The Lunar Geophysical Network

Final Report
Submitted in response to
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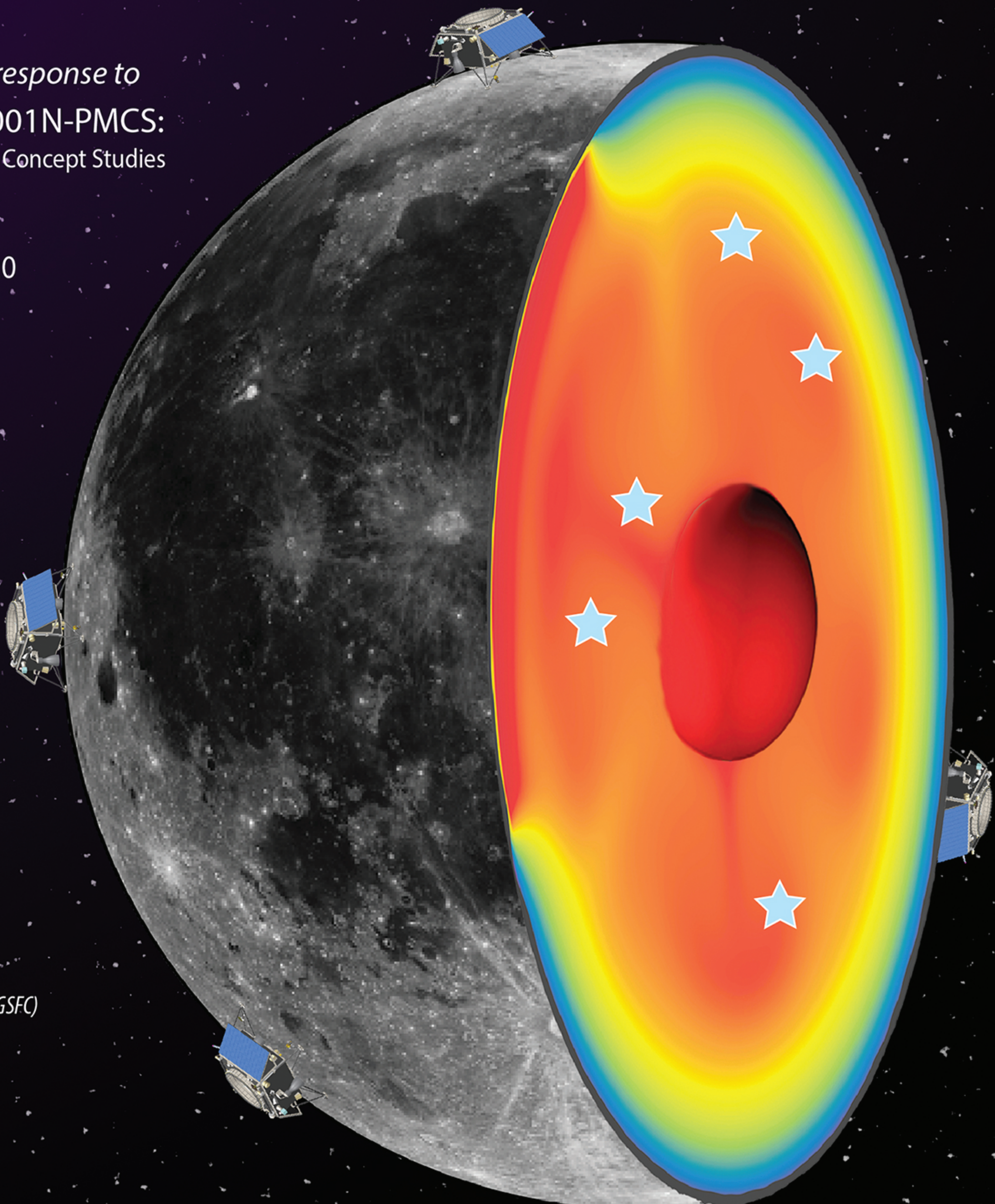
Planetary Mission Concept Studies

August 10, 2020



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LGN Decadal Study
Final Report

Presented to the Planetary Decadal Survey Steering Committee and Inner Planets Panel

August 10, 2020

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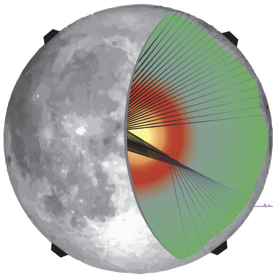

FACT SHEET

The Lunar Geophysical Network

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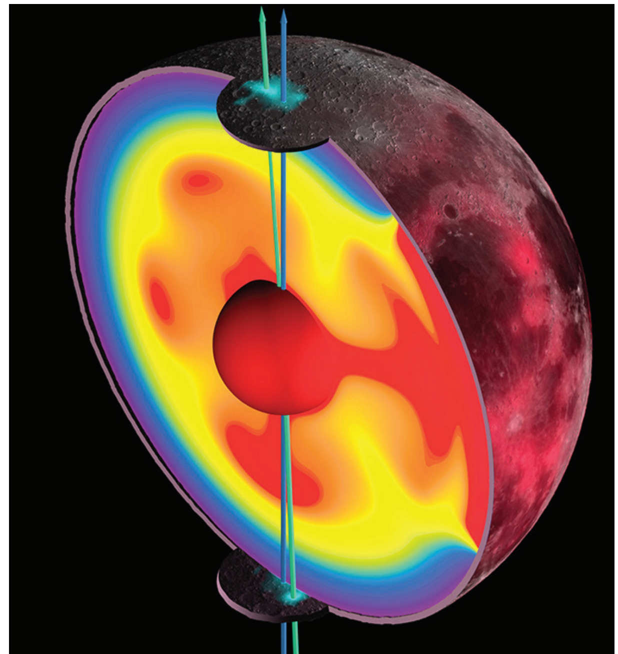
A Companion Piece to the 2020 NASA Planetary Mission Concept Study Report

MISSION OVERVIEW

The Lunar Geophysical Network (LGN) is a mission currently in formulation for NASA's New Frontiers 5 Announcement of Opportunity. The baseline mission consists of four solar-powered landers, broadly distributed across the Moon's surface, outfitted with identical instruments to make geophysical observations of the Moon's internal structure and thermal state within distinct lunar terranes.

