, regolith dynamics, thermal transitions, and topography constrain reliable surface navigation and payload performance. The mission is guided by four core science objectives.

The first objective investigates how extreme terrain and illumination gradients affect Terrain Relative Navigation (TRN) systems, utilizing the TAGCAMS Optical Navigation Suite and Stereoscopic Terrain Camera to generate 3D topographical point clouds at 0.03 m/pixel spatial resolution, slope maps across a 0°–90° range, and NIR albedo profiles (380–1050 nm) across illuminated crater surfaces and dark PSR environments, returning imagery data throughout each excursion for post-flight terrain-navigation correlation analysis.

Second, the mission characterizes the electrostatic regolith environment, micrometeorite impact, and surface radiation using a Dust Impact Mass Spectrometer and Radiation Dosimeter to record electrostatic charge accumulation (0.05 pA–150 nA), impactor mass spectra (1–200 amu), and cumulative Total Ionizing Dose (up to 130 krad(Si)) continuously across the mission's operational lifetime, evaluating airborne dust charging during mission excursions and flight, physical surface degradation, and long-term component survivability.

Third, platform flight dynamics, vibration spectra, and structural stability are evaluated using a Space-Grade Gyro and IMU pair to monitor acceleration profiles across a $\pm 20g$ dynamic range and vibration frequencies from 0.05–2500 Hz, generating flight-dynamics telemetry each excursion to quantify stability and maneuverability in lunar gravity.

Finally, automated flight protocols and live hazard avoidance are established using a Flight Computer Software & Power Monitor paired with a Time-of-Flight (ToF) Depth Camera, returning obstacle-detection data down to 5 cm feature sizes across a 0.50–15.0 m range, together enabling energy-optimal trajectory planning across each excursion cycle.

To accomplish these objectives, the mission relies on a complementary dual-platform operational framework. The stationary Base Station serves as the central command, thermal management, power distribution, environmental sensing, and Direct-to-Earth (DTE) communication hub. Meanwhile, the autonomous Scout Drone acts as an aerial asset, executing transects and entering PSRs to gather scientific, topological, and engineering data.

Science Goals	Science Objectives	Science Measurement Requirements		Instrument Performance Requirements		Predicted Instrument Performance	Instrument	Mission Requirements
		Physical Parameters	Observables					
Evaluate the lunar south pole's environmental hazards for navigation reference stations using illumination, dust mobilization, and terrain topology to develop mitigation strategies that preserve vehicle functionality over mission lifespan.	Determine how extreme the terrain and illumination gradients of the lunar south pole impact the operational functionality of Terrain Relative Navigation (TRN) systems.	Define the elevation (from a location-specific datum), terrain slope changes (using a raster with values per pixel), surface texture, surface albedo, and crater density.	Map 3D topographical point clouds to measure elevation (± 0.5 m vertical accuracy), slope (0° – 90°), and hazard geometry. Measure reflected NIR intensity (380–1050 nm) and surface albedo across the exploration zone.	Spatial Resolution:	≤ 0.10 m/pixel @ 10m range relative	0.03 m/pixel	TAGCAMS Optical navigation suite & FOV Stereoscopic Terrain Camera	The mission shall process telemetry from the Drone and downlink via the Base Station's DTE link whilst sustaining drone operations across the lunar thermal range (130°F to -334°F) during each excursion
				Field of View:	$\geq 60^\circ \times 60^\circ$	$75^\circ \times 75^\circ$		
				Spectral Wavelength:	400 - 1000 nm	380–1050 nm		
				Imaging Frame Rate:	≥ 30 fps real-time	30 fps continuous		
	Characterize the micrometeorite impact environment and electrostatic regolith mobilization at the lunar south pole to evaluate physical surface degradation and vehicle safety	Identify electrostatic properties (in nanoAmperes), the elemental material composition of impactors, and impact kinetic energy.	Record electrostatic current accumulation (nA) and collect mass spectra (1–200 amu) to determine impactor composition. Measure multi-angle albedo variations to detect micrometeorite impacts across (MeV) energy range.	Impact Charge Signal Sensitivity:	0.1 pA - 100 nA	0.05 pA – 150 nA	Dust Impact Mass Spectrometer	Instrument should be able to characterize the geological hazards resulting from meteor impacts, elevation changes and slope destabilization
				Energy Detection Range:	10 eV - 10 MeV	5 eV – 15 MeV		
				Elemental Composition Resolution:	Mass resolution ≤ 2 amu	1.2 amu, 1–200 amu range		
								The mission shall characterize dust hazards during excursions to protect

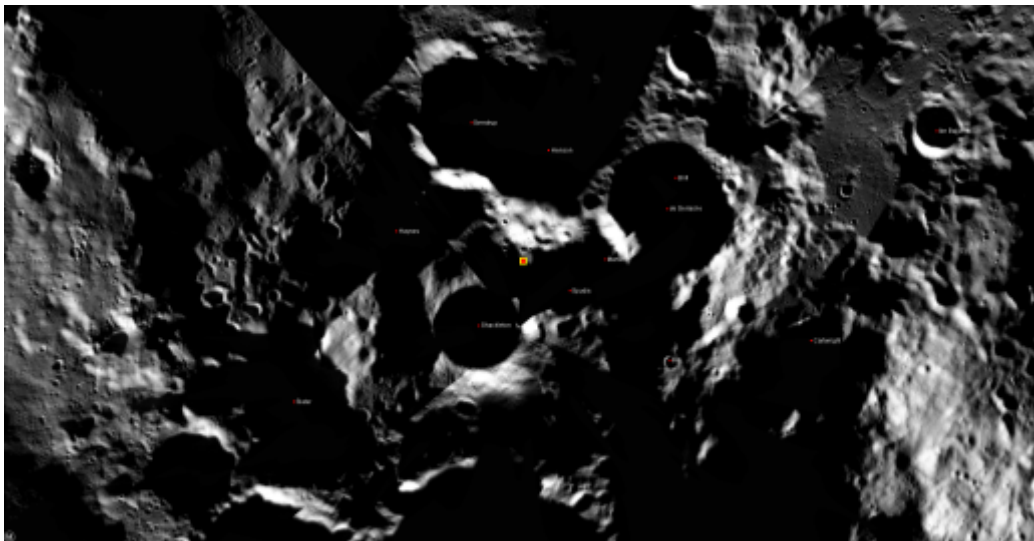
	mechanisms.			Spectral Resolution:	≤ 2.0 nm across 250–800 nm	1.5 nm / 0.2 s		platform survivability, with instrumentation measuring regolith electrostatic charge and dust mobilization.
Evaluate the capability of one or more mobility platforms to traverse lunar landscapes and avoid hazards by assessing terrain through sensors, adapting to difficult terrain changes, and safely transporting scientific samples, or instruments across the lunar surface.	Evaluate the capability of mobility platforms to accurately and reliably navigate the lunar environment while minimizing dynamic forces and vibrations acting on the craft and payload.	Identify the Total Ionizing Dose (TID), energetic particle flux, and energy spectrum in the lunar south pole radiation environment.	Measure radiation dose rates (Gy/hr) and cumulative TID (up to 130 krad) continuously across the mission's operational lifetime to determine electronics survivability.	Radiation Sensitivity:	TID Tolerance ≥ 100 krad (Si) cumulative	130 krad(Si)	Radiation Dosimeter	The system must be able to determine peak radiation zones exceeding safe operational thresholds for payload to ensure platform electronics within its total radiation dose threshold during mission lifespan
				Particle Energy Detection Range:	10 eV – 10 MeV	5 eV – 15 MeV		
				Dose Rate Sensitivity:	0.1 μ Gy/hr – 10 Gy/hr	0.05 μ Gy/hr – 15 Gy/hr		
	Determine automated routing protocols for robotic mobility platforms, connecting landing zones to points of interest to enable efficient mobility planning, ensuring hazard avoidance and the prevention of interference with	Identify vibration amplitude & frequency, linear & angular acceleration, and weight distribution.	Measure structural integrity and gyroscopic position changes (± 20 g acceleration) before and after each excursion to quantify flight-dynamics effects.	Vibration Frequency Range:	0.1 Hz - 2000 Hz	0.05 Hz – 2500 Hz	Space-Grade Gyro & IMU Suite	The system must be able to identify different gyroscopic positions to continue situational awareness during surface mobility and transition between illuminated and permanently shadowed regions
				Acceleration Range:	± 16 g	± 20 g		
				Weight sensing Accuracy:	≤ 0.10 N	0.04 N-equivalent resolution		
		Define path length, elevation gain, average and maximum slope, waypoint position, average	Record terrain suitability across energy-optimal, time-optimal, and hazard-adverse paths (≤ 2.0 m waypoint	Elevation Gain Computation Accuracy:	≤ 1.00 m	0.50 m	Flight Computer Software & Power Monitor	The mission shall autonomously plan and execute each excursion without real-time operator commanding, with onboard software

	human activities.	acceleration, energy usage, and transit time	tolerance, ≤ 0.50 m elevation accuracy, W·hr accuracy) to confirm route repeatability.	Waypoint Navigation Tolerance:	≤ 5.0 m waypoint tolerance	2.0 m		continuously updating the optimal path for energy savings and hazard avoidance
				Power Consumption Monitoring Accuracy:	≤ 0.50 W * hr	0.10 W·hr		
		Define obstacle detection ranges and accuracy, vehicle reaction times, stopping times, and potential damage variables upon hazard interaction.	Measure collision avoidance success rates (≥ 10 cm hazards detected at 0.50–15.0 m range), quantifying time (ms) and energy expended on hazard avoidance versus normal travel.	Minimum Hazard Resolution:	Detect obstacles ≥ 10 cm	Detect down to 5 cm	Time-of-Flight (ToF) Depth Camera	The system must be able to determine optimal reaction time for energy efficiency, optimizing system performance for vehicle survivability during planned excursions.
				Reaction time resolution:	≤ 33 ms	20 ms @ 30 fps		
				Obstacle detection Range:	1.00 m – 20.0 m (≥ 10 cm hazards)	0.50 m – 15.0 m		

Figure 1.2.1: Science Traceability Matrix

1.3. Summary of Mission Location

The target of this mission is the south polar region of Earth's Moon, a planetary body characterized by near-zero atmosphere, drastic temperature swings, rugged terrain, and constant exposure to meteorite impacts and radiation. The Sun's persistently low angle at the south pole, rarely exceeding a few degrees above the horizon, is what produces the region's defining feature: near-permanent illumination along elevated ridgelines paired with near-permanent darkness in adjacent crater floors, since low-angle sunlight is easily blocked by terrain that would receive full illumination at lower latitudes. This same geometry drives extreme temperature differentials between sunlit and shadowed terrain, from roughly $+120^{\circ}\text{C}$ in illuminated regions to below -233°C (40K) within Permanently Shadowed Regions, directly setting the thermal survivability requirement governing every subsystem's design.



South Pole region captured in JMars. The image shows the proximity to the craters of interest as well as nearby shadowed valleys.

Figure 1.3.1: PSRs Areas of Interest

The mission focuses on the South Pole of the Moon, specifically the Shackleton-de Gerlache Ridge and the adjacent Shackleton and de Gerlache craters. The region contains several "Peaks of Eternal Light," areas of near-continuous sunlight, making it well suited to a mission dependent on solar power. The landing site sits at approximately 89.46°S , 223.06°E , roughly 8 km from the rim of Shackleton crater and 21 km from the rim of de Gerlache crater, with multiple unnamed PSR valleys within 5 km of the site. The mission's exploration zone begins 250 m from the landing site and extends to the limits of the Scout Drone's operational range.

This location was selected as it satisfies both halves of the mission's science question: how illumination extremes, thermal transitions, and terrain constrain surface navigation and payload performance. The ridge's near-continuous sunlight allows sustained Base Station power generation, while its close proximity to multiple PSRs allow the Drone direct access to the illumination-to-darkness transition zone the mission's four science objectives require. This means generating the terrain, dust, radiation, and flight stabilization data the STM identifies.



The white areas show terrain with greater than 50% mean illumination and less than 5° mean slope within Shackleton-de-Gerlache Ridge. Site B is the chosen location. The surrounding red circle shows the 200m buffer.

Figure 1.3.2: Mission Chosen Landing Location

Customer-provided constraints require a landing site with slope under 5°, sustained Earth visibility, and sufficient solar illumination for power generation. The selected site satisfies all three: mean illumination exceeds 50% within 200 m of the site and nearly 80% within 30 m, supporting solar power as a viable primary energy source, with continuous darkness periods bounded to 70–120 hours and continuous daylight periods extending 233–263 days annually. Mean slope is under 5° within 200 m and roughly 2° within 30 m, making the ridge one of the few locations in the region combining usable sunlight with sufficiently flat terrain. These characteristics directly satisfy the site-selection constraints while independently supporting the mission's science objectives within the area of interest.

The area's polar characteristics, including the potential presence of water ice within PSRs, make it scientifically valuable beyond this mission's duration. The ridge provides a natural testing bed for route planning around the harsh topography of the surrounding craters while offering more continuous solar charging than a crater-floor or lower-latitude site would allow. The region is also a target of interest for the Artemis program, meaning the topographical and hazard baseline this mission produces has direct value for future crewed surface planning beyond the mission's own operational lifetime.

1.4. Mission Requirements

To ensure traceability between the science objectives and engineering design, a hierarchy is developed to flow from the mission goal down to mission requirements, then to system requirements, finally to subsystem requirements. The mission goal is to characterize the lunar south pole's environmental hazards and terrain conditions to establish a navigation and mobility baseline for future Artemis-era surface operations. This goal drives the mission-level requirements, captured in the customer's stated constraints on mass, volume, budget, and schedule. These mission requirements cascade into system requirements, setting the functional demands of the spacecraft, including a dual-platform architecture, autonomous surface mobility, radiation and thermal survivability, and Direct-to-Earth communication capability. These requirements outline the subsystem requirements that define specific performance parameters, from the thermal requirements ensuring survivability across the lunar south pole's temperature range, to the mechanical requirements enabling excursion into Permanently Shadowed Regions, to the power requirements sustaining operations, and to the CDH requirements governing data storage, processing, and transmission.

The mission flies as a secondary payload aboard a SpaceX Starship launch, a customer-imposed constraint that directly drives the system's mass, volume, and stowed-configuration limits below. The combined Base Station and Drone assembly shall not exceed a total launch mass of 175 kg, and shall fit within a stowed envelope of 1.5 m × 1.5 m × 1.0 m for launch integration. The system shall operate within the Lunar South Pole region at the Shackleton-de Gerlache Ridge, including terrain adjacent to and inside Permanently Shadowed Regions, and shall maintain functionality across a thermal range of -233°C (40K) to +120°C in both sunlit and PSR surface conditions.

The system shall generate and store a minimum of 1 TB of scientific/engineering data, and shall maintain Direct-to-Earth communication with Earth across distances up to 406,700 km without reliance on orbital relay satellites, consistent with the customer's clarification that

no orbital relay asset may be assumed for this mission. The system shall perform surface mobility and data collection through autonomous operations without real-time external assistance, and shall maintain partial operational capability, incorporating redundant secondary capabilities for mission-critical subsystems, in the event of a single subsystem malfunction or component failure.

The system shall not contain a Radioisotope Thermoelectric Generator or any radioisotope-derived power unit, and shall not contain more than 5 grams of radioactive material across all spacecraft systems. All electronics and enclosures shall provide radiation-hardened shielding limiting Total Ionizing Dose exposure to under 30 krad over the mission's operational lifetime, and shall use fully sealed enclosures to prevent ingress of abrasive regolith particles smaller than 100 μm .

The total cost of the mission shall not exceed \$250M across Phase A through Phase D. The integrated system assembly shall be delivered for launch integration by December 1, 2028, at Kennedy Space Center, and shall be configured and flight-ready for launch deployment by February 1, 2029, from Cape Canaveral.

Critical subsystems selected for the Base Station and Drone shall demonstrate direct flight or field heritage in a lunar or comparable deep-space environment, prioritizing components with documented history over undemonstrated technology, consistent with the mission's approach to minimizing mission schedule and technical risk.

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Req met?
SYS-001	The mission shall operate within the Lunar South Pole region at the Shackleton-de Gerlache Ridge, including areas adjacent to and inside Permanently Shadowed Regions (PSRs)	Defines the specific operational site and supports the mission task objective	STM-ESDMD-01	MECH-BS-005, MECH-DR-013	Analysis ▾	Yes ▾
SYS-002	The mission shall collect scientific data that characterizes terrain, illumination conditions, and environmental hazards relevant to lunar mobility operations	Supports the science traceability matrices objective for terrain and hazard mapping	STM-SMD-01	MECH-DR-015, CDH-003, CDH-009, CDH-BS-013, CDH-DR-019, CDH-DR-020, CDH-DR-021	Test ▾	Yes ▾
SYS-003	The mission shall utilize a robotic surface mobility platform (Lunar Drone) capable of traversing terrain features within the lunar south pole environment	Required for multi-point data collection and PSR navigation studies	STM-ESDMD-02	MECH-BS-008, MECH-BS-009, MECH-DR-014	Inspection ▾	Yes ▾
SYS-004	The mission shall acquire scientific and engineering data from multiple locations of scientific interest during the mission excursions	Supports the route planning and navigation study objectives	SYS-002	CDH-DR-020, CDH-DR-021	Test ▾	Yes ▾
SYS-005	The mission shall rely on provided launch, transit, and entry, descent, and landing (EDL) services for delivery to the lunar surface	EDL and transportation are provided externally by a third party	Customer Constraint	N/A	Inspection ▾	Yes ▾
SYS-006	The mission shall perform surface mobility and data collection tasks through autonomous operations without real-time operator intervention	Supports STM objective for safe automated mobility during PSR excursions	Customer Constraint	CDH-002, CDH-009, CDH-BS-015, CDH-DR-019	Demonstration ▾	Yes ▾
SYS-007	Critical subsystems selected for the lunar Base Station and Drone shall	Lowers engineering risk by prioritizing	Customer Constraint	PWR-BS-012, CDH-005	Analysis ▾	Yes ▾

	demonstrate direct flight or field heritage in a lunar or comparable deep-space surface environment	components with a documented operational history over undemonstrated technology				
SYS-008	The mission shall generate and store a minimum of 1 TB of scientific and engineering data to reduce operational uncertainty for future lunar missions	Quantifies data generation output supporting the mission task objective	STM-SMD-01	CDH-002, CDH-003, CDH-BS-016, CDH-DR-024	Analysis ▾	Yes ▾
SYS-009	The Base Station and Drone shall operate within a temperature range of -233C (40K) to +120C in sunlit and PSR surface conditions	Establishes baseline for external hazard protection and thermal protection range	Customer Constraint	MECH-003, MECH-BS-010, PWR-DR-014	Test ▾	Yes ▾
SYS-010	The mission shall maintain partially operational capability in the event of any single subsystem malfunction or component failure	Reduces risk of total mission loss through fault tolerance	Engineer Risk	PWR-BS-004, CDH-004, CDH-DR-025, CDH-DR-026	Analysis ▾	Yes ▾
SYS-011	The system shall incorporate redundant secondary capabilities or cold-spares for all mission-critical subsystems in case of a primary component malfunction or failure	Enhances mission success against external risks resulting in component failure	SYS-010, Engineer Risk	PWR-BS-003, PWR-DR-013, CDH-003, CDH-004	Demonstration ▾	Yes ▾
SYS-012	The mission shall maintain Direct-to-Earth (DTE) communication between Base Station and Earth across distances up to 406,700 km without reliance on orbital relay satellites	Directly supports the customer's clarification that no orbital relay asset can be assumed	Customer Constraint	CDH-001, CDH-006, CDH-BS-014	Demonstration ▾	Yes ▾
SYS-013	The system shall not contain more than 5 grams of radioactive material across all spacecraft systems	Enforces the customer's compliance limit on onboard radioactive materials	Customer Constraint	SYS-015	Inspection ▾	Yes ▾
SYS-014	The total launch mass of the combined Lunar Platform (Base Station and Drone assembly) shall not	Enforces the customer's mass limit constraint	Customer Constraint	MECH-001, MECH-BS-004, MECH-DR-012,	Inspection ▾	Yes ▾

	exceed 175 kg			CDH-010, CDH-011		
SYS-015	The system shall not contain a Radioisotope Thermoelectric Generator (RTG) or any radioisotope-derived power unit	Complies with prohibited materials constraints set by the customer	Customer Constraint	N/A	Inspection ▾	Yes ▾
SYS-016	The total cost of the mission system shall not exceed \$250M throughout the planned schedule (Phase A through Phase D)	Enforces the customer's financial budget constraint	Customer Constraint	Subsystem Budgets	Inspection ▾	Yes ▾
SYS-017	The integrated system assembly (all subsystems) shall be delivered for launch integration by December 1, 2028, at Kennedy Space Center	Directly supports the customer's system integration date constraint	Customer Constraint	Programmatics Schedule	Inspection ▾	Yes ▾
SYS-018	The system shall be configured and flight-ready for launch deployment by February 1, 2029, from Cape Canaveral	Directly supports the customer's launch date constraint	Customer Constraint	Programmatics Schedule	Inspection ▾	Yes ▾
SYS-019	The integrated Base Station and Drone assembly shall fit within an envelope of 1.5 m × 1.5 m × 1.0 m in its stowed launch configuration	Directly supports launch vehicle storage constraints	Customer Constraint	MECH-002, MECH-BS-005, MECH-DR-013	Inspection ▾	Yes ▾
SYS-020	The system shall tolerate abrasive regolith particles (<100 um) using fully sealed enclosures around all critical internal electronics and components (eg. CDH, Payload subsystem).	Prevents electrical short-circuits caused by lunar regolith accumulation	SYS-010	MECH-BS-011, MECH-DR-017, CDH-007, CDH-DR-023	Test ▾	Yes ▾
SYS-021	The system electronics and enclosures shall ensure radiation hardened shielding capable of limiting Total Ionizing Dose (TID) exposure to under 30 krad over the mission operational lifetime	Protects sensitive commercial-off-the-shelf (COTS) electronic components from lunar surface radiation	SYS-010	CDH-BS-017, CDH-DR-022	Test ▾	Yes ▾

Figure 1.4.1: Mission Requirements

1.5. Concept of Operations (ConOps)

Mission Overview

The Team 24 lunar south pole mission deploys a dual-platform system, a stationary Base Station and an autonomous Scout Drone, at a landing site on the Shackleton-de Gerlache Ridge. This location was selected for its combination of near-continuous solar illumination and close proximity to adjacent Permanently Shadowed Regions (PSRs), letting the Base Station sustain continuous power and thermal stability while the Scout Drone reaches terrain that a stationary platform cannot.

Following landing and system commissioning, the mission enters its primary operational phase: a repeating autonomous exploration cycle. During each cycle, the Scout Drone departs the Base Station and conducts an autonomous excursion into the adjacent PSRs, collecting 3D topographical data, surface thermal profiles, dust flux measurements, and radiation dose rates in direct support of the mission's four science objectives. Upon completing the excursion, the drone returns to the Base Station, docks to recharge, and transfers its accumulated science and engineering data. The Base Station stores the data and transmits it to Earth through its X-band Direct-to-Earth (DTE) communications link. Once recharging and downlink are complete, the drone departs for its next excursion.

This cycle repeats for the operational lifetime of both platforms, from the start of surface operations through End of Life (EOL), defined as the completion of downlinking a minimum of 1 TB of archived science and engineering data, or the point at which the platforms can no longer support additional excursions due to power, thermal, propulsion, or system limitations.

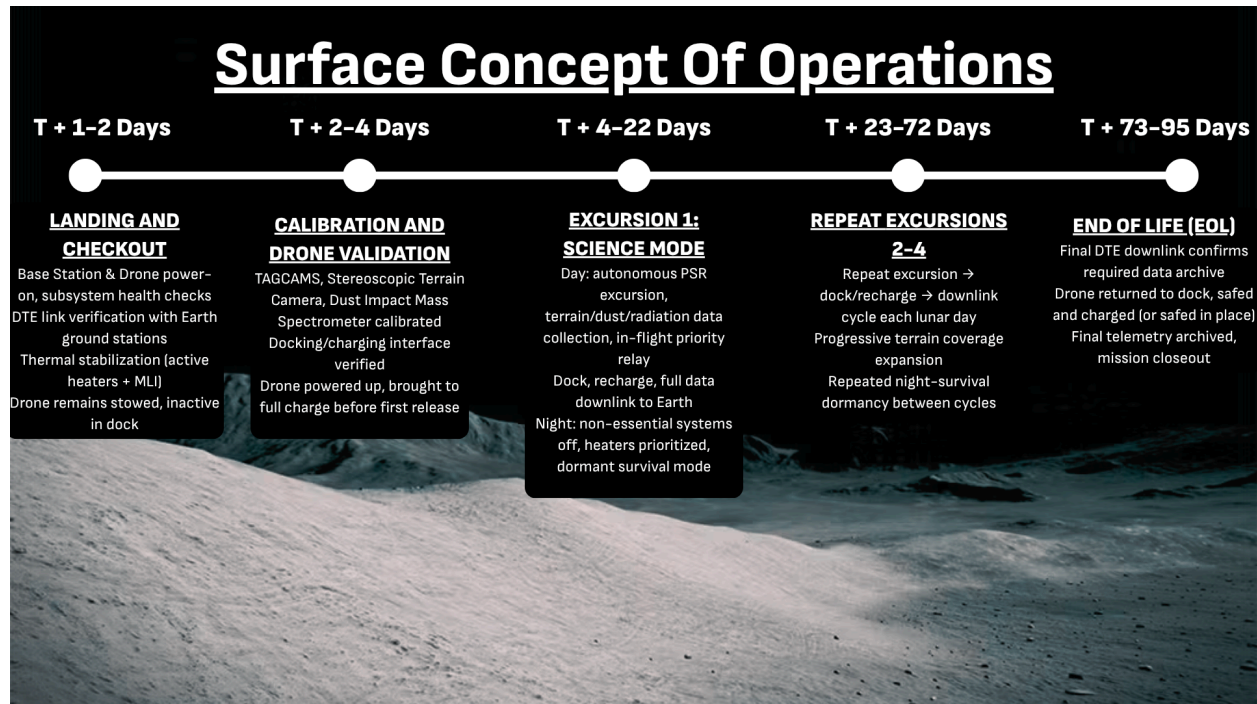


Figure 1.5.1: Concept of Operations Timeline

Mission Operations

Surface operations begin within 24 to 48 hours of landing (T+1 to 2 days), when the Base Station, positioned outside the PSR boundary to maintain solar power and thermal stability, performs a full system health check across both platforms, verifies DTE link closure with Earth ground stations, and confirms thermal stabilization of all electronics using active heaters and passive multi-layer insulation. The Scout Drone remains stowed and powered down within the Base Station docking mechanism throughout this period.

Following successful checkout, the payload's science instruments are powered on and calibrated against known reference conditions near the landing site over a 24 to 48-hour calibration period (T+2 to 4 days). Calibration is followed by a dedicated drone validation step, in which the drone is powered up, its docking and charging interface is verified, and it is brought to full charge before its first release from the Base Station (T+4 days), ensuring the excursion begins with full power margin rather than starting the mission's tightest-margin flight from a reduced state of charge.

Science operations begin with the first Scout Drone excursion into the PSR boundary region adjacent to the Shackleton-de Gerlache Ridge (T+4 days) and continue as a repeating excursion cycle for the remainder of the mission, occurring once per lunar day period. During each excursion, the Flight Computer Software and Power Monitor execute a

pre-planned autonomous flight path through the cFS-based flight management system, while the Time-of-Flight Depth Camera performs live hazard detection and trajectory planning as the drone transitions between illuminated terrain and PSR darkness, directly supporting STM Objective 4 (autonomous hazard avoidance and routing). Throughout flight, the Space-Grade Gyro and IMU Suite monitor acceleration, vibration spectra, and structural stability to maintain stable maneuvering, supporting STM Objective 3 (flight dynamics and platform stability). At the same time, the TAGCAMS Optical Navigation Suite and Stereoscopic Terrain Camera collect 3D topographical data and albedo profiles in support of STM Objective 1 (terrain and illumination effects on navigation), while the Dust Impact Mass Spectrometer and Radiation Dosimeter record electrostatic regolith charging, micrometeorite impact signatures, and radiation dose in support of STM Objective 2 (regolith and radiation hazard characterization). Throughout each excursion, the Base Station tracks drone health and status telemetry over the local link to support intervention in the event of an anomaly, and the drone relays priority science data to the Base Station in-flight ahead of its return, so a partial data set is preserved even if an excursion must be cut short.

At the end of each excursion (approximately T+14 days for the first cycle), the drone returns to dock with the Base Station over the passive spring-probe and target-contact charging interface, transfers its accumulated science and engineering data over the UHF local link for storage, and recharges before its next release. Recharge duration is bounded by the drone's ~75 Wh battery capacity and is sized to complete within the same lunar-day period as the excursion. Stored data is formatted and transmitted to Earth by the CDH and communications subsystems during scheduled X-Band DTE downlink windows; data volume per excursion is bounded by the payload suite's duty-cycled power draw and imaging rate, contributing toward the mission's 1 TB EOL data archive target across the full four-cycle excursion sequence. The Base Station's environmental sensors log dust mobilization, radiation dose, and local thermal state throughout all surface operations, concurrent with the excursion cycle.

Following the first excursion cycle, the system enters an extended dormancy period through the remainder of the first lunar night (T+15 to 22 days), during which non-essential Base Station and Drone systems are powered down and heater operation is prioritized to preserve thermal survival across the platform's -233°C to +120°C operating range. Excursions resume once power generation returns to nominal levels at the start of the next lunar day. Subsequent excursions (T+23 to 72 days, Cycles 2 through 4) repeat this excursion, dock and recharge, downlink, and dormancy pattern once per lunar day and night period, with each cycle progressively expanding terrain coverage across the

exploration zone and building the topographical, environmental hazard, and flight-performance baseline that supports future Artemis surface route planning and mobility platform development.

Mission operations conclude when the excursion cycle reaches its EOL conditions (T+73 to 95 days). The Scout Drone is commanded back to the Base Station and left docked and charged, or safed in place if it is unable to return. The Base Station performs a final DTE downlink confirming the required science and engineering data archive (approximately T+90 days), and mission closeout proceeds (approximately T+95 days).

Phase	Activity	Platform	Description
1 Checkout	System health verification	Both	Base Station and Drone undergo startup, subsystem health checks, and DTE link verification with Earth ground stations within 24–48 hours of landing.
1 Checkout	Thermal stabilization	Both	Active heaters and passive MLI bring all electronics to operating temperature range before science instrument activation.
2 Calibration	Instrument calibration	Drone	TAGCAMS Optical Navigation Suite, Stereoscopic Terrain Camera, and Dust Impact Mass Spectrometer calibrated over a 24–48-hour period.
2 Calibration	Drone validation & first charge	Both	Docking/charging interface verified; Scout Drone powered up and brought to full charge prior to first release from Base Station.
3 Ops	First PSR excursion	Drone	Scout Drone executes first autonomous excursion into the PSR boundary region, using the Flight Computer Software, Power Monitor, and Time-of-Flight Depth Camera for live hazard detection and trajectory planning.
3 Ops	Flight dynamics monitoring	Drone	Space-Grade Gyro and IMU Suite monitor acceleration, vibration spectra, and structural stability throughout each excursion to ensure stable flight and maneuverability.
3 Ops	Science & environmental data collection	Drone	TAGCAMS and Stereoscopic Terrain Camera collect 3D topographical data and albedo profiles; Dust Impact Mass Spectrometer and Radiation Dosimeter record electrostatic charging, impact signatures, and radiation dose.
3 Ops	Priority data relay (in-flight)	Both	Drone transmits priority information to the Base Station via Local Link during the excursion for DTE downlink ahead of return.
3 Ops	Drone health & status monitoring	Base Station	Base Station tracks drone telemetry and status during each excursion to support intervention in case of anomaly.

3 Ops	Dock, recharge & full data relay	Both	Drone returns to dock following each excursion, transfers remaining accumulated science and engineering data via UHF local link for storage, and recharges before its next release.
3 Ops	Science data downlink	Base Station	Base Station formats and transmits stored data via X-Band DTE to Earth at scheduled downlink windows.
3 Ops	Multi-excursion terrain mapping	Drone	Subsequent excursions expand terrain coverage across the exploration zone.
3 Ops	Environmental monitoring	Base Station	Continuous dust mobilization, radiation dose, and thermal logging throughout all surface operations phases.
3 Ops	Power conservation	Both	Non-essential systems powered down and heater operation prioritized during periods of reduced solar illumination or lower sun angle, preserving thermal survival; excursions resume at nominal power.
4 Closeout	Final science downlink	Base Station	Final DTE downlink confirming minimum required science and engineering data archived.
4 Closeout	End-of-life operations	Both	Drone commanded back to dock and safed/charged (or safed in place if unable to return); final telemetry archived; mission closeout proceeds.

Figure 1.5.3: Mission Operations Sequence

Operational Rationale

The sequential operations approach: checkout before calibration, calibration before science, and dedicated drone charge concurrent with a readiness step before first excursion. This ensures that all subsystems are verified healthy, all instruments are within their operable accuracy limits, and the drone begins science operations with full power, protecting the quality and traceability of the mission's science return. The dual-platform architecture directly supports the mission's operational objectives: the Base Station's fixed position allows continuous environmental monitoring, reliable DTE communications, and bulk data storage, while the Scout Drone's mobility enables data collection inside PSRs. The excursion cycle is hindered by two operational constraints: the Scout Drone's limited power budget per excursion, limiting maximum distance it can traverse and return with sufficient margin, and the thermal survival limits of PSR cold-soak exposure, which together set the practical duration and range of each excursion, and thus, the pace for which the terrain and hazard mapping can be built over the mission's lifetime.

1.6. Vehicle Design Summary

The Scout Drone is the mission's mobile platform, an autonomous aerial vehicle organized into seven subsystems: Payload Instrument Suite, Guidance, Navigation, and Control (GNC), Command and Data Handling (CDH), Electrical Power, Local Link Communication, Thermal

Control, and Mechanical. The drone carries a ten-instrument payload spanning optical/terrain-imaging, dust and radiation-sensing, inertial-sensing, and navigation/hazard-detection, executing autonomous excursions into and adjacent to Permanently Shadowed Regions to collect the terrain, environmental, and platform-performance data the mission's four science objectives require. Rather than carrying its own Direct-to-Earth communications, mass, and power infrastructure, the drone relies on the stationary Base Station for DTE relay, bulk data archiving, and recharging, docking after each excursion to offload science data and return to full charge. This dual-platform division of labor keeps the drone's own mass and power budget dedicated almost entirely to mobility, autonomy, and instrumentation rather than infrastructure the Base Station already provides. Major design features include a dual-channel, dual-camera-head TAGCAMS suite for optical navigation, a SCALPSS-derived stereoscopic camera paired with VIPER-derived illumination arrays for 3D terrain mapping inside PSR darkness, a dissimilar-redundant inertial suite (LN-200S and Honeywell MIMU) for flight-dynamics awareness, and a cFS-based autonomous flight computer that plans, executes, and adjusts the excursion path in real time using live hazard input from the Time-of-Flight Depth Camera.

Overall dimensions (mm)	Total mass	Total maximum power consumption
350 x 225 x 175	~50.7 kg	~205 W

Figure 1.6.1: Sole Drone Dimensions

The drone's final configuration was shaped directly by the Science Traceability Matrix: instrument placement was driven by the need for effectively 360° situational awareness rather than a single forward-facing sensor cone, with TAGCAMS' 75° × 75° field of view, the Stereoscopic Terrain Camera's independent stereo baseline, and the ToF Depth Camera's near-field detection covering navigation, terrain mapping, and close-range hazard avoidance as distinct, overlapping fields of vision that support STM Objectives 1 and 4 while giving the Stereoscopic Terrain Camera and TAGCAMS the ability to back up the ToF camera's hazard-sensing role. The most significant architectural tradeoff was pursuing a dual-platform system at all: splitting DTE communications, bulk storage, and primary power generation onto a stationary Base Station kept the drone's own budget (50.7 kg, 205 W max) concentrated on mobility and instrumentation, but introduced the docking interface and local link as new mission-critical dependencies a single-vehicle design wouldn't carry, a risk accepted because the alternative would have consumed mass and power the drone's PSR mobility and ten-instrument payload require, and because both interfaces are actively

monitored and recovered through the CDH subsystem's FDIR architecture rather than left unmitigated. The final configuration satisfies the mission's autonomy requirement, thermal and radiation survivability constraints (SYS-009, SYS-021), and the payload's mass and power allocation, while keeping the drone light and efficient enough to complete repeated excursions within a single lunar-day cycle.

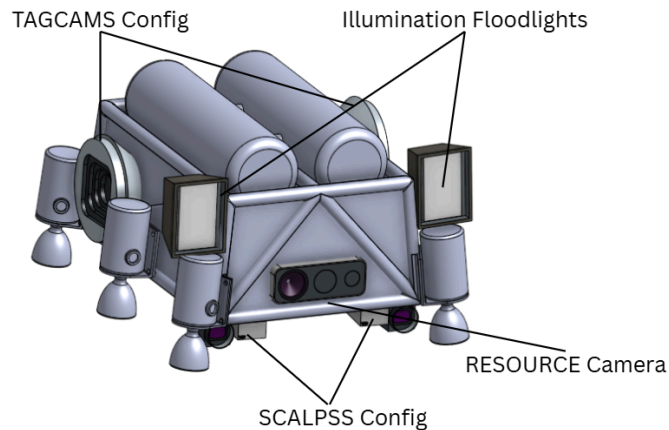


Figure 1.6.2: Autonomous Drone CAD Model

1.7. Science Instrumentation Summary

The Scout Drone carries a ten-instrument payload organized into four groups: optical/terrain-imaging, dust and radiation-sensing, inertial-sensing, and navigation/hazard-detection. Measurements include 3D topographical point clouds and albedo profiles, electrostatic dust charging and micrometeorite impact energy, cumulative radiation dose, flight vibration and acceleration, and obstacle geometry. Collected data points build the terrain, environmental, and platform-performance baseline the mission is designed to deliver.

TAGCAMS Optical Navigation Suite provides the drone's primary optical navigation and terrain imaging through a three-camera-head, dual-DVR architecture. It captures 2D intensity imagery at 0.03 m/pixel spatial resolution across a $75^\circ \times 75^\circ$ field of view, supporting STM Objective 1's evaluation of terrain and illumination effects on navigation.

Stereoscopic Terrain Camera (SCALPSS-derived) generates 3D photogrammetric point clouds and digital elevation models using a dual-CMOS stereo pair and Semi-Global Matching disparity processing. Operates simultaneously with Illumination Floodlight Arrays in PSR darkness to resolve slope, crater density, and surface texture, supporting STM Objective 1 alongside TAGCAMS.

Illumination Floodlight Arrays (VIPER-derived) are an enabling capability, providing the active LED lighting both cameras require once the drone leaves illuminated areas. Two independently switched units illuminate a path at least 5 m ahead of the drone, making the terrain-imaging suite's data collection possible inside PSRs.

Dust Impact Mass Spectrometer (SUDA-derived) is a time-of-flight, dual-polarity impact mass spectrometer that measures electrostatic charge accumulation and impactor kinetic energy from particle strikes on a biased target plate. It characterizes dust mobilization, directly supporting STM Objective 2's regolith and radiation hazard assessment.

Radiation Spectrometer/Dosimeter (LETS) uses a Timepix hybrid pixel detector to continuously measure Total Ionizing Dose, particle energy spectrum, and dose rate across sunlit and PSR terrain. Its data informs electronics survivability projections and can trigger an excursion abort or reroute, supporting STM Objective 2.

Space Grade Gyroscope-IMU (LN-200S) is a fiber-optic gyroscope and MEMS accelerometer package providing three-axis angular rate and acceleration data. Paired with the Honeywell MIMU in a dissimilar-redundancy configuration, it monitors vibration and structural stability throughout each excursion, supporting STM Objective 3.

Space Graded IMU Suite (Honeywell MIMU) provides an independent, radiation-hardened ring-laser-gyroscope attitude reference, cross-checked with the LN-200S. Its high bias stability supports flight-dynamics monitoring across the mission's excursion cycle under STM Objective 3.

Flight Processing Computer Software (cFS) executing the drone's pre-planned flight path, updating routing based on live hazard input from the ToF Depth Camera and Stereoscopic Terrain Camera. Running entirely without real-time operator commands, it directly enables STM Objective 4's autonomous hazard-avoidance requirement.

Power Monitor (INA901-SP + ADC128S102QML-SP) tracks real-time energy expenditure across the drone's power rails through a discrete current-shunt and ADC signal chain. It feeds consumption data to the Flight Computer Software for energy-optimal path selection, supporting STM Objective 4.

Time-of-Flight Depth Camera (RESOURCE Camera) is the drone's primary hazard-detection sensor, actively emitting pulsed infrared light to compute near-field depth directly. It detects obstacles as small as 10 cm across a 1.00–15.0 m range, supporting STM Objective 4's real-time obstacle avoidance.

Instrument name	Primary measurement(s)	Science objective(s) supported	Mass (kg)	Maximum power consumption
TAGCAMS Optical Navigation Suite	2D imagery, visual odometry	Objective 1	5	24
Stereoscopic Terrain Camera	3D point clouds, elevation, slope	Objective 1	6.5	54
Illumination Floodlight Arrays	Active illumination	Objective 1	1	64
Dust Impact Mass Spectrometer	Electrostatic charge, impact energy, mass spectra	Objective 2	12.57	15.3
Radiation Spectrometer/Dosimeter	TID, particle energy, dose rate	Objective 2	0.43	2.0
LN-200s IMU	Angular rate, linear acceleration	Objective 3	0.75	12.0
Honeywell MIMU	Altitude reference, acceleration	Objective 3	4.6	25.0
Flight Processing Computer Software	Autonomous routing, hazard response	Objective 4	0.00	0.00
Power Monitor	Real time energy usage	Objective 4	<0.01	<0.02
ToF Depth Camera	Near field obstacle range/size	Objective 4	0.35	6

Figure 1.7.1 Instrument Summary Table

Together, the ten-instrument payload gives the mission a complete measurement chain from terrain characterization to autonomous response. The optical/terrain-imaging suite and dust/radiation-sensing suite generate the environmental baseline the mission exists to produce, while the inertial-sensing and navigation/hazard-detection equipment ensure data can be collected safely, closing the loop between what the mission measures and how it survives measuring it. Several instruments provide meaningful cross-coverage beyond their primary role, the Dust Impact Mass Spectrometer and Radiation Spectrometer each contextualize their readings, and the Stereoscopic Terrain Camera and TAGCAMS both back up the ToF Depth Camera's hazard detection, so the science return is safely ensured.

1.8. Programmatic Summary

1.8.1. Team Introduction

Sid Gupta — Project Manager

Rising sophomore majoring in Mechanical Engineering at NC State University. He brings hands-on experience from high school FRC mechanical leadership, underwater robotics simulation for his university's Robosub team, and research on twin-screw rovers. His background coordinating cross-functional teams under FRC and Robosub deadlines directly

informs his role managing schedule, documentation consistency, and cross-subteam communication across Team 24's deliverables.

Jens Lloyd — Lead Systems Engineer

Third-year student studying Mechanical Engineering with minors in Computer Science and Electrical Engineering. He serves as the Vice President and Mechanical Lead for his university's robotics club. His cross-disciplinary coursework spanning mechanical, computer science, and electrical engineering gives him the systems-level fluency needed to reconcile interfaces between mechanical, CDH, and payload subsystems, the core responsibility of a Lead Systems Engineer.

Thomas Brady — Chief Scientist

Senior at Old Dominion University majoring in Geography and working as a student researcher and Geospatial Analyst with ODU GeoSEA. His geospatial analysis background directly supports the mission's terrain and site-selection work, including the Shackleton-de Gerlache Ridge landing site justification and Science Traceability Matrix development.

Henry Romero — Deputy Project Manager of Resources

Master's student pursuing a degree in Data Science Engineering and AI at the University of Houston, bringing past experience from the NPWEE Fall 2025 cohort. His data science background supports schedule and budget estimation work, where structured analysis of cost and lead-time data underpins the team's programmatic planning.

Jacob Boutin — Program Analyst

Junior studying Aerospace Engineering with a minor in Mechanical Engineering at the University of South Carolina, focusing on space policy and technical analysis. His combined aerospace and policy focus supports his work on schedule and budget basis-of-estimate development, where technical feasibility and programmatic constraints have to be weighed together.

Felicia Owens — Mission Assurance

Both an undergraduate Mechanical Engineering student and graduate Aerospace Engineering student. Her dual-level engineering training across both degree programs supports risk analysis and failure mode assessment work that spans mechanical hardware and broader system-level mission risk.

Jonathan Benton — Mission Assurance

Alonso Banos — Planetary Geologist

Master's Architecture student at the University of Texas at San Antonio. Experience on site analysis for environmental factors. His site-analysis background translates directly to evaluating the landing site's terrain, slope, and environmental constraints against the mission's science and safety requirements.

Citlali Alcala — Bioastronautics

Student at Texas Christian University studying Physics and Astronomy with a minor in Mathematics. Her physics and astronomy training supports payload success-criteria and instrument performance work, where the underlying physical measurement principles need to be evaluated alongside the mission's science objectives.

Massimo Sciaroni — Mechanical Engineer

Mechanical Engineering student at San Francisco State University and Aerodynamics Team Lead for his university's Formula SAE team. His aerodynamics leadership experience supports his work on the drone's mechanical design and written subsystem documentation, where structural and aerodynamic tradeoffs directly shape the vehicle's excursion performance.

Kavindu Wijesekara — Mechanical Engineer

Mechanical Engineering student at College of Staten Island. His mechanical engineering training supports his lead role developing the Base Station CAD design, translating subsystem requirements into a physically realized structural model.

Neha Bhaskarabhotla — Electrical Engineer

Electrical Engineering student at San Jose State University with experience in autonomous braking systems research, Formula SAE electronics design, and microcontroller hardware. Her electronics and microcontroller experience directly supports Power subsystem recovery, redundancy, and procurement planning, where component-level electrical reliability is the central design concern.

Marcos Olivas — Electrical Engineer

Electrical Engineering student at the University of Texas at El Paso. His electrical engineering background supports Power subsystem requirements work, and his contributions extending into the Mission Location summary reflect the same cross-disciplinary flexibility the team has relied on throughout the program.

Yahir Guerrero — Thermal Engineer

Aerospace Engineering graduate student entering his Master's program at San Jose State

University. His aerospace-level graduate training supports Thermal Control subsystem requirements development, where spacecraft-specific thermal environments demand more than general mechanical thermal analysis.

Aliyah Williams — Thermal Engineer

Mechanical Engineering student at the University of Central Florida with an interest in aerospace thermal systems. Their stated interest in aerospace thermal systems aligns directly with her role developing the Thermal Control subsystem's requirements and recovery/redundancy planning.

Alfonso Salas — Command & Data Handling Engineer

Computer Engineering Technology Junior at the University of Houston. He has prior research experience at Sugarland testing LiDAR mapping and routing, and partook in an internship at Thinkneuro LLC creating an LLM ASL pipeline. Additionally, he is currently serving as the Projects Co-Chair at SHPE-UH, planning workshops and events to teach students of varying skill levels. As a CDH engineer, utilized interdisciplinary planning, prior CAD software knowledge (Onshape, KiCAD) to communication and payload assembly.

1.8.2. Team Management Overview

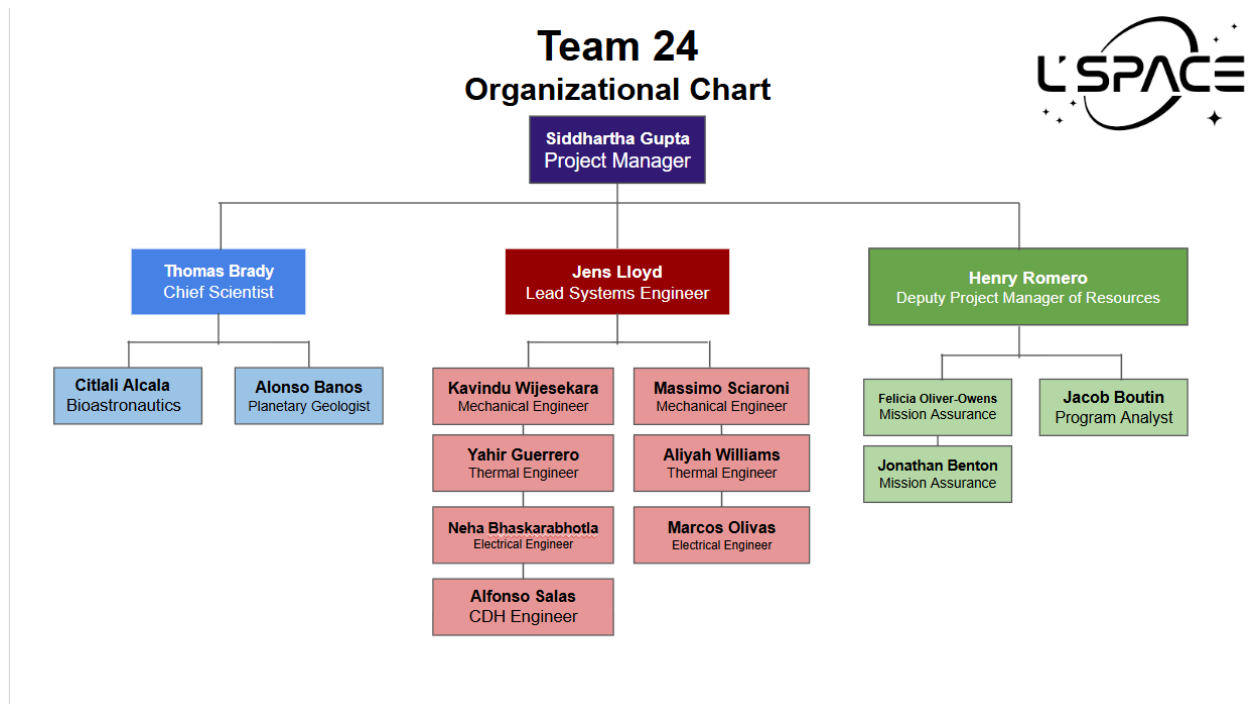


Figure 1.3.1: MCA Team Org Chart

Each deliverable is split into broad sections assigned to subteams by the Project Manager, with subteam leads then assigning individual tasks to members or small groups. Work is tracked in a shared Google Sheet, which is reviewed daily, with each task marked as “Not Started”, “In Progress”, or “Complete”, reinforcing mutual accountability and a clear completion pace. This tracking approach has evolved deliverable by deliverable to fix real problems as they surfaced: elective sections at MCR left workload unevenly distributed, prompting direct section assignment by SRR, followed by a soft deadline system at MDR, and an iterative review cycle at PDR to give members more time on each sections and catch gaps before becoming last-minute additions. Coordination across subteams runs through the Chief Scientist, Lead Systems Engineer, and Deputy PM of Resources directly. Team 24 instated a weekly Tuesday general meeting, Thursday subteam breakouts, and optional Discord voice sessions for members to work alongside one another.

Member	Role	Time Commitment	Key Responsibility
Siddhartha Gupta	Project Manager	8-10 hrs/week	Coordinates communication between leads and members; revisions across deliverables
Thomas Brady	Chief Scientist	6-8 hrs/week	Balances workload across remaining Science subteam
Jens Lloyd	Lead Systems Engineer	6-8 hrs/week	Propulsion system, Drone CAD Design, and overseeing workload across Engineering team
Henry Romero	Deputy PM of Resources	6-8 hrs/week	Oversees Programmatic workload; handling mission schedule
Citlali Alcala	Bioastronautics	4-6 hrs/week	Handling science objectives alongside instrumentation calibration, and STM
Alonso Banos	Planetary Geologist	4-6 hrs/week	Handling science objectives alongside payload instrumentation, and STM
Kavindu Wijesekara	Mechanical Engineer	6-8 hrs/week	Base Station CAD design; written mechanical sections
Massimo Sciaroni	Mechanical Engineer	6-8 hrs/week	CAD support; written mechanical sections
Yahir Guerrero	Thermal Engineer	4-6 hrs/week	Handling Thermal Subsection content (Overview, components, validation, requirements)
Aaliyah Williams	Thermal Engineer	4-6 hrs/week	Handling Thermal Subsection content (Overview, components, validation, requirements)

Neha Bhaskarabotla	Electrical Engineer	6-8 hrs/week	Handling Electrical Subsection content (Overview, components, validation, requirements)
Marcos Olivas	Electrical Engineer	4-6 hrs/week	Handling Electrical Subsection content (Overview, components, validation, requirements)
Alfonso Salas	CDH Engineer	10-12 hrs/week	Handling CDH and Payload Subsection content + revisions for STM and ConOps
Felicia Oliver-Owens	Mission Assurance	4-6 hrs/week	Handling FMEA and Risk Section
Jonathan Benton	Mission Assurance	4-6 hrs/week	Assisting with Mission Schedule and primarily working with Mission Outreach
Jacob Boutin	Program Analyst	6-8 hrs/week	Primary handler of Mission/System Budget

Figure 1.8.2.1: MCA Team Commitment and Responsibilities

Decision-making scales with a task's mission impact and urgency. Short, non-mission-critical decisions, such as naming or scheduling choices, are made autocratically by whichever lead owns that area (Lead Systems Engineer, Chief Scientist, Deputy PM, or PM). Decisions the team can weigh in on but a team lead finalizes, such as meeting logistics, follow a consultative model. Larger, mission specific decisions, such as deliverable content or design direction, are handled democratically: the whole team gives input and the direction is decided by majority vote. When time does not allow for full team input, the PM or affected lead makes the call directly and reports the decision and rationale to the team at the next meeting or via Discord announcement.

Peer-to-peer disagreements are resolved by the relevant subteam lead(s) until unanimous agreement is reached. Disagreements between team leads, or between a lead and a subordinate, are resolved through a team-wide survey ensuring majority agreement across all parties; if this does not resolve the conflict, it is brought to the team's PA mentor for an outside perspective. This process is applied consistently to both technical and non-technical disagreements.

The team holds the expertise needed across planetary science, bioastronautics, systems engineering, and programmatics, and closes gaps primarily through collaborative assistance. For example, Electrical Engineer Marcos Olivas contributed to the Mission Location summary and CDH Engineer Alfonso Salas supported ConOps and STM development outside their primary roles, supplemented by skill modules and research where gaps remain. Following the loss of both dedicated Mission Assurance specialists, this same cross-scope model was applied at the subteam level: risk identification and mitigation responsibility was distributed across all members as they complete their assigned tasks,

with the Deputy PM of Resources overseeing high-level risk tracking rather than a single dedicated role.

1.8.3. Schedule Overview

The mission schedule spans NASA Life Cycle Phases C through F, from final design and fabrication through system integration, launch, lunar surface operations, and mission closeout. Phase-level Gantt charts sequence subsystem design, procurement, assembly, integration, verification and validation, and environmental testing consistent with NPR 7120.5, ensuring the design matures through each major review before the mission commits to the next phase. This structure ties the schedule directly to the mission lifecycle, from the Critical Design Review that closes Phase C to the final data archival that closes Phase F.

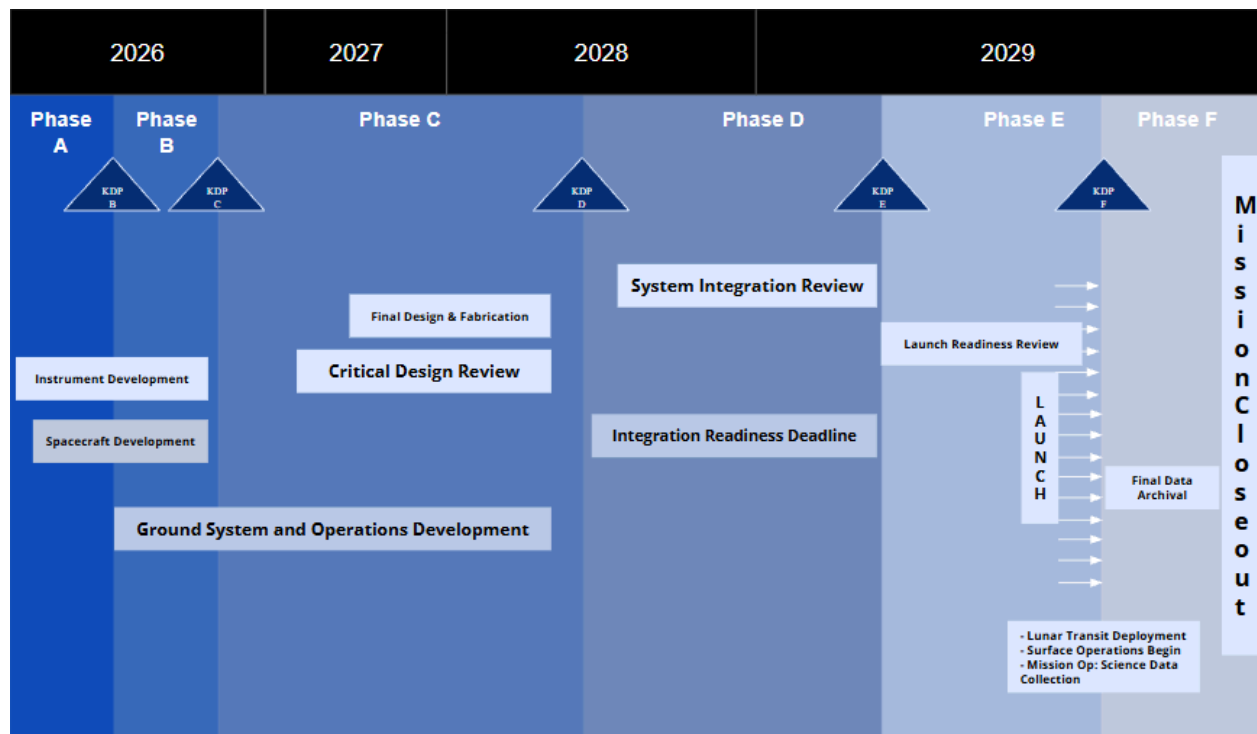


Figure 1.8.3.1: Team 24 Milestone Chart

Phase C (November 2026 – September 2027) carries both platforms from preliminary to final design, procuring long-lead components such as CDH's PolarFire and Xiphos Q8 OBCs and the Payload subsystem's heritage instrumentation, fabricating the Drone and Base Station structures, developing and porting flight software, and completing subsystem-level verification ahead of the CDR and KDP-D. Phase D (October 2027 – December 2028) then carries the design through assembly, integration, and test: Base Station and Drone system

integration, interface verification, thermal-vacuum and vibration environmental testing, EMI/EMC testing, system-level functional testing, flight software validation, and a mission operations rehearsal, culminating in the System Integration Review and shipment to Kennedy Space Center ahead of the December 2028 Integration Readiness Deadline. Phase E begins with the Launch Readiness Review and February 2029 launch, followed by lunar transit, landing, and surface commissioning, after which the Drone begins its excursion cycle to collect the terrain, illumination, dust, and radiation data supporting the mission's science objectives; Phase F then closes the mission with a final DTE downlink, data archival, and a closeout report documenting lessons learned.

The schedule was developed using an analogous-mission approach anchored to JPL's Ingenuity Mars Helicopter, with activity durations estimated from procurement lead times, subsystem complexity, and verification requirements for each phase rather than a single mission-wide multiplier. Ingenuity's one-month operational lifetime was not applied directly, since this mission's dual-platform architecture and 95-day, four-excursion sequence differ from Ingenuity's single-platform, short-flight profile, but Ingenuity's dusty, wide-diurnal-swing environment set a conservative lower bound on component survivability that this schedule builds margin around. The Payload subsystem's heritage instrumentation, the Drone's docking mechanism, and its regolith collection assembly carry the mission's longest procurement lead times and form the critical path from Phase C procurement through Phase D integration; scheduling SIR completion roughly three months ahead of the Integration Readiness Deadline preserves contingency time to invoke a backup supplier without threatening that date. The schedule will be refined at PDR and CDR as signed procurement contracts confirm actual vendor lead times and subsystem test durations are validated during TVAC and vibration testing, at which point the critical path will be reassessed against any backup-supplier substitutions used during Phase C or D.

1.8.4. Cost Overview

The lunar south pole dual platform mission carries a preliminary total estimated mission cost of \$240,796,376, developed using parametric Cost Estimating Relationships (CERs) sourced from The New SMAD and the NASA Instrument Cost Model (NICM), applied to the subsystem mass and power values baselined during the System Requirements Review. This total falls within the customer imposed budget cap of \$250,000,000 (SYS-017), leaving an available cost margin of \$9,203,624 to absorb cost growth as the design matures through Critical Design Review (CDR). The mission's dual platform architecture, which requires independently qualified Mechanical, Power, CDH, and Thermal subsystems for both the Base Station and Scout Drone, makes Direct Costs the dominant contributor to overall

mission cost at \$208,535,445, with Personnel Costs the second largest category at \$21,726,290. This cost distribution reflects the specialized radiation tolerant and cryogenically rated components required for PSR operations.

Budget Element	Value
Customer Budget Cap	\$250,000,000
Total Estimated Mission Cost	\$240,796,376
Personnel Costs	\$21,726,290
Travel Costs	\$158,562
Outreach Costs	\$30,693
Direct Costs (subsystem hardware, spacecraft margin, and facilities)	\$208,535,445
Facilities and Administrative (F&A) Costs	\$10,345,386
Available Cost Margin	\$9,203,624
Cost Margin Applied	30% of all Personnel Costs, 30% of all Travel Costs, 50% of all Direct Costs, 30% of all Outreach Costs
Mission Phases Covered	Phases C through F
Excluded Costs	Launch Vehicle, EDL System, Transit Vehicle

Figure 1.8.4.1: Budget Overview

The mission's budget is dominated by Direct Costs, which account for roughly 87 percent of the total estimated mission cost and reflect the hardware, spacecraft margin, and facility expenses associated with building two independently qualified vehicles rather than one. Personnel Costs form the second largest category at \$21,726,290, driven primarily by sustained Engineering and Science staffing through Phases C and D, before decreasing as the mission transitions into Phase E operations. Facilities and Administrative costs of \$10,345,386 support the office space, materials, and equipment associated with mission development, while Outreach and Travel costs are comparatively minor contributors, together representing well under one percent of the total mission cost. Detailed subsystem level cost drivers, CER methodology, and estimating assumptions for each category are

presented in the Cost Basis of Estimate and Direct Costs sections.

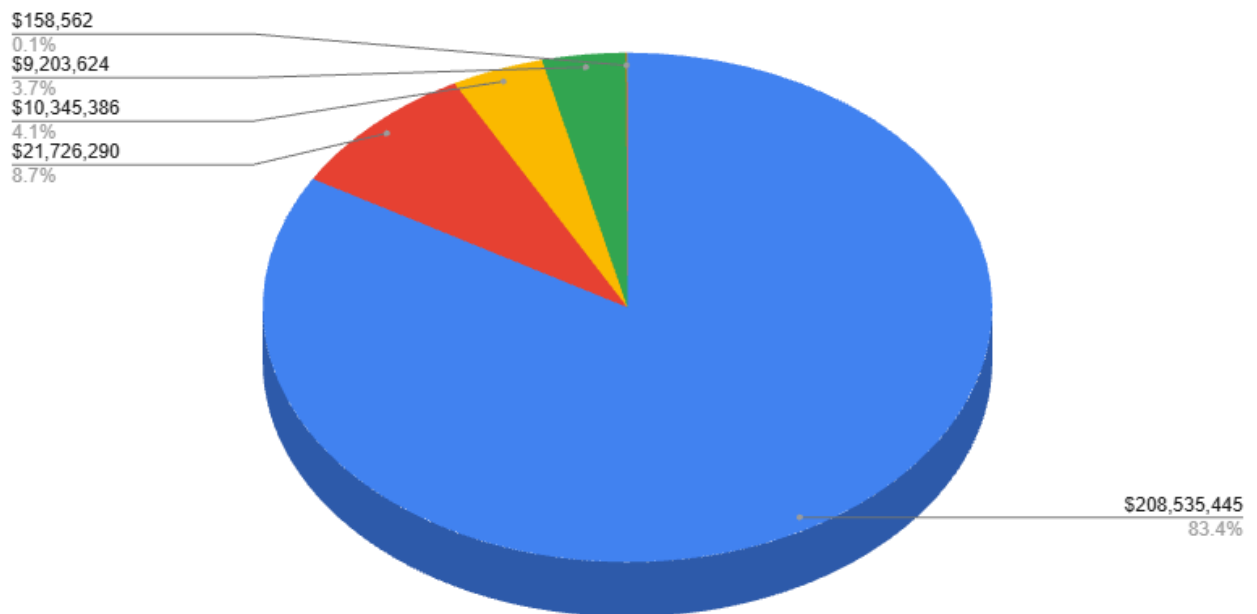


Figure 1.8.4.1: Team 24 Budget Visualization Pie Chart

2. Overall Vehicle and System Design

2.1. Spacecraft Overview

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Req met?
SYS-001	The mission shall operate within the Lunar South Pole region at the Shackleton-de Gerlache Ridge, including areas adjacent to and inside Permanently Shadowed Regions (PSRs)	Defines the specific operational site and supports the mission task objective	STM-ESD MD-01	MECH-BS-005, MECH-DR-013	Analysis ▾	Yes ▾
SYS-002	The mission shall collect scientific data that characterizes terrain, illumination conditions, and environmental hazards relevant to lunar mobility operations	Supports the science traceability matrices objective for terrain and hazard mapping	STM-SMD-01	MECH-DR-015, CDH-003, CDH-009, CDH-BS-013, CDH-DR-019, CDH-DR-020, CDH-DR-021	Test ▾	Yes ▾
SYS-003	The mission shall utilize a robotic surface mobility platform (Lunar Drone) capable of traversing terrain features within the lunar south pole environment	Required for multi-point data collection and PSR navigation studies	STM-ESD MD-02	MECH-BS-008, MECH-BS-009, MECH-DR-014	Inspecti... ▾	Yes ▾
SYS-004	The mission shall acquire scientific and engineering data from multiple locations of scientific interest during the mission excursions	Supports the route planning and navigation study objectives	SYS-002	CDH-DR-020, CDH-DR-021	Test ▾	Yes ▾
SYS-006	The mission shall perform surface mobility and data collection tasks through autonomous operations without real-time operator intervention	Supports STM objective for safe automated mobility during PSR excursions	Customer Constraint	CDH-002, CDH-009, CDH-BS-015, CDH-DR-019	Demon... ▾	Yes ▾
MECH-001	The total missions' mechanical subsystems shall not exceed 16.7 kg of the allocated mission mass.	Ensures that all mechanical subsystems will remain within the mission mass that was provided by the customer.	SYS-015	MECH-BS-004, MECH-DR-012	Inspecti... ▾	Yes ▾
MECH-002	The total missions' mechanical subsystems shall not exceed 1.5m x 1.5m x 1m of the allocated mission stowed volume	Ensures that all mechanical subsystems will remain within the mission volume allocation that was provided by the customer.	SYS-020	MECH-BS-005, MECH-DR-013	Inspecti... ▾	Yes ▾
MECH-DR-016	The drone structure shall withstand launch, landing, deployment, and operational loads	Ensures that the drone will survive all of the major mechanical loads that can occur.	SYS-010		Analysis ▾	Yes ▾

	without structural failure, and have a factor of safety greater than 1.5.					
MECH-DR-024	The drone docking assembly shall provide alignment features that allow for repeatable docking within $\pm 10\text{mm}$ positional and angular tolerance.	Ensures reliable docking and will minimize the chance of mechanical interference during recovery.	SYS-004, MECH-BS-009		Inspecti...	Yes
CDH-001	The CDH Subsystem shall transmit and receive data directly to/from Earth ground stations via X-band DTE communication (7.145–7.235 GHz uplink / 8.400–8.500 GHz downlink), without reliance on relay satellites.	Per stakeholders, all communication must be Direct-to-Earth (DTE), and must assume that relay satellites should not be used.	SYS-012	CDH-BS-014, CDH-BS-015	Test	Yes
CDH-002	The CDH subsystem shall receive, collect, and retain data onboard using storage providing at least 2,000 MB/s sustained write speeds.	Required to support high-throughput science-data collection and buffering	SYS-006, SYS-008	CDH-008, CDH-012, CDH-BS-016, CDH-DR-024	Demon...	Yes
CDH-004	The CDH subsystem shall incorporate autonomous FDIR loops for detecting and mitigating subsystem malfunctions, acting within ≤ 5 minutes of fault onset.	Directly supports backup/redundancy capabilities	SYS-010, SYS-011	CDH-DR-025, CDH-DR-026	Demon...	Yes
CDH-005	The CDH subassemblies shall demonstrate direct flight or field heritage in a comparable space, deep space, or lunar surface environment	Lowers engineering risk by prioritizing components with a documented operational history over undemonstrated technology	SYS-007		Inspecti...	Yes
PAY-SMD-1.1	The payload subsystem shall include an optical/terrain-imaging capability sufficient to characterize elevation, slope, surface texture, albedo, and crater density across illuminated and PSR terrain.	Directly supports STM Objective 1 (evaluating terrain/illumination gradient impact on TRN systems).	SYS-002, SYS-004	PAY-SMD-1.1.4, PAY-SMD-1.1.5,	Demon...	Yes
THERM-007	Materials for thermal systems should meet 173°C to $+127^{\circ}\text{C}$ thermal standards based on where passive and active cooling systems are needed for both Drone and Base stations.	Ensures the propulsion subsystem can generate reliable thrust and maintain performance during deployment, flight, landing, and lunar surface operations despite extreme lunar location temperature conditions.	SYS-004	THERM-BS-003, THERM-BS-004, THERM-DR-006.1	Demon...	Yes
THERM-BS-002	The base station thermal subsystem shall provide thermal isolation between heat generating components and temperature	This will prevent electronics, batteries, or science payloads from being damaged due to unwanted	THERM-BS-004	THERM-007	Inspecti...	Yes

	sensitive payloads.					
PWR-BS-002	The base station charging interface shall protect against overcurrent, overvoltage, undervoltage, and other battery abuse conditions	Ensures the scout drone and the charging interface are protected from adverse battery conditions and potential faults	PWR-BS-004	PWR-DR-019	Test ▾	Yes ▾
PWR-BS-005	The base station power subsystem shall distribute regulated electrical power to all subsystems	Ensures all base station subsystems receive stable and usable electrical power	PWR-BS-001	PWR-BS-006, PWR-BS-007, PWR-BS-008, PWR-BS-009, PWR-BS-010, PWR-BS-011	Test ▾	Yes ▾
PWR-BS-004	The base station shall support multiple power modes, including nominal, charging, low-power, and safe modes	Ensures complete functionality of the base station throughout different phases of the mission	SYS-011	PWR-BS-002	Demon... ▾	Yes ▾
PAY-SMD-1.2	The payload subsystem shall include a dust/impact-sensing capability sufficient to measure electrostatic charge accumulation and impact kinetic energy.	Directly supports STM Objective 2 (characterizing electrostatic regolith mobilization and micrometeorite impact environment).	SYS-002	PAY-SMD-1.2.1, PAY-SMD-1.2.2	Demon... ▾	Yes ▾
PAY-HSMD-2.1	The payload subsystem shall include an inertial-sensing capability sufficient to measure vibration amplitude/frequency and linear/angular acceleration during drone flight.	Directly supports STM Objective 3 (evaluating mobility platform navigation reliability while minimizing dynamic forces/vibration).	SYS-003	PAY-HSMD-2.1.2	Demon... ▾	Yes ▾
PAY-HSMD-2.2	The payload subsystem shall include a navigation-support capability sufficient to record elevation gain and power consumption during autonomous excursions.	Directly supports STM Objective 4 (establishing automated routing protocols connecting landing zones to points of interest).	SYS-004, SYS-006	PAY-HSMD-2.2.1, PAY-HSMD-2.2.2	Inspecti... ▾	Yes ▾
PAY-HSMD-2.3	The payload subsystem shall include a hazard-detection capability sufficient to identify obstacle range and size during flight.	Directly supports STM Objective 4 (hazard avoidance and prevention of interference with human activities).	SYS-004, SYS-006, SYS-010	PAY-HSMD-2.3.1	Demon... ▾	Yes ▾

Figure 2.1.1: Major Systems Requirement Table

The spacecraft consists of a dual-platform architecture: a stationary Base Station serving as the mission's power, communications, environmental sensing, and data-handling hub, and an autonomous Scout Drone providing mobility into terrain the Base Station cannot reach. The Base Station is organized into six subsystems, Direct-to-Earth (DTE) Communication, Command and Data Handling (CDH), Local Link Communication, Electrical Power, Mechanical, and Thermal Control, and is positioned outside the PSR boundary to maintain continuous solar illumination, providing power generation and storage, command authority, bulk data archiving, and the sole DTE relay path to Earth. The Drone is organized into seven subsystems, Payload Instrument Suite, Guidance, Navigation, and Control (GNC), CDH, Electrical Power, Local Link Communication, Thermal Control, and Mechanical, and executes autonomous excursions into and adjacent to PSRs, collecting science and engineering data, storing it onboard, and relaying it to the Base Station upon return.

This architecture directly supports the mission's operational concept: the Base Station's fixed position and continuous solar exposure sustain the power generation, DTE communications, and bulk storage the mission's four science objectives depend on, while the Drone's mobility is what makes PSR-interior data collection possible at all. The two platforms remain electrically independent during each drone flight but share a common CDH and power architecture, drawing on the same OBC, storage, and power-mode design pattern to reduce development and integration risk. The combined assembly is designed to fit within the mission's 175 kg mass allocation and 1.5 m x 1.5 m x 1.0 m stowed launch envelope, with each subsystem's mass and power budget flowing down from those top-level constraints.

Mechanical Subsystem

The Mechanical subsystem provides the structural, deployment, and protective hardware supporting every other subsystem on both platforms. The Base Station Structural Frame is the primary load-bearing structure, providing mounting locations for the power system, communication hardware, onboard computer enclosure, science payloads, and drone docking mechanism, built from Aluminum 6061-T6 for its balance of strength, manufacturability, corrosion resistance, and low mass. The Drone Structural Frame similarly supports the propulsion system, avionics, landing gear, payload, and communication hardware, using Aluminum 7075-T6 for its higher strength-to-weight ratio suited to a mass-constrained flight vehicle. The Drone Docking and Deployment Assembly mechanically secures the Drone during launch, transit, and surface operations while providing alignment features and a controlled release mechanism for interference-free deployment and repeatable post-excursion docking. The Onboard Computer Enclosure

houses and structurally protects the CDH electronics from lunar dust and environmental extremes using a sealed aluminum housing, while the Dust Protection System applies the same fully sealed enclosure approach around other critical mechanical and electronic interfaces. The Regolith Collection Assembly uses a passive funnel collection system, selected for reliability, low mass, and minimal moving parts, and the Solar Array Mounting Assembly structurally supports and transfers deployment and operational loads from the solar array into the Base Station structure. All seven mechanical subassemblies currently carry a TRL of 3, reflecting completed trade studies and preliminary concept selection; detailed CAD development, structural analysis, and physical testing remain ahead to mature each design.

Power Subsystem

The Electrical Power Subsystem (EPS) provides all power generation, storage, distribution, and fault protection functions for both platforms, including power modes designed to keep vital systems online during periods of low power generation. The Base Station and Drone each carry their own EPS unit built from similar components, but the two remain electrically independent during each Drone flight. Every subsystem depends on the EPS to function: across the mission's operational phases, the EPS ensures the Drone is brought to full charge before release, that all instruments operate nominally, and that every subsystem's power draw is met without exhausting the power supply. The subsystem was designed to sustain full power generation, distribution, and storage capability throughout the mission's duration, including a full lunar night cycle should the mission's lifespan align with one at the landing site, with degrading efficiency in any single subassembly not expected to pose a major concern across the mission's lifespan. Power generation is provided by a DHV deployable solar array, storage by EaglePicher NPD-002271 Li-Ion batteries sized separately for each platform, and regulation/distribution by the Ibeos Modular EPS, with the Drone recharging through a passive spring-probe and target-contact docking interface during each post-excursion dock.

Command & Data Handling (CDH)

The CDH subsystem manages all intra- and inter-platform communications, software execution, fault management, and data telemetry, split into Base Station (BS) and Drone (DR) requirement sets to reflect each platform's distinct power and mass constraints. The subsystem integrates five subassemblies, Flight Computer (OBCs), Data Storage, Internal Communication, Telecommunication, and Software Architecture, supporting the mission's operational phases from health verification during checkout through hazard-avoidance data support during PSR excursions and low-power monitoring during lunar night. The

Drone runs dual PolarFire SoCs in an Active/Cold-Spare configuration for autonomous flight control, while the Base Station runs a Xiphos Q8 Zynq UltraScale+ for telemetry formatting and DTE traffic, backed by a low-power Vorago VA41630 that continuously monitors baseline health while the Xiphos idles. Both platforms use Space-Graded NVMe SSDs exceeding 3,000 MB/s sustained write speed, with the Base Station pooling storage into a redundant 8 TB RAID 1 array ahead of downlink and the Drone falling back to a memory card if its primary drive fails. SpaceWire (200 Mbps) serves as the internal instrument and telemetry bus on both platforms, UHF over the CCSDS Proximity-1 protocol carries the Base Station-Drone local link, and an Iris V2 X-Band transponder, JHU/APL Frontier SSPA, and high-gain patch antenna provide the Base Station's sole DTE path to Earth. Flight software runs NASA's cFS atop RTEMS for fault isolation and multi-core support.

Thermal Subsystem

The Thermal Control Subsystem (TCS) provides all internal and external heating and cooling for both platforms, keeping CDH, Electrical, Mechanical, and Payload hardware within operating temperature range despite the lunar south pole's swing from +120°C in sunlit terrain to -233°C within PSRs. Because the Base Station and Drone face different exposure profiles, each carries its own TCS architecture, split between passive and active components. Passive TCS includes Aluminized Kapton multilayer insulation (MLI), selected for its radiation and UV resistance, thermal performance, and durability, layered across both platforms' exterior and select interior surfaces; Pyrolytic Graphite Sheet (PGS) thermal straps, which link heat-generating electronics to other TCS components to regulate temperature while adding minimal mass; and a deployable panel radiator that passively rejects excess heat. Active TCS includes Lockheed Martin Space MICRO1-1 cryocoolers for cooling temperature-sensitive cameras and sensors, Minco HK6905 Polyimide Thermofoil heaters to counter cold-soak exposure during PSR excursions and lunar night, and Tark Thermal Solutions thermoelectric coolers for localized heating or cooling as needed.

Science Instrumentation / Payload

The Payload subsystem collects all scientific and engineering data supporting the mission's four science objectives: characterizing terrain and illumination effects on navigation, characterizing regolith and radiation hazards, monitoring platform flight dynamics, and enabling autonomous hazard avoidance. Because the core science objectives require measurement inside and adjacent to PSRs, all payload instrumentation is carried aboard the Scout Drone. The subsystem integrates ten instruments across four functional groupings, each tied to a specific objective in the Science Traceability Matrix. The optical/terrain-imaging suite, TAGCAMS, the SCALPSS-derived Stereoscopic Terrain Camera,

and VIPER-derived illumination floodlight arrays, generates the 3D topographical point clouds, slope maps, and surface albedo profiles used across illuminated and PSR terrain. The dust and radiation-sensing suite, the LDEX-derived Dust Impact Mass Spectrometer and LETS-heritage Radiation Spectrometer, characterizes dust mobilization, micrometeorite impact behavior, and long-term electronics survivability. The inertial-sensing suite, the LN-200S Gyro-IMU and Honeywell MIMU, monitors vibration, acceleration, and structural stability to maintain stable maneuvering between illuminated terrain and PSR darkness. The navigation/hazard-detection suite, the Flight Processing Computer Software, Power Monitor, and RESOURCE ToF Depth Camera, executes and updates the Drone's autonomous flight path, tracks energy expenditure, and supplies real-time obstacle detection, enabling hazard avoidance without operator intervention where communication delay makes real-time commanding impractical.

Subsystem	Mass (kg)	Dimensions (m)	Maximum Power Draw (W)
Mechanical	~16.7kg	1.5 x 1.5 x 1	~10W
Power	~54.3 kg	0.4 x 0.4 x 0.55	40W
CDH	~3.330 kg	6,128.61 cm ³	122.6 W
Thermal	~29 kg		~ 85W
Payload	~31.21 kg	32,450.73 cm ³	~202.32 W
Total Spacecraft	121.10 kg	1.5 x 1.5 x 1.0	459.92 W

Figure 2.1.0.1: Total Spacecraft Dimensions

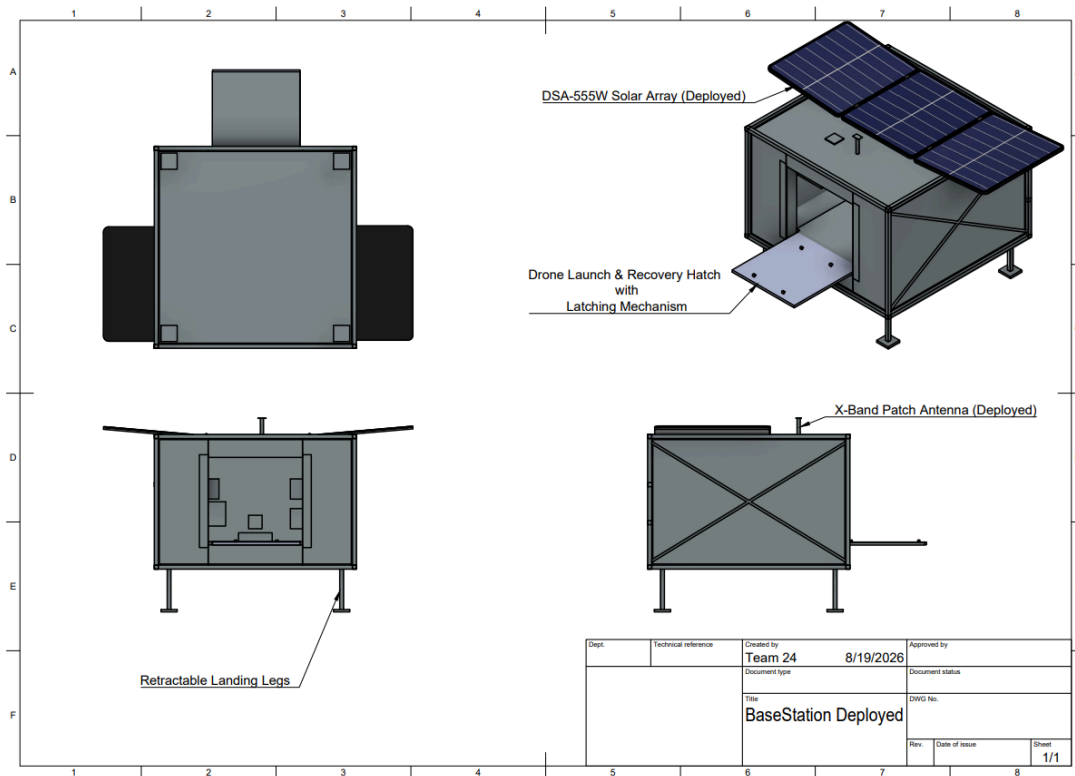


Figure 2.1.0.2: Base Station Deployed CAD

2.1.1. Mechanical Subsystem Overview

Mechanical Subsystem Overview

The mechanical subsystem provides the structural foundation and physical interfaces that are required for the successful operation of the lunar base station and drone throughout all mission phases. The subsystem is responsible for the supporting of onboard subsystems, maintaining structural integrity during launch, landing, deployment, and surface operations, and finally protecting hardware from hard lunar conditions. The mechanical subsystem also provides the necessary mounting features and interfaces for the power, communication, CDH, propulsion, and science subsystems while maintaining compatibility with the overall spacecraft design.