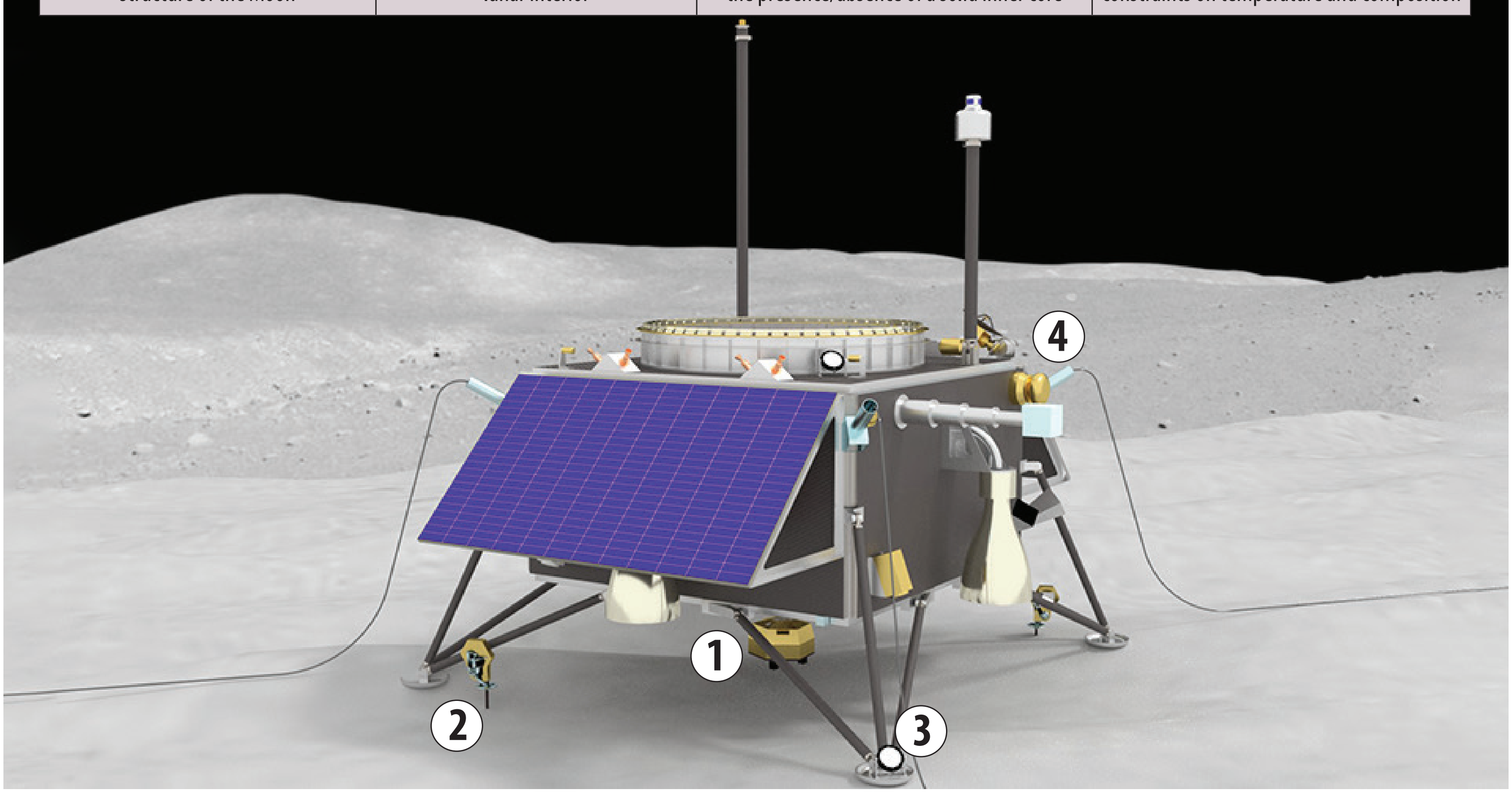
The goal of the mission is to understand the initial stages of terrestrial planet evolution. Terrestrial planets all share a common structural framework (crust, mantle, core) which is developed very shortly after formation and that determines subsequent evolution. While much of Earth's early structural evidence has been destroyed by plate tectonics, the so-called "ancient" planetary bodies, including the Moon, retain more information about their early interior structure. The Moon's small size – and resulting heat engine – means that the initial differentiation event is likely preserved. This is supported by source region modeling of Apollo mare basalt samples.



Credit: James Keane (JPL)

INSTRUMENT COMPLEMENT

1 Seismometer	2 Heat Flow Probe	3 Retroreflector	4 Magnetotelluric Sounder
Understand the current seismic state and determine the detailed internal structure of the Moon	Measure the heat flow to characterize the temperature structure of the lunar interior	Constrain the deep mantle environment, fluid core/solid mantle boundary conditions, and the presence/absence of a solid inner core	Determine the electrical conductivity structure of the interior to establish joint constraints on temperature and composition



MISSION CONCEPT

The LGN mission will deploy four landers to permit global distribution (including the far side) and allow for redundancy, as a threshold of two landers can still achieve the goal of global coverage. The landers should be long-lived (6 years with a goal of 10 years) to maximize science and allow other nodes to be added by international and commercial partners during the lifetime of the mission, thus increasing the fidelity and value of the data obtained.

The four landers will be launched on one launch vehicle and sent into lunar orbit, where the landers will be deployed sequentially from a parent spacecraft. The lander-carrying spacecraft will remain in orbit to serve as the communications relay, thus allowing a lander to be placed on the far side of the Moon. Each lander will also be able to send data direct to Earth, so the communications orbiter acts as a back-up for near side landers.

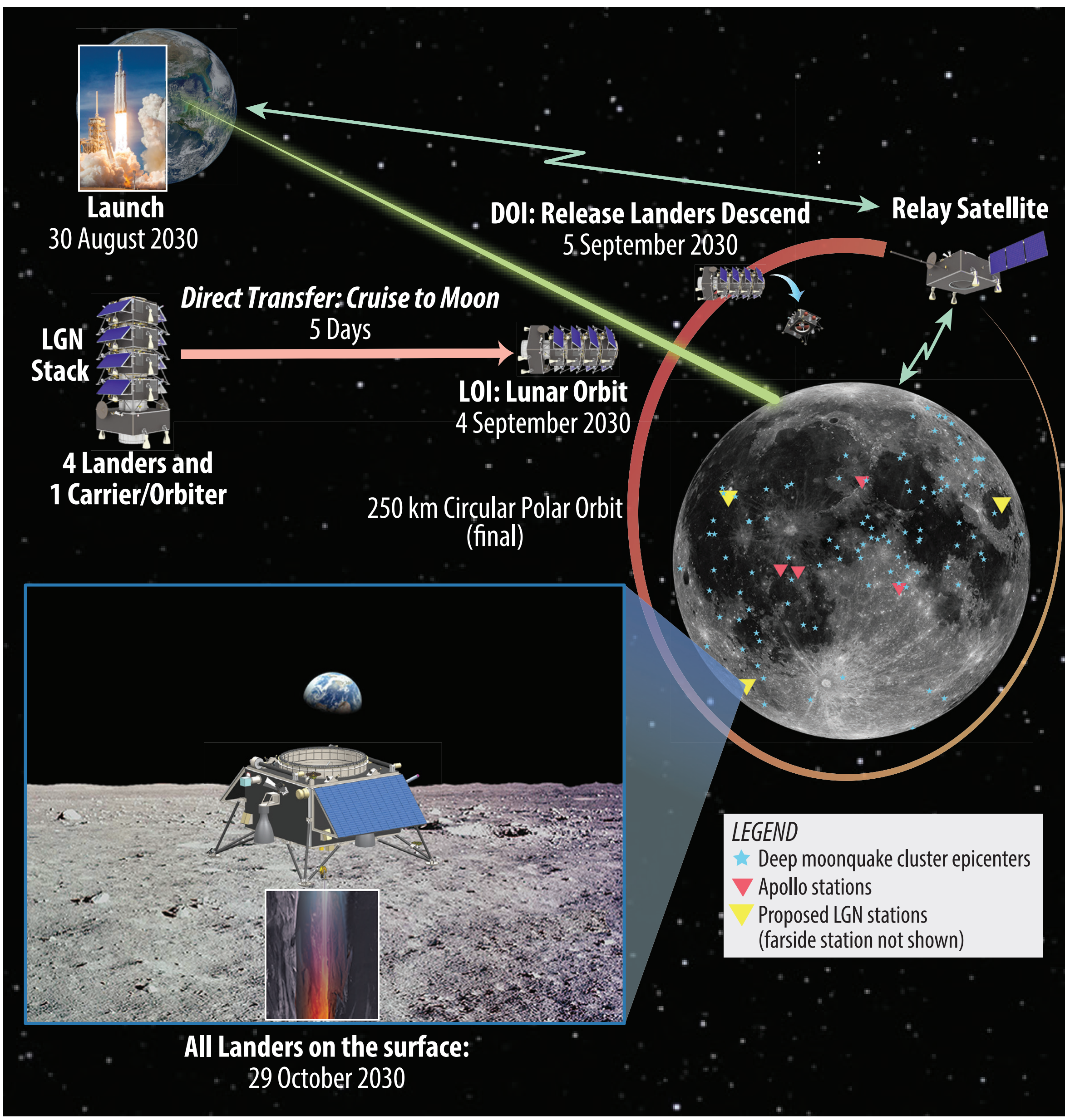
SCIENCE OBJECTIVES

Objectives:

- Define the interior structure of the Moon.
- Constrain the interior and bulk composition of the Moon.
- Delineate the vertical and lateral heterogeneities within the interior of the Moon as they relate to surface features and terranes.
- Evaluate the current seismo-tectonic activity of the Moon.

Investigations:

- Determine the size, state and composition of the lunar core.
- Determine the state of as well as the chemical/physical stratification in the lunar mantle.
- Determine the thickness of the lunar crust and characterize its vertical and lateral variability.
- Determine the thermal state of the lunar interior and elucidate the workings of the planetary heat engine.
- Monitor impacts on to the lunar surface as an aid to exploring the lunar interior.



The proposed landing sites for Lunar Geophysical Network greatly expand the footprint of the prior Apollo geophysical array. The stations are located far from the boundary between the Procellarum KREEP Terrain and the Feldspathic Highlands Terrain, therefore providing unambiguous geophysical signals.

A wider geographical spread of stations permits improved global structure determination from seismology and laser ranging. And a station on the farside will optimize new assessments of global seismicity and the structure of the Moon's lower mantle and core.

1. SCIENTIFIC OBJECTIVES

The goal of the Lunar Geophysical Network (LGN) mission is to understand the evolution of terrestrial planets, from their initial stages of formation, differentiation, and subsequent persistence (or lack) of internal dynamics into the present. The Moon is a natural target for a geophysical network mission as it presents an opportunity to study an internal heat engine that waned early in planetary evolution, and thereby enabled preservation of the initial differentiation event, information that has been lost on Earth due to crustal recycling and weathering of our most ancient rocks. This lunar model is supported by analyses of returned Apollo basaltic samples, which are consistent with derivation from a source comprised of cumulates that crystallized from an initial magma ocean and subsequently underwent an overturn event (Taylor and Jakes, 1974; Snyder *et al.*, 1997).

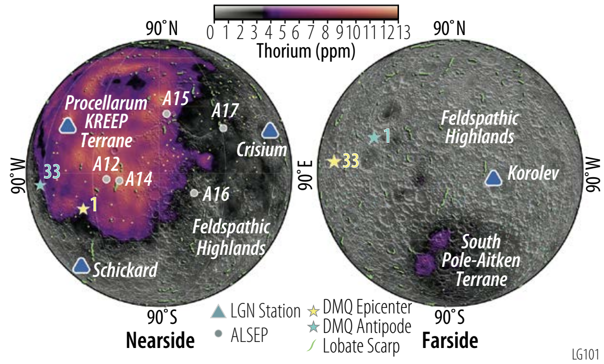


Figure 1: The Lunar Geophysical Network will place stations that sample across major lunar terranes and enable new investigation of the deep interior and tectonic evolution. The proposed LGN stations (blue triangles) are positioned to take advantage recent lobate scarp seismicity (green lines, Watters *et al*., 2019) and deep moonquake (DMQ) clusters (yellow stars/dots) and their antipodal locations (cyan stars/dots), both types of seismicity detected by the Apollo ALSEP stations (gray circles). Two deep moonquake clusters (01 and 33) are highlighted (see text). The LGN is designed to geophysically interrogate the internal structure, temperature, composition, and tectonics/seismicity both in the Feldspathic Highlands (unsampled by Apollo) and within the Procellarum KREEP terranes, outlined by the Lunar Prospector thorium abundance *e.g.*, Lawrence *et al.*, 2000. Shaded surface relief is derived from LOLA topography (Smith *et al*., 2017). See **Appendix B, Figure B-55** for enlarged view.