The preliminary mechanical design consists of a base station, and drone. The base station is the primary structural platform for the mission. The base station supports the electrical systems, science payloads, onboard computer enclosure and solar array assembly, and the drone docking interface. The drone provides mobility for mission operations and is structurally designed to support its propulsion system, avionics, payload, and landing system while trying to minimize the overall mass.

The mechanical subsystem was designed to use requirements that were supported by multiple trade studies. These studies were evaluated on structural materials, landing gear configurations, dust mitigation strategies, onboard computer enclosure concepts,

regolith collection methods, and fastener selection to help properly identify the best preliminary concepts that will satisfy the mission requirements.

Mechanical Capabilities

The Mechanical Subsystem provides the primary structural support and physical interfaces that are required for successful mission operations during all phases of the mission. It maintains the structural integrity of both the lunar base station and drone during all launch, landing deployment, flight and surface operations. During this it needs to support the integration of the power, CDH, communications, thermal, propulsion, and science payload substeams. The subsystem that enables the deployment, docking, and recovery of the drone provides mounting interface and structural support for mission hardware. It accommodates the regolith collection system and onboard computer enclosure as well. In addition, it also enhances the environmental survivability by protecting the critical components from lunar dust, extreme temperatures, and operational loading through the sealed enclosure, dust mitigation features, and robust structural design.

Subassembly	Mass (kg)	Dimensions (m)	Maximum Power Draw (W)	TRL
Base Station Frame Structure	5-6 kg	1.4 x 1.4 x 0.9m	0	3
Drone Structure	2-3 kg	0.35 x 0.225 x 0.175m	0	3
Drone Docking platform	0.8-1.5 kg	0.5 x 0.5 x 0.15m	~5	3
Onboard Computer Enclosure	0.8-1.2 kg	0.35 x 0.30 x 0.18m	0	3
Dust Protection System	0.5-1 kg	1.5 x 1.5 x 1m	0	3
Regolith Collection assembly	0.5-1.5 kg	TBR	0	3
Solar Array/mount	1-2.5 kg	TBR	~5	3
Total Subsystem	~10.6-16.7 kg	1.5 x 1.5 x 1m	10	3

Figure 2.1.2: Mechanical Subsystem Summary Table

2.1.1.1. Mechanical Subsystem Requirements

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Req met?
MECH-001	The total missions' mechanical subsystems shall not exceed 16.7 kg of the allocated mission mass.	Ensures that all mechanical subsystems will remain within the mission mass that was provided by the customer.	SYS-015	MECH-BS-004 , MECH-DR-01 2	Inspection ▾	Yes ▾
MECH-002	The total missions' mechanical subsystems shall not exceed 1.5m x 1.5m x 1m of the allocated mission stowed volume	Ensures that all mechanical subsystems will remain within the mission volume allocation that was provided by the customer.	SYS-020	MECH-BS-005 , MECH-DR-01 3	Inspection ▾	Yes ▾
MECH-003	The total mission's mechanical subsystems shall survive the expected lunar south pole temperature range of -173°C - +127°C degrees during the mission.	Ensures that all mechanical subsystems will remain functional when exposed to harsh conditions.	SYS-010	MECH-BS-010 , MECH-DR-01 8	Test ▾	No ▾
MECH-004	The mechanical subsystem shall provide cable routing paths and strain-relief features that do not interfere with subsystem operation or maintenance.	Protects electrical wiring harnesses while maintaining accessibility and subsystem integration	SYS-004, MECH-BS-00 8, MECH-DR-01 5		Inspection ▾	Yes ▾
MECH-005	The mechanical subsystem shall accommodate thermal expansion and contraction without the loss of structural integrity or subsystem alignment over the mission's temperature range.	Prevents structural distortion and binding caused by extreme temperature ranges.	SYS-010, MECH-003		Analysis ▾	No ▾
MECH-BS-006	The base station mechanical subsystem shall not exceed 5-6 kg of	Ensures that the base station will stay within the total mission	SYS-015		Analysis ▾	Yes ▾

	the allocated mission mass.	mass budget.				
MECH-BS-007	The base station shall fit within a stowed volume of 2.25 m ³ for lunar delivery.	Ensures that the compatibility with commercial lunar delivery constraints.	SYS-020		Inspection ▾	Yes ▾
MECH-BS-008	The base station structure shall maintain static stability for all orientations while on the lunar surface, with slopes up to 5 degrees.	Prevents tipping or instability after landing on uneven lunar terrain.	SYS-010		Analysis ▾	No ▾
MECH-BS-009	The base station structure shall support all mounted subsystems under all expected loading conditions, and have a factor of safety greater than 1.5.	Ensures that mounted hardware can safely be supported during mission operations.	SYS-015		Analysis ▾	No ▾
MECH-BS-010	The base station structure shall provide mechanical mounting interfaces for the drone, payload, power, communication, and thermal subsystems.	Ensures that all required subsystems can be properly and physically integrated onto the base station.	SYS-004, PAY-001, PAY-002, PWR-BS-012		Inspection ▾	No ▾
MECH-BS-011	The base station shall provide a mechanical deployment interface that will allow a drone to be released without physical interference.	Ensures that a drone can deploy safely and successfully from the base station.	SYS-004		Demonstr... ▾	No ▾
MECH-BS-012	The base station shall survive the expected lunar south pole surface temperature of -173°C - +127°C degrees.	Ensures that the structure will remain functional in the lunar thermal environment.	SYS-010, PWR-BS-012		Test ▾	No ▾
MECH-BS-013	The base station exterior mechanical components, including external instruments, shall maintain structural functionality after exposure to lunar regolith without degradation that affects performance.	The lunar regolith is very abrasive and can degrade exposed components quickly.	SYS-010, CDH-010		Analysis ▾	No ▾
MECH-DR-014	The drone mechanical subsystem	Ensures that the drone will stay	SYS-015		Analysis ▾	Yes ▾

	shall not exceed 2-3 kg of the allocated mission mass.	within the total mission mass budget.				
MECH-DR-015	The drone shall fit within the stowed volume of TBR m ³ while it is attached to the base station.	Ensures that the drone can be packaged for delivery and deployment.	SYS-020		Inspection ▾	Yes ▾
MECH-DR-016	The drone structure shall withstand launch, landing, deployment, and operational loads without structural failure, and have a factor of safety greater than 1.5.	Ensures that the drone will survive all of the major mechanical loads that can occur.	SYS-010		Analysis ▾	No ▾
MECH-DR-017	The drone's physical structure shall provide the mechanical mounting interfaces that are required for all science, navigation, communication, and power components.	Ensures that all required drone subsystems can be physically integrated properly.	SYS-002, PAY-001, PAY-002, PAY-003, PAY-005, CDH-013		Inspection ▾	Yes ▾
MECH-DR-018	The drone mechanical subsystem shall protect mounted payloads from mechanical interference during the deployment and landing.	Prevents damage or obstruction to payloads during mechanical operations.	SYS-004, PAY-001, PAY-002, PAY-003, PAY-004, PAY-005, PAY-006, PAY-007, PAY-008,		Test ▾	No ▾
MECH-DR-019	The drone's exterior mechanical components shall tolerate the lunar regolith and dust exposure during mission operations.	Ensures that the drone components can operate in dusty lunar conditions	SYS-010, CDH-010		Analysis ▾	No ▾
MECH-DR-020	The drone shall survive the expected lunar south pole surface temperature of -173°C - +127°C degrees.	Ensures that the drone structure will remain functional in extreme temperature conditions.	SYS-010, PWR-DR-003		Test ▾	No ▾

MECH-DR-021	The drone shall maintain structural integrity after 5 deployment and landing cycles.	Ensures that the drone can complete the required number of mission operations that are needed.	SYS-004		Test ▾	No ▾
MECH-DR-022	The drone's mechanical subsystem shall provide structural support and mounting interfaces for the propulsion system throughout all mission phases.	Ensures that the propulsion system will remain securely mounted and capable of operating as intended during launch, deployment, flight, landing, and surface operations.	SYS-004		Inspection ▾	Yes ▾
MECH-DK-023	The drone docking assembly shall mechanically restrain the drone during launch, transit, and surface standby operations without unintended release under loading conditions.	Ensures that the drone remains secured and attached to the base station until deployment.	SYS-004, MECH-BS-008		Test ▾	No ▾
MECH-DK-024	The drone docking assembly shall provide alignment features that allow for repeatable docking within ± 10 mm positional and angular tolerance.	Ensures reliable docking and will minimize the chance of mechanical interference during recovery.	SYS-004, MECH-BS-009		Inspection ▾	Yes ▾
MECH-DK-025	The drone deployment mechanism shall maintain a minimum clearance of ± 10 mm from adjacent hardware throughout deployment.	Prevents collisions between the drone and surrounding base station hardware during deployment.	SYS-004, MECH-BS-009		Inspection ▾	No ▾
MECH-OBC-026	The onboard computer enclosure shall provide mounting interferences for all CDH hardware while allowing access for assembly, inspection, and maintenance.	Ensures that the CDH subsystem can be securely integrated and serviced.	SYS-002, MECH-BS-008		Inspection ▾	No ▾
MECH-OBC-027	The onboard computer system shall prevent lunar regolith intrusion into the enclosed electronics during mission operations.	Protects sensitive electronics from lunar dust contamination.	SYS-010, MECH-BS-011		Test ▾	No ▾

MECH-DUST-028	The dust protection system shall prevent the lunar regolith from degrading the performance of critical mechanical interfaces and electronic hardware.	Preserves long term reliability of mission critical components.	SYS-010, MECH-BS-01 1, MECH-DR-01 7		Test ▾	No ▾
MECH-RC-029	The regolith collection assembly shall provide a mechanical interface that is compatible with the selected science payload without obstructing the payload operation.	Ensures a successful integration between the mechanical and science subsystems.	SYS-002, PAY-001, MECH0BS-00 8		Demonstr... ▾	No ▾
MECH-RC-030	The regolith collection assembly shall collect and transfer the lunar regolith without mechanical blockage during nominal operation.	Ensures that a reliable sample collection will be made during the mission.	SYS-002, PAY-001		Test ▾	No ▾
MECH-SA-031	The solar array mounting assembly shall support all expected operational loads without permanent deformation or structural failure.	Ensures that the structural support is reliable for the deployed solar array.	SYS-015, MECH-BS-00 7		Analysis ▾	No ▾
MECH-SA-32	The solar array mounting assembly shall allow deployment without interference with the drone, payloads, communications hardware, and or base station structure.	Prevents deployment failures that are caused by mechanical interference.	SYS-004, MECH-BS-00 8		Demonstr... ▾	No ▾
MECH-SA-33	The solar array mounting assembly shall securely maintain the deployed solar array orientation throughout the mission operations.	Ensures a stable solar array position after it is deployed.	SYS-015, MECH-BS-00 7		Analysis ▾	No ▾

Figure 2.1.1.1.1: Mechanical Subsystem Requirements

2.1.1.2. Mechanical Sub-Assembly Overview

Base Station Structural Frame

The base station structural frame serves as one of the primary load bearing structures for the spacecraft. It provides mounting locations for all of the major subsystems. This includes the power system, communication hardware, onboard computer enclosure, science payloads and drone docking mechanism. The structure is designed to maintain stability during lunar landing and all other operational loads that are encountered during the mission. Based on the structural frame trade study, Aluminum 6061-T6 was selected because it provides the best balance between strength, manufacturability, corrosion resistance, and finally low mass. This subassembly was assigned a TRL of 3 because the structural concept and material have been selected through trade studies and preliminary designs. Detailed CAD development, structural analysis, and physical testing will be needed to help mature this design and raise the TRL level.

Drone Structural Frame

The drone structural frame is the primary support system for the propulsion system, avionics, landing gears, payload, and communication hardware. The frame needs to withstand launch, deployment, flight, landing, and operational loads while trying to be as light as possible, but maintaining structural integrity to maximize flight performance. During this trade study Aluminum 7075-T6 was the selected material due to its high strength to weight ratio. This makes it well suited for lightweight aerospace structures such as the drone frame. This subassembly was given a TRL of 3 because the structural concept has been established and the preliminary material selection was completed using trade studies. Before the TRL can increase, design refinement, structural verification and prototype testing needs to be completed.

Drone Docking and Deployment Assembly

The docking and deployment assembly is going to mechanically secure the drone to the base station during the launch transit and surface operations that will take place. This assembly will provide alignment features and a controlled release mechanism that will allow the drone to deploy with interference from any neighboring subsystems. The design also provides repeatable positioning for docking after mission operations. This assembly was given a TRL of 3 because the concept for deployment has been defined and evaluated at the design level. Future work includes detailed mechanism design, prototype fabrication, and functional real world testing to validate the performance.

Onboard Computer Enclosure

The onboard computer enclosure will house the mission computer and other associated electronics within the base station. This enclosure will provide the structural support while also protecting these sensitive electronic components from the lunar dust, and other extreme conditions that could arise during mission operations. This enclosure will also have mounting interfaces for the CDH subsystem. This will allow for convenient

access for assembly and maintenance during ground operations. For the onboard computer enclosure a sealed aluminum enclosure was selected during the trade study as it allows for excellent structural integrity, environmental protection, and ease of manufacturing. This was given a TRL of 3 because the enclosure concept and material have been chosen through preliminary trade studies. A detailed design, interface verification, and environmental testing is needed to help increase the TRL level.

Dust Protection System

The dust protection system will minimize the amount of abrasive lunar regolith that will harm the mechanical and electronic components. Dust mitigation is completed through protective enclosures, seals, and barriers that will be placed around each critical interface and/or electronic hardware. Following from the dust mitigation trade study the, the option that was picked was a fully sealed enclosure as the baseline because it provides the highest level of protection for these mission critical components. The dust protection system was given a TRL of 3 because the dust mitigation approach was selected through trade studies, and for it to increase in TRL level it needs

Regolith Collection Assembly

The regolith collection assembly will provide the structural support and collection interface that is required for the science subsystem. The purpose of this assembly is to help analyze the lunar soil samples. The assembly is integrated into the base station to provide the structural support for the science instrument while also enabling efficient regolith sample collection with the least amount of moving parts. This is to ensure that there are less places for the collection assembly to fail. Based on the trade study that was completed, a passive funnel collection system was selected because it offers a high reliability, low mass, and simple integration while satisfying the science objectives. This received a TRL level of 3 because this concept was only chosen through trade studies. It needs more mechanical design, integration and testing before it can increase.

Solar Array Mounting Assembly

The solar array mounting assembly provides the structural support for deployable solar panels and will transfer the loads into the base station structure. The assembly is designed to securely support the solar array during launch, landing, deployment, and all surface operations. The solar array mount needs to maintain structural rigidity and minimize the overall mass. The design prioritizes simplicity, reliability, and ease of integration with the electrical subsystems. This subassembly was assigned a TRL of 3 because the mounting concept has been established through the conceptual design phase and preliminary trade studies. To increase the TRL level the mounting assembly will require

a detailed Structural design and integration with the dollar array.

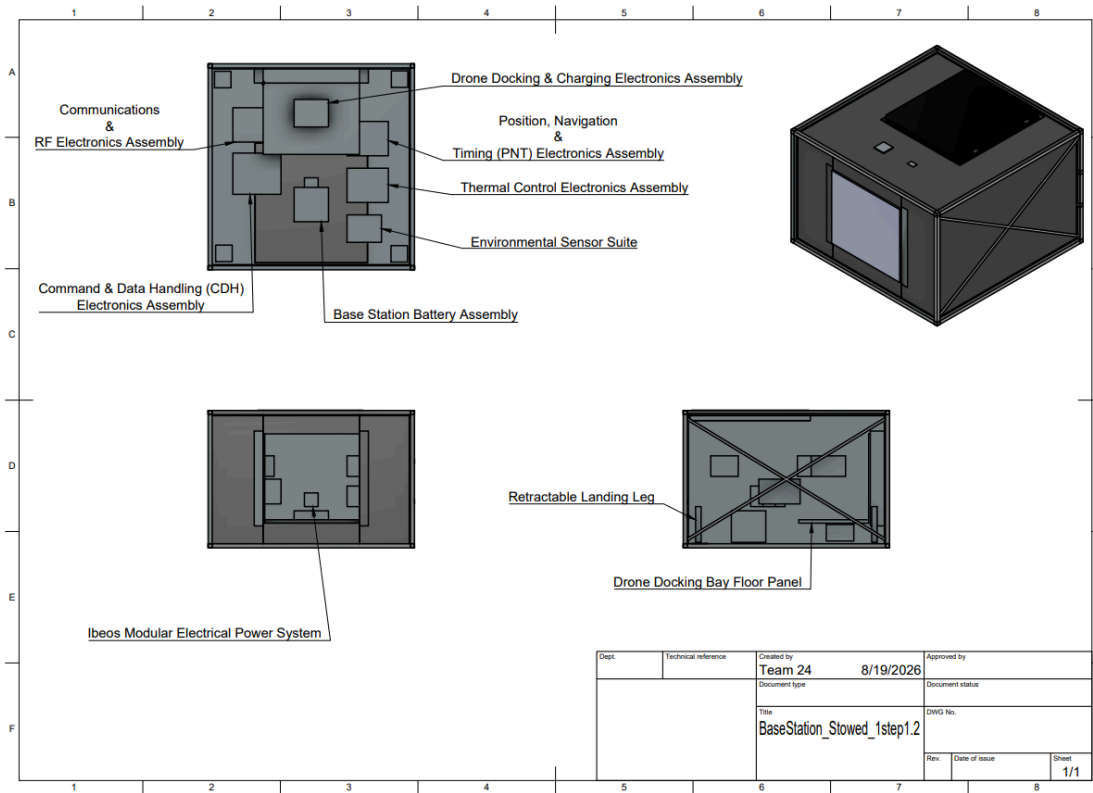


Figure 2.1.2.1: Base Station Interior CAD

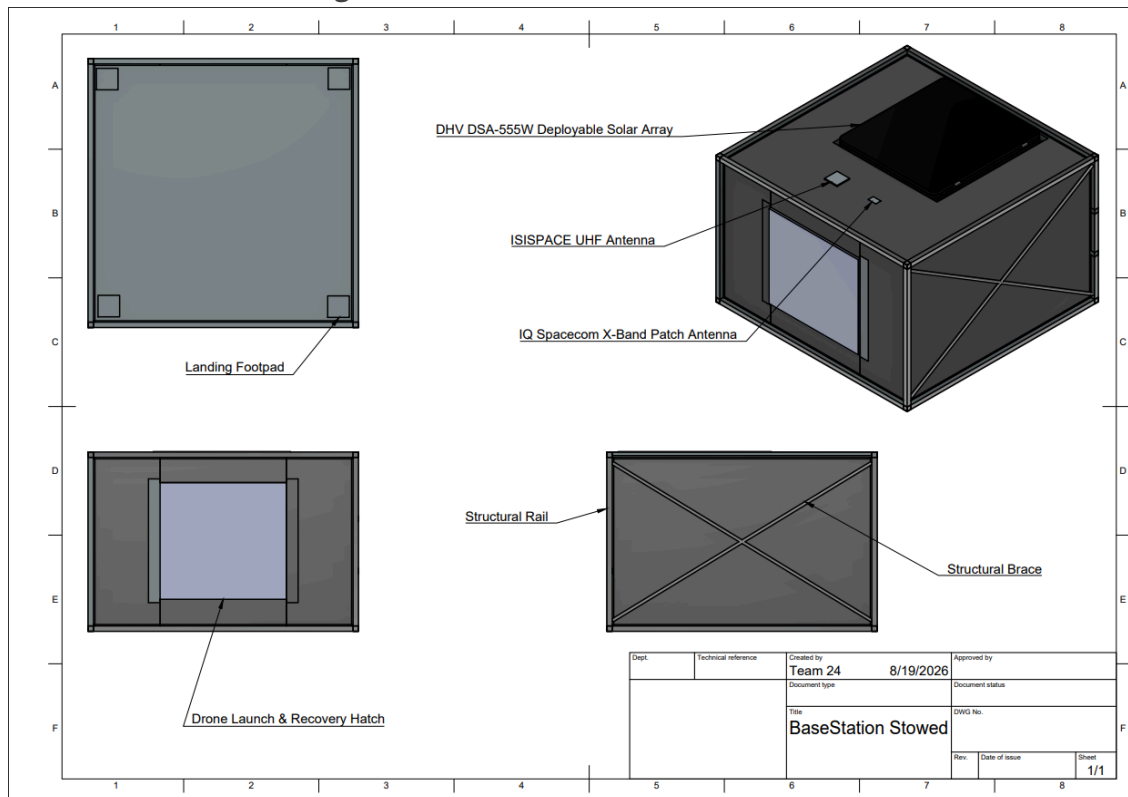


Figure 2.1.2.2: Base Station Exterior CAD

2.1.1.3. Mechanical Subsystem Recovery and Redundancy

Redundancy

The mechanical subsystem uses primarily passive redundancy and design rather than a fully duplicated structure or mechanism. This is because duplicating major mechanical hardware would significantly increase the mass and complexity of the designs. Redundancy is incorporated through the conservative structural sizing, multiple load paths, standardized fasteners, and secondary retention features that are the interfaces. The base station and drone structures are designed so that the damage or loss of a single noncritical attachment point does not immediately result in a total failure. The critical mounted components may also use multiple fasteners, backup brackets, locking hardware, and mechanical stops to help reduce the risk of separation during different phases of the mission.

Additionally using a fixed landing gear, sealed protective enclosures, and secondary dust barriers will also help with redundancy. Fixed landing legs help reduce the number of moving components that could fail. Sealed enclosures and brush seals provide the layered protection that is needed to help with lunar regolith. The drone docking assembly also includes an alignment feature and a secondary retention method to help keep the drone secured if the primary latch does not fully engage correctly.

Recovery

Mechanical recovery is focused on maintaining a safe configuration and preserving all mission critical functions after a fault occurs. If a feature on the drone docking or deployment mechanism were to fail the system may prevent deployment and keep the drone mechanically secured to prevent major damage to other nearby hardware. Mechanical stops and retention features can also help isolate the failure mechanism and prevent unintended motion.

If damage were to occur to a noncritical bracket, cover, seal, or mounting interface, the structure is intended to continue supporting all essential equipment through the remaining load paths that were designed. A failure of the regolith collection assembly would not prevent the base station from continuing communication, power generation, and other mission critical operations. Similarly, if damage were to occur to a secondary dust seal it would be partially mitigated by the fully sealed enclosure, thus protecting the internal components that are there. Recovery actions would prioritize the affected assembly and

make sure it gets to a stable condition, which limits any further mechanical movement, or damage.

2.1.1.4. Mechanical Subsystem Manufacturing and Procurement Plans

Base Station Structure

The base station structure will be procured using a build-to-print method. GRACE Aerospace was selected as the primary supplier, with Northrop Grumman as the backup supplier. This component requires a large structural aluminum assembly that can support the power, CDH, payload, thermal, communication, and drone docking hardware. GRACE Aerospace was selected because the base station structure requires CNC machining, fabrication, inspection, and aerospace systems integration. Northrop Grumman was selected as the backup supplier because it has large-scale aerospace manufacturing and integration capability.

The estimated lead time for the base station structure is 16–20 weeks. If the backup supplier must be used, additional time may be required for drawing review, supplier onboarding, and manufacturing schedule availability.

Drone Structure

The drone structure will be procured using a build-to-print method. Accurate Machine and Tool Corp. was selected as the primary supplier, with Protolabs as the backup supplier. The drone frame requires a precision-machined lightweight aluminum structure with tight tolerances. Since the drone structure directly affects flight mass, payload support, landing loads, and propulsion mounting, the selected supplier must be capable of high-quality machining and fabrication.

The estimated lead time for the drone structure is 12–18 weeks. Protolabs is a suitable backup because it can support rapid low-volume prototype manufacturing if the primary supplier is unavailable or delayed.

Onboard Computer Enclosure

The onboard computer enclosure will be procured using a build-to-print method. Protolabs was selected as the primary supplier, with Xometry as the backup supplier. This enclosure will be a precision-machined sealed aluminum housing designed to protect the onboard computer and related electronics from lunar dust, contamination, and mechanical damage.

The estimated lead time for the onboard computer enclosure is 6–10 weeks. This is shorter than the larger structural assemblies because the enclosure is a smaller machined component and can be produced through rapid low-volume manufacturing. Xometry was selected as a backup supplier because it can provide distributed manufacturing support for machined parts if schedule risk occurs.

Drone Docking Mechanism

The drone docking mechanism will be procured using a build-to-print method. Moog was selected as the primary supplier, with Lockheed Martin as the backup supplier. The docking mechanism must secure the drone during launch, transit, and surface operations while also allowing controlled release and repeatable docking. This component may require supplier support for motion control, latching reliability, alignment features, and qualification testing.

The estimated lead time for the drone docking mechanism is 20–28 weeks. This is one of the longer lead-time components because it is a more complex mechanical assembly and may require additional testing before final integration.

Solar Array Mount

The solar array mount will be procured using a build-to-print method. Honeybee Robotics was selected as the primary supplier, with Moog as the backup supplier. The solar array mount must support the solar array, transfer operational loads into the base station structure, and allow deployment or pointing motion if required by the power subsystem.

The estimated lead time for the solar array mount is 18–24 weeks. Honeybee Robotics was selected because the component may require planetary robotics and electromechanical actuation experience. Moog is a strong backup supplier because of its experience with precision motion-control hardware.

Regolith Collection Assembly

The regolith collection assembly will be procured using a build-to-print method. Honeybee Robotics was selected as the primary supplier, with Lockheed Martin as the backup supplier. This assembly supports the mission's sample collection and science objectives. It requires experience with planetary sampling, regolith acquisition, containment, and contamination control.

The estimated lead time for the regolith collection assembly is 18–26 weeks. This longer lead time is due to the specialized nature of lunar regolith collection hardware and the need to verify that the assembly can operate reliably in a dusty lunar environment.

Dust Protection Hardware

The dust protection hardware will be procured using a build-to-print method. Accurate Machine and Tool Corp. was selected as the primary supplier, with GRACE Aerospace as the backup supplier. This hardware includes machined covers, seals, brackets, and protective enclosures that reduce lunar dust intrusion into critical mechanical and electronic interfaces.

The estimated lead time for the dust protection hardware is 8–12 weeks. Accurate Machine and Tool Corp. was selected because these parts mainly require conventional aerospace machining and fabrication methods. GRACE Aerospace is a suitable backup because it can support aerospace mechanical fabrication and integration work.

Cold Gas Thrusters

The thrust for the drone is created using cold gas thrust, the design was created around existing products from current manufacturers developing cold gas thrust for use in small satellites. The primary supplier is IBB, they were selected for form factor, ease of integration, and small mass, though they are a relatively small supplier. The backup supplier is Moog, who develops slightly less apt cold gas thrusters. The estimated lead time is 8–12 weeks.

Cold Gas Plumbing

The cold gas plumbing system will be designed using components provided by VACCO with moog selected as a backup supplier. This includes the tanks, plumbing, pumps, and control devices needed to run cold gas thrust.

The estimated lead time for the cold gas plumbing system is 6–8 weeks. VACCO was selected as the primary supplier for ease of integration with the primary CGT selection. Moog is selected as the backup supplier to prepare for the possibility that IBB will not be a viable supplier for the CGT's

Lead Time Summary

Mechanical subsystem procurement lead times are estimated to range from 6–28 weeks, depending on component complexity. Smaller machined parts, such as the onboard computer enclosure and dust protection hardware, have shorter lead times. Larger or more complex assemblies, such as the drone docking mechanism, solar array mount, and

regolith collection assembly, require longer lead times because they may involve deployment features, motion control, alignment verification, and qualification testing.

If backup suppliers must be used, the expected schedule impact is approximately 4–8 additional weeks. This delay would come from supplier onboarding, drawing review, manufacturability review, contract setup, and manufacturing slot availability. The highest schedule-risk items are the drone docking mechanism and regolith collection assembly because they require more specialized supplier expertise.

Subsystem / Component	Procurement Method	Primary Supplier	Backup Supplier	Estimated Lead Time	Justification
Base station structure	Build-to-Print... ▾	GRACE aerospace	Northrop Grumman	16-20 weeks	Large structural aluminum assembly requiring CNC machining, fabrication, inspection, and aerospace systems integration.
Drone structure	Build-to-Print... ▾	Accurate machine and tool corp.	Protolabs	12-18 weeks	Precision machined lightweight aluminum frame requiring tight manufacturing tolerance and aerospace quality fabrication.
Onboard computer enclosure	Build-to-Print... ▾	Protolabs	Xometry	6-10 weeks	Precision machined sealed aluminum enclosure with rapid manufacturing capability and low volume aerospace production.
Drone docking mechanism	Build-to-Print... ▾	Moog	Lockheed Martin	20-28 weeks	Precision deployment and latching mechanism that may require supplier support, motion control expertise, and qualification testing.
Solar array mount	Build-to-Print... ▾	Honeybee Robotics	Moog	18-24 weeks	Precision deployment and pointing mechanism that needs planetary robotics and electromechanical actuation experience.
Regolith collection assembly	Build-to-Print... ▾	Honeybee Robotics	Lockheed Martin	18-26 weeks	Planetary sampling mechanism that needs specialized expertise in regolith acquisition, containment, and contamination control.
Dust protection hardware	Build-to-Print... ▾	Accurate Machine and	GRACE aerospace	8-12 weeks	Precision machined covers, seals, brackets,

		Tool Corp			and protective enclosures manufactured using conventional aerospace fabrication methods.
Cold Gas Thrusters	Select Method ▾				
Cold Gas Plumbing	Select Method ▾				

Figure 2.1.1.4.1: Mechanical Procurement Summary Table

2.1.1.5. Mechanical Subsystem Verification Plans

The mechanical subsystem will be verified using a combination of analysis, inspection, testing, and then demonstration to confirm that each mechanical requirement has been met. The verification method for each requirement was selected based on the type of performance or design feature that is being evaluated. The “analysis” method will primarily be used to help with structural loading, stability, mass, and environmental design requirements. The “inspection” method will be used to help verify that the dimensions, mounting interfaces, clearances, cable routing and other physical design features are correct. The “testing” method will help demonstrate that the mechanical hardware must achieve an acceptable performance under representative loading, thermal, dust, deployment, or any other operational conditions. The “Demonstration” method will be used for requirements that involve integrated mechanical functions, such as the deployment and operation of major assemblies. The “verification” method will come from CAD and the analytical evaluation during the design phase to prototype and integrate subsystem testing as the mechanical design grows.

Req #	Requirement Summary	Verification Method	Rationale for Method	Preliminary Verification Plan
MECH-001	Mechanical Subsystem $\leq 16.7\text{kg}$	Inspection ▾	Confirms the total mechanical mass against the allocated mass limit.	Review CAD mass properties and verify final assembled mechanical mass is $\leq 16.7\text{ kg}$.
MECH-002	Mechanical subsystem fits within $1.5 \times 1.5 \times 1\text{m}$ stowed volume	Inspection ▾	Physical dimensions can be directly checked against the required envelope.	Inspect CAD and final assembly dimensions to confirm the stowed configuration remains within the allocated envelope.
MECH-003	Mechanical subsystem survives -173°C to $+127^{\circ}\text{C}$	Test ▾	Thermal survivability requires operation under representative temperature conditions.	Perform environmental/thermal testing and inspect the mechanical hardware for failure, excessive deformation, or loss of function.
MECH-004	Provide cable routing and strain	Inspection ▾	Routing and clearances can be directly evaluated	Inspect cable paths, attachment points, strain relief, and accessibility

	relief without interference		from CAD and the integrated assembly.	during CAD integration and final assembly.
MECH-005	Accommodate thermal expansion/contraction without loss of structural integrity or alignment	Analysis ▾	Thermal deformation can be predicted analytically before environmental testing.	Perform thermal expansion/deformation analysis across the specified temperature range and verify acceptable alignment and stresses.
MECH-BS-006	Base station mechanical mass $\leq 5-6$ kg	Analysis ▾	CAD mass properties can determine whether the design meets its mass allocation.	Calculate mass using CAD material assignments and compare the result with the 5-6 kg allocation.
MECH-BS-007	Base station fits within allocated stowed volume	Inspection ▾	Dimensions can be directly checked against the allocated envelope.	Inspect CAD dimensions and the integrated stowed configuration against the final allocated volume.
MECH-BS-008	Base station remains statically stable on required lunar slopes of 5°	Analysis ▾	Stability can be evaluated using center-of-mass and tipping calculations.	Determine the center of mass and evaluate tipping stability at the required maximum surface slope.
MECH-BS-009	Base station supports expected loads with FoS >1.5	Analysis ▾	Structural analysis can determine stress, deformation, and factor of safety.	Perform structural/FEA analysis using expected loading conditions and verify FoS >1.5 .
MECH-BS-010	Provide mounting interfaces for required subsystems	Inspection ▾	Interfaces can be directly confirmed in CAD and the integrated assembly.	Inspect CAD and interface drawings to confirm mounting provisions for drone, payload, power, communications, and thermal hardware.
MECH-BS-011	Provide interference-free drone deployment interface	Demonstrations ▾	Successful deployment requires demonstrating the integrated mechanism.	Demonstrate drone release/deployment using the integrated base station and verify no physical interference occurs.
MECH-BS-012	Base station survives -173°C to $+127^\circ\text{C}$	Test ▾	Environmental testing is needed to confirm thermal survivability.	Subject representative hardware to the required thermal range and verify structural functionality is maintained.
MECH-BS-013	Exterior components tolerate lunar regolith exposure	Analysis ▾	Material and design compatibility with the dust environment can initially be evaluated analytically.	Evaluate exposed materials, seals, and interfaces for expected regolith exposure and confirm no predicted performance-limiting degradation.
MECH-DR-014	Drone mechanical mass $\leq 2-3$ kg	Analysis ▾	CAD mass properties can verify compliance with the drone mass allocation.	Calculate the drone mechanical mass from CAD and compare it with the 2-3 kg allocation.
MECH-DR-015	Drone fits within allocated stowed volume	Inspection ▾	Packaging can be directly verified geometrically.	Inspect the stowed drone CAD while attached to the base station and compare dimensions with the final allocated volume.

MECH-DR-016	Drone survives expected loads with FoS >1.5	Analysis ▾	Structural analysis can determine stresses and factor of safety under expected loads.	Perform FEA for launch, deployment, flight, and landing load cases and verify FoS >1.5.
MECH-DR-017	Drone provides mounting interfaces for required hardware	Inspection ▾	Physical mounting provisions can be directly verified.	Inspect CAD and interface drawings for all required science, navigation, communication, and power mounting locations.
MECH-DR-018	Protect payloads from mechanical interference	Test ▾	Integrated operation must show payloads remain unobstructed during deployment and landing.	Test representative deployment/landing motions and verify adequate clearance around payload hardware.
MECH-DR-019	Drone exterior tolerates lunar regolith/dust	Analysis ▾	Materials and exposed interfaces can initially be evaluated for compatibility with expected dust exposure.	Analyze exposed mechanical components and protective features for expected regolith conditions and identify unacceptable degradation risks.
MECH-DR-020	Drone survives -173°C to +127°C	Test ▾	Thermal survivability requires environmental testing.	Subject representative drone mechanical hardware to the required thermal range and verify continued structural functionality.
MECH-DR-021	Drone maintains structural integrity after 5 required deployment/landing cycles	Test ▾	Repeated-cycle performance requires physical testing.	Perform the required number of deployment and landing cycles and inspect for damage, loosening, or structural degradation.
MECH-DR-022	Provide structural support/mounting for propulsion system	Inspection ▾	Mounting interfaces can be directly verified against the propulsion configuration.	Inspect propulsion mounting points and interfaces in CAD and the integrated drone assembly.
MECH-DK-023	Docking assembly restrains drone without unintended release	Test ▾	Retention performance must be verified under representative loads.	Apply representative loading to the docked drone and confirm the retention mechanism does not release unintentionally.
MECH-DK-024	Provide repeatable docking alignment within required tolerance	Inspection ▾	Alignment features and achieved position can be measured directly.	Perform repeated docking operations and measure positional/angular alignment against the final specified tolerances.
MECH-DK-025	Deployment mechanism maintains required clearance	Inspection ▾	Clearance can be directly measured throughout the deployment envelope.	Inspect CAD motion envelopes and prototype deployment to verify minimum clearance from adjacent hardware.
MECH-OBC-026	Enclosure provides CDH mounting and maintenance access	Inspection ▾	Mounting provisions and accessibility are directly observable.	Inspect enclosure CAD and integrated hardware for secure CDH mounting and required assembly/maintenance access.

MECH-OBC-027	Enclosure prevents regolith intrusion	Test ▾	Sealing effectiveness requires representative dust exposure.	Expose the enclosure to simulated regolith/dust and inspect the interior for particle intrusion.
MECH-DUST-028	Dust protection prevents performance degradation	Test ▾	Protection effectiveness must be evaluated under simulated dust exposure.	Expose protected mechanical/electronic interfaces to simulated regolith and verify continued functionality.
MECH-RC-029	Collection assembly interfaces with payload without obstruction	Demons... ▾	Integrated operation is needed to show compatibility between the assemblies.	Demonstrate the collection assembly with representative payload hardware and verify unobstructed operation.
MECH-RC-030	Collect and transfer regolith without blockage	Test ▾	Collection performance must be physically evaluated with representative material.	Operate the collection system using lunar-regolith simulant and verify successful collection and transfer without blockage.
MECH-SA-031	Solar array mount supports operational loads without permanent deformation/failure	Analysis ▾	Structural analysis can predict stresses and deformation under expected loading.	Perform structural/FEA analysis of the mount under expected load cases and verify no structural failure or permanent deformation.
MECH-SA-032	Solar array deploys without interference	Demons... ▾	Deployment clearance is best verified by demonstrating the complete motion.	Demonstrate the solar-array deployment sequence and verify clearance from the drone, payload, communications hardware, and base station.
MECH-SA-033	Solar array mount maintains deployed orientation	Analysis ▾	Structural stiffness and expected deflection can be evaluated analytically.	Analyze the deployed mount under expected operational loads and confirm orientation remains within the allowable design tolerance.

Figure 2.1.1.5.1: Mechanical Subsystem Verification Table

Verification Method Rationale

The verification methods for the mechanical subsystem were selected based on the type of requirement that was being evaluated. Analysis was primarily used for requirements that involved mass, structural loading, stability, thermal expansion, and expected deformation. This was because these characteristics can be evaluated using CAD mass properties, engineering calculations, and structural analysis before any physical hardware is available. Inspection was used for the requirements that involve dimensions, mounting interfaces, cable routing, accessibility, and mechanical clearances because the characteristics can be directly confirmed using CAD models, engineering drawings, and the assembled hardware. Testing was used for requirements that involve thermal survivability, dust protection, repeated mechanical cycles, structural retention, and regolith collection because these requirements depend on the physical response of the hardware under its representative condition that each requirement will face. Demonstration is used for the integrated

mechanical functions such as drone deployment, regolith collection integration, and the solar array deployment, where successful operation and freedom from any interference must be shown to complete the assembly.

Preliminary Verification Plan

Mechanical verification will occur throughout the design, fabrication, and integration phases of the mission. During the preliminary and detailed design phases the CAD, models, engineering drawings, mass properties, structural calculations, and FEA will be used to verify the mass, volume, stability, structural loading, thermal deformation, and interface requirements. Successful verification will require the calculated values to remain within the limits that were established by the applicable mechanical requirement. As the mechanical designs grow, inspection will be performed on the CAD models and fabricated assemblies to verify its dimensions, mounting locations, cable routing, accessibility, and mechanical clearances.

Physical testing and demonstrations will occur once the prototypes or integrated assemblies are available. Structural, deployment, thermal, dust, docking, and regolith collection assemblies will be tested, and they will be performed using the appropriate measurement, environmental, lunar regolith testing equipment where applicable. The integrated demonstrations will be used to verify drone deployment, regolith collection integration, and solar array deployment without any interference. A requirement will be considered successfully verified, when the applicable analysis inspection test, or demonstration confirms that its stated performance criteria has been achieved.

2.1.2. Power Subsystem Overview

Power Subsystem Overview

The electrical power subsystem (EPS) provides all power generation, storage, distribution, and fault protection functions for the base station and the drone. Its baseline architecture includes various power modes designed to keep vital systems online during times of low power generation. Both the base station and the drone have separate units. Both vehicles contain similar components, but they are electrically independent during each drone flight.

Every subsystem depends on the EPS in order to function. Throughout the different phases of the mission, the EPS provides critical functions such as ensuring the drone is fully charged, and that all instruments are operating nominally. It certifies that every

subsystem's power draw needs are met without fully exhausting the power supply and endangering the mission.

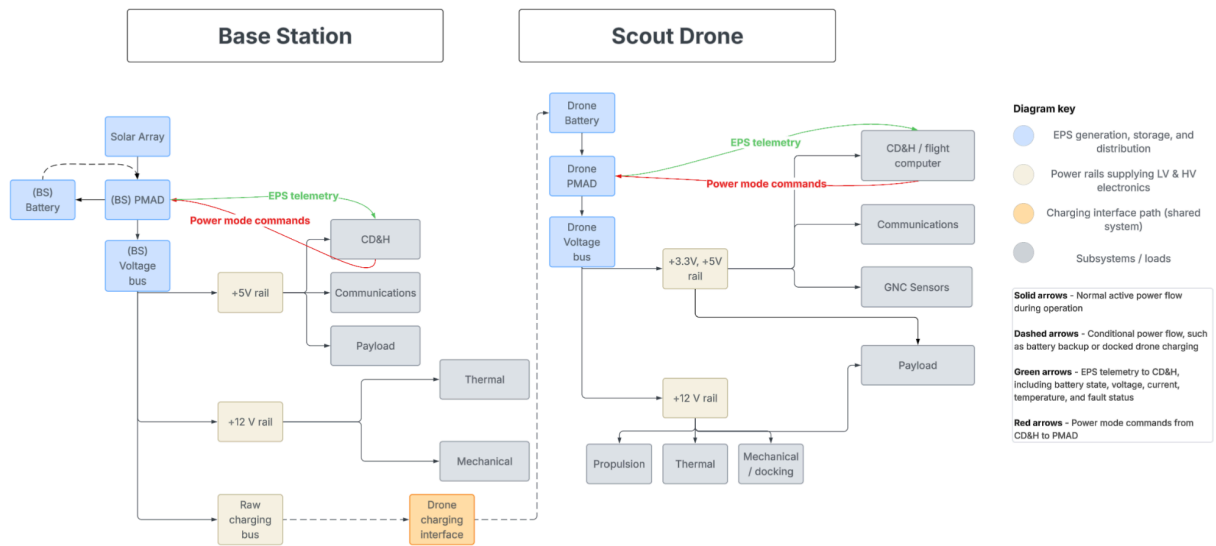


Figure 2.1.2.01: EPS Flow Chart

Power Capabilities

The Electrical Power Subsystem was designed with the intention of being able to sustain sufficient power generation levels, distribute power efficiently, and store power throughout the mission duration, including during periods of lunar night. Each subassembly is capable of performing at full efficiency throughout the mission. Still, the EPS was designed to survive a lunar night cycle should the lifespan align with lunar night at the landing site. Degrading efficiency within any subassembly will not be of major concern throughout the mission's lifespan.

Subassembly	Mass (kg)	Dimensions (m)	Maximum Power Draw (W)	TRL
DHV Solar Arrays	~27.8 kg	762.51 mm x 850 mm	0 W (Passive)	9
Ibeos Modular EPS	~1 kg	0.1 x 0.09 x 0.1	~10 - 40 W (Dependent on configuration)	8
EaglePicher NPD-002271 Base Station	~25 kg	0.25m×0.25m×0.23m	0 W (Passive)	7

EaglePicher NPD-002271 Drone	0.49 kg	0.12 m × 0.10 m × 0.03 m	0 W (Passive)	7
Smiths Interconnect IDI Spring Probe	0.01-0.05kg	~0.0041 × 0.0014 × 0.0014 m	0 W (Passive)	8
Smiths Interconnect IDI target contacts	~0.02-0.05kg	~0.0050 × 0.0011 × 0.0011 m	0 W (Passive)	8
Total Subsystem	~54.3 kg	0.4 m x 0.4 m x 0.55 m	~10-40 W	7.8

Figure 2.1.2.02: EPS Subsystem Spec Sheet

2.1.2.1. Power Subsystem Requirements

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Req met?
PWR-BS-001	The base station power subsystem shall store at least 3.84 kWh of electrical energy to support critical operations during low-power generation periods.	Ensures the base station can remain operational at times of low-power generation in low light conditions	SYS-010	PWR-BS-005	Analysis	Yes
PWR-BS-002	The base station charging interface shall protect against overcurrent, overvoltage, undervoltage, and other battery abuse conditions	Ensures the scout drone and the charging interface are protected from adverse battery conditions and potential faults	PWR-BS-004	PWR-DR-019	Test	Yes
PWR-BS-003	The base station power subsystem shall monitor and report system status and health at regular intervals	Ensures long-term functionality and notifies operators of faults	SYS-012	PWR-DR-018	Test	Yes
PWR-BS-004	The base station shall support multiple power modes, including nominal, charging, low-power, and safe	Ensures complete functionality of the base station throughout different phases of the mission	SYS-011	PWR-BS-002	Demonstration	Yes

	modes					
PWR-BS-005	The base station power subsystem shall distribute regulated electrical power to all subsystems	Ensures all base station subsystems receive stable and usable electrical power	PWR-BS-001	PWR-BS-006, PWR-BS-007, PWR-BS-008, PWR-BS-009, PWR-BS-010, PWR-BS-011	Test	Yes
PWR-BS-006	The base station shall allocate 200 W to the payload subsystem during nominal operations	Ensures the science payload receives sufficient electrical power to collect mission data.	PWR-BS-005		Analysis	Yes
PWR-BS-007	The base station power subsystem shall provide up to 40 W to the EE subsystem for EPS/PMAD overhead, control electronics, and power monitoring.	Ensures electrical support equipment, power control hardware, and secondary electronics receive sufficient power	PWR-BS-005		Analysis	Yes
PWR-BS-008	The base station power subsystem shall provide up to 100 W to the mechanical subsystem for deployment, docking, and mechanism operations.	Ensures mechanical components, actuators, deployment systems, and related hardware receive sufficient power when needed	PWR-BS-005		Analysis	Yes
PWR-BS-09	The base station power subsystem shall provide up to 345 W to the thermal subsystem during peak heater operation.	Ensures thermal control hardware can maintain required operating temperatures in the lunar surface environment	PWR-BS-005	THERM-BS-004	Analysis	Yes
PWR-BS-010	The base station power subsystem shall provide up to 70 W to the CDH subsystem during peak processing, monitoring, and communication support.	Ensures the base station computer, data storage, and control electronics remain powered for mission operations	PWR-BS-005	CDH-010	Analysis	Yes
PWR-BS-011	The base station power subsystem shall provide sufficient electrical power to support thermal control	Ensures base station electronics, batteries, payload, and other temperature-sensitive components remain	PSR-BS-005	THERM-BS-004	Analysis	Yes

	components, including a 29.5 W survival-mode thermal load, required to maintain subsystem operating temperatures within allowable limits.	functional during lunar surface operations.				
PWR-BS-012	All Power subsystem subassemblies shall meet or exceed TRL 6.	Meets system TRL requirement	SYS-008		Analysis	Yes
PWR-DR-013	The scout drone's onboard flight computer shall complete a power-system pre-flight checklist before each flight, including battery state of charge, voltage, current, temperature, and fault status.	Ensures subsystem conditions are nominal before each flight, mitigating risk	SYS-012	PWR-DR-004	Test	Yes
PWR-DR-014	The scout drone power subsystem shall provide sufficient electrical power to support onboard thermal control components required to maintain battery, electronics, and payload operating temperatures within allowable limits	Ensures the scout drone remains functional while operating in the lunar surface thermal environment	SYS-010		Analysis	Yes
PWR-DR-015	The scout drone power subsystem shall maintain a minimum battery state of charge of 80% before deployment from the base station.	Ensures the scout drone can perform all required operations during flight	PWR-DR-013		Analysis	Yes
PWR-DR-016	The scout drone power subsystem shall enter low-power mode when battery state of charge falls below 30%.	Preserves remaining energy for communication, recovery, or return-to-base operations.	PWR-DR-014 , SYS-007		Demonstration	Yes
PWR-DR-017	The scout drone power subsystem shall maintain sufficient reserve energy to return	Prevents loss of the scout drone during science operations.	PWR-DR-014 , SYS-007		Analysis	Yes

	to the base station using the 75 Wh battery capacity and mission-defined flight power profile.					
PWR-DR-018	The scout drone power subsystem shall report battery state of charge, voltage, current, temperature, and power fault status to the base station.	Allows the base station and operators to monitor drone power health before and during sorties.	PWR-BS-003		Test	Yes
PWR-DR-019	The scout drone power subsystem shall be rechargeable to at least 80% battery state of charge through the base station charging interface.	Allows the drone to complete multiple flights throughout the mission.	SYS-004	PWR-BS-002	Demonstration	Yes

Figure 2.1.2.1.1: EPS Requirements chart

2.1.2.2. Power Sub-Assembly Overview

Solar Array Subassembly

The primary power source for the base station is the DHV DSA-1268W Sextuple Deployable Solar Array with a power generation capability of 1.3kW at the start of the mission. Due to the low angle of the sun at the selected landing site, the arrays will be deployed horizontally relative to the base station while the cells are positioned in a near vertical fashion, directly at the Sun's expected angle, in order to maximize output. DHV solar arrays have flight heritage in other small satellites and spacecraft, earning it a TRL of at least 8. The 1.3kW rating per array is sufficient for daytime operations of all subsystems. The generated power is then directed into the power management and distribution (PMAD) subassembly. This subassembly integrates with the solar array mounting assembly, which provides adequate structural support for the array throughout the mission's duration.

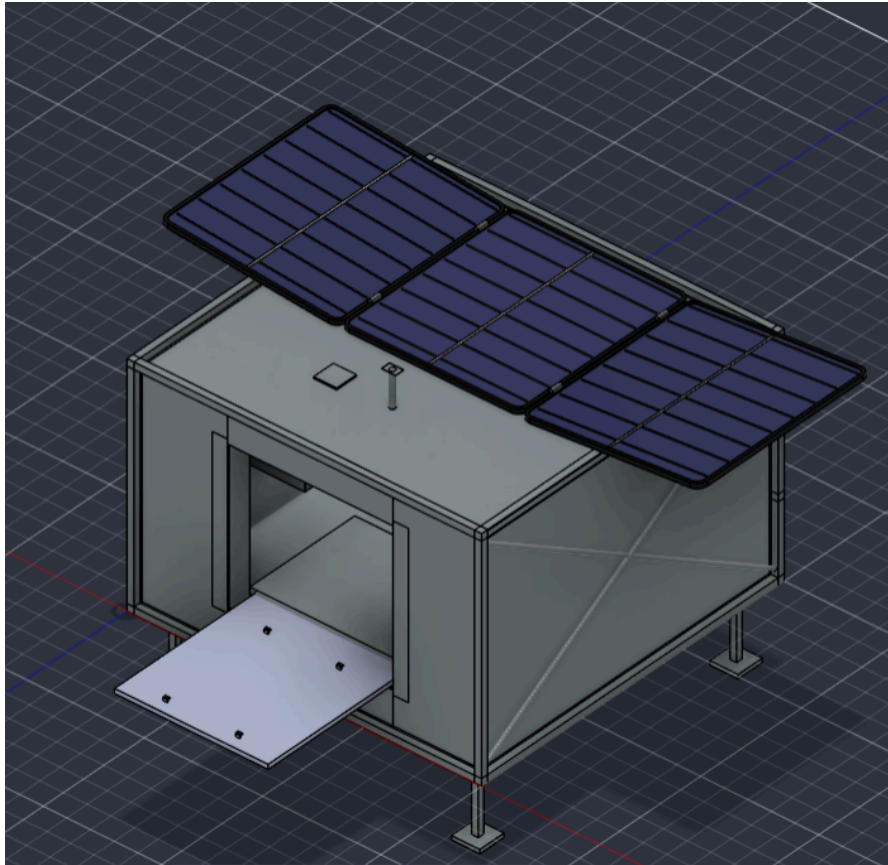


Figure 2.1.2.2.1: Deployed Solar Array Base Station Configuration

PMAD Subassembly

The selected PMAD is the Ibeos Modular EPS. Rated for 200-2500W, this PMAD unit can be configured to handle the output from the solar arrays. This unit has four primary functions. First, it tracks the output from the solar array, extracting maximum power output regardless of the angle of the Sun. Second, it regulates the charging of the base station's battery, monitoring the battery's conditions to protect it from overcharging or overheating. Third, it regulates its power output using a hybrid voltage bus, converting the primary 28V output down into 12V, 5V, and 3.3V secondary rails capable of powering low power instruments. Lastly, it contains a built-in fault protection system and power mode switching using its configurable number of latching current limiters (LCL). Each channel can be routed to a specific subassembly, ensuring that each load is electrically isolated from every other. Should the current exceed its configured threshold, the individual LCL will trip, isolating the fault and protecting other systems until the fault can be restored. The LCL channels can be grouped into tiers based on each subsystem's needs. Some channels can be configured to supply critical functions only; useful for times of low power generation during lunar night.

Other channels can support pre-flight operations depending on the configuration. This unit will be used in both the base station and the drone; each containing two for redundancy. The current mass per unit is estimated at 1kg, meaning an extra unit doesn't significantly alter the mass budget of either vehicle. This unit has an extensive flight history, including CubeSats and other LEO missions. Specifically, this unit flew on NASA Goddard's petitSat, earning it a TRL of at least 8.

Battery Subassembly

The selected battery for both the base station and the drone is the EaglePicher NPD-002271 Space-grade Lithium Ion battery, though the size, mass, and capacity of the two batteries is different for each vehicle. For the base station, the battery must be capable of supplying a continuous 29.5W over the maximum dark period of 120 hours. The 29.5W figure is the minimum power required to keep essential thermal systems alive during lunar night. In order to err on the side of caution, the base station battery's capacity will be rated to supply the necessary wattage for 130 hours; 10 more than the longest dark period. This storage reservoir can ease the power demand during transitions between day and night. The selected battery is rated at 153.5 Wh/kg, giving it a mass of roughly 25 kg while being approximately the size of a typical shoebox. For the drone, the average wattage of all other subsystems is expected to be near 75W, with nominal wattage being about 15W and peak loads being about 200W. This gives the battery a mass of 0.49 kg while being the size of a large smartphone. EaglePicher batteries have an extensive flight history, including GPS III and Spirit and Opportunity rovers. This specific model was given a TRL of 7-9 by NASA. For the purpose of this mission, it has been assessed the low end of this estimate due to a limited lunar history.

Drone Charging Interface Subassembly

The charging interface between the drone and the base station consists of the Smiths Interconnect IDI Spring Probe on the base station, and the Smiths Interconnect IDI target contacts. The subassembly's primary purpose is to transfer power from the base station onto the drone with minimal input and complexity after each flight. The drone will land on spring-loaded contact pads attached to the base station's landing area, where it will mate passively with gold-plated beryllium copper pads on the drone's landing gear after touchdown. This architecture has proven effective on past missions, such as NASA's Astrobee mission, where small robots autonomously docked to the ISS for charging. Smiths

Interconnect has an extensive flight history of various NASA missions, but the lack of a direct lunar analog earns these units a TRL of 8.

2.1.2.3. Power Subsystem Recovery and Redundancy Plans

Redundancy

The power subsystem exhibits various layers of redundancy, most notably the inclusion of the Ibeos Modular EPS on each vehicle. The unit is designed to monitor the flow of power through each Latching Current Limiter (LCL), and it contains the ability to switch off within a few milliseconds of a catastrophic failure, keeping the flow of energy to the other latches active. This unit acts as a passive form of redundancy rather than a new integrated design, maintaining a high TRL and reducing the EPS mass budget in the process.

The base station battery is designed to supply power for up to 10 extra hours than the longest dark period. This extra supply will help offset lowered efficiency caused by battery degradation over the course of the mission. In addition to this, the total mission duration of approximately 95 days as well as the sunlight exposure at the selected landing site ensures that the mission will likely only encounter a single dark period during its active lifespan, making the battery storage sufficient for the expected power budget.

The drone charging interface was designed to be a passive mechanism with minimal human intervention required to begin charging. Therefore, the design itself is resistant to critical faults and failures and is expected to outlast the mission's lifespan.

Recovery

The LCLs within each Ibeos Modular EPS unit act as a fault detection and recovery mechanism. The EPS unit can isolate a fault and trip that particular channel before any other channels can be affected. The channel will remain offline until a fix can be found. The LCLs can also be organized in such a way as to include specific power modes for different phases of the mission. Power modes include a hibernation mode for lunar night that can keep only essential systems online to conserve battery life while protecting sensitive instruments, as well as a safe mode after a fault is detected and isolated.

2.1.2.4. Power Subsystem Manufacturing and Procurement Plans

The mission procurement approach prioritizes commercially available and space-qualified COTS/SQCOTS hardware whenever suitable products exist, rather than developing components in-house. Commercial suppliers were selected for subsystems where existing qualified components meet mission needs, reduces scheduling delays, and provides flight heritage. In-house manufacturing is used for mission-specific integration hardware, mounting structures, and harness routing where commercial components must be customized to fit the base station, scout drone, or docking configuration.

For the power subsystem, commercial or vendor-configured suppliers are used for the solar array, EPS/PMAD, battery packs, and docking/charging contacts. These components are procured rather than built in-house because they involve specialized space power hardware, battery safety requirements, and electrical interface reliability that are better handled through qualified suppliers. In-house work is limited to subsystem integration, mounting, harnessing, and verification testing.

EPS Component Procurement Details

Solar Array

The primary solar array supplier is DHV Technology, using the DHV DSA-555W deployable solar array as the selected power generation component. The DSA-555W was selected because it provides 556 W Peak BOL power generation while maintaining a more mass-conscious baseline than the 1268 W growth option. The solar array supports the current base station average power generation requirement of approximately 400–600 W and provides a deployable architecture compatible with the vehicle CAD allocation.

The solar array is procured as a vendor-configured SQCOTS component. The vehicle CAD model represents the DSA-555W as a triple-deployable wing with three panel sections. Based on the available DSA-555W reference image, the deployed wing is approximately 2416.44 mm × 850 mm, with each panel section approximately 762.51 mm × 850 mm. The array is stowed on the top deck of the base station and deployed outward during surface operations. The CAD model also accounts for hinge placement, deployment sweep, mounting clearance, and harness routing. Airbus Sparkwing is the backup supplier for solar array procurement. If the backup supplier is used, the solar array electrical interface, mounting interface, and deployment envelope will be verified during interface control and integration testing.

PMAD / Power Regulation and Control

The primary PMAD supplier is Ibeos, using the Ibeos Modular EPS as the selected power management, distribution, and regulation component. The Ibeos Modular EPS combines power distribution, voltage regulation, solar input control, battery interface, telemetry, and fault protection into one subsystem. This supports the selected hybrid voltage bus architecture and reduces the need for separate PMAD and regulation/control hardware.

Argotec ZEUS PCDU is the designated backup supplier. Argotec was retained as the backup because ZEUS provides modular power conversion and distribution, battery rail support, CAN/UART interfaces, overcurrent protection, and FDIR capability. If the backup supplier is used, EPS rail allocation, voltage compatibility, and C&DH command/telemetry interfaces will be verified during interface control and integration testing.

Battery Packs

The primary battery supplier is EaglePicher, using the EaglePicher NPD-002271 Li-ion battery family for the base station and scout drone battery selections. The base station battery configuration supports survival-mode operation, thermal loads, and drone recharge support. The scout drone uses a lower-capacity customized configuration sized for onboard flight, payload, and recharge requirements. Selected as a space-grade Li-ion battery option for base station energy storage, survival-mode operation, and support for drone recharge. The base station battery is sized to support a 29.5 W survival load for 130 hours, corresponding to a minimum required storage capacity of approximately 3.84 kWh.

Saft is the designated backup supplier for battery procurement contingency. Saft is retained as the backup because it provides space-qualified Li-ion battery products with flight heritage. If the backup supplier is used, battery capacity, discharge rate, charge control, thermal interface, and mechanical mounting will be verified during interface control and integration testing.

Docking / Charging Electrical Interface

The docking/charging interface uses Smiths Interconnect IDI spring probes and target contacts. These components provide a compact electrical contact solution for transferring power between the base station and scout drone after docking. The spring probe and target contacts are procured as COTS/SQCOTS electrical interface hardware.

Mill-Max is the designated backup supplier for the docking/charging contact interface. Mill-Max is retained as the backup because it provides commercial spring-loaded contact and interconnect products suitable for compact electrical interfaces. If the backup supplier

is used, contact resistance, current rating, docking alignment tolerance, and mechanical interface will be verified during integration testing.

Lead Time Summary

The Power subsystem procurement schedule is driven by the longest-lead vendor-configured items: the DHV solar array, Ibeos Modular EPS, and EaglePicher battery packs. The overall Power subsystem procurement timeline is estimated at approximately 4-5 months, with 3-4 weeks for assembly, 4-6 weeks for subsystem testing, and 3-5 weeks for integration with the rest of the spacecraft. If a backup supplier is required, more time may be needed to verify its specific interface.

Subsystem / Component	Procurement Method	Primary Supplier	Backup Supplier	Estimated Lead Time	Justification
<i>DHV DSA-555W Deployable Solar Array</i>	SQCOTS (S... ▾)	DHV Technology	Airbus Sparkwing	12-20 weeks	Selected for deployable solar generation, 556 W Peak BOL, and strong mass efficiency compared with larger arrays. The 1268 W option remains a growth/backup option if final power sizing requires more generation.
<i>Ibeos Modular EPS</i>	SQCOTS (S... ▾)	Ibeos	Argotec ZEUS	10-16 weeks	Selected to combine PMAD, power regulation/control, battery interface, telemetry, and fault protection into one scalable EPS architecture.
<i>EaglePicher NPD-002271 Base Station Battery</i>	SQCOTS (S... ▾)	EaglePicher	Saft	10-18 weeks	Selected as a space-grade Li-ion battery option for base station energy storage, survival-mode operation, and support for drone recharge.
<i>EaglePicher NPD-002271 Drone Battery</i>	SQCOTS (S... ▾)	EaglePicher	Saft	10-18 weeks	Selected as a custom lower-mass Li-ion variant for scout drone onboard energy storage.

<i>Smiths Interconnect IDI Spring Probe</i>	COTS (Com... ▾)	Smiths Interconnect	Milli-Max	6-10 weeks	Selected for the docking/charging electrical interface because spring probes provide a compact, passive, and low mass solution.
<i>Smiths Interconnect IDI Target Contacts</i>	COTS (Com... ▾)	Smiths Interconnect	Milli-Max	6-10 weeks	Selected as the mating contact surface for the overall charging interface.

Figure 2.1.2.4.1: Power Procurement Table

2.1.2.5. Power Subsystem Verification Plans

The Power subsystem verification approach is built from the Power subsystem requirements and uses four verification methods: Analysis, Inspection, Test, and Demonstration. Verification demonstrates that each Power requirement has been satisfied by correlating each method back to the specific requirement it verifies. Since the Power subsystem uses mostly COTS/SQCOTS and vendor-configured hardware, including the DHV DSA-555W solar array, Ibeos Modular EPS, EaglePicher Li-ion batteries, and Smiths Interconnect charging contacts, several requirements can be verified through calculations, datasheet review, CAD/interface inspection, and supplier documentation. Requirements involving measurable electrical behavior, such as voltage regulation, battery charge/discharge, telemetry reporting, fault protection, and drone recharge, are verified through testing or demonstration during subsystem and integrated spacecraft testing.

Req #	Requirement Summary	Verification Method	Rationale for Method	Preliminary Verification Plan
<i>PWR-BS-001</i>	Base station stores at least 3.84 kWh for critical survival-mode operation during low-power generation periods.	Analysis ▾	Energy storage capacity is verified through calculation using survival load and required duration.	Verify battery sizing using $29.5 \text{ W} \times 130 \text{ hr} = 3.84 \text{ kWh}$. Success is defined as the selected base station battery configuration meeting or exceeding 3.84 kWh energy storage.
<i>PWR-BS-002</i>	Base station charging interface	Test ▾	Protection behavior must be confirmed	Conduct controlled overcurrent, overvoltage, and undervoltage

	protects against overcurrent, overvoltage, undervoltage, and battery abuse conditions.		under electrical fault conditions.	fault testing through EPS/BMS charging interface. Success is defined as the system isolating or limiting the fault without damaging the drone battery or charging hardware.
<i>PWR-BS-003</i>	Base station power subsystem monitors and reports system status and health at regular intervals.	Test ▾	Telemetry reporting must be verified through measured EPS/C&DH communication.	During integrated testing, verify reporting of voltage, current, battery state of charge, temperature, and power fault status. Success is defined as power health telemetry being received by C&DH at the required update interval.
<i>PWR-BS-004</i>	Base station supports nominal, charging, low-power, and safe power modes.	Analysis ▾	Power modes are system-level behaviors that must be shown through command and response.	Demonstrate transitions between nominal, charging, low-power, and safe modes using C&DH commands. Success is defined as correct load switching, telemetry reporting, and retention of critical loads in low-power/safe mode.
<i>PWR-BS-005</i>	Base station distributes regulated electrical power to all required subsystems.	Test ▾	Regulated power distribution must be measured under representative subsystem loads.	Bench test EPS output rails using representative loads subsystems. Success is defined as all required voltage rails remaining within acceptable limits during nominal and peak cases.
<i>PWR-BS-006</i>	Base station provides up to 200 W to the payload subsystem during nominal science operations.	Analysis ▾	Payload allocation is verified by comparing the payload power budget against EPS output capability and operating modes.	Review payload power budget and EPS rail allocation. Success is defined as the payload maximum planned draw remaining at or below 200 W during nominal science mode.
<i>PWR-BS-007</i>	Base station provides up to 40 W to the EE subsystem for	Analysis ▾	EE allocation is verified through power-budget comparison and EPS configuration review.	Compare EE subsystem estimated draw against the 40 W allocation. Success is defined as EE loads remaining within the

	EPS/PMAD overhead, control electronics, power monitoring, and secondary electronics.			allocated power level during applicable modes.
<i>PWR-BS-008</i>	The base station provides up to 100 W to the mechanical subsystem for deployment, docking, and mechanism operations.	Analysis ▾	Mechanical loads are dependent on modes and verified through budget allocation.	Compare deployment, docking, and mechanism load cases against the 100 W allocation. Success is defined as mechanical power demand remaining within the allocation during commanded mechanism operations.
<i>PWR-BS-009</i>	Base station provides up to 345 W to the thermal subsystem during peak heater operation.	Analysis ▾	Thermal allocation is verified by comparing heater and thermal-control loads against the power budget.	Analyze thermal subsystem peak heater demand against the 345 W allocation. Success is defined as thermal peak demand remaining within the allocated value or being controlled through duty cycling.
<i>PWR-BS-010</i>	Base station provides up to 70 W to the CDH subsystem during peak processing, monitoring, and communication support.	Analysis ▾	CDH allocation is verified by comparing CDH peak draw against allocated EPS power.	Compare CDH peak power demand against the 70 W allocation. Success is defined as CDH loads remaining within the allocation during peak processing and communication-support modes.
<i>PWR-BS-011</i>	Base station provides sufficient power to thermal-control components, including a 29.5 W survival-mode thermal load, to	Analysis ▾	Thermal survival support is verified by calculating survival-mode energy and confirming EPS/battery support.	Verify that survival-mode loads are powered from the base station battery for 130 hours. Success is defined as the battery supporting the 29.5 W critical thermal load through the required survival duration.

	maintain subsystem operating temperatures.			
<i>PWR-BS-012</i>	Power subsystem subassemblies meet or exceed TRL 6.	Analysis ▾	TRL is verified through supplier documentation, NASA tables, and component maturity review.	Review documented TRL values for the DHV solar array, Ibeos EPS, EaglePicher batteries, and Smiths Interconnect contacts. Success is defined as each selected power subassembly meeting or exceeding TRL 6.
<i>PWR-DR-013</i>	Scout drone onboard flight computer completes a power-system pre-flight checklist before each flight.	Test ▾	Checklist completion must be verified using actual command and telemetry behavior.	Test the pre-flight checklist sequence before simulated deployment. Success is defined as confirmed battery SOC, voltage, current, temperature, and fault status before release.
<i>PWR-DR-014</i>	Scout drone power subsystem supports onboard thermal-control components within the lunar surface thermal environment.	Analysis ▾	Drone thermal power support is verified by comparing thermal-control demand against the drone battery and power budget.	Drone thermal power support is verified by comparing thermal-control demand against the drone battery and power budget.
<i>PWR-DR-015</i>	Scout drone maintains at least 80% battery state of charge before deployment from the base station.	Analysis ▾	SOC threshold is verified through battery energy budget and deployment-readiness rules.	Verify pre-flight battery telemetry before each release. Success is defined as deployment being permitted only when drone battery SOC is greater than or equal to 80%.
<i>PWR-DR-016</i>	Scout drone enters low-power mode when battery state of charge falls below 30%.	Demonstra... ▾	Low-power transition is an autonomous behavior that must be demonstrated.	During integrated testing, simulate or discharge the drone battery below 30% SOC and verify automatic transition into low-power mode. Success is defined as nonessential loads being shed while

				communications and recovery functions remain powered.
PWR-DR-017	Scout drone maintains reserve energy to return to the base station using the 75 Wh battery capacity and mission-defined flight profile.	Analysis ▾	Return reserve is verified by comparing the flight energy model against available battery capacity.	Analyze outbound, science, return, and reserve energy demands using the drone flight power profile. Success is defined as the energy model preserving sufficient reserve for return-to-base operations.
PWR-DR-018	Scout drone reports battery SOC, voltage, current, temperature, and power fault status to the base station.	Test ▾	Telemetry reporting must be confirmed through active communication between drone and base station.	Test drone-to-base-station power telemetry during simulated drone conditions. Success is defined as the base station receiving all required power health parameters without loss of critical telemetry.
PWR-DR-019	Scout drone recharges to at least 80% battery state of charge through the base station charging interface.	Demonstra... ▾	Recharge requires both measurable charge transfer and end-to-end docking/contact behavior.	Demonstrate charging through Smiths Interconnect spring probes and target contacts. Success is defined as stable electrical contact and recharge to greater than or equal to 80% SOC through the base station interface.

Figure 2.1.4.5.1: Power Verification Matrix Table

Analysis is used for requirements that can be verified through calculations, power-budget comparison, energy-storage sizing, or review of documented component capability. This method is utilized for requirements such as the 3.84 kWh base station survival energy requirement, subsystem power allocations, TRL confirmation, and drone return-energy reserve. Inspection is used when verification depends on reviewing physical configurations, CAD layout, harness routing, connector placement, supplier documentation, or interface compatibility. Test is used for measurable electrical performance, including EPS voltage regulation, battery charge/discharge behavior, charging-interface protection, telemetry reporting, and power-transfer performance. Demonstration is used for behaviors that

require the integrated system to respond correctly, including transitions between power-modes, autonomous low-power mode entry, and base-station-to-drone recharge through the docking interface.

Power subsystem verification occurs in stages across subsystem testing and integrated spacecraft testing. Early verification will focus on analysis and inspection of power budgets, battery sizing, TRL documentation, CAD packaging, solar array deployment geometry, EPS rail allocation, harness routing, and interface compatibility. Subsystem-level testing will verify regulated EPS outputs, load switching, battery charge/discharge performance, overcurrent/overvoltage/undervoltage protection, power health telemetry, and charging-contact continuity. Integrated testing will verify C&DH-commanded power modes, survival-mode load shedding, drone pre-flight power checks, drone low-power mode entry below 30% SOC, and recharge to at least 80% SOC through the base station charging interface. Success criteria include meeting the 3.84 kWh survival-energy requirement, maintaining stable regulated voltage rails under representative loads, protecting the charging interface during fault cases, reporting required power telemetry, and demonstrating correct operation of nominal, charging, low-power, and safe modes.

2.1.3. CDH Subsystem Overview

The Command and Data Handling (CDH) subsystem manages all intra- and inter-communications, software execution, fault management operations, and data telemetry for Team 24's lunar mission. Given the dual-platform architecture of this mission, requirements are categorized as stationary lunar Base Station (BS) and the mobile Drone (DR) to ensure a clear distinction for their respective power and mass constraints. This subsystem integrates five subassemblies: Flight Computer (OBCs), Data Storage, Internal Communication, Telecommunication, and Software Architecture. The CDH subsystem works together to ensure the mission's operational phases such as health verification during Checkout, instrument calibration and drone system validation during Calibration, data telemetry and hazard-avoidance support during PSR excursions, and to low-power health and idle monitoring during lunar-nights.

The mission utilizes an assorted computing architecture. The drone is equipped with dual PolarFire SoCs (System on a Chip) arranged in an Active/Cold-Spare configuration, each running a 667 MHz multi-core APU and 667 MHz RPU. The PolarFire provides enough multi-core performance necessary for autonomous flight control loops whilst only drawing 2.5W nominally. Additionally, during each excursion, the Drone OBC utilizes live data from the Space-Grade Gyro and IMU Suite to monitor acceleration, vibration spectra, system stability, using this state data to inform propulsion commands and maintain stable flight as the drone transitions between illuminated terrain and PSR darkness. The Base Station utilizes a Xiphos Q8 Zynq UltraScale+, running a 1.2 GHz quad-core APU and 600 MHz dual-core RPU, to manage heavy telemetry formatting from the lunar drone alongside DTE downlinks, uplinks, and various multitasking procedures. Additionally, to ensure failsafe measures, the low-power Vorago VA41630 operates continuously to monitor baseline system health whilst the Xiphos Q8 idles to conserve power.

To support high-resolution topographical mapping and mission instrumentation tools, both platforms utilize Space-Graded NVMe SSDs, each capable of sustained write speeds exceeding 3,000 MB/s. The Drone uses a single NVMe drive strictly for excursion logging, paired with a Memory Card for low-power nominal telemetry, acting as a temporary failsafe, though also capable of write speeds exceeding 3,000 MB/s. When local link bandwidth is hindered/limited during an excursion, the CDH subsystem prioritizes transmitting fault and health logs over science payload data to the Base Station, protecting subsystem awareness and drone safety ahead of return. The Base Station pools this data into dual NVMe drives configured in a RAID 1 array with a combined 8TB usable capacity, ensuring complete data redundancy prior to DTE (Direct-to-Earth) communication.

The internal communication between the mission's two platforms is facilitated by a UHF architecture, operating on the CCSDS Proximity-1 protocol. This local link allows the Base Station OBC (Onboard Computer) to relay Earth originated commands to the drone. Likewise, it provides the bandwidth necessary for the drone to transmit scientific data collected to the Base Station's NVMe storage array after a PSR excursion. SpaceWire, operating at a 200 Mbps line rate, serves as the instrument, sensor, and telemetry bus between instruments and subsystems in both the Base Station and Drone. The telemetry bus architecture routes data from the payload subsystem's imaging and environmental instruments alongside system command/state data to and from the drone's GNC and propulsion subsystems.

The lunar Base Station is solely responsible for all Direct-to-Earth (DTE) communications, completely offloading the mass and power constraints of deep space transmissions from the drone system. This subassembly, consisting of X-Band architecture, utilizes an Iris Version 2.0 Transponder paired with a JHU/APL Frontier Solid State Power Amplifier (SSPA) and a high-gain X-Band patch antenna, which operates at data rates around 50 Mbps across the maximum Earth-Moon distance (~406,700 km). The Iris V2 manages baseband processing and RF generation, while the SSPA amplifies the signal between the transponder and the antenna. The subassembly is likewise capable of transmitting the scientific instrumentation data directly to Earth's Deep Space Network (DSN) during scheduled downlink windows.

The mission utilizes RTEMS (Real Time Executive for Multiprocessor Systems), an open-source operating system with support for C/C++ and other languages. RTEMS natively supports multi-core architectures and provides FDIR process isolation and open-source availability. The chosen flight software is NASA's cFS (Core Flight System) for its extensive flight heritage and lightweight software load whilst providing necessary flight loops and coverage.

Subassembly	Mass (kg)	Dimensions (m)	Maximum Power Draw (W)	TRL
DR: Primary OBC (PolarFire SoC)	~0.06	~0.08638 × 0.05461 × 0.0188	5.0 W	6
DR: Failsafe OBC (PolarFire SoC)	~0.06	~0.08638 × 0.05461 × 0.0188	5.0 W	6
DR: Primary Storage (Exascend NVMe SSD)	0.01	0.080 × 0.022 × 0.004	7.5 W	5

DR: Sec. Storage (Prograde Digital cFexpress)	0.001	$0.0385 \times 0.0298 \times 0.0038$	1.25 W	6
DR: UHF Transceiver (TRX-U)	0.125	$0.083 \times 0.057 \times 0.016$	8.0 W	6
DR: Quadrifilar Helix Antenna	~0.60	$\sim 0.127 \times 0.127 \times 0.2159$	0 W (passive)	6
BS: Master OBC (Xiphos Q8)	0.064	$0.086 \times 0.080 \times 0.0112$	25.0 W	6
BS: Failsafe OBC (Vorago VA41630)	0.10	$\sim 0.08001 \times 0.08382 \times 0.01$	0.25 W	6
BS: Storage (2x NVMe RAID 1)	0.020	$2 \times (0.080 \times 0.022 \times 0.0035)$	15.0 W	5
BS: UHF Transceiver (TRX-U)	0.125	$0.083 \times 0.057 \times 0.016$	8.0 W	6
BS: Iris V2 X-Band Transponder	0.875	$0.101 \times 0.101 \times 0.056$	33.6 W	6
BS: JHU/APL Frontier SSPA	1.1	$0.184 \times 0.128 \times 0.050$	14.0 W	6
BS: IQ Spacecom X-Band Patch Antenna	0.040	$0.10 \times 0.10 \times 0.0018$	0 W (passive)	6
BS: UHF Antenna (Turnstile)	~0.065	$0.224 \times 0.222 \times 0.0077$	0 W (passive)	6
Total Subsystem	~3.245 kg	6,128.61 cm ³	122.6 W (Max both platforms) Max DR: 26.75 W Max BS: 95.85 W	5

Figure 2.1.3.0.1 : CDH Subsystem Summary Table

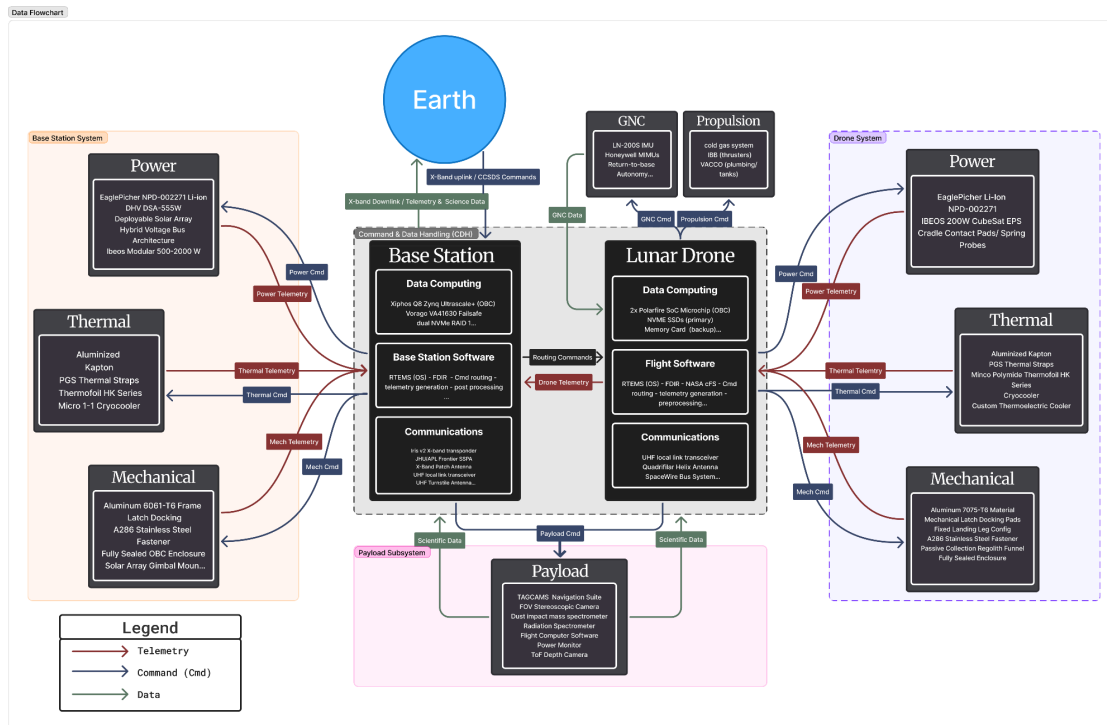


Figure 2.1.3.0.2 : Software Architecture Flowchart

The data flowchart for Team 24's mission depicts a dual system architecture consisting of a stationary Base Station and an autonomous Lunar Drone operating at the lunar south pole. The flowchart illustrates command and data paths across four layers: the X-Band DTE communication link with Earth, the Base Station CDH hub, the lunar Drone CDH hub, and the Payload Subsystem. Within the CDH subsystem, functionality is organized into four groupings: Data Computing (the OBCs and onboard storage), Software Execution (RTEMS, FDIR, NASA's cFS), Internal Communication, and Telecommunications (DTE and local-link hardware). Together these manage the collection, processing, prioritization, and transmission of all command and telemetry data across the system. Additionally, subsystems including Power, Thermal, and Mechanical are shown with bidirectional command and telemetry connections with their respective CDH subsystems. GNC and Propulsion, which are managed directly by the Drone's flight computer, receive commands from and return data solely to the Drone CDH subsystem. All communication between the Base Station and Earth is conducted without orbital relay assistance and remains primarily autonomous.

Prior to each scheduled window, flight controllers prepare command sequences that are uplinked via the X-Band carrier and received by the Iris V2 transponder, which demodulates the signal and passes command packets to the Base Station OBC. The Iris additionally handles two way ranging, generating frequencies used by the DSN for surface position verification. Validated commands enter cFS, where they are checked for integrity and routing source. Fault recovery executes immediately, whilst science and configuration updates are queued through Stored Command. Drone specified commands are transmitted via UHF, while remaining commands are distributed internally within the Base Station via the SpaceWire bus system.

The Drone's cFS validates and routes commands to the appropriate applications. Payload instruments, including the TAGCAMS Optical Navigation Suite and Stereoscopic Terrain Camera for terrain and albedo mapping, the ToF Depth Camera for obstacle detection, and the Dust Impact Mass Spectrometer for regolith and impact depiction, stream data via SpaceWire to the Drone OBC and stored on the primary NVMe SSD. Additionally, the Space-Grade Gyro and IMU Suite, though procured as a Payload instrument, is primarily utilized by the GNC subsystem, fusing IMU and terrain camera data to maintain real time in relation to the Base Station. Additionally, IMU and camera data is utilized for preventing crash detection, and thus system stability, primarily for the Drone system. This state estimate informs propulsion commands controlling thrust trajectories during PSR excursions.

Instrumentation data is temporarily stored on the Drone's primary NVMe SSD until confirmed transmission to the Base Station is achieved, with the Base Station's dual NVMe drives configured in a RAID 1 array, providing redundancy against single drive failure prior to DTE communication. As a Drone side failsafe, if the NVMe experiences difficulties/failures, a Memory Card is used for continued data storage, though at a reduced data rate for local link transmission. Critical telemetry, such as fault logs, power levels, and system health, is assigned transmission priority ahead of science payload data to prevent overwriting during extended excursions where downlink opportunities may be constrained.

Beyond GNC and Propulsion, the Base Station and Drone CDH subsystems also maintain telemetry exchange with respective Power and Thermal subsystems. For the power subsystem, Battery state, charge status, and draw across active instruments, is reported to the OBC, which can issue power commands to reallocate load or transition between operational modes during PSR excursions and idle cycles. Thermal telemetry from sensors on either Drone or Base Station platforms informs active heater and MLI cycling commands, critical during PSR excursions where external temperatures may reach -230

degrees Celsius, causing internal components to potentially fall below their minimum operating temperatures. Additionally, for FDIR, sustained deviation from programmed operating ranges is treated by cFS as a fault condition.

The lunar environment presents thermal and radiation exposure hazards that directly affect CDH hardware and software survivability. The Xiphos Q8's 30 krad TID rating, while providing a degree of protection for the Base Station OBC, remains susceptible to cumulative buildup causing mission failure. A radiation hardened enclosure mitigates this exposure whilst keeping temperatures low for operational usage. Aluminized Kapton MLI provides passive thermal isolation across both platforms, supplemented by active thermal cycling commands during PSR excursions. Power conservation is managed through operational modes, with the Vorago VA41630 continuously monitoring baseline system health whilst the Xiphos Q8 idles to reduce prime power draw.

During scheduled DSN contact windows, cFS releases transfer frames to the Iris V2, which modulates the downlink carrier, amplifies it through the JHU/APL Frontier SSPA, and transmits it through the X-Band patch antenna directly to Earth. Received packets are demodulated and routed to the Ground Data System, where science packets and spacecraft health telemetry are monitored and archived. As failsafe, if a contact window is shorter than planned, science data is stored in bulk for the next scheduled cycle.

2.1.3.1. CDH Subsystem Requirements

The CDH subsystem requirements establish the constraints that the Base Station and Drone design must adhere to, and define the essential functionalities needed to fulfill the mission's goal. The subsystem is organized around five subassemblies, Flight Computer (OBCs), Data Storage, Internal Communication, Telecommunication, and Software Architecture, each of which carries its own set of requirements. Because the mission operates a dual platform architecture, requirements are further split between the Base Station (CDH-BS-XXX) and the mobile Drone (CDH-DR-XXX) to account for differing physical characteristics and constraints.

The majority of CDH requirements trace directly to customer constraints set by stakeholders. CDH-001 and CDH-BS-014 establish DTE communication parameters, requiring the Base Station to close the link across the maximum Earth Moon distance of approximately 406,700 km without reliance on orbital relay assets, per SYS-012. By choosing the maximum Earth-Moon distance, the system ensures a stable link budget during mission lifetime and informs the Trade Study selection of the X-Band architecture. CDH-002, CDH-BS-016, and CDH-DR-024 establish the subsystem's data storage and throughput requirements, in relation to the mission's 1 TB minimum data requirement

(SYS-008). This allows the Drone to buffer data locally during excursions while the Base Station retains the 8 TB archive prior to downlink.

Radiation tolerance requirements, CDH-BS-017 and CDH-DR-022, were derived from differing exposure environments of each platform rather than a fixed figure. The Base Station's 30 krad TID requirement reflects its stationary position outside the PSR boundary, while the Drone's 100 krad requirement accounts for its repeated excursions into the PSR itself, where prolonged radiation exposure is more severe. Both figures flow down from the system level shielding requirement, SYS-021, and were validated against the selected hardware's flight heritage during the Onboard Computer trade study. While the Polarfire SoC achieved high radiation tolerance (100 krad), Xiphos Q8+ was chosen mainly for its computational power, not overall radiation tolerance (30 krad).

Power and mass allocations, CDH-010 and CDH-011, were derived from the overall 175 kg system mass constraint (SYS-014) alongside the PSR environment limiting solar power generation. Thus, to confine power consumption, the CDH subsystem must remain within a 45 W average and 100 W peak draw per platform. These constraints directly shaped the selection of the low power PolarFire SoC for the Drone and the Vorago VA41630 failsafe for the Base Station, both of which were weighted for power efficiency during their respective trade studies.

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Req met?
CDH-001	The CDH Subsystem shall transmit and receive data directly to/from Earth ground stations via X-band DTE communication (7.145–7.235 GHz uplink / 8.400–8.500 GHz downlink), without reliance on relay satellites.	Per stakeholders, all communication must be Direct-to-Earth (DTE), and must assume that relay satellites should not be used.	SYS-012	CDH-BS-014, CDH-BS-015	Test	Yes
CDH-002	The CDH subsystem shall receive, collect, and retain data onboard using storage providing at least 2,000 MB/s sustained write speeds.	Required to support high-throughput science-data collection and buffering	SYS-006, SYS-008	CDH-008, CDH-012, CDH-BS-016, CDH-DR-024	Test	Yes
CDH-003	The CDH subsystem shall retain collected science and engineering data onboard, with a data loss rate $\leq 0.1\%$, until confirmation of successful relay is received.	Protects against data loss during DTE communication in the case of hurdles in the lunar south pole and PSRs.	SYS-002, SYS-008, SYS-011	CDH-BS-013, CDH-DR-020	Test	Yes
CDH-004	The CDH subsystem shall incorporate autonomous FDIR loops for detecting and mitigating subsystem malfunctions, acting within ≤ 5 minutes of fault onset.	Directly supports backup/redundancy capabilities	SYS-010, SYS-011	CDH-DR-025, CDH-DR-026	Demonstration	Yes
CDH-005	The CDH subassemblies shall demonstrate direct flight or field heritage in a comparable space, deep space, or lunar surface environment	Lowers engineering risk by prioritizing components with a documented operational history over undemonstrated technology	SYS-007		Inspection	Yes
CDH-006	The CDH subsystem shall sample and log the RSSI for all active Local and DTE communication	Supports requirement to monitor communication signal strength, availability, and data	SYS-012		Test	Yes

	links at a rate of 1 Hz during all operational modes	accuracy				
CDH-007	The CDH subsystem enclosures shall prevent ingress of lunar regolith particles $\leq 20 \mu\text{m}$ throughout full mission duration.	Traces to specific dust and environmental hazard mitigation needed for CDH hardware and electronics	SYS-020	CDH-DR-023	Inspection	Yes
CDH-008	The CDH subsystem shall perform onboard data compression/pre-processing, achieving a minimum data-volume reduction of at least 30%, prior to DTE transmission	Reduces required transmission bandwidth and ETA DTE communication	CDH-002	CDH-BS-018	Analysis	Yes
CDH-009	The CDH subsystem shall provide a systemized internal data interface, supporting a minimum sustained throughput of 100 Mbps, compatible with all instrument/ subsystem data tools.	Ensures all instrument data can be accumulated within a common internal communication architecture	SYS-002, SYS-006	CDH-012	Test	Yes
CDH-010	The CDH subsystem's electrical power draw shall not exceed 45 W average / 100 W peak PER platform	Supports minimizing power consumption due to the PSR environment hindering solar energy conversion.	SYS-014		Analysis	Yes
CDH-011	The collective CDH subsystem shall not exceed a mass allocation of 4 kg each for the Base Station and Drone element	Supports the overall 175kg system mass constraint; specific allocation TBD	SYS-014		Inspection	Yes
CDH-012	The Drone and Base Station OBCs shall provide storage write throughput independent of, and not limited by, the SpaceWire data interface's 200 Mbps throughput ceiling. throughput ceiling.	Resolves the throughput conflict between storage requirement and data interface cap within the subsystem.	CDH-002, CDH-009		Inspection	Yes

Base Station CDH (CDH-BS-XXX)						
CDH-BS-013	The Base Station CDH subassembly shall receive and accumulate data and telemetry from all platform instruments, with a data loss rate $\leq 0.1\%$ under nominal link.	Required for data accumulation at the element responsible for Earth relay.	SYS-002, CDH-003		Test	Yes
CDH-BS-014	The Base Station CDH subsystem shall close the DTE link with a minimum of ≥ 3 dB across the maximum Earth-Moon distance (~406,700 km) during all scheduled downlink windows.	Establishes a specific link budget threshold needed for DTE communication	SYS-012, CDH-001		Analysis	Yes
CDH-BS-015	The Base Station CDH subsystem shall serve as a localized surface network hub, relaying commands, telemetry, and payload data packets to and from the Drone, within ~6km (line-of-sight range).	The base station acts as its line-of-sight communication gateway.	SYS-006, CDH-001		Test	Yes
CDH-BS-016	The CDH Base Station subsystem shall provide onboard data storage of at least 8 TB usable capacity, to retain collected payload data until confirmed Earth downlink is achieved.	Establishes bulk data buffering needs at a primary archive	SYS-008, CDH-002		Inspection	Yes
CDH-BS-017	The CDH Base Station subsystem electronics shall tolerate Total Ionizing Dose (TID) of at least 30 krad over mission lifespan	Flows to specific radiation tolerance needed for onboard electronics including CDH and other subsystems	SYS-021		Test	Yes
CDH-BS-018	The CDH Base Station subsystem shall perform onboard compression/ pre-processing for	Reduces required transmission bandwidth while preserving reserve storage margin.	CDH-008		Analysis	Yes

	instrument data prior to every Earth transmission window, maintaining a minimum reserve of 2 TB (~25%) of total storage capacity at all times.					
Drone CDH (CDH-DR-XXX)						
CDH-DR-019	The Drone CDH subsystem shall process local active sensor data streams to detect obstacles ≥ 10 cm at ranges of 1.00–20.0 m continuously during all PSR excursions.	Supports the mission objective for autonomous hazard avoidance and safe navigation during PSR excursions.	SYS-002, SYS-006		Test	Yes
CDH-DR-020	The Drone subassembly shall retain collected science/engineering data onboard, with a data loss rate $\leq 0.1\%$, until Base Station transmission is confirmed.	Protects against data loss in case the Drone-to-Base-Station link is interrupted	SYS-002, SYS-004, CDH-003		Test	Yes
CDH-DR-021	The Drone CDH subassembly shall receive and accumulate data and telemetry from all Drone instruments, including navigation and science sensors, continuously during operation.	Supports capturing all science and engineering data onboard prior to transmission.	SYS-002, SYS-004		Demonstration	Yes
CDH-DR-022	The Drone CDH electronics shall tolerate a Total Ionizing Dose (TID) of at least 100 krad(Si) over the mission lifetime.	Flows to specific radiation tolerance needed for onboard electronics including CDH and other subsystems	SYS-021		Test	Yes
CDH-DR-023	The Drone CDH subassembly shall prevent ingress of fine, electrically charged lunar particles ≤ 20 μm in diameter into internal	Repeated launch/landing cycles create dust disturbances compared to static systems.	SYS-020, CDH-007		Inspection	Yes

	components during takeoff and landing cycles.					
CDH-DR-024	The Drone CDH subassembly shall provide onboard data storage of at least 1 TB to buffer collected science data during mission cycles	Sized for shorter duration during buffer times, since bulk archiving will occur at the Base Station	SYS-008, CDH-002		Inspection	Yes
CDH-DR-025	The Drone CDH subassembly shall detect and respond to local link loss within ≤ 5 minutes, initiating procedures to restore communication within that window.	Supports backup capabilities in case of system failure or malfunction	SYS-010, CDH-004		Test	Yes
CDH-DR-026	The Drone CDH subsystem shall prioritize transmission of fault/health logs over science payload data whenever local link bandwidth is hindered.	Supports mitigation strategies to ensure subsystem health, preventing mission loss.	SYS-010, CDH-004		Demonstration	Yes

Figure 2.1.3.1.1: CDH Subsystem Requirement

2.1.3.2. CDH Sub-Assembly Overview

The Command and Data Handling (CDH) subsystem acts as the brain for the dual platform architecture of the mission. For the Base Station, the CDH subsystem handles routing commands to the Drone, receiving Drone telemetry such as payload and instrumentation data and health logs through the local link, and sending archived data through DTE communication during scheduled downlink windows. For the Drone, the CDH subsystem handles flight software via RTEMS and NASA's cFS to ensure system stability and proper propulsion, data computing, recording payload telemetry, and relaying it to the Base Station. The CDH subsystem is broken down into five subassemblies: Onboard Processing, Data Storage, Internal Communication, Telecommunications, and Software Architecture (flight software and FDIR). This system ensures science objectives are fulfilled throughout the planned mission lifetime.

Onboard Processing

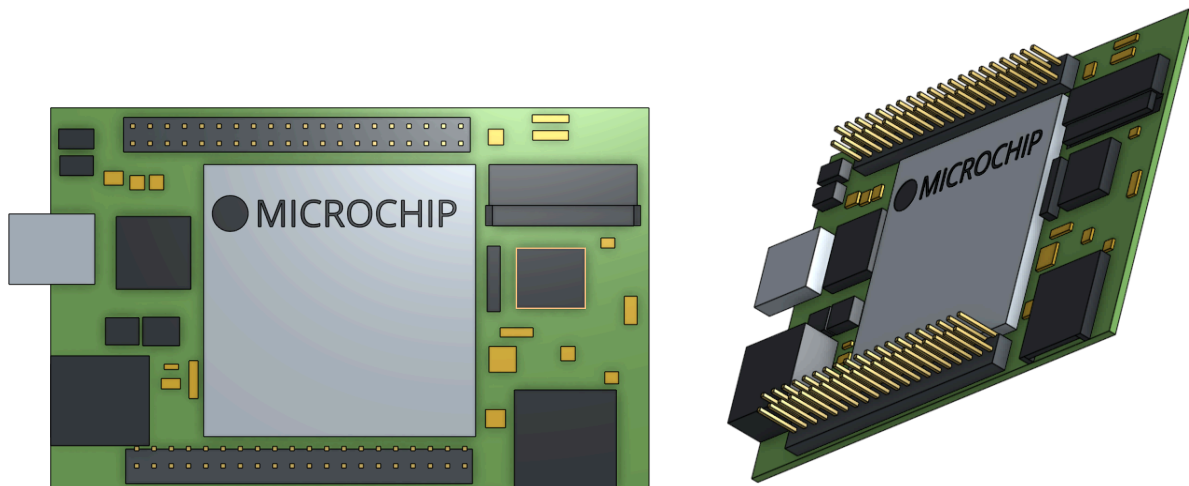


Figure 2.1.3.2.1: CAD Polarfire OBC board

The Flight Computer subassembly provides the processing power for both platforms, executing flight software, managing autonomy routines, and formatting data for archive storage and transmission. The Drone OBC uses a dual PolarFire SoC (RTPF500ZT) configuration in an Active/Cold-Spare arrangement[1]. Each unit integrates a 64-bit multi-core RISC-V Microprocessor Subsystem running a 667 MHz multi-core APU alongside a 667 MHz real-time processor[1]. The SoC draws approximately 2.5W nominal power, scaling to 5.0W peak, and measures a compact $40 \times 40 \times 1.0$ mm at just 52g bare[1], though with a miniaturized board (like the Beagle-V fire), can reach up to ~ 86.38 mm $\times$ 54.61mm $\times$ 18.8mm. As a result, a change of dimensions was accounted for on the Polarfire OBC, the

Vorago OBC, and Xiphos OBC. Going back, the Polarfire SoC is equipped with 4 GB of volatile RAM and 32 GB of non-volatile storage onboard for efficient data and command processing[1]. Its radiation-tolerant material is built on non-volatile SONOS flash switches, immune to configuration Single Event Upsets from heavy ion strikes[2]. Ground testing confirms Total Ionizing Dose tolerance exceeding 100 krad alongside Single Event Latch-Up immunity above 60 to 75 MeV·cm²/mg[2,3]. This radiation and mass profile is what allows the drone to fly two units for redundancy without exceeding platform power and mass constraints.

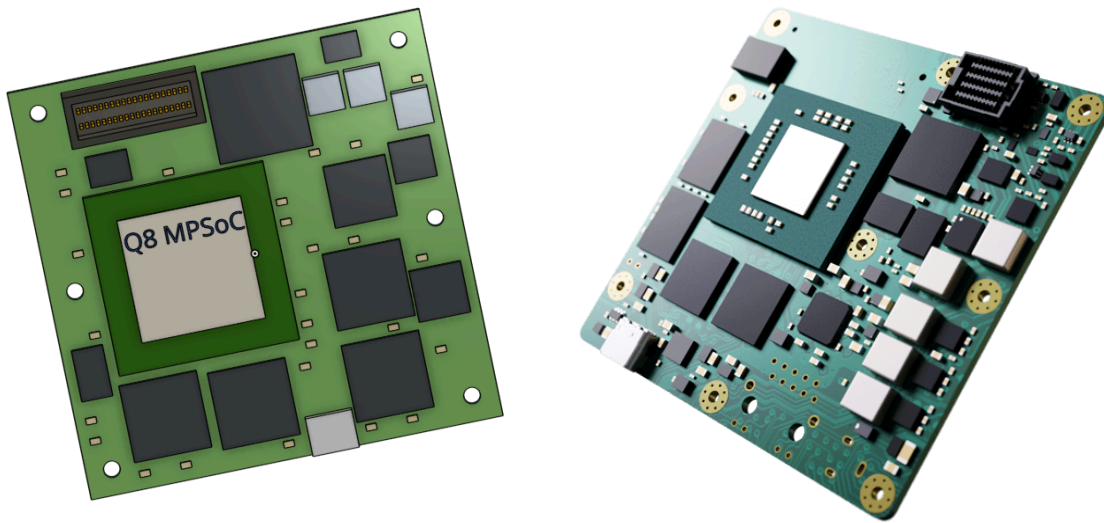


Figure 2.1.3.2.2: Xiphos Q8 Zynq Ultrascale CAD + Physical Design

The Base Station OBC uses a Xiphos Q8 Zynq UltraScale+, built around a quad-core ARM Cortex-A53 Application Processing Unit running at 1.2 GHz alongside a dual-core ARM Cortex-R5 Real-Time Processing Unit[4]. The board carries 4 GB of LPDDR4 RAM with error detection and correction, dual 128 MB QSPI NOR Flash boot modules, and dual 128 GB eMMC data storage drives operating on independent power and data buses, all stored within an 86 mm × 80 mm × 11.2 mm footprint and 64g mass[4]. Radiation mitigation is handled by an integrated Microsemi ProASIC3 control and FPGA providing bitstream scrubbing and overcurrent monitoring[4,5]. Through external testing, the OBC received a Total Ionizing Dose rating in the 25 to 30 krad range[4,5]. This rating is an accepted weakness for the mission, offset by a radiation hardened enclosure alongside using the Vorago VA41630, a 32-bit Arm Cortex-M4 microcontroller running up to 100 MHz, as the continuous failsafe system monitor[6]. While the Vorago VA41630 chip dimensions are 20mm x 20mm x 2.5mm, a procured board from the manufacturer is around 80.01 mm × 83.82 mm × 10 mm[6]. The Vorago is equipped with HARDSIL silicon-level radiation

hardening that pushes its own Total Ionizing Dose tolerance beyond 200 krad(Si), Single Event Latch-Up immunity above 110 MeV·cm²/mg, hardware Error Detection, and Correction scrubbing[6].

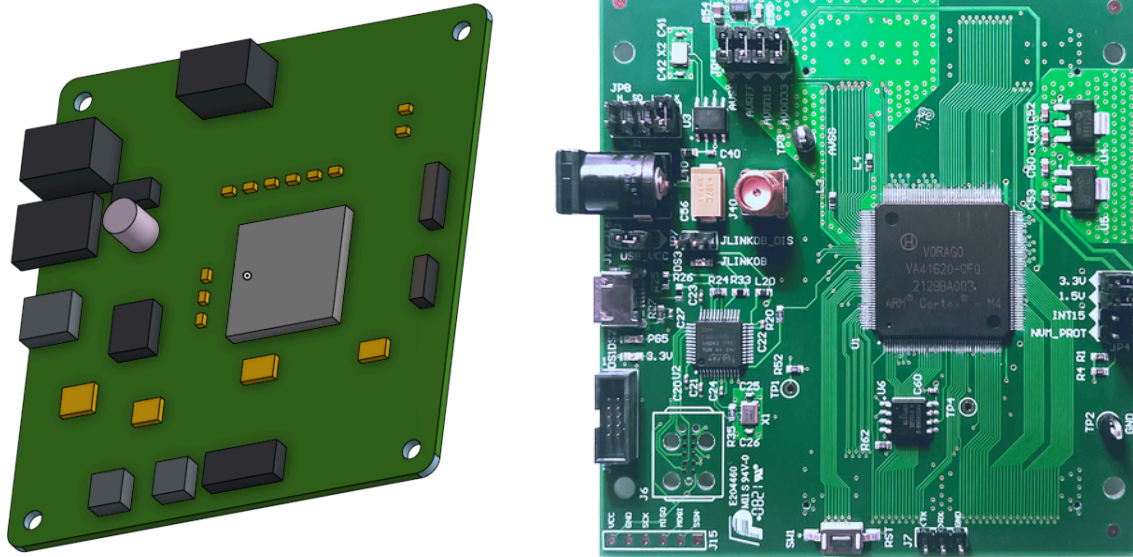


Figure 2.1.3.2.3: Vorago VA41630 CAD + OBC Design

Both OBCs interface with Data Storage and Internal Communications directly; alternatively, Payload, GNC, and Propulsion are interfaced through the SpaceWire data interface architecture. Environmentally, both platforms operate within a -40°C to +100°C hardware rating[1,4], requiring active thermal management to remain functional during PSR excursions where temperatures may approach -230°C.

Data Storage

The Data Storage subassembly retains science and engineering data onboard until scheduled periods. Thus, each platform utilizes different storage capacities to reflect respective roles in the mission's data pipeline. Both platforms use the Exascend PR4-series space-graded NVMe SSD, built around a PCIe Gen4 x4 NVMe 1.4 interface, delivering sustained write speeds of 3,000 MB/s and read speeds between 3,200 and 3,500 MB/s, with random read throughput up to 500,000 IOPS[7,8].

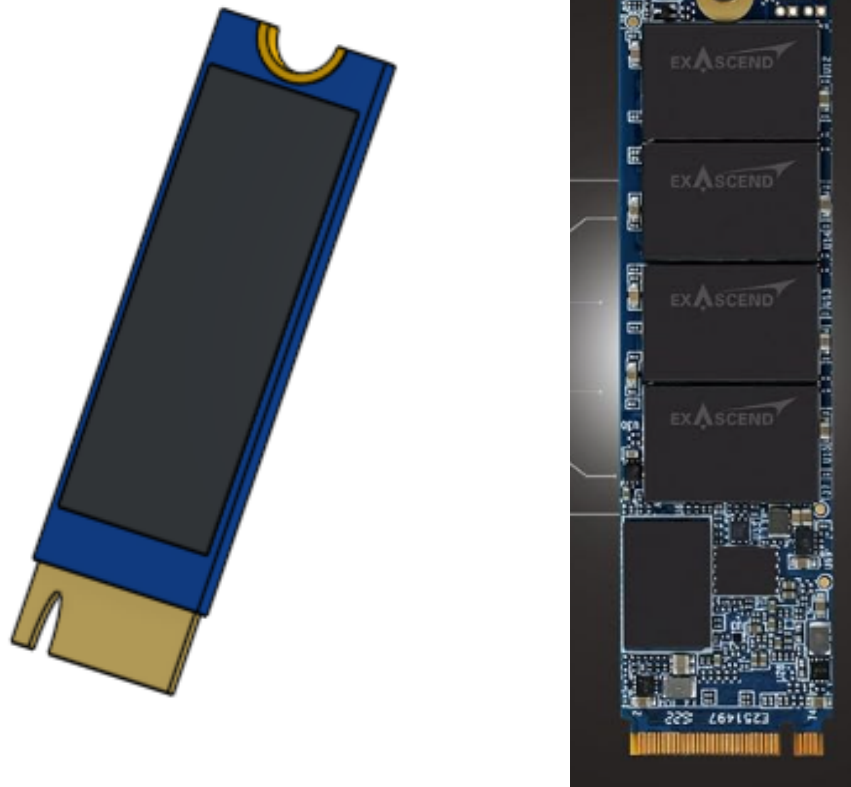


Figure 2.1.3.2.4: Exascend PCIe CAD + Characteristics

The drive's Neutron Shield 2.0 firmware provides dual-copy, auto-reboot hardening alongside radiation-induced bit-flip correction and LDPC error management, giving it a Total Ionizing Dose (TID) tolerance of up to 500 krad through combined hardware/firmware mitigation[7,8]. Physically, the drive draws a peak of 7.5W and idles under 1W[7]. The drive is protected by conformal coating and underfill encapsulation against shock and vibration, alongside carrying dual power-loss-protection to preserve data integrity during power interruptions, a needed feature given the mission's PSR hazardous environments[7].

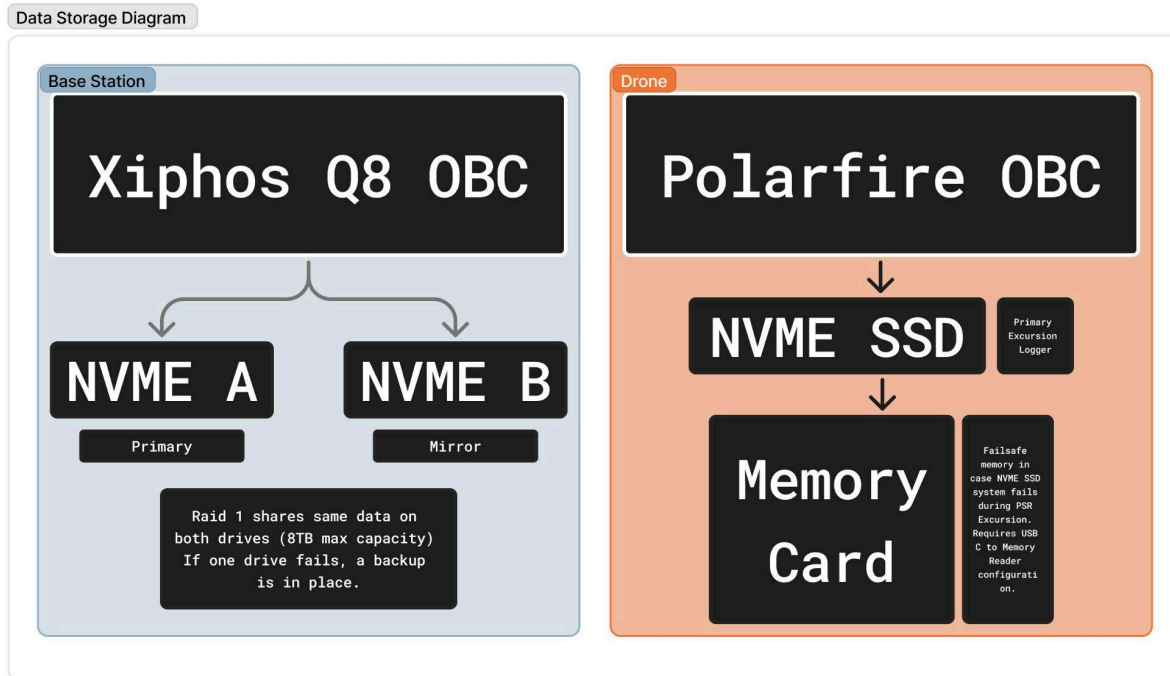


Figure 2.1.3.2.5: Data Storage Subsystem Configuration

The Drone uses a single NVMe drive for excursion logging, paired with a CFexpress Type B memory card retained as a low-power backup, capable of read speeds up to 3,550 MB/s and rated for sustained 4K and 8K-equivalent data capture, ensuring nominal telemetry logging continues even in the event of primary NVMe failure[9,10]. Due to the Polarfire OBC not having a dedicated memory card reader, an external (USB C config) is provided. The Base Station pools all mission data into a RAID 1 array using both NVMe drives, providing complete data redundancy and up to 8 TB of combined capacity[7]. This subassembly interfaces with the OBC directly for read and write operations whilst interfacing with SpaceWire for incoming instrument data streams. Environmentally, the PR4's radiation-hardened firmware and physical ruggedization are specifically what allow it to operate through the radiation and thermal extremes of repeated PSR excursions[7,8].

Internal Communication

The Internal Communications Network routes commands and telemetry between the OBC, instruments, and transceivers on both platforms using SpaceWire, built to the ECSS-E-ST-50-52C standard and delivering a 200 Mbps line rate[11,12]. SpaceWire uses Low-Voltage Differential Signaling to maintain signal integrity and noise immunity against the lunar south pole's radiation and thermal cycling, and its standardized routing switches allow all subsystems, OBC, Data Storage, Telecommunications, Payload, GNC, and

Propulsion, to share a single network architecture rather than requiring heavy point-to-point cabling between every pair of components[11]. This shared-bus approach directly supports the mass constraints of both platforms while giving the subsystem a single, well-characterized interface standard to test and qualify.

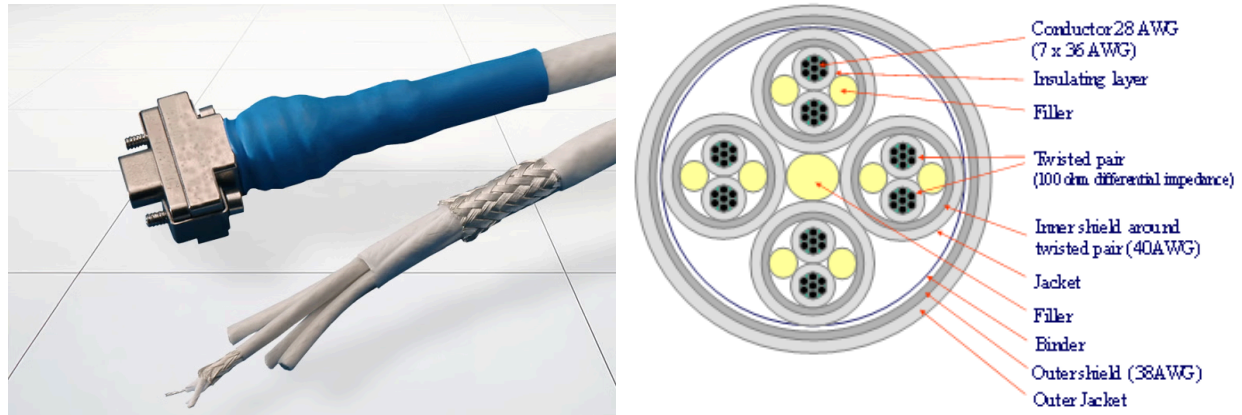


Figure 2.1.3.2.6: SPACEWIRE hardware data Cable

This subassembly interfaces with every other CDH subassembly as well as Power, Thermal, Mechanical, Payload, GNC, and Propulsion, making it the backbone connecting the entire spacecraft's command and data paths.

Telecommunications

The Telecommunications subassembly is split between the Base Station's Direct-to-Earth (DTE) chain and the shared UHF local link connecting the two platforms. This ensures reliable communication from Drone to Base Station and Base Station to Earth with different frequency and operational ranges from the X-Band and UHF architecture.

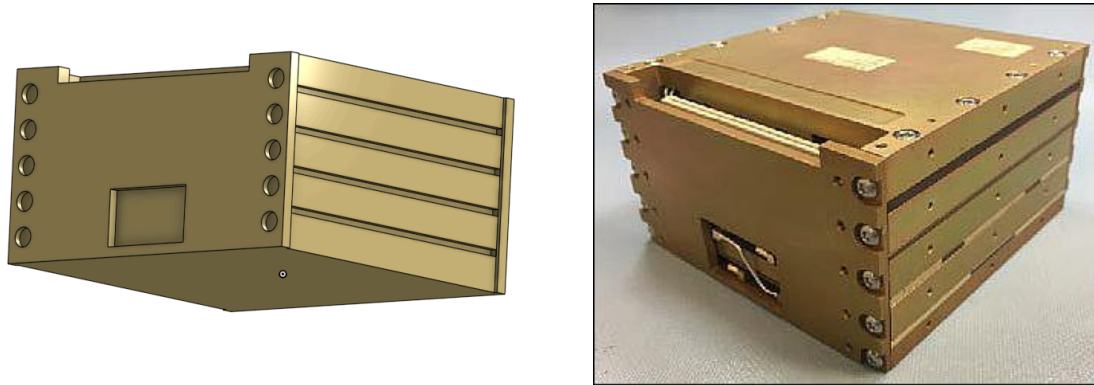


Figure 2.1.3.2.7: Iris V2 CAD + Physical Characteristics

The DTE architecture is built around an Iris V2 X-Band Transponder, a Gaisler LEON3-FT processor-based unit carrying 32 Mbit of non-volatile NOR flash and 4 Mbit of EDAC-protected volatile SRAM, massing 875g within a 101 mm × 101 mm × 56 mm frame[13]. The Iris V2 handles baseband processing, RF generation, and two-way ranging for surface position verification, operating across a 7.145 to 7.235 GHz uplink and 8.400 to 8.500 GHz downlink, drawing 10.3W in receive-only mode and up to 33.6W during simultaneous transmit/ receive[13,14]. The transponder's processing power and EDAC protection ensures stable operational usage during the mission lifetime[13,14].

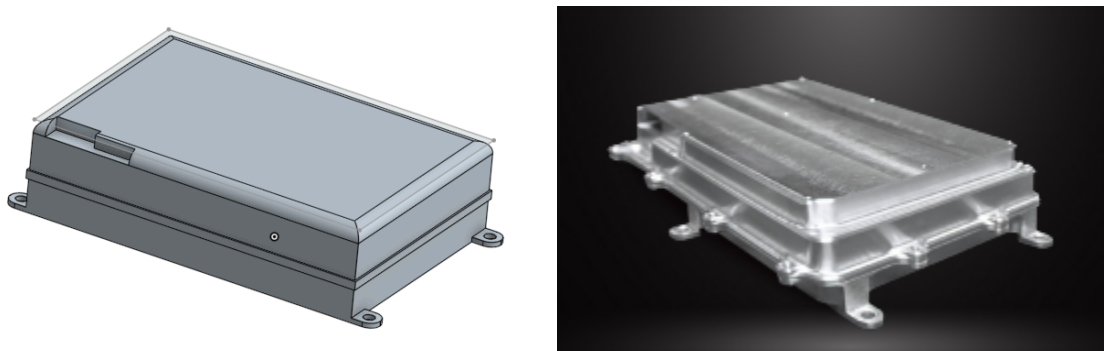


Figure 2.1.3.2.8: JHU/APL SSPA CAD + Physical Characteristics

Its signal is amplified by a JHU/APL Frontier Solid State Power Amplifier, a two-box system built on Gallium Nitride HEMTs across an RF Amplifier Box and a separate Power Converter Unit, delivering an output power of approximately 11.5W and massing under 1.1 kg total. The link rate for the JHU APL SSPA reads a 6 dB rate, meeting the CDH-BS-014 requirement [15,16]. The previously chosen General Dynamics SSPA, after further inspection, was found to have significantly higher power consumption despite providing similar X-Band functionality. The dimensions, system interface, functionality, mass, and operating temperature remain similar, though the JHU/APL Frontier SSPA was procured due to its

significantly lower nominal power consumption of 14W compared to 63W for the General Dynamics SSPA, as well as its use of newer GaN HEMT technology. This change reduces a substantial power demand while maintaining similar overall characteristics and preserves the Base Station's required power margin. [15,81].

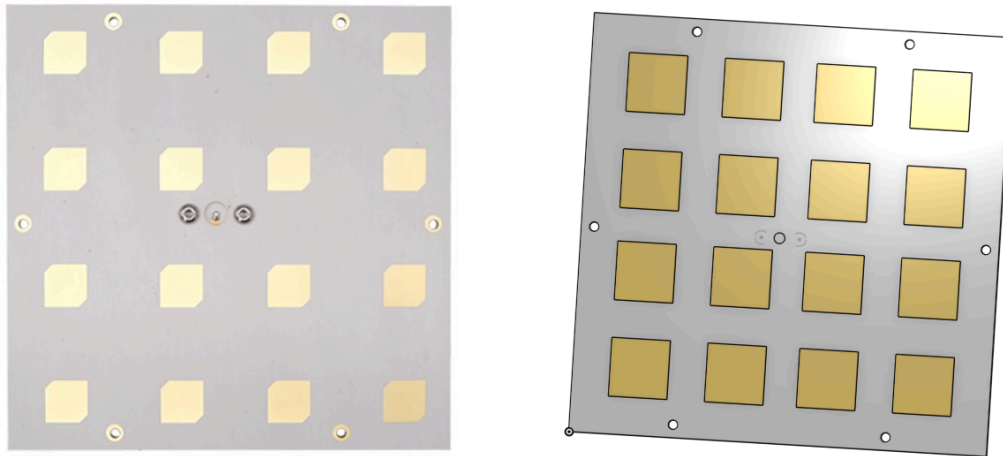


Figure 2.1.3.2.9: X-Band Patch Antenna

The signal is then radiated through an IQ Spacecom X-Band patch antenna, an ultra-small circularly polarized patch built on space-grade Rogers laminate, massing just 40g at 100 mm × 100 mm × 1.8 mm with maximum gain of 16 dBi[17,18]. It is to note that the CDH subsystem had previously accounted for the quad patch antenna with a 60 x 40 x 2 mm dimension and 20g mass[19]. However, the quad patch antenna provided higher dBi gain and beamwidth for slightly elevated size and mass[17,19]. Together this DTE chain transmits at data rates around 50 Mbps, sufficient for the mission's scheduled downlink windows[20].

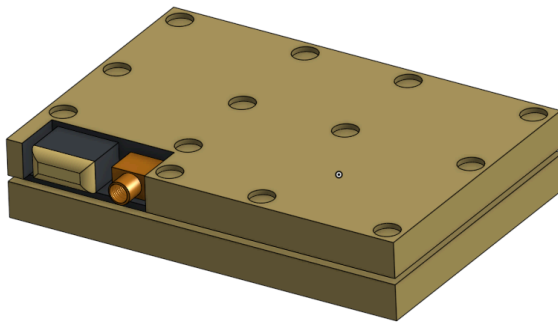


Figure 2.1.3.2.10: TRX-U CAD + Physical Characteristics

The local link interface within both platforms include the AAC SpaceQuest TRX-U transceiver[21]. The component comes with built-in automatic gain and frequency control, and digital RSSI reporting across a -120 to 60 dBm range[21]. Operating across the 390 to 470 MHz UHF band on the CCSDS Proximity-1 protocol at data rates up to 19.2 kbps per channel, the transceiver draws approximately 230 mW in receive mode and up to 8W during transmission, massing 125g in an $83 \times 57 \times 16$ mm housing[21,22]. This supports the store-and-forward model where science data collected during PSR excursions, given transmission link is available, is able to transmit when possible. If not, only priority logs are transmitted and collected data is offloaded to the Base Station during post-excursion recharging windows.

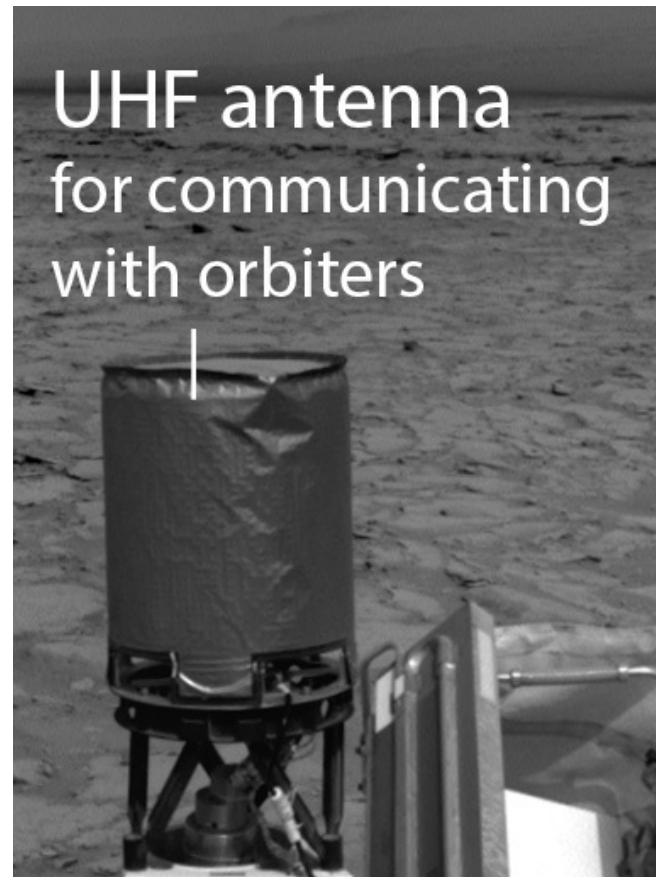
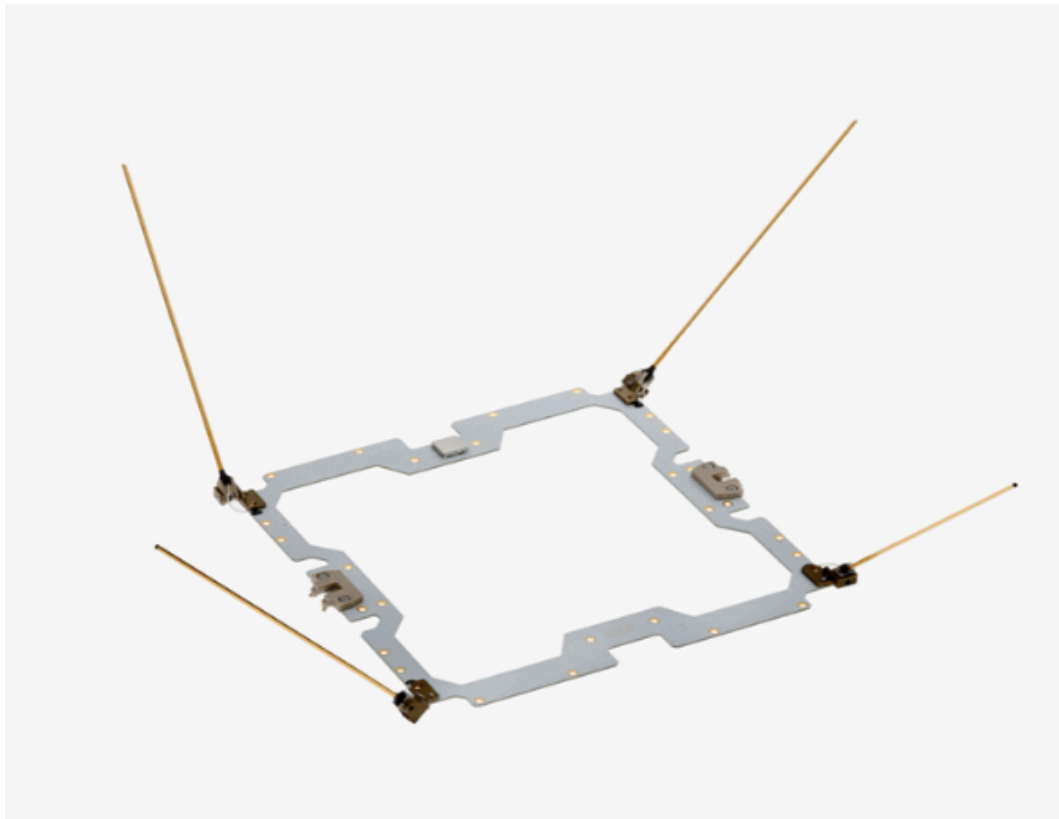


Figure 2.1.3.2.11: Cylindrical Helix Antenna CAD + Design

The Drone's local link antenna was revised during subassembly review from the originally proposed ISISPACE tape-measure antenna to a Cylindrical Quadrifilar Helix Antenna[23]. The ISISPACE unit's flight heritage reflects standard Low Earth Orbit operation only, with no documented lunar surface or PSR specific qualification, a gap identified during the subsystem's flight history and satisfied CDH requirement reassessment[23]. The quadrifilar helix instead draws on flight heritage from Perseverance's Rover UHF communications system, giving it credible flight history in the same UHF band the drone operates in[24,25]. The antenna radiates a circularly polarized signal, delivering up to 13.8 dBic gain in axial directivity mode with an axial ratio below 2 dB, or between 2.23 and 3.8 dBi across a wide 34° to 55° beamwidth in its omnidirectional configuration, remaining immune to polarization mismatch losses as the drone maneuvers during PSR excursions[26]. Additionally, the antenna's hemispherical pattern eliminates the dead zone directly overhead[26]. The drone's antenna is sized at approximately 127 mm in diameter by 216 mm in height, smaller than Perseverance's roughly 300 × 350 mm assembly, a deliberate scaling tradeoff given the drone's substantially tighter mass and stowage constraints[27]. It

is to note, most dimensions are approximations due to Perseverance's classified physical



and descriptive information[27].

Figure 2.1.3.2.12: Turnstile Antenna Antenna Design

The Base Station retains a NanoAvionics Turnstile UHF antenna, a passive quad-monopole phasing array requiring no onboard power, delivering 1.37 to 3.0 dBi of circularly polarized gain across a 400 to 500 MHz band from a 224 x 222 x 7.7 mm, 65g footprint[28]. Its stationary role outside the PSR boundary does not carry the same harsh PSR-specific qualification driving the Drone antenna change. Furthermore, procurement of a more radiation hardened and lunar space-qualified component is possible.

Flight Software Architecture

The Flight Software Architecture governs command validation, routing, and task execution across both platforms, built on RTEMS, an open-source operating system providing hard bounded interrupt latencies and preemptive scheduling mandatory for stabilizing autonomous drone control loops. RTEMS integrates natively with NASA's Core Flight System, and its verified Board Support Packages support both the Zynq UltraScale+ and

PolarFire SoC multi-core architectures used for both platforms. cFS runs atop RTEMS as the mission's chosen flight software framework, validating and routing all commands, executing fault recovery immediately upon detection, and queuing science and configuration updates through Stored Command for execution during scheduled windows.

Fault Detection, Isolation & Recovery (FDIR)

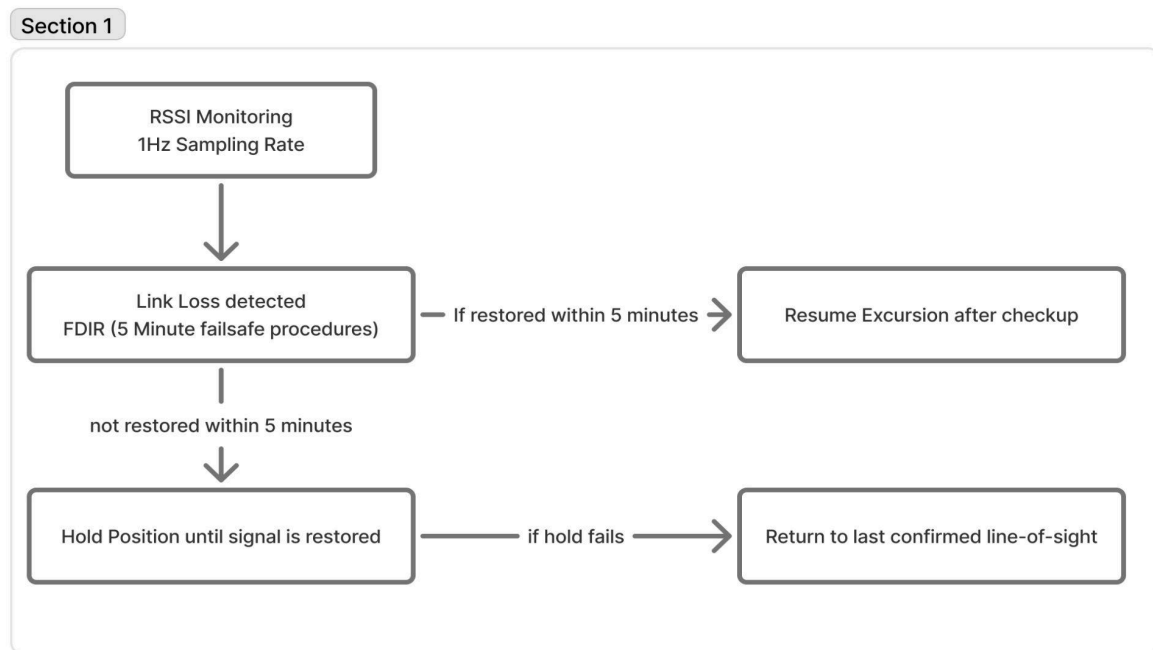


Figure 2.1.3.2.13: FDIR system pipeline

FDIR is not a physical subassembly but rather a capability implemented across the Flight Computer, Data Storage, and Telecommunications subassemblies through RTEMS and cFS. Its purpose is to detect subsystem malfunctions and initiate a mitigating response autonomously, without requiring real-time operator intervention[29]. Key design features include the Drone OBC's Active/Cold-Spare failover, the Base Station's Vorago VA41630 continuous health monitor, automatic power draw cutoff, and RSSI-based local link loss detection triggering a hold-position or return-to-last-contact behavior if the link is not restored within the mandated 5 minute window. FDIR interfaces with every CDH subassembly by design, since its entire function is to monitor and respond to failure across the subsystem.

Technology Readiness Level (TRL)

Vendor-published TRL ratings for CDH hardware largely reflect flight heritage in near-Earth orbit, cislunar space, or general deep space, rather than the lunar south pole's specific combination of PSR and south-pole regolith conditions. A component's TRL for this mission is assessed against its heritage environment's similarity to the lunar south pole, not simply carried over from its highest achieved rating, thus no CDH component is assessed above TRL 6 unless it carries direct flight heritage from a PSR excursion mission specifically.

The PolarFire SoC (Drone Primary and Failsafe OBC) is assessed at TRL 6. It has demonstrated operational flight heritage in Low Earth Orbit aboard the ISS through Skycorp's ISSIFQE payload, executing multi-bus communications and fault-tolerant boot switching, and its underlying silicon is qualified under MIL-STD-883 Class B and QML Class Q standards [2,30]. It has not yet accumulated lunar surface flight heritage, placing it at TRL 6 for this mission. The Exascend PR4 NVMe SSD (Drone Primary Storage and Base Station Storage) is assessed at TRL 5, having passed third-party ESCC 22900 Total Ionizing Dose and ESCC 25100 Single Event Effects laboratory testing, but remaining built primarily for Low Earth Orbit and tactical defense applications rather than deep space or lunar surface profiles [7,8,32]. The CFexpress secondary storage card is assessed at TRL 6, having completed over 18 months of simulated space environment qualification testing at NASA Marshall Space Flight Center and delivered under a formal Space Act Agreement for use with NASA's lunar surface camera systems, though has not yet flown on any Artemis mission [9,10]. The Drone and Base Station UHF Transceivers (AAC SpaceQuest TRX-U) are each assessed at TRL 6. Their nearly two decades of flight heritage across INSPIRESat-1, IDEASSat, and numerous CubeSat constellations, including full qualification-level vibration and thermal vacuum cycling testing, is extensive but confined to Low Earth Orbit CubeSat missions, with no PSR excursion heritage to support a higher rating under this mission's assessment criterion [21,32,33].

The Drone's Cylindrical UHF Relay Antenna is assessed at TRL 6. Its underlying quadrifilar helix architecture carries strong heritage through deep space and planetary surface missions, including a deck-mounted implementation on the Mars Science Laboratory (Curiosity) rover, but its application inside a lunar PSR specifically requires further verification against PSR cold-trap temperatures and against dielectric coupling effects from surrounding lunar regolith, placing it at TRL 6 for this mission's specific operating environment [24,25,34].

The Base Station's Master OBC (Xiphos Q8) is assessed at TRL 6. The underlying Zynq UltraScale+ hardware carries flight heritage across Low Earth Orbit constellations, ISS crew-class telemetry hubs, and cislunar CLPS lander manifests including Intuitive Machines

and ispace missions, but its deployment specifically for continuous lunar south pole surface operations, including sustained exposure to south pole thermal cycling well beyond a lander's brief descent and landing window, has not yet been demonstrated [4,5,35]. The Base Station's Failsafe OBC (Vorago VA41630) is assessed at TRL 6. Its flight heritage on the QASCOM/NASA/ASI Lunar GNSS Receiver Experiment and qualification to MIL-PRF-38534 Class K standards support a mature rating, but this heritage does not include a PSR excursion mission, so the component is capped at TRL 6 under this mission's assessment criterion despite its stationary role outside the PSR boundary [6,35,36].

The Base Station's Iris V2 X-Band Transponder is assessed at TRL 6. Its flight heritage across MarCO's real-time relay communications during the InSight Mars EDL phase, CAPSTONE's X-band tracking into lunar Near-Rectilinear Halo Orbit, and multiple Artemis I secondary payloads including Lunar IceCube and LunaH-Map demonstrates strong deep-space and cislunar performance, and LunaH-Map and Lunar IceCube specifically targeted lunar polar volatile mapping in terrain adjacent to PSR environments, but none of this heritage comes from a PSR excursion mission itself, so the component is rated TRL 6 for this application [14,37,38]. The JHU/APL Frontier SSPA is assessed at TRL 6, reflecting flight heritage on deep space missions including the Van Allen Probes, Parker Solar Probe, and Europa Clipper, alongside its baseline flight demonstration aboard NASA's Interstellar Mapping and Acceleration Probe; this heritage is entirely deep-space rather than lunar-surface or PSR-adjacent, placing the component at TRL 6 under this mission's assessment criterion [15,16,39]. The Base Station's IQ Spacecom X-Band patch antenna is assessed at TRL 6, reflecting flight-proven heritage on numerous Low Earth Orbit missions but requiring further validation against lunar surface thermal cycling and the substantially longer Earth-Moon link path this mission requires [17,18,19]. The Base Station's NanoAvionics Turnstile UHF antenna is assessed at TRL 6, carrying extensive Low Earth Orbit CubeSat and SmallSat heritage but requiring custom assembly, radiation screening, and thermal enhancement to survive deep-space lunar surface deployment [28]. SpaceWire is assessed at TRL 6, despite extensive flight heritage across ESA, NASA, and JAXA missions including Gaia, ExoMars, and the James Webb Space Telescope, since it has not been demonstrated in a PSR-analogous environment specifically. RTEMS, as flight software rather than physical hardware, and while attaining its vendor-rated TRL 9 due to Mars and Perseverance, has never flown on a PSR lunar excursion; as a result, it is also rated TRL 6.

Following the principle that the subsystem TRL is limited by its lowest rated subassembly, the CDH subsystem's overall TRL is set at TRL 5. This reflects a subsystem built mainly from flight-proven or near-flight-proven COTS and SQCOTS hardware, where the limiting factor across almost every non-software component is not "new" technology but the absence of

direct lunar south pole PSR missions specifically, an environment no prior mission has fully replicated [34,30]. This meets the flight or deep-space history threshold established in CDH-005 and SYS-007.

2.1.3.3. CDH Subsystem Recovery and Redundancy Plans

Redundancy

The CDH subsystem's redundancy is tailored to each platform's mass and power constraints. The Drone uses dual PolarFire SoCs in an Active/Cold-Spare configuration, while the Base Station pairs a Xiphos Q8 primary processor with a low-power Vorago VA41630 that continuously monitors baseline health while the primary idles. The Base Station preserves science data in dual Space-Grade NVMe SSDs configured as a RAID 1 array ahead of DTE downlink, and the Drone falls back to a ProGrade Digital memory card if its primary NVMe fails.

Communications hardware and the internal SpaceWire bus are single-string on both platforms; the DTE chain (Iris V2, SSPA, X-Band antenna) and UHF local link hardware carry backup suppliers for procurement risk rather than onboard spares. This is an accepted risk offset by design margin and recovery procedure: DTE degradation is schedule-impacting rather than data-destroying, since bulk storage retains data for the next contact window, and the UHF link's omnidirectional antenna reduces the likelihood of signal loss before hardware failure becomes relevant. SpaceWire's backup path, RS-422 or CAN-FD, provides reduced-throughput fallback (below 50 Mbps versus 200 Mbps), an accepted tradeoff since this failure mode is rated Low risk and results in bottlenecked, not lost, data.

Recovery

Failure detection and recovery are governed by RTEMS and NASA's cFS, with all autonomous FDIR loops required to detect and act within 5 minutes of fault onset. To prevent battery drain during a fault, FDIR sends power commands to the PMAD bus and severs power if a platform's CDH draw exceeds its 100 W peak allocation, isolating the fault before it cascades. If the Drone's primary PolarFire SoC loses heartbeat telemetry or raises a fault flag during command validation, the system triggers a safe-mode reset, isolates the unit, and boots the cold-spare to resume navigation; if the primary NVMe fails mid-excursion, detected through a failed write acknowledgment, software workarounds redirect data to the memory card to ensure a safe return.

CDH receives dose-rate and cumulative TID telemetry from the Payload radiation spectrometer, compared against the Base Station OBC's 30 krad and Drone OBC's 100 krad ratings so an approaching threshold is flagged before hardware damage occurs. Loss of the BS-Drone UHF link is detected via 1 Hz RSSI monitoring; if contact isn't restored within the 5-minute FDIR window, the Drone holds position or returns to its last confirmed line-of-sight location. If a scheduled X-Band downlink window is missed, detected through a failed link-closure confirmation from the Iris V2, a store-and-forward workaround keeps science data safely stored in the RAID array until the next contact window.

2.1.3.4. CDH Subsystem Manufacturing and Procurement Plans

The mission's procurement approach prioritizes commercially available, space-qualified hardware over in-house development wherever a qualified product exists, consistent with NASA policy NPD 1370.1. The CDH subsystem applies this by minimizing cost, schedule, and technical risk through Space-Qualified Commercial-Off-The-Shelf (SQCOTS) suppliers for every major hardware component, including onboard computers, data storage, and communication hardware, a decision reinforced by recent price volatility in volatile memory and onboard storage markets. Leveraging established contractors with extensive flight heritage in avionics gives the mission high reliability, predictable lead times, and a streamlined software integration phase.

CDH Component Procurement Details

Onboard Computers (OBCs)

The Drone OBC uses a dual PolarFire SoC configuration in an Active/Cold-Spare setup, procured as SQCOTS hardware from Microchip Technology. PolarFire was selected over the BAE RAD750, BAE RAD5545, and Cobham (Frontgrade) GR712R for its radiation-tolerant SONOS flash architecture, immune to single event upsets and rated beyond 100 krad TID, a margin over the GR712RC's comparable rating at lower mass and power. BAE Systems is the backup supplier, offering the RAD750 or RAD5545 as drop-in alternatives with a multi-decade flight record if Microchip's lead time slips.

The Base Station OBC uses a Xiphos Q8 Zynq UltraScale+, procured as SQCOTS hardware from Xiphos Technologies for its 1.2 GHz quad-core processing and 4 GB RAM/256 GB storage, sized for the Base Station's heavy telemetry-formatting load and backed by Xiphos's integration heritage on CLPS lunar lander manifests. Its 30 krad TID tolerance is an accepted risk, offset by a radiation-hardened enclosure and the Vorago VA41630 (SQCOTS,

VORAGO Technologies), selected for its HARDSIL silicon-level hardening beyond 200 krad(Si), needed for its role as a continuously-powered monitor rather than an intermittently duty-cycled unit. Frontgrade Gaisler's GR716A/B is the backup, offering comparable radiation-tolerant performance with ESA flight heritage, accepted despite higher power draw since the monitoring role's total contribution stays small regardless of supplier.

Data Storage

Both platforms use the Exascend PR4-series space-graded NVMe SSD (SQCOTS), selected over comparable vendors for its 500 krad TID tolerance via combined hardware/firmware mitigation (Neutron Shield 2.0), the highest radiation margin evaluated, alongside a 3,000 MB/s write speed needed for payload instrumentation during PSR excursions and a sub-1W idle draw. The Base Station configures dual PR4 drives in a RAID 1 array for complete redundancy prior to DTE transmission. Phison's uSSD is the backup, offering comparable long-duration exposure heritage through flight on Perseverance and the ISS, accepted despite a narrower TID rating.

As a low-cost, low-mass backup, the Drone incorporates a CFexpress memory card from Prograde Digital, selected under a documented NASA partnership agreement providing schedule certainty and write speeds comparable to the primary NVMe drive, ensuring boot images and safe-mode telemetry survive a primary drive failure. Sandisk is the backup, qualified through Artemis-program hardware and offered at lower per-unit cost.

Internal Communication

SpaceWire hardware is procured as SQCOTS/COTS hardware from Glenair, selected for its established ECSS-E-ST-50-52C-compliant product line and decades of flight-qualified interconnect heritage across ESA, NASA, and JAXA missions, rather than sourcing connectors from a general-purpose aerospace harness vendor without SpaceWire-specific qualification history. SpaceWire itself was selected over RS-422 and CAN-FD alternatives for its 200 Mbps throughput. RS-422 and CAN-FD remain the backup procurement path if SpaceWire routing hardware is delayed, since both were evaluated in the same trade study.

Telecommunication (DTE and Local Link)

The Base Station's DTE transponder, an Iris V2 X-Band unit (SQCOTS, L3Harris), is selected for its flight heritage on MarCO's real-time relay during InSight's Mars EDL and CLPS IM-1, demonstrating reliability in the same autonomous, unattended DTE role this mission

requires. General Dynamics' Small Deep Space Transponder is the backup, offering comparable deep-space flight heritage despite higher power draw.

The signal is amplified by a JHU/APL Frontier Solid State Power Amplifier, procured through Rocket Lab's commercial licensing of the Frontier architecture and selected for its Gallium Nitride HEMT design, delivering higher efficiency at lower mass than legacy traveling-wave-tube SSPAs in this power class, backed by APL's own flight heritage integrating the architecture on deep-space missions ahead of licensing. Aethercomm Space is the backup, offering a comparable mass/volume envelope despite a less established GaN-specific flight record.

The signal radiates through an IQ Spacecom X-Band patch antenna, selected for its directional 16 dBi gain in an ultra-compact, low-mass footprint suited to the Base Station's fixed pointing geometry, an improvement over the mission's earlier quad-patch configuration, whose higher gain came at a mass and size penalty the trade study ruled unjustified for a stationary platform. EnduroSat's AD antenna is the backup, offering comparable performance and constraints.

For the local link, the AAC SpaceQuest TRX-U transceiver (SQCOTS, SpaceQuest, Ltd.) is selected over comparable UHF transceivers for nearly two decades of flight heritage across INSPIRESat-1, IDEASSat, and numerous CubeSat missions, including full qualification-level vibration and thermal vacuum testing, the most extensive validated record of any local-link transceiver evaluated, plus built-in gain/frequency control and digital RSSI reporting the mission's FDIR loop directly depends on for local-link-loss detection. NanoAvionics is the backup, offering comparable UHF performance across a broader CubeSat deployment base rather than matching AAC SpaceQuest's qualification depth.

The Drone's local link antenna is a Cylindrical Quadrifilar Helix Antenna, procured via Heritage Hardware Reuse from NASA's Jet Propulsion Laboratory, selected for its flight heritage on Perseverance's Rover UHF system, giving it credible planetary-surface history in the same UHF band the Drone operates in. Helical Communication Technologies is the backup, offering a comparable helical product line with far less flight history.

The Base Station retains a NanoAvionics Turnstile UHF antenna, selected for its passive, no-power quad-monopole design, preventing signal loss within a low weight/volume footprint suited to its stationary role. AAC SpaceQuest's ANT-100 is the backup, offering a whip antenna configuration with its own flight history.

UHF was selected over S-Band for its omnidirectional coverage, important to the Drone's multidirectional maneuvering; S-Band remains an alternate architecture if UHF hardware lead times slip.

Lead Time Summary for CDH

CDH lead times are estimated from vendor-published data where available, and from comparable SQCOTS components' historical delivery times where not. At the subsystem level, CDH totals a maximum of 25 weeks build time plus 3 weeks delivery, plus 3 weeks additional if a backup supplier is invoked, for 25 to 31 weeks overall.

A backup supplier is invoked under two circumstances: non-delivery from the primary vendor past its quoted lead time, or failure of incoming hardware to pass testing during Design/Finalization. Backup paths require additional verification against the Base Station and Drone before integration can proceed, consuming most or all of the standard 3-week contingency.

Subsystem / Component	Procurement Method	Primary Supplier	Backup Supplier	Estimated Lead Time	Justification
<i>Drone OBC (PolarFire Soc)</i>	SQCOTS (S... ▾)	Microchip Technology	BAE systems	20-25 weeks	SONOS flash gives >100 krad TID tolerance and SEU immunity at lower mass/power than GR712RC; BAE backup on RAD750/RAD5545 flight record
<i>BS OBC (Xiphos Q8)</i>	SQCOTS (S... ▾)	Xiphos Technologies	BAE systems	21 weeks	Xiphos Q8 offers highest processing/memory headroom for telemetry formatting load; established CLPS lander integration heritage; BAE backup on flight heritage (SpaceVPX architectures) and shorter lead time
<i>BS Failsafe OBC (Vorago VA41630)</i>	SQCOTS (S... ▾)	VORAGO Technologies	Frontgrade Gaisler GR716A/B	12-20 weeks	HARDSIL hardening gives >200 krad(Si) TID, deepest margin in CDH, needed for continuous monitoring role; Gaisler backup on ESA heritage despite higher power draw
<i>Drone/BS Data Storage (NVMe SSD)</i>	SQCOTS (S... ▾)	Exascend	Phison Electronics Corporation	6-8 weeks	500 krad TID via Neutron Shield 2.0, highest margin

					evaluated, sub-1W idle draw; Phison backup on Perseverance/ISS heritage
<i>DR Memory Card</i>	COTS (Com... ▾)	Prograde Digital	Sandisk	1-3 weeks	Recent NASA partnership agreement, comparable write speeds to main NVMe SSD; Sandisk used in Artemis missions, much more cost efficient, and whilst different architecture, provides similar dimensions and functionality.
<i>Internal Data Interface (SpaceWire Hardware)</i>	SQCOTS (S... ▾)	Glenair	RS-422 / CAN-FD	6-9 weeks	SpaceWire - 200 Mbps throughput and flight heritage; RS-422 or CAN-FD is a contingency alternate/ backup due to lower throughput and outdated technology
<i>DTE Transponder (Iris V2)</i>	SQCOTS (S... ▾)	L3Harris + Space Dynamic Laboratory (Iris V2 program)	General Dynamics (SDST)	8-15 weeks	Direct flight heritage on MarCO EDL relay and CLPS IM-1; GD backup based on deep-space network heritage and RF carrier frequencies
<i>Spaceborne X-Band SSPA (Frontier SSPA)</i>	Technology T... ▾	Johns Hopkins University Applied Physics Laboratory (JHU/APL)	Aethercomm Space	12-16 weeks	GaN HEMT architecture beats legacy GaAs MMIC-based SSPAs on efficiency/mass; APL heritage predates commercial licensing; Aethercomm backup on comparable envelope
<i>DTE Patch Antenna</i>	COTS (Com... ▾)	IQ Spacecom	EnduroSat AD	13-17 weeks	IQ Spacecom: directional high-gain patch design given stationary pointing geometry + TRL heritage; EnduroSat AD backup due to similar constraints and performance
<i>UHF Antenna (Quadrifilar Helix)</i>	Heritage Har... ▾	NASA's Jet Propulsion Laboratory	Helical Communication Technologies (HCT)	18-24 weeks	Direct heritage on Perseverance RUHF, same UHF band; HCT backup on comparable helical product line

<i>UHF Antenna (Turnstile)</i>	MCOTS (Mo... ▾)	NanoAvionics	AAC SpaceQuest ANT-100	13-17 weeks	Passive, no-power design prevents signal loss in low weight/volume footprint; SpaceQuest backup on whip configuration and flight history
<i>Local Link Transceiver (TRX-U)</i>	SQCOTS (S... ▾)	AAC SpaceQuest, Ltd.	NanoAvionics	16 weeks	Nearly two decades of flight heritage (INSPIRESat-1, IDEASSat, CubeSat constellations) with qualification-level vibe/TVAC testing; built-in RSSI reporting feeds FDIR directly; NanoAvionics backup on broader deployment base

Figure 2.1.3.4.1: CDH Procurement and Manufacturing Table

2.1.3.5. CDH System Verification Plans

The CDH subsystem's verification approach is built directly off the CDH requirements, using four verification methods: Analysis, Inspection, Test, and Demonstration. Verification demonstrates that each CDH requirement has been met by tracing a documented, repeatable method back to its parent system requirement, ensuring that hardware selections, software behavior, and subsystem performance can all be confirmed similarly to the CDH trade studies. Since several CDH requirements are structured as parent and child reqs (eg. CDH-002's storage requirement flowing down to CDH-BS-016 and CDH-DR-024's platform capacity requirements) verification of the parent requirement in these cases is confirmed once its associated child requirements have each also been verified, rather than requiring a separate verification effort. Since the CDH subsystem is almost entirely composed of SQCOTS and COTS hardware, a large portion of verification relies on documentation and datasheet confirmation through Analysis and Inspection. Test and Demonstration are reserved for behaviors that depend on other subsystems, such as FDIR responses, data throughput under load, and DTE link closure. Verification is scheduled across the CDH Testing phase (6 to 10 weeks) and Integration phase (10 to 14 weeks), with the majority of Test and Demonstration items concentrated in final integrated testing (TVAC, EMI, VIBE, and Ambient) conducted through the MCCET facility. Verification priority is also indicated by CDH risks, with requirements tied to the subsystem's highest rated risks, radiation degradation (CDH-R01), DTE link closure (CDH-R02), and local link loss (CDH-R03), receiving earlier and extensive verification attention.

Req #	Requirement Summary	Verification Method	Rationale for Method	Preliminary Verification Plan
CDH-001	X-Band DTE communication without relay satellites	Test	Link closure and frequency compliance can only be confirmed through active RF transmission/reception	Conducted during the final integrated test using the Iris V2 transponder and SSPA to confirm uplink/downlink frequencies (7.145 to 7.235 GHz and 8.400 to 8.500 GHz ranges), success defined as closed link.
CDH-002	Onboard storage sustained write speed $\geq 2,000$ MB/s	Test	Sustained write speed is a measurable performance that must be confirmed under realistic data load	Bench test of the Exascend PR4 NVMe SSD using representative payload data streams, success defined as sustained write throughput meeting or exceeding 2,000 MB/s
CDH-003	Onboard data retention with $\leq 0.1\%$ data loss until relay confirmation	Test	Data loss rate under nominal and degraded link conditions can only be confirmed through simulated transmission testing	Simulated DTE and local link interruption testing during integration, comparing transmitted versus received data volume, success defined as data loss remaining under 0.1%
CDH-004	Autonomous FDIR loops acting within ≤ 5 minutes of fault onset	Demonstration	FDIR behavior is a live system response that must be shown operating end to end rather than pure confirmation.	Fault testing during integration testing, timing intervals between induced fault and FDIR response, success defined as recovery action initiated within 5 minutes
CDH-005	CDH subassemblies demonstrate field heritage in comparable environment	Inspection	Flight heritage is a documented status confirmed through vendor and records	Review of vendor documentation and flight heritage records for each CDH subassembly prior to procurement finalization, success defined as all subassemblies meeting criteria
CDH-006	RSSI sampled and logged at 1 Hz across all active links	Test	Sampling rate and logging is a direct, measurable software function	Bench test of RSSI output during simulated Local and DTE link activity, success defined as consistent 1 Hz sample logging during operational mode
CDH-007	Enclosures prevent ingress of regolith particles ≤ 20 μm	Inspection	Enclosure sealing is a physical characteristic confirmed through direct examination	Physical inspection of CDH enclosure seals and gaskets following dust exposure testing, success defined as no particle ingress observed at or below 20 μm
CDH-008	Onboard compression/pre processing achieves $\geq 30\%$ data volume reduction	Analysis	Compression performance can be calculated directly from representative data sets	Analysis of compressed versus raw data volume using test payload data sets, success defined as a minimum 30% reduction prior to DTE transmission
CDH-009	Internal data interface supports ≥ 100 Mbps sustained throughput	Test	Throughput must be confirmed under actual data transfer conditions across the internal bus	Bench test of the SpaceWire bus using simulated data streams, success defined as sustained throughput meeting or exceeding 100 Mbps
CDH-010	CDH power draw ≤ 45 W average / 100 W max per platform	Analysis	Power draw can be calculated from component-level datasheets and summed across the subsystem	Power budget analysis summing power draw for each CDH subsystem, success defined as average and peak draw within specified limits

CDH-011	CDH mass allocation ≤4 kg per platform	Inspection	Mass is a fixed physical property confirmed through direct measurement	Physical weighing of assembled CDH hardware per platform prior to integration, success defined as total mass remaining at or below 4 kg
CDH-012	OBC storage write speeds independent of data interface throughput	Inspection	Interface independence is confirmed through architecture and datasheet inspection.	Ensuring OBC storage architecture is properly configured to ensure write speeds are independent and relating to the NVMe SSD rather than the data interface system (spaceWire). success defined as confirmed separation between storage and internal bus paths.
CDH-BS-013	BS receives/accumulates data from all instruments at ≤0.1% loss	Test	Data accumulation accuracy must be confirmed through active simulated data transfer	Bench test transmitting simulated instrument data to the Base Station under nominal link conditions, success defined as data loss remaining under 0.1%
CDH-BS-014	BS closes DTE link with ≥3 dB margin at max Earth Moon distance	Analysis	Link margin at maximum range is calculated through link budget analysis	Link budget analysis using Iris V2 and SSPA specifications against the 406,700 km maximum Earth Moon distance, success defined as link margin at or above 3 dB
CDH-BS-015	BS serves as local network hub within ~6 km line of sight	Test	Local link range must be confirmed through active transmission testing between platforms	Field or bench range test of the UHF local link between Base Station and Drone hardware, success defined as reliable command and telemetry relay maintained within 6 km
CDH-BS-016	BS provides ≥8 TB usable storage	Inspection	Storage capacity is a fixed, datasheet confirmed specification	Review of Exascend PR4 RAID 1 array configuration and capacity documentation, success defined as capacity at or above 8 TB
CDH-BS-017	BS electronics tolerate ≥30 krad TID	Test	Radiation tolerance must be confirmed through direct radiation exposure testing of hardware	TID exposure testing of CDH hardware at a radiation test facility, success defined as continuous operation after 30 krad exposure
CDH-BS-018	BS compresses instrument data while maintaining ≥2 TB reserve	Analysis	Reserve capacity margin can be calculated from storage allocation and compression performance data	Analysis of storage allocation against compression ratios, success defined as a minimum 2 TB reserve maintained at all times
CDH-DR-019	Drone detects obstacles ≥10 cm at 1.00 to 20.0 m continuously	Test	Obstacle detection range and continuity is a measurable sensor characteristic	Bench and field test of the ToF Depth Camera against obstacle targets across specified range, success defined as consistent detection of obstacles at or above 10 cm
CDH-DR-020	Drone retains data with ≤0.1% loss until BS transmission confirmed	Test	Data retention accuracy under link interruption must be confirmed through simulated transmission testing	Simulated local link interruption testing during Drone to Base Station transfer, success defined as data loss remaining under 0.1%
CDH-DR-021	Drone accumulates data from all instruments continuously	Demonstration	Accumulation across multiple instrument streams is best shown through live operation	Demonstration of instrument data streaming to Drone OBC storage during simulated environments, success defined as accumulation with no dropped instrument stream

CDH-DR-022	Drone electronics tolerate ≥ 100 krad(Si) TID	Test	Radiation tolerance must be confirmed through direct radiation exposure testing of hardware	TID exposure testing of PolarFire SoC hardware at a radiation test facility, success defined as continued operation after 100 krad exposure
CDH-DR-023	Drone enclosure prevents charged particles $\leq 20 \mu\text{m}$ entering during takeoff/landing	Inspection	Enclosure sealing is a physical characteristic confirmed through direct examination	Physical inspection of Drone CDH enclosures following simulated takeoff and landing dust exposure cycles, success if no particle enters system at or below $20 \mu\text{m}$
CDH-DR-024	Drone provides ≥ 1 TB buffer storage	Inspection	Storage capacity is a fixed, datasheet specification	Review of Drone NVMe SSD capacity documentation, success defined as usable buffer capacity at or above 1 TB
CDH-DR-025	Drone detects/responds to local link loss within ≤ 5 minutes	Test	Response timing to a link loss event is a measurable system behavior	Simulated local link loss testing during integration. Time during link loss and Drone recovery initiation is a success if marked within 5 minutes
CDH-DR-026	Drone prioritizes fault/health logs when bandwidth is hindered	Demonstration	Prioritization behavior under constrained bandwidth must be shown fully operational	Bandwidth-constrained demonstration transmitting simultaneous fault log and science data streams, success defined as fault/health logs consistently transmitted first

Figure 2.1.3.5.1: CDH Verification Matrix

Verification Method Rationale

Verification methods are selected to efficiently confirm requirements without over-testing. Analysis applies to mathematically calculated or datasheet-derived values (e.g., CDH-010 power budget, CDH-BS-014 link margin) before hardware is available. Inspection covers fixed physical traits (e.g., mass, storage, sealing) via direct review. Test assesses active hardware under simulated conditions (e.g., throughput, radiation, link closure) to confirm performance under realistic mission environments. Demonstration verifies real-time subsystem responses (e.g., FDIR fault recovery, data prioritization). Parent requirements are satisfied once all child requirements pass individual Inspection, eliminating duplicate effort.

Preliminary Verification Plan

Verification aligns with the subsystem schedule, starting with Analysis and Inspection during Design (8–12 weeks) and Procurement (25–31 weeks) to confirm physical metrics via vendor documentation. Test items (radiation, throughput, link closure) occur during the Testing phase (6–10 weeks) using external facilities for integrated environmental windows (TVAC, EMI, VIBE, Ambient). Demonstration items (FDIR, prioritization) occur during Integration (10–14 weeks) once hardware and software are combined. If backup hardware

is required, a 3-week schedule contingency may compress these windows, requiring datasheet reconfirmation before testing begins.

2.1.4. Thermal Control Subsystem Overview

Thermal Control Subsystem (TCS) handles all internal and external cooling or heating for the mission. TCS is very important for all missions and all systems that deal with mechanisms that will eventually overheat due to environmental effects or usage of energies affecting its long term survivability and completion of the missions it is assigned. TCS systems are meant to effectively keep the spacecraft operational while facing the harsh conditions that are in space. Thermal Controls for these missions will exist on to a Dual-Architecture that is proposed which is the BS and DR. Each facing their own issues and also need different TCSs.

TCS will support the other subsystems in this mission (CDH, Electrical, Mechanical), keeping balanced at their respective operating temperatures to minimize the issues that can arise during space exploration.

Environmental hazards can include the sun's effects on the systems while beginning on the moon's south pole which will reach temperatures as high as +120°C and as low as -233°C. That is why TCS is a very crucial subsystem in order to maintain other systems functional.

The components for the TCS were split off into passive and active types of TCS. Passive TCS have many kinds of components that work differently to each other such as Multi-Layer Insulations (MLI), Thermal Straps (TS), and Radiators. These will help not only reduce the usage of power needed to be used but also dissipate heat or bring in heat into areas needing it. There has to be a balance with temperatures to have the systems operating.