1.1 Science Questions and Objectives

Our first look into the Moon's interior came from the Apollo Lunar Surface Experiment Packages (ALSEPs) that deployed surface magnetometers, placed laser NGLR arrays, installed seismometers that detected moonquakes and meteorite impacts, and took heat flow measurements – key geophysical information that has advanced our knowledge of the Moon's internal structure, evolution, and dynamics. However, it is very evident that our understanding of the lunar interior remains incomplete. The identification of different lunar surface terranes from new global surface compositional data have produced a paradigm shift in our understanding for the global evolution model of the Moon (Jolliff *et al.*, 2000; Laneuville *et al.*, 2018). These data demonstrate that there is a fundamental limitation to the ALSEP geophysical datasets, as all were collected in or very near one anomalous region, the Procellarum KREEP Terrane (PKT) (**Figure 1**). Subsequent orbital missions have expanded the global geophysical picture of the interior (*e.g.*, GRAIL, Kaguya, LRO, etc.), but only a landed long-lived geophysical network can address the significant questions that remain unanswered by Apollo:

- How does the overall composition and structure of the Moon inform us about initial differentiation of terrestrial planets?
- What is the state, structure, and composition of the mantle and is it consistent with the lunar magma ocean hypothesis (or are there resolvable discontinuities)?
- What is the present heat budget and how could the Moon experience magmatism for >3 b.y.?
- What is the crust and mantle heterogeneity within and between different terranes?
- Based on a constrained size, state and composition, how did the lunar core form and could it have supported a global magnetosphere (as indicated by sample analyses – *e.g.*, Weiss and Tikoo, 2014)
- What is the bulk composition of the Moon?

New geophysical data are needed to address these questions and add greater fidelity to data sets already obtained. For example, the GRAIL gravity data that inform on crustal thickness are constrained by Apollo seismic data at depth, but the fidelity across the lunar surface is poor due to the narrow aperture of the Apollo passive seismic network and localization in the thinned and likely anomalous crust of the PKT (*e.g.*, Hood, 1986). The large discrepancies between the size (and nature) of the lunar core defined by seismic, lunar laser retroreflector (LLR), and GRAIL data reflect the lack of fidelity in Apollo seismic and ongoing LLR data (*e.g.*, Williams *et al.*, 2014). Therefore, the fundamental purpose of the LGN mission is to distribute Landers with seismometers, heat flow probes, electromagnetic sounders, and laser NGLRs around the Moon, including on the farside, that sample a representative suite of lunar terranes. A similar analogy can be drawn for the Earth: unless geophysical information can be drawn across a representative number of terranes on our own planet (*e.g.*, oceanic plates, continents, cratons, plate margins, etc.), it would have been impossible to formulate an accurate geophysical picture of the different internal processes at work within the Earth. Indeed, it was shortly after geophysical exploration of the oceanic plates commenced that the paradigm shift to plate tectonics took place within the geosciences. Thus, a widely dispersed network of geophysical stations will enable LGN to fully interrogate the deep interior of the Moon, more accurately locate hypocenters of large moonquakes and impacts, and constrain crustal thickness variations across a wide range of lunar terranes – none of which are possible with the Apollo data. The LGN mission will allow more intricate questions to be addressed that have resulted from previous work, such as:

- Do shallow moonquakes represent movement along thrust faults (*e.g.*, Watters *et al.*, 2019)?
- Do moonquakes present a threat to future human infrastructure (Oberst and Nakamura, 1992)?
- Do deep moonquakes occur on the farside of the Moon (Nakamura *et al.*, 1982; Nakamura, 2005)?
- What is the mechanism for triggering deep moonquakes (Weber *et al.*, 2009; Kawamura *et al.*, 2017)?
- Are there global discontinuities in the mantle and do they relate to the lunar magma ocean (Nakamura *et al.*, 1982; Lognonné *et al.*, 2003)?
- Do different lunar terranes have unique heat flow budgets and what does this imply about the bulk geochemical composition of the Moon (Laneuville *et al.*, 2018)?
- What is the lateral/vertical structure and composition as revealed by electrical conductivity (Hood *et al.*, 1982; Grimm, 2013)?

1.2 Science Traceability

LGN's primary goal is to: Understand the initial stages of terrestrial planet evolution. To achieve this goal, LGN has four Objectives explored through five Investigations.

Objectives:

- Evaluate the interior structure and dynamics of the Moon.
- Constrain the interior and bulk composition of the Moon.
- Delineate the vertical and lateral heterogeneities within the interior of the Moon as they relate to surface features and terranes.
- Evaluate the current seismo-tectonic activity of the Moon (Kumar *et al.*, 2016, 2019; Watters *et al.*, 2019).

Investigations:

- Determine the size, state, and composition of the lunar core (building on the work of Weber *et al.*, 2011).
- Determine the state and chemical/physical stratification of the lunar mantle (is there garnet in the lower mantle; Neal, 2001?).
- Determine the thickness of the lunar crust and characterize its vertical and lateral variability (refining and adding fidelity to the GRAIL results of Wieczorek *et al.*, 2013).
- Determine the thermal state of the lunar interior and elucidate the workings of the planetary heat engine.
- Monitor impacts on the lunar surface as an aid to exploring the lunar interior.

The instruments selected for LGN have the capabilities required to make the measurements that enable its investigations and objectives and ultimately answer the primary goal (see the **Science Traceability Matrix**).

2. MISSION CONCEPT

2.1 Overview

The Lunar Geophysical Network mission will deploy an Orbiter and four solar-powered Landers with instrumentation as described by the International Lunar Network report (ILN, 2009) and outlined in **Section 1**. Based on the LGN Concept Study conducted as part of the previous decadal survey (Shearer & Tahu, 2013), four Landers are baselined. Deploying four Landers enables global distribution (including the farside) and allows for redundancy, as a threshold of two Landers still achieves the threshold mission. The four Landers are long-lived (6 years – covering one primary tidal cycle – with a goal of 10 years) to maximize science and allow other nodes to be added by international and commercial partners during the lifetime of the mission, thus increasing data fidelity.