Subassembly	Mass (kg)	Dimensions (m)	Maximum Power Draw (W)	TRL
Aluminized Kapton Multilayer-Insulation	~15.5kg	~430m ² x .0003	0	6
Pyrolytic Graphite Sheet (PGS) Thermal Straps x23	1.65kg	0.3 x 0.04 x 0.002	0	6

Q Rad Deployable Panel Radiator	8-10kg	Panel Length: 0.5 x ~0.25 x ~.002	0	6
Lockheed Martin Space MICRO1-1 Cryocooler x2	~0.70kg	Length: 0.205 x ~1.0 x ~1.0	15W	6
Minco Polyimide Thermofoil HK Series Heater x23	~0.115kg	0.09 X 0.07 X 0.0003	3W-15W	6
Tark Thermal Solutions CP14-7-10-L1-W4.5 Thermoelectric Cooler x6	~0.015kg*6 = 0.09kg	0.0095 X 0.0095 X 0.0047	15W-40W	6
Thermal Sensors Banner Engineering S15S-T-MQ x10	~0.030kg	0.0569 X 0.015 X 0.015	1.5 W	6
Total Subsystem	~29kg		Min = 48W, Max = 85W	6

Figure 2.1.4.1.1 Subassembly Summary Table

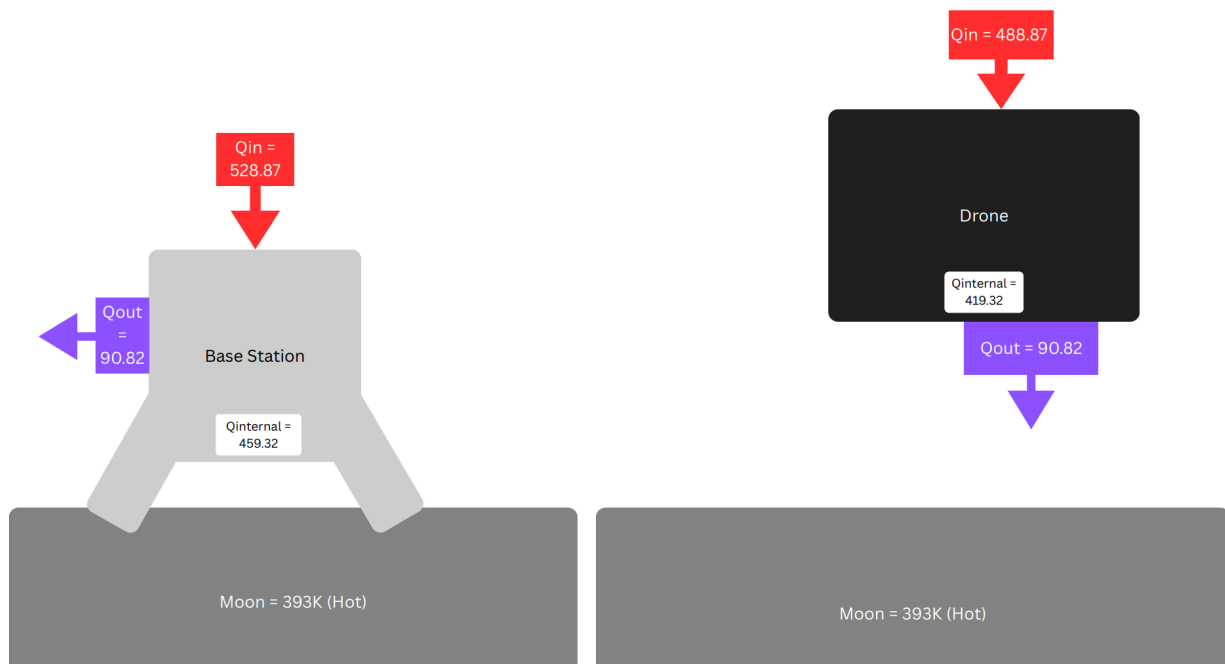


Figure 2.1.4.1.2: Heat Flow Hot Case

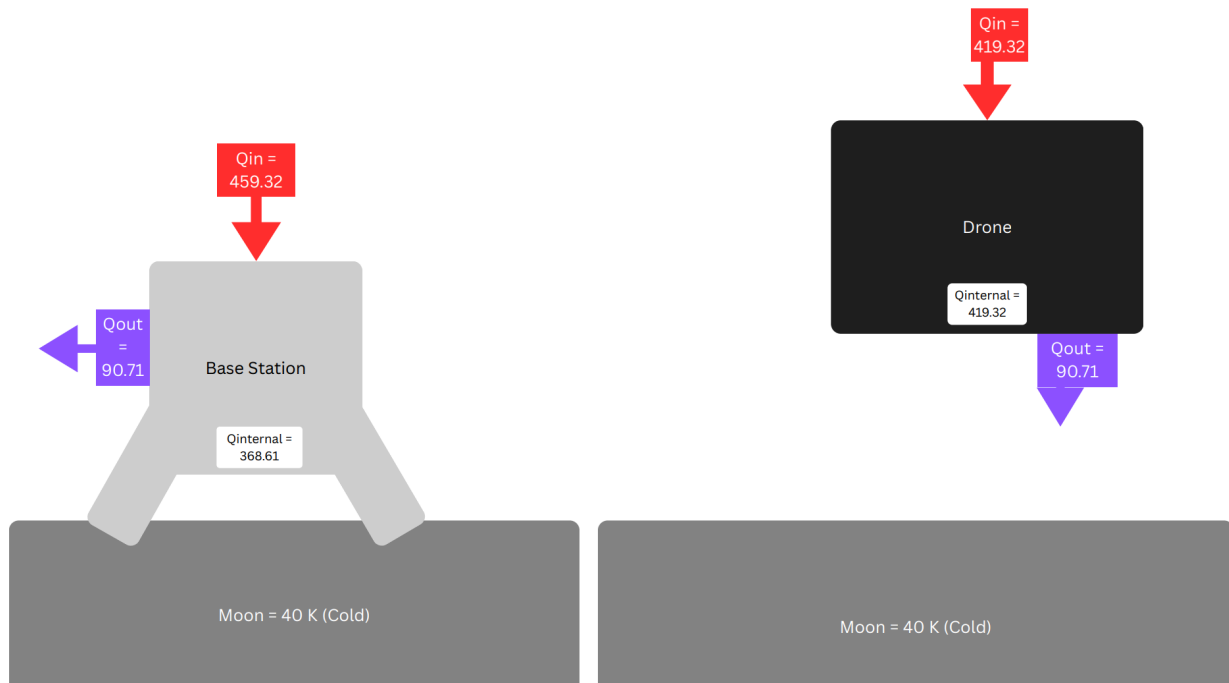


Figure 2.1.4.1.3: Heat Flow Cold Case

2.1.4.1. Thermal Control Subsystem Requirements

Req #	Requirement	Rationale	Parent Req	Child Req	Verification Method	Req met?
THERM-007	Materials for thermal systems should meet thermal standards based on where passive and active cooling systems are needed for both Drone and Base stations.	Ensures the propulsion subsystem can generate reliable thrust and maintain performance during deployment, flight, landing, and lunar surface operations despite extreme lunar location temperature conditions.	SYS-004	THERM-BS-003, THERM-BS-004, THERM-DR-006. 1	Demonstration	Met
Base Station THERM (THERM-BS-XXX)						
THERM-BS-002	The base station thermal subsystem shall provide thermal	This will prevent electronics, batteries, or science payloads from	THERM-BS-004	THERM-007	Inspection	Met

	isolation between heat generating components and temperature sensitive payloads.	being damaged due to unwanted				
THERM-BS-003	The base station thermal subsystem shall maintain thermal performance in lunar vacuum conditions where conduction is unavailable.	Thermal design should rely on radiation and conduction systems	THERM-BS-004	THERM-DR-006	Test	Met
THERM-BS-004	The base station thermal subsystem shall be able to withstand the expected lunar south pole temperature range of +120 C to -233 C.	Ensures bases system can endure extreme hot and extremely cool temperatures	SYS-010	THERM-BS-001, THERM-BS-002, THERM-BS-003, THERM-DR-006	Test	Met
THERM-BS-005	Active cooling systems should operate while taking less then 95W from the power supply for efficient base station operation	Active Cooling systems take less energy from other systems.	SYS-010 , SYS-004	THERM-007	Analysis	Met
Drone THERM (THERM-DR-XXX)						
THERM-DR-006.1	Hardware from thermal subsystems should be able to face Lunar South Pole Regions with 100% survivability once on surface	Ensures reliable operation under extreme lunar thermal conditions	THERM-DR-006	THERM-007	Test	Not Met
Req #	Requirement	Rationale	Parent Req	Child Req	Verification Method	Req met?
THERM-007	Materials for thermal systems should meet thermal standards based on where passive and active cooling systems are needed for both Drone and Base stations.	Ensures the propulsion subsystem can generate reliable thrust and maintain performance during deployment, flight, landing, and lunar surface operations despite extreme lunar location temperature	SYS-004	THERM-BS-003, THERM-BS-004, THERM-DR-006.1	Demonstration	Met

		conditions.				
Base Station THERM (THERM-BS-XXX)						

2.1.4.2. Thermal Control Sub-Assembly Overview

MLI is a great passive TCS and is a highly regarded form of TCS for its long history from previous missions. So with that this mission will follow and also use MLI. The specific material going to be used is Aluminized Kapton, selected for its excellent radiation and UV resistance, thermal performance, and durability. Standing out for its capabilities, it was a decision compared to many other materials. MLI comes in sheets and each will be layered wherever needed for example on the exterior parts of the BS and DR. As well in the interior portion to maintain heat for components. Aluminized Kapton MLI comes with reflective and heat-insulating sides. The TRL for this is rated at 9 since this has been used on previous NASA missions.

Pyrolytic Graphite Sheets (PGS) Thermal Straps are other subassembly components used to transfer heat from electronics that are sensitive to the effect of heat coming from the sun. PGS Thermal Straps are flexible components and can be placed anywhere acting as links between internal systems helping the DR or BS complete its task. One side is connected to the electrical components that could be heating or cooling down and the other side of the strap is connected to other TCS subsystems to regulate its temperatures. Compared to other materials PGS is light weight and excellent at conducting heat from system to system and resisting radiation while on the moon all due to the graphite materials it using. The TRL rating given is a 9 due to its flight heritage on Space missions.

Deployable Panel Radiator helps remove any excess heat coming into the systems. Another method to cool the components passively helps us conserve energy usage.

Active TCS work the same as the passive TCS just they will require power in order to cool down or heat up the system inside the BS and DR. The subassemblies for this mission would be Cryocoolers, Heaters, and Thermoelectric Coolers. Each one serves an importance to keeping Dual-Architecture operating through the mission's life cycle.

Lockheed Martin Space MICRO1-1 Cryocooler is a specific component meant to cool down systems using helium gas. Helpful for any cameras and sensors for both the DR and BS. The active TCS won't need to be continuously on permanently removing power from the

systems but once it's on it will take about 15W-40W of power. The power is still in discussion with other subteams and their components.

MINCO HK6905 Polyimide WA Thermofoil™ Heaters will help with boosting the heating power to electronics that are feeling the effect of the cold lunar nights in the PRCs of interest. Since there will be issues with cooling, having heaters close in contact with the electronics will help.

Tark Thermal Solutions Thermoelectric Coolers are another cooling device but they use eclectic current to transfer temperatures from components and also can help with heating when need be.

1. Technology Readiness Level (TRL)

MLI Aluminized Kapton will be rated at TRL 6 as bar to pass should be able to successfully work in a simulated environment. Has been recommended by NASA articles.

PGS Thermal Straps are given a TRL 6 as well since they are space grade and can be simulated in an environment like that on the moon.

Deployable Q-Rad Panel Radiator at TRL 6 but their are company rating of TRL 5 yet the push to make it a 6 will be through all forms

Lockheed Martin Space MICRO1 Series Cryocooler TRL 6 does have space history with previous NASA Missions but still will be reduced to this rating for this specific mission.

MINCO HK6905 Polyimide WA Thermofoil™ Heaters TRL 6 but important when facing the cold temperatures that will occur as this system lands in the PSRs. Will help the mission feel secured when this can be tested in a similar environment to the mission's POI.

Tark Thermal Solutions CP14-7-10-L1-W4.5 Thermo electric coolers are also given a TRL 6 for their cooling capabilities if any point components from all subsystems begin to overheat by the amount of power they are using.

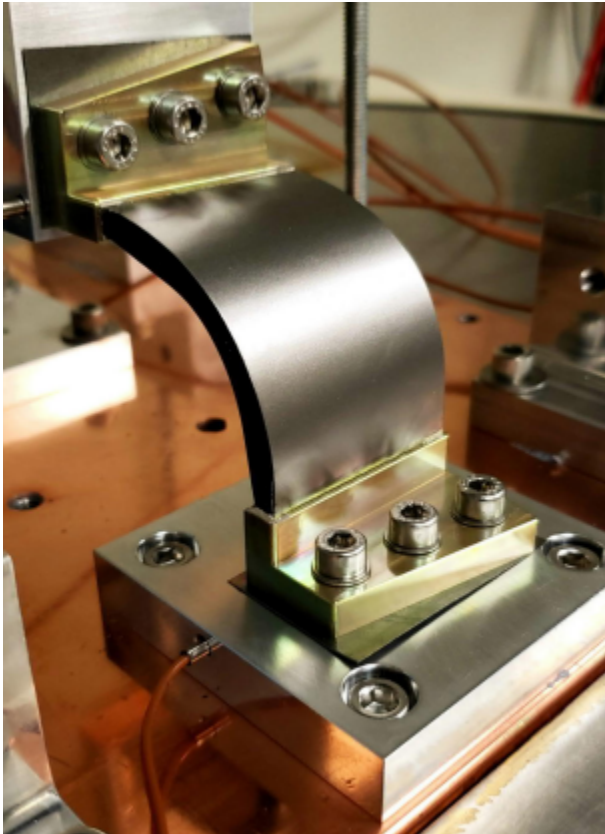


Figure 2.1.4.2.4: PGS Thermal Straps Technological Application Inc.

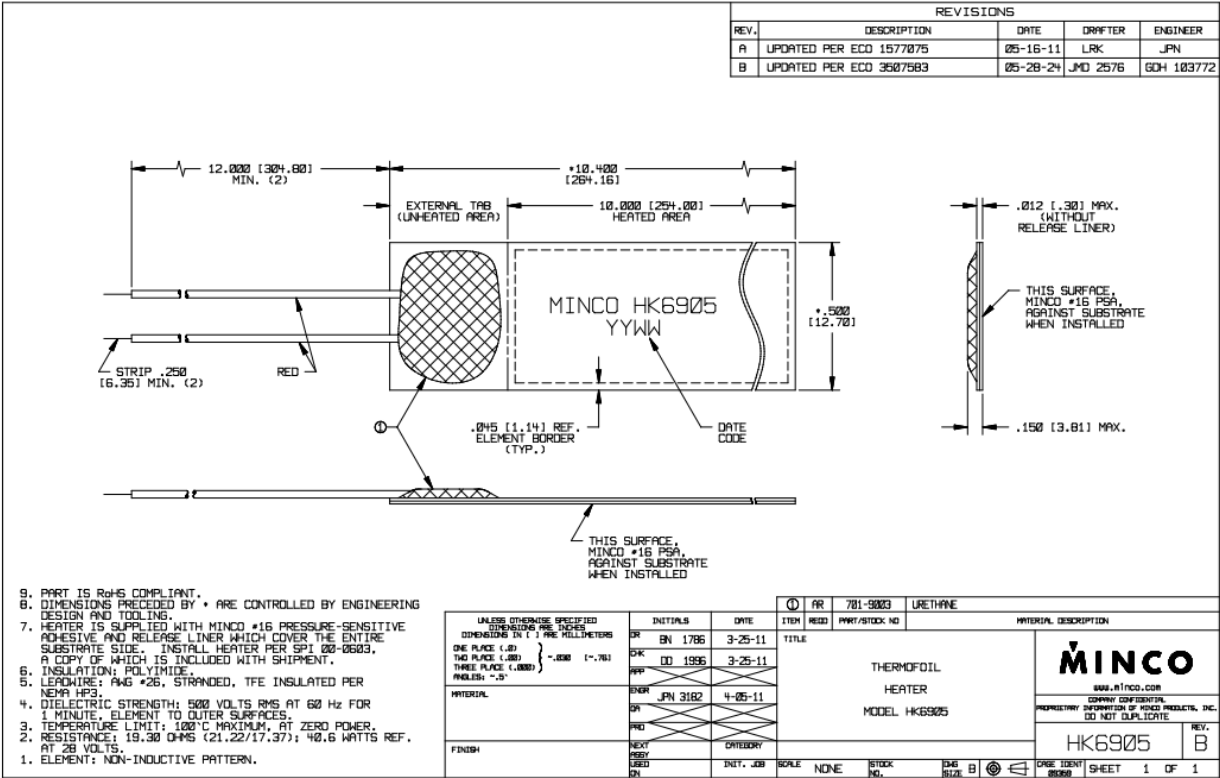


Figure 2.1.4.2.5: Thermofoil Electric Heaters MK6905 Minco Dimension Drawing

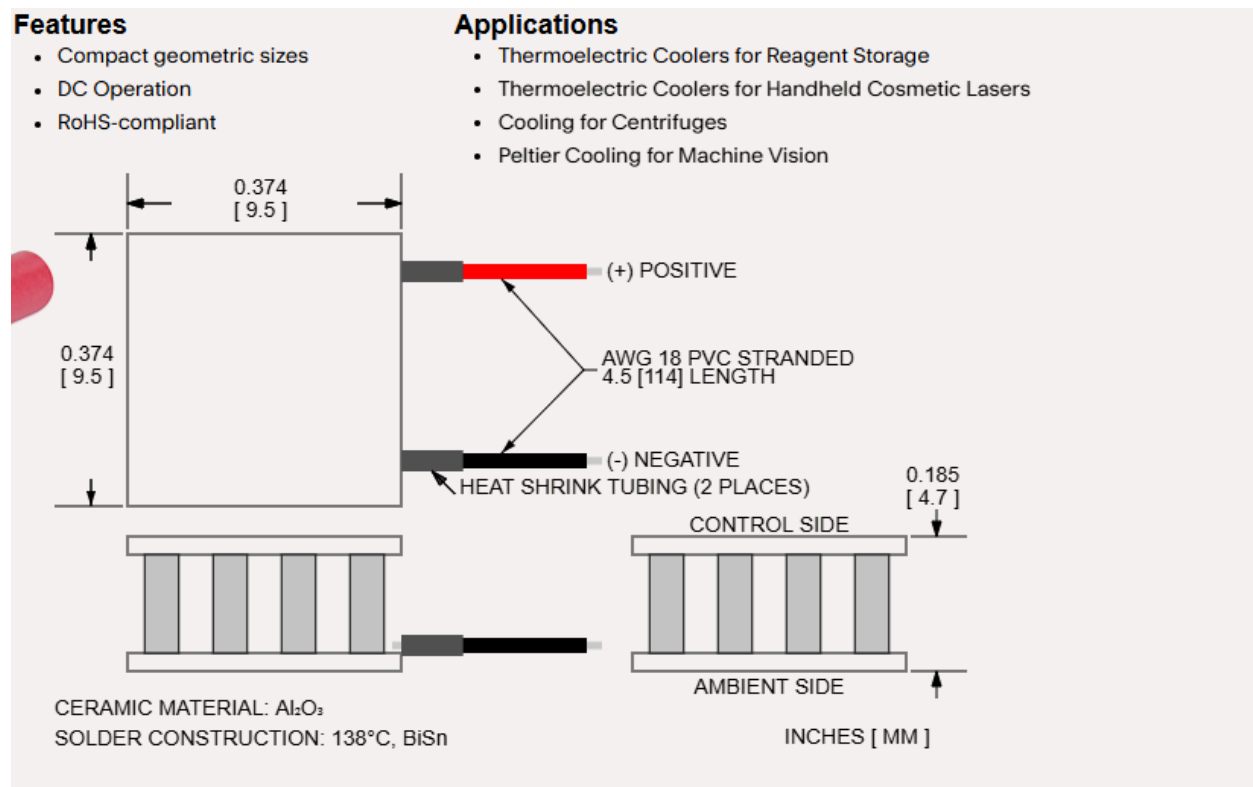


Figure 2.1.4.2.6: Thermoelectric Coolers CP14-7-10-L1-W4.5 Dimensions Drawing Tark Thermal Solutions

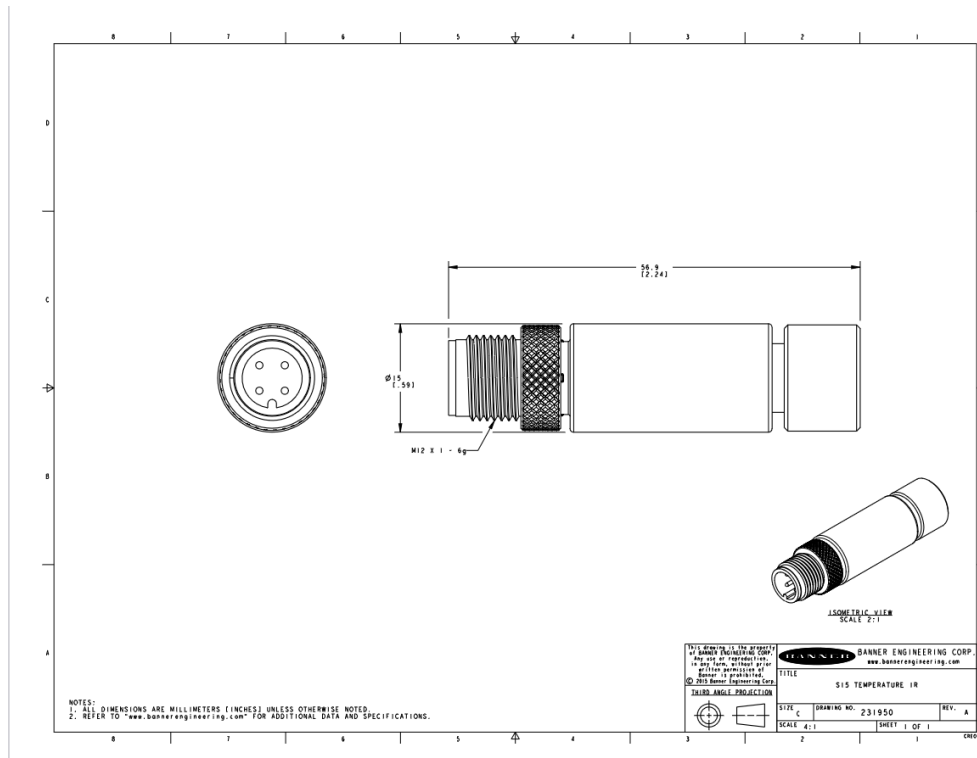


Figure 2.1.4.2.7: Thermal Sensors Drawing Banner Engineering S15S-T-MQ

2.1.4.3. Thermal Control Subsystem Recovery and Redundancy Plans

Redundancy

Heaters and temperature sensors are dual string on both the drone and base station, so a single failure would not cause loss of thermal stability or monitoring in critical zones. Heaters contain independent heater circuits and controllers that protect critical thermal zones such as batteries, computers, and scientific payload compartments, so the use on both platforms are redundant. Coolers will be the least used since the mission takes place in an environment predominantly cold. They will all be monitored using thermal sensors for all subsystem components and instruments.

Recovery

If both heaters in a zone fail, adjacent zone heaters and passive thermal margins such as the MLI and radiators slow temperature decline, allowing time for the base station to pause operations or the drone is able to operate a safe procedure before limits are exceeded. As all are monitored systems and protocols are in place to maintain operating temperatures for all components, because there are multiple sensors on the systems in case they fail.

2.1.4.4. Thermal Control Subsystem Manufacturing and Procurement Plans

Thermal Component Procurement Details

The thermal subsystem will primarily utilize commercially available thermal control hardware (COTS) to reduce development time, minimize technical risk, and leverage technology that has already demonstrated reliability during flight operations. Components such as flexible heaters, multilayer insulation (MLI), thermal straps, cryocoolers, radiators, and thermoelectric coolers are readily available from aerospace-qualified suppliers. Custom manufacturing will be limited to subsystem integration, mounting brackets, insulation sizing, and interfaces designed specifically for the mission. Commercial suppliers were selected based on technical performance, availability of aerospace-qualified products, and compatibility with the extreme thermal environment of the lunar south pole.

Thermal Sensors

Environmental and component monitoring will be performed using Banner Engineering as the primary supplier and FCI Aerospace as the backup supplier. Banner engineering provides commercially available industrial sensors with short procurement lead times and integrated temperature and humidity monitoring, while FCI offers aerospace qualified RTD sensors with high precision and flight proven reliability. Sensors provided from either suppliers will provide real time thermal data from both the Base Station and Drone, enabling the Thermal Control System to maintain operating temperatures within a safe range and provide protection to critical spacecraft hardware.

Multilayer Insulation

Aluminized Kapton MLI will be bought commercially, with specific details of how much of the film is needed when manufacturing of the BS and Drone begins. Yet having enough for full exterior coverage and insulation onto the surfaces. Dunemore Aerospace will be able to provide our needs for this mission.

Deployable Radiator

Q-Rad Deployable Radiator will be bought by the manufacturer and needed to be mounted onto the BS currently due to its longer exposure to radiation from the sun. Having the bigger space to work for all internals in the BS will benefit from its placement there. Redwire has many instruments that have been used for previous missions, and these components will be enough to operate for this mission.

Thermal Straps

PGS Thermal Straps bought from the supplier benefit the BS and DR systems to mitigate overheating or also the need to transfer heat from heat sources to systems that will experience the colder temperatures in the lunar PSR. Technology Application Inc. was able to provide commercial prices and data to back up its Primary Supplier status.

Cryocoolers

Cryocoolers will need some customized specifics to fit well into this mission. Yet as one of the Active TCS needed power to this system is crucial when internal systems/instruments begin to overheat when operating on BS and DR. Flight heritage used on previous Artemis missions and the company innovated from foundation technology. Lockheed Martin Space division focused on these components and future development

Electric Heaters

Heaters will be frequently used for this mission, as this Active TCS will be seen using them the most power in comparison to the rest of the TCS. Needing many and also constant attention to ensure their survivability on the Lunar Surface. The primary supplier was Minco as they had a wide range of heaters to select from.

Thermoelectric Coolers

For additional localized cooling, the subsystem will incorporate thermoelectric coolers supplied by Tark Thermal Solutions, with Coherent serving as the backup supplier. Thermoelectric coolers provide precise temperature regulation for heat-sensitive electronics and payload components without any required moving elements, improving reliability while reducing maintenance requirements. Ceramic Plate Series Thermoelectric Coolers offer many options to choose from that will satisfy mission needs.

Lead Time Summary

The estimated schedule for the Thermal Control Systems will go as follows in terms of having components ready for testing and integration at scheduled timing. TCS is broken into 6 components to help the systems operate while carrying out the mission and 1 current component to monitor temperatures to stay operational in the Permanently Shadowed Region (PSR) on the moon. Lead times are estimated to be 1 year for all components to be completed and ready for operations. Designing would be the first 15% of component lead time. Testing could change, but is expected to be a TRL 6 and have taken at least 25% of the lead time.

Thermal					
Subsystem / Component	Procurement Method	Primary Supplier	Backup Supplier	Estimated Lead Time	Justification
Aluminized Kapton MLI Film	COTS (Com... ▾)	Dunmore Aerospace	CAPLINQ	8-20 weeks	MLI Insulation that will be on both BS and DR exterior. Aluminized Kapton was the better-selected material for its reflective properties and radiation resistance.
Q-Rad Deployable Radiator	COTS (Com... ▾)	Redwire	Northrop Grumman	3-5 months	Deployable Radiators will help with heat rejection inside the BS. Instruments and components can overheat, and TCS is needed to reduce this possibility.
PGS Thermal Straps	COTS (Com... ▾)	Technology Application Inc.	Thermotive	18-25 weeks	Transferring heat from heat sources to different components using the PGS Thermal Straps. PGS material is more effective at transferring that heat, needed for the -230 degrees Celsius in PSR on the moon's south pole.
Cryocoolers	MCOTS (Mo... ▾)	Lockheed Martin	Sunpower/Ametek	12 months	Cryocoolers will be custom made to fit the needs of this mission. Lead times extend longer than 12 months, but best case scenario is this time frame. Another TCS to help create
Electric Heaters	COTS (Com... ▾)	Minco	Zoppas Industries Space	12-22 weeks	Electric Heater will help maintain the operating temperatures of the internal components and instruments for both BS

					and DR systems
Thermoelectric Coolers	MCOTS (Mo... ▾)	Tark Thermal Solutions	Coherent	25-46 weeks	Custom manufacturing. CP Series of Thermal Coolers will help provide that necessary cooling in areas/instruments that experience constant operation of the BS and DR.
Thermal Sensors	COTS (Com... ▾)	Banner Engineering	FCI Aerospace	5-24 weeks	Temperature and humidity sensors will monitor environmental and component temperatures on both the BS and DR. These sensors will provide accurate data to the thermal control system, allowing for safe operating conditions

Figure 2.1.4.4.8: Thermal Procurement Table

2.1.4.5. Thermal Control Subsystem Verification Plans

The Thermal Control System (TCS) verification approach is built directly using the thermal system requirements through a combination of analysis, test, inspection and demonstration. The verification process will allow for evaluating the ability of both the Base Station and Drone thermal control architectures to maintain critical components within their operable temperature ranges. This approach is necessary because the TCS uses both passive and active thermal control methods. The purpose of verification is to demonstrate that these components are able to operate together to maintain sensitive spacecraft and payload hardware within its allowable operating range in extreme temperature and radiative conditions. Passive components such as MLI, PGS, and the Q-Rad deployable radiator will be verified primarily through analysis and inspection depends on material properties, geometry, placement, and thermal interfaces. Active components that depend on applied electrical power and thermal loads will require verification testing.

Req #	Requirement Summary	Verification Method	Rationale for Method	Preliminary Verification Plan
THERM-BS-002	The base station thermal subsystem shall provide thermal isolation between heat generating components and temperature sensitive payloads.	Analysis ▾	Conductive and radiative isolation performance from MLI, standoffs, and thermal breaks can be predicted using established material properties..	Model conduction paths (structure, mounts, harnesses) and radiative view factors between identified heat generating components and temperature sensitive payload. Then, verify whether predicted payload temperature rise remains within allowable limits with isolation features.
THERM-BS-003	The base station thermal subsystem shall maintain thermal performance in lunar vacuum conditions where convection is unavailable.	Analysis ▾	MLI, PGS thermal straps, and radiators have predictable thermal properties and physical configurations that can be evaluated analytically	Verify the material and configuration of the Aluminized Kapton MLI, PGS thermal straps, Q-Rad radiator. Perform thermal modeling to confirm adequate insulation, heat transfer, and heat rejection under expected conditions
THERM-BS-004	The base station thermal subsystem shall be able to withstand expected lunar south pole temperature range of -233 degrees Celsius to +230 degrees Celsius.	Analysis ▾	Thermal modeling can determine whether the proposed MLI, thermal straps, radiator, heaters, and active cooling provide sufficient thermal performance.	Perform thermal analysis using expected Base Station heat loads and environmental conditions. Following integration, perform thermal testing while monitoring temperature sensitive electronics and comparing measured temperatures against analytical predictions.
THERM-BS-005	Active cooling systems should operate while taking less than 10% from the power supply for	Test ▾	Actual power draw of pumps, fans, and valves depend on real hardware efficiency, so measured performance is needed to confirm the	Compare total active cooling power consumption to the base station power allocation to confirm <10% threshold is met.

	efficient base station operation		power budget requirement.	
THERM-DR-006.1	Hardware from thermal subsystems should be able to face Lunar South Pole Regions with 100% survivability once on surface	Analysis ▾	Long duration cool weather survivability can be evaluated through thermal analysis using worse case scenario environmental conditions.	Confirm no component drops below its minimum non operating temperature limit for the required exposure duration
THERM-DR-007	Thermal subsystems should be at a level 6 TRL rating before further procedures	Inspection ▾	TRL is established by reviewing document test and demonstration evidence against standard TRL definitions.	Review test reports, demonstration records, and design documentation for each thermal subsystem component against TRL definitions, then confirm components have reached TRL 6 before proceeding to next development phase

Figure 2.1.4.4.9: Thermal Validation and Verification Table

Analysis will be used for requirements that can be evaluated using thermal models, calculations, or components specifications before system integration. This includes heat transfer, expected component temperatures, thermal margins, and TCS power consumption. Inspection will be used for fixed physical characteristics like component installation, sensor placement, MLI coverage, thermal interfaces, and compliance with supplier specifications. Test will be used where hardware performance must be demonstrated under representative thermal or environmental conditions, particularly for the -233 degree celsius to +120 degree celsius mission environment and active thermal control components. Demonstration will be reserved for integrated thermal control functions that require multiple components to operate with one another.

Thermal verification will begin during the Design and Procurement phases with analysis and inspection of component specifications and the preliminary thermal model. Vendor data for the selected COTS and MCOTS components will be used to establish expected thermal performance and power requirements. Component level testing will verify heater, cooler, and sensor performance. Integrated thermal testing will then evaluate the TCS under representative hot and cold conditions. The thermal model will be updated using measured test data to ensure that analytical predictions accurately represent the

integrated hardware. Final verification will occur during subsystem integration, where the TCS will be operated alongside electrical, mechanical, CDH, and payload loads. Successful verification will require all measured temperatures to remain within the applicable component and thermal system limits while the TCS operates within its predicted and allowable power range. The current procurement plan anticipates long lead times particularly the cryocoolers having procurement estimated at approximately 12 months. Therefore, thermal analysis and vendor data verification should begin to ensure proper integration and requirement approval.

2.1.5. Payload Subsystem Overview

The Payload subsystem collects all scientific and engineering data supporting the mission's four science objectives: characterizing terrain and illumination effects on navigation, characterizing regolith and radiation hazards, monitoring platform flight dynamics, and enabling autonomous hazard avoidance. Because the core science objectives require measurement inside and adjacent to Permanently Shadowed Regions (PSRs), all payload instrumentation is carried aboard the Scout Drone rather than the stationary Base Station. The subsystem integrates ten instruments across four functional groupings, an optical/terrain-imaging suite, a dust and radiation-sensing suite, an inertial-sensing suite, and a navigation/hazard-detection suite, each tied to a specific objective in the Science Traceability Matrix.

The optical/terrain-imaging suite consists of the TAGCAMS Optical Navigation Suite, the SCALPSS-derived Stereoscopic Terrain Camera, and VIPER-derived illumination floodlight arrays. TAGCAMS provides high-resolution optical navigation imagery, while the Stereoscopic Terrain Camera provides stereo depth data used to resolve elevation and slope changes at the resolution needed for Terrain Relative Navigation. Because PSRs receive no direct sunlight, both cameras depend on the illumination floodlight arrays to return usable imagery once the drone leaves illuminated terrain. Together, these three instruments generate the 3D topographical point clouds, slope maps, and surface albedo profiles used across illuminated and PSR terrain.

The dust and radiation-sensing suite consists of the LADEE LDEX-derived Dust Impact Mass Spectrometer and the LETS-heritage Radiation Spectrometer. The Mass Spectrometer records electrostatic charge accumulation and impactor kinetic energy to characterize dust mobilization and micrometeorite impact behavior, while the Radiation Dosimeter logs Total Ionizing Dose, particle energy spectrum, and dose rate to evaluate long-term electronics

survivability. Together, these instruments support the mission's second science objective of quantifying physical surface degradation risk to the platform.

The inertial-sensing suite, the Northrop Grumman LN-200S Gyro-IMU and Honeywell MIMU, monitors vibration amplitude, acceleration, and structural stability throughout each excursion, informing the flight controller in real time to maintain stable maneuvering between illuminated terrain and PSR darkness. The navigation/hazard-detection suite pairs the Flight Processing Computer Software with the Power Monitor and the RESOURCE Time-of-Flight Depth Camera: the flight software executes and updates the drone's autonomous flight path based on live hazard input, the Power Monitor tracks energy expenditure for energy-optimal path selection, and the ToF Depth Camera supplies real-time obstacle detection, allowing hazard avoidance without operator intervention where communication delay makes real-time commanding impractical.

Instrument	Mass (kg)	Dimensions (m) (LWH)	Maximum Power Draw (W)	TRL
TAGCAMS Optical Navigation Suite	5	~0.078 × 0.058 × 0.164 (full camera ×3) ~0.20 × 0.20 × 0.06 (DVR system)	~24	5
Stereoscopic Terrain Camera (SCALPSS-derived)	6.5	0.197 × 0.142 × 0.152 (MSE-lite) 0.039×0.029×0.029 (per camera x2) 0.050×0.030×0.018 (per laser x2) 0.200×0.050×0.050 (per optic x2)	54	5
Illumination Floodlight Arrays (VIPER)	1	~0.08×0.06×0.05 (×2)	64	5
Radiation Spectrometer/Dosimeter (LETS)	0.43	~0.12 × 0.052 × 0.0045	2.0	6
Dust Impact Mass Spectrometer (SUDA derived)	12.57	0.268 × 0.250 × 0.171	15.3	5
Space Grade IMU-Gyro (LN-200S)	0.75	0.089 × 0.089 × 0.085	12.0	6
Space Graded IMU (Honeywell MIMU)	4.6	0.234 × 0.234 × 0.170	25.0	6
Power Monitor (INA901-SP + ADC128S102QML-SP)	<0.01	0.0065×0.0065×0.0028 (×2 chips)	<0.02	5
Time-of-Flight (ToF) Depth Camera (RESOURCE Camera)	~0.35	~0.126 × 0.103 × 0.039	6	6

Total Subsystem	~31.21 kg	32,450.73 cm ³	MAX: ~202.32 W Avg Power: ~142.61W	5
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Figure 2.1.5.0.1: Payload Mass, Dimensions, Power Draw, and TRL

Payload Capabilities

The Payload subsystem's primary capability is optical/stereoscopic surface imaging, provided by TAGCAMS, the Stereoscopic Terrain Camera, and the illumination floodlight arrays that make imaging possible once the drone enters unlit terrain, together generating the topographical, slope, and albedo datasets needed to evaluate how terrain and illumination gradients affect surface navigation. A second core capability is environmental hazard sensing, provided by the Dust Impact Mass Spectrometer and Radiation Spectrometer, which characterize electrostatic regolith behavior, micrometeorite impact energy, and cumulative radiation dose across the mission's operational lifetime. A third capability is inertial and flight-dynamics monitoring, provided by the fused Space-Grade IMU configuration (MIMU and LN-200S), supporting evaluation of platform stability and maneuverability during autonomous flight in lunar gravity. The fourth capability is autonomous navigation and real-time hazard avoidance, provided jointly by the Flight Processing Computer Software, Power Monitor, and ToF Depth Camera, allowing the drone to plan, execute, and adjust its excursion path without real-time operator commanding. Collectively, the Payload subsystem occupies approximately 31.2 kg and ~34,068.47 cm³ of the drone's mass and volume budget, drawing an average of 142.6 W against a peak of 202.3 W, and builds the baseline supporting future Artemis-era surface mobility planning at the lunar south pole.

2.1.5.1. Science Instrumentation Requirements

The payload subsystem requirements establish the constraints the Scout Drone's instrumentation must satisfy to fulfill the mission's science and mobility objectives, organized into three tiers: general requirements (PAY-XXX) applying across all instrumentation, science requirements (PAY-SMD-X.XX) tracing to Science Mission Directorate objectives, and mobility requirements (PAY-HSMD-X.XX) tracing to Exploration Systems Development Mission Directorate objectives.

The general requirements, PAY-001 through PAY-006, trace to stakeholder and system-level constraints. PAY-001 requires instrumentation to pass simulated space environment testing or demonstrate prior flight heritage, lowering engineering risk ahead of PSR excursions. PAY-002 and PAY-003 derive from the mission's operating environment (SYS-009, SYS-021): since excursions carry the Drone between sunlit terrain and PSR

cold-soak, electronics must survive -233°C to $+120^{\circ}\text{C}$ while tolerating under 30 krad cumulative TID, consistent with the system-level shielding budget. PAY-004 flows down the autonomy constraint (SYS-006), requiring excursions and data collection without real-time operator commanding. PAY-005 requires data transfer at a rate compatible with the UHF local link and X-Band DTE schedule, so payload output does not bottleneck the mission's 1 TB data return requirement (SYS-008). PAY-006 requires a lighting system illuminating 5 m ahead of the drone, ensuring visibility for imaging and hazard avoidance during PSR excursions.

The SMD-derived requirements, PAY-SMD-1.1 through PAY-SMD-1.3.3, trace to the mission's first two science objectives. PAY-SMD-1.1 establishes terrain-imaging capability to characterize elevation, slope, albedo, and crater density, requiring 0.03 m/pixel resolution and full operability across the mission's thermal extremes, satisfied by TAGCAMS and the Stereoscopic Terrain Camera. PAY-SMD-1.2 establishes dust/impact-sensing capability to characterize electrostatic regolith charging and micrometeorite impact energy, satisfied by the Dust Impact Mass Spectrometer. PAY-SMD-1.3 establishes radiation-sensing capability to characterize TID, particle energy, and dose rate; its ≥ 100 krad tolerance was set to meet or exceed PAY-003's shielding requirement, satisfied by the Radiation Dosimeter.

The HSMD-derived requirements, PAY-HSMD-2.1 through PAY-HSMD-2.3.1, trace to the mission's third and fourth objectives, supporting safe autonomous mobility rather than direct science return. PAY-HSMD-2.1 establishes inertial-sensing capability to monitor flight dynamics and structural stability, requiring a $\pm 16\text{g}$ range consistent with the Gyro and IMU Suite's specifications. PAY-HSMD-2.2 establishes navigation-support capability to track elevation gain and power consumption for autonomous route planning, satisfied by the Flight Computer Software and Power Monitor. PAY-HSMD-2.3 establishes hazard-detection capability for real-time obstacle avoidance, requiring detection of hazards down to 10 cm across a 1.00–15.0 m range, satisfied by the ToF Depth Camera.

Req #	Requirement	Rationale	Parent Req	Child Req	Verification method	Req met?
PAY-001	The payload subsystem instrumentation shall demonstrate successful completion of simulated space environment testing or presenting verified space flight heritage prior.	Ensures payload instruments pass simulated environments or prior flight heritage, lowering engineering risk before PSR excursions	SYS-007	N/A	Analysis	Met
PAY-002	The payload subsystem shall operate within a temperature range of -233°C (40K) to +120°C in sunlit and PSR surface conditions through thermal regulation and adequate component specification.	Flows down mission thermal environment constraint to ensure functionality across illuminated and PSR terrain.	SYS-009	PAY-SMD-1.1.5	Test	Met
PAY-003	The payload subsystem's electronics shall be shielded to limit Total Ionizing Dose (TID) exposure to under 30 krad over the mission operational lifetime.	Derives from radiation-hardening requirements to protect instrument electronics from surface radiation.	SYS-021	PAY-SMD-1.3.1	Test	Met
PAY-004	The payload subsystem shall acquire and process science data autonomously, without requiring real-time operator commanding.	Flows down customer autonomy constraint to instrument data-acquisition behavior.	SYS-006	N/A	Demonstration	Met
PAY-005	The payload subsystem shall interface with the Command and Data Handling subsystem to transfer science and engineering data at a rate compatible with the mission's UHF local-link and X-Band DTE downlink schedule.	Supports Instrumentation data flow between platforms and Direct-to-Earth (DTE) communication for successful mission operations	SYS-012 CDH-001	N/A	Inspection	Met

PAY--006	The payload subsystem shall include a lighting system capable of illuminating a path 5 meters ahead of the rover, ensuring visibility for navigation and instrumentation during PSR excursions.	Ensures payload instrumentation to clearly see around clearly for efficient topographical mapping, navigation, and hazard avoidance.	SYS-002	N/A	Test	Met
PAY-SMD-1.1	The payload subsystem shall include an optical/terrain-imaging capability sufficient to characterize elevation, slope, surface texture, albedo, and crater density across illuminated and PSR terrain.	Directly supports STM Objective 1 (evaluating terrain/illumination gradient impact on TRN systems).	SYS-002, SYS-004	PAY-SMD-1.1.4, PAY-SMD-1.1.5,	Demonstration	Met
PAY-SMD-1.1.4	The terrain-imaging capability shall capture data at 0.03 m/pixel spatial resolution	Supports continuous TRN evaluation during drone flight excursions.	PAY-SMD-1.1	N/A	Demonstration	Met
PAY-SMD-1.1.5	The terrain-imaging capability shall remain fully operational across the payload's full thermal envelope, -233°C (40K) to +120°C, during active imaging.	Ensures optical/terrain-imaging data collection continues uninterrupted through PSR hazards since imaging degraded by thermal stress would compromise STM Objective 1 data.	PAY-SMD-1.1, PAY-002	N/A	Test	Met
PAY-SMD-1.2	The payload subsystem shall include a dust/impact-sensing capability sufficient to measure electrostatic charge accumulation and impact kinetic energy.	Directly supports STM Objective 2 (characterizing electrostatic regolith mobilization and micrometeorite impact environment).	SYS-002	PAY-SMD-1.2.1, PAY-SMD-1.2.2	Demonstration	Met
PAY-SMD-1.2.1	The dust/impact-sensing capability shall detect electrostatic charge signals across a sensitivity range of 0.1 pA to 100 nA.	Defines detection threshold needed to characterize regolith electrostatic charging behavior.	PAY-SMD-1.2	N/A	Test	Met

PAY-SMD-1.2.2	The dust and impact-sensing capability shall detect impactor kinetic energy across a range of 10 eV to 10 MeV.	Enables characterization of micrometeorite impact energy relevant to surface degradation.	PAY-SMD-1.2	N/A	Test	Met
PAY-SMD-1.3	The payload subsystem shall include a radiation-sensing capability sufficient to measure cumulative Total Ionizing Dose, particle energy spectrum, and dose rate in the lunar south pole environment.	Directly supports STM Objective 2 (evaluating radiation risk to electronics survivability).	SYS-002, SYS-010	PAY-SMD-1.3.1, PAY-SMD-1.3.2, PAY-SMD-1.3.3	Demonstration	Met
PAY-SMD-1.3.1	The radiation-sensing capability shall tolerate a cumulative TID of ≥ 100 krad (Si).	Ensures instrument survivability matches or exceeds the mission's own shielding threshold.	PAY-SMD-1.3, PAY-003	N/A	Test	Met
PAY-SMD-1.3.2	The radiation-sensing capability shall detect Particle Energy Detection at a range of 10 eV to 10 MeV.	Defines the energy spectrum needed to identify peak radiation zones exceeding safe operational thresholds.	PAY-SMD-1.3	N/A	Test	Met
PAY-SMD-1.3.3	The radiation-sensing capability shall measure dose rate across a sensitivity range of 0.1 $\mu\text{Gy/hr}$ to 10 Gy/hr.	Enables continuous dose-rate monitoring to determine electronics survivability during surface operations.	PAY-SMD-1.3	N/A	Test	Met
PAY-HSMD-2.1	The payload subsystem shall include an inertial-sensing capability sufficient to measure vibration amplitude/frequency and linear/angular acceleration during drone flight.	Directly supports STM Objective 3 (evaluating mobility platform navigation reliability while minimizing dynamic forces/vibration).	SYS-003	PAY-HSMD-2.1.2	Demonstration	Met
PAY-HSMD-2.1.2	The inertial-sensing capability shall measure linear/angular acceleration across a range of $\pm 16g$.	Establishes the dynamic range needed to characterize flight maneuvering forces in lunar gravity.	PAY-HSMD-2.1	N/A	Test	Met

PAY-HSMD-2.2	The payload subsystem shall include a navigation-support capability sufficient to record elevation gain and power consumption during autonomous excursions.	Directly supports STM Objective 4 (establishing automated routing protocols connecting landing zones to points of interest).	SYS-004, SYS-006	PAY-HSMD-2.2.1, PAY-HSMD-2.2.2	Demonstration	Met
PAY-HSMD-2.2.1	The navigation-support capability shall compute elevation gain to an accuracy of ≤ 1.00 m.	Ensures route repeatability and terrain-suitability confirmation across energy-optimal and hazard-adverse paths.	PAY-HSMD-2.2	N/A	Test	Met
PAY-HSMD-2.2.2	The navigation-support capability shall monitor power consumption to an accuracy of ≤ 0.50 W·hr.	Supports continuous optimal-path updates for energy savings during autonomous excursions.	PAY-HSMD-2.2, PAY-004	N/A	Test	Met
PAY-HSMD-2.3	The payload subsystem shall include a hazard-detection capability sufficient to identify obstacle range and size during flight.	Directly supports STM Objective 4 (hazard avoidance and prevention of interference with human activities).	SYS-004, SYS-006, SYS-010	PAY-HSMD-2.3.1	Demonstration	Met
PAY-HSMD-2.3.1	The hazard-detection capability shall detect obstacles/hazard features ≥ 10 cm across a range of 1.00 m to 15.0 m.	Defines the minimum feature size and range needed for reliable collision avoidance.	PAY-HSMD-2.3	N/A	Test	Met

Figure 2.5.1.1: Payload Requirements

2.1.5.2. Payload Instrument Overview

The Payload subsystem carries out the drone's science and hazard-sensing role for the mission, organized into the four functional groupings: optical/terrain-imaging, dust/radiation-sensing, inertial-sensing, and navigation/hazard-detection.

TAGCAMS Optical Navigation Suite

TAGCAMS handles the drone's primary optical navigation and terrain imaging, producing 2D imagery used for visual odometry and feature tracking, and, alongside the Stereoscopic Terrain Camera, the elevation, slope, and albedo data required for STM Objective 1.



Figure 2.1.5.2.1: TAGCAMS DVR and Camera Configuration

The instrument uses a three camera head, dual-channel DVR architecture housed in a single stacked unit: two heads dedicated to navigation and a third repurposed for secondary feature tracking and calibration support, all built on a heritage CMOS detector array[41,42]. Each head captures 10 or 12 bit raw imagery compared to 8 bit, keeping downlink volume manageable without discarding the dynamic range needed for low light PSR boundary conditions[41]. Global shutter windowing and binning extend integration time and raise signal to noise ratio in the same low light setting[41].

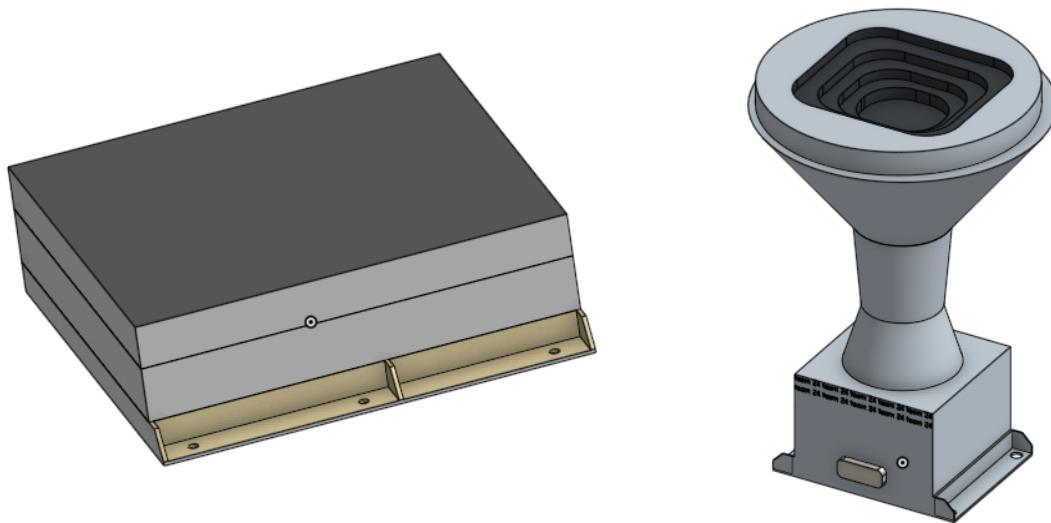


Figure 2.1.5.2.2: TAGCAMS DVR + Camera CAD Model

Spectral range is extended from the 400–700 nm heritage baseline to 380–1050 nm to support near-IR albedo characterization at the south pole[41,43]. Optics are adapted to a combined $75^\circ \times 75^\circ$ field of view against a $60^\circ \times 60^\circ$ requirement, wider than the heritage single head's $44^\circ \times 32^\circ$, since one navigation head's field of view alone could not cover both navigation and terrain simultaneously[41]. TAGCAMS carries a 30 fps design rating, sustaining 7–10 fps once synchronized with the Illumination Floodlight Arrays in PSR darkness; the sensor's 0.03 m/pixel spatial resolution at range keeps this trade from compromising the terrain-mapping objective shared with the Stereoscopic Terrain Camera. TAGCAMS interfaces with Command and Data Handling over SpaceWire for image storage and downlink formatting, and feeds navigation output directly into the Flight Computer Software's route planning logic. The dual-channel, dual-head configuration is redundant by design: losing one head or channel drops the system to single-channel optical navigation without ending an excursion in progress[41]. Camera heads sit inside an actively heated, insulated enclosure to remain within the platform's -233°C to $+120^\circ\text{C}$ envelope, with electronics shielded to the subsystem's 30 krad TID budget.

Stereoscopic Terrain Camera (SCALPSS-derived)

The Stereoscopic Terrain Camera provides the drone's primary 3D photogrammetric mapping, generating stereo derived point clouds and digital elevation models to characterize slope, crater density, and surface texture across illuminated and PSR terrain.

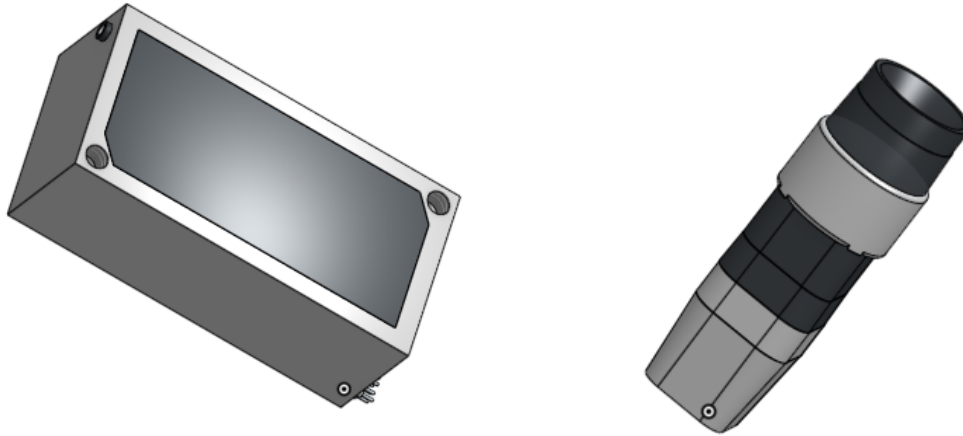


Figure 2.1.5.2.3: SCALPSS Laser and Optic CAD

A dual CMOS stereo pair with fixed baseline separation feeds a Semi-Global Matching disparity processor that reconstructs 3D geometry from paired imagery[45,46]. Since PSR interiors carry no ambient light, the camera operates jointly with the Illumination Floodlight Arrays, and can be paired with structured light projection over featureless terrain to generate synthetic correlation targets[45].



Figure 2.1.5.2.4: SCALPSS Camera CAD + Characteristics

Its rated 54 W peak draw and 6.5 kg mass reflect the dual-camera, illuminator-integrated configuration adapted from SCALPSS's stationary lander baseline, heavier than the lander version since the drone's illumination requirement did not exist in the original design[45]. Photogrammetric accuracy scales with mounting altitude, focal length, and camera separation; processing takes advantage of low grazing-angle sunlight near PSR rims, which

enhances rather than degrades surface contrast for cross correlation[47,48]. The camera interfaces with Command and Data Handling via SpaceWire for point cloud storage and with the Flight Computer Software for real-time hazard geometry, acting as a coarse-rate backup to the ToF Depth Camera if that sensor degrades. It is single-string at the hardware level, an accepted risk offset through cross-instrument coverage with TAGCAMS rather than physical duplication. Camera placement enforces partial self-shading against direct solar thermal loading, and lenses are positioned away from regolith ejecta exposure[49].

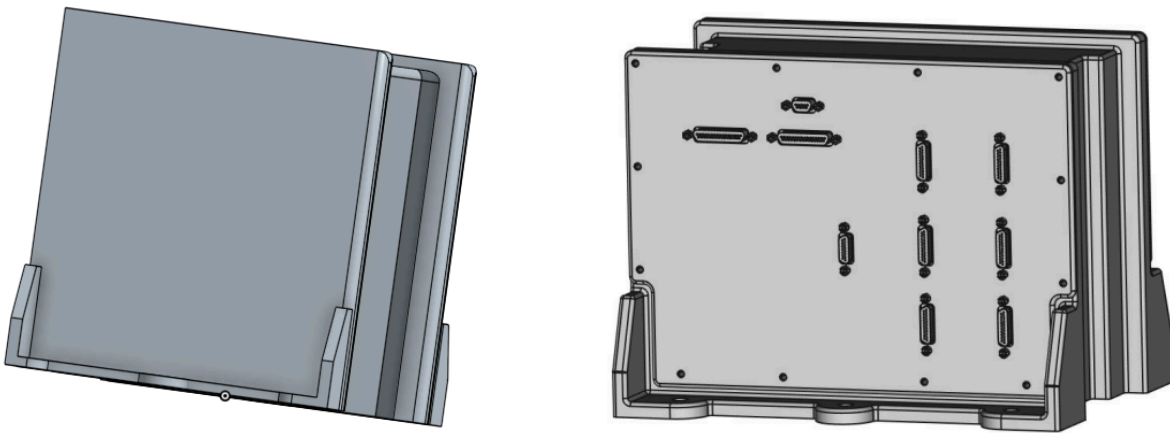


Figure 2.1.5.2.5: SCALPSS MSE-lite + Characteristics

SCALPSS's underlying stereo camera architecture builds on prior NASA robotic lunar lander development work establishing terminal descent and surface-imaging design practices for CLPS-class platforms[50].

Illumination Floodlight Arrays (VIPER-derived)

The Illumination Floodlight Arrays are not a science measurement instrument but an enabling one. Without active lighting, PSR interiors are unusable by either camera in the terrain imaging suite, since PSRs receive zero direct sunlight[51,52]. The arrays consist of two independently switched LED floodlight units synchronized to the imaging cadence of TAGCAMS and the Stereoscopic Terrain Camera, sized to illuminate a path at least 5 m ahead of the drone per PAY-006[52]. Each unit measures approximately 0.08 by 0.06 by 0.05 m and 1 kg combined, small enough that mass is not the driving constraint; the 64 W peak draw is the driving constraint, since it is the highest peak load of any single instrument in the subsystem[52]

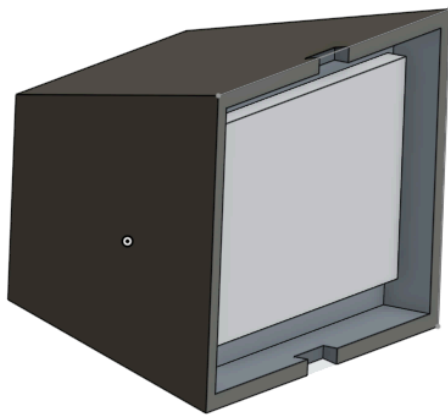


Figure 2.1.5.2.6: Floodlight Array + Characteristics

Since illumination is only needed during active PSR flight segments, an estimated 50% duty cycle drops average draw to roughly 32 W, the figure that governs the subsystem's sustained power budget[52]. The dual-unit configuration is independently switched, so a single array failure reduces illuminated coverage rather than eliminating it. The arrays interface with the optical suite's imaging triggers, with the Power Monitor for duty-cycle-aware energy budgeting, and receive commands over the same SpaceWire architecture as the rest of the payload. Environmental qualification is weighted toward cold survival and optical output stability across the platform's full thermal range rather than TID shielding, since the design driver is thermal/optical performance in permanent cold-dark conditions[51]. This pairing of active illumination with autonomous localization in total darkness mirrors the broader PSR navigation approach validated for lunar darkness operations generally[53].

Dust Impact Mass Spectrometer (LDEX/SUDA-derived)

The Dust Impact Mass Spectrometer characterizes electrostatic regolith charging and micrometeorite impact behavior at the lunar south pole, directly supporting STM Objective 2. It is a time of flight, dual polarity, reflectron type impact mass spectrometer: particles strike a biased target plate, generating an impact plasma whose ion time of flight is measured to derive elemental composition, while charge sensitive amplifiers and velocity grids separately measure the impactor's electrostatic charge and kinetic energy[55,57]. This gives the mission direct in situ characterization of dust mobilization and micrometeorite flux, rather than an inference drawn from remote sensing[56].

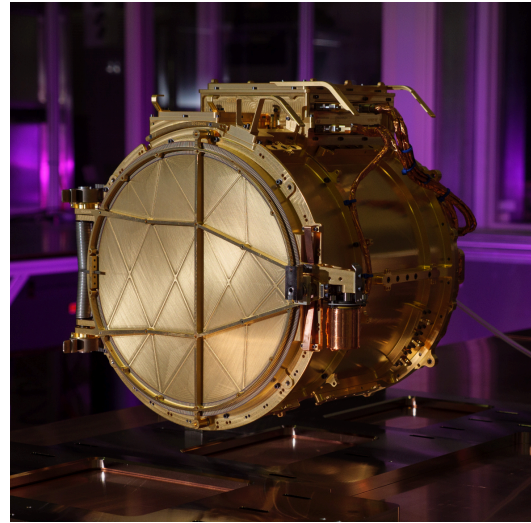
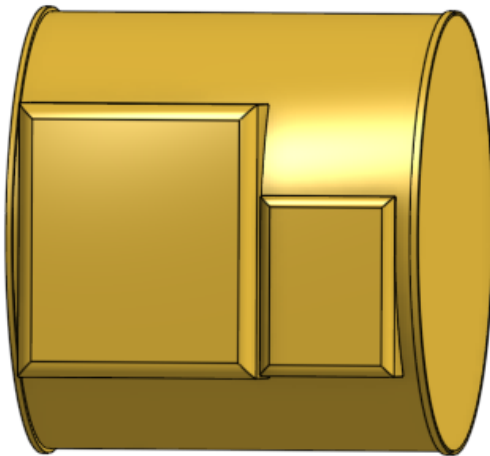


Figure 2.1.5.2.7: SUDA CAD + Characteristics

At 12.57 kg, this is the largest mass allocation in the payload suite, reduced from its interplanetary baseline by stripping roughly 2 kg of Jovian-environment radiation shielding the lunar radiation environment does not require[55,57]. Its 15.3 W peak draw includes decontamination heaters that double as target-conditioning elements, clearing condensed volatile frost between excursions[57]. Calibration is adapted for the lower impact velocities expected from lofted PSR regolith and micrometeorite ejecta, versus the higher-velocity flyby impacts its heritage missions were calibrated against[54,56]. It interfaces with Command and Data Handling over SpaceWire for spectral data storage ahead of DTE downlink, and with the Radiation Spectrometer, which provides secondary context on the charged-particle environment driving electrostatic dust behavior. The sensor head is thermally isolated from the drone chassis by titanium kinematic mounting struts and survives to its rated cold limit through onboard heaters, consistent with the sub-40 K PSR floor temperatures the mission operates through[54].

Radiation Spectrometer/Dosimeter (LETS)

The Radiation Spectrometer continuously measures Total Ionizing Dose, particle energy spectrum, and dose rate at the lunar south pole, directly supporting STM Objective 2's evaluation of long term electronics survivability. Built around a CERN derived Timepix hybrid pixel detector, it registers individual ionizing particle interactions across its silicon detector area, letting it reconstruct both cumulative dose and instantaneous dose rate as the drone moves between sunlit terrain and PSR interiors, where secondary particle fluxes from galactic cosmic ray interactions with regolith can differ from surface fluxes outside the PSR boundary[58].

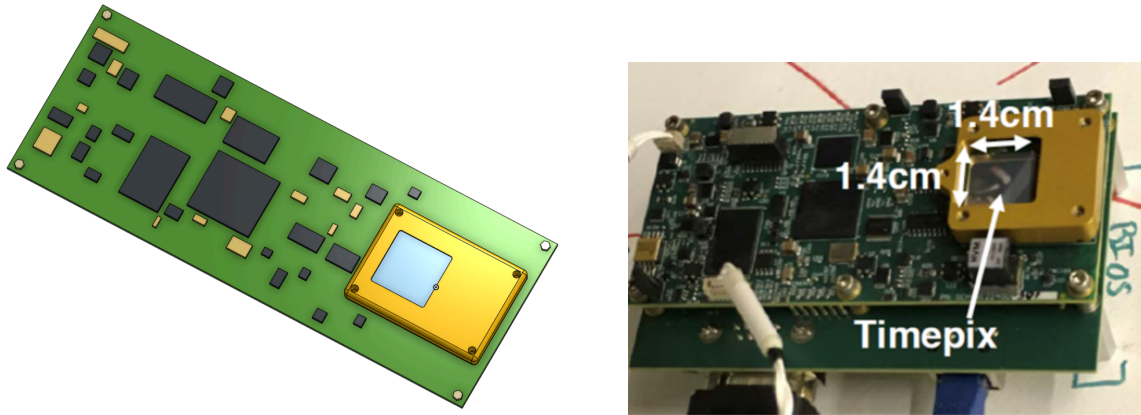


Figure 2.1.5.2.8: LETS Radiation Spectrometer + Characteristics

At 0.43 kg and 2.0 W peak draw, it is the lightest and lowest-power instrument in the dust/radiation suite, reflecting its compact solid-state detector architecture[58]. Its qualified TID tolerance of roughly 130 krad(Si) comfortably exceeds both the 100 krad(Si) requirement in PAY-SMD-1.3.1 and the subsystem's 30 krad shielding budget in PAY-003, the widest design margin in its suite[58]. It shares a CDH interface with the rest of the payload and provides real-time dose-rate telemetry that can inform an excursion abort or reroute if radiation near a PSR feature exceeds safe thresholds, and provides secondary context to the Dust Impact Mass Spectrometer's charging measurements, since particle flux and electrostatic regolith behavior are physically linked[59,60].

Space Grade Gyroscope-IMU (LN-200S)

The LN-200S is one half of the inertial sensing suite's dissimilar redundancy pair, providing three axis angular rate and linear acceleration data used to monitor vibration, structural stability, and flight dynamics throughout each excursion for STM Objective 3. It uses three solid state fiber optic gyroscopes and three MEMS accelerometers in a non dithered, hermetically sealed package with no moving parts, outputting incremental velocity and incremental angle over an RS 422/RS 485 serial interface[61]. Its low white noise and angle random walk let the flight computer dead reckon through PSR darkness when optical sensors are degraded, and its 1,000 degree per second dynamic range and 200 Hz bandwidth support the rapid attitude corrections needed while maneuvering between illuminated and shadowed terrain[61,62].

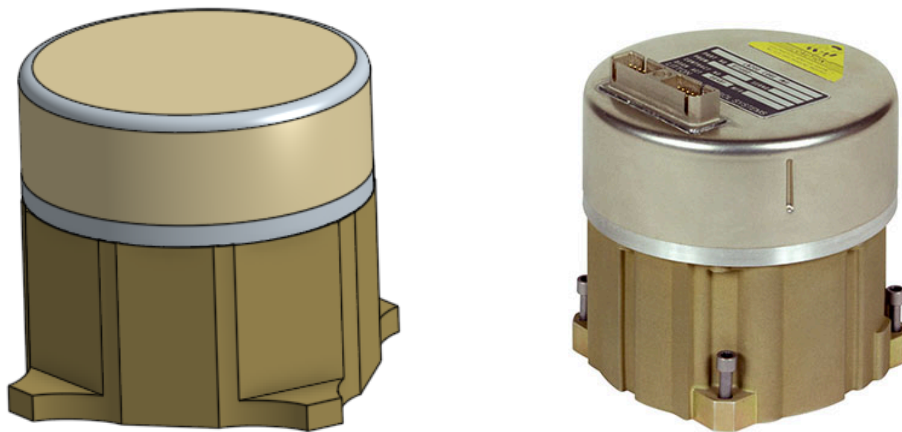


Figure 2.1.5.2.9: LN-200S CAD + Characteristics

At 0.75 kg and 12 W nominal draw, it imposes a small mass and power cost for the flight-dynamics awareness it provides[61]. It is paired with the Honeywell MIMU in a dissimilar-technology configuration, fiber-optic gyroscopes on one unit and ring-laser gyroscopes on the other, with the flight computer fusing both outputs during nominal operations and cross-checking for divergence; losing either unit degrades precision without eliminating flight-dynamics awareness, since the surviving unit operates alone until re-convergence at the next docking period. The unit interfaces with the Flight Computer Software and, indirectly, the Power Monitor, since stable flight dynamics affect energy-optimal path execution. Its hermetically sealed enclosure protects the fiber-optic and MEMS elements from fine lunar dust, and sits inside an insulated enclosure to stay above its operating temperature floor through PSR cold-soak[63,64].

The LN-200S's own rated TID tolerance is 10 krad(Si), below the subsystem's 30 krad shielding budget in PAY-003. This is an accepted risk, offset by housing the unit within the same radiation-shielded enclosure used for thermal protection, keeping effective cumulative exposure under its rated threshold rather than exposing it to the full subsystem-level allowance.

Space Graded IMU Suite (Honeywell MIMU)

The Honeywell MIMU is the second half of the inertial sensing suite's dissimilar pair, using radiation hardened GG1320 ring laser gyroscopes rather than the LN-200S's fiber optic architecture, providing an independent three axis attitude and acceleration reference for STM Objective 3. Its measurement principle relies on the Sagnac effect within its ring laser

cavities to sense rotation, paired with accelerometer channels for linear acceleration, and it outputs data over RS422 and MIL STD 1553B interfaces[65]. Long term bias stability under 0.02 degrees per hour per year and sub 1 ppm scale factor drift give it strong precision across the mission's full multi lunar day excursion cycle, which is the timescale since bias drift compounds over repeated excursions rather than within a single one[65].

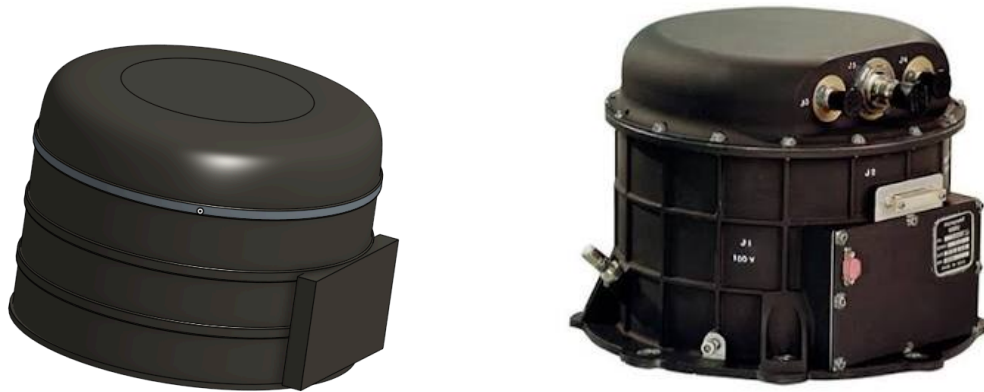


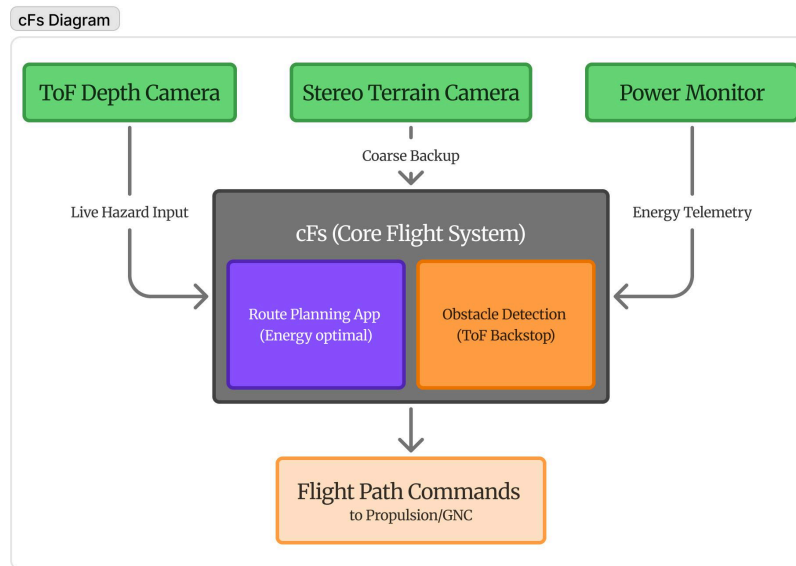
Figure 2.1.5.2.10: Honeywell MIMU CAD + Characteristics

At 4.6 kg and 25 W nominal draw, the MIMU is the heavier, more power-intensive inertial unit, a trade accepted for its radiation-hardened microelectronics and 100 krad(Si) TID qualification, well above the subsystem's 30 krad requirement[65]. The flight computer compares LN-200S and MIMU output continuously; persistent divergence, rather than an internal fault flag, triggers the flight computer to flag the divergent unit and continue on the higher-confidence sensor. The MIMU interfaces with the Flight Computer Software the same way the LN-200S does, and its Ring Laser Gyroscope health telemetry (laser intensity monitoring) gives an early indicator of degradation. Duty-cycling the MIMU off during quiescent, non-maneuvering phases preserves operational life and is carried into the mission's power-conservation dormancy periods [66,67].

Flight Processing Computer Software (cFS)

The Flight Processing Computer Software is the payload's autonomy backbone, executing the drone's pre-planned excursion flight path and updating routing based on live hazard input from the ToF Depth Camera and Stereoscopic Terrain Camera, directly enabling STM Objective 4's autonomous hazard-avoidance requirement. The mission adopts NASA's open-source Core Flight System, a layered, publish-subscribe architecture separating mission-specific logic from the underlying real-time operating system, letting payload

navigation applications run as modular, independently testable components rather than a



single monolithic program.

Figure 2.1.5.2.11: cFs Software Architecture

cFS applications fuse Power Monitor telemetry with live hazard and terrain data to execute energy-optimal path selection, satisfying PAY-HSMD-2.2's navigation support requirement, and its onboard obstacle-detection loops serve as a software-level backstop to the ToF Depth Camera's primary hazard-avoidance role if that sensor degrades. Because cFS runs entirely without real-time operator commanding, it is the component most directly responsible for satisfying PAY-004's autonomy requirement; its behavior must be demonstrated end-to-end during a simulated excursion rather than verified through a single measurable value. As flight software, cFS carries no mass, power, or physical envelope of its own; its environmental consideration is robustness against radiation-induced single-event upsets, addressed through the framework's built-in memory scrubbing and checksum applications rather than physical shielding.

Power Monitor (INA901-SP + ADC128S102QML-SP)

The Power Monitor tracks real-time energy expenditure across the drone's power rails, feeding the Flight Computer Software the consumption data needed for energy-optimal path selection and satisfying PAY-HSMD-2.2.2's requirement to monitor power consumption to within 0.50 W·hr accuracy. It pairs a radiation-hardened current-shunt monitor with a companion ADC in a discrete, physically separated signal chain: the shunt

monitor conditions the small differential voltage across a low-drift foil shunt resistor, and the ADC digitizes that signal for delivery to the flight computer over a four-wire SPI interface, integrating instantaneous power draw into cumulative energy usage[72,73].

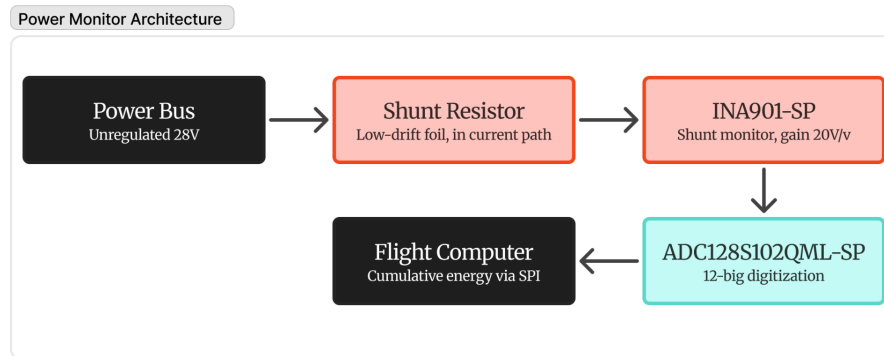


Figure 2.1.5.2.12: Power Monitor Architecture

At under 0.01 kg combined and under 0.02 W across both chips, it is the smallest and lowest-power instrument in the payload suite, reflecting its role as a telemetry sensor rather than an active science instrument[72,73]. Keeping the high-current shunt resistor physically separate from the sensitive analog conditioning electronics avoids the thermal coupling and single-event functional interrupt risk monolithic digital power-monitor ICs carry, typically avoided in critical deep-space applications[74,75]. The Power Monitor interfaces directly with the Flight Computer Software and, functionally, with every powered instrument and subsystem whose current draw it observes. Its role is safety- and efficiency-critical to every excursion despite its minimal footprint, so it is treated as single-string but low-risk, with loss degrading energy-optimal routing precision rather than eliminating flight capability.

Time-of-Flight (ToF) Depth Camera (RESOURCE Camera)

The ToF Depth Camera is the drone's primary real time hazard detection sensor, providing the near field depth data used for autonomous obstacle avoidance under STM Objective 4. Unlike the passive stereo imaging TAGCAMS and the Stereoscopic Terrain Camera perform, the ToF camera actively emits pulsed infrared light and measures return timing to compute distance directly, which makes it functionally independent of ambient or artificial visible spectrum lighting and well suited to total PSR darkness[76,79]. It combines a 1 megapixel time of flight depth sensor, a 12 megapixel RGB camera, and an integrated IMU in one

compact package, delivering sub-centimeter near field resolution for hazard geometry directly in front of the drone[76,78].



Figure 2.1.5.2.13: ToF Resource Camera CAD + Characteristics

At approximately 0.35 kg and 6 W peak draw, it mounts with minimal impact on the drone's mass and power budget, and its active-illumination principle carries lower onboard computational cost than the stereo-matching the passive cameras use, since depth comes directly from pulse timing rather than image correlation[76,79]. It satisfies PAY-HSMD-2.3.1's requirement to detect obstacles of at least 10 cm across a 1.00–15.0 m range, with predicted performance extending detection to smaller features across a wider minimum range[77]. The ToF camera interfaces directly with the Flight Computer Software, supplying the primary real-time input to the hazard-avoidance routing loop, and is the subsystem's primary single-string fault case: if telemetry drops or falls outside expected range, the flight computer's onboard obstacle-detection software and the Stereoscopic Terrain Camera take over at a reduced update rate while the drone slows and increases stand-off distance from terrain features. It retains the Azure Kinect's commercial housing externally, while internal hardware and firmware are substantially modified for spaceflight, including vacuum-compatible internals and flight-qualified processing electronics not present in the COTS baseline[78,80].

Technology Readiness Level (TRL)

Each payload instrument's TRL is assessed against its relevance to the lunar south pole PSR environment specifically, not carried over directly from a vendor's highest achieved flight rating elsewhere, consistent with PAY-001's constraint that no payload instrument exceed TRL 6.

TAGCAMS is assessed at TRL 5. Its underlying ECAM-C50 camera architecture is flight proven at TRL 9 aboard OSIRIS-REx[41], but the mission specific configuration, active illumination synchronization, extended near IR spectral response, and wide FOV optics for PSR operation, has not been demonstrated in a directly relevant environment.

The Stereoscopic Terrain Camera is assessed at TRL 5. Its baseline hardware carries flight heritage on CLPS lunar landers[45], but its adaptation for active illumination stereo imaging aboard a mobile excursion platform, rather than a stationary lander, has not been demonstrated at flight readiness.

The Illumination Floodlight Arrays are assessed at TRL 5. The underlying LED architecture was analog tested for VIPER's PSR operations[51,52], but integration as a synchronized, dual unit array supporting two independent camera systems aboard this drone has not been flight qualified.

The Dust Impact Mass Spectrometer is assessed at TRL 5. Its detection principle is flight proven at TRL 9 through LDEX and SUDA[54,56], but recalibration for the low velocity impact regime expected from lofted PSR regolith, rather than the higher velocity flyby impacts of its heritage missions, has not been demonstrated in a relevant environment.

The Radiation Spectrometer is assessed at TRL 6. Its core Timepix detector operates successfully aboard the ISS and has direct lunar relevant flight heritage through Astrobotic's Peregrine Mission One[58], giving it the strongest environmental relevance of the dust/radiation sensing suite.

The LN-200S is assessed at TRL 6. Its underlying technology is flight proven at TRL 9 across multiple planetary rovers and landers[61,63,64], and while it has not specifically flown inside a lunar PSR, its dead reckoning role and thermal/dust protections are consistent with prior lunar and planetary lander applications.

The Honeywell MIMU is assessed at TRL 6, for the same reason as the LN-200S: flight proven inertial technology with strong lunar lander heritage through InSight and Artemis

l[65], applied here in a role consistent with its established flight history despite the absence of PSR specific demonstration.

cFS is assessed at TRL 6 for its role within the Payload subsystem. The underlying framework is vendor rated at TRL 8-9 across 40 plus missions[68,69], but no prior mission has demonstrated fully autonomous, closed loop hazard avoidance flight control within a lunar PSR excursion profile, so the mission specific application is down-rated relative to the framework's general flight heritage.

The Power Monitor is assessed at TRL 5. Both constituent chips are individually flight proven at TRL 9 in planetary rover power telemetry applications[72,73], but their specific integration supporting real time, energy optimal autonomous routing aboard a PSR excursion platform has not been demonstrated as a combined application, placing it at the subsystem's more conservative rating.

The ToF Depth Camera is assessed at TRL 6. It has direct, relevant flight qualification for the IM-2 mission targeting the Shackleton Connecting Ridge[77], including analog testing specifically for PSR representative low angle lighting and dust conditions, giving it the strongest PSR specific heritage of any instrument in the subsystem.

Following the mission's convention that a subsystem's overall TRL is capped by its lowest rated element, the Payload subsystem's overall TRL is set at TRL 5. Five of the ten instruments, TAGCAMS, the Stereoscopic Terrain Camera, the Illumination Floodlight Arrays, the Dust Impact Mass Spectrometer, and the Power Monitor, carry this rating because their PSR configured operating modes have not yet been demonstrated in a fully relevant environment, even where the underlying hardware individually carries higher flight heritage elsewhere. This is consistent with the PAY-001 ceiling of TRL 6 and points the subsystem's verification planning toward iterative PSR representative testing for these five instruments ahead of PDR, while the TRL 6 components, the Radiation Spectrometer, both inertial units, cFS, and the ToF Depth Camera, carry comparatively lower first pass verification risk.

2.1.5.3. Payload Subsystem Recovery and Redundancy Plans

The Payload subsystem employs a mix of instrument-level and cross-instrument redundancy, tailored to the drone's mass and power envelope rather than duplicating every unit. TAGCAMS is internally redundant through its dual-camera, dual-DVR-channel architecture, so a single camera-head or channel loss retains optical navigation on the surviving path. The LN-200S and Honeywell MIMU form a dissimilar-technology pair,

fiber-optic gyroscopes and MEMS accelerometers on one unit, ring-laser gyroscopes on the other, fusing values during nominal operations; loss of either degrades precision but not flight-dynamics awareness, since the flight computer falls back to the surviving unit. The Illumination Floodlight Arrays carry the same independently switched dual-unit configuration, so a single array failure reduces coverage rather than causing total shutdown.

The Stereoscopic Terrain Camera, Dust Impact Mass Spectrometer, Radiation Spectrometer, Power Monitor, and ToF Depth Camera are single-string, an accepted risk given the subsystem's 31.21 kg and 202.32 W budget, offset through cross-instrument coverage rather than physical duplication. TAGCAMS is similarly accepted at a sustained 7-10 fps rather than its full 30 fps requirement, since its higher native resolution preserves the terrain-mapping objective it shares with the Stereoscopic Terrain Camera.

Legend	
	Does not contribute to objectives
	Does not meet baseline
	Partial contribution to objectives
	Instrument can fully meet objectives

Figure 2.1.5.3.1: Instrument Redundancy Matrix Legend

Science Measurement Requirements		Baseline Performance Requirements		Instrument					
Physical Parameters	Observables			TAGCAM S Optical navigation suite & FOV Stereoscopic Terrain Camera	Dust Impact Mass Spectrometer	Radiation Spectrometer	Space-Gyro & IMU Suite	Flight Computer Software & Power Monitor	RESOU RCE Time-of-Flight (ToF) Depth Camera
Define the elevation (from a location-specific datum), terrain slope changes (using a raster with values per pixel), surface texture, surface albedo, and crater density.	Map 3D topographical point clouds to measure elevation, slope, and hazard geometry. Measure reflected NIR intensity & surface albedo to evaluate how terrain affects navigation performance.	Spatial Resolution:	≤0.10 m/pixel @ 10m range relative						
		Field of View	≥ 60° × 60°	●	○	○	○	●	●
		Spectral Wavelength	400 - 1000 nm						
		Imaging Frame Rate	≥ 30 fps real-time						
Identify electrostatic properties (in nanoAmperes), the elemental material composition of impactors, and impact kinetic energy.	Record electrostatic current accumulation (in nanoAmperes) and collect mass spectra data to determine impactor composition. Measure multi-angle albedo variations to detect micrometeorite impacts.	Impact Charge Signal Sensitivity:	0.1 pA - 100 nA						
		Energy Detection Range	10 eV - 10 MeV						
		Elemental Composition Resolution:	Mass resolution ≤ 2 amu	○	●	●	○	●	○
		Spectral Resolution:	≤ 2.0 nm across 250–800 nm						
Identify the Total Ionizing Dose (TID), energetic particle flux, and energy spectrum in the lunar south pole radiation environment.	Measure radiation dose rates and dose rate continuously to determine electronics survivability during continuous surface operations.	Radiation Sensitivity:	TID Tolerance ≥ 100krad (Si) cumulative						
		Particle Energy Detection Range	10 eV - 10 MeV	○	●	●	○	●	○
		Dose Rate Sensitivity	0.1 µGy/hr – 10 Gy/hr						
Identify vibration amplitude & frequency, linear & angular acceleration, and weight distribution.	Measure structural integrity and gyroscopic position changes before and after transport to quantify how it affects these parameters.	Vibration Frequency Range:	0.1 Hz - 2000 Hz						
		Acceleration Range:	±16g	○	○	○	●	●	○
		Weight sensing Accuracy:	≤ 0.10 N						
Define path length, elevation gain, average and maximum slope, waypoint position, average acceleration, energy usage, and transit time.	Record terrain suitability across energy-optimal, time-optimal, and hazard adverse paths to confirm route repeatability and performance consistency.	Elevation Gain Computation Accuracy:	≤ 1.00 m						
		Waypoint Navigation Tolerance:	≤ 5.0 m waypoint tolerance	○	○	○	○	●	○
		Power Consumption Monitoring Accuracy:	≤ 0.50 W * hr						
Define obstacle detection ranges and accuracy, vehicle reaction times, stopping times, and potential damage variables upon hazard interaction.	Measure collision avoidance success rates, quantifying the specific time and energy expended on hazard avoidance compared to normal travel parameters.	Minimum Feature/Hazard Resolution:	Detect obstacles ≥ 10 cm						
		Reaction time resolution:	≤ 33 ms	●	○	○	○	●	●
		Obstacle detection Range:	1.00 m – 20.0 m (≥10 cm hazards)						

Figure 2.1.5.3.2: Instrument Redundancy Matrix

No instrument fully carries a second objective outside its own design purpose, but several provide partial coverage: the Dust Impact Mass Spectrometer contributes secondary insight into radiation-driven charging behavior, the Radiation Spectrometer contributes partial context to dust testing, and the Flight Computer Software's obstacle-detection loops backstop the ToF Depth Camera's hazard-avoidance role. The IMU pair and Power Monitor satisfy only their own objectives, since flight-dynamics sensing and power telemetry have no elsewhere in the payload suite to fall back on, an acceptable gap given both are already covered by the hardware-level redundancy above.

Recovery

Failure detection is governed by the same CDH fault-management framework used across the mission: system health logs and Limit Checker cFS continuously monitor instrument telemetry and power draw, with all FDIR responses required to detect and act within the mission's 5-minute fault window. For the IMU pair, detection is cross-sensor: the flight computer compares LN-200S and MIMU output in real time, and a persistent divergence,

not an internal fault flag, triggers recovery. For TAGCAMS, loss of a camera head or channel is detected through the same heartbeat mechanism, and the system reverts to single-channel navigation without interrupting the excursion.

If the Radiation Spectrometer or Dust Impact Mass Spectrometer faults, detected through a failed telemetry acknowledgment or an out-of-range value, CDH commands an automated power-cycle and restart; if recovery fails after a set number of attempts, the instrument is flagged for troubleshooting at the next DTE window and the excursion continues with that data stream marked lost. Loss of cross-check agreement between the IMU pair triggers a calibration-recovery step instead: the flight computer flags the divergent unit, continues on the higher-confidence unit for the remainder of the excursion, and re-attempts convergence during the next docking period, when both units can be re-referenced against a known-stationary baseline.

The ToF Depth Camera is the subsystem's primary fault-management case, since its loss directly affects hazard avoidance. If ToF telemetry drops or falls outside expected range, the flight computer's onboard obstacle-detection loops assume interim hazard-sensing responsibility while the drone reduces flight speed and increases stand-off distance, shifting reliance onto the Stereoscopic Terrain Camera and TAGCAMS for coarse detection at a slower update rate. If stereo imaging is degraded at the same time, the drone defaults to aborting the excursion and returning to dock along its most recently verified safe path, consistent with the mission's autonomy constraint since no step requires real-time operator commanding.

2.1.5.4. Payload Subsystem Manufacturing and Procurement Plans

Procurement Overview

The Payload subsystem's procurement strategy prioritizes heritage and flight-proven hardware over novel development wherever the mission's science and mobility objectives allow, reflecting the customer's TRL 6 ceiling on the payload subsystem (PAY-001) and the schedule risk associated with developing new instrumentation from scratch. Suppliers were selected using four criteria applied consistently across the subsystem: direct flight or field heritage relevant to the lunar south pole environment (particularly PSR darkness and cryogenic cold-soak), schedule compatibility with the mission's integration date, mass and power compatibility with the drone's 175 kg and 202.32 W (max) allocation, and cost reasonableness relative to the mission's overall budget cap. Where a primary supplier's heritage hardware requires adaptation for lunar PSR conditions, such as TAGCAMS'

visible-spectrum optics or the Dust Impact Mass Spectrometer's high-velocity Jovian calibration, a backup supplier was identified specifically for comparable underlying sensing technology rather than an identical off-the-shelf match, since no COTS instrument currently exists that was designed for this mission's exact operating envelope. This approach allowed the team to build a payload suite almost entirely from SQCOTS, heritage-reuse, and licensed technology-transfer sources, reducing the schedule and technical risk that would come with funding original instrument development within the mission's Phase A through Phase D timeline.

Component Suppliers

TAGCAMS Optical Navigation Suite is procured via Heritage Hardware Reuse, based on the TAGCAMS design flown on OSIRIS-REx, with Malin Space Science Systems (MSSS) as the primary supplier given their role in the original TAGCAMS development. Johns Hopkins University Applied Physics Laboratory (APL) is identified as the backup supplier, based on their development of DRACO (Didymos Reconnaissance and Asteroid Camera for Optical navigation), a comparable autonomous real-time optical navigation camera flown on DART; APL's demonstrated space-qualified imaging hardware and prior involvement managing the OSIRIS-REx TAGCAMS payload make it a credible fallback with directly applicable design experience rather than an unfamiliar vendor.

Stereoscopic Terrain Camera (SCALPSS-derived) is procured via Heritage Hardware Reuse, with NASA's Langley Research Center as the primary developer, based on the SCALPSS payload's flight heritage on Intuitive Machines' Nova-C (IM-1) Firefly's Blue Ghost lander, and its camera design lineage from the Mars 2020 EDLCam system. The Japan Aerospace Exploration Agency (JAXA) is identified as the backup supplier, based on its stereoscopic terrain-mapping camera utilized on the SELENE lunar orbiter; JAXA's direct lunar terrain-imaging heritage, rather than a general-purpose stereo camera vendor, keeps the backup option technically aligned with similar instrumentation application produced.

Illumination Floodlight Arrays (VIPER-derived) are procured via Heritage Hardware Reuse, with NASA Ames Research Center as the primary developer, based on their design and analog testing of the LED headlight/floodlight system built for the VIPER rover specifically to address PSR darkness at the lunar south pole. Spectrolab (a Boeing company) is identified as the backup supplier, given their flight-qualified LED and optical component manufacturing heritage across multiple NASA and commercial spacecraft programs; while Spectrolab has not developed a lunar-specific floodlight array, their space-qualified LED die

and optics product line provides a credible fallback path for reproducing VIPER's illumination approach.

Dust Impact Mass Spectrometer (SUDA-derived) is procured via University-Led Development, with the University of Colorado Boulder's Laboratory for Atmospheric and Space Physics (LASP) as the primary supplier, based on their design and flight heritage with both the Lunar Dust Experiment (LDEX) aboard LADEE and the Surface Dust Analyzer (SUDA) currently flying aboard Europa Clipper. INFICON is identified as the backup supplier, given their MSolo instrument, a flight-proven COTS mass spectrometer already flown on IM-2 and selected for the VIPER mission. This dual heritage gives LASP a direct design lineage for a dust/impact spectrometer, while INFICON's flight-proven lunar-surface deployment makes it a genuinely comparable backup for this mission.

Radiation Spectrometer (LETS) is procured via Technology Transfer/Licensing, with NASA Johnson Space Center's RadWorks Group as the primary source, based on their design of the Linear Energy Transfer Spectrometer (LETS), which flew on Astrobotic's Peregrine Mission One. The German Aerospace Center's (DLR) M-42 detector is identified as the backup supplier, given comparable radiation-sensing characteristics and a similar Timepix-class detector architecture; although M-42 carries less direct lunar-surface flight heritage than LETS, it shares enough underlying sensing technology that a supplier switch would not require a redesign of the instrument's science output format.

Space Grade Gyroscope-IMU (LN-200S) is procured as SQCOTS hardware, with Northrop Grumman as the primary supplier, selected for the LN-200S's extensive flight heritage across the Mars Exploration Rovers, Curiosity, and Perseverance. Safran Sensing Technologies is identified as the backup supplier, based on the flight heritage of their Astrix-series fiber-optic gyros on ESA-class missions; both suppliers offer fiber-optic gyroscope technology with similar performance specs, preferably for its main emphasis on gyroscope integration.

Space Graded IMU Suite (Honeywell MIMU) is procured as SQCOTS hardware directly from Honeywell Aerospace, selected for the MIMU's radiation-hardened design (100 krad TID tolerance) and entry-descent-landing heritage on Mars 2020 and InSight. Safran Sensing Technologies Norway AS is identified as the backup supplier, given a comparable qualified ring-laser-gyro IMU with similar bias stability and TID tolerance specifications; this keeps the dissimilar-redundancy pairing with the LN-200S intact regardless of which supplier is ultimately used.

Flight Processing Computer Software (cFS) is procured via Open Source Software, with NASA Goddard Space Flight Center as the primary source for the Core Flight System (cFS), a reusable, open-source flight software framework with broad mission heritage including direct lunar-lander deployment on LADEE and Nova-C/EagleCam. Simulink (MathWorks) is identified as the backup supplier, given its established simulation and code-generation tooling for flight software development; because cFS is open-source rather than proprietary, this backup functions less as a drop-in replacement and more as a fallback should the team need to create custom flight-control logic outside the standard cFS application library.

Power Monitor (INA901-SP + ADC128S102QML-SP) is procured as SQCOTS hardware, with Texas Instruments as the primary supplier for both the current-shunt monitor and the companion analog-to-digital converter, selected for the discrete signal-chain architecture's demonstrated flight precedent on planetary rovers including Curiosity and Perseverance, and its avoidance of the Single Event Functional Interrupt risk. Analog Devices is identified as the backup supplier, given their comparable radiation-hardened current-shunt monitor and SAR ADC product lines with similar TID ratings and SPI-compatible digital interfaces; keeping both stages of the signal chain sourced from a single backup vendor, rather than mixing suppliers across the shunt monitor and ADC.

Time-of-Flight (ToF) Depth Camera (RESOURCE Camera) is procured via Technology Transfer/Licensing, based on the RESOURCE payload developed by MIT Media Lab's Space Exploration Initiative, which adapts the Microsoft Azure Kinect time-of-flight sensor for lunar PSR hazard detection and was flight-qualified for the IM-2 mission aboard the Lunar Outpost MAPP rover. Teledyne Space is identified as the backup supplier, given their ToF and imaging sensor component manufacturing product lines; because RESOURCE itself is a modified COTS device rather than a fully custom instrument, Teledyne's sensor-component manufacturing background offers a credible path to reproducing a similar depth-sensing capability.

Lead Time Summary

Payload instrument lead times vary considerably due to the mix of procurement methods used across the subsystem. SQCOTS-sourced hardware, including the Space Grade Gyroscope-IMU, Space Graded IMU Suite, and Power Monitor components, carries the most predictable lead times, since these are catalog parts with established qualification records rather than mission-specific adaptations. Heritage- and university-developed instruments, including the TAGCAMS derivative, Stereoscopic Terrain Camera, Dust Impact

Mass Spectrometer, Illumination Floodlight Arrays, and ToF Depth Camera, carry longer and less certain lead times, since each requires design adaptation from a previous mission's configuration, whether that means integrating active illumination into TAGCAMS' visible-spectrum optics, recalibrating the Dust Impact Mass Spectrometer for low-velocity PSR impacts, or reproducing VIPER's floodlight design outside NASA Ames' own development pipeline, rather than pulling a directly compatible catalog part. At the subsystem level, Payload/Science Instruments is estimated at a maximum of 28 weeks build/development time, plus 4 weeks delivery/integration, plus 6 weeks additional if a backup supplier or institution must be used, for a total of 28–38 weeks. If a backup supplier is invoked for any single component, the added schedule contingency does not necessarily delay the full subsystem, since most instruments can be integrated independently, but it does compress the available margin ahead of the December 1, 2028 integration date if more than one backup path is triggered concurrently, particularly for the Stereoscopic Terrain Camera and Dust Impact Mass Spectrometer, whose backup suppliers (JAXA and INFICON, respectively) sit outside the primary supplier's existing design lineage and would likely require additional interface verification before integration could proceed.

Subsystem / Component	Procurement Method	Primary Supplier	Backup Supplier	Estimated Lead Time	Justification
<i>TAGCAMS Optical Navigation Suite</i>	Heritage Har... ▾	Malin Space Science Systems (MSSS)	Johns Hopkins University Applied Physics Laboratory (APL)	18-24 weeks	MSSS holds extensive NASA flight-camera design/manufacturing heritage; APL backup based on DRACO's comparable autonomous optical navigation heritage and prior TAGCAMS program involvement
<i>StereoScopic Terrain Camera (SCALPSS)</i>	Heritage Har... ▾	NASA's Langley Research Center	Japan Aerospace Exploration Agency - JAXA	24-30 weeks	SCALPSS was developed at NASA Langley and has flown on IM-1 (Nova-C) with camera design heritage from the Mars 2020 EDLCam system; JAXA backup based on its stereoscopic terrain camera utilization on the SELENE lunar orbiter.
<i>Dust Impact Mass Spectrometer (SUDA)</i>	University-L... ▾	Development University of Colorado	INFICON Technology (msolo)	14-18 weeks	SUDA/LDEX heritage designed and built at LASP, with direct flight

		Boulder - LASP			heritage on LADEE and Europa Clipper; INFICON backup based on MSolo, a flight-proven COTS mass spectrometer flown on IM-2 (PRIME-1) and selected for VIPER.
<i>Illumination Floodlight Arrays (VIPER)</i>	Heritage Har... ▾	NASA Ames Research Center	Spectrolab (Boeing)	16-22 weeks	Ames developed and analog-tested the VIPER floodlight system specifically for PSR darkness conditions; Spectrolab backup based on flight-qualified LED and optical component manufacturing heritage.
<i>Space Grade Gyroscope/IMU (LN-200S)</i>	SQCOTS (S... ▾	Northrop Grumman	Safran (Astrix 200)	16-20 weeks	LN-200S has extensive flight heritage (MER, Curiosity, Perseverance); Safran backup based on Astrix-series flight heritage on ESA-class missions.
<i>Space Graded IMU Suite (Honeywell MIMU)</i>	SQCOTS (S... ▾	Honeywell Aerospace	Safran Sensing Technologies Norway AS	13-18 weeks	MIMU is radiation-hardened to 100 krad TID with EDL flight heritage on Mars 2020; Safran backup based on comparable ring-laser-gyro IMU specifications.
<i>Flight Processing Computer Software (cFs)</i>	Open Sourc... ▾	NASA's Goddard Space Flight Center	SimuLink (MathWorks)	1-6 week (to acquire)	Core Flight System (cFS) is NASA's open-source, reusable flight software framework with broad mission heritage; Simulink as backup development tooling for custom flight-control logic.
<i>Radiation Dosimeter/Spectrometer (LETS)</i>	Technology ... ▾	NASA JSC RadWorks Group	German Aerospace Center (DLR)	12-20 weeks	LETS was developed by NASA JSC RadWorks Group and used on Astrobotic's Peregrine lander; DLR M-42 backup based on

					comparable Timepix-class detector characteristics.
<i>Power Monitor (INA901-SP + ADC128S102QML-SP)</i>	SQCOTS (S... ▾)	Texas Instrument	Analog Devices	10-16 weeks	INA901-SP/ADC128S102QML-SP discrete signal chain has direct flight precedent on Curiosity and Perseverance power telemetry; Analog Devices backup based on comparable rad-hard shunt-monitor and SAR ADC product lines.
<i>Time-of-Flight (ToF) Depth Camera (RESOURCE Camera)</i>	Technology ... ▾	MIT Media Lab Space Exploration Initiative	Teledyne Space	22-28 weeks	RESOURCE adapts Microsoft Azure Kinect ToF sensing for lunar PSR hazard detection; Teledyne backup based on their ToF/imaging sensor component manufacturing.

Figure 2.1.5.4.1: Procurement Summary Table

2.1.5.5. Payload Subsystem Verification Plans

The Payload subsystem's verification approach is built directly off the payload requirements, using the same four verification methods applied across the mission's subsystems: Analysis, Inspection, Test, and Demonstration. Verification demonstrates that each payload requirement has been met by tracing a documented, repeatable method back to its parent mission requirement, ensuring that instrument selection, environmental survivability, and autonomous behavior can all be confirmed consistently with the instrument trade studies conducted during procurement. Since several payload requirements are structured as parent and child reqs (such as PAY-SMD-1.1's terrain-imaging capability being a parent req to PAY-SMD-1.1.4 and PAY-SMD-1.1.5's spatial resolution and thermal survival thresholds), verification of the parent requirement in these cases is confirmed once its associated child requirements have each also been verified. The payload subsystem is primarily composed of heritage vendors, university led teams, and hardware reuse, and thus requires further testing and analysis for system verification. Other procurements such as SQCOTS also require further testing, but are verified through vendor data specs and overall inspection. Verification priority is also indicated by the subsystem's TRL constraint, with instruments assigned TRL 5 under PAY-001 (TAGCAMS, Stereoscopic Terrain Camera, Illumination Floodlight Arrays, Dust Impact Mass Spectrometer, Power Monitor) receiving earlier and more extensive verification attention,

since these represent the subsystem's least-matured components and the greatest risk to PSR excursions.

Req #	Requirement Summary	Verification Method	Rationale for Method	Preliminary Verification Plan
<i>PAY-001</i>	Payload instrumentation TRL shall not exceed 6	Analysis	TRL is a documented status confirmed through vendor and heritage records	Review of vendor TRL documentation and test heritage records for each instrument, psr excursion environment known to be demanding/challenging, thus most instrumentation does not pass TRL6.
<i>PAY-002</i>	Payload operates within -233°C to +120°C	Test	Thermal survival under PSR cold-soak and sunlit extremes can only be confirmed through active environmental exposure	Thermal-vacuum chamber testing of assembled payload hardware across -233°C to +120°C range, success defined as continued nominal operation through range
<i>PAY-003</i>	Payload electronics shielded to <30 krad TID	Test	Radiation tolerance must be confirmed through direct radiation exposure testing of hardware	TID exposure testing of payload electronics at a radiation test facility, success defined as continued operation after 30 krad exposure
<i>PAY-004</i>	Payload acquires/processes data autonomously	Demonstration	Autonomous behavior is a live system response that must be shown operating end to end	Demonstration of instrument data acquisition and processing during a simulated excursion with no external aid, success defined as completed acquisition cycle without manual intervention
<i>PAY-005</i>	Payload interfaces with CDH at UHF/X-Band compatible rates	Inspection	Interface compatibility is confirmed through architecture and datasheet review	CDH ensures prior testing, double checking configuration and ensuring components are physically compatible, success defined as confirmed rate compatibility
<i>PAY-006</i>	Lighting system illuminates 5 m ahead of drone	Test	Illumination range and intensity are measurable characteristics confirmed through active testing	Bench and field test of the Illumination Floodlight Arrays against a marked 5 m target distance in simulated PSR darkness, success defined as sufficient illumination confirmed for TAGCAMS and Stereoscopic Terrain Camera imaging
<i>PAY-SMD-1.1</i>	Optical/terrain-imaging capability to measure elevation, slope, texture, albedo, crater density	Demonstration	Capability is best confirmed through live operation combining multiple instruments	Demonstration of combined TAGCAMS and Stereoscopic Terrain Camera output against known terrain features in an analog test bed, success defined as accurate characterization across all parameters
<i>PAY-SMD-1.1.4</i>	Terrain-imaging captures data at 0.03 m/pixel spatial resolution	Demonstration	Resolution performance is confirmed through direct imaging output during simulated operation	Demonstration of imaging output against a calibrated resolution target at 10 m range, success defined as resolution meeting or exceeding 0.03 m/pixel

<i>PAY-SMD-1.1.5</i>	Terrain-imaging remains operational across -233°C to +120°C	Test	Thermal survival during active imaging can only be confirmed through direct environmental exposure while the instrument is operating, not through datasheet review alone	Thermal-vacuum chamber testing of TAGCAMS and the Stereoscopic Terrain Camera while operational across the full -233°C to +120°C range, success defined as continued image capture and data quality within spec throughout the range.
<i>PAY-SMD-1.2</i>	Dust/impact-sensing capability measures charge accumulation and impact energy	Demonstration	best confirmed through live sensor operation under simulated impact conditions	Demonstration of the Dust Impact Mass Spectrometer against a calibrated particle source, success defined as detected charge and energy values within expected ranges
<i>PAY-SMD-1.2.1</i>	Electrostatic charge signal sensitivity 0.1 pA - 100 nA	Test	Sensitivity threshold is a measurable performance value confirmed under controlled input	Bench test of the Dust Impact Mass Spectrometer using a calibrated charge source across the full sensitivity range, success defined as accurate detection
<i>PAY-SMD-1.2.2</i>	Impactor kinetic energy detection 10 eV - 10 MeV	Test	Energy detection range is a measurable performance value confirmed under controlled input	Bench test using a calibrated particle accelerator or equivalent source across the specified energy range, success defined as accurate detection
<i>PAY-SMD-1.3</i>	Radiation-sensing capability measures TID, particle energy, dose rate	Demonstration	Capability is best confirmed through live sensor operation under a simulated radiation environment	Demonstration of the Radiation Spectrometer against a calibrated radiation source, success defined as accurate reporting of dose rate, TID, and particle energy
<i>PAY-SMD-1.3.1</i>	Radiation-sensing tolerates ≥ 100 krad (Si) cumulative TID	Test	Radiation tolerance must be confirmed through direct radiation exposure testing of hardware	TID exposure testing of the Radiation Spectrometer at a radiation test facility, success defined as continued operation after 100 krad exposure
<i>PAY-SMD-1.3.2</i>	Particle energy detection range 10 eV - 10 MeV	Test	Energy detection range is a measurable performance value confirmed under controlled input	Bench test using a calibrated radiation source across the specified energy range, success defined as accurate detection at expected tolerances
<i>PAY-SMD-1.3.3</i>	Dose rate sensitivity 0.1 μ Gy/hr - 10 Gy/hr	Test	Dose rate sensitivity is a measurable performance value confirmed under controlled input	Bench test using a calibrated radiation source producing known dose rates across range, success defined as accurate detection at known tolerances
<i>PAY-HSMD-2.1</i>	Inertial-sensing capability measures vibration and linear/angular acceleration	Demonstration	Capability is best confirmed through live sensor operation during simulated flight dynamics	Demonstration of the Space Grade Gyro and IMU pair on a vibration/shaker table simulating drone flight conditions, success defined as accurate reporting of vibration and acceleration data
<i>PAY-HSMD-2.1.2</i>	Linear/angular acceleration range $\pm 16g$	Test	Dynamic range is a measurable performance value confirmed under controlled input	Bench test on a controlled acceleration/shaker table across the full $\pm 16g$ range, success defined as accurate measurement at tolerable ranges

<i>PAY-HSMD-2.2</i>	Navigation-support records elevation gain and power consumption	Demonstration	Capability is best confirmed through live operation combining flight software and power telemetry	Demonstration of the Flight Computer Software and Power Monitor during a simulated excursion, success defined as accurate elevation and power data recorded
<i>PAY-HSMD-2.2.1</i>	Elevation gain computation accuracy ≤ 1.00 m	Test	Computation accuracy is a measurable performance value confirmed against known terrain	Bench/field test comparing computed elevation gain against surveyed reference terrain, success defined as accuracy at or below 1.00 m
<i>PAY-HSMD-2.2.2</i>	Power consumption monitoring accuracy ≤ 0.50 W·hr	Test	Monitoring accuracy is a measurable performance value confirmed against a known power draw	Bench test comparing Power Monitor readings against a calibrated reference load, success defined as accuracy at or below 0.50 W·hr
<i>PAY-HSMD-2.3</i>	Hazard-detection capability identifies obstacle range and size during flight	Demonstration	Capability is best confirmed through live sensor operation against representative obstacles	Demonstration of the ToF Depth Camera against staged obstacles of varying size, success defined as consistent detection and range reporting
<i>PAY-HSMD-2.3.1</i>	Obstacle detection ≥ 10 cm across 1.00-15.0 m range	Test	Detection range and minimum feature size are measurable sensor characteristics confirmed under controlled conditions	Bench and field test the ToF Depth Camera against calibrated obstacle targets across specified range, success defined as consistent detection of obstacles at or above 10 cm

Figure 2.1.5.5.1: Payload Verification Matrix

Verification Method Rationale

Each verification method was selected based on which approach most directly and efficiently confirms that a requirement has been met, without over-testing requirements that can be confirmed simply through documentation. Analysis is applied to PAY-001's TRL constraint, since maturity status can be verified against vendor and heritage documentation before hardware is even procured. Inspection is applied to PAY-005's interface compatibility, where architecture and datasheet review against CDH's documented bandwidth capacity is sufficient without requiring active data transmission. Test is applied to requirements that depend on active hardware operation under simulated or environmental conditions, such as thermal survival (PAY-002), radiation tolerance (PAY-003, PAY-SMD-1.3.1), and sensitivity/range thresholds across the dust, radiation, inertial, and hazard-detection instruments, since these values can only be properly confirmed once hardware is assessed under conditions representative of the lunar south pole environment. Demonstration is reserved for capability-level parent requirements where multiple instruments or subsystem behaviors must be shown working together in real time, such as PAY-004's autonomous acquisition behavior and the combined-instrument capability requirements (PAY-SMD-1.1, PAY-SMD-1.2, PAY-SMD-1.3, PAY-HSMD-2.1, PAY-HSMD-2.2, PAY-HSMD-2.3), rather than confirming a single measurable

value in isolation. Where a requirement is verified through its child requirements rather than independently, the parent requirement is stated as satisfied once each associated child has individually passed its own Test or Analysis. This avoids duplicating verification effort across a requirement and its breakdown, while ensuring the parent-level capability is only confirmed once every underlying performance threshold has been checked.

Preliminary Verification Plan

Payload verification follows the same phased approach used across the mission, beginning with vendor documentation-based Analysis and Inspection items during instrument procurement, since TRL status (PAY-001) and interface compatibility (PAY-005) can both be confirmed before hardware is delivered. Requirements verified through Test, including thermal survival (PAY-002), radiation tolerance (PAY-003, PAY-SMD-1.3.1), and instrument-level sensitivity and range thresholds, require access to environmental test facilities (thermal-vacuum chamber, radiation test facility, calibrated particle/radiation sources) and are planned for the subsystem's dedicated Testing phase, ideally consolidated into the same integrated test window (TVAC, radiation exposure, vibration/shaker table) used elsewhere in the mission to reduce facility scheduling overhead. Requirements verified through Demonstration, including autonomous data acquisition (PAY-004) and the combined-instrument capability requirements, are planned for the Integration phase once individual instruments are mounted to the drone and connected to the Flight Computer Software, since these behaviors depend on the full operational payload architecture rather than individual components in isolation. A key limitation is that several instruments currently carry TRL 5 under PAY-001 (TAGCAMS, Stereoscopic Terrain Camera, Illumination Floodlight Arrays, Dust Impact Mass Spectrometer, Power Monitor), meaning their PSR-specific configuration, particularly active-illumination integration for the optical suite, has not yet been demonstrated in a relevant environment; verification planning for these instruments should therefore include margin for iterative testing and configuration adjustment ahead of PDR, rather than assuming first-pass success as may be reasonable for the subsystem's TRL 6 heritage components (Radiation Spectrometer, Gyro-IMU pair, Flight Computer Software).

2.2. Interface Control

The Base Station serves as the mission's stationary hub, organized into six subsystems: Direct-to-Earth (DTE) Communication, Command and Data Handling (CDH), Local Link Communication, Electrical Power, Mechanical, and Thermal Control. The Base Station is positioned outside the PSR boundary to maintain continuous solar illumination, providing power generation and storage, command authority, bulk data archiving, and the sole DTE relay path to Earth, while relaying commands to and receiving telemetry and science data from the Drone over local UHF link.

The Drone is the mission's mobile platform, organized into seven subsystems: Payload Instrument Suite, Guidance, Navigation, and Control (GNC), Command and Data Handling (CDH), Electrical Power, Local Link Communication, Thermal Control, and Mechanical. The Drone executes autonomous excursions into and adjacent to PSRs, collecting science and engineering data, storing onboard and relays to the Base Station upon return, drawing on the same CDH and power architecture used at the Base Station to sustain operation.

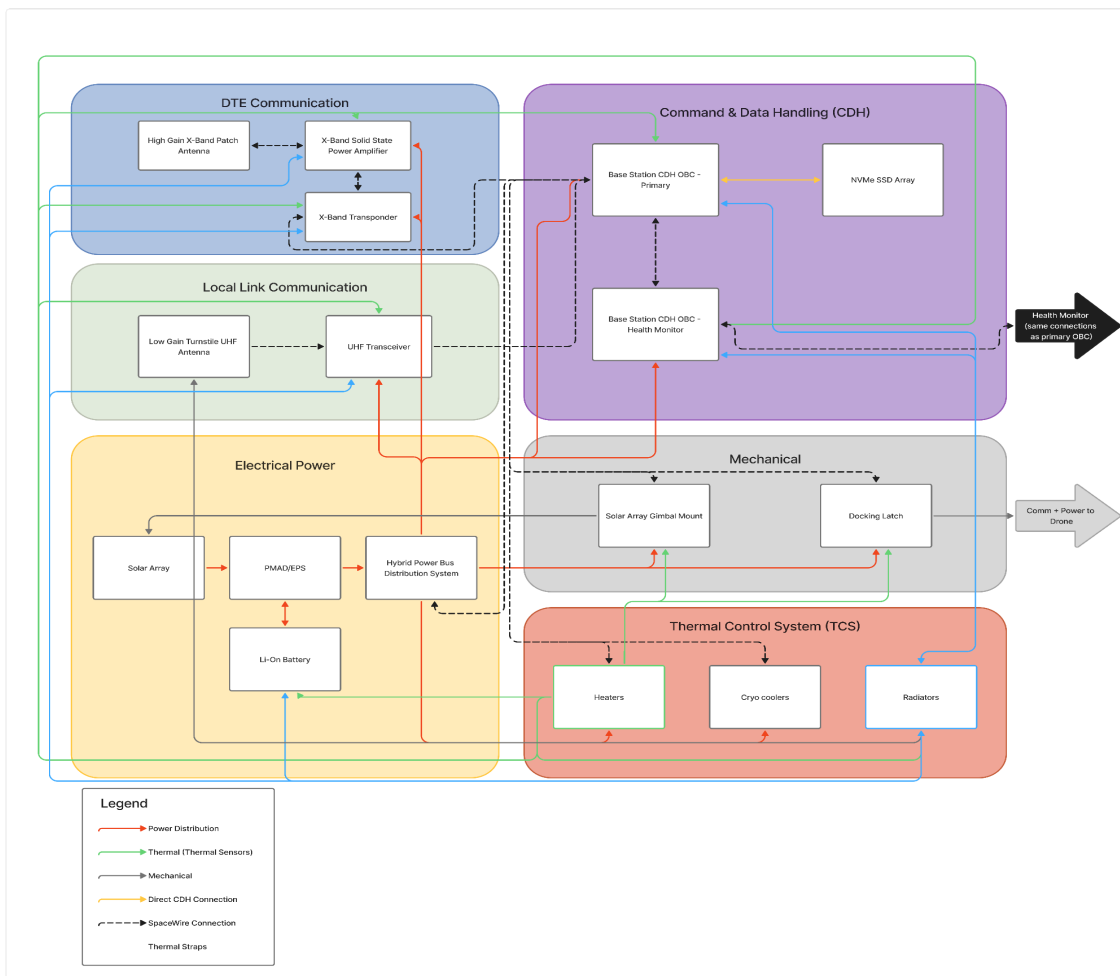


Figure 2.2.1: Base Station Block Diagram

Power originates at the solar array, which feeds the PMAD/EPS. The PMAD/EPS charges the Li-Ion battery and drives the hybrid power bus, which distributes regulated power to every active subsystem, including the UHF and X-Band transceivers/transponder, the SSPA, both CDH OBCs, the solar array gimbal mount, the docking latch, and the thermal control hardware.

Commands flow in two directions. Uplinked commands arrive via the X-Band Transponder, are demodulated, and pass to the primary OBC over a direct CDH connection for validation and routing. Drone-directed commands are then relayed out over the UHF Transceiver to the local link. Telemetry follows the reverse path: subsystem health data is collected by the OBC and formatted for downlink through the Transponder-SSPA-Antenna chain.

Science data arrives at the Base Station over the local UHF link from the Drone following each excursion, is written to the dual NVMe SSD array, and is queued for DTE downlink during scheduled contact windows. The direct (non-SpaceWire) CDH connection reserved for the primary OBC and SSD array reflects this path's priority: bulk science data write throughput must not compete with the general SpaceWire bus.

Thermal control is applied selectively. Heaters and radiators, monitored via thermal sensors, connect to the electronics that are temperature-sensitive: the X-Band Transponder, SSPA, UHF Transceiver, Li-Ion battery, and both CDH OBCs. Heaters extend to the solar array gimbal mount and docking latch, since bearing tolerances in moving mechanisms are vulnerable to the mission's -233°C floor. Consistent with standard practice, neither antenna carries a power or thermal connection: both are passive radiating elements with no active electronics, connected only by an RF feed.

Mechanical support ties the solar array gimbal mount to the solar array through a direct mechanical linkage, while the docking latch provides the physical and power/data interface to the Drone, supplying charging current and communication access during each docking period. This docking interface is a second critical path for the mission: it is the only point at which the Drone receives power and offloads data, so its reliability directly gates the excursion cycle.

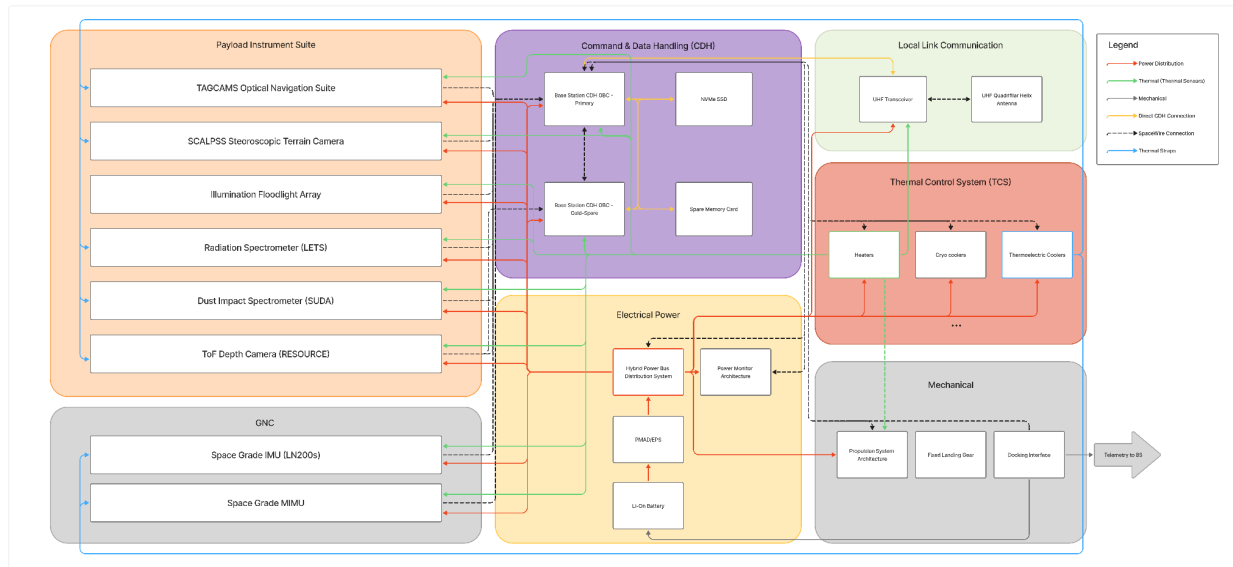


Figure 2.2.2: Scout Drone Block Diagram

Power is drawn from the Drone's own Li-Ion battery through the PMAD/EPs to a hybrid power bus, which distributes to every active load: the payload instrument suite (TAGCAMS, SCALPSS, the illumination floodlight arrays, both spectrometers, and the ToF Depth Camera), the Space-Grade IMU and MIMU, the propulsion system, the Power Monitor, the thermal control hardware, and the UHF Transceiver. During docking, the docking interface itself becomes a power interface, since it carries charging current from the Base Station back into the Li-Ion battery.

Commands and telemetry route through the dual CDH OBCs (primary and cold-spares), which interface directly with the NVMe SSD and spare memory card over a dedicated, non-SpaceWire connection, keeping bulk storage write throughput independent of spacewire throughput. Every other subsystem, including the payload suite, GNC, propulsion, and the Power Monitor, connects to the OBC over SpaceWire for command and status exchange. The UHF Transceiver-to-Quadri-filar Helix Antenna link is the one exception: this is an RF feed carrying local-link signal, not a SpaceWire data connection, and thus labeled as direct CDH connection.

Science data is generated by the payload instrument suite during each excursion and streamed to the OBC over SpaceWire for storage ahead of the return-and-relay cycle. GNC data follows a related but distinct path: the IMU and MIMU report to the OBC for sensor fusion rather than direct connection to propulsion, since the fused attitude and acceleration state, not raw sensor output, is what informs propulsion commands and makes this critical for flight safety.

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Thermal control is provided by heaters, cryocoolers, and thermoelectric coolers. Heaters and thermoelectric coolers connect to the payload instrument suite and to GNC (IMU/MIMU), covering the electronics most exposed to the excursion's swing between illuminated terrain and PSR cold-soak. Cryocoolers are reserved specifically for the optical sensors most sensitive to thermal noise, TAGCAMS and the SCALPSS-derived Stereoscopic Terrain Camera. Propulsion also carries a heater connection given its thermally sensitive valves and lines, since -233°C exposure during PSR excursions poses the same freeze/binding risk to propulsion hardware as it does to any other mechanism on the vehicle.

Mechanical support includes the propulsion system architecture, the fixed landing gear, and the docking interface. The fixed landing gear carries no functional interfaces, being a passive structure. The docking interface provides both the mechanical latch and the dual power/data path back to the Base Station.

N² chart Narrative

The N² chart provides a systematic method for identifying and verifying every interface between subsystems on a given platform. Each system on the diagonal represents a subsystem, and any populated off-diagonal cell represents a real interface between the row (output) and column (input) subsystem, read left to right. Blank cells assert that no interface exists between two subsystems. Applied to the Base Station and Drone, the chart surfaces where a given subsystem's function depends on receiving something from another subsystem before it can operate.

Inputs from External	Docking (latch from base station)	Dust/radiation/thermal exposure	commands + relay window (UHF)		Thermal Load (lunar surface)	Chargin current (From base station)
Drone Interfaces						
Internal Interfaces	Spacecraft Bus Structure	Mechanical interface	Mechanical interface	Mechanical interface	Mechanical interface	Mechanical interface
		Payload	Science + engineering data			
		Commands/control (spacewire)	Command and Data Handling (CDH)	Telemetry + data for relay		
			Received commands	Telecomms		
			Sensor Fusion		Guidance Navigation and Control	
	Heating (propulsion valves/lines)	Power (instruments, Power Monitor)	Heating (OBCs)	Heating (Transceiver)	Heating/TEC (IMU/MIMU)	Thermal
	Power (propulsion)	Power (instruments, power monitor)	Power (OBCs)	Power (UHF Transceiver)	Power (IMU and MIMU)	Power (Heaters/TECs/cryocooler)
Outputs to External	Docking (latch to base station)		Science + engineering data telemetry		Waste heat (radiated)	

Inputs from External	Mechanical (Mounting to Launch Vehicle)	Dust/radiation/thermal exposure	Data (X-band uplink; local link data (drone)		Thermal load (lunar surface)	Solar illumination (array)
Base Station Interfaces						
Internal Interfaces	Spacecraft Bus Structure		Mechanical interface	Mechanical interface	Mechanical interface	Mechanical interface
		Payload	Env data (dust, radiation, thermal)			
	Commands (Systems to deploy)		Command & Data Handling (CDH)	Telemetry + science data (for DTE)	Commands (Heating and cooling)	Commands (Power distribution to systems)
			Uplinked commands	Telecomms		
					Guidance Navigation & Control (GNC)	
	Heating (gimbal, docking latch)	Power (sensors)	Thermal control (Heating) Data (Temperatures)	Heating/cooling (transponder, SSPA, UHF)	Thermal	Heating (battery)
	Power (gimbal, docking latch)		Power (OBCs, comms)	Power (transponder, SSPA, UHF)	Power (heaters/radiators)	Power
Outputs to External	Docking Latch (mech./data to drone)		Telemetry/science, commands (UHF to drone)		waste heat (radiated to space)	charging current (to drone via docking)

Figure 2.2.3: Base Station and Drone N2 Chart

The Base Station's chart depicts Power as the most heavily connected subsystem, distributing to every subsystem on the platform. CDH and Thermal each depend on Power, marking these as one-directional dependencies rather than closed loops. The interfaces requiring the closest coordination are the docking latch (Structure and Power teams both own pieces of this interface, since it is simultaneously the mechanical mate to the Drone and its sole charging/data path) and the Telecoms-to-CDH command path, where uplinked commands must be validated and routed before any Drone-directed command can be relayed out over UHF. A breakdown in either interface stalls the excursion cycle at a different point: docking failure prevents recharge and data offload, while a Telecoms-CDH fault prevents new commands from ever reaching the Drone.

The Drone's chart depicts the Power dependency structure, but its more safety-critical coordination point is the GNC-to-CDH interface, where the IMU and MIMU report fused attitude and acceleration state to the OBC rather than to propulsion, meaning any latency or fault in that single link directly affects flight control. Thermal coordination is also tighter on the Drone than the Base Station, since cryocooling is reserved specifically for payload instrumentation rather than uniformly.

3. Science Mission Plan

3.1. Science Objectives Mission Objectives

Mission Objectives

The primary mission objective is to analyze the terrain and illumination of the selected research area. This location, the Shackleton-de Gerlache Ridge, is of scientific interest to NASA due to the abundance of Permanently Shadowed Regions (PSRs) and areas of continuous sunlight as well as rugged terrain. The Science Mission Directorate goal of how the study of the lunar environment might affect exploration objectives and the secondary Human Exploration Goal of expanding mobility and addressing hazards on the lunar surface shape the mission objectives. First, assessing the terrain and illumination of the lunar south pole in order to determine how well Terrain Relative Navigation (TRN) systems can perform. Second, the vehicle, payload, and critical systems can be significantly affected by hazards including micrometeorites and electrostatic regolith dust; this objective involves the measurement of such hazards through both remote sensing and via wear on components over time. Next, addressing the capability of a relatively novel vehicle system, the lunar drone, evaluating the ability of the vehicle to navigate while addressing the prior mentioned hazards of micrometeorites and adverse effects from regolith dust. Finally, advancing navigation capabilities with waypoints and landing zone mapping encompasses the last goal. These four objectives were selected because each maps directly to a distinct, unresolved constraint on lunar south pole surface operations: no prior mission has characterized TRN performance across this specific illumination extreme, quantified dust and radiation hazard accumulation on flight hardware over repeated PSR transitions, or demonstrated autonomous route planning inside a PSR, making them the objectives most directly needed to reduce risk for future Artemis-era surface activity.

Scientific Significance

Each objective addresses a gap current remote-sensing data cannot close on its own. Terrain and illumination characterization matters because low-angle sunlight at the south pole produces extreme, spatially abrupt contrast between illuminated ridgelines and shadowed crater floors, a lighting geometry orbital imagery can map from above but cannot validate from the ground-level perspective a navigation system actually operates in. Regolith and radiation hazard characterization matters because electrostatic dust mobilization and cumulative radiation exposure are known from Apollo-era and LADEE observations to affect exposed hardware, but the specific accumulation rates and behavior

at this landing site's PSR boundaries have not been measured in situ. Platform flight-dynamics monitoring matters because a small aerial mobility platform's stability in lunar gravity, particularly during transitions between illuminated and shadowed terrain, has not been demonstrated at this scale in this environment. Autonomous hazard avoidance matters because communication delay to Earth makes real-time operator commanding impractical during PSR excursions, so any future crewed or robotic asset operating in this environment will depend on validated autonomous routing. Collected data will reveal how these four factors interact and constrain one another at a specific, previously uncharacterized site, directly informing the mitigation strategies and platform designs needed for sustained lunar south pole exploration.

Science Traceability

The STM introduces the metrics that define each science goal in terms of the expected data and the process of retrieving the data. Instrumentation with estimated performance metrics allow the STM to display the suitability for the vehicle to achieve the different science goals.

The physical parameters and observables section most explicitly ties the individual goals to what data the vehicle is collecting. First, studying the terrain, elevation, surface albedo, and surface texture can give the vehicle data to benefit the goal of determining the suitability for the Terrain Relative Navigation Systems to perform. Next, the various instrumentation collects data on environmental hazards. In the observable section, row two, the STM describes the process for meeting the goal: "Record electrostatic current accumulation (in nanoAmperes) and collect mass spectra data to determine impactor composition. Measure multi-angle albedo variations to detect micrometeorite impacts." This way, the hazards of electrostatic dust and micrometeorites are studied in the area of interest at the same time. Taking radiation measurements, as seen in the first section of the HSMD portion of the STM, complements the goal of minimizing the environmental effects on mobility systems, since radiation can affect the critical electronic and computational components enabling the vehicle to be controlled and collect data. Finally, flight diagnostic data and instruments monitoring the performance, attitude, and altitude of the vehicle appear in the final section of the observables and physical parameters, where the vehicle collects this data passively while supporting the other mission goals.

Instrument Support

Each of the ten payload instruments ties to at least one mission objective through the measurements it collects. TAGCAMS and the Stereoscopic Terrain Camera together support the terrain and illumination objective, collecting optical imagery and stereo-derived elevation, slope, and albedo data across both sunlit and PSR terrain to characterize TRN performance directly. The Illumination Floodlight Arrays enable this same objective by making PSR imaging possible at all, since neither camera can operate in total darkness without active lighting. The Dust Impact Mass Spectrometer and Radiation Spectrometer support the hazard-characterization objective, collecting electrostatic charge and impact-energy data alongside continuous TID and dose-rate measurements to quantify how micrometeorite impacts and radiation exposure affect vehicle and payload survivability over time. The Space Grade Gyroscope-IMU and Honeywell MIMU support the mobility-platform evaluation objective, collecting vibration, acceleration, and attitude data that characterize how the drone maneuvers and remains stable while navigating around the same dust and micrometeorite hazards as the second objective measures. Finally, the Flight Processing Computer Software, Power Monitor, and Time-of-Flight Depth Camera support the navigation and waypoint-mapping objective, collecting routing, energy-consumption, and real-time obstacle-detection data that together let the drone plan and execute excursions autonomously while avoiding hazards, directly answering the mission's science question of how the south-pole environment constrains reliable surface navigation.

3.2. Experimental Logic, Approach, and Method of Investigation

The science problem that the team is trying to investigate is the analysis of the south pole of the moon. Understanding the permanently shadowed areas in order to investigate the area for the creation of future infrastructure and architecture of Artemis missions. The mission is trying to observe and analyze the terrain and illumination to try to characterize the regoliths and the impacts that meteorites have. By analysing the site, the team was able to determine if the selected side is perfect for the continuation of missions. The selected site is Shackleton de Gerlache connecting Ridge due to the different characteristics that the site offers. The rims of the crater have enough to charge the drone to have a certain amount of energy to continue the science missions. The water and ice deposit could help with future human missions that will expand the Artemis explorations. The instrumentation that the team chose was based on mission readiness and heritage of past missions. By having instruments that can adapt to the coldness of the south pole and the radiation impact that it will take from the sun.

Drone and base station will be deployed and be set on the main location where the most flat area of the area of interest is. The base station will serve as the main hub of

communication between the moon and earth, hold larger instruments, and serve as a charging/upload station for the drone. The Drone will serve as a scout and be the primary source of analysis for the site and environment. The target will identify the terrain and illumination by the use of TAGCAMS Optical navigation and FOV Stereoscopic Terrain Camera. For the characterization of the creators and meteor impact the use of a dust impact mass spectrometer will be used to record any electrostatic property from the dust. The rest instruments added on the drone will determine precise location for tracking of different data regarding time of flight and potential hazards found to be avoided in future scout missions. All data will be transferred back to earth when the drone returns back to the base station for fuel and charge.

First the team will have an initial health check of the base station and drone, by checking all the different subsystem communications. If everything goes as planned a DTE link verification will be sent to the earth ground station within 24-48 hours of landing. Once communication has been established, the heaters will activate in order to allow instruments to adapt to the new environment, and be fully functional. When the health checks have been completed and no issues occur the calibration process will begin with the different instruments found in the drone. A simple test of the different instruments will be done to calibrate against the known references of the landing site. As soon as the instrument calibration is done the station base will charge the drone fully and check for connection between base station and drone. From here the drone will be ready for the first flight mission and science can begin.

The first measurement that the team will collect will be the first flight data and the reaction of the drone. By analyzing this data the different components will be able to adapt to the known information from the moon. The second will be the actual environmental analysis from the analysis of the soil, the change of slope, and the change in temperature/radiation. Most of the observations will be different scans of the site in order to compare with what it has already been studied in the past. No samples will be collected due to the drone not being capable of carrying anything due to the conservation of power. Most data will be pictures taken and readings that the instruments capture in order to be shared with the base station then sent to earth.

The data will be secured by the drone for most of the excursion as it moves between areas and collects the different technical information. The drone will have its own system explained in the CDH subsystem where it will have limited connection with the base station. Some of the information will be processed through the different instruments that the drone has access to. When the drone returns to charge all the information that was captured will start to be uploaded and processed through the based station. While the system is compressing the information DTE link will be connecting to the earth station and data will be sent when the connection is strong. The base station will maintain all the information as a second storage but the team does share most of the information to the earth stations.

The drone is expected to take more than one excursion analysing the same information on different areas around the craters and permanent shadow regions. Measuring the changes of illuminated areas towards the shadow regions, the different electromagnetic charges that are found on the regolith, and the change of slope for mapping of the site. In addition, the drone will be taking measurements of travel time to see how long it takes to travel between points. While doing that, measuring the reaction time for hazard avoidance and adaptability of the drone. It is important the same measurements get repeated in order to obtain more accurate and precise data for correct assumptions. It will allow the team to figure out the change in radiation, temperature and even regolith analysis. Gives a better understanding for the pre establishment of architecture for future Artemis missions.

The final observation will be the last test for the movement and mapping of the site with the different analyses that were made through the mission. Most of the science objectives should be accomplished as well as any engineering objectives. The establishment of architecture and mapping for future moon missions, and explorations around the craters. The mission will ensure the highest measure data by having the same measurements be taken in different areas and limiting the areas of explorations. By measuring the data of the drone while calibrating and adaptability it can smooth the mission. The mission will fully end once the storage is full or there is no longer any fuel on the drone for more scout missions. The drone will stay charging in the base station as the station sends the final recording to earth.

The reason for the sequence of steps is to assure a successful mission and understanding of what it needs to be accomplished. If the health checks and calibrations are not done ahead of time the mission could completely fail. If the instruments are not able to obtain the correct data then none of the objectives can be actually accomplished. The cameras will be used to take a picture of the different areas in order to identify the different changes in slope, regolith and meteorite impacts. The dust meter will allow analysis on the dust and help prevent the issue with dust on the instruments. The gyro, the Radiation and the flight computer will analyse location tracking of the drone and the changes it faces during temperature changes.

The main interest of the team is to examine the south pole region due to the interest of the client on the permanent shadow regions. The site has a great potential for the future expansion of future Artemis missions that can help establish different bases. The main focus is the exploration shows the transition between the areas of illumination and permanent shadow regions. The slope does affect the amount of light the craters will have due to the angle the moon has, allowing for ice to be formed and water to collect. The figure below is also mentioned in section 1,3 that shows the landing area and where the plan will take place due to the low slope degree. The Base station will need a small area where it can laid as flat as it possibly can,



The white areas show terrain with greater than 50% mean illumination and less than 5° mean slope within Shackleton-de-Gerlache Ridge. Site B is the chosen location. The surrounding red circle shows the 200m buffer.

Figure 3.2.1 Mission Location

3.3. Payload Success Criteria

The Payload subsystem's overall success is defined by its ability to return the terrain, environmental, and flight-performance dataset the mission's four science objectives require, while sustaining the autonomous operation needed to safely complete each PSR excursion. At minimum, mission success requires the subsystem to archive and downlink the mission's required 1 TB of science and engineering data (SYS-008) across at least one completed excursion cycle, with each of the four instrument suites, optical/terrain-imaging, dust/radiation-sensing, inertial-sensing, and navigation/hazard-detection, contributing usable data toward its corresponding STM objective. Because the subsystem's single-string instruments (the Stereoscopic Terrain Camera, Dust Impact Mass Spectrometer, Radiation Spectrometer, Power Monitor, and ToF Depth Camera) are backed by cross-instrument coverage rather than physical duplication, payload success does not require every instrument to survive the full mission; it requires the subsystem as a whole to keep collecting and safely returning data even as individual components degrade. Successful payload operation directly enables the mission's broader purpose: establishing a topographical, thermal, and environmental hazard baseline at the lunar south pole that future Artemis-era crewed surface missions can build upon.

TAGCAMS Optical Navigation Suite. TAGCAMS' primary objective is generating the 2D optical imagery used for visual odometry and, jointly with the Stereoscopic Terrain Camera, the elevation, slope, and albedo data required by STM Objective 1. Success requires sustained imaging at 0.03 m/pixel spatial resolution across a $75^\circ \times 75^\circ$ field of view throughout active excursions, satisfying PAY-SMD-1.1.4. Expected deliverables include a continuous visual-navigation dataset and near-IR albedo profiles across both illuminated and PSR terrain. This data directly supports the mission's terrain/illumination TRN characterization objective, and its scientific significance lies in demonstrating whether south-pole illumination extremes degrade optical navigation performance in a way that constrains future surface mobility platforms.

Stereoscopic Terrain Camera (SCALPSS-derived). The Stereoscopic Terrain Camera's primary objective is 3D photogrammetric mapping, generating stereo-derived point clouds and digital elevation models to characterize slope, crater density, and surface texture. Success requires functional stereo imaging jointly with the Illumination Floodlight Arrays across PSR terrain, meeting PAY-SMD-1.1's terrain-imaging capability. Expected deliverables are digital elevation models across illuminated and PSR terrain, which, combined with TAGCAMS imagery, form the primary dataset for STM Objective 1. The scientific significance is direct: elevation and slope data from inside a PSR is data no prior mission has collected in situ, making this instrument's output a first-of-kind contribution to lunar polar topography.

Illumination Floodlight Arrays (VIPER-derived). As an enabling instrument, the floodlight arrays' primary objective is making PSR imaging possible at all, since neither camera in the optical suite can return usable data in total darkness without them. Success requires at least one of the two independently switched units illuminating a path 5 m ahead of the drone (PAY-006). The deliverable is not a science product itself but the operational condition that makes TAGCAMS' and the Stereoscopic Terrain Camera's science return possible inside PSRs. Its significance is therefore indirect but foundational: without functioning illumination, STM Objective 1's PSR-specific data collection cannot occur at all.

Dust Impact Mass Spectrometer (SUDA-derived). This instrument's primary objective is characterizing electrostatic regolith charging and micrometeorite impact behavior at the lunar south pole. Success requires detecting electrostatic charge signals across its 0.1 pA–100 nA sensitivity range and impactor kinetic energy across 10 eV–10 MeV, satisfying PAY-SMD-1.2.1 and PAY-SMD-1.2.2. Expected deliverables include electrostatic current accumulation logs and mass spectra characterizing dust mobilization and micrometeorite flux. This data directly supports STM Objective 2, and its scientific significance lies in

providing the mission's only direct, in-situ characterization of dust hazards, information current remote-sensing methods cannot substitute for.

Radiation Spectrometer/Dosimeter (LETS). The Radiation Spectrometer's primary objective is continuously measuring Total Ionizing Dose, particle energy spectrum, and dose rate to evaluate long-term electronics survivability. Success requires tolerating a cumulative TID of at least 100 krad(Si) while measuring dose rate across a 0.1 $\mu\text{Gy/hr}$ –10 Gy/hr sensitivity range, satisfying PAY-SMD-1.3.1 through 1.3.3. Expected deliverables are a continuous radiation dose-rate log spanning the full mission lifetime, directly supporting STM Objective 2's radiation-hazard evaluation. Its scientific significance extends beyond this mission: cumulative dose data from repeated PSR transitions directly informs radiation-shielding requirements for future crewed hardware operating in the same environment.

Space Grade Gyroscope-IMU (LN-200S). The LN-200S's primary objective is providing three-axis angular rate and linear acceleration data to monitor vibration, structural stability, and flight dynamics. Success requires measuring acceleration across a $\pm 16g$ range (PAY-HSMD-2.1.2) throughout each excursion. The expected deliverable is a continuous flight-dynamics dataset characterizing platform stability as the drone transitions between illuminated terrain and PSR darkness, supporting STM Objective 3. Its scientific significance lies in quantifying how lunar gravity and terrain transitions affect a small aerial platform's maneuverability, a direct input to future mobility platform design.

Space Graded IMU Suite (Honeywell MIMU). The MIMU's primary objective is providing an independent, radiation-hardened attitude and acceleration reference to cross-check against the LN-200S. Success requires the same $\pm 16g$ dynamic range and continuous cross-validation with the LN-200S throughout the mission's multi-lunar-day excursion cycle. The expected deliverable is a redundant, high-confidence flight-dynamics dataset with the precision to detect sensor divergence. Supporting STM Objective 3 alongside the LN-200S, its scientific significance is primarily one of data confidence: cross-validated flight-dynamics data carries substantially more weight for future platform design than a single-sensor dataset would.

Flight Processing Computer Software (cFS). cFS's primary objective is autonomously executing and updating the drone's flight path based on live hazard input, directly enabling STM Objective 4. Success requires demonstrated end-to-end autonomous operation across a full simulated excursion without real-time operator commanding (PAY-004). As flight software, its deliverable is not a science dataset but the demonstrated capability itself: a

fully autonomous hazard-avoidance and routing behavior functioning in a PSR-representative environment. Its significance is establishing whether autonomous flight control is viable for lunar surface mobility platforms operating beyond real-time Earth communication range.

Power Monitor (INA901-SP + ADC128S102QML-SP). The Power Monitor's primary objective is tracking real-time energy expenditure to enable energy-optimal path selection. Success requires monitoring power consumption to within 0.50 W·hr accuracy. Its deliverable is continuous energy-consumption telemetry correlated with route data, supporting STM Objective 4's automated routing evaluation. Though its footprint is minimal, its scientific significance is that it makes the mission's energy-optimal routing objective measurable and verifiable rather than assumed.

Time-of-Flight (ToF) Depth Camera (RESOURCE Camera). The ToF Depth Camera's primary objective is real-time obstacle detection for autonomous hazard avoidance. Success requires detecting obstacles at least 10 cm in size across a 1.00–15.0 m range (PAY-HSMD-2.3.1). Its deliverable is a continuous near-field hazard-geometry dataset used both operationally and, post-mission, to characterize obstacle density and collision-avoidance performance across the exploration zone, directly supporting STM Objective 4. Its scientific significance is establishing quantitative hazard-avoidance performance data at the lunar south pole, a direct input to sizing future autonomous mobility platforms' safety margins.

Instrument Failure Modes

TAGCAMS. The most likely failure mode is loss of a single camera head or DVR channel. Because TAGCAMS uses a dual-channel, dual-head architecture, this failure degrades the instrument to single-channel optical navigation rather than eliminating its function, so STM Objective 1 data collection continues at reduced resolution. Complete TAGCAMS loss would remove the mission's primary optical navigation source, though the Stereoscopic Terrain Camera provides partial coarse-rate coverage. No further mitigation beyond this internal redundancy is planned, since a fully independent third optical system was not justified against the drone's mass and power budget.

Stereoscopic Terrain Camera. As a single-string instrument, its most significant failure mode is a hard fault in the camera pair or MSE-lite electronics, which would eliminate the drone's primary 3D photogrammetric capability. This risk is accepted and mitigated through cross-instrument coverage: TAGCAMS continues providing 2D navigation imagery,

and the ToF Depth Camera continues providing near-field hazard geometry, so hazard avoidance is not lost even though STM Objective 1's stereo-depth data collection would be. The mission's overall science return would be reduced but not ended, since TAGCAMS alone can partially support terrain characterization at lower fidelity.

Illumination Floodlight Arrays. The dual-unit, independently switched design means a single array failure reduces illuminated coverage rather than eliminating it outright, so TAGCAMS and the Stereoscopic Terrain Camera retain partial PSR imaging capability. Loss of both units would be more severe, effectively ending PSR-specific optical data collection for the remainder of the mission, since neither camera can function in total darkness without active lighting. No instrument-level backup exists for total illumination loss; the mission would fall back to illuminated-terrain-only data collection for its remaining excursions.

Dust Impact Mass Spectrometer. A fault is detected through a failed telemetry acknowledgment or an out-of-range reading, triggering an automated power-cycle and restart attempt. If recovery fails after a set number of attempts, the instrument is flagged for troubleshooting at the next DTE window and the excursion continues with that data stream marked lost, rather than aborting the excursion. Loss of this instrument would end STM Objective 2's dust-characterization contribution specifically, though the Radiation Spectrometer continues providing partial secondary context on the charged-particle environment driving dust behavior.

Radiation Spectrometer. This instrument follows the same fault-detection and automated power-cycle recovery process as the Dust Impact Mass Spectrometer. Its loss would remove the mission's only continuous TID and dose-rate monitoring capability, directly affecting STM Objective 2's radiation-hazard evaluation and removing the mission's ability to flag an excursion-threatening radiation zone in real time. The Dust Impact Mass Spectrometer provides only partial secondary context in this scenario, since it was not designed as a radiation-sensing backup.

LN-200S. Detection is cross-sensor: the flight computer compares LN-200S and MIMU output continuously, and a persistent divergence, rather than an internal fault flag, triggers recovery. If the LN-200S fails or diverges, the flight computer continues on the MIMU alone until re-convergence can be attempted at the next docking period. This degrades precision but does not eliminate flight-dynamics awareness, so STM Objective 3 data collection continues at reduced confidence rather than stopping.

Honeywell MIMU. The MIMU shares the LN-200S's cross-sensor detection and recovery process. Its loss leaves the LN-200S as the sole flight-dynamics sensor, which is functionally sufficient for safe flight but removes the dissimilar-redundancy cross-check that gives the mission's flight-dynamics dataset its highest confidence level. STM Objective 3 data collection continues, though the scientific dataset carries reduced validation weight for the remainder of the mission.

Flight Processing Computer Software (cFS). As flight software, cFS's failure modes are functional rather than physical: radiation-induced single-event upsets are the primary risk, addressed through the framework's built-in memory scrubbing and checksum applications rather than physical shielding. A sustained cFS fault would be the subsystem's most severe failure mode, since it is the component most directly responsible for autonomous operation; the mission's FDIR architecture is designed to detect and recover from this within the standard 5-minute fault window, defaulting to a safe hold-position or return-to-dock behavior if recovery is not achieved.

Power Monitor. Given its minimal footprint and telemetry-only role, its loss degrades energy-optimal routing precision rather than eliminating flight capability, since the flight computer can continue operating on a pre-planned path without live consumption feedback. STM Objective 4's energy-optimal routing evaluation would lose its supporting dataset, but hazard avoidance and general navigation continue unaffected, since the Power Monitor has no role in obstacle detection.

Time-of-Flight Depth Camera. As the subsystem's primary single-string fault-management case, a ToF failure is detected through dropped telemetry or out-of-range output, triggering the flight computer's onboard obstacle-detection software and the Stereoscopic Terrain Camera to assume interim hazard-sensing responsibility at a reduced update rate while the drone slows and increases stand-off distance from terrain features. If stereo imaging is degraded at the same time, the drone defaults to aborting the excursion and returning to dock along its most recently verified safe path. This is the subsystem's most safety-critical failure mode, since it directly affects the drone's ability to avoid collision during an active excursion.

Category	Minimum Success	Optimum Success	Stretch Goal
Mission - Science Return	1 TB of science and engineering data archived	1 TB archived and downlinked across all four	Data return exceeding 1 TB with mapping coverage extending

	and downlinked across at least one completed excursion cycle	planned excursion cycles through EOL (T+95 days)	beyond the nominal exploration zone
Payload - Optical/Terrain-Imaging Suite	At least one of TAGCAMS or the Stereoscopic Terrain Camera returns usable imagery meeting 0.03 m/pixel resolution during at least one excursion	Both cameras and the Illumination Floodlight Arrays remain operational across all four excursion cycles	Sub 0.03 m/pixel resolution mapping across the full exploration zone with multi-cycle time-series comparison of terrain change
Payload- Dust/radiation - sensing suite	At least one of the Dust Impact Mass Spectrometer or Radiation Spectrometer returns valid data meeting its PAY-SMD-1.2/1.3 thresholds	Both instruments remain operational across the full mission, returning a correlated dust-charging and radiation dataset	A dataset sufficient to support a predictive PSR environmental hazard model usable in future Artemis surface planning
Payload - Inertial Sensing Suite	At least one of the LN-200S or Honeywell MIMU provides flight-dynamics data sufficient for safe autonomous navigation	Both units remain cross-validated throughout the mission, fully satisfying PAY-HSMD-2.1	A flight-dynamics dataset detailed enough to refine lunar-gravity mobility-platform models beyond this mission
Payload - Navigation/Hazard Detection Suite	cFS, Power Monitor, and ToF Depth Camera together complete at least one full excursion with autonomous hazard avoidance and no vehicle loss	All four planned excursion cycles are completed autonomously with energy-optimal routing sustained throughout	A routing and hazard-avoidance dataset sufficient to establish a reusable autonomous navigation protocol for future Artemis-era mobility platforms

Figure 3.3.1: Success Criteria Success Summary Table

The complete payload, taken together, directly advances all four of the mission's science objectives: characterizing how terrain and illumination gradients affect surface navigation, quantifying regolith and radiation hazards, monitoring platform flight dynamics, and enabling autonomous hazard avoidance. If every payload success criterion above is met, the mission delivers the lunar south pole's first integrated dataset combining PSR-interior topography, in-situ dust and radiation characterization, and flight-dynamics performance from a single platform operating across repeated illumination cycles, data no prior mission has collected in this combination. This directly advances understanding of the Shackleton-de Gerlache Ridge and its adjacent PSRs as a target environment, resolving open questions about optical navigation reliability, dust mobilization thresholds, and

radiation exposure that current remote-sensing data cannot answer on their own. The scientific value of achieving even minimum success is substantial, since a single completed excursion cycle already returns first-of-kind PSR interior data; achieving optimum or stretch-level success across all four excursion cycles would establish a topographical, environmental, and platform-performance baseline detailed enough to directly inform Artemis-era crewed surface route planning and future robotic mobility platform design at the lunar south pole.

3.4. Testing and Calibration Measurements

Testing & Calibration Overview

Following successful landing and Base Station checkout, the payload's science instruments undergo a dedicated 24 to 48-hour calibration period (T+2 to 4 days) before the drone's first excursion, during which each instrument is powered on and verified against known reference conditions near the landing site. This in-situ calibration serves a fundamentally different purpose than the Verification & Validation performed prior to launch: pre-launch V&V (TVAC, radiation exposure, vibration/shaker table testing) confirms that hardware meets its design specifications under controlled, simulated conditions, while in-situ calibration confirms that each instrument survived launch and landing intact and is producing valid, correctly referenced data in the mission's actual operating environment, something no ground test can fully replicate given the lunar south pole's unique combination of illumination geometry, thermal cycling, and regolith composition. Only after this calibration period confirms nominal instrument health does the mission proceed to its first PSR excursion and begin nominal science operations.

Instrument Testing & Calibration

TAGCAMS Optical Navigation Suite. Initial health checks confirm all three camera heads and both DVR channels respond to command and return valid imagery. In-situ calibration images known, unobscured terrain features near the landing site under controlled illumination to verify focus, exposure, and geometric alignment against the mission's pre-characterized landing-site imagery as a reference. Correct operation is confirmed when returned imagery meets the instrument's 0.03 m/pixel spatial resolution at the reference target distance.

Stereoscopic Terrain Camera (SCALPSS-derived). Initial checks confirm both cameras in the stereo pair power on and stream imagery to the MSE-lite electronics. Calibration images a fixed reference target of known geometry near the landing site to verify stereo baseline alignment and disparity-processing accuracy, using the target's surveyed dimensions as the

control reference. Correct operation is confirmed when reconstructed point-cloud geometry matches the reference target's known dimensions within expected tolerance.

Illumination Floodlight Arrays (VIPER-derived). Initial checks confirm both LED units respond to command and reach rated output. Calibration verifies illumination intensity and coverage against the PAY-006 requirement by measuring illuminated distance ahead of the drone in a shadowed test area near the landing site. Correct operation is confirmed when both units, and each independently, illuminate a path of at least 5 m sufficient for TAGCAMS and Stereoscopic Terrain Camera imaging.

Dust Impact Mass Spectrometer (SUDA-derived). Initial checks confirm the target plate, charge-sensitive amplifiers, and velocity grids respond to a command self-test sequence. Calibration exposes the instrument to background particle flux near the landing site as a baseline reference, since a dedicated calibration source is not available in situ. Correct operation is confirmed when baseline charge and mass-spectra readings fall within the instrument's expected background noise floor.

Radiation Spectrometer/Dosimeter (LETS). Initial checks confirm the Timepix detector responds to command and reports a baseline count rate. Calibration compares the instrument's initial dose-rate reading against the expected ambient lunar surface radiation environment as a reference value. Correct operation is confirmed when reported TID and dose-rate readings are consistent with expected ambient background levels at the landing site.

Space Grade Gyroscope-IMU (LN-200S). Initial checks confirm the unit powers on and outputs incremental velocity and angle data over its serial interface. Calibration holds the drone stationary at the Base Station to verify zero-rate bias and gravity-vector alignment, using the known stationary state as the reference condition. Correct operation is confirmed when zero-rate output remains within the unit's rated bias stability while stationary.

Space Graded IMU Suite (Honeywell MIMU). Initial checks confirm the ring-laser gyroscope cavities and accelerometer channels respond to command. Calibration follows the same stationary zero-rate procedure as the LN-200S, with cross-validation between the two units serving as the primary control reference rather than an independent external source. Correct operation is confirmed when MIMU output agrees with the LN-200S within expected cross-check tolerance.

Flight Processing Computer Software (cFS). Initial checks confirm all cFS applications load and execute correctly on the OBC. Calibration consists of a simulated short-range navigation sequence near the Base Station to verify route-planning and obstacle-response

logic execute correctly, using the drone's known stationary surroundings as the reference scenario. Correct operation is confirmed when the software completes the simulated sequence without manual intervention.

Power Monitor (INA901-SP + ADC128S102QML-SP). Initial checks confirm the shunt monitor and ADC report valid readings over SPI. Calibration compares reported power draw against the Base Station's known charging current during the drone's pre-excursion charge cycle as a reference load. Correct operation is confirmed when readings remain within the instrument's 0.50 W·hr accuracy requirement against this known reference.

Time-of-Flight Depth Camera (RESOURCE Camera). Initial checks confirm the depth sensor, RGB camera, and integrated IMU all respond to command. Calibration images a staged obstacle of known size and distance near the landing site as the reference target. Correct operation is confirmed when the instrument detects the reference obstacle within its 1.00–15.0 m range at the correct reported distance.

Data Quality Assessment

Data quality is assessed primarily through comparison against known reference values established during each instrument's in-situ calibration, repeated observations across successive excursion cycles, and cross-instrument agreement where multiple instruments observe related phenomena, such as the Dust Impact Mass Spectrometer and Radiation Spectrometer's shared context on charged-particle behavior, or the LN-200S and MIMU's continuous cross-check of flight-dynamics data. Anomalous measurements are identified through the same CDH fault-management framework used across the mission: telemetry falling outside an instrument's expected calibrated range triggers an automated flag, prompting a power-cycle and restart attempt before the reading is either accepted, re-verified against the next excursion's data, or marked as a lost data point rather than treated as valid science return. Data consistency across excursion cycles, rather than any single measurement in isolation, is the team's primary indicator of whether an instrument continues operating within its validated performance envelope.

Instrument	Initial Functionality Test	Calibration Method	Success Criteria
TAGCAMS Optical Navigation Suite	Confirm all 3 camera heads and 2 DVR channels respond to command	Image known terrain near landing site under controlled illumination	0.03 m/pixel resolution achieved at reference target distance

Stereoscopic Terrain Camera	Confirm stereo pair streams imagery to MSE-lite	Image surveyed reference target of known geometry	Reconstructed geometry matches known target dimensions within tolerance
Illumination Floodlight Arrays	Confirm both LED units respond and reach rated output	Measure illuminated distance in shadowed test area	Each unit independently illuminates ≥ 5 m path
Dust Impact Mass Spectrometer	Confirm target plate and amplifiers pass command self-test	Measure background particle flux near landing site	Baseline readings within expected background noise floor
Radiation Spectrometer/Dosimeter	Confirm Timepix detector reports baseline count rate	Compare reading against expected ambient lunar background	TID/dose-rate consistent with expected ambient levels
Space Grade Gyroscope-IMU (LN-200S)	Confirm unit outputs incremental velocity/angle data	Stationary zero-rate and gravity-vector check	Zero-rate output within rated bias stability
Space Graded IMU Suite (MIMU)	Confirm RLG cavities and accelerometers respond to command	Stationary zero-rate check, cross-validated with LN-200S	Output agrees with LN-200S within cross-check tolerance
Flight Processing Computer Software (cFS)	Confirm all applications load and execute on OBC	Simulated short-range navigation near Base Station	Simulated sequence completes without manual intervention
Power Monitor	Confirm shunt monitor and ADC report valid SPI readings	Compare against known Base Station charging current	Readings within 0.50 W·hr accuracy of known reference
Time-of-Flight Depth Camera	Confirm depth sensor, RGB camera, and IMU respond to command	Image staged obstacle of known size and distance	Obstacle detected within 1.00–15.0 m range at correct distance

Figure 3.4.1: Payload Testing and Calibration Summary

3.5. Precision and Accuracy of Instrumentation

Precision & Accuracy Overview

Instrument precision and accuracy directly determine whether the mission's science return is usable for its intended purpose: establishing a topographical, environmental, and platform-performance baseline future Artemis surface missions can build on.

Measurement uncertainty that is too large or poorly characterized undermines the ability to distinguish real terrain, dust, radiation, and flight-dynamics effects from instrument noise, weakening the mission's core science conclusions even if data volume targets are met. Understanding each instrument's uncertainty is therefore as important as meeting its detection range or resolution requirement, since uncharacterized uncertainty can silently invalidate an otherwise complete dataset.

Instrument Performance

TAGCAMS Optical Navigation Suite collects 2D intensity imagery for visual odometry and albedo characterization. Its predicted spatial accuracy is 0.03 m/pixel, exceeding the ≤ 0.10 m/pixel requirement, based on the STM's predicted instrument performance derived from the ECAM-C50 heritage architecture's flight-proven optical characteristics. Uncertainty from low-light PSR imaging is mitigated through global-shutter windowing and binning, though residual noise at extended range may reduce effective resolution below the design value. The instrument meets its performance requirement with margin, supporting STM Objective 1.

Stereoscopic Terrain Camera (SCALPSS-derived) collects stereo-derived 3D point clouds and elevation data. Accuracy scales with mounting geometry per the SCALPSS photogrammetric accuracy model, with sub-millimeter to centimeter-level precision predicted at typical excursion altitudes based on published SCALPSS validation testing on lunar-analog terrain. Uncertainty increases with camera separation error and low-contrast, featureless regolith, partially mitigated by structured-light projection. The instrument's predicted performance supports STM Objective 1's terrain-mapping requirement.

Illumination Floodlight Arrays provide enabling illumination rather than a direct science measurement, so accuracy/precision does not apply in the traditional sense; performance is instead characterized by illuminated range, engineering-estimated at ≥ 5 m per PAY-006 based on VIPER heritage LED output. Uncertainty in beam falloff at the far edge of the

illuminated path could marginally affect imaging quality at maximum camera range, a secondary rather than primary uncertainty source for the mission.

Dust Impact Mass Spectrometer (SUDA-derived) collects electrostatic charge and impactor mass-spectra data. Predicted mass resolution is 1.2 amu against a ≤ 2 amu requirement, and charge sensitivity extends to 0.05 pA, both drawn from the STM's predicted instrument performance based on SUDA/LDEX flight heritage. Uncertainty arises primarily from recalibration to the mission's lower-velocity PSR impact regime relative to heritage flyby calibration, which may shift absolute composition accuracy until validated in situ. The instrument meets its performance requirements with margin, supporting STM Objective 2.

Radiation Spectrometer/Dosimeter (LETS) collects TID, particle energy, and dose-rate data. Predicted TID tolerance is 130 krad(Si) against a ≥ 100 krad(Si) requirement, with dose-rate sensitivity to 0.05 $\mu\text{Gy/hr}$, based on manufacturer-published LETS specifications and Peregrine Mission One flight heritage. Uncertainty stems from secondary particle flux variation between sunlit and PSR terrain, which the instrument's continuous sampling is designed to characterize rather than average out. The instrument exceeds its performance requirements, supporting STM Objective 2.

Space Grade Gyroscope-IMU (LN-200S) collects three-axis angular rate and linear acceleration data. Manufacturer datasheet specifications list a dynamic range of $1,000^\circ/\text{sec}$ against the $\pm 16g$ mission requirement, with bias stability supporting reliable dead-reckoning. Uncertainty from long-duration bias drift is bounded through cross-validation against the MIMU rather than relied upon in isolation. The instrument meets its performance requirement, supporting STM Objective 3.

Space Graded IMU Suite (Honeywell MIMU) collects an independent three-axis attitude and acceleration reference. Manufacturer specifications list bias stability under $0.02^\circ/\text{hr}$ per year and sub-1 ppm scale-factor drift, exceeding mission needs. Uncertainty is minimized through continuous cross-check against the LN-200S, with persistent divergence itself serving as the fault-detection signal rather than a tolerated error source. The instrument meets its performance requirement, supporting STM Objective 3.

Flight Processing Computer Software (cFS) does not collect a physical measurement; its "performance" is functional accuracy in routing and hazard response, verified through demonstration rather than a quantified precision figure. Uncertainty is primarily behavioral (timing latency in fault response) rather than measurement-based, bounded by the mission's 5-minute FDIR window. The software's demonstrated behavior determines whether STM Objective 4 is met, independent of traditional accuracy/precision metrics.

Power Monitor (INA901-SP + ADC128S102QML-SP) collects real-time current and cumulative energy data. Predicted accuracy is 0.10 W·hr against a ≤ 0.50 W·hr requirement, based on manufacturer-published shunt-monitor and ADC specifications. Uncertainty from thermal drift in the shunt resistor is minimized by physically isolating it from the sensitive analog electronics. The instrument exceeds its performance requirement, supporting STM Objective 4.

Time-of-Flight Depth Camera (RESOURCE Camera) collects near-field depth data for obstacle detection. Predicted detection extends to 5 cm features against a ≥ 10 cm requirement, with reaction time of 20 ms at 30 fps, based on Azure Kinect COTS baseline specifications and IM-2 analog testing. Uncertainty from pulse-timing noise in high-contrast PSR lighting transitions is a known limitation of ToF sensing generally. The instrument exceeds its performance requirement, supporting STM Objective 4.

Measurement Uncertainty

The subsystem's primary uncertainty sources are calibration drift between the pre-excursion reference check and end-of-excursion conditions, environmental effects from the PSR's extreme thermal swing and low-light imaging conditions, and instrument-specific limitations such as the Dust Impact Mass Spectrometer's uncalibrated low-velocity impact regime and the ToF camera's sensitivity to high-contrast lighting transitions. These uncertainties are minimized through in-situ calibration ahead of each excursion, cross-instrument validation (the IMU pair, and TAGCAMS/Stereoscopic Terrain Camera cross-coverage), and continuous telemetry monitoring that flags out-of-range readings for reprocessing or exclusion rather than allowing unflagged noise into the final dataset.

Instrument	Primary Measurement(s)	Accuracy	Precision	Source	Science Objective(s)
TAGCAMS	Optical imagery, albedo	0.03 m/pixel	380–1050 nm spectral	STM predicted performance	Objective 1
Stereoscopic Terrain Camera	3D point clouds, elevation	Geometry-dependent	Sub-cm at nominal altitude	Published SCALPSS validation	Objective 1

		(SCALPSS model)			
Illumination Floodlight Arrays	N/A (enabling)	≥ 5 m illuminated range	N/A	Engineering estimate, VIPER heritage	Objective 1
Dust Impact Mass Spectrometer	Charge accumulation, mass spectra	0.05 pA sensitivity	1.2 amu	STM predicted performance	Objective 2
Radiation Spectrometer	TID, dose rate	0.05 μ Gy/hr	130 krad(Si) tolerance	Manufacturer spec, flight heritage	Objective 2
LN-200S	Angular rate, acceleration	± 16 g range	1,000°/sec dynamic range	Manufacturer datasheet	Objective 3
Honeywell MIMU	Attitude, acceleration	$< 0.02^\circ$ /hr/yr bias	< 1 ppm scale factor	Manufacturer datasheet	Objective 3
cFS	N/A (autonomy)	N/A	N/A	Demonstration-based	Objective 4
Power Monitor	Power draw, energy	0.10 W·hr	N/A	Manufacturer datasheet	Objective 4
ToF Depth Camera	Near-field depth	5 cm feature detection	20 ms @ 30 fps	COTS baseline, IM-2 heritage	Objective 4

Figure 3.5.1: Payload Instrument Performance Summary

3.6. Expected Data & Analysis

The mission's science data return spans four core products: soil and terrain composition mapping, dust impact analysis and its effect on drone/instrument hardware, drone location and timing data supporting distance and time-of-flight characterization between points of interest, and instrument power/energy consumption data characterizing each instrument's operational envelope within the lunar south pole environment. Together, this data characterizes the hazards present at the lunar south pole, quantifies how dust impact and mobilization affect instrument and platform functionality, and establishes a mobility and hazard-avoidance baseline that future lunar surface platforms can build on.

Instrument Data & Analysis

TAGCAMS Optical Navigation Suite. TAGCAMS is expected to produce grayscale intensity imagery similar in format to the navigation-camera imagery returned by OSIRIS-REx during its approach to asteroid Bennu, where the same underlying ECAM-C50 architecture supported optical navigation and natural feature tracking[41]. Whereas OSIRIS-REx's imagery characterized an asteroid's surface from a standoff distance, this mission's imagery will be captured at close range during low-altitude PSR excursions, producing higher relative feature density per frame but over a smaller instantaneous footprint. Imagery will be analyzed through feature-tracking and photogrammetric correlation software to derive terrain slope, albedo, and crater density. The expected conclusion is a characterization of how illumination angle affects feature-tracking reliability across the illuminated-to-shadowed transition, directly improving understanding of how south-pole lighting geometry constrains optical navigation at the target site.