The Orbiter and four Landers will be launched together and the Orbiter will provide the ΔV needed to place all flight systems into lunar orbit, where the Landers will be deployed sequentially. Each Lander will contain a very broadband and short period seismometer package (beneath the Lander), one short period seismometer on the Lander deck (primarily to measure Lander noise), one buried short period seismometer (deployed from one of the legs), two heat flow probes (deployed from two other legs), two laser NGLRs (one on the Lander deck and one on a leg near the lunar surface), and a magnetotelluric sounder that uses four remote electrodes (deployed via a spring-loaded mechanism) and a fluxgate magnetometer. The sounder is supplemented by an electrostatic analyzer and a search coil magnetometer, both attached to the body of the Lander.

The Orbiter carries a communications payload that provides a relay for all the Landers. This allows a Lander to be placed on the farside of the Moon by providing a relay back to Earth for deployment operations and data transfer. As each Lander will be capable of sending data directly to Earth, the Orbiter also provides redundancy for the nearside Landers. In addition, the Orbiter carries a magnetometer on a two-meter boom to supplement surface electromagnetic measurements. The mission design is summarized in the **LGN Fact Sheet**.

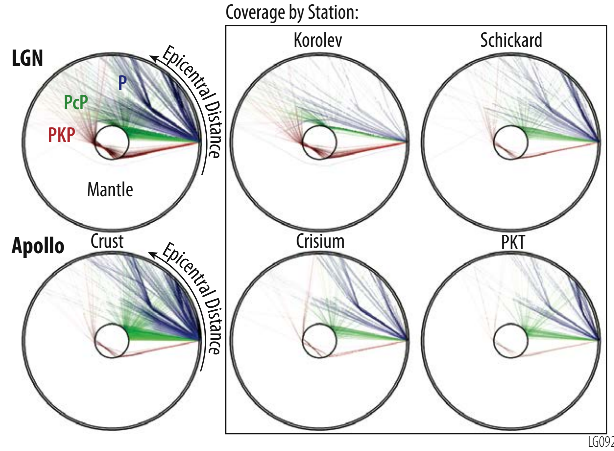


Figure 2: The LGN stations are situated to vastly improve our knowledge about the lunar deep mantle and core. The wider aperture of the LGN array and geographical distribution of the stations compared to Apollo provides ray path sampling of the entirety of the lunar interior. Using the seismicity catalog of Apollo, we calculated raypath densities for P, PcP, and PKP to the LGN stations across epicentral distances using the Weber *et al.*, (2011) velocity model and proposed LGN station locations (**Figure 1**). Consistent seismic raypath density across epicentral distance is crucial for providing a full picture of lunar internal structure. The LGN provides a significantly denser sampling for core traversing waves (e.g., PKP), waves that sample the deep mantle (PcP, P), and more uniform sampling of the crust and upper mantle than Apollo. The more complete coverage is enabled by the deployment of four stations, particularly from a farside station in Korolev crater that provides deep mantle and core sampling, and will allow for the detection of farside seismicity that was unobserved by Apollo.

For purposes of this study, landing sites have been chosen to reflect a global distribution around the Moon, greatly increasing the footprint of the former Apollo array (**Figure 1**). Four landers will be deployed sequentially, three on the nearside and one on the farside. Sites were selected in order to 1) maximize the recording of seismic signals from known deep moonquake clusters as they pass through the deepest mantle and core (**Figure 2** and Yamada *et al.*, 2013), 2) unambiguously sample heat flow both inside and well outside the boundaries of the PKT, 3) push the footprint of the existing laser NGLR array further towards the limbs, improving accuracy of lunar rotation and tide determinations, and 4) avoid crustal magnetic anomalies that would contaminate the magnetotelluric sounding measurements. Rationale for these sites are as follows:

PKT ((P-60 young basalt field) site just south of the Aristarchus Plateau (latitude = 20.7˚; longitude = -47.4˚): Relatively young and flat volcanic terrain, with few craters and boulders. This landing site is well within the boundaries of the PKT and just south of the thorium anomaly at the Aristarchus Plateau, well-situated to detect both direct and core-reflected arrivals from the known nearside deep moonquake clusters.

Schickard Basin (latitude = -44.3˚; longitude = -55.1˚): This site is in the southern hemisphere of the Moon and the floor is partially flooded with basaltic lava flows that form a relatively flat landing site, with few craters and boulder fields. This landing site is well outside the PKT, situated ideally to detect seismic phases transmitted through the core by known deep moonquake clusters in the north-eastern quadrant of the nearside, and refracted arrivals from the farside A-33 deep moonquake cluster (**Figure 2**).

Crisium Basin (latitude = 18.5˚; longitude = 61.8˚): The basaltic lavas on the floor of the basin form a relatively flat terrain, but contain secondary crater populations that will need to be avoided. According to the latest crustal thickness maps, the primary crust is essentially absent (Wieczorek *et al.*, 2013) allowing mantle heat flow to be directly measured. Seismic measurements of the mantle at this site will also benefit from not being distorted by the fracture crust. For magnetotelluric measurements, the known magnetic anomalies within Mare Crisium (*e.g.*, Richmond and Hood, 2008) need to be avoided.

Korolev Basin (latitude = -2.4˚; longitude = -159.3˚): This site will allow the first surface geophysical measurements to be made on the farside of the Moon. The Korolev Basin affords a relatively flat and boulder-free landing area that is in the vicinity of a lobate scarp. It is situated well within the Feldspathic Highlands Terrane and in the highest topographic area of the Moon, which represents the thickest crust. The site is approximately antipodal to many nearside deep moonquake clusters, again improving ray coverage for core-traversing seismic phases (**Figure 2**).

2.2 Concept Maturity Level

This study was conducted with a goal of Concept Maturity Level (CML) 5, however, certain aspects of the design concept could be classified as a CML 3 or 4 (see **Table 1** for CML definitions). It presents an implementation concept at the subsystem level, as well as science traceability, mission requirements traceability, key technologies, heritage, risks and mitigations. Detailed cost models were developed. Further development is necessary to mature the final design architectures and approaches, especially to the Lander power and thermal concepts.

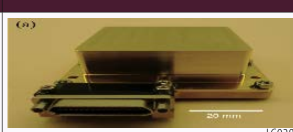
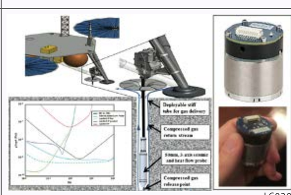
Table 1: LGN maturity is 5 in NASA's definitions of Concept Maturity Level (CML).

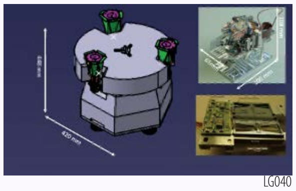
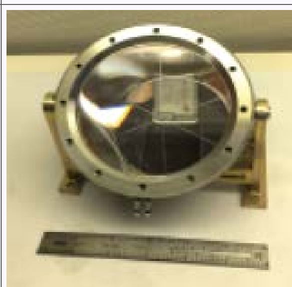
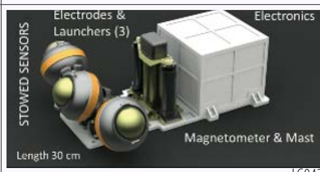
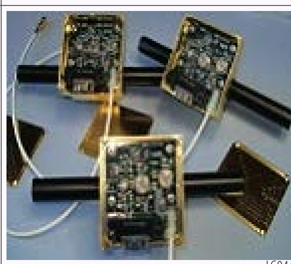
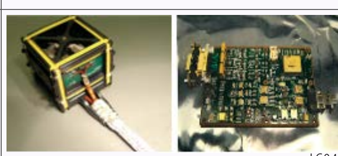
Concept Maturity Level	Definition	Attributes
CML 6	Final Implementation Concept	Requirements trace and schedule to subsystem level, grassroots cost, V&V approach for key areas
CML 5	Initial Implementation Concept	Detailed science traceability, defined relationships and dependencies: partnering, heritage, technology, key risks and mitigations, system make/buy
CML 4	Preferred Design Point	Point design to subsystem level mass, power, performance, cost, risk
CML 3	Trade Space	Architectures and objectives trade space evaluated for cost, risk, performance
CML 2	Initial Feasibility	Physics works, ballpark mass and cost
CML 1	Cocktail Napkin	Defined objectives and approaches, basic architecture concept

2.3 Current Technology Maturity Level

The proposed LGN mission plans to use modified instruments either currently in development or with flight heritage (*i.e.,* Mars InSight and Maven). **Table 2** below shows the current instrument TRL for the Moon and development plans. Risks associated with the Si-Audio Seismometer, LMSS-MT, and LMSS-Plasma current TRLs are anticipated to be retired by CLPS or SIMPLEx flights by the time of flight system development. The study recognizes the Moon thermal environment is much harsher than Mars; **Section 3** describes the thermal constraints and mitigation in more detail for each instrument. All Lander and Orbiter hardware have extensive heritage from multiple missions and have current TRLs greater than 6 as detailed in **Appendix B**.

Table 2: Instrument Development Plans : LGN instrument development plans actively mitigate risk by advancing all instruments to TRL 6 or higher by the end of Phase A.

Name	Photo	Current TRL for the Moon	TRL Development Plan or Heritage	Deployment Risk	Mitigation / Comments
Short Period Seismometer (on the deck and on the lunar surface)	 LG038	~7	InSight mission: Currently operating on Mars.	Low - deployment entails switching on the seismometer.	N/A
Si-Audio Seismometer	 LG039	5 (sensor) 4 (burial system)	For Si-Audio, its funded by DALI to advance the TRL, we are expecting to have TRL 6 hardware by September 2022, the end of the DALI contract.	Moderate - pneumatic burial may encounter subsurface rocks.	Ensure landing sites are rock-free and devoid of young craters.

VBB-SP Seismometer Package: - 4 VBB sensors in oblique conf. - 3 SP sensors in Z-H conf.	 LG040	~7	InSight mission: Currently operating on Mars. The lunar VBB will need a larger proof mass (249g), increasing the pendulum period to 5 sec, and a larger voltage of the displacement transducer, decreasing its electronic noise.	Low - deployment is relatively simple. Need to ensure that the surface under the Lander is rock-free and relatively flat.	Ensure landing sites are rock-free and devoid of young craters. Examination of surface under previous landers (inc. Apollo) shows rocket exhaust is not an issue.
Heat Flow Probe	 LG041	~5	Will fly on the CLPS lander to Mare Crisium in late 2022/early 2023.	Low to Moderate - pneumatic burial may encounter subsurface rocks. Deployment and all other systems will have been tested on the lunar surface before LGN.	Moderate - pneumatic burial may encounter subsurface rocks. Two heat flow probes are deployed to reduce this risk.
Next Generation Lunar Retoreflector (NGLR)	 LG042	>6	Will fly on the CLPS lander to Mare Crisium in late 2022/early 2023.	Low - deployment and all other systems will have been tested on the lunar surface before LGN. Surrounding rocks may inhibit full deployment of gimbal.	Ensure landing sites are rock-free and devoid of young craters.
LMSS-MT (Threshold)	 LG043	5	Will fly on the CLPS lander to Mare Crisium in late 2022/early 2023.	Low - deployment and all other systems will have been tested on the lunar surface before LGN. Surrounding rock may inhibit full deployment of electrodes.	Ensure landing sites are rock-free and devoid of young craters.
LMSS-SCM (Baseline)	 LG044	8	Heritage: the recently flown DSX TASC.	Low - boom deployment has high heritage.	N/A
LMSS - Plasma (Baseline)		4-5	Heritage: this is a new compact design based on the MAVEN SWIA.	Low - deployment entails switching on the instrument.	N/A
Orbiter Magnetometer	 LG045	9	A magnetomater that has been proven by previous space missions will be used.	Low - high heritage instrument and deployment mechanisms will be used.	N/A

2.4 Key Mission Trades

This study builds upon the LGN Concept Study conducted as part of the previous decadal survey (Shearer & Tahu, 2013). The previous decadal study baselined the use of an Advanced Stirling Radioisotope Generator (ASRG) in order to meet lunar night power needs. The current effort focused on developing a solar powered architecture concept and then comparing it to a concept that uses a Next Generation Radioisotope Thermoelectric Generator (NGRTG) to determine the optimal design that would fit within a New Frontiers cost cap. **Table 3** shows the comparison. Included in the effort was the exploration of using a carrier to act as an Orbiter that provides a communication relay for a farside Lander and a platform for an orbital magnetometer.