Stereoscopic Terrain Camera (SCALPSS-derived). The camera is expected to produce stereo-derived digital elevation models similar in form to the pre- and post-landing 3D surface morphology products generated by SCALPSS aboard Intuitive Machines' Nova-C lander[45]. SCALPSS's heritage data characterized a static landing site's morphology; this mission's data will instead capture DEMs across a moving excursion path, producing a continuous corridor of elevation data rather than a single fixed-site model. Point clouds will be processed through the same Semi-Global Matching disparity approach to derive slope and crater density maps. The expected conclusion is a quantified relationship between terrain roughness and safe traverse corridors near PSR boundaries, improving understanding of which terrain features at the target site are navigable by a small aerial platform.

Illumination Floodlight Arrays. As an enabling instrument, the floodlight arrays produce no independent science data product; their contribution is measured indirectly through the

imagery quality of TAGCAMS and the Stereoscopic Terrain Camera under active illumination, comparable to how VIPER's NavLight/HazLight system was validated by its effect on downstream NavCam and HazCam image usability rather than as a standalone dataset[52]. Analysis is therefore limited to confirming illuminated-image usability meets the same quality thresholds as sunlit imagery, improving understanding of what illumination intensity is required to make PSR interiors scientifically accessible at all.

Dust Impact Mass Spectrometer (SUDA-derived). The instrument is expected to produce mass spectra and impact-charge time series similar in structure to the compositional surface-mapping data generated by SUDA aboard Europa Clipper and its predecessor LDEX aboard LADEE[54,56]. Those missions characterized dust environments around an icy moon and the lunar exosphere at orbital altitude; this mission's data will instead characterize surface-adjacent dust mobilization from a low-altitude excursion platform, a measurement regime neither heritage mission directly sampled. Spectra will be analyzed for elemental composition signatures and correlated against excursion location to map dust charging behavior spatially. The expected conclusion is a spatial map of electrostatic dust mobilization risk across the exploration zone, improving understanding of how dust hazards vary with proximity to PSR boundaries.

Radiation Spectrometer/Dosimeter (LETS). The instrument is expected to produce continuous TID and dose-rate time series comparable to the radiation environment characterization LETS returned aboard Astrobotic's Peregrine Mission One[58]. That mission characterized radiation exposure during transit and brief surface contact; this mission's data will instead characterize sustained, cyclical exposure across repeated transitions between sunlit terrain and PSR interiors over the full mission lifetime. Dose-rate data will be analyzed against the excursion timeline to identify whether PSR interiors present elevated secondary-particle flux relative to sunlit terrain. The expected conclusion is a quantified radiation exposure profile specific to PSR transit, improving understanding of electronics survivability requirements for future hardware operating in the same environment.

Space Grade Gyroscope-IMU (LN-200S) and Space Graded IMU Suite (Honeywell MIMU). These instruments are expected to produce three-axis angular rate and acceleration time series consistent with the flight-dynamics telemetry both units have returned across their respective heritage platforms, including Mars Exploration Rover EDL data for the LN-200S and Mars 2020/InSight entry-descent-landing telemetry for the MIMU[61,65]. Both heritage datasets characterized single-event entry and landing dynamics; this mission's data will instead characterize sustained, repeated low-altitude flight dynamics across many excursion cycles in lunar gravity. Cross-validated output from both units will be analyzed

for vibration spectra and stability trends correlated against terrain type. The expected conclusion is a characterization of how lunar gravity and terrain transitions affect small-platform flight stability, improving understanding of maneuverability constraints for future lunar aerial mobility platforms.

Flight Processing Computer Software (cFS). cFS produces no independent measurement dataset; its output is behavioral, an autonomous routing and hazard-response log comparable in structure to the fault-recovery and command-routing telemetry cFS has generated across its more than 40 flight missions, including LADEE's lunar surface deployment[68,69]. Unlike those missions' general-purpose telemetry, this dataset will specifically characterize closed-loop, real-time hazard-avoidance decisions inside a PSR excursion profile, a scenario not previously demonstrated. Analysis will compare planned versus executed flight paths to quantify autonomous routing accuracy. The expected conclusion is a demonstrated feasibility (or identified limitation) of fully autonomous hazard avoidance for lunar surface platforms operating beyond real-time Earth communication range.

Power Monitor (INA901-SP + ADC128S102QML-SP). The Power Monitor is expected to produce continuous current and cumulative energy-usage telemetry consistent in format with the power-rail monitoring data this discrete signal-chain architecture has returned on planetary rovers including Curiosity and Perseverance[72,73]. Those platforms characterized power consumption across long-duration, multi-year surface operations; this mission's data will instead characterize power consumption per discrete excursion cycle, informing near-term route planning rather than long-term system aging. Energy data will be analyzed against route length and terrain type to determine energy cost per unit distance across different terrain classes. The expected conclusion is an energy-cost model for PSR excursions, improving understanding of the energy budget required for future mobility platforms operating in the same environment.

Time-of-Flight Depth Camera (RESOURCE Camera). The camera is expected to produce near-field depth maps and RGB imagery similar to the sub-centimeter depth data validated for the RESOURCE payload during Svalbard field testing and parabolic-flight lunar-lighting simulations[76,78]. Those tests characterized instrument performance under controlled analog conditions; this mission's data will instead characterize performance under actual PSR darkness and lunar dust conditions, closing the gap between analog validation and flight performance. Depth data will be analyzed for obstacle detection consistency and false-positive/false-negative rates across terrain types. The expected conclusion is a validated obstacle-detection performance envelope specific to lunar PSR conditions,

improving understanding of what hazard-detection range and resolution are sufficient for safe autonomous lunar mobility.

Error & Uncertainty Analysis

Instrument accuracy and precision directly determine the confidence level attached to each science conclusion drawn from the mission's dataset: an instrument operating within its rated accuracy produces data that can be compared directly against mission requirements and heritage benchmarks, while data falling outside characterized uncertainty bounds must be flagged and either reprocessed or excluded rather than treated as valid science return. The instruments with the widest performance margins relative to their requirements, the Radiation Spectrometer (130 krad(Si) against a 100 krad(Si) requirement) and the Power Monitor (0.10 W·hr against a 0.50 W·hr requirement), carry the lowest interpretive risk, since their measurement uncertainty is small relative to the phenomena being characterized. Instruments operating closer to their requirement threshold, such as the ToF Depth Camera's reaction-time performance or the Stereoscopic Terrain Camera's geometry-dependent accuracy, carry correspondingly higher interpretive risk, and their data will be weighted accordingly during cross-instrument analysis rather than treated with equal confidence. As a general principle, the more precisely and accurately an instrument's data is captured, the more directly it supports downstream mission planning; consistent, well-characterized data return makes conclusions about terrain mapping, hazard zones, and future base-site selection more directly actionable, while uncertain or unflagged anomalous data introduces risk into any mobility or siting decision built on it.

Instrument	Expected Data Product(s)	Expected Scientific Outcome	Associated Science Objective(s)
TAGCAMS	Grayscale navigation imagery, albedo profiles	Illumination-angle effect on optical navigation reliability	Objective 1
Stereoscopic Terrain Camera	Stereo-derived DEMs, slope maps	Terrain roughness vs. safe traverse corridors	Objective 1

Illumination Floodlight Arrays	Illuminated-image usability data (indirect)	Minimum illumination needed for PSR data access	Objective 1
Dust Impact Mass Spectrometer	Mass spectra, impact-charge time series	Spatial map of dust mobilization risk	Objective 2
Radiation Spectrometer	TID/dose-rate time series	Radiation exposure profile across PSR transit	Objective 2
LN-200S / Honeywell MIMU	Angular rate, acceleration time series	Flight-stability constraints in lunar gravity	Objective 3
cFS	Autonomous routing/hazard-response log	Feasibility of autonomous PSR hazard avoidance	Objective 4
Power Monitor	Current/energy-usage telemetry	Energy-cost model for PSR excursions	Objective 4
ToF Depth Camera	Near-field depth maps, RGB imagery	Validated obstacle-detection envelope in PSR conditions	Objective 4

Figure 3.6.1: Expected Data and Analysis Summary

4. Mission Risk Management

4.1. Safety and Hazard Overview

Safety and Hazard Overview

The Lunar South Pole Surface Mobility Mission approaches risk and safety as a continuous, whole-program responsibility rather than a single design review checkpoint. Throughout mission development, the team identifies technical, programmatic, and personnel hazards early and tracks them through design maturation so that risk is treated as something to be actively reduced rather than discovered late and reacted to. Risk management matters at every stage of this mission as the dual-platform architecture depends on interlinked subsystems operating reliably in the lunar south pole's PSR boundary, a function that no prior mission has fully replicated. Thus, a failure in one subsystem can cascade into loss of science return, loss of the drone, or complete mission loss. The remainder of this section works together to support that goal: the Risk Management Approach describes how the team identifies, evaluates, and tracks risk across the mission lifecycle, the accompanying Risk Log and Risk Matrix document the mission's current technical and programmatic risks and their priority, the Failure Mode and Effects Analysis (FMEA) examines the mission's highest-consequence failure modes in greater depth, the Personnel Hazards and Mitigations section addresses safety during manufacturing, assembly, integration, and test, and this section's Planetary Protection discussion addresses the mission's contamination-control obligations as a robotic lunar mission.

Risk Management Approach

Risks are identified throughout the systems engineering process using design reviews, requirements analysis, trade studies, subsystem analysis, FMEA, verification planning, environmental assessments, and Mission Assurance evaluations, with each subsystem team tracing its own platform decisions and trade study trade-offs to plausible failure conditions. Every identified risk is documented in a standardized Condition-Departure-Asset-Consequence (CDAC) statement to keep risk language consistent and traceable across subsystems, then scored on a 5×5 Likelihood × Consequence scale, where Likelihood reflects how probable the risk is to occur during the mission and Consequence reflects its impact on performance, schedule, cost, or mission success. The product of these two scores places each risk into a Low, Moderate, High, or Critical band on the Risk Matrix, which the team uses to prioritize mitigation effort toward highest-rated first. Risks are monitored continuously: each risk is assigned a responsible subsystem owner who tracks its mitigation actions and trend (increasing, decreasing, or neutral) through the Mission Design Review phase, with status reviewed periodically during technical interchange meetings, subsystem reviews, and configuration management activities so that the risk posture reflects the current state of the design. Mitigation strategies are applied based on the most appropriate response to each risk, design margin, hardware or functional redundancy, autonomous fault detection and recovery, procedural

controls, or, for lower-priority items, formal acceptance, and are updated as component selections, test results, and design maturity evolve, keeping the mission's overall risk posture within an acceptable range as the project advances toward Critical Design Review.

Planetary Protection

As a robotic mission targeting the lunar surface, this mission falls under NASA's planetary protection guidelines for solar system bodies without indigenous life potential, per NPR 8020.12D, *Planetary Protection Provisions for Robotic Extraterrestrial Missions*. The Moon is not considered a body of significant biological interest under current planetary protection categorization, so the mission's primary obligation is forward contamination control, preventing terrestrial biological or chemical contamination from compromising the scientific integrity of in-situ measurements, rather than the more stringent sample-return or life-detection protocols applied to bodies like Mars or Europa. This concern is captured directly in the mission's risk log (PP-R01, Forward Contamination), which identifies that inadequate contamination control procedures during assembly and integration could compromise hardware cleanliness and, in turn, the scientific integrity of the environmental measurements. The mission addresses this through standard cleanroom assembly procedures, equipment, and contamination-control protocols during payload integration, and protective instrument covers that remain closed until active data acquisition begins.

4.1.1. Risk Analysis

The Lunar South Pole Surface Mobility Demonstration Mission uses NASA's Continuous Risk Management (CRM) process to identify, evaluate, mitigate, monitor, and communicate technical, programmatic, and environmental risks throughout mission development (NASA/SP-2011-3422, *NASA Risk Management Handbook*). Risk-Informed Decision Making (RIDM) supports engineering trade studies and subsystem selection by balancing mission performance, reliability, technical maturity, schedule, and acceptable risk (NASA/SP-2010-576, *NASA Risk-Informed Decision Making Handbook*). Mission risks are documented in the risk log using the Condition–Departure–Asset–Consequence (CDAC) statement format, prioritized using a 5×5 Likelihood × Consequence Risk Matrix, and continuously updated during design reviews, subsystem testing, and Mission Assurance activities.

Risk Identification

Mission risks are identified throughout the systems engineering process using design reviews, requirements analysis, trade studies, subsystem analyses, Failure Modes and

Effects Analysis (FMEA), verification planning, environmental assessments, and Mission Assurance evaluations. Each identified risk is documented using the CDAC risk statement format ("Given [condition], there is a possibility of [departure], which could affect [asset], resulting in [consequence]") to ensure consistency and traceability across subsystems. Risks are evaluated according to their likelihood and consequence using a 5×5 scale, where Likelihood ranges from 1 (very unlikely to occur during the mission) to 5 (near-certain to occur), and Consequence ranges from 1 (minimal impact on performance, schedule, or cost) to 5 (loss of mission or crew-relevant safety impact). The product of these two scores ($\text{Rating} = L \times C$) determines whether a risk falls into the Low, Medium, High, or Critical band on the Risk Matrix. Each risk is assigned to a responsible subsystem owner who develops and tracks mitigation actions throughout the Mission Design Review (MDR) phase, and risk status is reviewed periodically during technical interchange meetings, subsystem reviews, and configuration management activities to ensure mitigation strategies remain effective as the design matures.

Each subsystem team identifies risks by tracing its own platform decisions and trade study trade-offs to plausible failure conditions.

The CDH subsystem's risks stem primarily from the subsystem's operating environment (radiation, thermal extremes, dust) and single-point failure conditions in its command and data architecture (loss of local link, storage failure, OBC anomalies), alongside programmatic risk from long-lead-time components.

The EPS subsystem's risks stem from the selected solar array, EPS/PMAD, battery storage, voltage bus architecture, and docking/charging interface, with major drivers including limited solar generation during low-illumination periods, thermal stress on batteries and electronics, solar array deployment, dust degradation, EPS overload, battery capacity growth, and charging-interface failure between platforms, alongside programmatic risk from vendor-configured components such as the solar array, PMAD, and battery packs.

The Mechanical subsystem's risks stem from structural integrity, mass growth, manufacturing, integration, environmental exposure, and subsystem interfaces, including risk inherent to the drone's cold-gas propulsion system. The Thermal subsystem's risks stem from radiation degradation effects on component materials (particularly MLI), and from active and passive TCS elements failing to complete their intended heating, cooling, or power-conservation functions.

The Payload subsystem's risks stem from the selected science and navigation instruments and the external lunar south pole environment they operate in, including dust, thermal

extremes, and vibration during flight and landing, with major drivers including optical surface dust contamination, PSR thermal exposure, calibration drift from vibration and thermal cycling, mounting misalignment, and illumination interference affecting hazard-detection performance. Once identified, each risk is evaluated in terms of its likelihood during nominal mission operations and its impact on mission success, data return, science quality, or schedule, and assigned a mitigation category based on whether the risk is best addressed through design margin, redundancy, or operational procedures.

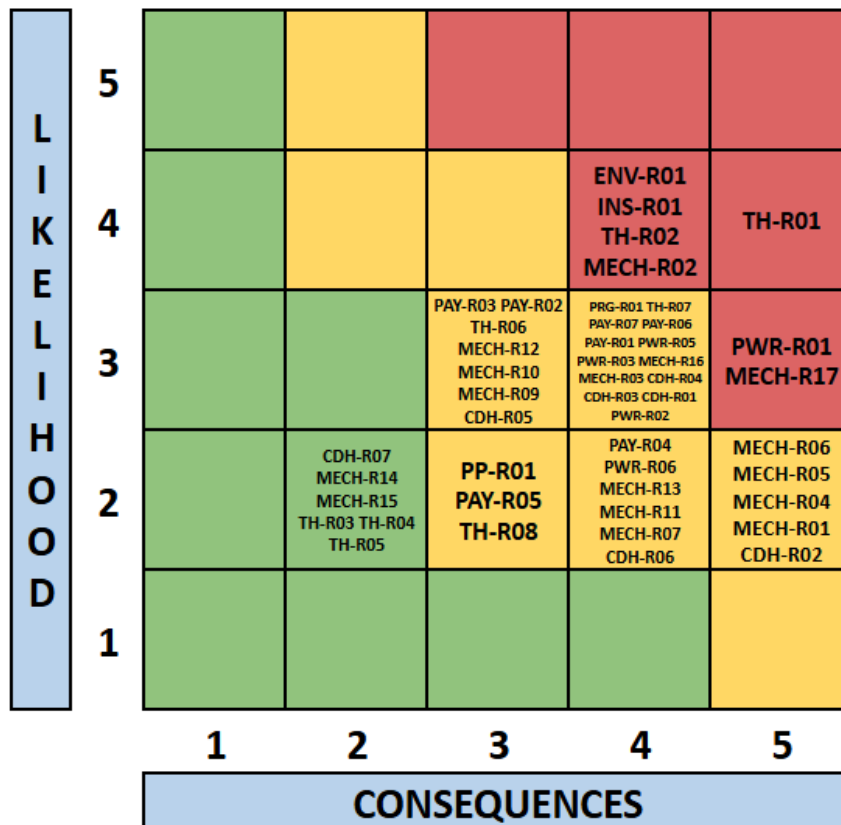


Figure 4.1.1.1: Subsystem Risk Matrix

Rank	Risk ID #	Risk Owner	Risk Title	Risk Statement	L	C	Rating	Approach	Trend
1	MECH-R02	Mechanical Team	Lunar Dust Intrusion	The lunar south pole contains highly abrasive and electrostatically charged regolith, which means there is a possibility that lunar dust could infiltrate bearings, seals, hinges, docking interfaces, or other moving mechanical components, ultimately causing failure.	4	4	High	Mitigate	→ Neut...
2	MECH-R17	Mechanical Team	Fuel plumbing failure	The fuel system on either the base station or the drone could experience a leak. The leak could range from minor to total decompression. This could severely cut mission life, or end the mission outright	3	5	High	Mitigate	↓ Decr...
3	PWR-R01	Power Team	Insufficient Energy During Darkness	Given that Site B may experience extended low-illumination periods, there is a chance that battery storage is insufficient to support survival loads, which could result in loss of thermal control, C&DH safe-mode operation, or mission interruption.	3	5	High	Mitigate	↓ Decr...
4	TH-R01	Thermal Lead	Thermal Survival Failure	Given that prolonged operation in Permanently Shadowed Regions (PSRs) may exceed thermal design capability, there is a possibility that battery and electronics operating limits are exceeded, which can result in mission degradation or loss.	4	5	High	Mitigate	→ Neut...
5	TH-R02	Thermal Team	Extreme Cold	For BS and DR systems, they are in PSR where temperatures can reach -230 degrees Celsius, impacting all instruments and subsystem components. Having heating TCS will help reduce that from happening	4	4	High	Mitigate	↓ Decr...
6	INS-R01	Payload Lead	Dust Contamination	Given that lunar dust may accumulate on payload optical instruments, there is a possibility of degraded imaging/sensing performance, which can result in science and navigation data quality falling below mission requirements.	4	4	High	Mitigate	→ Neut...
7	ENV-R01	Lead Systems Engineering	Lunar Dust Accumulation	Given that abrasive lunar dust may enter mechanical interfaces, there is a possibility of subsystem performance degrading over time.	4	4	High	Mitigate	→ Neut...
8	CDH-R01	CDH Team	Radiation Degradation	Given that the Base Station OBC (Xiphos Q8) is rated to only 30 krad TID while operating in a sustained lunar radiation environment, there is a possibility of cumulative Total Ionizing Dose buildup impacting the onboard processor and flash memory, which can result in memory corruption or permanent hardware failure of the primary Base Station computer.	3	4	Moderate	Mitigate	→ Neut...
9	CDH-R02	CDH Team	DTE Link Closure Failure	Given that Base Station pointing errors, RF interference, or insufficient link margin occur at the maximum Earth-Moon distance (~406,700 km), there is a possibility during DTE communication link closure failure impacting the X-Band communications subsystem, which can result in a missed scheduled downlink window and/or delayed telemetry data return.	2	5	Moderate	Watch	→ Neut...
10	CDH-R03	CDH Team	BS-Drone Local	Given that the line-of-sight can be obstructed by terrain or lunar dust during a	3	4	Moderate	Mitigate	↓ Decr...

			Link Loss	PSR excursion, there is a possibility of a local UHF link drop between the Base Station and Drone. As a result, this can impact the local communication network, resulting in autonomous mission hold, delayed command relay, or lost drone telemetry until FDIR recovery is completed.					
11	CDH-R04	CDH Team	Thermal Extremes PSR	Given that mission lifespan occurs inside a Permanently Shadowed Region (PSR) where temperatures may approach -230°C, there is a possibility of CDH electronics operating outside qualified thermal ranges adversely impacting onboard processing and bus timing, which can result in component failure or degraded SpaceWire bus synchronization.	3	4	Moderate	Mitigate	↓ Decr...
12	MECH-R01	Mechanical Team	Structural Failure	Since the base station and drone structural assemblies are currently in the preliminary design phase and have not yet gone through detailed structural sizing or verification, there is a possibility that the structure could fail. This could result in a loss of structural integrity, payload damage, or mission failure.	2	5	Moderate	Watch	↓ Decr...
13	MECH-R03	Mechanical Team	Mechanical Procurement Delay	Several of the mechanical assemblies require very specialized manufacturing, which leads to long lead times that could cause a delay in procurement. This delay in procurement could cause delays in subsystem integration and verification testing, impacting the overall project schedule and deadlines.	3	4	Moderate	Mitigate	↓ Decr...
14	MECH-R04	Mechanical Team	Launch vibration and Resonance	Given that the spacecraft will experience significant launch vibration and dynamic loading, it is possible that the structural members or mounted components could resonate near their natural frequencies.	2	5	Moderate	Watch	↓ Decr...
15	MECH-R05	Mechanical Team	Landing Impact Loads	Landing loads may differ from the preliminary design assumptions, and their impact forces may exceed the subsystem's design limits.	2	5	Moderate	Watch	↓ Decr...
16	MECH-R06	Mechanical Team	Environmental Test Failure	If mechanical components fail vibration, thermal cycling, dust exposure, or deployment testing, redesign or rework may be required before system integration.	2	5	Moderate	Watch	→ Neut...
17	MECH-R16	Mechanical Team	Propulsion engine failure	The engines for the cold gas thrust system may experience major or minor failure, greatly impacting the mission. Minor failure may mean a reduced operational life or small course readjustments. Major failure could include catastrophic mechanical failure, completely eliminating the maneuverability of the drone	3	4	Moderate	Mitigate	↓ Decr...
18	PWR-R02	Power Team	Reduced Solar Generation Margin	Given that the baseline DHV DSA-555W solar array provides lower generation margin than the 1268 W growth option, there is a chance that dust accumulation, shadowing, off-nominal sun angle, or deployment issues reduce available power below mission needs, which could result in insufficient power, delayed battery recharge, reduced drone sortie frequency, or forced power conservation.	3	4	Moderate	Mitigate	→ Neut...
19	PWR-R03	Power Team	EPS / PMAD Overload	Given that updated thermal, payload, and drone recharge estimates may exceed early EPS assumptions, there is a possibility of PMAD overload or load-shedding, which could interrupt power to critical subsystems.	3	4	Moderate	Mitigate	↓ Decr...

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20	PWR-R04	Power Team	Battery Thermal Degradation	Given that Li-ion batteries are sensitive to charge rate, temperature, and depth of discharge, there is a possibility of thermal degradation, which could reduce mission lifetime or cause safety hazards.	2	5	Moderate ▾	Watch ▾	→ Neut... ▾
21	PWR-R05	Power Team	Drone Charging Interface Failure	Given that the scout drone relies on physical docking contacts for recharge, there is a chance of misalignment, contamination, or wear and tear preventing recharge, which could reduce capabilities.	3	4	Moderate ▾	Mitigate ▾	↓ Decr... ▾
22	PAY-R01	Payload Team	Payload Thermal Survival in PSR	Given that payload electronics and optics may operate inside Permanently Shadowed Regions with temperatures approaching -230°C, there is a possibility of components falling outside qualified thermal ranges, which can result in reduced sensor sensitivity or instrument failure.	3	4	Moderate ▾	Mitigate ▾	→ Neut... ▾
23	PAY-R06	Payload Team	Payload Procurement/Integration Delay	Given that several payload instruments rely on heritage or university-developed hardware with longer and less predictable lead times, there is a possibility of delayed instrument delivery, which can result in schedule impacts to assembly, integration, and test.	3	4	Moderate ▾	Mitigate ▾	↓ Decr... ▾
24	PAY-R07	Payload Team	ToF Camera Illumination Interference	Given that active NIR illumination or high-contrast lighting near PSR boundaries may occur, there is a possibility of signal degradation, which can result in reduced hazard detection reliability during landing or flight.	3	4	Moderate ▾	Mitigate ▾	
25	CDH-R05	CDH Programmatics Member	Data Storage Overflow	Given that Drone payload instruments (sensors, cameras) generate data faster than the UHF local link can transfer to the Base Station during an excursion, there is a possibility of onboard data storage overflow impacting the data management system, which can result in the overwriting or loss of critical science/engineering data before Base Station transfer is confirmed.	3	3	Moderate ▾	Watch ▾	↓ Decr... ▾
26	CDH-R06	CDH Team	Component Procurement Delay	Given that flight-grade CDH hardware (e.g., Iris V2 Transponder, Drone OBC) carries estimated lead times of up to 25–32 weeks, there is a possibility of component procurement delays adversely impacting the manufacturing and integration schedule, which can consequently affect hardware testing.	2	4	Moderate ▾	Mitigate ▾	↓ Decr... ▾
27	MECH-R07	Mechanical Team	Docking Mechanism Misalignment	If the manufacturing tolerances are exceeded or there are assembly errors that cause misalignment of the drone docking mechanism, the drone may fail to dock securely or deploy reliably during operations.	2	4	Moderate ▾	Mitigate ▾	↓ Decr... ▾
28	MECH-R08	Mechanical Team	Fastener loosening	During the mission, the subsystems will experience vibration, landing loads, and repeated thermal cycling. This could possibly lead to a threaded fastener coming loose over time.	2	4	Moderate ▾	Mitigate ▾	↓ Decr... ▾
29	MECH-R09	Mechanical Team	Regolith collection failure	Since the regolith collection assembly operations are in an abrasive lunar environment, there is a chance that the collected regolith obstructs or interferes with the collection mechanism.	3	3	Moderate ▾	Watch ▾	→ Neut... ▾
30	MECH-R10	Mechanical Team	Mechanical Mass Growth	If the structural design matures beyond the preliminary estimates and additional brackets, supports, fasteners, or reinforcements are needed, the mechanical subsystem could exceed its allocated mass budget.	3	3	Moderate ▾	Mitigate ▾	↓ Decr... ▾
31	MECH-R11	Mechanical	Thermal	If there is thermal expansion and/or contraction that causes distortion of the	2	4	Moderate ▾	Watch ▾	↓ Decr... ▾

		Team	Distortion	structural members or mounting interfaces, the subsystem alignment may become affected. This will reduce the docking accuracy and/or instrument performance.					
32	MECH-R12	Mechanical Team	CAD and Interface Changes	The subsystem interface definitions are still being refined, which means there may be changes to mounting locations, payload interfaces, and/or subsystem packaging that could require a component redesign. This could ultimately result in additional design iterations and schedule delays.	3	3	Moderate ▾	Mitigate ▾	→ Neut... ▾
33	MECH-R13	Mechanical Team	Structural Verification Delay	The structural analysis and verification activities are in progress. There may be deficiencies that are identified during analysis or testing that could require there to be a redesign of structural components.	2	4	Moderate ▾	Mitigate ▾	↓ Decr... ▾
34	PWR-R06	Power Programmatic	Power Component Delay	Given that the solar array, EPS, and battery packs are vendor-configured items, there is a chance of procurement delay, which could affect assembly, integration, and test schedule.	2	4	Moderate ▾	Mitigate ▾	→ Neut... ▾
35	TH-R06	Thermal Team	Radiator Damage / Performance	Once fully extended out to take advantage of the radiation hitting the solid state panel. The buildup will reduce its performance, and we're already dealing with the BS location being at PSR	3	3	Moderate ▾	Watch ▾	→ Neut... ▾
36	TH-R07	Thermal Team	Regolith Build Up	Regolith collects on the TCS components that are causing harm to the operation to help the BS and DR station. The internal TCS can run into performance cooling and heating issues.	3	4	Moderate ▾	Watch ▾	→ Neut... ▾
37	TH-R08	Thermal Team	Sensor Failures	TCS sensors monitor all operating temperatures of the BS and DR. If there are no readings, TCS will have no way of knowing what temperatures the components need to maintain operational. Multiple sensors needed to reduce the probability of this occurring.	2	3	Moderate ▾	Mitigate ▾	↓ Decr... ▾
38	PRG-R01	Project Manager	Schedule Delay	Given that subsystem design maturation, verification, or testing may extend beyond planned milestones, there is a possibility of delayed integration and testing, separate from vendor procurement delays.	3	4	Moderate ▾	Watch ▾	↓ Decr... ▾
39	PAY-R02	Payload Team	Instrument Calibration Drift	Given that thermal cycling and vibration during flight and surface operations may occur, there is a possibility of camera and IMU calibration drifting from pre-flight values, which can result in reduced measurement and navigation accuracy.	3	3	Moderate ▾	Mitigate ▾	↓ Decr... ▾
40	PAY-R03	Payload Team	Payload Mass/Power Growth	Given that payload instrument design is still maturing, there is a possibility of the subsystem exceeding its allocated mass or power budget, which can result in reduced margin for other subsystems or mission-level mass/power constraints.	3	3	Moderate ▾	Mitigate ▾	→ Neut... ▾
41	PAY-R04	Payload Team	Instrument Vibration Misalignment	Given that launch loads, landing loads, and drone flight vibration may occur, there is a possibility of camera or IMU mounting misalignment, which can result in reduced pointing accuracy or degraded navigation performance.	2	4	Moderate ▾	Watch ▾	↓ Decr... ▾
42	PP-R01	Mission Assurance	Forward Contamination	Given that contamination control procedures may not be followed, there is a possibility of compromised mission hardware cleanliness and scientific integrity.	2	3	Moderate ▾	Watch ▾	→ Neut... ▾

43	PAY-R05	Payload Team	Mass Spectrometer Contamination	Given that particulate buildup on the Dust Impact Mass Spectrometer aperture may occur, there is a possibility of instrument response bias, which can result in inaccurate composition data.	2	3	Moderate ▾	Watch ▾	→ Neut... ▾
44	CDH-R07	CDH Team	Reduced Bandwidth Backup Data Interface	Given that the primary SpaceWire data bus fails or becomes unavailable, there is a possibility of the system falling back to the RS-422 or CAN-FD backup (each rated below 50 Mbps vs. 200 Mbps for SpaceWire) impacting the data handling architecture, resulting in severely bottlenecked science telemetry throughput.	2	2	Low ▾	Accept ▾	→ Neut... ▾
45	MECH-R14	Mechanical Team	Backup Supplier Availability	The backup suppliers have already been identified for each of the major mechanical assemblies. There is a chance that the primary supplier becomes unavailable as well as the backup, which could result in large delays.	2	2	Low ▾	Accept ▾	→ Neut... ▾
46	MECH-R15	Mechanical Team	Minor Surface Damage and Wear	The exposed surface may experience handling damage, minor impacts, or superficial abrasion during manufacturing, integration, and the mission. If it is during manufacturing and integration, it could lead to localized rework or inspection, causing delay. If it occurs during the mission in the lunar environment, it could cause degradation of protective coatings or expose interfaces if the damage becomes more severe over time.	2	2	Low ▾	Accept ▾	→ Neut... ▾
47	TH-R03	Thermal Team	Heating Component Failure	Heating components such as the Electric Heaters and Thermal Straps connected to the sources.	2	2	Low ▾	Accept ▾	→ Neut... ▾
48	TH-R04	Thermal Team	Cooling Component Failure	Cooling components such as Cryocooler, Thermoelectric Coolers, and Deployable Radiator.	2	2	Low ▾	Accept ▾	→ Neut... ▾
49	TH-R05	Thermal Team	MLI Damage	MLI film will be on the exterior of needed systems; for this mission, it will be located on the BS and DR exterior. Film can be damaged by the buildup of lunar regolith or long exposure to radiation.	2	2	Low ▾	Watch ▾	↓ Decr... ▾

Figure 4.1.1.2: Subsystem Risk Log

1.1.1. Failure Mode and Effect Analysis (FMEA)

The Failure Modes and Effects Analysis (FMEA) evaluate the mission's most critical technical risks by examining potential failure modes, their causes, their effects on individual subsystems, and their overall impact on mission success. The highest-priority risks were selected directly from the Basic Risk Log based on their Risk Priority Number (RPN), likelihood, consequence, and importance to achieving the mission objectives.

The FMEA should focus on the mission's highest-risk items and use the required template with the following columns:

The discussion following the table should explain how these failure modes are mitigated through redundancy, environmental testing, qualification, autonomous fault recovery, contamination control, and operational procedures, while identifying any remaining high-risk areas that will require continued mitigation as the project advances beyond the Mission Design Review.

Command & Data Handling (CDH)

The Xiphos Q8's 30 krad TID rating is the single largest radiation vulnerability in the CDH subsystem, since it directly manages DTE formatting and command routing for the entire Base Station. If radiation degradation disables the primary processor, the mission loses its capability to prepare and transmit science data to Earth unless ground intervention restores partial function or the Vorago Health monitor logs and boots a repair, if possible.

Radiation-induced failures within the Command and Data Handling (CDH) subsystem may occur due to single-event upsets, total ionizing dose effects, or memory corruption. These failures can interrupt autonomous navigation, onboard data processing, command execution, and communications. Temporary processor resets may be recoverable, while permanent electronic failures could prevent the rover from performing autonomous operations. Because nearly every subsystem depends on CDH functionality, this failure represents one of the highest mission risks.

The Drone payload instruments generate data faster than the UHF link offloads it, which during extended excursions, create a risk of overwriting unconfirmed science data before it reaches the Base Station's data storage. Thus, during long period excursions, this bottleneck may hinder excursion data offloads from Drone to Base Station and cause overwriting and loss of data.

Command processing anomalies or memory corruption within the cFS stem from radiation-induced effects during command validation. This directly impacts the software subsystem across the Base Station and Drone, potentially resulting in erratic system behaviors, command errors, or hazardous commands delivered to the system overall. The

impact ranges from temporary software halts to permanent component damage, threatening total mission degradation if FDIR isolation loops fail to contain the fault and safely contain/verify the problem.

A primary unit failure in the Drone's SoC is caused by radiation component degradation during PSR excursions or severe software corruption that prevents the cold-spare interface from activating. This failure negatively impacts the Drone's autonomous flight loops and payload data processing. If the Drone system fails to connect to the secondary spare unit, drone mobility is halted, resulting in an immediate halt of mobility and a total loss of PSR data collection.

A UHF contact loss during a PSR excursion cuts off the Drone from both command uplink and telemetry downlink, halting autonomous operations line-of-sight between Drone and Base Station is restored. While the omnidirectional antenna and FDIR recovery loop reduce likelihood and detection time, external and unaccounted for factors during an excursion could force a mission hold, directly affecting science data collection timelines.

A failed link closure at maximum Earth-Moon range delays rather than destroys data, since bulk storage retains science data for the next contact window, thus not providing a drastic, but scrutiny risk factor. The primary mission impact is schedule/latency in data return rather than permanent loss, provided onboard storage capacity is not simultaneously compromised by another external risk factor. In turn, DTE communication must be closely monitored for reliable data extraction.

Thermal Control

Thermal control failure may occur when survival heaters, thermal insulation, or temperature control electronics are unable to maintain acceptable operating temperatures during extended operations inside permanently shadowed regions. This failure may result from insufficient heater capacity, prolonged exposure to extreme cold, excessive heat loss, or electrical power limitations. Affected subsystems include the battery, avionics, scientific payloads, and communications hardware, all of which depend on stable operating temperatures. Loss of thermal control could significantly degrade rover operations and potentially result in complete mission failure if critical hardware becomes permanently inoperable.

Sustained PSR temperatures near -230°C threaten CDH electronics on both platforms, making this a cross-platform failure mode rather than a single system critical issue. Passive MLI and active thermal cycling drastically reduce likelihood, but any margin erosion (such as MLI degradation) would affect Base Station and Drone computing/bus timing at the same time.

Power Systems

Insufficient electrical power may result from reduced solar illumination, battery degradation, solar-array contamination, deployment anomalies, or unexpectedly high electrical loads. Reduced available power directly impacts thermal control, communications, rover mobility, command and data handling, and scientific operations. Extended power shortages may force the rover into survival mode or prevent completion of planned traverses and science activities. If adequate power cannot be restored, primary mission objectives may not be achieved.

Mechanical & Structure

The base station's structural frame supports all major mechanical, electrical, and science subsystems, making it one of the most critical structural elements of the mission. Failure due to excessive loading, manufacturing defects, or inadequate design margins could compromise multiple subsystems simultaneously and prevent mission completion. This risk is mitigated through material trade studies, preliminary structural analysis, and planned verification via detailed CAD development and finite element analysis.

The drone's structural frame supports the propulsion system, payload, and avionics throughout all mission phases. Excessive loading during launch, landing, or operation could result in structural failure, compromising vehicle stability or leading to complete mission loss. This risk is mitigated prior to fabrication through careful material selection, finite element analysis, and the application of conservative structural margins.

Failure of the landing gear could result in unstable landings, structural damage, or loss of the science payload following touchdown. Uneven lunar terrain and the cumulative effect of repeated landing loads increase the likelihood of this failure mode if the structure is not adequately designed. The landing gear design will be evaluated through structural analysis and repeated landing simulations to verify that it can safely withstand expected operational loads.

Wheel slip or sinkage may occur while traversing loose lunar regolith or steep terrain near the lunar south pole. Excessive wheel slip reduces mobility efficiency, increases power consumption, and may eventually immobilize the rover. The mechanical subsystem experiences the greatest impact; however, reduced mobility also limits navigation performance, science operations, and communications opportunities. A permanently immobilized rover would prevent completion of planned exploration routes and significantly reduce mission success.

A jam or misalignment in the docking mechanism would prevent the drone from deploying from, or returning to, the base station, substantially reducing the mission's operational capability. This failure mode may arise from regolith accumulation, tolerance stack-up, or mechanical misalignment over repeated docking cycles. The design addresses this risk

through alignment guides, adequate mechanical clearances, and a simplified docking mechanism. These measures will be validated through prototype testing and repeated deployment demonstrations.

Lunar dust intrusion represents one of the highest-risk failure modes for the mechanical subsystem. The abrasive nature of lunar regolith poses a direct threat to mechanical interfaces, seals, and sensitive electronics. The dust that enters these critical components can speed up wear, jam mechanisms, and degrade subsystem performance to the point of preventing successful mission operations. To mitigate this risk, the mechanical subsystem design incorporates sealed enclosures, protective barriers, and a reduction in the number of exposed moving components.

Lunar dust may accumulate on optical sensors, cameras, navigation sensors, and scientific instruments due to electrostatic adhesion and rover operations. Dust contamination reduces image quality, measurement accuracy, and science return while potentially increasing thermal absorption. Although this failure generally does not threaten rover survival, it directly reduces the mission's ability to characterize the lunar environment.

Mitigation Summary

The identified failure modes are mitigated through a combination of robust engineering design, subsystem redundancy, autonomous fault management, environmental qualification testing, and operational planning. Thermal risks are reduced by using multilayer insulation, survival heaters, thermal-vacuum qualification, and autonomous safe-mode operation. Power-system risks are minimized through battery management systems, power budgeting, load prioritization, and conservative energy margins. Mobility risks are mitigated through regolith mobility testing, terrain-relative navigation, slip monitoring, and recovery procedures. Radiation effects are reduced through radiation-tolerant electronics, watchdog timers, error-correcting memory, and autonomous system recovery. Dust contamination is minimized using protective covers, contamination-control procedures, cleanroom assembly, and periodic instrument calibration.

Although these mitigation strategies substantially reduce mission risk, several high-priority failure modes continue to require additional refinement during future design phases. Thermal survival within permanently shadowed regions, rover mobility across unconsolidated regolith, long-term battery degradation, and cumulative lunar dust contamination remain the most significant technical risks. These risks will continue to be monitored through NASA's Continuous Risk Management (CRM) process and further reduced through subsystem testing, environmental qualification, verification activities, and future design iterations prior to Critical Design Review (CDR).

Function	Failure Mode	Effects	Sev	Cause	Occ	Prevention	Det	RPN	Actions
Command & Data Handling (CDH)	BS OBC (Xiphos Q8) radiation fault	Loss of primary telemetry processing, command routing, and DTE preparation at Base Station; degraded data return	9	Cumulative TID buildup exceeding Xiphos Q8 30 krad rating	5	Radiation-hardened enclosure; continuous autonomous health monitoring via Vorago VA41630 OBC	4	180	Failover to Vorago VA41630 for safe-mode operation; review enclosure shielding.
	Drone OBCs (PolarFire SoC) primary unit failure	Loss of autonomous flight control loops and payload data handling; if cold spare fails, mission halts	8	Component defect exceeding radiation tolerance in PSR; software corruption	2	Active/cold spare PolarFire SoC configuration (>100 krad tolerance); flight control loop validation	3	48	Automatic failover to spare PolarFire; post-excursion diagnostic log review.
	Onboard data storage overflow	Overwriting or loss of critical science & engineering data before BS transfer	7	Drone payload generates data faster than UHF offload rate during excursions	3	MicroSD failsafe backup; priority assignment protecting critical health/fault logs	3	63	Confirm-before-delete protocol; revisit buffer sizing margin if excursion durations increase.
	Flight Software command/memory corruption	Erratic command execution or unintended actions on BS or Drone; hazardous execution	9	Radiation-induced bit flip in volatile memory during command validation	3	cFS command integrity and routing source verification; RTEMS FDIR process isolation	3	72	Immediate fault recovery execution per cFS validation; queued commands re-verified before execution.
Communications	Loss of BS-Drone local UHF link	Interruption of command relay and telemetry/science data return during excursion	7	Line-of-sight obstruction by terrain/dust; RF interference during drone movements	5	Omnidirectional UHF antenna configuration; autonomous FDIR contact-loss recovery loop	2	70	Autonomous hold position or return to last confirmed LOS if contact lost past FDIR window; RSSI logging.
	X-Band DTE link fails to close	Missed downlink opportunity; delayed telemetry & science data return to Earth	6	BS pointing error, RF interference, or insufficient link margin at Earth-Moon range	3	X-Band architecture selected for link closure margin at DSN 34m stations; RSSI monitoring	2	36	Bulk data storage retains science data for transmission during the next available contact window.
Thermal Control	Thermal survival failure / extremes in PSR	Electronics and batteries fall below operating limit (-230°C in PSR); local bus timing/sync failure	8	Prolonged exposure to extreme cold exceeding thermal protection capacity	4	Aluminized Kapton MLI passive isolation; active heater control loops; thermal testing	2	64	Continuous thermal telemetry monitoring; constrain excursion durations based on thermal operational limits.

Power Systems	Insufficient battery energy during darkness	Interruption of thermal/C&DH survival modes; potential total mission loss	9	Underestimated night loads or extended low-illumination duration in Site B	4	Survival power mode, duty cycling, conservative battery capacity margin	3	108	Confirm survival loads; size battery to worst-case low-power mode; update duty cycles.
	Solar array failure or deployment jam	Reduced recharge ability; limited science and drone operations	8	Dust accumulation, mechanical hinge obstruction, or deployment actuator failure	4	Deployment testing, dust protection, active solar tracking, generation margin	4	128	Verify deployment clearances; include power generation margin; define cleaning/parking modes.
	EPS overload or rail fault	Loss of power to critical loads; forced emergency load shedding	8	Unanticipated load growth, short circuit, or incorrect rail allocation	3	Ibeos Modular EPS margin, current-limited outputs, hardware fault protection	3	72	Finalize load table; allocate rails; test load switching and fault response.
	Li-ion battery thermal overheating/degradation	Permanent capacity loss, thermal runaway safety hazard, mission failure	10	High charge/discharge rates, ambient thermal extremes, overcharging	2	Dedicated BMS, active thermal control, strict charge limits and telemetry monitoring	3	60	Define battery thermal limits; verify charge profile; coordinate EPS-Thermal limits.
	Drone docking contacts fail to transfer power	Drone cannot recharge at Base Station; reduced operational capability	7	Regolith dust contamination or mechanical misalignment	4	Spring probes, self-aligning docking funnel, contact wiping/protection	3	84	Test docking cycles; add contact protection; verify charge continuity during qualification.
Structure & Mobility	Base Station structural frame or tipping failure	Structural loss, mounted hardware damage, loss of platform stability	9	Insufficient strength, unexpected loading, uneven lunar slope/terrain	3	Wide support footprint, low center of gravity, structural margins, FEA modeling	4	108	Perform detailed load analysis, static testing, and add deployable leveling/stabilizing feet.
	Fastener loosening	Component misalignment or separation due to launch/landing forces	7	High launch vibration, thermal cycling, or torque installation error	3	Locking fasteners, strict torque specs, secondary retention features	5	140	Inspect and retorque fasteners; utilize redundant attachment points in high-stress areas.
	Drone frame fracture or landing gear failure	Loss of propulsion, structural instability, inability to conduct excursions	10	Landing impact velocity, material fatigue, or severe ground obstacle impact	3	High-strength materials, landing-load margins, reinforced fixed landing gear	4	120	Reinforce high-stress frame regions; implement reduced-risk landing control mode.
	Propulsion mounting failure	Loss of thrust alignment, unstable flight, or catastrophic drone crash	10	Fastener failure, fatigue, or motor excitation vibration	3	Reinforced motor mounts, locking	5	150	Shut down the affected thruster unit autonomously and execute safe landing mode.

						hardware, modal vibration analysis			
Deployment Interfaces	Docking mechanism jam / misalignment	Drone cannot deploy, redock, recharge, or return sample	9	Regolith buildup, actuator failure, tolerance stack-up, or mechanical obstruction	4	Dust barriers, guide funnels, repeatable docking features, manual release paths	5	180	Attempt secondary release; retry docking with positioning adjustments; enter safe state.
Environmental & Dust Protection	Lunar dust intrusion / seal degradation	Mechanical binding, thermal seal failure, overheating, component jamming	9	Regolith exposure, seal wear, damaged enclosure interfaces	5	Hermetically sealed enclosures, non-contact magnetic seals, dust barriers	6	270	Isolate affected hardware; switch to backup internal components; restrict exposed motion.
	Surface abrasion / thermal distortion	Degradation of coatings, optical contamination, mechanism binding	8	Abrasive regolith contact, differential thermal expansion across materials	4	Compatible material CTE matching, wear-resistant coatings, expansion allowances	5	160	Adjust operational temperature limits; isolate affected mechanisms; inspect wear limits.
Payload & Regolith System	Regolith collection blockage or interface failure	Reduced science return; inability to transfer sample to payload instruments	10	Regolith compaction, particle clogging, mechanism misalignment	4	Passive funnel design, large mechanical clearances, minimal moving parts	5	200	Retry collection using vibration/alternate orientation; proceed with backup science modes.
	Payload dust contamination	Reduced science data quality, degraded optical/sensor performance	8	Electrostatic dust adhesion on lenses, apertures, or mass spec inlets	4	Protective deployable covers, electrostatic covers, contamination control	3	96	Execute periodic sensor calibration; actuate protective covers when not acquiring data.

Figure 4.1.1.1: System FMEA

1.1.2. Personnel Hazards and Mitigations

Personnel safety is an integral component of the Mission Assurance program throughout the manufacturing, assembly, integration, and testing of the Lunar South Pole Surface Mobility Demonstration Mission. The project follows established NASA system safety principles by identifying workplace hazards, implementing engineering and administrative controls, providing appropriate Personal Protective Equipment (PPE), and continuously monitoring safety performance throughout mission development. Hazards are identified and managed through a combination of scheduled safety meetings, Job Hazard Analyses (JHAs) completed prior to any new hazardous task, and routine facility walkthroughs conducted alongside laboratory safety assessments, manufacturing process reviews, integration planning, and environmental test planning. Hazard controls identified through this process are incorporated into standard operating procedures, personnel training, configuration management, and Test Readiness Reviews (TRRs) to ensure personnel safety while protecting flight hardware during all development activities. The project's safety program is structured to comply with applicable Occupational Safety and Health Administration (OSHA) workplace standards, ISO cleanroom and quality standards, and NASA system safety and Mission Assurance requirements throughout every phase of development.

Safety and risk management remain fundamental throughout every phase of mission development: Manufacturing → Assembly → Integration → Testing → Verification → Flight Hardware Acceptance. By continuously identifying workplace hazards, implementing appropriate mitigation strategies, and fostering a strong safety culture through regular review and team accountability, the project protects personnel, preserves critical flight hardware, and supports NASA's commitment to Mission Assurance and Continuous Risk Management (CRM).

Personnel Hazards & Mitigations

Manufacturing activities involve machining structural components, fabricating brackets, modifying mechanical assemblies, and preparing hardware for integration. These activities expose personnel to rotating machinery, moving equipment, flying debris, excessive noise, pinch points, and sharp metal edges. For example, a bracket being milled on the CNC machine could eject metal shavings at high velocity if the machine guard is bypassed. Mitigation measures include machine-specific operator training, machine guards and emergency stop switches, safety glasses with side shields, hearing protection when required, protective footwear, lockout/tagout procedures during maintenance, and routine equipment inspections.

Assembly and integration of rover electronics, batteries, communications hardware, and

avionics introduce electrical hazards including electric shock, short circuits, arc flashes, and electrostatic discharge (ESD). A realistic scenario includes a technician contacting an energized OBC test board during live-circuit troubleshooting, or an ungrounded technician handling a flight NVMe drive and inducing an ESD event that damages the component before the failure is even detected. Mitigation measures include qualified electrical personnel, insulated tools, electrical isolation during servicing, ground-fault protection, ESD workstations, antistatic wrist straps, and electrical safety inspections.

Lithium-ion batteries used by the rover and scout drone present potential fire, chemical, and thermal hazards if damaged, improperly charged, or short-circuited. A cell subjected to over-discharge or physical damage during handling could enter thermal runaway, producing toxic off-gassing and fire risk within an enclosed lab space. Mitigation measures include a Battery Management System (BMS), thermal monitoring, fire suppression equipment, approved charging procedures, dedicated charging locations, manufacturer handling guidelines, battery inspections before and after testing, and emergency response procedures.

Mission development requires the use of solvents, adhesives, lubricants, cleaning agents, conformal coatings, and other chemicals that may present health hazards, such as respiratory irritation from uncured conformal coating fumes in a poorly ventilated area. Mitigation measures include Safety Data Sheets (SDS), chemical hygiene procedures, proper labeling, ventilation systems, spill response kits, protective gloves, eye protection, and approved storage cabinets.

Personnel may encounter hot soldering equipment, thermal-vacuum chambers, environmental test chambers, cryogenic cooling systems, and heated electronic components. A technician removing hardware from a thermal-vacuum chamber immediately after a hot-soak cycle without verifying surface temperature could sustain a contact burn. Mitigation measures include heat-resistant gloves, thermal warning labels, controlled access during testing, equipment cooldown periods, temperature monitoring, and thermal emergency procedures.

Large rover assemblies, support fixtures, batteries, and environmental test equipment present lifting hazards that may result in musculoskeletal injuries or dropped hardware, such as a two-person lift of the Base Station structural frame resulting in a dropped-hardware incident if load distribution is not planned in advance. Mitigation measures include mechanical lifting devices, team lifting procedures, ergonomic lifting techniques, load limits, personnel training, and lift planning for oversized hardware.

Compressed gas cylinders, pneumatic systems, and environmental testing equipment may present explosion or projectile hazards, including during propulsion-related ground testing of the drone's cold-gas system, where an over-pressurized line or fitting failure could result in a rapid gas release. Mitigation measures include pressure vessel inspections, pressure relief devices, cylinder restraints, qualified operators, leak testing, and safe storage

procedures.

Cutting tools, drill bits, sheet metal, machined parts, and unfinished hardware present laceration hazards, particularly when handling freshly machined aluminum brackets with unfinished edges. Mitigation measures include cut-resistant gloves, deburring components, proper tool storage, safe handling procedures, and routine inspections.

Sensitive flight hardware is assembled within controlled laboratories and cleanroom environments where contamination and electrostatic discharge can affect both personnel safety and hardware reliability. Potential hazards include electrostatic discharge, Foreign Object Debris (FOD), cleanroom contamination, chemical exposure, and restricted movement. Cleanrooms also introduce a confined-space consideration during payload integration, where limited egress and restricted movement around sensitive hardware increase the consequence of an incident such as a fall or equipment collision. Mitigation measures include ISO cleanroom procedures, gowning protocols, shoe covers, hair coverings, ESD grounding, contamination monitoring, controlled access, and routine cleanroom inspections.

Engineering personnel spend significant time performing design work, documentation, software development, and systems engineering in office environments. Potential hazards include slips and trips, electrical cords, fire hazards, emergency evacuation, fatigue, and workspace clutter. Mitigation measures include housekeeping inspections, fire extinguishers, emergency evacuation plans, first-aid equipment, proper cable management, and emergency lighting.

Personnel will wear PPE appropriate for each task performed during mission development. Typical PPE includes safety glasses, face shields, laboratory coats, cut-resistant gloves, heat-resistant gloves, electrical gloves (when required), hearing protection, respirators (if required), steel-toe footwear, cleanroom garments, shoe covers, and antistatic wrist straps. PPE requirements are determined through task-specific hazard assessments, formalized in each activity's JHA, and reviewed before manufacturing, assembly, integration, and testing activities begin.

Personnel involved in prolonged computer work, hardware assembly, inspections, and repetitive laboratory activities may experience ergonomic hazards such as repetitive strain injuries, muscle fatigue, and back pain. Mitigation measures include adjustable workstations, ergonomic chairs, proper monitor placement, anti-fatigue mats, stretch breaks, job rotation, and lifting assistance, along with periodic ergonomic assessments.

Hazard	Activity/Location	Potential Consequence	Mitigation
Machining equipment	Machine shop	Cuts, crush injuries, eye injuries	Machine guarding, operator training, PPE
Electrical hazards	Electronics laboratory	Electric shock, arc flash	Lockout/tagout, insulated tools, qualified personnel
Battery handling	Battery laboratory	Fire, thermal runaway, chemical exposure	Battery management, fire suppression, thermal monitoring
Hazardous chemicals	Laboratory	Chemical burns, respiratory irritation	SDS compliance, ventilation, gloves, eye protection
Thermal hazards	Thermal-vacuum facility	Burns, cold injuries	Heat-resistant gloves, warning signs, controlled access
Lifting operations	Assembly area	Back injuries, dropped hardware	Mechanical lifting devices, team lifts, lifting training
Pressurized systems	Environmental testing	Explosion, flying debris	Pressure relief devices, inspections, operator certification
Sharp tools	Machine shop	Cuts and punctures	Cut-resistant gloves, proper tool storage
Cleanroom hazards	Cleanroom	Hardware contamination, ESD damage	Gowning procedures, contamination control, ESD protection
Inadequate lighting	Assembly area	Missed defects, eye strain	Adjustable task lighting, lighting inspections
Ergonomic hazards	Office and laboratory	Repetitive strain injuries, fatigue	Adjustable workstations, scheduled breaks
Extended work hours	Integration activities	Fatigue, reduced situational awareness	Workload management, staffing plans, scheduled rest periods
Electrical Shock	testing of OBC boards, NVMe drives, and transceivers	Electric shock, burns, or internal/physical damage to hardware from live circuit contact	De-energize circuits before handling; require electrical safety training before testing; follow and be knowledgeable of proper electrical regulations
RF/EM Exposure	Bench/integration testing of the Iris V2 Transponder and SSPA	RF radiation exposure exceeding safe limits at close range during active transmission	Test into a dummy load or shielded enclosure rather than open-air; maintain minimum safe distance during RF test; require RF safety training
Ionizing Radiation Exposure	TID/radiation-hardness testing of CDH boards	radiation exposure operating or present near the test source	Follow facility-provided radiation safety protocols and shielding; limit exposure time per the ALARA (As Low As Reasonably Achievable) principle
Solder/Rework Burns	PCB assembly, rework, and soldering during CDH board integration	Thermal burns from soldering irons/rework equipment	fume extraction at all soldering stations; require heat-resistant gloves and protection; soldering safety training; ensure proper ventilation
Cryogen Handling	Thermal-vacuum chamber	Frostbite/cold burns from Liquid Nitrogen contact; asphyxiation risk in enclosed test spaces	Use insulated cryogenic gloves and body shields; adequate lab ventilation and oxygen-level monitoring

Figure 1.1.2.1: Team 24 System Hazard Summary Table

Safety Management

Personnel safety procedures will be implemented throughout manufacturing, assembly, integration, and testing using documented safety plans, standard operating procedures, hazard analyses, and Test Readiness Reviews (TRRs). Before any hazardous task begins, a Job Hazard Analysis (JHA) will be completed and reviewed to identify task-specific risks and required controls, and all hazardous activities will be preceded by safety briefings, equipment inspections, and verification that appropriate engineering controls and PPE are in place. The team will hold scheduled weekly safety reviews to track open hazards, confirm mitigation status, and coordinate corrective actions, and will conduct routine facility walkthroughs to proactively identify emerging hazards before they result in an incident. These reviews will be coordinated with an Environmental Health & Safety (EH&S) Engineer to ensure continuous improvement of shop safety practices and alignment with applicable OSHA, ISO, and NASA safety standards. Configuration management and quality assurance processes will ensure that hardware modifications, testing activities, and safety documentation remain current throughout mission development.

Personnel participating in mission development will complete required training before working with mission hardware. Training will include machine shop safety, electrical safety, lockout/tagout procedures, chemical hygiene, battery handling, cleanroom protocols, electrostatic discharge (ESD) protection, emergency response, fire extinguisher use, and proper PPE. Facility requirements such as certified cleanrooms, controlled laboratory environments, calibrated test equipment, and emergency safety systems will be maintained throughout manufacturing and integration to ensure both personnel safety and hardware integrity.

2. Activity Plan

2.1. Project Management Approach

1. Mission Team Overview

The lunar south pole dual-platform mission is led by a four-person senior leadership team with the Project Manager (PM), Chief Scientist (CS), Lead Systems Engineer (LSE), and Deputy Project Manager of Resources (DPMR) who hold collective decision-making authority and remain consistent across all mission phases. Beneath the senior leadership, the mission is organized into three functional subteams (Science, Engineering, and Programmatic) each led by one of the senior leaders and staffed with specialists whose responsibilities evolve across the mission lifecycle as the primary activities shift from design and fabrication through integration, testing, launch, and operations.

This organizational structure supports the mission's dual-platform architecture by ensuring that the Base Station and Scout Drone are developed, integrated, and operated with clearly defined subsystem ownership, traceable requirements, and coordinated interface management. The Science subteam drives the mission's scientific objectives and instrument requirements, Engineering translates those requirements into hardware and software designs, and Programmatic maintains the mission's cost, schedule, risk, procurement, and configuration-control baselines.

2. Personnel and Functional Teams

The full mission team required to execute Phases C through F is larger than the current MCA team of 20. Based on the mission's personnel cost estimates and functional requirements across all project phases, the mission requires an estimated 22 personnel at peak staffing during Phases B and C, tapering to 10 during Phase F closeout. The table presents the staffing plan by functional role and mission phase.

Role / Functional Team	Phase B	Phase C	Phase D	Phase E	Phase F	Notes
Project Manager	1	1	1	1	1	Consistent across all phases
Chief Scientist	1	1	1	1	1	Consistent across all phases
Lead Systems Engineer	1	1	1	1	1	Consistent across all phases
DPMR (Programmatic)	1	1	1	1	1	Consistent across all phases
Scientists	4	4	2	2	1	Reduced at onset of Phase D
Engineers	8	8	4	2	1	Reduced at onset of Phase D
Technicians	2	2	4	2	2	Peak during Phase D I&T
Administrators	2	2	2	3	3	Increased Phase E–F for ops/closeout

TOTAL	20	20	15	14	10	
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The Science subteam consists of four to six personnel through Phase C, including planetary scientists responsible for STM development and landing site selection, bioastronautics specialists supporting environmental hazard characterization and human exploration objectives, payload instrument scientists responsible for instrument selection and performance validation, and instrument systems engineers managing science-engineering interfaces. Science staffing is reduced at the onset of Phase D as the team's primary deliverables shift from design and instrument selection to hardware integration oversight.

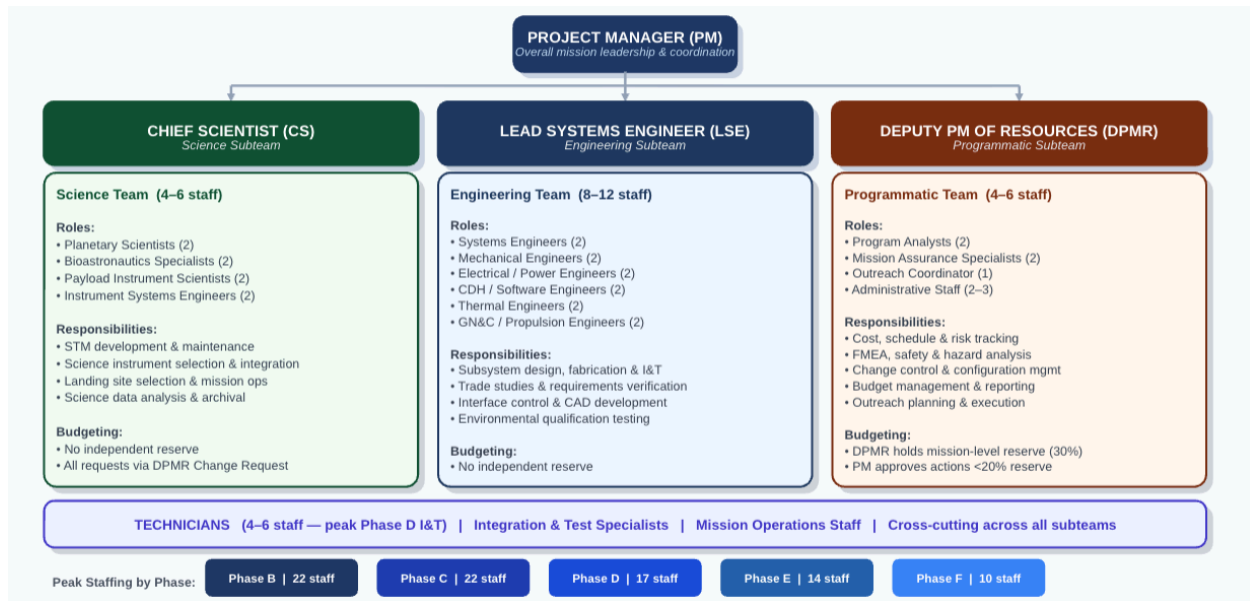
The Engineering subteam is the largest functional team, consisting of eight to twelve personnel through Phases C and D. The subteam includes systems engineers responsible for requirements management and interface control, mechanical engineers managing Base Station and Drone structural fabrication and qualification, electrical and power engineers overseeing EPS design and integration, CDH and software engineers responsible for onboard computing, FDIR logic, and ground software development, thermal engineers managing MLI design and heater integration, and GN&C and propulsion engineers responsible for drone flight dynamics and cold-gas propulsion qualification. Engineering staffing is reduced in Phase D as fabrication activities complete and the team transitions to technician-led integration and test operations.

The Programmatic subteam consists of four to six personnel across all phases, including program analysts responsible for cost and schedule estimation using the MCCET tool, mission assurance specialists managing the risk register, FMEA, and personnel hazard documentation, an outreach coordinator responsible for executing the community engagement activities, and administrative staff supporting logistics, reporting, and documentation management. Programmatic staffing remains relatively constant across phases, with administrative headcount increasing slightly during Phase E and F as reporting and closeout activities intensify.

Technicians form a cross-cutting workforce that spans all three subteams, peaking at six personnel during Phase D system integration and test. Technicians are responsible for hands-on hardware assembly, environmental test setup and operation, cleanroom activities, and mission operations during Phase E surface operations. Their staffing is reduced during Phase F as the remaining closeout activities are primarily administrative and analytical.

3. Personnel and Functional Teams

The chart represents the mission team organizational chart organized by a functional team. The chart reflects the full mission team structure required for Phase C and onward organized by functional roles rather than individual personnel.



3.1 Budgeting Authority and Reserves

The DPMR maintains subsystem budget allocations and mission-level reserve. Subteam leads manage expenditures within approved allocations, while requests requiring project-level reserve are reviewed by the DPMR and Project Manager through the change-control process in §2.4.1. Any change threatening the customer cost cap or another controlled programmatic baseline is escalated according to the mission's formal change-classification process.

3.2 Decision Making and Team Coordination

Technical decisions within a subteam are made by the relevant subteam lead without requiring full leadership team review. Decisions that affect multiple subteams, carry cost or schedule implications, or involve changes to a baselined system requirement (SYS-001 through SYS-022) are escalated to the full leadership team and resolved by majority consensus of PM, DPMR, CS, and LSE. Time-sensitive decisions where a full meeting is not feasible are made by the PM with 24-hour notification to all other leads.

Team coordination is maintained through weekly all-hands meetings, subteam-level working sessions, and the Change Control Board process. Meeting notes are maintained by

the PM and serve as the authoritative record of all decisions and action items. The DPMR maintains the Change Log, which tracks all baseline modifications across the mission lifecycle.

2.2. Mission Schedule

2.2.1. Schedule Basis of Estimate

Ground Rules & Constraints

The mission schedule is developed for Phases C through F, with Phase A and Phase B treated as sunk. The project must be completed within a \$250 million budget, inclusive of all team salaries, logistical expenses, and development costs through all post-PDR phases. The system must remain under 175 kg, fit within a 1.5 m × 1.5 m × 1.0 m stowed volume, and be delivered to Kennedy Space Center for launch integration by December 1, 2028 (SYS-017, SYS-018, SYS-019). The system must be launch-ready by February 1, 2029 (SYS-019). Planning costs factor in the use of contractors for the Launch Vehicle, Entry, Descent, and Landing (EDL) system, and Transit Vehicle, which are excluded from the Team 24 cost estimate. All Key Decision Point reviews (KDP-C, KDP-D, KDP-E, KDP-F) are assumed to occur at NASA Headquarters in Washington, D.C., requiring approximately five days of travel per review for key personnel. Schedule planning uses calendar-year increments aligned to NASA fiscal year boundaries where applicable.

Key Cost Assumptions

Inflation is assumed at a static rate of 2.4% from an average of predicted 2026–2028 inflation rates, and budgets incorporate a margin of 30% consistent with NASA standards for early-phase missions (Elkins-Tanton). Mission systems are qualified at a minimum TRL-6 (SYS-008) to buy down technical risk, as technologies at this level have demonstrated functionality in a relevant environment (NASA Technology Assessment and Insertion Process). Due to the confidence this provides, the engineering and flight unit are treated as the same system.

- Procurement of all long-lead hardware CDH components (25–31 weeks), EPS solar arrays and batteries (10–20 weeks), and payload instruments are initiated at the start of Phase C. Delays to procurement initiation directly compress Phase D integration margin.
- The Base Station and Scout Drone undergo separate integration and qualification test campaigns in Phase D before system-level integration, consistent with the dual-platform architecture.
- Environmental testing: thermal-vacuum (TVAC), vibration (VIBE), and electromagnetic compatibility (EMI/EMC) is performed at a NASA Center or qualified

test facility. Facility scheduling is treated as a schedule risk and buffered with two weeks of margin per test campaign.

- The Phase C/D boundary is set by KDP-D authorization following the Critical Design Review, targeted for July 2027, providing approximately 16 months for hardware procurement and design finalization from the KDP-C entry point.
- Travel estimates are developed using standard procedure for KDP meetings, with mission leadership traveling to Washington, D.C. for reviews, and the full team traveling to Cape Canaveral for launch.

Major Cost Drivers

Specialized labor costs, particularly for scientists and engineers, are expected to be the largest cost driver, as the complex dual-platform mission task will require a higher-than-average number of technical staff across Phases C and D. Travel expenses for Mission Lifecycle Reviews and launch will further influence total mission cost through housing, transportation, and per diem costs across multiple review cycles. Payload complexity is expected to present significant cost and schedule impacts, as specialized radiation-tolerant and cryogenically-rated instruments required for PSR operations introduce extended procurement and qualification timelines.

The three primary schedule drivers are CDH procurement lead times (up to 31 weeks for the Iris V2 X-Band transponder and PolarFire SoC OBCs), EPS procurement (DHV solar arrays at 12–20 weeks, EaglePicher batteries at 10–18 weeks), and the dual-platform Phase D integration campaign. The dual-platform architecture adds an estimated four to six weeks to Phase D compared to a single-vehicle baseline, as both platforms require independent TVAC and vibration qualification before system-level integration. The drone propulsion system remains the highest-risk open trade; if not closed before CDR, it introduces the greatest schedule uncertainty in Phase C. Schedule margin of two weeks is applied before each major review milestone consistent with NASA Schedule Management Handbook guidance.

Future Refinement

These estimates and assumptions are preliminary given the early phase maturity of the mission. Budget fluctuations will result from changes in staffing across project phases, component price changes from inflation and TRL progression, and subsystem design decisions not yet baselined. Schedule estimates will be updated at each subsequent gate review as hardware selections are finalized, test facility availability is confirmed, and KDP travel dates are established. The Mission Gantt Charts in §1.9.3 present the current phase-level schedule and will be revised through PDR as design matures.

2.2.2.1. Mission Phase C Overview

Activity durations were based on expected procurement, design, fabrication, and verification timelines, along with NASA schedule guidance and engineering judgment where direct estimates were not available. The order of activities was based on the dependencies between major tasks, such as completing detailed designs before fabrication and having the necessary hardware available before verification can be completed. One of the main schedule risks during this phase is the Scout Drone propulsion system, which remains an open trade and could affect the schedule if it is not resolved before CDR. Schedule margin is also maintained before major review milestones to allow time for potential delays and corrective actions. Phase C depends heavily on the timely delivery of long-lead components, completion of subsystem designs and verification, and closure of CDR actions before entering Phase D. Delays in these areas could reduce the time available for the separate Base Station and Scout Drone integration and qualification activities planned for the next phase.

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[illegible]

2.2.2.2. Mission Phase D Overview

Phase D begins after KDP-D and focuses on system assembly, integration, verification, and preparation for launch. During this phase, the Base Station and Scout Drone are integrated and qualified separately before being brought together for system-level testing. This sequence was chosen because both platforms contain different hardware and operational functions that need to be verified independently before their interfaces can be tested as a complete system. Environmental testing includes thermal-vacuum, vibration, and EMI/EMC testing at a NASA Center or other qualified facility, with additional time included in the schedule to account for facility availability and possible testing delays. Once both platforms have completed their individual qualification activities, system-level integration will verify interfaces between the Base Station and Scout Drone, including communications, power,

Phase D durations are based on the expected time needed for assembly, subsystem and system-level integration, environmental qualification, verification, and resolution of issues found during testing. The dual-platform architecture is one of the main factors affecting the length of this phase because the Base Station and Scout Drone require separate qualification campaigns before full system integration, adding an estimated four to six weeks compared with a single-vehicle approach. A two-week schedule margin is also included for each environmental test campaign because access to qualified test facilities is an external dependency that could affect the planned testing sequence. Phase D depends on the successful completion of Phase C hardware procurement and subsystem verification, since delays in long-lead hardware or unresolved design issues would reduce the time available for integration and testing. Testing activities are also dependent on successful completion of earlier assembly and integration work, meaning significant failures during environmental or system-level testing could require corrective work and retesting before the system can proceed toward launch. These dependencies make the completion of integration and qualification activities a major schedule driver leading into launch preparation and Phase E.

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2.2.2.3. Mission Phase E Overview

Phase E begins with launch and continues through deployment, checkout, calibration, and science operations on the lunar surface. After landing, the Base Station and Scout Drone will complete the initial system checkout before beginning science activities. The operational sequence places checkout and calibration before the first excursion so that subsystem health, instrument performance, communications, and the docking and charging interface can be verified before the Scout Drone enters the permanently shadowed region. Once checkout is complete, the Scout Drone will begin repeated autonomous excursions to collect terrain, environmental, and engineering data while the Base Station remains outside the PSR to support communications, data storage, environmental monitoring, and power-related functions. Following each excursion, the Scout Drone will return to the Base Station to recharge and transfer its remaining data, while the Base Station will downlink stored science and engineering data to Earth during scheduled communication windows. This cycle will continue throughout the operational period to gradually expand coverage of the exploration area.

The Phase E timeline is based on the order required to safely begin and maintain surface operations rather than on hardware procurement or fabrication estimates. Checkout must be completed before calibration, and calibration must be completed before the first science excursion, making these activities direct dependencies at the beginning of the phase. During routine operations, the pace of Scout Drone excursions will depend primarily on its available power and the amount of time it can safely operate in the cold conditions of the PSR. These limits affect how far the Scout Drone can travel while maintaining enough power and thermal margin to return to the Base Station, recharge, and prepare for the next excursion. Communications and data handling are also important schedule dependencies because science and engineering data must be transferred to the Base Station and downlinked during available Earth communication windows. The operational schedule therefore allows repeated excursions and data-return cycles to continue as

conditions permit while maintaining enough margin for charging, thermal recovery, data transfer, and response to potential anomalies.

Task	Owner	Jan'29	Feb'29	Mar'29
Launch Readiness Review (LRR)	All			
♦ LAUNCH - Cape Canaveral	All			
Lunar transit & EDL (contractor)	Contractor			
Surface deployment & checkout	CS/CDH			
Instrument calibration	CS			
PNT reference initialization	CDH			
First PSR excursion	CS/CDH			
Science data collection begins	CS			
DTE downlink - first data return	CDH			
♦ KDP-F - Phase F Authorization	All			

2.2.2.4. Mission Phase F Overview

Phase F begins after the completion of nominal science operations and focuses on mission closeout, final data return, and end-of-life activities. Before surface operations end, the Base Station will complete a final science downlink to confirm that the required science and engineering data have been returned and archived. The Scout Drone will then return to the Base Station and be placed in a safe configuration when possible; if it cannot return, it will instead be safed at its final location. Final telemetry from both platforms will be collected and archived so that the mission team has the information needed to complete post-mission analysis and documentation. Once these activities are complete, the mission will transition away from active lunar operations and into final data processing and project closeout.

The Phase F schedule is driven mainly by the completion of science operations and the amount of time needed to return, process, archive, and document the mission's remaining data. Final downlink and telemetry collection must occur before active operations end, making successful data return one of the main dependencies for closeout. Data processing, archiving, and final documentation can begin as usable data become available rather than waiting for every operational activity to end, allowing some closeout work to occur in parallel. The schedule also assumes that

sufficient communications capability remains available near the end of the mission to complete the final data return. Unexpected spacecraft conditions, communication issues, or additional data-processing needs could extend closeout activities, so the Phase F timeline should retain enough flexibility to address these issues before the mission is formally completed.

Task	Owner	Mar'29
Mission operations final downlink	CDH	
Final science data archival	CS	
Mission closeout report	PM/DPMR	
Lessons learned documentation	All leads	

2.3. Mission Budget

2.3.1. Budget Basis of Estimate

Ground Rules & Constraints

The system must be ready for integration with other systems by December 1st, 2028 and ready for launch from Cape Canaveral by February 1st, 2029. All components of the mission outside of the primary vehicle (the Launch Vehicle, Entry, Descent, and Landing (EDL) System, and Transit Vehicle) will be supplied by contractors, allowing the MCA team to focus on the Lunar Vehicle. Outside of these completion deadlines, the NASA Mission Lifecycle and the travel its milestones require present hard limitations on when the project must reach certain progress levels. The Mission Lifecycle includes four formal design reviews that occur after the completion of the Preliminary Design Review (PDR): the Critical Design Review (CDR), System Integration Review (SIR), and Operational Readiness Review (ORR). The system's total mass must not exceed 175kg and it must fit within a 1.5m x 1.5m x 1m volume to be properly stowed for deployment.

Key Assumptions

The team will implement fiscal year planning to best model the experience of working as an official NASA team, with staffing and development varying across the Science, Engineering, and Programmatic teams between fiscal years based on the main priority of the mission during that fiscal year (NASA Fiscal Year 2027 Budget Request). Acquisition timelines will consider the maximum provided lead time for a product intended to be integrated in the

Lunar Vehicle System as a means of implementing margin into the schedule budget. Technologies used for the mission will be a minimum of TRL-6 to buy down risk on the mission, as technologies at this level have been demonstrated as functional in a relevant testing environment as determined by NASA (NASA Technology Assessment and Insertion Process). Travel estimates will be developed implementing the standard procedure for KDP meetings, to which mission leadership and a few additional supporting team members will travel, and costs for travel to Cape Canaveral for launch will be computed using the maximum number of staffers allotted for in the budget to ensure the complete project team can witness the culmination of their efforts. Travel costs related to testing are assumed to be negligible, as the project is considered to be based out of the NASA Jet Propulsion Laboratory and will have access to its testing facilities.

Major Cost/Schedule Drivers

The major schedule drivers are GNC hardware maturity/drone mobility development, thermal/radiation/dust-tolerance testing, instrument integration, and communications interface verification. GNC hardware maturity/drone mobility development are expected to influence the cost of the mission because the cold gas system used for drone mobility and the GNC hardware to interface with it will likely have to be adapted from existing systems to meet mission needs, introducing additional R&D and testing expenses. Testing the payload for thermal, radiation, and dust resilience is expected. Payload complexity is also expected to have a major influence on mission cost and complexity due to the nature of needing to withstand the environmental conditions within permanently shadowed regions (PSRs) while collecting terrain data. Further payload complexity arises from the inclusion of multiple instruments into the system, which consists of a lunar radiometer, time-of-flight camera for depth measurements, and a stereoscopic camera. These instruments serve as a driver of mission cost because they are vital to the completion of science objectives, and thus each can be expected to receive heavy budgetary investment to ensure consistent functionality and high-resolution data acquisition. Landing requirements are also expected to have significant cost and schedule implications as positioning the payload to access PSRs will require careful consideration and specific positioning technologies that will introduce additional system costs. Instrument integration is expected to serve as a major cost driver for the system, as the aforementioned payload complexity will likely extend the integration timeframe compared to a normal mission, thus increasing costs. Ensuring consistent communications between the payload and data storage system is also likely to introduce significant schedule and cost concerns, as acquiring equipment capable of withstanding PSR conditions and performing the necessary testing to ensure these standards are met

will require extensive searching for testing facilities and acquisition of higher-end communications equipment, which will come attached to higher costs.

Future Refinement

It is understood that these estimates and assumptions are preliminary since the mission is in its early phases. In accordance with this understanding, estimates and assumptions will be refined as the mission develops. Expected changes to the schedule estimate include the expansion and shortening of times allotted for certain mission-critical tasks as situational factors, such as staffing and component acquisition, fluctuate over the course of the mission's development.

2.3.2. Total Mission Cost

Budget Overview

The lunar south pole dual-platform mission carries a preliminary total estimated mission cost that falls within the imposed budget cap of \$250 million (SYS-017, Figure XX), covering all development, integration, testing, operations, and closeout activities across Phases C through F. The preliminary cost estimate was developed using parametric Cost Estimating Relationships (CERs) sourced from The New Space Mission Analysis and Design (SMAD) and the NASA Instrument Cost Model (NICM), applied to the subsystem mass and power values baselined during the System Requirements Review. A cost margin of 50% has been applied to all direct cost categories; although this exceeds the 30% margin recommendation from the NASA Cost Estimating Handbook guidance for missions at pre-Phase C maturity, this number was chosen as a means of providing consideration to the additional expense of the dual-system payload architecture. This ensures adequate margin is available to absorb cost growth as the design matures through Critical Design Review (CDR). Costs associated with the Launch Vehicle, Entry Descent and Landing (EDL) System, and Transit Vehicle are excluded from this estimate as they are contractor-provided per customer requirements (SYS-006).