The architecture trade study revealed the pairing and combination of the various options (Solar Arrays, Batteries, NGRTG and Orbiter) have significant impact to the mass of the Landers and marginal

Table 3: The LGN team performed multiple architecture trade studies to refine the mission design, revealing the optimal use of resources is a single Orbiter/carrier that enables eliminating Lander propulsion mass.

Trade	Launch Mass (kg)	Launch Margin Falcon 9 Heavy Expendable	Launch Margin SLS Block 1	Number of Landers	Lander Dry Mass (kg)	Lander Mass (wet) (kg)	Orbiter Dry Mass (kg)	Orbiter Mass (wet) (kg)	Fits within Falcon 9 Heavy Expendable 5m fairing	Fits within SLS Block 1 5.1 m Fairing	Regulatory Approval Difficutly	Launch Site Integration and Test Complexity	Landing Operations Complexity	Landing Site Flexibility	Thermal Design Complexity	Farside Comm
Carrier, Comm, Battery and Solar Array Power	14,240.0	8.8%	89.6%	3	1,263.6	2,550.0	1,386.7	6,590.1	Yes	Yes	Low	Low	Low	High	Low	Yes
Carrier, Comm, Battery and Solar Array Power	16,879.8	-8.2%	60.0%	4	1,263.6	2,550.0	1,386.7	6,679.9	Yes	Yes	Low	Low	Low	High	Low	Yes
No Comm w/Carrier, Battery and Solar Array Power*	12,826.8	20.8%	110.5%	3	1,263.6	2,550.0	1,322.6	5,176.9	Yes	Yes	Low	Low	Low	High	Low	No
No Comm w/Carrier, Battery and Solar Array Power*	16,430.0	-5.7%	64.3%	4	1,263.6	2,550.0	1,322.6	6,230.1	No	Yes	Low	Low	Low	High	Low	No
No Comm, No Carrier w/Battery and Solar Array Power	13,991.9	10.8%	93.0%	3	1,658.5	4,664.0	N/A	N/A	Yes	Yes	Low	Low	High	Low	Low	No
No Comm, No Carrier w/Battery and Solar Array Power	18,655.9	-16.9%	44.7%	4	1,658.5	4,664.0	N/A	N/A	No	Yes	Low	Low	High	Low	Low	No
Carrier, Comm, RTG Power	9,589.3	61.6%	181.6%	3	892.3	1800.6	1,389.3	4,187.5	Yes	Yes	High	Medium	Low	High	High	Yes
Carrier, Comm, RTG Power	12,094.5	28.2%	123.2%	4	892.3	1800.6	1,389.3	4,892.1	Yes	Yes	High	Medium	Low	High	High	Yes
No Comm, w/Carrier, RTG Power*	9,500.1	63.2%	184.2%	3	892.3	1800.6	1,325.2	4,098.3	Yes	Yes	High	Medium	Low	High	High	No
No Comm, w/Carrier, RTG Power*	12,005.4	29.1%	124.9%	4	892.3	1800.6	1,325.2	4,802.9	Yes	Yes	High	Medium	Low	High	High	No
No Comm, No Carrier w/RTG Power	9,403.0	64.8%	187.1%	3	1,237.2	3134.3	N/A	N/A	Yes	Yes	High	Medium	High	Low	High	No
No Comm, No Carrier w/RTG Power	12,537.3	23.6%	115.4%	4	1,237.2	3134.3	N/A	N/A	Yes	Yes	High	Medium	High	Low	High	No

** No Comm w/Carrier options use one of the Landers to communicate with Earth during Lunar Transfer and Deployment of Landers.*

impact on cost. The cost is somewhat normalized by the increased pricing and launch costs of using an NGRTG or increased cost of the Landers with higher mass due to batteries. The overall technical result that is best suited for launching 4 Landers is the NGRTG with Orbiter concept. However, 3 Landers provide a graceful descope that is acceptable to meet science objectives (see **Section 3.5.3**). The 3 Lander solar array concept provides excellent performance with less complexity and more flexibility at a lower cost than the NGRTG.

A key factor in the architecture trade is the approach of using the Orbiter as a carrier to perform the Lunar Orbit Insertion (LOI) maneuver that places the Landers into a circular orbit. This eliminates additional propulsion masses that would have been required for each Lander to perform its own maneuver. Using a single carrier, which remains in orbit after delivering the Landers, yields a mass/propellant efficient Lander design and provides a communication relay to enable a farside Lander. Additional factors evaluated included the complexity introduced by both regulatory and integration/test of an NGRTG, thermal design complexity, landing site flexibility and landing operations complexity. The solar array concepts offered the lowest regulatory, integration/test, and thermal design complexity. Lander release from the Orbiter can be spaced out over months to minimize the operations team size and allow them to apply lessons learned to each subsequent Lander. The Orbiter also enables the ability to retry landing at a site with another Lander if a landing failed, providing redundancy.

Table 3 shows the results of the architecture trade study. Overall, the solar array Lander option with Orbiter was the least complex and most flexible but at the expense of a 4th Lander for the Falcon 9 Heavy launch vehicle. Options exist, such as to launch the 4th Lander separately on a second launch vehicle or to use a larger launch vehicle, but these were not explored in this study. The team performed additional engineering trades, including: Power, Attitude Control, and Communications. Analysis and rationale for these trades is detailed in **Appendix B**, **Section 1.4.2**.

2.5 Planetary Protection

The Lander is assumed to be categorized as Class I-L as per NID 8715.128 "Planetary Protection Categorization for Robotic and Crewed Missions to the Earth's Moon".

Table 4: Lander Payload Mass, Power and Mission Data Table.

	Mass			Average Power			Mission Data Volume For 6 Years (Gbits)
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)	
VBB/SP Seismometer	25	30	32.5	11	30	14.3	3,217
SP Seismometer (On Deck)	1	30	1.3	2	30	2.6	1,514
Silicon Audio/MEMS Seismometer	9.4	30	12.3	4.6	30	6.0	1,779
Heat Flow Probe (2)	14.2	30	18.5	12	30	15.6	284
Lunar Magnetotelluric Sounder Suite	6.1	30	7.9	8.6	30	11.2	11,353
Next Generation Lunar Retroreflector (2)	10	30	13.0	0.0	30	0.0	0
Close Range Imager (4), Panoramic Imager	9.8	30	12.7	25.8	30	33.5	1,779
Payload Totals	**76.5**	**30**	**99.5**	**64**	**30**	**83.2**	**19,926**
Mass and power estimates include instrument electronics and deployment systems as appropriate.							