Budget

Additional Information					
	Phase C	Phase C	Phase C-D	Phase D-E	Phase E-F
# People on Team	FY 1 (26-27)	FY 2 (27-28)	FY 3 (28-29)	FY 3 (28-29)	FY 3 (28-29)
Science Personnel:	8	8	6	4	4
Engineering Personnel:	10	10	8	6	4
Technicians:	12	12	12	8	4
Administration Personnel:	5	5	5	3	3
Management Personnel:	4	4	4	4	4

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Mission Phase	Phase C	Phase C	Phase C-D	Phase D-E	Phase E-F	
Year	FY 1 (26-27)	FY 2 (27-28)	FY 3 (28-29)	FY 3 (28-29)	FY 3 (28-29)	Cumulative Total
PERSONNEL						
Science Personnel	\$ 640,000	\$ 656,640	\$ 504,960	\$ 344,960	\$ 353,280	\$ 2,499,840
Engineering Personnel	\$ 800,000	\$ 820,800	\$ 673,280	\$ 517,440	\$ 353,280	\$ 3,164,800
Technicians	\$ 720,000	\$ 738,720	\$ 757,440	\$ 517,440	\$ 264,960	\$ 2,998,560
Administration Personnel	\$ 300,000	\$ 307,800	\$ 315,600	\$ 194,040	\$ 198,720	\$ 1,316,160
Project Management	\$ 480,000	\$ 492,480	\$ 504,960	\$ 517,440	\$ 529,920	\$ 2,524,800
Total Salaries	\$ 2,940,000	\$ 3,016,440	\$ 2,756,240	\$ 2,091,320	\$ 1,700,160	\$ 12,504,160
Total ERE	\$ 820,554	\$ 841,888	\$ 769,267	\$ 583,687	\$ 474,515	\$ 3,489,911
Personnel Margin	\$ 1,128,166	\$ 1,157,499	\$ 1,057,652	\$ 802,502	\$ 652,402	\$ 4,798,221
TOTAL PERSONNEL	\$ 4,888,720	\$ 5,146,238	\$ 4,821,483	\$ 3,748,755	\$ 3,121,093	\$ 21,726,290

TRAVEL						
Total Flights Cost	\$ 2,250	\$ -	\$ 22,050	\$ 2,250	\$ 2,250	\$ 28,800
Total Hotel Cost	\$ 6,875	\$ -	\$ 39,245	\$ 6,875	\$ 6,875	\$ 59,870
Total Transportation Cost	\$ 945	\$ -	\$ 2,870	\$ 945	\$ 945	\$ 5,705
Total Per Diem Cost	\$ 2,070	\$ -	\$ 15,057	\$ 2,070	\$ 2,070	\$ 21,267
Travel Margin	\$ 3,642	\$ -	\$ 23,767	\$ 3,642	\$ 3,642	\$ 34,693
Total Travel Costs	\$ 15,782	\$ -	\$ 108,344	\$ 17,013	\$ 17,423	\$ 158,562
OUTREACH						
Total Outreach Materials	\$ -	\$ -	\$ -	\$ -	\$ 1,825	\$ 1,825
Total Outreach Venue Costs	\$ -	\$ -	\$ -	\$ -	\$ -	\$ -
Total Outreach Travel Costs	\$ -	\$ -	\$ -	\$ -	\$ -	\$ -
Total Outreach Services Costs	\$ -	\$ -	\$ -	\$ -	\$ -	\$ -
Total Outreach Personnel Costs	\$ -	\$ -	\$ -	\$ -	\$ 17,636	\$ 17,636
Outreach Margin	\$ -	\$ -	\$ -	\$ -	\$ 8,341	\$ 8,341
Total Outreach Costs	\$ -	\$ -	\$ -	\$ -	\$ 30,693	\$ 30,693

DIRECT COSTS						
Mechanical Subsystem	\$ 4,769,000	\$ 4,769,000	\$ 9,663,500	\$ 4,894,500	\$ -	\$ 24,096,000
Power Subsystem	\$ 3,458,000	\$ 3,458,000	\$ 7,007,000	\$ 3,549,000	\$ -	\$ 17,472,000
Thermal Control Subsystem	\$ 2,451,000	\$ 2,451,000	\$ 4,966,500	\$ 2,515,500	\$ -	\$ 12,384,000
Comms & Data Handling Subsystem	\$ 1,197,000	\$ 1,197,000	\$ 2,425,500	\$ 1,228,500	\$ -	\$ 6,048,000
Guidance, Nav, & Control Subsystem	\$ 2,185,000	\$ 2,185,000	\$ 4,427,500	\$ 2,242,500	\$ -	\$ 11,040,000
Science Instrumentation	\$ 5,206,000	\$ 5,206,000	\$ 10,549,000	\$ 5,343,000	\$ -	\$ 26,304,000
Spacecraft Cost Margin	\$ 9,633,000	\$ 9,633,000	\$ 19,519,500	\$ 9,886,500	\$ -	\$ 48,672,000
Total Spacecraft Direct Costs	\$ 28,899,000	\$ 29,650,374	\$ 61,603,542	\$ 31,972,941	\$ -	\$ 152,125,857
Manufacturing Facility Cost	Included in Spacecraft Cost	Included in Spacecraft Cost	Included in Spacecraft Cost	Included in Spacecraft Cost	\$ -	\$ -
Test Facility Cost	\$ 7,144,000	\$ 7,144,000	\$ 14,476,000	\$ 7,332,000	\$ -	\$ 36,096,000
Facility Cost Margin	\$ 3,572,000	\$ 3,572,000	\$ 7,238,000	\$ 3,666,000	\$ -	\$ 18,048,000
Total Facilities Costs	\$ 10,716,000	\$ 10,994,616	\$ 22,843,128	\$ 11,855,844	\$ -	\$ 56,409,588
Total Direct Costs	\$ 39,615,000	\$ 40,644,990	\$ 84,446,670	\$ 43,828,785	\$ -	\$ 208,535,445
Total MTDC	\$ 28,899,000	\$ 29,650,374	\$ 61,603,542	\$ 31,972,941	\$ -	\$ 152,125,857

FINAL COST CALCULATIONS						
Total F&A	\$ 1,926,600	\$ 2,001,737	\$ 4,208,404	\$ 2,208,644	\$ -	\$ 10,345,386
Total Projected Cost	\$ 46,446,102	\$ 47,792,966	\$ 93,584,901	\$ 49,803,197	\$ 3,169,210	\$ 240,796,376
Total Cost Margin	\$ 14,336,808	\$ 14,362,499	\$ 27,838,919	\$ 14,358,644	\$ 664,385	\$ 71,561,255
	30.9%	30.1%	29.7%	28.8%	21.0%	
Total Project Cost	\$ 46,446,102	\$ 47,792,966	\$ 93,584,901	\$ 49,803,197	\$ 3,169,210	\$ 240,796,376
***** Do not change the numbers in the cells below unless mission concept instructions specify otherwise.						
F&A %	10%	10%	10%	10%	10%	
ERE - Staff	28%	28%	28%	28%	28%	
Inflation Rate	0.0%	2.6%	5.2%	7.8%	10.4%	
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 5.3.2.1: Complete Mission Budget

Cost Summary

Budget Element	Value
Customer Budget Cap	\$250,000,000
Total Estimated Mission Cost	\$240,796,376
Available Cost	\$9,203,624
Cost Margin Applied	30% of all Personnel Costs 30% of all Travel Costs 50% of all Direct Costs 30% of all Outreach Costs
Mission Phases Covered	Phases C through F
Excluded Costs	Launch Vehicle, EDL System, Transit Vehicle

Figure 5.3.2.2: Budget Overview

The mission's dual-platform architecture comprising an independent Base Station and autonomous scout drone introduces hardware and qualification testing costs beyond those typical of a single-vehicle CLPS-class rover mission. These additional costs are captured within the direct cost and integration and test line items. The 30% cost margin for personnel, travel, and outreach costs and the 50% direct cost margin reflects the early design maturity of the mission at the PDR stage (Figure XX).

2.3.3. Personnel Budget

Personnel Cost Overview

Personnel designated to support the mission consist of scientists, engineers, technicians, administrators, and project management staff, consisting of the Project Manager (PM), Chief Scientist (CS), Lead Systems Engineer (LSE), and Deputy Project Manager of Resources (DPMR). Project management staff remain consistent across all project phases, but staffing varies across other departments due to mission needs (Figure XX). Scientists, engineers, and technicians staffing levels remain equivalent until the onset of the Phase D-E component of the project, where they are reduced in accordance with the transition away from conceptual engineering and scientific work towards system assembly, integration, and testing in preparation for launch (Figure XX). Staffing numbers are further reduced during phases E-F, where an equal number of engineers, scientists, and technicians are retained along with three administrators and the project management team (Figure XX). This change is due to the focus on operations during these phases, where scientists, engineers, and technicians have less involvement due to the lack of physical access to the spacecraft and its payload.

	Phase C	Phase C	Phase C-D	Phase D-E	Phase E-F
# People on Team	FY 1 (26-27)	FY 2 (27-28)	FY 3 (28-29)	FY 3 (28-29)	FY 3 (28-29)
Science Personnel:	8	8	6	4	4
Engineering Personnel:	10	10	8	6	4
Technicians:	12	12	12	8	4
Administration Personnel:	5	5	5	3	3
Management Personnel:	4	4	4	4	4

Figure 5.3.3.1: Personnel Counts

Budget Template

Mission Phase	Phase C	Phase C	Phase C-D	Phase D-E	Phase E-F	
Year	FY 1 (26-27)	FY 2 (27-28)	FY 3 (28-29)	FY 3 (28-29)	FY 3 (28-29)	Cumulative Total
PERSONNEL						
Science Personnel	\$ 640,000	\$ 656,640	\$ 504,960	\$ 344,960	\$ 353,280	\$ 2,499,840
Engineering Personnel	\$ 800,000	\$ 820,800	\$ 673,280	\$ 517,440	\$ 353,280	\$ 3,164,800
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Administration Personnel	\$ 300,000	\$ 307,800	\$ 315,600	\$ 194,040	\$ 198,720	\$ 1,316,160
Project Management	\$ 480,000	\$ 492,480	\$ 504,960	\$ 517,440	\$ 529,920	\$ 2,524,800
Total Salaries	\$ 2,940,000	\$ 3,016,440	\$ 2,756,240	\$ 2,091,320	\$ 1,700,160	\$ 12,504,160
Total ERE	\$ 820,554	\$ 841,888	\$ 769,267	\$ 583,687	\$ 474,515	\$ 3,489,911
Personnel Margin	\$ 1,128,166	\$ 1,157,499	\$ 1,057,652	\$ 802,502	\$ 652,402	\$ 4,798,221
TOTAL PERSONNEL	\$ 4,888,720	\$ 5,146,238	\$ 4,821,483	\$ 3,748,755	\$ 3,121,093	\$ 21,726,290

Figure 5.3.3.2: Personnel Budget

Personnel Cost Narrative

Personnel estimates were developed through evaluating the major functions of each phase of the NASA Mission Life Cycle's relationship to the functions of mission staffers. Evaluations of the Mission Life Cycle affirmed that each phase necessitates the involvement

of four key staffers who represent the decision making-authorities for the whole mission and its individual sub-teams, considered “Project Management Personnel”: the PM, CS, LSE, and DPMR (Figure 5.3.3.1). Scientists and Engineers make up most of the staff, for a total of twenty personnel per phase, through the onset of Phase D (Figure 5.3.3.1); this reflects their importance in establishing the conceptual components of the mission, final design of the system, and fabrication processes before their role decreases during the assembly, integration/testing, and pre-launch phases, as technicians become especially valuable for doing hands-on work with the system (Figure 5.3.3.1). This reflects their higher staffing numbers compared to other project departments.

Project staffing decreases across all departments with the exception of project management at the onset of Phase E, as the shift towards operations occurs and the spacecraft is no longer physically accessible for modifications or repairs (Figure 5.3.3.1). Technicians are the primary staffing force during this phase, as they will be responsible for conducting mission operations and handling on-the-fly system issues (Figure 5.3.3.1). Staff is further reduced after the completion of phase E, as the closeout phase of the mission will consist largely of administrative tasks and data archival that can be addressed by a comparatively minimal amount of staffers within their individual departments (Figure 5.3.3.1).

Personnel Cost Summary

Personnel costs are dominated by the engineering team during Phase C, before being picked up by the technician team during phases D-E. Overall expenditures are highest throughout Phases C-D, where they sit ~\$5,000,000 per period, before dropping to ~\$3 million to \$4 million per period as a result of reduced staffing (Figure 5.3.3.2). These metrics include employee benefit costs in addition to their salaries, and reflect the consideration of the standard 30% margin utilized by the Jet Propulsion Laboratory for Phase B Missions (Elkins-Tanton). The overall cost of personnel, including the aforementioned 30% margin, is projected to be \$21,726,290 (Figure 5.3.3.2). While a considerable cost for the mission, it is overshadowed by the direct costs, which make up a majority of the expenses associated with the mission (Figure 5.3.3.2). Major assumptions of the program include the use of a single engineering and flight unit; while risky, this is an essential component of allowing a balanced budget due to the expensive nature of the spacecraft’s mechanical subsystem and science instrumentation.

2.3.4. Travel Budget

Travel Cost Overview

Travel operations for the mission are focused on around the NASA Mission Life Cycle, launch, and the beginning of mission operations. NASA Mission Life Cycle reviews will be handled by a small team of five individuals, four of which are designated as project leadership consisting of the PM, DPMR, LSE, and Chief Scientist. The whole team is scheduled to travel to Cape Canaveral for launch and the beginning of operations as a form of acknowledgement for providing their services to the development of the mission, which culminates in a final travel cost of \$158,562 (Figure 5.3.4.1).

Budget Template

TRAVEL												
Total Flights Cost	\$	2,250	\$	-	\$	22,050	\$	2,250	\$	2,250	\$	28,800
Total Hotel Cost	\$	6,875	\$	-	\$	39,245	\$	6,875	\$	6,875	\$	59,870
Total Transportation Cost	\$	945	\$	-	\$	2,870	\$	945	\$	945	\$	5,705
Total Per Diem Cost	\$	2,070	\$	-	\$	15,057	\$	2,070	\$	2,070	\$	21,267
Travel Margin	\$	3,642	\$	-	\$	23,767	\$	3,642	\$	3,642	\$	34,693
Total Travel Costs	\$	15,782	\$	-	\$	108,344	\$	17,013	\$	17,423	\$	158,562

Figure 5.3.4.1: Travel Budget

Travel Cost Narrative

Cost estimates were developed through consideration of travel through NASA Mission Life Cycle Reviews in Washington, DC and launch activities in Cape Canaveral, Florida. All travel was considered to be based out of the Los Angeles airport due to its proximity to the team's base of operations being designated as JPL at CalTech. This choice is tied to the similarities between our project and the former Ingenuity Mars Helicopter mission, which eliminates travel costs for manufacturing, integration, and testing as the site is assumed to be equipped with the resources to carry out these operations based on its previous mission portfolio. Travel cost numbers for flights, hotels, and per diem rates were sourced from federal rates in accordance with NASA procedure, and transportation rates reflect costs of the minimum number of vehicle rentals for present staff in the area where they have traveled to.

Travel Cost Summary

Total travel costs come out to \$158,562, and are tied to the travel of the entire project team to the Cape Canaveral area for launch and the travel of project leadership and an additional staff member for KDP reviews at NASA Headquarters (Figure 5.3.4.1). Hotel cost is the primary cost driver, as consideration for staff members merits that they are each given their own private room. The assumption significant to travel is that all travel was

based out of the Los Angeles airport due to the project work site having been designated as JPL for reasons explained above.

2.3.5. Outreach Budget

Outreach Cost Overview

Outreach activities for this mission are strongly rooted in the local community, and involve forming partnerships with local institutions such as planetariums and community colleges. These programs are focused on educating the public about the lunar environment that our mission will explore, and include a workforce development element where project staff share about their journey to the space industry.

These activities serve to stoke interest in lunar exploration and space exploration, which support the mission's education objectives by helping build a foundation for the future space workforce that will contribute to later robotic and manned spaceflight missions throughout the remainder of the Artemis program and beyond.

Budget Template

OUTREACH							
Total Outreach Materials	\$	-	\$	-	\$	-	\$ 1,825
Total Outreach Venue Costs	\$	-	\$	-	\$	-	\$ -
Total Outreach Travel Costs	\$	-	\$	-	\$	-	\$ -
Total Outreach Services Costs	\$	-	\$	-	\$	-	\$ -
Total Outreach Personnel Costs	\$	-	\$	-	\$	-	\$ 17,636
Outreach Margin	\$	-	\$	-	\$	-	\$ 8,341
Total Outreach Costs	\$	-	\$	-	\$	-	\$ 30,693

Figure 5.3.5.1: Outreach Budget

Outreach Cost Narrative

Outreach cost estimates were developed with consideration to the maximum amount of expenses that could potentially be allocated to these categories to ensure flexibility in outreach plan development as the project progresses. These operations are rooted in the local community, and will occur at various planetariums and community colleges in the area surrounding the project worksite in the form of educational workshops and presentations.

Resources to support the development of the planetarium sessions include an informative video focused on the value of exploring PSRs and the team's mission, as well as supplemental lectures and presentation materials that will aid in educating the attendees of these public lecture sections. The community college workshops will be more technical,

geared towards equipping and inspiring the next generation of space industry professionals; materials include presentation and demonstration materials focused on illustrating the process of how the drone and base station collaborate and the simulations of lunar rover operations.

Outreach Cost Summary

Outreach costs are budgeted to be equally spread out across materials, venue, travel, services, and personnel costs across phases E-F of the mission with \$1,000,000 being allocated to each category for the phases. This is a generous estimate for each, but is designed to take advantage of the remaining budgetary resources to provide flexibility for sudden, unexpected costs related to outreach operations. The primary assumptions used in developing the estimate are that outreach operations will be conducted by in-house staff and that operations will be based in the area surrounding JPL, the base of operations for the project team.

2.3.6. Direct Costs

Direct Cost Overview

The mission's direct costs are expenses related to the development of the spacecraft, and consist of hardware and facility costs associated with manufacturing / integration and testing. Cost estimation was approached using parametric methods sourced from Cost-Estimating Relationships (CERs) provided in *The New SMAD*, which were then fed into the NICM to account for inflation, wrap costs, and testing costs.

The main components of direct costs are spacecraft costs, facility costs (for both manufacturing and integration / testing), and the margin for both larger categories. Margin was computed to be 50% of the cost of each category to operate in line for the margin convention used for the spacecraft direct costs category. The major assumption driving the assessment of these costs is the use of a single engineering and flight unit; while it is understood that it introduces extra risk, the complex nature of the spacecraft necessitates this choice in order to allow the project to adhere to its budgetary cap.

Budget Template

DIRECT COSTS						
Mechanical Subsystem	\$ 4,769,000	\$ 4,769,000	\$ 9,663,500	\$ 4,894,500	\$ -	\$ 24,096,000
Power Subsystem	\$ 3,458,000	\$ 3,458,000	\$ 7,007,000	\$ 3,549,000	\$ -	\$ 17,472,000
Thermal Control Subsystem	\$ 2,451,000	\$ 2,451,000	\$ 4,966,500	\$ 2,515,500	\$ -	\$ 12,384,000
Comms & Data Handling Subsystem	\$ 1,197,000	\$ 1,197,000	\$ 2,425,500	\$ 1,228,500	\$ -	\$ 6,048,000
Guidance, Nav, & Control Subsystem	\$ 2,185,000	\$ 2,185,000	\$ 4,427,500	\$ 2,242,500	\$ -	\$ 11,040,000
Science Instrumentation	\$ 5,206,000	\$ 5,206,000	\$ 10,549,000	\$ 5,343,000	\$ -	\$ 26,304,000
Spacecraft Cost Margin	\$ 9,633,000	\$ 9,633,000	\$ 19,519,500	\$ 9,886,500	\$ -	\$ 48,672,000
Total Spacecraft Direct Costs	\$ 28,899,000	\$ 29,650,374	\$ 61,603,542	\$ 31,972,941	\$ -	\$ 152,125,857
Manufacturing Facility Cost	Included in Spacecraft Cost	Included in Spacecraft Cost	Included in Spacecraft Cost	Included in Spacecraft Cost	\$ -	\$ -
Test Facility Cost	\$ 7,144,000	\$ 7,144,000	\$ 14,476,000	\$ 7,332,000	\$ -	\$ 36,096,000
Facility Cost Margin	\$ 3,572,000	\$ 3,572,000	\$ 7,238,000	\$ 3,666,000	\$ -	\$ 18,048,000
Total Facilities Costs	\$ 10,716,000	\$ 10,994,616	\$ 22,843,128	\$ 11,855,844	\$ -	\$ 56,409,588
Total Direct Costs	\$ 39,615,000	\$ 40,644,990	\$ 84,446,670	\$ 43,828,785	\$ -	\$ 208,535,445
Total MTDC	\$ 28,899,000	\$ 29,650,374	\$ 61,603,542	\$ 31,972,941	\$ -	\$ 152,125,857

Figure 5.3.6.1: Direct Costs

Subsystem Cost Estimates

The estimated cost for the mechanical subsystem which consists of the drone enclosure, base station enclosure, drone frame, base station frame, and necessary fasteners is \$24,096,000 (Figure 5.3.6.1, Appendix C). This cost estimate was derived using a parametric approach that combines the NICM CERs and *The New SMAD* CERs. This combined approach accounts for inflation and wrap costs based on the direct cost from the NICM CER. Major assumptions used for the estimate are that the aluminium and steel selected for enclosures, frame, and fasteners are suitable for surviving PSRs. This influences the cost estimate by driving down expenses, as these materials are commonly used in industrial contexts. Major cost drivers include the need for the payload to be entirely dust-proof, and which raises the cost by necessitating the use of more material to create fully-sealed enclosures. Sources of uncertainty in the cost estimate are the true durability of the material. They impact the estimate by introducing the possibility of a substantial cost increase if the selected materials prove to be insufficient.

The estimated cost for the power subsystem which consists of space-grade lithium ion batteries, two Ibeos 200W CubeSat EPS systems, and a DHV Technologies is \$17,472,000 (Figure 5.3.6.1, Appendix C). This cost estimate was derived using a parametric approach that combines the SMAD and NICM CERs, which accounts for inflation and wrap costs using the direct cost from the CERs and scaling metrics from the NICM. The major assumption used for the estimate is that EPS hardware utilized for a Cubesat can be suitably adapted for use with our payload, which influences the cost estimate by eliminating the need for expensive R&D efforts. The major cost driver is the need to ensure the electronics are protected from radiation which influences the cost estimate by necessitating the purchase of dedicated shielding and hardened electronics. Uncertainty arises from the ability to adapt the aforementioned Cubesat hardware, which could increase cost of the system if

adaptation is not possible due to the need for further R&D efforts to develop a custom solution for our mission.

The estimated cost for the thermal subsystem which consists of a custom thermoelectric cooling system, Lockheed Martin Space MICRO 1-1, Minco Polyimide Thermofoil HK Series, Pyrolytic Graphite thermal straps, and Aluminized Kapton insulation is \$12,384,000 (Figure 5.3.6.1, Appendix C). This cost estimate was derived using a parametric approach that combines the SMAD and NICM CERs, which accounts for inflation and wrap costs using the direct cost from the CERs and scaling metrics from the NICM. The major assumption used for this estimate is that the thermal straps can be made large enough with their limited area allotment to reject the needed amount of heat. The major driver for the cost estimate is the thermal conditions present on PSRs, which impacts the cost estimate by driving the selection of the materials used for heat rejection. Uncertainty in the cost estimate arises from the lack of knowledge of the exact behavior of our selected thermal system under PSR conditions, which impacts the cost estimate by introducing the potential for additional expenses should testing inform that further thermal refinement is needed.

The estimated cost for the Command and Data Handling (CD&H) subsystem which consists of a Spacewire data interface, three space-grade SSD and radiation shielding arrays, a microSD, a Xiphos Q8 Zynq UltraScale+, and two Polarfire SoCs is \$5,568,000 (Figure 5.3.6.1, Appendix C). This cost estimate was derived using a parametric approach that combines the SMAD and NICM CERs, which accounts for inflation and wrap costs using the direct cost from the CERs and scaling metrics from the NICM. The major assumption used for the estimate is that the selected components can handle PSR conditions which reduces cost by removing the need for R&D costs associated with custom components. The major cost driver is the need to protect the CD&H system from radiation, which increases cost through requiring the purchase of dedicated shielding for components. Uncertainty arises from the lack of use of these systems within lunar PSRs, which could increase cost should any of the systems be demonstrated as incapable of surviving during system testing.

The estimated cost for the Guidance, Navigation, and Control (GN&C) subsystem which consists of cold gas propellant, cold gas storage and plumbing, ten cold gas thrusters, and two IMUs is \$11,040,000 (Figure 5.3.6.1, Appendix C). This cost estimate was derived using a parametric approach that combines the SMAD and NICM CERs, which accounts for inflation and wrap costs using the direct cost from the CERs and scaling metrics from the NICM. The primary assumption for this system is that cold gas thrusters used for Cubesats can be

reasonably adapted for use with the drone, which drives down costs by preventing the need to undertake extensive R&D efforts.

The estimated cost for the payload subsystem which consists of a time-of-flight camera for depth measurement, stereoscopic camera, radiation dosimeter, and mass spectrometer is \$26,304,000 (Figure 5.3.6.1, Appendix C). This cost estimate was derived using a parametric approach that combines the SMAD and NICM CERs, which accounts for inflation and wrap costs using the direct cost from the CERs and scaling metrics from the NICM. The major assumption used and cost driver for the estimate is that the payload components will be able to survive PSR conditions, which increases the cost of the system as components rated for these conditions are harder to come by and more expensive. Uncertainty in the cost estimate rises from the unique PSR environment, and increases the cost estimate by necessitating the use of harder, more expensive components.

Other Direct Costs

Manufacturing, assembly, and integration costs are directly associated with the completion of the spacecraft and preparing it for pre-flight testing, a prerequisite for launch readiness. These costs were accounted for through the implementation of the NICM, which estimates these expenses as part of the program-level wrap expenses based on the cost input retrieved from the NICM Cost Estimating Relationships (CERs) sourced from *The New SMAD*.

Pre-flight testing costs are estimated through the same method since it is also considered a program wrap cost according to the NICM. These expenses are necessary to satisfy the verification and validation loop of the Systems Engineering process and ensure the customer that the spacecraft is a valuable use of economic resources and space aboard the launch vehicle

Cost Estimating Methods

Cost estimation was performed via a parametric approach, with analysis being performed using a combination of the NICM CERs and CERs from *The New SMAD*. CER results for each spacecraft subsystem were fed into their own NICM page, which accounted for facility cost and the impact of inflation between 2004 and 2026 while converting the cost metrics to millions of dollars. This process also provided information on the breakdown of costs across Phases B, C, & D; only direct costs associated with phases C and D were considered for the budget, as any Phase A and B costs are considered "sunk".

The use of these tools to build out an entirely parametric estimation approach was deemed appropriate for this mission largely due its unique nature that prompted concerns about the inability to locate and properly adapt analogous missions to accurately reflect the current mission.

Direct Cost Summary

Major contributors to direct costs include the mechanical subsystem, the science instrumentation, and testing facility costs. Mechanical subsystem costs are associated with the high-grade equipment needed to protect the science instrumentation from lunar dust; as such, these costs are essentially a method of buying down risk on the mission in accordance with NASA procedure. Science instrumentation and testing costs are closely related, as it is essential to ensure the lunar radiometer, time-of-flight camera, and stereoscopic cameras function reliably to maximize the likelihood of data acquisition.

The highest uncertainty areas are the costs associated with the electrical power system and facilities; uncertainty in the electrical power system's cost results from the potential need to size up the solar panel used for power generation. This is related to the uncertainty in facilities costs as the modeling approach used positions them to fluctuate directly on the cost of the overall spacecraft. The cost estimate is expected to fluctuate as the design maturity increases as component selection becomes baselined ahead of manufacturing efforts in Phase C.

2.4. Scope Management

2.4.1. Change Control Management

Change Control Overview

Change control maintains a single authoritative technical baseline as the dual-platform architecture matures, so that every requirement, interface, allocation, and component selection referenced in one section of this deliverable agrees with every other. Configuration management is essential to this mission because the base station and the scout drone share command and data handling, power, and thermal interfaces; an unmanaged modification on either platform propagates into the opposite platform's interface definitions, mass and power allocations, verification approach, and cost estimate

(NASA, 2016; NASA, 2008). Changes are therefore separated into internal working changes, which are dispositioned inside the team and affect no controlled item, and baseline changes, which require review by the Change Control Board (CCB) chaired by the customer representative and staffed by assigned subject matter experts external to the team. Formal board review is required for any modification to a system requirement, a science objective, an approved subsystem design parameter, a mass or power allocation, a component or supplier selection, a verification method, a controlled milestone date, or a budget line item.

Two quantities govern classification and are used consistently throughout this deliverable: margin is the unallocated technical capability held at the subsystem and system level in mass, power, data volume, and link budgets, while reserve is the unallocated cost and schedule held at the project level for future authorization. A change that consumes margin is Class II; a change that consumes reserve beyond its established limit or alters a customer constraint is Class I. Table 2.4.1-1 defines the three change classes, their thresholds, the approving authority for each, and the forum in which each is dispositioned.

Class	Definition and threshold	Approval authority	Forum and timeline
Class III (internal working change)	Affects no controlled baseline item; no cost, schedule, mass, or power impact. Includes editorial revisions, internal analysis refinement, tool changes, and work reassignment.	Responsible subteam lead, with Project Manager notification	Dispositioned at the weekly subteam meeting; recorded in the Internal Design Note (IDN) log within 48 hr
Class II (baseline change within reserve)	Modifies a controlled baseline item, with cost, schedule, mass, or power impact contained within the allocated subsystem margin and the project cost and schedule reserve.	Engineering Change Panel: Project Manager, Deputy Project Manager of Resources, Chief Scientist, Lead Systems Engineer (majority disposition)	Standing agenda item at the weekly systems engineering meeting; disposition within one week of classification; CCB informed in the monthly status report
Class I (controlled baseline change)	Alters a system requirement, a science objective, the mission architecture, a customer constraint (cost cap, launch date, mission duration), or an impact that exceeds the project reserve.	Change Control Board: customer representative (chair) and assigned subject matter experts, supported by the Project Manager and Lead Systems Engineer as presenters	Dedicated board session within five business days of classification; unanimous board concurrence required before implementation

Table 2.4.1-1. Change classification, approval authority, and disposition forum.

Change Request Process

Any team member may originate a change request, which is documented on the standard form presented in Table 2.4.1-3 and submitted to the Configuration Manager, a role held by the Deputy Project Manager of Resources, within 24 hours of identification. The Configuration Manager screens each request for completeness, returns incomplete submissions to the originator within 48 hours, and assigns a classification of Class I, II, or III within the same period; every affected subteam lead then provides written technical concurrence on the stated cost, schedule, mass, power, risk, and verification impacts before disposition. Class III requests are dispositioned by the responsible subteam lead at the weekly subteam meeting with notification to the Project Manager and are recorded in the internal design note log. Class II requests are dispositioned by majority vote of the Engineering Change Panel, comprising the Project Manager, the Deputy Project Manager of Resources, the Chief Scientist, and the Lead Systems Engineer, as a standing agenda item at the weekly systems engineering meeting. Class I requests are presented by the Project Manager and the Lead Systems Engineer to the CCB in a dedicated session convened within five business days, where unanimous board concurrence is required; the board may approve, approve with conditions, defer, or reject, and each disposition is recorded with its rationale. No change is implemented before its disposition is recorded, and any preparatory analysis performed in anticipation of approval is held outside the controlled baseline until the request is approved and logged. The complete step, owner, and timeline sequence is presented in Table 2.4.1-6.

Change Tracking and Communication

Approved changes are recorded in the change log, presented as Table 2.4.1-4 in Section 2.4.1.5, which is the authoritative record of every baseline change and is carried forward in each submitted deliverable. Controlled artifacts are stored in a revision-controlled team repository under a sequential revision index, and the configuration tracking register, presented as Table 2.4.1-5, records the current revision, owner, last approved change, and open request status for each controlled item so that configuration status is verifiable at any point in development. Requirements changes are traced through the requirements traceability matrix, where each affected requirement carries the identifier of the change request that last modified it, and design changes are traced through the corresponding drawing or component list revision. Internally, the Configuration Manager notifies all affected subteam leads through the team communication channel within 24 hours of

disposition, assigns an implementation owner and completion date, and records the action in the weekly meeting minutes.

Stakeholder communication is deliberate rather than incidental. A configuration status accounting report is issued to the customer representative and assigned subject matter experts each month, summarizing requests approved, pending, and rejected during the period, the resulting change in cost, schedule, mass, power, and remaining margin and reserve, and any open corrective actions (NASA, 2008). Class I dispositions are additionally transmitted to the customer representative within 24 hours in a written impact memorandum, and any request that would affect a customer constraint is raised to the board before implementation rather than reported after the fact. Functional and physical configuration audits are planned prior to hardware acceptance so that the delivered configuration is demonstrated to match the approved baseline (NASA, 2016).

Requests for Action and Advisory Comments

Requests for action (RFAs) and advisory comments (ADVs) received from stakeholders are entered into the feedback register within 24 hours of receipt, assigned to the responsible subteam lead with a closure date set before the next deliverable submission, and dispositioned as concur, partially concur, or non-concur with recorded rationale (NASA, 2020). Any item whose resolution alters a controlled baseline item is converted into a change request, so that stakeholder feedback enters the same classification, approval, and verification path as internally originated change and cannot be implemented informally. Closure is verified by the Lead Systems Engineer and the Project Manager against the responsible section, with the specific table, figure, or narrative that satisfies the item recorded as closure evidence; a partially concurred item remains open, with the outstanding portion tracked to a later milestone. Closure statements are returned to the originating stakeholders in the following deliverable and in the monthly configuration status accounting report, and recurring feedback themes are captured in a lessons-learned log that is reviewed during the planning of each subsequent deliverable so that repeated findings are prevented rather than re-answered. Table 2.4.1-2 presents the feedback incorporation matrix, which records the disposition, resulting change request, closure evidence, and status of each item received from the preceding review.

Feedback ID	Disposition	Action taken and resulting change request	Closure evidence	Status
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RFA-14	Concur	Change control restructured into three change classes with an external Change Control Board and an internal Engineering Change Panel (CR-020)	Section 2.4.1 and Figure 2.4.1-1	Closed
RFA-15	Concur	Scope reduction and growth options extended to all five subsystems with prioritized ordering and traceability (CR-019)	Tables 2.4.2-1 and 2.4.2-2	Closed
RFA-03	Concur	Risk matrix reinstated and risk handling approaches and owners revised (CR-009, CR-010)	Risk analysis section and risk log	Closed
ADV-05	Concur	Margin and reserve defined as distinct quantities throughout the budget narrative (CR-015)	Budget overview section	Closed
ADV-03	Partially concur	Supplier justification expanded and thermal backup suppliers added; two long-lead ranges remain broad pending vendor confirmation (CR-016)	Procurement summary table	In work

Table 2.4.1-2. Feedback incorporation matrix: disposition and closure of stakeholder requests for action and advisory comments from the preceding review.

Supporting Documentation

Four artifacts constitute the change control record and are maintained on the team repository: the change request form, the change log, the configuration tracking register, and the change approval workflow. Each is updated after every panel or board action, and all four are reviewed by the Lead Systems Engineer before submission to confirm that no approved change remains unincorporated and that no unapproved change has been introduced.

Change Request Form

Table 2.4.1-3 presents the standard form used for every change submission. The Configuration Manager assigns the identifier on receipt, and every field is completed before classification, so that the cost, schedule, mass, power, risk, and verification consequences of the change are documented before any authority acts on it.

Field	Entry
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CR identifier	Assigned by the Configuration Manager upon receipt (sequential, e.g. CR-021)
Date submitted	Date the request was delivered to the Configuration Manager
Submitted by	Name, role, and subteam of the originator
Affected baseline items	Requirements, drawings, component selections, interfaces, allocations, or deliverable sections affected
Change description	Statement of the modification, including the as-is and to-be configuration
Technical rationale	Technical or programmatic justification, including the analysis, trade study, or stakeholder feedback that prompted the request
Cost impact	None / within reserve / exceeds reserve, with the estimated dollar value and the affected fiscal year
Schedule impact	None / within float / consumes critical path float / extends a controlled milestone, with the estimated duration
Mass impact	None / within subsystem allocation / within system margin / exceeds margin, in kilograms
Power impact	None / within subsystem allocation / within system margin / exceeds margin, in watts
Risk impact	New, retired, or revised risk identifiers, with the resulting change in likelihood and consequence
Verification impact	Verification methods or test activities added, deleted, or revised as a result of the change
Classification	Class I / II / III, assigned by the Configuration Manager within 48 hr of receipt
Affected lead concurrence	Signature and date from each subteam lead whose baseline is affected
Disposition	Approved / approved with conditions / deferred / rejected, with date, deciding body, and recorded rationale
Implementation owner	Subteam lead accountable for executing the change and the committed completion date
Verification of incorporation	Lead Systems Engineer confirmation that all affected artifacts were updated and remain internally consistent
Status	Open / classified / in board review / approved-in-work / verified / closed / rejected

Table 2.4.1-3. Change request form.

Change Log

Table 2.4.1-4 is the authoritative record of approved baseline changes and lists the twelve changes approved since the previous review, each traced to the analysis or stakeholder

comment that prompted it. Requests CR-001 through CR-008, which established the dual-platform architecture, the TRL-6 technology readiness floor, the 175 kg mass baseline, the flight computer selections, and the power subsystem baseline, remain closed and are carried in the deliverable appendix.

CR ID	Affected section	Description of approved change	Rationale or originating feedback	Class	Status
CR-009	Risk analysis	Risk matrix reinstated as a required figure with likelihood and consequence placement for every logged risk, and matched line-for-line to the risk log	Stakeholder request for action on risk documentation completeness	III	Closed
CR-010	Risk log	Risk handling approaches diversified across mitigate, watch, and accept; trends updated; named individual risk owners substituted for subsystem owners	Stakeholder feedback: uniform mitigate, neutral, and open entries did not reflect the current risk posture	III	Closed
CR-011	Failure modes and effects analysis	Seven subsystem worksheets consolidated into a single mission-level analysis with cascading failure paths and recomputed risk priority numbers	Stakeholder feedback: per-subsystem worksheets obscured cross-subsystem failure propagation	III	Closed
CR-012	Team management	Organizational chart updated to current composition; time-sensitive decision authority defined; escalation path added for lead-to-lead and lead-to-member disagreement	Stakeholder request for action on team management completeness	III	Closed
CR-013	Project management approach	Phase C through Phase F workforce increased above the previous 20-person estimate, with a floating technician pool and a dedicated verification and validation team added	Stakeholder feedback: staffing insufficient for the dual-platform architecture	II	Closed
CR-014	Mission schedule	Schedule expanded to include all robotic mission lifecycle reviews; launch relocated from Phase E to Phase D; critical path identified and annotated on the milestone chart	Alignment with the NASA lifecycle sequence and stakeholder request for action on schedule completeness	II	Closed

CR-015	Budget	Margin and reserve defined as distinct quantities; the 30 percent unallocated future expense held at project level reclassified explicitly as reserve	Stakeholder advisory on interchangeable use of the two terms	III	Closed
CR-016	Procurement	Supplier selection rationale expanded beyond vendor qualification status; backup suppliers added for thermal components; lead-time ranges narrowed and tied to switchover triggers	Stakeholder advisory on procurement justification depth	II	Closed
CR-017	Mission requirements	Technology readiness floor confirmed at TRL-6; requirement identifiers restructured to distinguish parent from child system requirements	Stakeholder advisory: TRL-8 floor unachievable for permanently shadowed region hardware	II	Closed
CR-018	Concept of operations	Mission timeline graphic expanded with per-phase activities; drone excursion cadence, recharge duration, and terrain point count quantified	Stakeholder advisory on operational detail	III	Closed
CR-019	Scope control	Scope reduction and growth options extended from Command and Data Handling alone to all five subsystems, with traceability to requirements and risks and a prioritized ordering	Stakeholder request for action on scope control breadth	III	Closed
CR-020	Change control	Change control restructured into three change classes; Change Control Board membership redefined as the customer representative and external subject matter experts; Engineering Change Panel established for internal disposition	Stakeholder request for action on change control feasibility and board composition	III	Closed

Table 2.4.1-4. Change log: baseline changes approved since the previous review.

Configuration Tracking

Configuration control is maintained at the item level rather than the subteam level, so that configuration status can be demonstrated on demand rather than reconstructed at

submission. Table 2.4.1-5 records the current revision, responsible owner, last approved change, and open request status for each controlled baseline item; the Lead Systems Engineer reconciles the register against the change log before every submission and reports any discrepancy as a corrective action.

Controlled item	Rev	Owner	Last approved change	Open CRs	Configuration status
System requirements table (SYS series)	C	Lead Systems Engineer	CR-017	None	Baselined and current
Science traceability matrix	B	Chief Scientist	CR-005	None	Baselined and current
Mass and power allocation budget	C	Lead Systems Engineer	CR-013	CR-021	Baselined, change in work
CDH architecture and component list	C	CDH Lead	CR-007	None	Baselined and current
Power subsystem architecture	B	Power Lead	CR-008	None	Baselined and current
Integrated mission schedule	C	Deputy Project Manager of Resources	CR-014	None	Baselined and current
Cost estimate and reserve posture	C	Deputy Project Manager of Resources	CR-015	None	Baselined and current
Risk log and risk matrix	C	Project Manager	CR-010	None	Baselined and current

Table 2.4.1-5. Configuration tracking register for controlled baseline items.

Change Approval Workflow

Table 2.4.1-6 presents the step-by-step workflow from identification of a change through classification, technical concurrence, disposition by the appropriate authority, implementation, verification of incorporation, and closure, with the responsible owner and committed timeline stated for each step.

Step	Action	Owner	Timeline
1	Change identified and change request form completed with a supporting impact assessment	Any team member	As identified

2	Request submitted to the Configuration Manager	Originator	Within 24 hr of identification
3	Request screened for completeness and classified as Class I, II, or III; incomplete requests returned	Configuration Manager	Within 48 hr of receipt
4	Affected subteam leads provide technical concurrence on the stated cost, schedule, mass, power, risk, and verification impacts	Affected subteam leads	Within 3 business days of classification
5a	Class III dispositioned and recorded in the Internal Design Note log	Subteam lead, Project Manager notified	Within 48 hr of classification
5b	Class II dispositioned by majority vote of the Engineering Change Panel	Engineering Change Panel	Next weekly systems engineering meeting
5c	Class I presented to the Change Control Board; unanimous board concurrence required	Change Control Board	Dedicated session within 5 business days
6	Disposition and rationale recorded in the change log; revision index and configuration tracking register updated	Configuration Manager	Within 24 hr of disposition
7	Affected leads notified through the team channel; implementation owner and completion date assigned	Configuration Manager	Within 24 hr of disposition
8	Affected deliverable sections, drawings, allocations, and the requirements traceability matrix updated	Implementation owner	Before the next submission
9	Cross-section consistency and traceability verified; corrective action assigned if incomplete	Lead Systems Engineer	Pre-submission review
10	Request closed and reported to stakeholders in the configuration status accounting report	Configuration Manager	Monthly

Table 2.4.1-6. Change approval workflow from identification through closure.

Change Control Flowchart

Figure 2.4.1-1 presents the change control process as six sequential steps, from identification of a change through submission, classification, disposition, logging, and implementation with verification, with the responsible party and the committed timeline stated at each step. Three decision groups branch from the sequence: the classification outcome at Step 3, which routes the request to the subteam lead, the Engineering Change Panel, or the Change Control Board; the disposition outcome at Step 4, which may approve,

approve with conditions, defer, or reject; and the verification outcome at Step 6, where a change found incompletely incorporated returns to its implementation owner for corrective action before submission rather than closing. Requests for action and advisory comments enter the same sequence at Step 1, so stakeholder feedback that alters a controlled baseline item cannot be implemented outside this path. The process is scaled to a three-subteam organization with five active subsystems: internal working changes are resolved without board overhead, while architectural and constraint-affecting changes receive full board scrutiny before implementation, as occurred for the architecture revision recorded as CR-001.

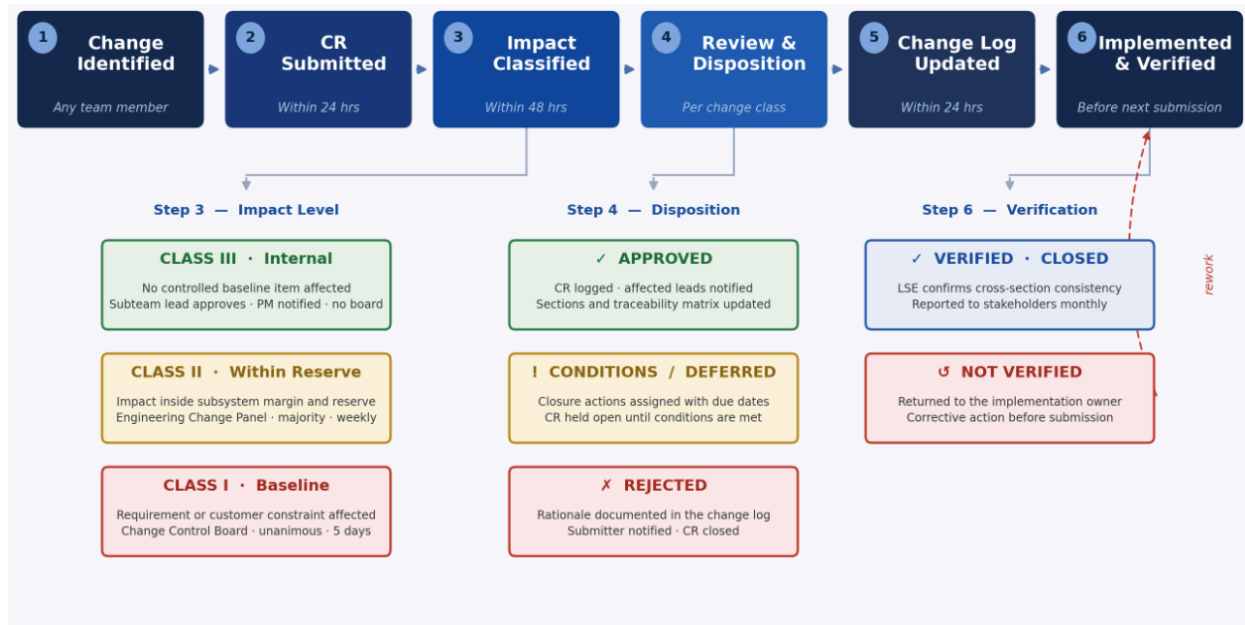


Figure 2.4.1-1. Change control process through implementation & verification

2.4.2 Scope Control Management

Scope Control Overview

Mission scope is defined by the science objectives carried in the science traceability matrix, the system requirements derived from them, and the customer constraints on cost, launch date, and mission duration; a proposed change to any of these is a scope change and is processed as a Class I or Class II change request rather than as an informal adjustment. Every proposed scope change is evaluated against five criteria before disposition: traceability to the affected science objective and the resulting change in science return, cost against the mission cost cap and the remaining project reserve, schedule against the launch date and the remaining critical path float, technical performance against mass,

power, data volume, thermal, and link margins, and the resulting change in risk posture expressed as a revised likelihood and consequence in the risk log. The evaluation is prepared by the Lead Systems Engineer with the Chief Scientist and is presented with the change request, so that the deciding authority sees the science consequence and the programmatic consequence together. Scope changes that preserve all Level 1 science objectives may be dispositioned by the Engineering Change Panel; any change that reduces a Level 1 objective or affects a customer constraint requires CCB concurrence.

Managing Scope Reduction

A prioritized descope ladder is maintained jointly by the Lead Systems Engineer and the Chief Scientist and is reviewed at each lifecycle review, so that reduction decisions are made against a pre-analyzed ordering rather than under schedule pressure. The ladder is invoked when a projected cost growth would consume more than half of the remaining project reserve, when critical path float falls below four weeks, or when a technical shortfall cannot be closed within the affected subsystem's margin. Reduction actions are applied in the order presented in Table 2.4.2-1, which is ordered by ascending impact to mission success: actions that slow the rate of science return or reduce implementation effort are taken first, actions that remove redundancy and therefore raise risk consequence are taken next, and removal of a science instrument is taken last. Each entry is traced to the requirement and risk it affects, states the relief obtained, and states the consequence the team would accept, so that stakeholders can see what is being traded rather than only what is being saved. Two boundaries are held firm: environmental qualification of hardware exposed to the permanently shadowed region is not eligible for descope at any point on the ladder, because the qualification evidence is the basis of the survival case for both platforms, and any action affecting a Level 1 science objective requires customer concurrence before implementation.

The ladder deliberately spans procurement method, subsystem design, testing, science capability, and mission operations rather than concentrating on a single subsystem. On the procurement side, the first action converts the base station primary structure and the drone airframe from separate qualification and flight builds to a protoflight approach, which removes a full structural build and its qualification campaign at the cost of reduced structural test margin. Operations reductions follow, because excursion cadence and contact scheduling can be relaxed without changing what is measured, only how quickly the measurement set is completed. Design simplification is then applied across command and data handling, thermal, and power: deferral of onboard compression, substitution of passive thermal control for thermoelectric coolers on the radiometer, and reversion of the drone flight computer and the base station store to single-string configurations, in that

order, because the first two reduce capability while the third reduces fault tolerance. Testing tailoring is applied only after design simplification is exhausted, and the regolith characterization instrument, the lowest-ranked contributor to the science traceability matrix, is the final action before the mission would be declared non-compliant with its constraints.

#	Area	Descope action	Estimated relief	Traceability	Consequence accepted
1	Manufacturing and procurement	Adopt a protoflight build in place of separate qualification and flight units for the base station primary structure and the drone airframe	Two structural builds and one qualification campaign removed	Structural qualification requirement; mechanical procurement risk	Reduced structural test margin; qualification and acceptance loads applied to flight hardware
2	Operations	Reduce drone excursions per lunation and consolidate direct-to-Earth contact windows, extending the same data return over a longer period	Operations staffing and ground station contact hours reduced	Terrain mapping objective; downlink volume requirement	Slower science return; no loss of measurement content
3	CDH	Defer onboard data compression and pre-processing to a later phase and transmit at lower bandwidth efficiency	Software development and verification effort deferred	Data handling requirement; data storage overflow risk	Higher storage utilization and longer downlink duration
4	Thermal	Replace thermoelectric coolers on the lunar radiometer with a passive radiator and heater set, and reduce the thermal sensor count to the minimum needed for control authority	Component cost, power draw, and integration effort reduced	Radiometer thermal stability requirement; thermal control risk	Narrower instrument operating temperature range and reduced diagnostic resolution
5	CDH	Revert the drone flight computer from the dual active and cold-spare configuration to a single unit with software-only fault recovery, and the base station store from mirrored drives to a single solid-state drive	Unit cost, mass, and standby power reduced on both platforms	Fault tolerance requirement; single-point-failure and data-loss risks	Loss of hardware redundancy; consequence of a flight computer or drive failure rises to mission-critical

6	Power	Resize the array area and battery capacity to the shortest survival duration consistent with the reduced excursion cadence, and reduce the power conditioning unit to a single string	Array area, battery mass, and unit cost reduced	Survival duration requirement; battery depth-of-discharge risk	Reduced eclipse survival margin; loss of power distribution redundancy
7	Testing	Reduce unit-level thermal vacuum cycling to the protoflight minimum while retaining full-system thermal vacuum testing and permanently shadowed region qualification of all cold-exposed hardware	Test facility days and test support labor reduced	Environmental verification requirement; workmanship defect risk	Lower confidence in workmanship screening; environmental qualification is retained and is not eligible for descope
8	Science instruments	Defer the regolith characterization instrument, the lowest-priority contributor to the science traceability matrix, retaining the atomic clock, inertial measurement units, and lunar radiometer	Instrument procurement, accommodation, and calibration effort removed	Lowest-ranked science objective; payload accommodation requirements	One secondary science objective is not met; navigation and thermal objectives are unaffected. Requires customer concurrence

Table 2.4.2-1. Prioritized descope ladder.

ordered by ascending impact to mission success. Relief estimates are engineering estimates to be updated against the current cost, mass, and power models at each lifecycle review; requirement and risk identifiers are maintained in the requirements table and risk log.

Managing Scope Growth

If additional cost or schedule reserve becomes available, it is applied in a fixed priority order that places risk retirement ahead of margin recovery and margin recovery ahead of new capability, as presented in Table 2.4.2-2. The first allocation extends radiation and thermal qualification of the flight computers and power electronics beyond the TRL-6 minimum, because those items carry the highest-rated subsystem risks and because qualification evidence obtained before hardware commitment is the least expensive form of risk reduction available (NASA, 2016). The second allocation restores redundancy where the failure modes analysis identifies remaining single-point failures, specifically a second

base station flight computer, redundant survival heater circuits, and spare antennas for the local and direct-to-Earth links. The third restores technical margin consumed during detailed design by increasing storage above the 8 TB floor, adding battery capacity for longer dwell in the permanently shadowed region, and raising link margin. Expanded science operations follow, since additional excursions and duty cycles increase data return against existing objectives without generating new requirements, and additional mission capability is considered last and only while schedule reserve remains before the design is released, because new capability generates new requirements, accommodation, and verification scope at the phase of development where that scope is most expensive to absorb.

#	Area	Enhancement	Benefit realized	Traceability
1	Verification	Extend radiation and thermal qualification of the flight computers and power electronics beyond the TRL-6 minimum toward flight qualification of the delivered units	Retires the highest-rated CDH and power risks before hardware commitment, when correction remains inexpensive	Highest-rated subsystem risks; technology readiness requirement
2	Redundancy	Add a second base station flight computer, redundant survival heater circuits, and a spare antenna for both the local link and the direct-to-Earth link	Removes remaining single-point failures identified in the failure modes analysis	Fault tolerance requirement; communication and thermal failure modes
3	Design margin	Increase base station storage above the 8 TB floor, add battery capacity for longer permanently shadowed region dwell, and raise link margin with a higher-gain antenna	Restores margin consumed during detailed design and reduces the likelihood of storage overflow and link closure failure	Storage, survival duration, and link margin requirements
4	Science operations	Increase drone excursion cadence, extend the radiometer duty cycle, and add ground contact windows to increase data return	Increases measurement density against existing objectives without new hardware	Terrain mapping and thermal characterization objectives
5	Mission capability	Add a third drone flight computer or an additional science instrument	Adds capability but also adds accommodation, integration, and verification scope; considered only with schedule reserve remaining before the design is released	New requirements would be generated and require board approval

Table 2.4.2-2. Prioritized scope growth ladder, ordered by risk retired per unit of resource applied.

Scope Management Throughout the Mission Lifecycle

The latitude available for scope change narrows continuously as the design matures, and the cost of a given change rises with the number of downstream artifacts already committed to it. Through Phase B, scope changes are documentary and are the least expensive to implement, because architecture, instrument complement, requirements, and allocations exist only as analysis and drawings. In Phase C, changes remain practical until the critical design review; afterward, released drawings and placed long-lead orders mean that a design change carries re-release, re-analysis, and potential reprocurement cost. In Phase D, hardware exists and verification is underway, so a scope change requires rebuild, requalification, and retest and directly threatens the launch date constraint, which makes this the phase in which scope discipline matters most and in which descope is preferred over accommodation. In Phase E, scope adjustment is limited to what the delivered hardware supports, principally excursion cadence, instrument duty cycles, contact scheduling, uplinked software, and extended mission planning, while Phase F admits only closeout scope. Three decision gates are therefore fixed in the schedule: a descope review immediately preceding the critical design review, a long-lead procurement confirmation at the system integration review, and an operations reprioritization checkpoint following the post-landing assessment review. Table 2.4.2-3 summarizes the latitude, practical change set, and relative cost of change by phase.

Phase	Scope change latitude	Changes that remain practical	Relative cost of change
Pre-Phase A through B	High	Architecture, instrument complement, requirements, allocations, and operations concept; changes are documentary	Lowest; analysis and documentation rework only
Phase C, before the critical design review	Moderate	Subsystem simplification, redundancy level, test program content, supplier selection, and scope of deferred capabilities	Moderate; drawing revision and re-analysis, but no procured hardware affected
Phase C, after the critical design review	Limited	Test program tailoring and operations planning; hardware changes require re-release of drawings and reassessment of long-lead orders	High; rework of released design and possible reprocurement
Phase D	Very limited	Flight software, sequences, procedures, and operations staffing; hardware changes require rebuild, requalification, and threaten the launch date	Highest; hardware rework, retest, and direct schedule impact to a customer constraint

Phase E	Operational only	Excursion cadence, instrument duty cycles, contact schedules, uplinked software, and extended mission planning	Low in cost but bounded by the delivered hardware capability
Phase F	Closeout only	Data archiving, disposal execution, and final reporting scope	Low; no design change is available

Table 2.4.2-3. Scope changes latitude and relative cost of change by mission phase.

Scope Management Summary

Mission objectives are protected by controlling scope through the same classified change process that governs the technical baseline, by maintaining pre-analyzed and prioritized reduction and growth ladders that are traced to specific requirements and risks, and by fixing decision gates at the points in the lifecycle where change remains affordable. Reduction actions are ordered so that rate of science return is traded before fault tolerance and fault tolerance before science content, while growth resources are directed first at retiring the highest-rated risks and only afterward at margin and capability. Managing scope in this order keeps the mission within its cost cap and launch date, preserves the technical margins on which the Level 1 science objectives depend, and ensures that any change to what the mission delivers is a deliberate, documented, and stakeholder-approved decision rather than a consequence of accumulated pressure.

2.5. Outreach Summary

The mission's outreach efforts aim to increase public awareness and understanding of lunar south pole exploration while creating accessible educational opportunities for students and the general public. Through workshops, presentations, hands-on activities, and educational materials, the outreach program will introduce participants to the scientific and engineering challenges associated with operating in the lunar south pole environment, provide opportunities to interact with mission scientists and engineers, and encourage greater interest in STEM education and careers. Outreach content will directly connect these activities to the mission's objectives by explaining how the stationary Base Station and autonomous Scout Drone work together to characterize terrain, environmental hazards, and mobility challenges in and around permanently shadowed regions. By presenting these concepts through accessible and interactive experiences, the outreach program will help participants understand the purpose of the mission, appreciate the challenges

involved in lunar exploration, and recognize how the information collected by the mission could contribute to safer and more reliable future exploration of the Moon.

The outreach program will primarily engage community college students and members of the general public through activities conducted at local community colleges, museums, and planetariums. Community college students, particularly those from historically underrepresented or underserved backgrounds in STEM, were selected because they may have fewer opportunities to interact directly with aerospace professionals or participate in activities connected to space missions. Museums and planetariums will provide opportunities to reach a broader audience, including students, families, and community members with varying levels of familiarity with lunar science and engineering. These audiences were selected because they allow the outreach program to combine focused STEM engagement with broader public education, reaching both individuals considering future STEM careers and those interested in learning more about lunar exploration. Together, these groups provide opportunities to increase awareness of the mission among audiences that may otherwise have limited exposure to its science, technology, and objectives.

Activity 1: Community College STEM Workshops

The mission team will conduct STEM workshops at local community colleges, with an emphasis on reaching students from historically underrepresented or underserved backgrounds in STEM. Each workshop will combine a presentation on the mission with interactive activities that allow students to explore some of the same scientific and engineering challenges addressed by the mission team. Activities may include navigating a simulated lunar surface, interpreting maps of lunar terrain, identifying potential hazards, and developing solutions to challenges that an autonomous vehicle could encounter while operating near the lunar south pole. These activities will be directly connected to the mission by introducing students to the stationary Base Station and autonomous Scout Drone and explaining how the two platforms work together to investigate areas that are difficult or hazardous to access. Students will also learn about the challenges associated with permanently shadowed regions and how observations of terrain and environmental hazards can provide information relevant to future lunar surface operations. Mission scientists and engineers will participate in the workshops when possible to explain how science objectives influence engineering decisions and answer questions about the different

disciplines involved in developing a lunar mission. By combining mission-specific information with hands-on problem solving, the workshops are expected to improve students' understanding of lunar exploration while demonstrating how concepts learned in STEM courses can be applied to real spaceflight challenges.

Activity 2: Museum and Planetarium Programs

The mission team will partner with local museums and planetariums to provide public lectures, educational presentations, and an informational video designed for a general audience, including families, students, and community members with varying levels of prior knowledge about space exploration. The programs will use visual explanations and planetarium capabilities to introduce visitors to the lunar south pole, explain the importance of exploring this region, and demonstrate some of the environmental and operational challenges associated with conducting a mission near permanently shadowed regions. Mission-specific content will focus on the Base Station and autonomous Scout Drone, including how the two platforms support exploration of difficult-to-access areas and collect information about terrain, environmental hazards, and mobility conditions that can affect future lunar operations. Presentations will use mission imagery, diagrams, informational posters, and other educational materials to make these concepts easier to understand, while the planetarium format can provide an immersive way to place the mission within the broader context of lunar exploration. Public lectures will also provide opportunities for mission team members to discuss their individual roles and experiences working on the mission, helping visitors understand the range of scientific, engineering, and programmatic disciplines required to develop and operate a space mission. The expected outcome is for visitors to leave with a clearer understanding of why the lunar south pole is an important region for exploration, what scientific and engineering challenges the mission is designed to address, and how the resulting information can support future lunar activities. By combining mission science with visual learning and personal interaction, the museum and planetarium programs will encourage broader public appreciation of lunar exploration and the work required to support future exploration of the Moon.

Outreach activities will be designed to reduce barriers that may prevent underrepresented or underserved communities from engaging with space science and aerospace-related opportunities. Events will be offered at no cost to participants, with hands-on demonstrations, videos, visual explanations, and

accessible language used to accommodate different levels of technical knowledge. Digital versions of educational materials will provide an additional way for individuals who cannot attend in person to access mission information. Participants will also have opportunities to ask team members about STEM education and career pathways, providing a more personal connection to the people and disciplines involved in space exploration. Together, these approaches will make the outreach program accessible to audiences with a range of backgrounds, experiences, and levels of STEM familiarity.

Outreach planning will begin during the mission development phases, when educational materials, presentations, workshop content, and other resources will be prepared and updated as the mission design matures. During mission operations, community college workshops and museum and planetarium programs will incorporate current mission information and operational results when appropriate. Local community colleges will provide venues for student-focused workshops, while participating museums and planetariums will provide facilities and educational settings for broader public programming. Following mission operations, presentation materials and educational resources will remain available digitally to continue supporting STEM education and public interest. This lifecycle approach allows outreach content to mature alongside the mission and retain educational value after mission completion.

The outreach plan supports the mission by making its scientific and engineering objectives understandable and relevant to audiences beyond the mission team. By combining educational experiences, public engagement, and continued access to mission resources, the program will strengthen community understanding of the challenges associated with lunar exploration while encouraging continued interest in STEM and space science. These efforts will extend the mission's educational value beyond its scientific results and help connect the public with the role this work can play in future lunar exploration.

3. Conclusion

1. PDR Summary

The Preliminary Design Review advances Team 24's lunar south-pole mission to a preliminary dual-platform design, with a stationary Base Station supporting power, communications, data storage, and docking while an autonomous Scout Drone characterizes terrain, illumination, dust, radiation, and mobility hazards near permanently shadowed regions. Major accomplishments during this phase include development of the integrated vehicle CAD, refinement of subsystem architectures and requirements, selection and characterization of the ten-instrument science payload, and definition of Base Station and Scout Drone interfaces through system block diagrams and N² analysis. The team also matured subsystem procurement and verification plans, mission risk and FMEA analyses, and the Phase C through F cost and schedule baselines, with a current estimated mission cost of approximately \$240.8 million within the \$250 million customer constraint. Together, these products establish the preliminary technical and programmatic baseline required to proceed into Phase C detailed design and preparation for the Critical Design Review.

2. Future Work and Upcoming Milestones

Following PDR, the team will prepare for CDR by converting the preliminary vehicle architecture into a detailed design through final component packaging, engineering drawings, structural and thermal analysis, propulsion sizing, and refinement of integrated mass, power, thermal, and interface margins. Subsystem teams will advance procurement of long-lead hardware and mature verification activities into detailed test procedures for structural, thermal-vacuum, vibration, electromagnetic, communications, autonomy, and integrated vehicle testing. Science development will refine instrument calibration procedures, uncertainty models, data-processing methods, and mission operations as hardware and environmental models mature, while programmatic baselines will be updated using supplier, staffing, and test-facility information. With additional development time, the team would perform higher-fidelity coupled simulations and hardware-in-the-loop testing of autonomous flight, docking, PSR thermal survival, and mission-level fault recovery to further reduce technical uncertainty before CDR.

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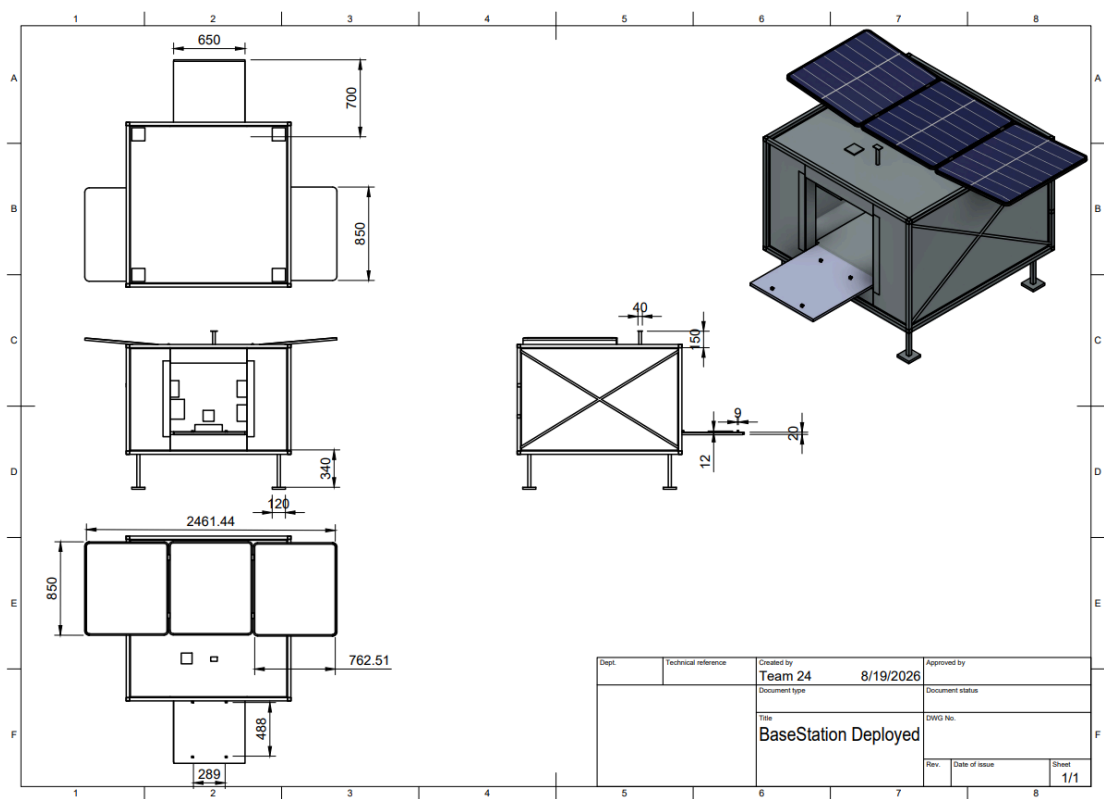
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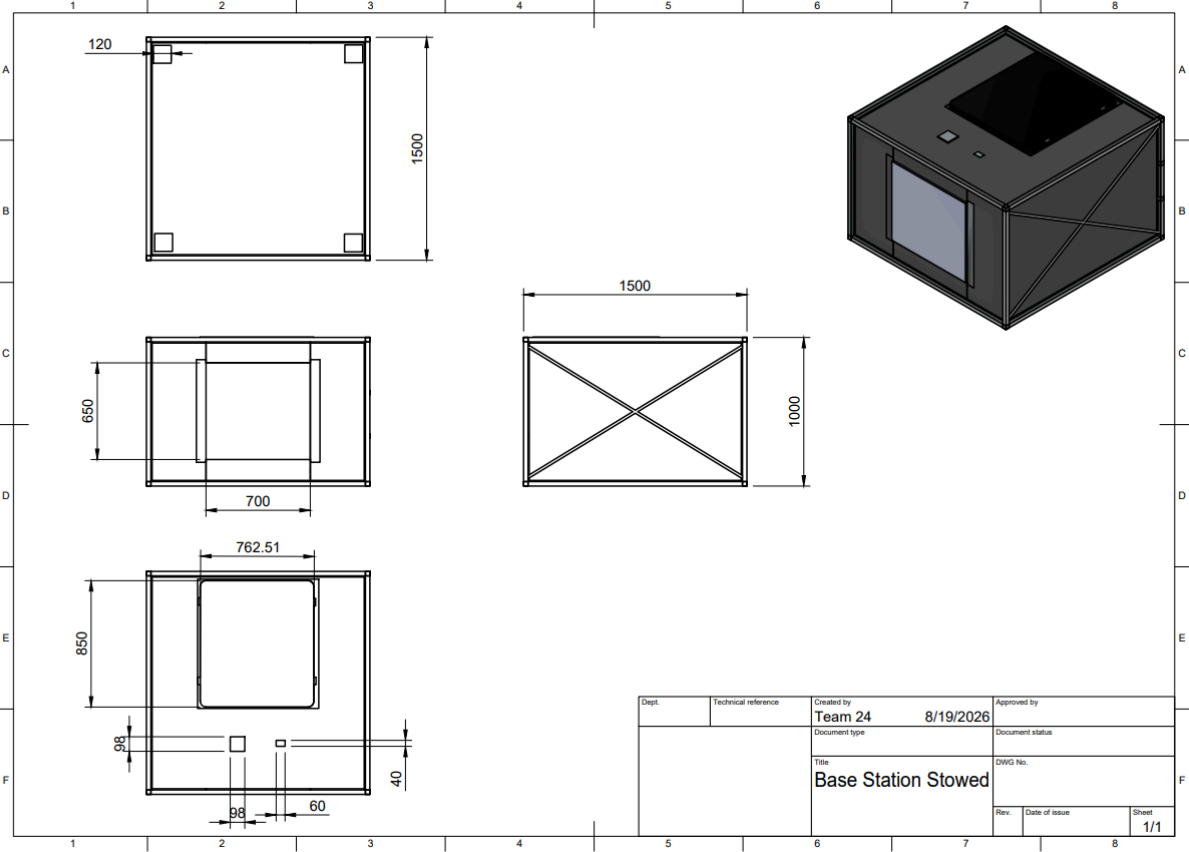
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The team has not used generative AI for any of the deliverables except for the MCR. During the preparation of the MCR, the team used ChatGPT to concisely summarize the requirements and rationale wording in section 1.4 (Mission Requirements). After using this tool, the team reviewed and edited the content as needed and took full responsibility for the content of the deliverable. The input used to generate the content was to "concisely edit the lengthy sentences provided in requirements and rationale of the table." The goal was to simply make the requirements and rationale portion of 1.4 more readable and unrepetitive to preserve proper grammar.

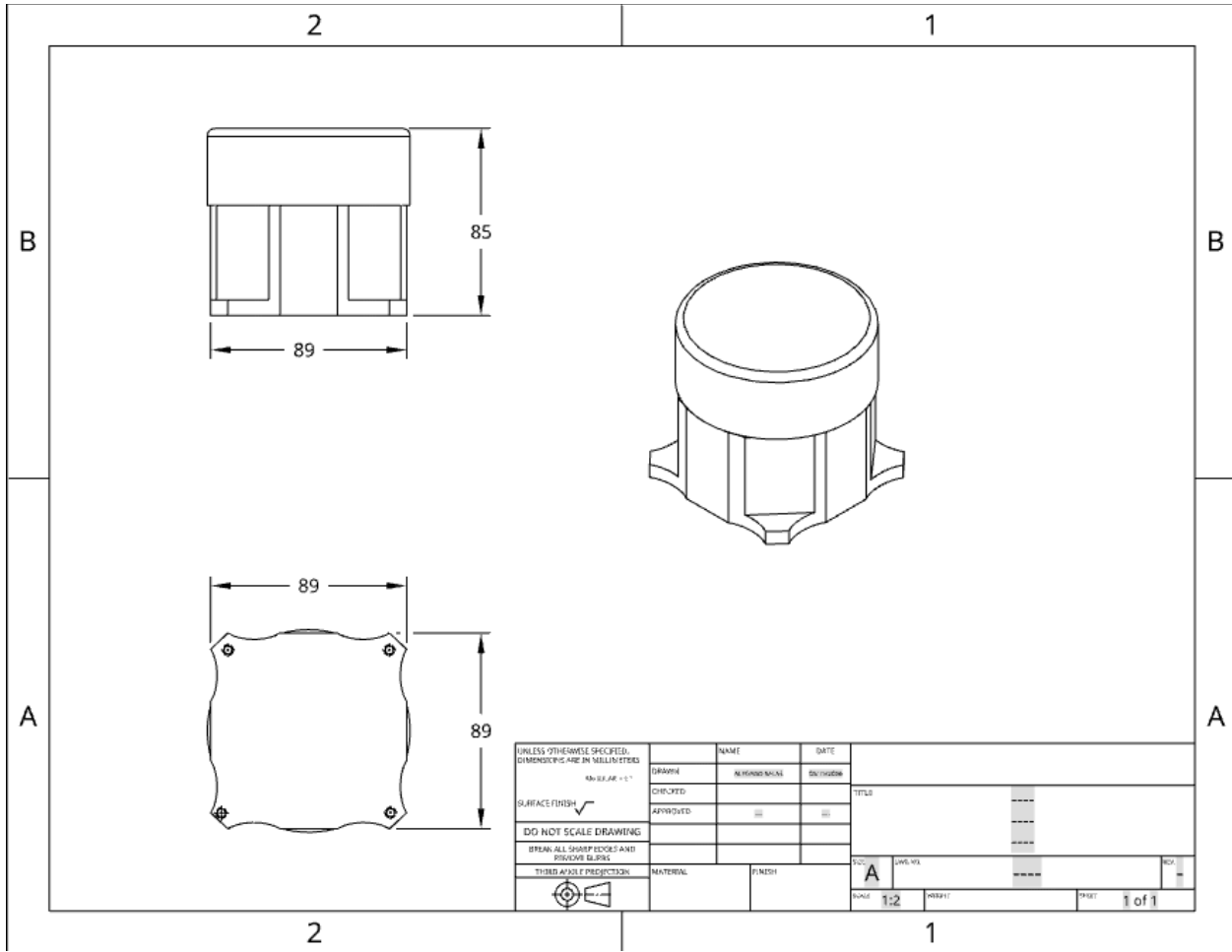
Appendix A: CAD Designs



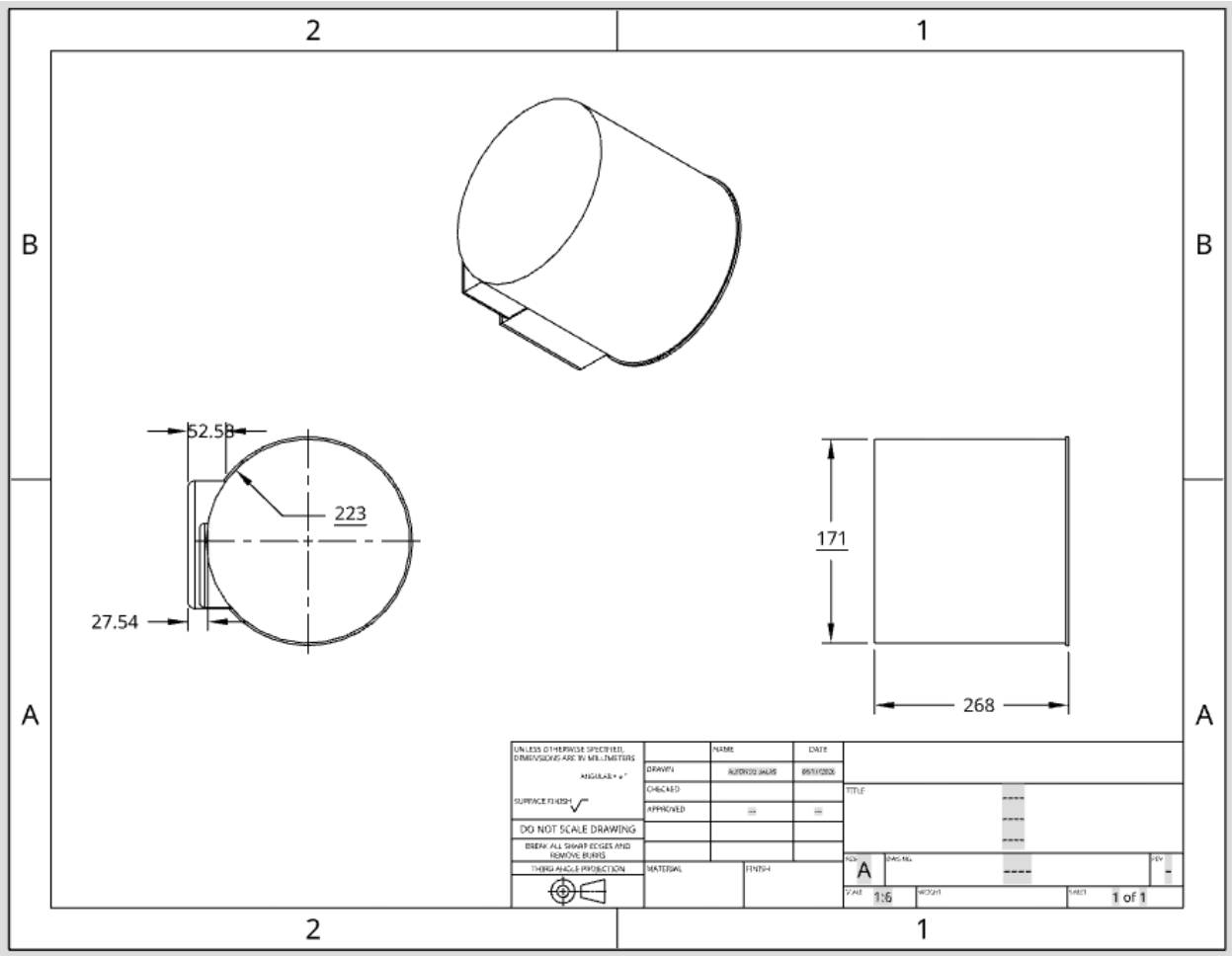
Appendix A.1.1 - Deployed Base Station CAD Drawing



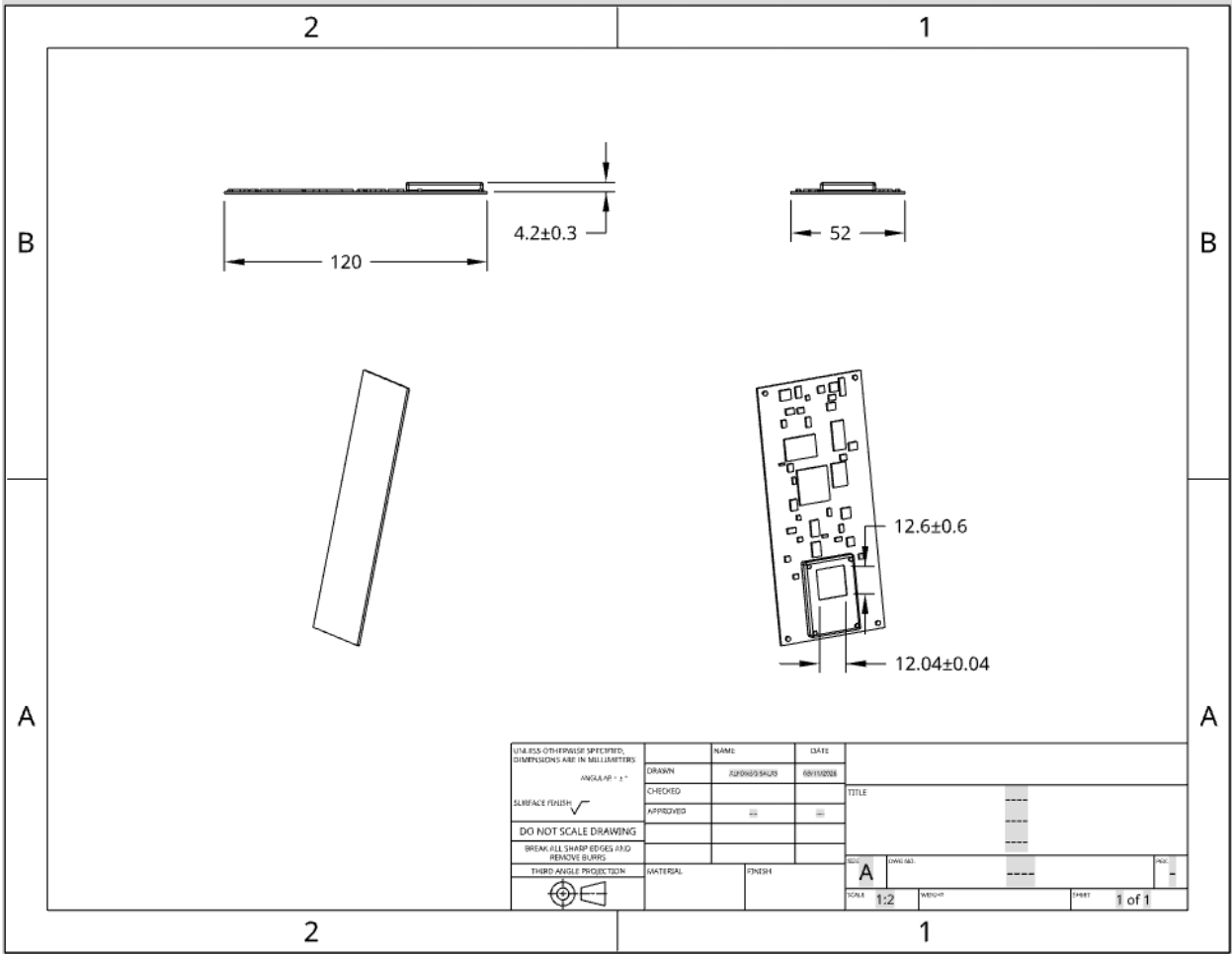
Appendix A.1.2 - Stowed Base Station CAD Drawing



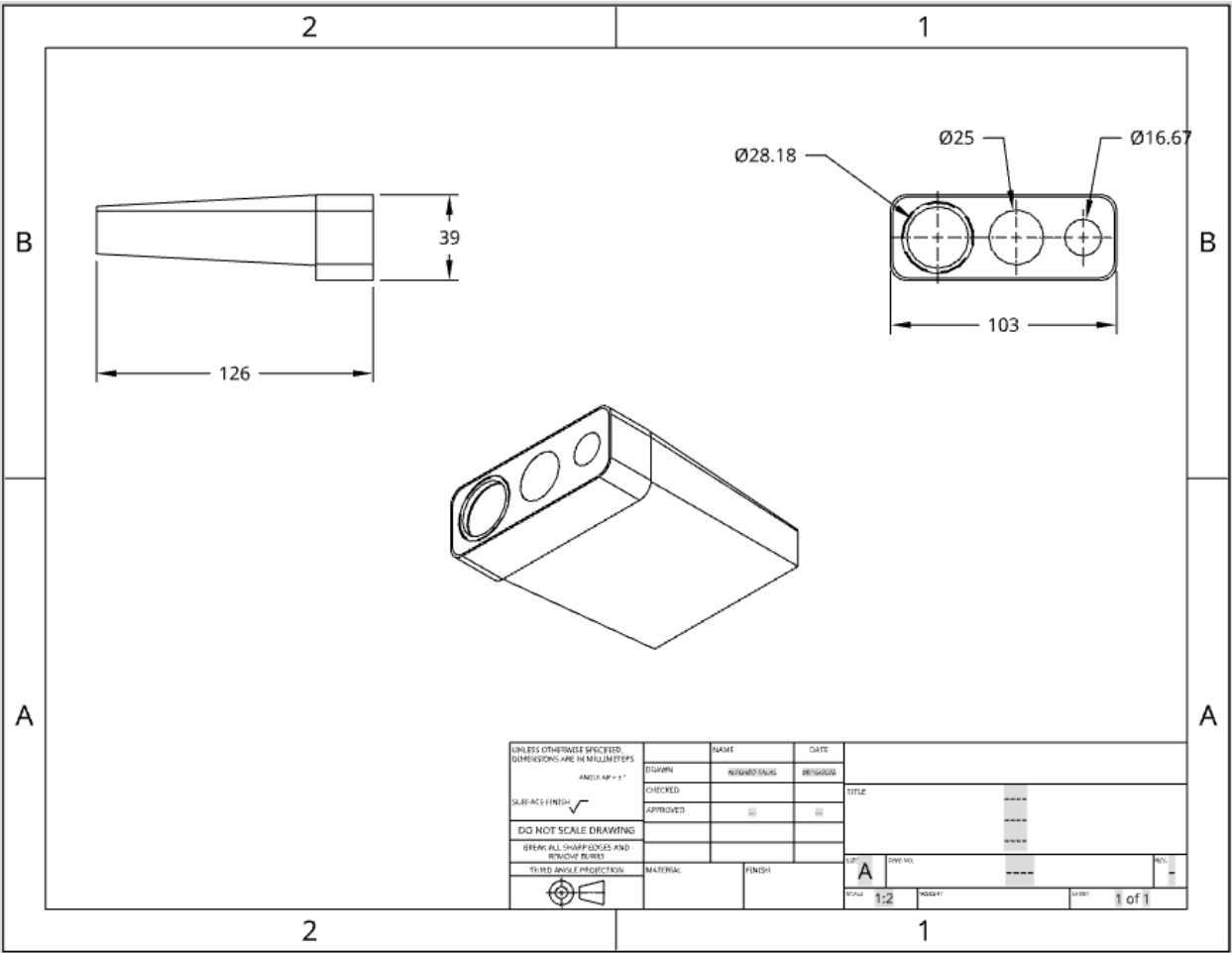
Appendix A.2.1 - LN200s CAD Drawing



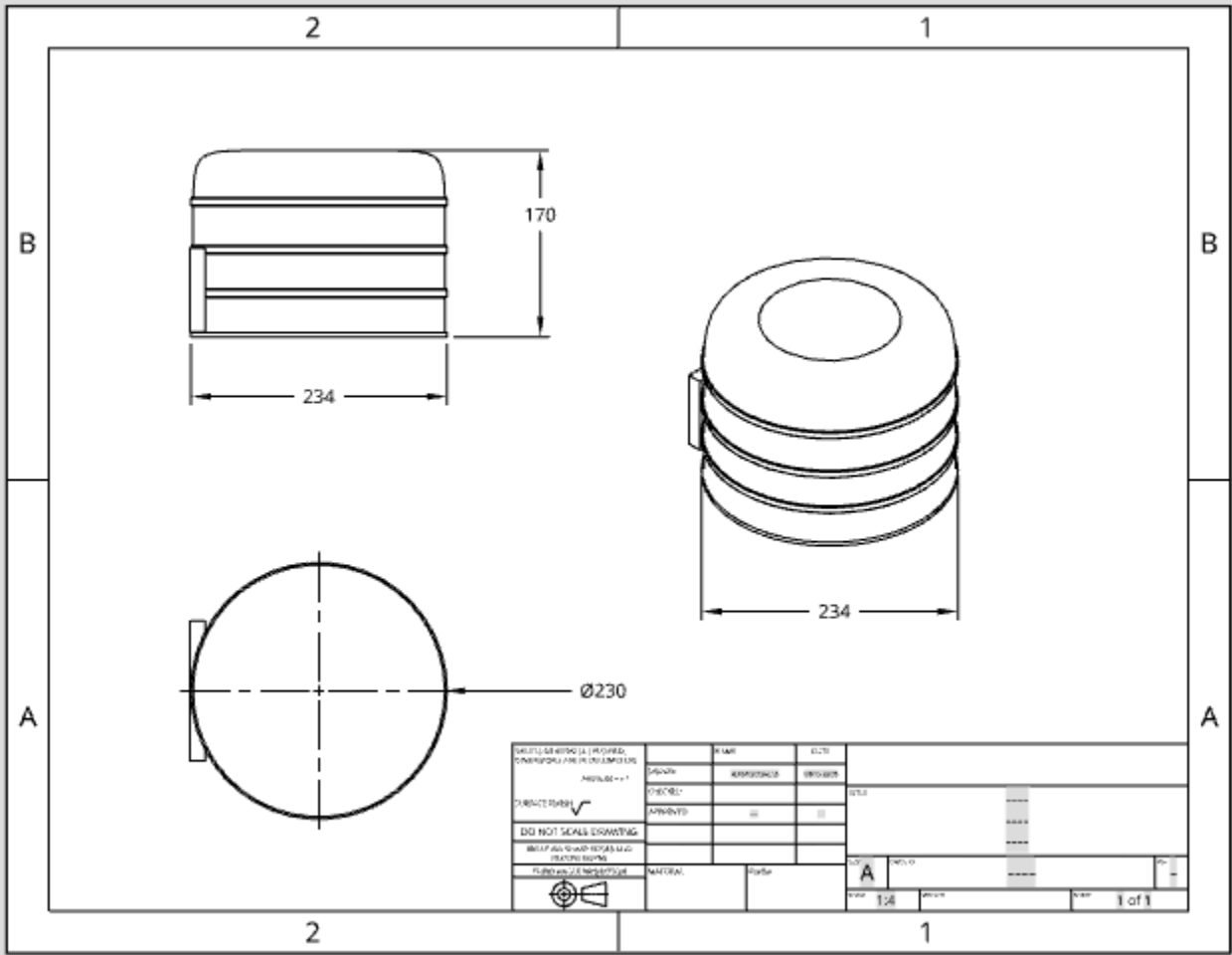
Appendix A.2.2 - SUDA Dust Mass Spectrometer CAD Drawing



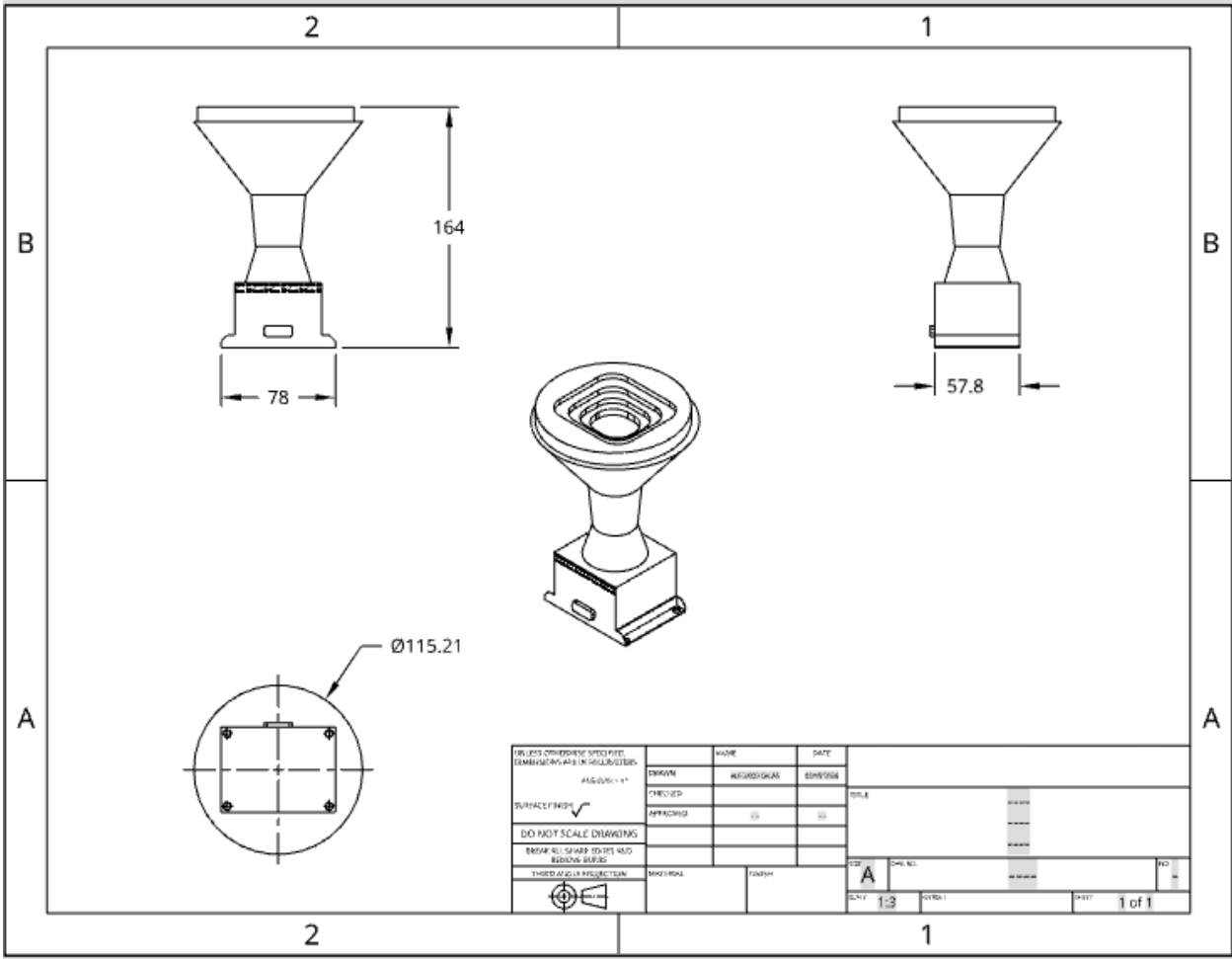
Appendix A.2.3 - LETS Spectrometer CAD Drawing



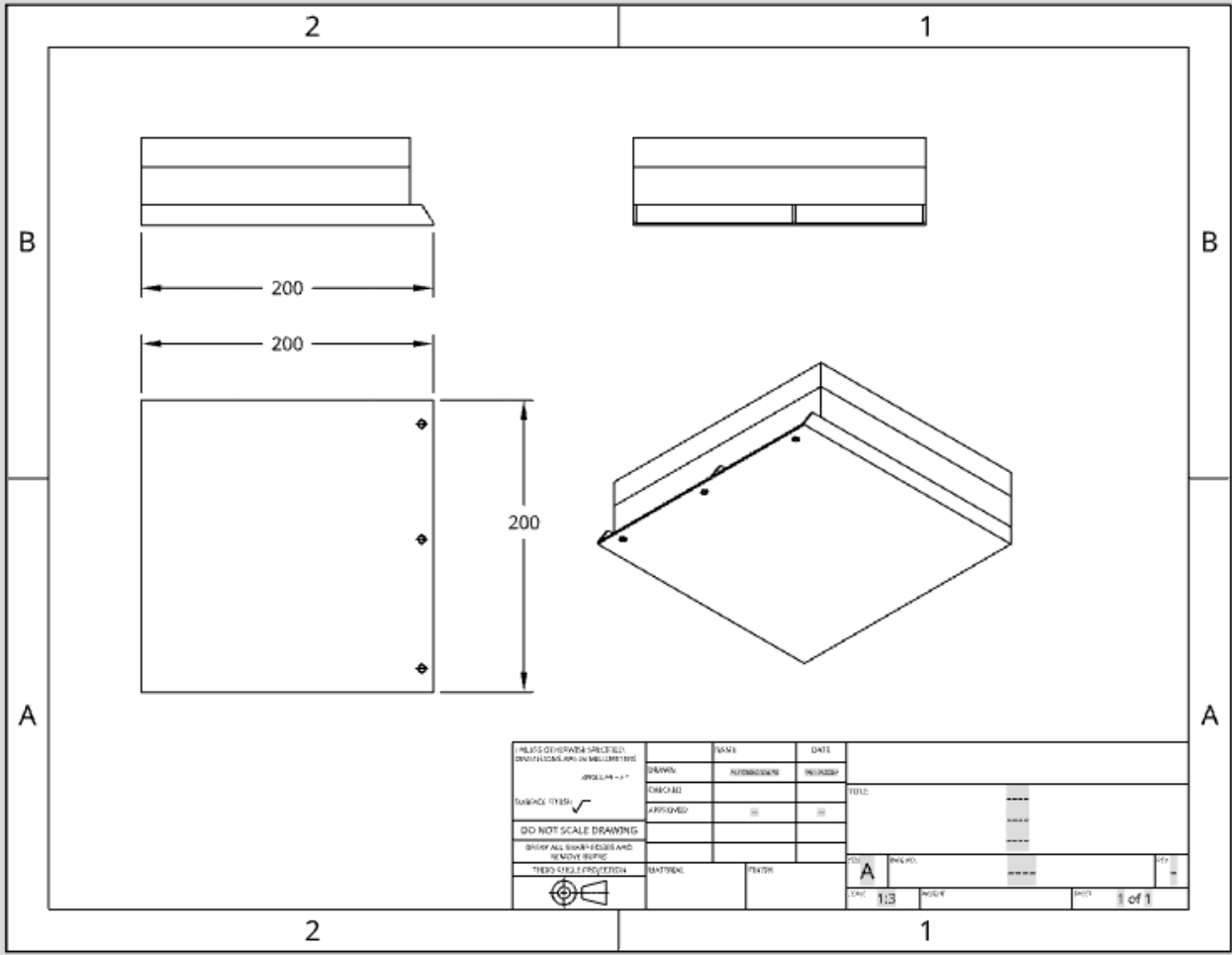
Appendix A.2.4 - RESOURCE Camera CAD Drawing



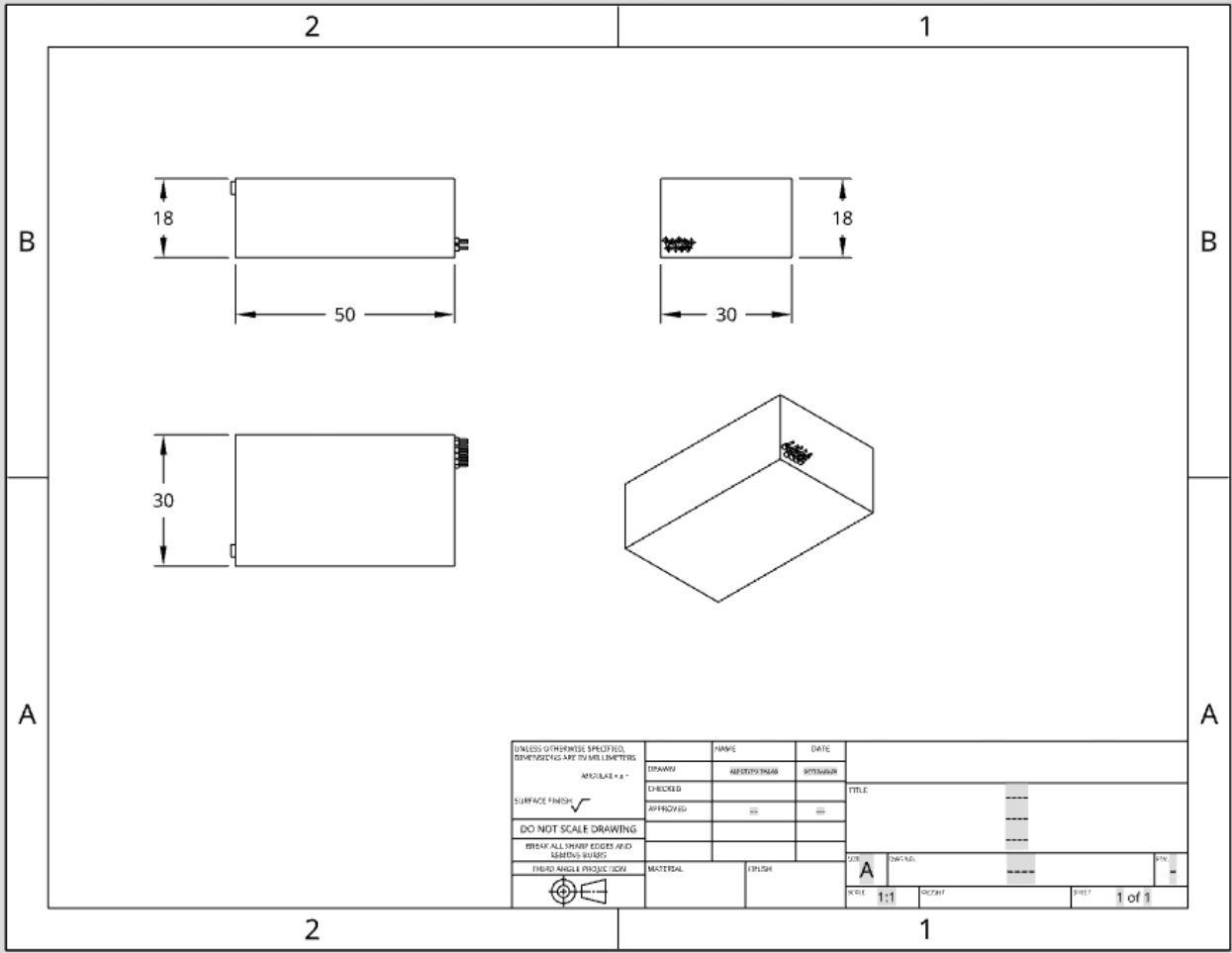
Appendix A.2.5 - Honeywell MIMU CAD Drawing



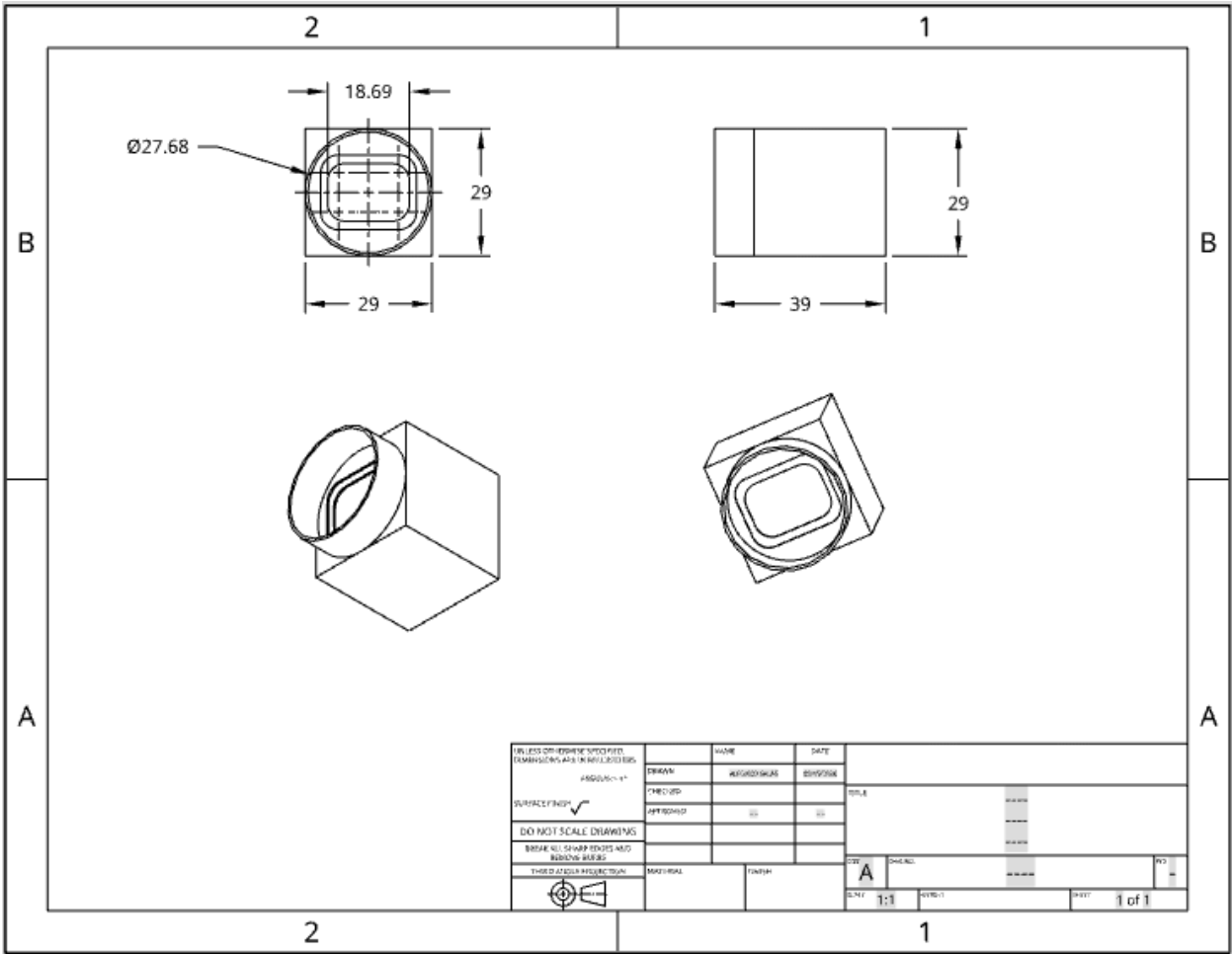
Appendix A.2.6 - TAGCAMS Camera CAD Drawing



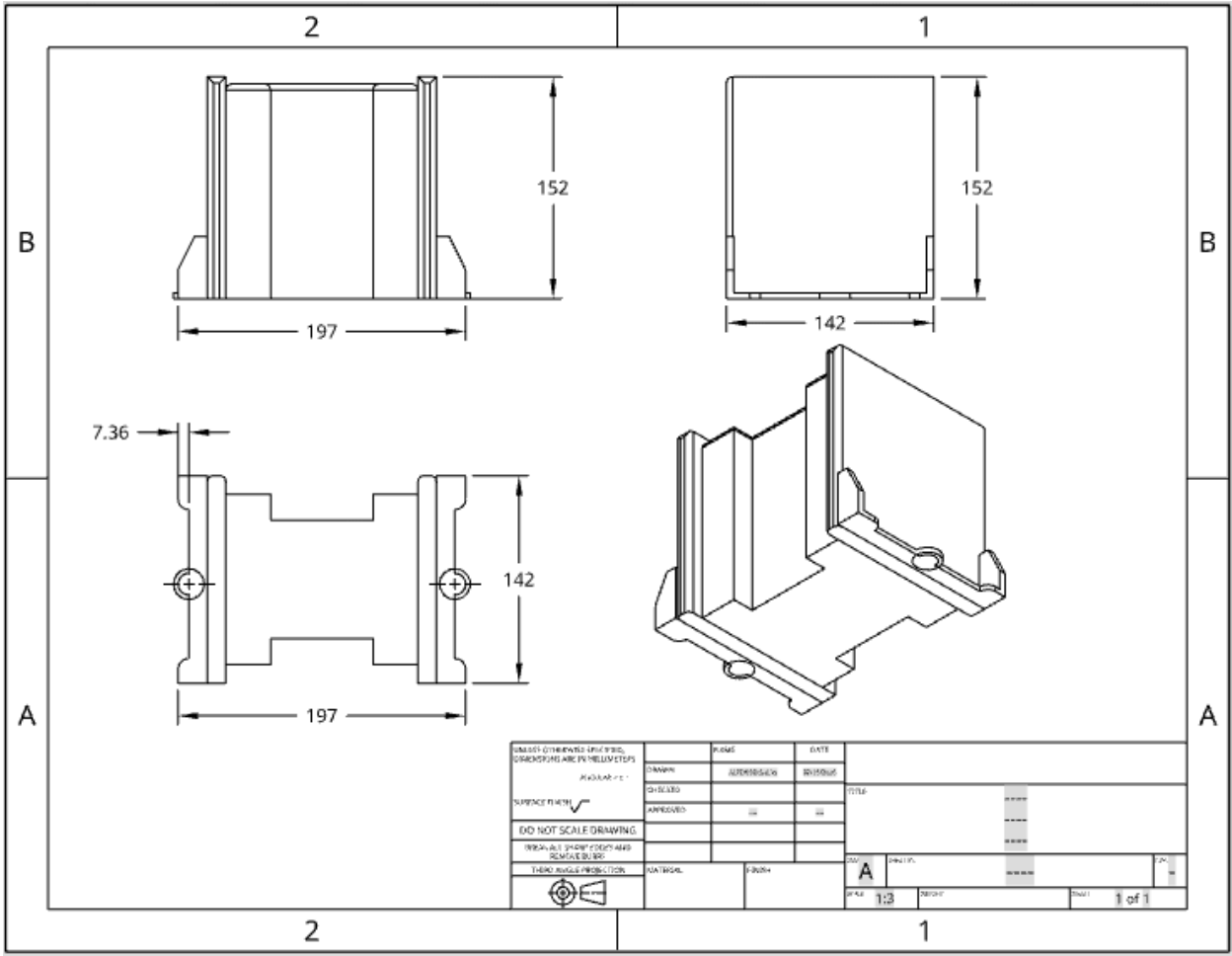
Appendix A.2.7 - TAGCAMS DVR CAD Drawing



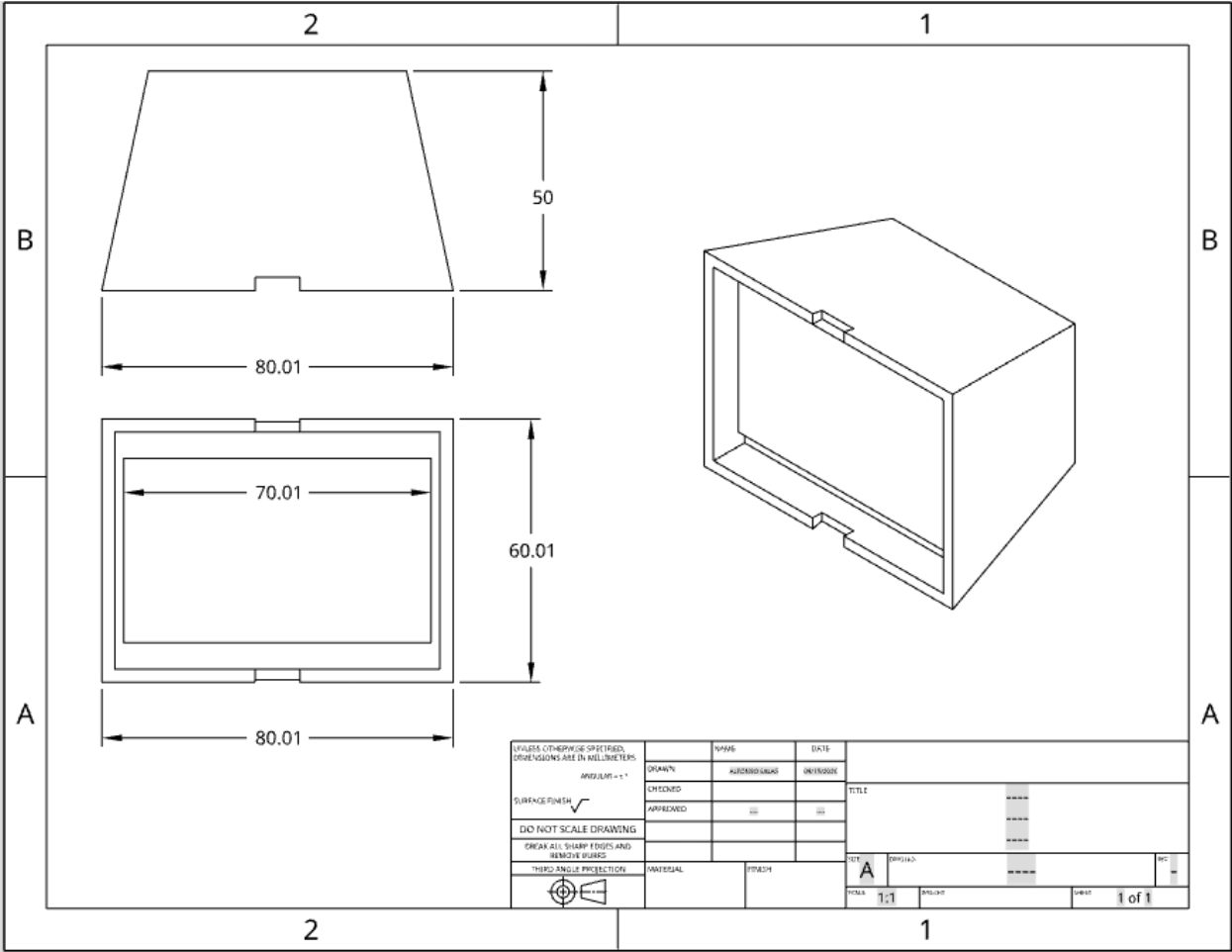
Appendix A.2.8 - SCALPSS Laser CAD Drawing



Appendix A.2.9 - SCALPSS Camera CAD Drawing

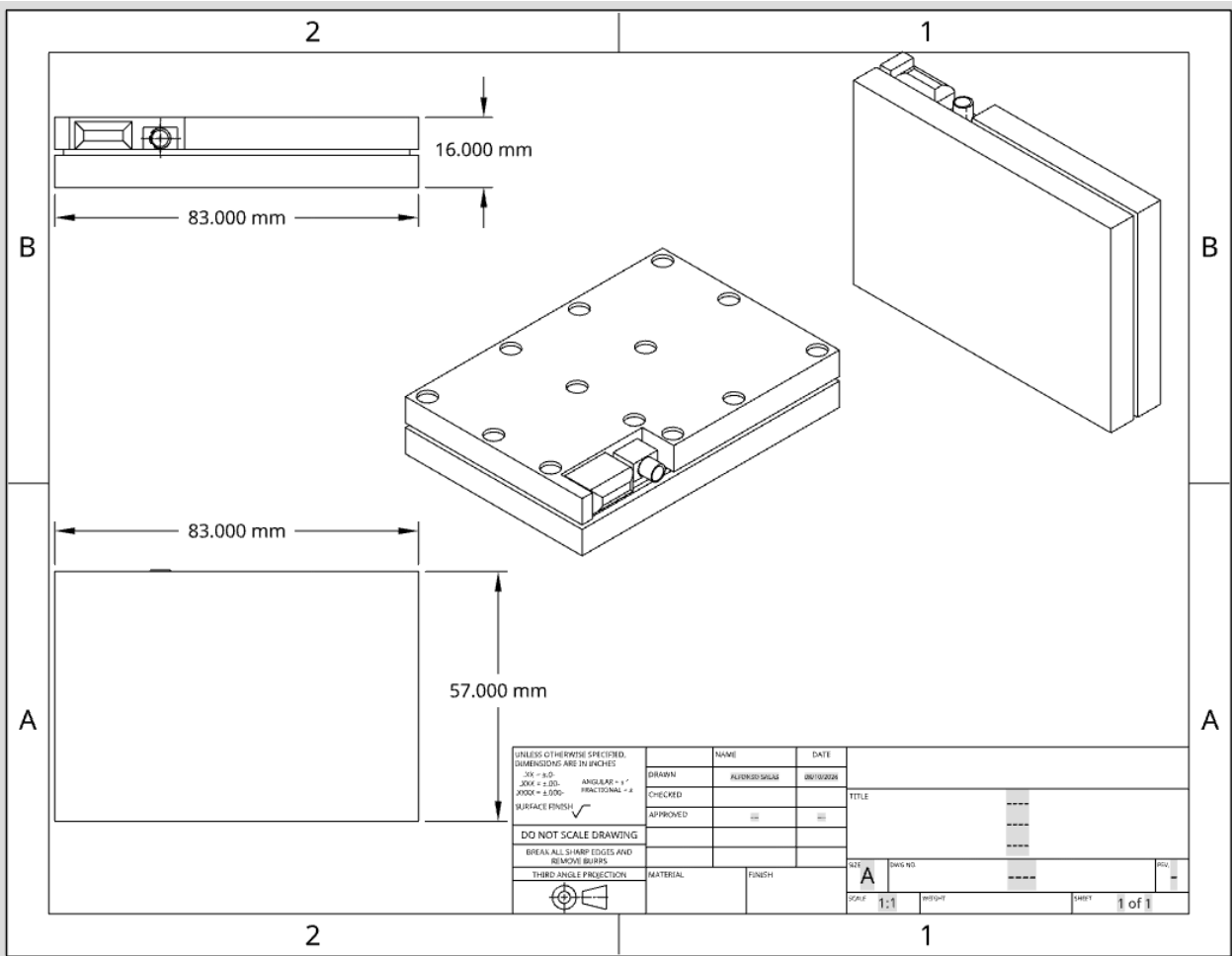


Appendix A.2.10 - SCALPSS MSE-Lite CAD Drawing

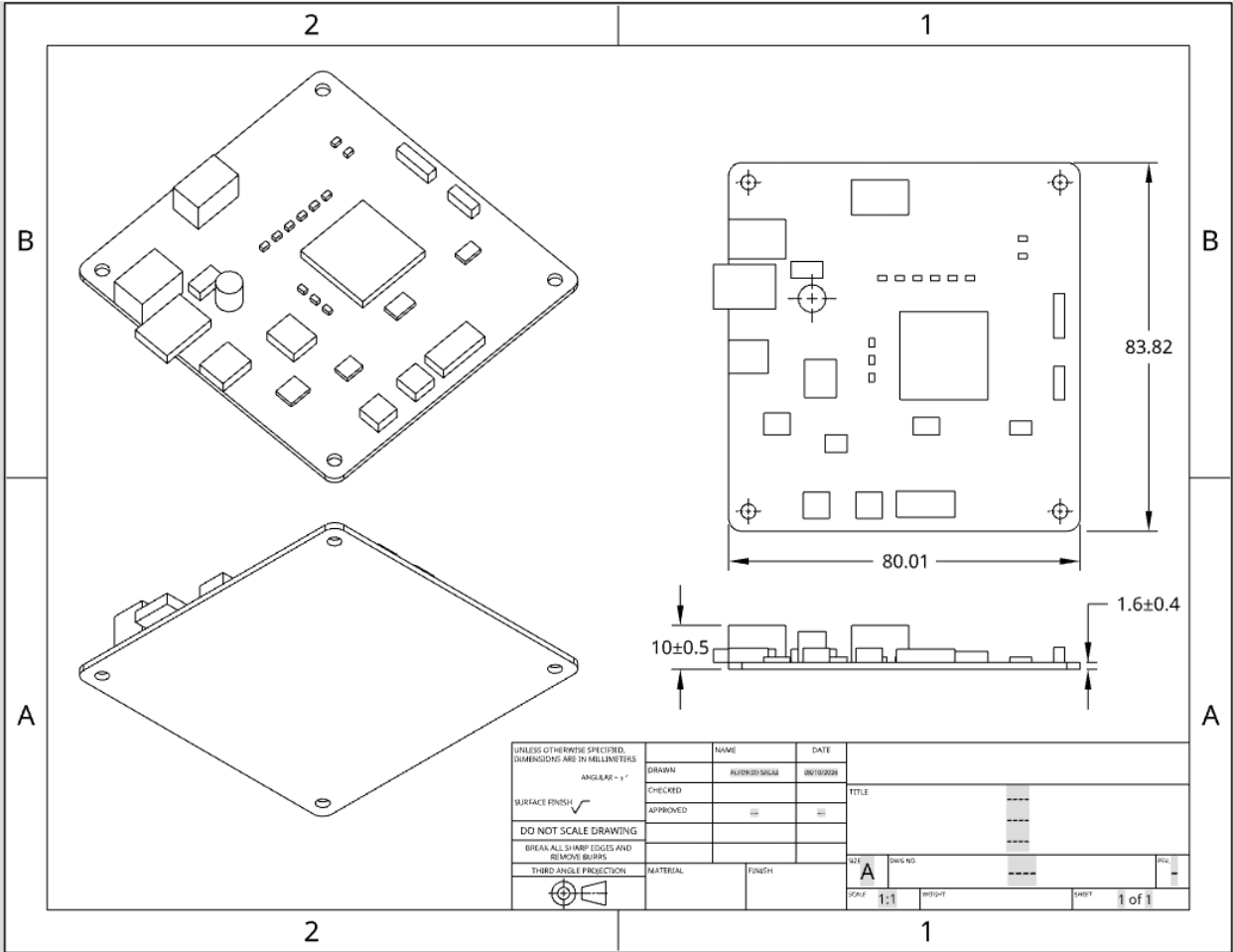


Technical drawing of a mechanical part, likely a bush or sleeve, showing three views: a top view, a front view, and a side view. The top view shows a circular cross-section with a diameter of 58.88 mm. The front view shows a cylindrical shape with a height of 200 mm and a diameter of 50 mm. The side view shows a cross-section with a width of 50 mm. The drawing is labeled "SCALE 1:2".

Appendix A.2.12 - SCALPSS Laser Optics CAD Drawing



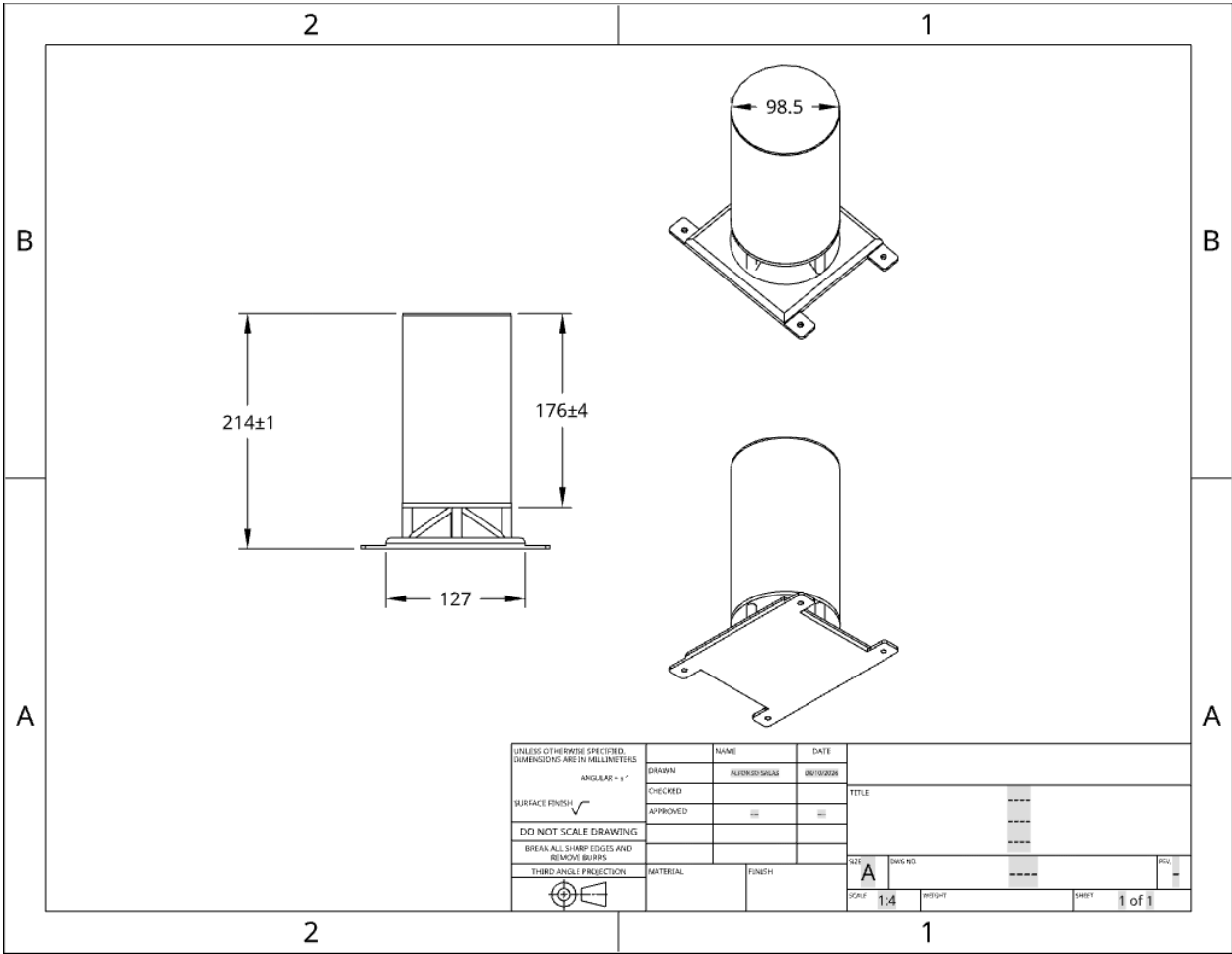
Appendix A.3.1 - TRX-U CAD Drawing



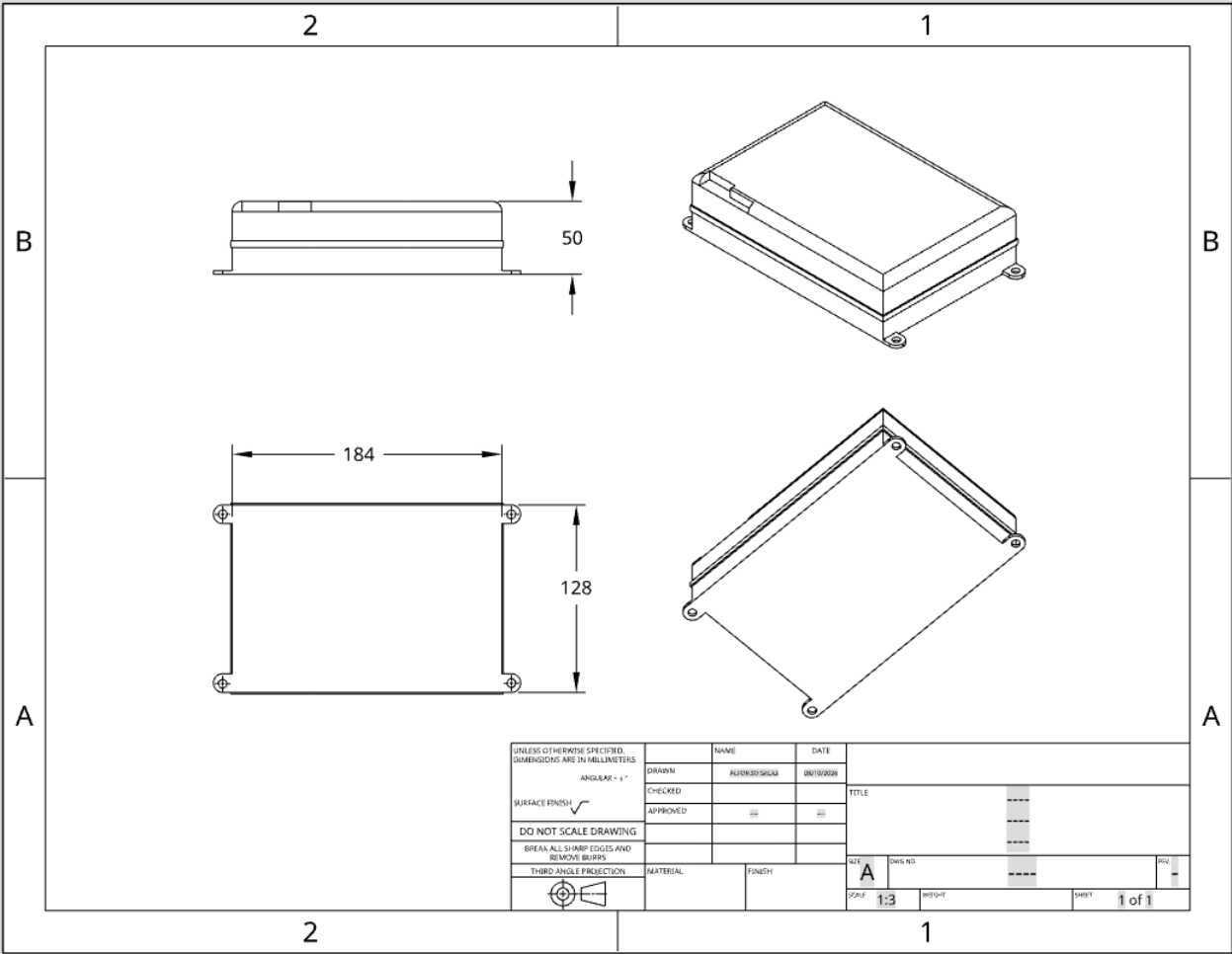
Technical drawing of a square plate with dimensions 100.000 mm x 100.000 mm. The drawing includes a top view, a side view, and a perspective view. The top view shows a central hole with a diameter of 2.261 mm and a distance of 47.800 mm from the center to the edge. The side view shows a flange with a width of 1.800 mm and a thickness of 5.50 mm. The perspective view shows the plate with a grid of 10x10 squares.

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Appendix A.3.3 - IQ Spacecom X-Band Antenna CAD Drawing



Appendix A.3.4 - Quadrifilar Helix Antenna CAD Drawing



Technical drawing of a rectangular box, showing three views: front, top, and isometric.

Dimensions:

- Height: 56
- Width: 101
- Depth: 10.32

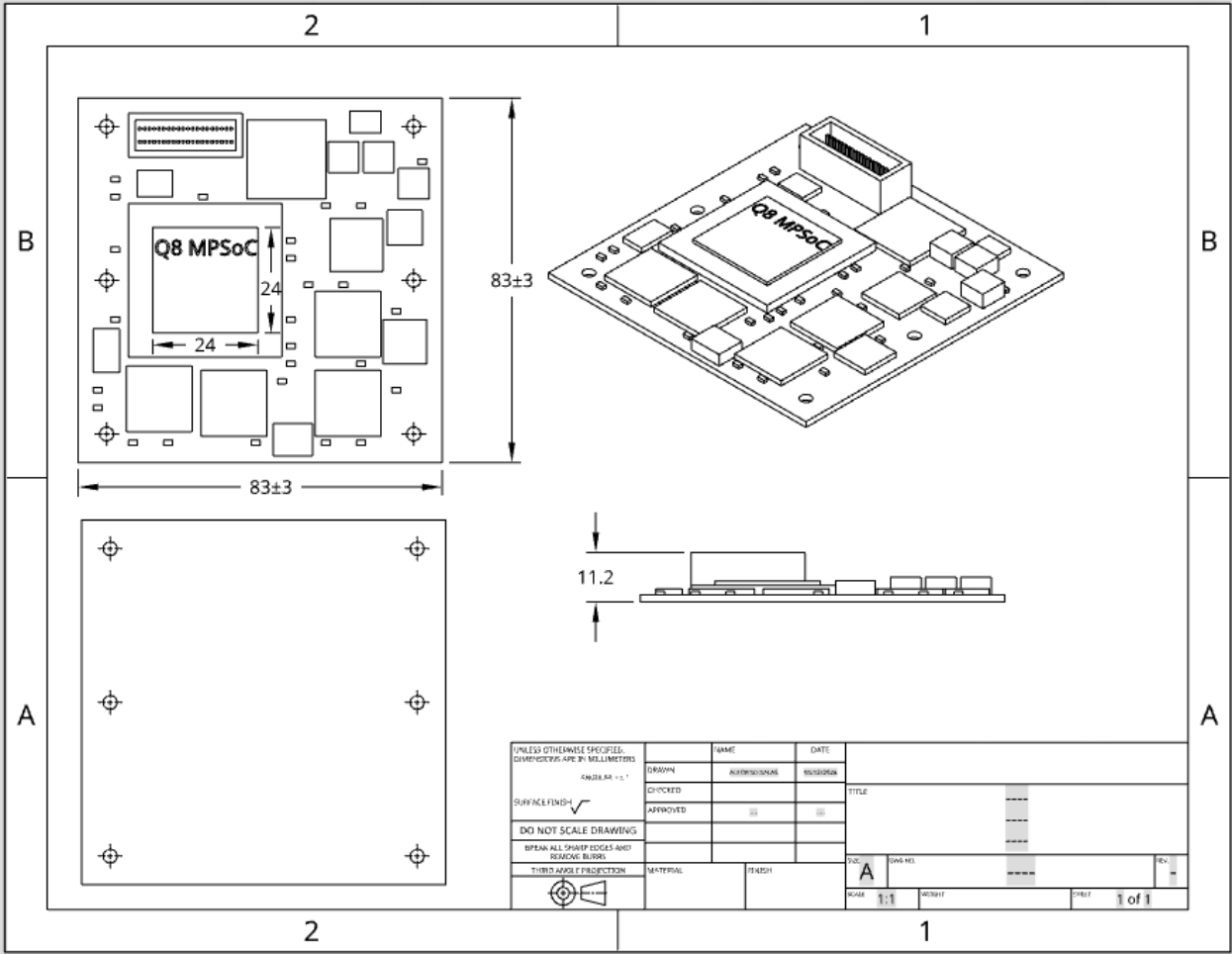
Assembly Details:

- Top view shows 8 screws (Ø5.77) around the perimeter.
- Front view shows 4 screws (Ø5.77) along the top edge.
- Isometric view shows the box with screws on the top and front edges.

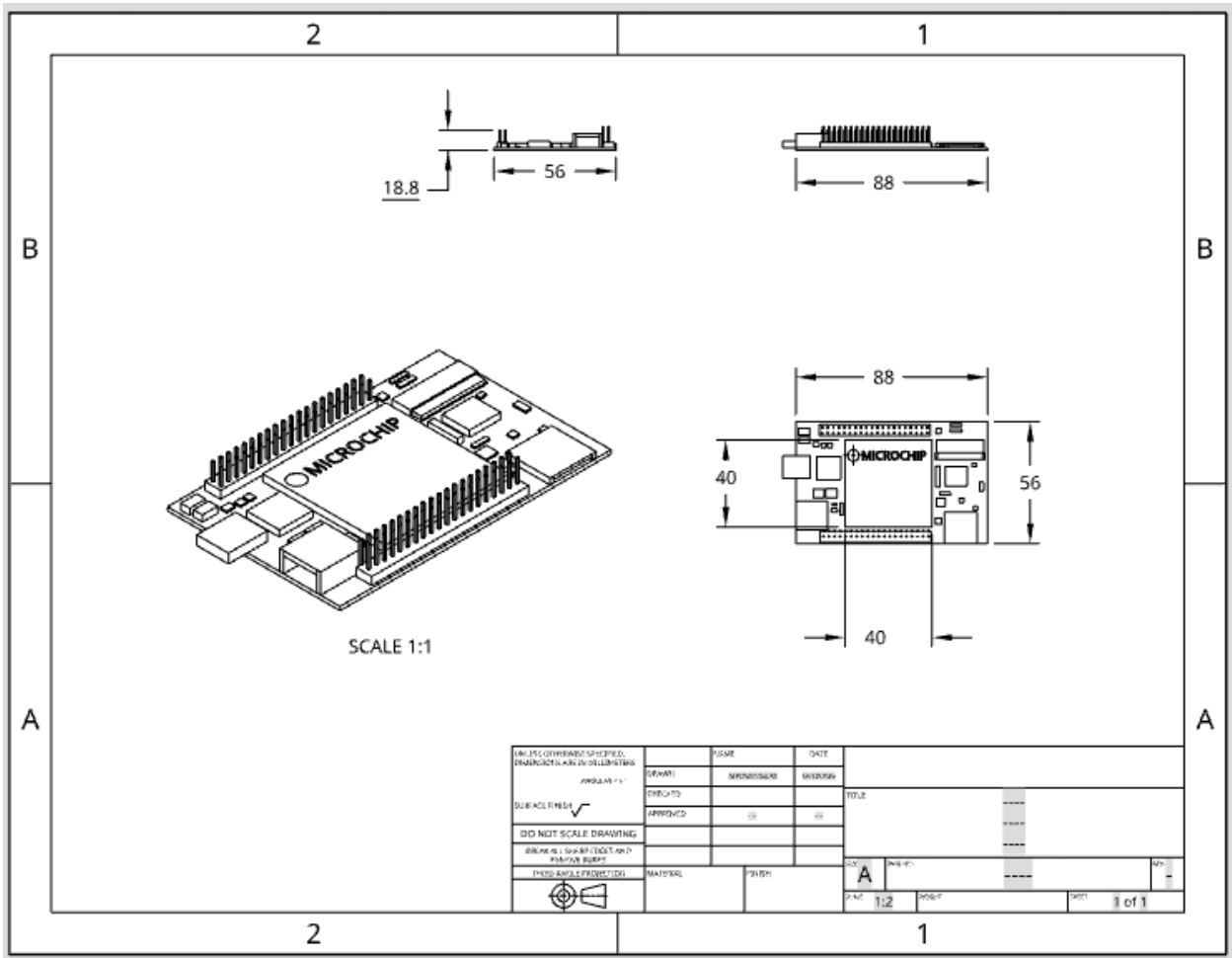
Table:

ITEM	QTY	UNIT	REMARKS
1	1	PC	BOX
2	1	PC	BOX
3	1	PC	BOX
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99	1	PC	BOX
100	1	PC	BOX

Appendix A.3.6 - Iris V2 CAD Drawing



Appendix A.3.6 - Xiphos Q8 OBC CAD Drawing



Appendix A.3.7 - PolarFire OBC CAD Drawing

Appendix B: Subsystem Trade Studies

Onboard Computer (OBC) - Drone							
Criteria	Explanation	Grade	Weight	BAE systems RAD750	PolarFire SoC	GR712RC	RAD5545
Processing Power	Determines the processor's ability to execute autonomous navigation, instrument control, and flight software at sufficient speeds to support drone operations.	10: >1GHz multi-core 1-9: Linear (200MHz-1GHz) 0: <200MHz (Fail)	20%	1	6	1	4
Radiation (TID)	Assess the computer's resistance to environmental hazards during PSR excursions	10: >100 krad TID 1-9: Linear (5k rad-100 krad) 0: <5k rad (Fail)	25%	10	10	10	10
TRL & Heritage	Indicated reliability and previous sufficient reliability to meet minimum TRL-6 threshold	10: TRL 9 6-9: TRL 7 to TRL 8 1-5: TRL 6 0: <TRL 6 (Fail)	20%	10	6	10	10
Mass Volumetrics	Evaluates combined physical parameters directly influencing mass and volume hard constraints	10: <200g (Best) 1-9: Linear (200g-1kg) 0: >1kg (Fail)	20%	5	10	10	0
Power Efficiency	Evaluates power draw of the processor as it directly influence available flight time and mission range on Drone	10: <2W (Best) 1-9: Linear (2W to 10W) 0: >10W (Fail)	15%	5	9	6	0
TOTALS:			100%	64.50%	82.50%	76.00%	53.00%

Appendix B.1.1 - OBC Drone Trade Study

Onboard Computer (OBC) - Base Station							
Criteria	Explanation	Grade	Weight	BAE Systems RAD750	Vorago VA41630	Q8 Zynq UltraScale +	GR-VPX-G R740 (GR740)
Processing Power	Determines the processors' ability to manage various tasks including DTE downlink, drone telemetry, and instrumentation without bottlenecks	10: >1GHz multi-core 1-9: Linear (200MHz-1GHz) 0: <200MHz (Fail)	30%	1	0	10	2
Radiation (TID)	Assesses computer's resistance to Total Ionizing Dose accumulation over full mission lifetime at South pole	10: >100krad TID 1-9: Linear (5krad-100krad) 0: <5krad (Fail)	20%	10	10	3	10
Memory Capacity	Determines whether the computer's onboard memory architecture can support multi tool data responses	10: >2GB 1-9: Linear (128MB-2GB) 0: <128MB (Fail)	30%	5	0	8	2
Mass Volumetrics	Evaluates combined physical parameters directly influencing mass and volume hard constraints	10: <200g (Best) 1-9: Linear (200g-1kg) 0: >1kg (Fail)	10%	5	10	10	10
Power Efficiency	Evaluates power draw of the processor as it directly influence available flight time and mission range on Drone	10: <2W (Best) 1-9: Linear (2W to 10W) 0: >10W (Fail)	10%	5	10	6	6
TOTALS:			100%	48%	40.00%	76.00%	48%

Appendix B1.2 - OBC Base Station Trade Study

Data Storage - Drone						
Criteria	Explanation	Grade	Weight	Micro Secure Digital (SD) Cards	NAND flash-based Solid State Drives (SSDs)	Space-Graded NVMe SSDs
Data Writing Speed	The Drone storage must write incoming science instrumentation and data sensor streams at an adequate pace during missions	10: >2000 MB/s 1-9: Linear (100-2000 MB/s) 0: <100 MB/s (Fail)	25%	1	4	8
Radiation (TID)	Storage hardware must resist radiation induced environments to prevent bit flips and data corruption during PSR missions	10: >100krad TID 1-9: Linear (5krad-100krad) 0: <5krad (Fail)	30%	1	8	10
Storage Capacity	Must provide adequate storage to retain all science and PNT data collected during PSR missions	10: >256GB 1-9: Linear (16GB-256GB) 0: <16GB (Fail)	10%	8	10	10
Physical Footprint	Measures combined impact of component mass and volume metrics to ensure hard constraints are met	10 = <50g, <5cm ³ 5 = 50–150g, 5–15cm ³ 1 = 150–300g, 15–30cm ³ 0 = >300g, Fail	20%	10	5	5
TRL & Flight Heritage	Indicates reliability and compliance with the L'SPACE minimum TRL 6 mandate	10: TRL 9 (>5 missions) 6-9: TRL 7 to TRL 8 1-5: TRL 6 0: <TRL 6 (Fail)	15%	10	10	6
		TOTALS:	100%	48.50%	69.00%	79%

Appendix B1.3 - Data Storage Drone Trade Study

Data Storage - Base Station						
Criteria	Explanation	Grade	Weight	Rad-Hard 3D NAND Flash SSRs	Space-Graded NVMe SSD Arrays	MRAM + Rad-Tolerant Flash hybrid Array
Data Writing Speed	The base station storage must write incoming instrument data without pipeline bottlenecking during data transfer periods	10: >2000 MB/s 1-9: Linear (100-2000 MB/s) 0: <100 MB/s (Fail)	25%	4	8	5
Radiation (TID)	Storage hardware must resist TID degradation over full mission lifetime to protect mission data archive	10: >100krad TID 1-9: Linear (5krad-100krad) 0: <5krad (Fail)	25%	8	10	9
Storage Capacity	Must provide sufficient capacity to archive all BS instrument data relayed from drone during full mission duration	10: >1TB 1-9: Linear (64GB-1TB) 0: <64GB (Fail)	30%	9	10	6
Physical Footprint	Measures combined impact of component mass and volume metrics to ensure hard constraints are met	10 = <200g, <20cm ³ 5 = 200–500g, 20–50cm ³ 1 = 500g–1kg, 50–100cm ³ 0 = >1kg, Fail	10%	9	5	8
TRL & Flight Heritage	Data storage must meet minimum TRL-6 and demonstrate deep space/lunar heritage for reliability	10: TRL 9 (>5 missions) 6-9: TRL 7 to TRL 8 1-5: TRL 6 0: <TRL 6 (Fail)	10%	10	5	8
		TOTALS:	100%	76%	85.00%	69.00%

Appendix B1.4 - Data Storage Base Station Trade Study

Data Interfaces (Internal)						
Criteria	Explanation	Grade	Weight	SpaceWire (ECSS-E-ST-50-52C)	RS-422	CAN-FD
Data Throughput	Determines sufficient data transfer rates to handle multitasking without creating pipeline bottlenecks	10: >100 Mbps 1-9: Linear (1 Mbps-100 Mbps) 0: <1 Mbps (Fail)	30%	10	1	4
Hardware Complexity	Physical complexity must remain simple to integrate and scope with mass and volume constraints of lunar drone and base station	10: Simple implementation 6-9: Moderate, minor overhead 1-5: Complex, bulky layout 0: Prohibitive (Fail)	25%	7	10	8
TRL & Heritage	Selected interface must meet minimum TRL-6 threshold and demonstrate spaceflight heritage	10: TRL 9 (>5 missions) 6-9: TRL 7 to TRL 8 1-5: TRL 6 0: <TRL 6 (Fail)	20%	10	10	7
Environment Reliability	Protocol must maintain signal integrity when exposed to lunar south pole conditions (radiation, thermal cycling)	10: >100krad, -230°C 8-9: 20-100krad, -150°C 5-7: 5-20krad, -80°C 2-4: <5krad, unshielded 0: lack of protection (Fail)	25%	9	6	5
TOTALS:			100%	90.00%	63.00%	58.50%

Appendix B1.5 - Data Interfaces Trade Study

Direct-To-Earth Communication (BS)						
Criteria	Explanation	Grade	Weight	S Band Architecture	X-Band Architecture	Ka-Band Architecture
Data Rate	Indicates communication system reliability for transmitting sufficient science and telemetry data volumes	10: >100 Mbps 1-9: Linear (1 Mbps-100 Mbps) 0: <1 Mbps (Fail)	30%	2	5	10
Hardware Complexity	Antenna and transceiver system must remain simple enough to integrate within base stations structural volume and mass allocation	10: <500g, no pointing 6-9: 500g-2kg, minimal align 1-5: 2-5kg, moderate align 0: >5kg, complex point (Fail)	10%	10	6	3
Link Closure	Must demonstrate sufficient transmit power, antenna gain, and link margin to maintain closed communication link at distance from Lunar South Pole to Earth	10: >6dB margin 1-9: Linear (0dB-6dB margin) 0: No link closure (Fail)	30%	6	8	5
Environment Reliability	DTE system must maintain signal integrity and transmission capability when exposed to external conditions (radiation, thermal cycling)	10: >100krad, -230°C 8-9: 20-100krad, -150°C 5-7: 5-20krad, -80°C 2-4: <5krad, unshielded 0: System exposure (Fail)	20%	9	9	7
TRL & Heritage	Communication hardware must meet minimum TRL-6 and demonstrate deep space/lunar heritage for reliability	10: TRL 9 (>5 missions) 6-9: TRL 7 to TRL 8 1-5: TRL 6 0: <TRL 6 (Fail)	10%	10	10	7
TOTALS:			100%	62.00%	73.00%	69.00%

Appendix B1.6 - Direct-to-Earth Base Station Trade Study

Local Link - Shared						
Criteria	Explanation	Grade	Weight	UHF Space Transceiver	S-Band SDR Transceiver	Space-Ruggedized Wireless LAN (Wi-Fi)
Data Rate	Must transfer sufficient data and telemetry between drone and base station within offload windows to prevent malfunctions	10: >100 Mbps 1-9: Linear (1 Mbps-100 Mbps) 0: <1 Mbps (Fail)	15%	1	5	8
Hardware complexity	Must remain lightweight and compact enough to integrate within drone and base station without overengineering	10: <50g 6-9: 50g-200g 1-5: 200g-500g 0: >500g (Fail)	15%	9	5	7
TRL & Heritage	Communication hardware must meet minimum TRL-6 and demonstrate deep space/lunar heritage for reliability	10: TRL 9 (>5 missions) 6-9: TRL 7 to TRL 8 1-5: TRL 6 0: <TRL 6 (Fail)	10%	10	10	1
Antenna Directionality	Local link must maintain strong connection between systems despite natural landforms, drone maneuverability, and long range distances	10: Omnidirectional, no pointing needed 6-9: Wide beam, $\pm 45^\circ$ tolerance 1-5: Semi-directional, active orientation needed 0: Linear, gimbal required (fail)	25%	10	4	5
Environment Reliability	Local link must maintain sufficient strength and data integrity for communication within PSR excursions	10: >100krad, -230°C 8-9: 20-100krad, -150°C 5-7: 5-20krad, -80°C 2-4: <5krad, unshielded 0: Link failure (Fail)	25%	7	8	2
Power Consumption	Important for minimizing power consumption to increase flight time, excursion range and overall mission lifespan	10: <1W 6-9: 1W-5W 1-5: 5W-10W 0: >10W (Fail)	10%	8	7	7
TOTALS:			100%	72.50%	66.65%	51.10%

Appendix B1.7 - Local Link Trade Study

SSPA						
Criteria	Explanation	Grade	Weight	SSPA 8.4-8.6-30	General Dynamics X-Band SSPA	JHU/APL Frontier SSPA
Output Power / Link Margin	RF output power available for the DTE link budget	5: ≥ 25 W output power 1-4: Linear (10 W-25 W) 0: <10 W (Fail)	10%	5	3	1
TRL & Heritage	Flight heritage relevance to this mission's operating environment	5: Direct lunar surface flight heritage (TRL 9) 1-4: Linear (TRL 6-8, general deep-space/near-Earth heritage) 0: No documented flight heritage (Fail, below TRL 6)	20%	5	4	2
Radiation Tolerance	Qualified Total Ionizing Dose rating	5: ≥ 100 krad(Si) TID 1-4: Linear (15 krad-100 krad(Si)) 0: <15 krad(Si) or unpublished (Fail)	10%	0	5	1
Power Efficiency	Composite PAE / DC-to-RF conversion efficiency	5: $\geq 30\%$ composite PAE / DC-to-RF efficiency 1-4: Linear (20%-30%) 0: <20% (Fail)	15%	4	4	5
Mass Volumetrics	Evaluates combined physical parameters directly influencing mass and volume hard constraints	5: <0.5 kg 1-4: Linear (0.5 kg-1.5 kg) 0: >1.5 kg (Fail)	20%	5	2	3
Power Consumption	DC input draw against the platform's 100 W peak / 45 W average ceiling	5: <40 W DC input draw 1-4: Linear (40 W-100 W) 0: >100 W (Fail)	25%	0	3	5
TOTALS:			100%	55.00%	67.00%	73.00%

Appendix B1.8 - SSPA Revised (MDR-ADV-003) Trade Study

Subsurface Sensing & Navigation						
Criteria	Explanation	Grade	Weight	IMUs (Inertial Measurements Units)	High-Frequency Ground Penetrating Radar	Atomic Clock
Power Consumption	Evaluates active electrical power required. Passive optical sensors must minimize draw to save drone battery.	10: < 2W nominal power 6-9: 2W - 5W nominal power 1-5: 5W - 10W nominal power 0: > 10W (Fails)	20%	9	3	6
Physical Footprint	Assesses mass and volume. Sensor must remain lightweight to maintain drone thrust-to-weight ratio.	10: < 0.2 kg + highly compact 6-9: 0.2 kg - 1.0 kg 1-5: 1.0 kg - 3.0 kg 0: > 3.0 kg (Fails)	15%	10	4	5
Data Processing	Rates raw data amount the sensor outputs. High data volume requires heavy processing, which could overwhelm the drone's onboard computer	10: Internal pre-processing + Low burden 6-9: Moderate data volume + standard interface 1-5: High Raw Data Volume 0: Incompatible	15%	9	3	10
Navigation accuracy & Precision	Ability to precisely locate drone location during PSR excursions in case of mission hurdles	10: < 10 cm positional error (High precision) 6-9: 10 cm - 1.0 m error 1-5: 1.0 m - 5.0 m error 0: > 5.0 m error (Fails)	25%	8	7	9
TOTALS:			75%	88.67%	45.33%	76.00%

Appendix B2.1 - Subsurface Sensing & Navigation

Dust Diagnostic						
Criteria	Explanation	Grade	Weight	EDA (Electrostatic Dust Analyzer)	Microscope Imager	Regolith Characterization Spectrometer
Power Consumption	Evaluates active and standby electrical power. Low power is critical to avoid draining the drone's battery.	10: < 1W Max power (Passive) 6-9: 1W - 3W Max power 1-5: 3W - 7W Max power 0: > 7W (Fails)	20%	5	5	7
Physical Footprint	Assesses mass and volume. The instrument must be extremely lightweight to maintain drone flight capability.	10: < 0.2 kg + highly compact 6-9: 0.2 kg - 0.5 kg 1-5: 0.5 kg - 1.5 kg 0: > 1.5 kg (Fails)	15%	2	8	7
STM Measurement	Evaluates ability to directly measure scientific required parameters: electrostatic charge or surface texture	10: Can directly and simultaneously measure scientific parameters 6-9: High accuracy and resolution 1-5: little to no ability to directly measure scientific parameters	30%	9	7	8
Data Processing	Rates raw data amount the sensor outputs. High data volume requires heavy processing, which could overwhelm the drone's onboard computer	10: Internal pre-processing + Low burden 6-9: Moderate data volume + standard interface 1-5: High Raw Data Volume 0: Incompatible	20%	7	7	7
TRL & Heritage	Evaluates flight heritage and reliability. Payload must meet the mission's minimum TRL-6 requirement.	10: TRL 9 (>5 missions) 6-9: TRL 7 to TRL 8 1-5: TRL 6 0: < TRL 6 (Fail)	15%	5	5	5
TOTALS:			100%	61.50%	64.50%	70.00%

Appendix B2.2 - Dust Diagnostic Trade Study

Illumination						
Criteria	Explanation	Grade	Weight	Stereoscopic Camera	SWIR Spectrometer	Hyperspectral Camera
Power Consumption	Evaluates active electrical power required. Passive optical sensors must minimize draw to save drone battery.	10: < 3W Max power 6-9: 3W - 8W Max power 1-5: 8W - 15W Max power 0: > 15W (Fails)	20%	8	7	4
Physical Footprint	Assesses mass and volume. Sensor must remain lightweight to maintain drone thrust-to-weight ratio.	10: < 0.5 kg + highly compact 6-9: 0.5 kg - 1.5 kg 1-5: 1.5 kg - 3.0 kg 0: > 3.0 kg (Fails)	15%	9	7	5
Environmental Resilience	Ability to survive radiation (TID) and extreme lunar thermal shifts without heavy, bulky shielding.	10: >100krad, -230°C to +120°C 8-9: 20-100krad, -150°C 5-7: 5-20krad, -80°C 0: <5krad, unshielded (Fails)	15%	8	8	7
Hazard & Albedo Detection	Evaluates ability to accurately map 3D surface roughness, hazards, and albedo changes in grazing lunar sunlight.	10: High-res 3D depth & albedo 6-9: Good albedo, moderate depth 1-5: Spectral only, poor depth 0: Cannot detect hazards (Fails)	30%	10	5	8
Data Processing	Rates the raw data bandwidth. Complex scientific data can easily overwhelm the drone's SoC.	10: < 10 Mbps (Easy processing) 6-9: 10 - 50 Mbps (Moderate) 1-5: 50 - 100 Mbps (Heavy burden) 0: > 100 Mbps (Overwhelms)	20%	8	6	3
TOTALS:			100%	87.50%	63.50%	56.00%

Appendix B2.3 - Illumination Detection Trade Study

Power Generation Architecture - Base Station						
Criteria	Rationale	Grade	Weight	Body-mounted	Simple deployable	Multi-panel deployable
Generated Power vs. Requirements	Ensures the architecture can meet all base station and drone power demands.	1: Does not meet power requirements 2: Meets minimum requirements with no margin 3: Meets requirements with acceptable margin 4: Exceeds requirements with strong margin	35%	1	3	4
Reliability / Deployment Risk	Evaluates risk of failure, especially deployment reliability for solar arrays.	1: High risk of failure or deployment issues 2: Moderate reliability; some risk 3: Good reliability; low risk 4: Very reliable; minimal risk	25%	4	3	2
Mass / Specific Power	Assesses how efficiently the architecture delivers power relative to its mass.	1: Heavy system; poor W/kg 2: Moderate mass; average W/kg 3: Good mass efficiency; good W/kg 4: Very lightweight; excellent W/kg	10%	4	3	3
Integration Complexity	Measures how difficult the architecture is to integrate structurally and electrically.	1: Very complex; difficult integration 2: Moderate complexity; requires careful integration 3: Manageable complexity; standard integration 4: Simple integration; minimal effort	15%	4	3	1
Dust & Shadow Mitigation	Measures how easily the system can clear dust or adjust to low sun angles/terrain shadows.	1: No adaptation. Fixed flat; dust collects and shadows block power completely 2: Low adaptation. Cannot move, but large size slightly offsets dust losses 3: Good adaptation. Semi-adjustable; can reduce dust or shadow impact 4: Excellent adaptation. Actively tilts to trace low sun, avoid shadows, and dump dust	10%	1	4	2
Landing Safety	Evaluates the physical crash hazard the arrays pose to a returning scout drone.	1: High hazard. Large, sprawling wings can create a major crash risk for the drone 2: Moderate hazard. Wide structures limit the drone's safe landing path 3: Low hazard. Single, compact panel footprint is easy for the drone to avoid 4: Zero hazard. Leaves the docking area completely clear	5%	4	3	1
TOTAL			100%	58.75%	63.75%	58.75%

Appendix B2.4 - Power Generation Architecture Trade Study

PMAD/Power Regulation Component Selection - Base Station							
Criteria	Rationale	Grade	Weight	2NDSpace SOLO - EPS 8	Argotec PCDU ZEUS	Argotec PCDU PHOENIX	Ibeos Modular EPS, 500–2000 W
Power Capability	Measures capability of handling the combined electrical loads from the base station, solar tracking mechanism, payloads, communications, thermal control, and scout drone charging path.	1: Under 150 W 2: 150-199 W 3: 200-249 W 4: 250 W+	30%	2	3	4	4
Mass	Ensures that the power system should not take unnecessary space or weight from the base station design.	1: Heavy / bulky 2: Moderate 3: Low mass 4: Very low mass	15%	3	4	2	2.5
Efficiency	Ensures that wasted energy and heat is reduced in a system.	1: <85% 2: 85-90% 3: 91-95% 4: 96%+	15%	3	2	2	4
Voltage Compatibility	Measures voltage levels needed by various subsystems.	1: Limited outputs 2: Some useful outputs 3: 3.3/5/12 V supported 4: 3.3/5/12 V plus battery/unregulated or 28 V support	15%	3	4	4	4
Fault Protection	Protects loads and allows health monitoring.	1: Limited or unclear protection 2: Basic protection only 3: Good protection and telemetry 4: Strong fault-tolerant design with protection, telemetry, and fault isolation/recovery features	15%	4	4	3	4
Integration Complexity	Monitors how PMAD can affect wiring, testing, voltage interfaces, and schedule risk.	1: Difficult integration / many unknowns 2: Moderate complexity 3: Manageable integration 4: Simple, low-risk integration	10%	4	4	2	3
TOTAL			100%	73.75%	85.00%	76.25%	91.88%

Appendix B2.5 - PMAD, Power Regulation, and Control Trade Study

Voltage Bus Architecture - Base Station						
Criteria	Rationale	Grade	Weight	Low-Voltage Regulated Bus	Unregulated Battery Bus	Hybrid Voltage Bus
Voltage Compatibility	Must support avionics, sensors, communications, controls, and high-power loads.	1: Limited outputs 2: Some useful outputs 3: 3.3/5/12 V supported 4: 3.3/5/12 V plus battery/unregulated or 28 V support	30%	2	2	4
Power Efficiency	Lower current and fewer conversions reduce losses and heat.	1: High losses expected 2: Few unnecessary conversion losses 3: Good efficiency 4: Best efficiency by matching loads to rails	25%	2	3	4
Integration Complexity	Should align with selected Ibeos EPS outputs.	1: Difficult integration / many unknowns 2: Moderate complexity 3: Manageable integration 4: Simple, low-risk integration with Ibeos EPS outputs	20%	3	2	4
Growth Flexibility	Should allow future loads like drone charging electronics, gimbal motors, or payload changes.	1: Little room for future loads 2: Limited growth 3: Some flexibility 4: Supports future loads	15%	2	3	4
Fault Protection	Protects loads and allows health monitoring.	1: Limited or unclear protection 2: Basic protection only 3: Good protection and telemetry 4: Strong fault-tolerant design with protection, telemetry, and fault isolation/recovery features	10%	3	2	4
TOTAL			100%	57.50%	60.00%	100.00%

Appendix B2.6 - Voltage Bus Architecture Trade Study

EE - Base Station Battery Type Trade Study						
Criteria	Rationale	Grade	Weight	Li-ion 18650 COTS	Space-Grade Li-ion	Solid-State
Energy Storage per Kilogram	Ensures efficient power storage within mission mass constraints	1: <110 Wh/kg 2: 110+ Wh/kg 3: 130+ Wh/kg 4: 150+ Wh/kg	35%	4	4	4
Thermal Performance	Assess battery performance across the range of temperatures at the Lunar south pole	1: Not Viable 2: Requires significant heating 3: Minor/ moderate heating required 4: No heating required	25%	4	4	2
Safety	Assess risk of catastrophic failure	1: High/unacceptable risk of combustion / catastrophic failure 2: Moderate risk of combustion / catastrophic failure 3: Low risk of combustion / catastrophic failure 4: No / Very Low risk of combustion / catastrophic failure	15%	3	4	4
Cycle Life	Measures battery health across various charge/discharge cycles throughout the mission	1: <200 cycles 2: 200+ cycles 3: 500+ cycles 4: 1000+ cycles	10%	4	4	4
TRL / Flight Heritage	Certifies compliance with given system requirements of a minimum of TRL 6	1: Untested 2: Ground testing 3: LEO heritage 4: Lunar/ Deep space heritage	10%	4	4	2
Cost / Availability	Certifies compliance with mission budget of \$250 million	1: High cost/ Inadequate supply 2: Moderate cost/ scarce availability 3: Low premium cost/ moderately available 4: Low price/widely available	5%	4	2	2
TOTAL			100%	96.25%	97.50%	80.00%

Appendix B2.7 - Base Station Battery Type Trade Study

Battery Type - Scout Drone						
Criteria	Rationale	Grade	Weight	LiPo Pouch COTS	Space-Grade Li-Ion	21700 High-Drain
Packaging	Evaluates battery mass and storage space given the relatively small margin	1: Cells are bulky and take up significant space 2: Fixed size; requires a structural workaround 3: Cells are small and can fit within constrained spaces 4: Cells are flexible/ conformal	30%	4	3	1
Energy Storage per Kilogram	Ensures efficient power storage within mission mass constraints	1: <110 Wh per kilogram 2: 110-129 Wh per kilogram 3: 130-149 Wh per kilogram 4: >150 Wh per kilogram	25%	3	3	3
Environmental Tolerance	Measures battery tolerance of extreme temperature swings and constant vibration	1: No flight heritage, extreme sensitivity 2: Limited flight heritage, highly sensitive 3: Qualified with elevated environmental sensitivity 4: Qualified with extensive flight heritage	20%	3	4	3
Pulse Performance	Weights performance during spikes in power demand during flights	1: Cannot deliver reliably without a significant voltage drop 2: Significant performance drop during high demand 3: Meets peak loads; some voltage drop 4: Handles high demand even when cold soaked	15%	3	4	4
Cycle Life	Assess battery longevity throughout the duration of the mission	1: <100 cycles before capacity degradation 2: 100-199 cycles before capacity degradation 3: 200-499 cycles before capacity degradation 4: >500 cycles before capacity degradation	10%	2	4	4
TOTAL			100%	80.00%	86.25%	66.25%

Appendix B2.8 - Drone Battery Type Trade Study

PMAD/Power Regulation Component Selection - Drone							
Criteria	Rationale	Grade	Weight	IBEOS Modular EPS	EnduroSat	GomSpace P60	Pumpkin AMPS
Power Capability	Measures capability of handling the combined electrical loads from the base station, solar tracking mechanism, payloads, communications, thermal control, and scout drone charging path.	1: Under 150 W 2: 150-199 W 3: 200-249 W 4: 250 W+	30%	4	2	3	2
Mass	Ensures that the power system should not take unnecessary space or weight from the base station design.	1: Heavy / bulky 2: Moderate 3: Low mass 4: Very low mass	15%	4	4	2	3
Efficiency	Ensures that wasted energy and heat is reduced in a system.	1: <85% 2: 85-90% 3: 91-95% 4: 96%+	15%	4	2	3	2
Fault Protection	Protects loads and allows health monitoring.	1: Limited or unclear protection 2: Basic protection only 3: Good protection and telemetry 4: Strong fault-tolerant design with protection, telemetry, and fault isolation/recovery features	15%	4	2	3	4
Interface Compatibility	Evaluates integration complexity with Base Station units	1: Difficult integration / many unknowns 2: Moderate complexity 3: Manageable integration 4: Simple, low-risk integration	15%	4	3	3	4
Radiation/ Environmental Tolerance	Measures unit tolerance to radiation and temperature swings encountered during flights into PSRs	1: Extreme sensitivity to radiation/ temperature swings 2: High sensitivity to adverse environmental conditions 3: Some heritage with proven tolerance 4: Radiation hardened/ tolerant of constant vibration	10%	3	2	3	3
TOTAL			100%	97.50%	61.25%	71.25%	71.25%

Appendix B2.9 - Drone PMAD/PDU Component Selection Trade Study

Charging Interface - Base Station and Drone						
Criteria	Rationale	Grade	Weight	Cradle Contact Pads/ Spring Probes	Robotic Arm	Hardwired Umbilical
Reliability and Simplicity	Ensures convenient and easy charging. Fewer moving parts means a longer lifespan	1: Motor-driven robotic mechanism required 2: No motors, but requires assistance with alignment 3: Minimal moving parts; springs only 4: No moving parts; passive contact only	25%	4	1	2
Autonomy	Low levels of human input decreases risk	1: Requires human action to charge 2: Autonomous alignment, but requires human verification to charge 3: Autonomous alignment, but requires sensor verification to charge 4: Charging begins on contact	20%	4	3	1
Power Transfer Efficiency	Measures how much power moves from the base station EPS to the drone; critical given mass constraints	1: <80% 2: 80-89% 3: 90-95% 4: 96%+	20%	4	4	4
Environmental Tolerance	Measures interface tolerance of extreme temperature swings and constant vibration	1: Extreme sensitivity to dust; degradation expected 2: High sensitivity to dust; degradation over time 3: Some tolerance of dust 4: Tolerant to dust accumulation; protected contacts	15%	3	2	2
Mass and Volume	Verifies compliance with mission mass constraint	1: Requires significant structural accommodations; weight >1kg 2: Moderate hardware; weight between 0.6-1kg 3: Light hardware; weight between 0.2-0.5kg 4: Minimal hardware; combined weight <0.2kg	10%	4	1	4
Spaceflight Heritage/ TRL	Certifies compliance with given system requirements of a minimum of TRL 6	1: TRL 5 or lower; not suitable for flight 2: TRL 6; borderline suitability 3: TRL 7-8; some heritage 4: TRL 9; extensive heritage	10%	4	3	4
TOTAL			100%	96.25%	58.75%	65.00%

Appendix B2.10 - Charging Interface Trade Study

Appendix C: Direct Cost Calculations

Subsystem	Equation	Cost Drivers	Cost ('04 \$K)	Cost ('26 \$K)	Source
Structure & Mechanisms Subsystem	$Y = 407 + 19.3 X * \ln(X)$	X = Structure & Mechanisms Mass (kg) = 17	\$7081.7	\$25100	<i>The New SMAD</i>
Thermal Control Subsystem (TCS)	$Y = 335 + 5.7 X^2$	X = Thermal Control Mass (kg) = 13.7	\$3657.633	\$9722.964	<i>The New SMAD</i>
Attitude Determination & Control Subsystem (ADCS)	$Y = 1850 + 11.7 X^2$	X = ADCS Mass (kg) = 11	\$3265.7	\$8681.028	<i>The New SMAD</i>
Electrical Power Subsystem (EPS)	$Y = 1261 + 539 X^{0.72}$	X = EPS Mass (kg) = 28.53	\$8188.360638	\$13733.5091	<i>The New SMAD</i>
Telecomms Subsystem	$Y = 456 + 55.5 X^{1.35}$	X = Telecom Mass (kg) = 1.255	\$1389.968847	\$3694.62	<i>The New SMAD</i>
Command & Data Handling (CDH) Subsystem	$Y = 658 + 75 X^{1.35}$	X = CDH Mass (kg) = 2.67	\$1794.427096	\$4768.452	<i>The New SMAD</i>
Payload - Mass Spectrometer	$Y = 652 * \text{TotalMass}^{0.62} * \text{TotalMaxPwr}^{0.44}$	TotalMass (kg) = 4.5 TotalMaxPwr (W) = 193	\$4055.771403	\$10780.24039	NICM
Payload - Rad. Dosimeter	$Y = 1,470 * \text{TotalMass}^{0.29} * \text{TotalMaxPwr}^{0.36}$	TotalMass (kg) = 0.16 TotalMaxPwr (W) = 26.8	\$375.1969235	\$997.27342	NICM
Payload - Stereoscopic Camera	$Y = 392 * \text{TotalMass}^{0.45} * \text{TotalMaxPwr}^{0.49} * \text{DsgnLife}^{0.32}$	TotalMass (kg) = 2 TotalMaxPwr (W) = 2.43 DsgnLife (months) = 3	\$1504.126139	\$3997.96728	NICM
Payload - ToF Camera	$Y = 392 * \text{TotalMass}^{0.45} * \text{TotalMaxPwr}^{0.49} * \text{DsgnLife}^{0.32}$	TotalMass (kg) = 0.7 TotalMaxPwr (W) = 3.88 DsgnLife (months) = 3	\$1844.719691	\$4903.26494	NICM