3. TECHNICAL OVERVIEW

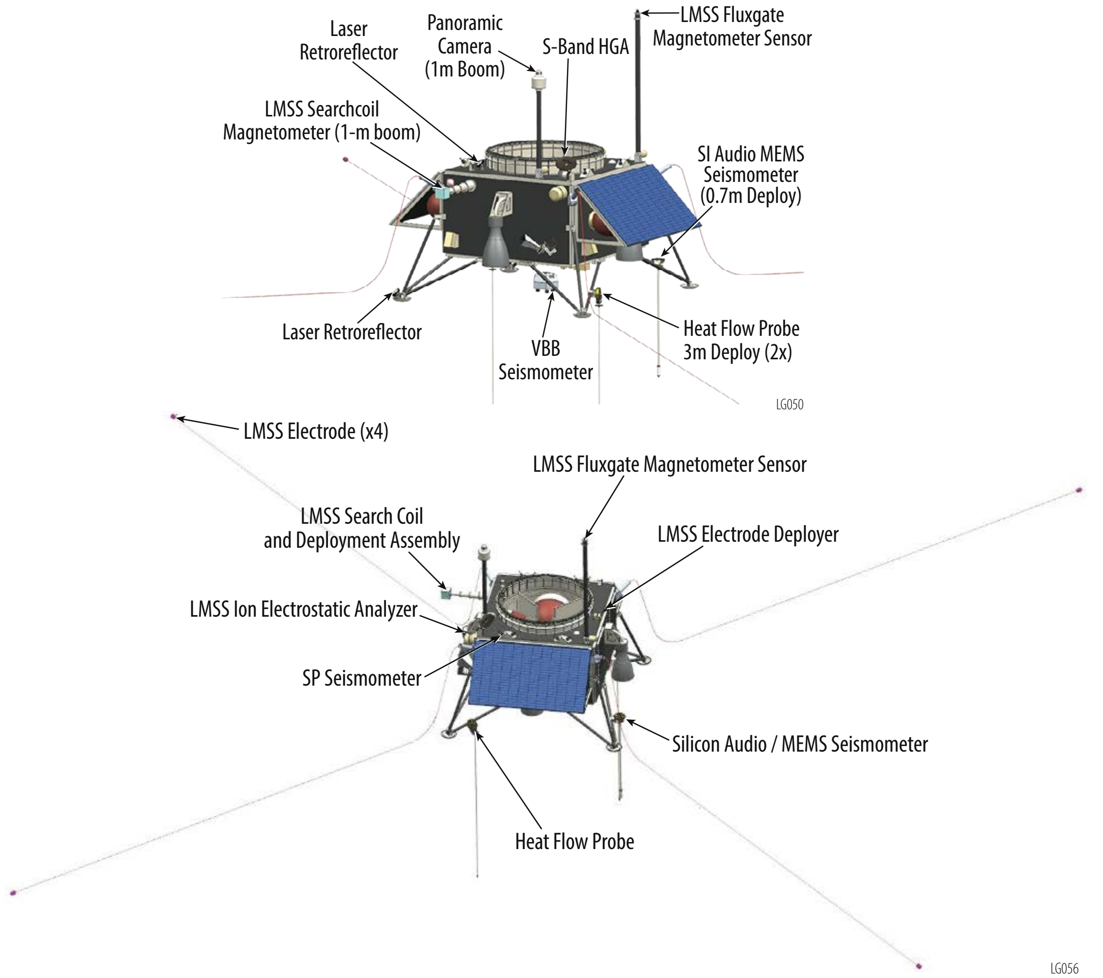


Figure 3: LGN Lander deployed.

3.1 Instruments Payload Descriptions

The LGN instruments are shown in **Figure 3**, the mass, power and mission data volume are given in **Table 4**. The instrument details are shown in **Table 5** and discussed in the following paragraphs.

Very Broadband Seismometer (VBB) (developed by CNES and IPGP in France): The proposed instrument is based on the InSight SEIS VBB seismometer (Lognonné *et al.*, 2019), with some adaptation the Moon. VBB will contain 4 sensors in an oblique configuration that measures ground acceleration in the 1mHz /25 Hz bandwidth. This seismometer requires a thermal regime between -55°C to +125°C and will be deployed on the surface under the Lander, complete with a thermal sheath to ensure thermal stability. With InSight heritage, it is currently TRL ~7 for the Moon. The seismometer will be lowered onto the lunar surface by a "crane" developed by Honeybee Robotics, which has a single actuator, single spool, and two cables (similar to the MSL Descent Brake Deployment in spool winding). The VBB instrument will be positioned at the center of the Lander, which will provide additional thermal protection from the lunar heat by taking advantage of the Lander shading. The VBB system is lowered using stainless-steel cables (1/8-inch diameter, 24 inches) on a bridle, with two cables connected to the bridle to stabilize rotational motion as it is lowered. The cables are retracted after cable separation from the VBB is complete, and the deployment mechanism is mounted directly above the VBB on the spacecraft, with pulleys on opposite sides of the cylinder. Cable deployment heritage comes from MSL SkyCrane, Mars2020 SkyCrane, and JWST sun-shield deployment.

Table 5: Instrument details.

Item *Instrument Type*	**VBB/SP Seismometer** *Seismometer*	**SP Seismometer** *Seismometer*	**Silicon Audio/ MEMS Seis** *Seismometer*	**Heat Flow Probe** *Subsurface Deployment*	**LMSS** *Electromagnetic / Plasma*	**Lunar Reflector** *Retroreflector*	**Cameras**	**Orbiter Magnetometer**
Size/Dimensions (m x m x m)	Deployed : 0.42 x .448 On Lander: 0.42 x 0.42 x .10	.03 x (.05 x .05 x .05)	05 diameter x .20 long	Deployment: .33 x .33 x .15 Avionics: .064 x .16 x.24	LMSS-MT (Threshold): Electronics .15 x .12 x .12; Electrodes (4) .15 x .08 x .08; Magnetometer .12 x .06 x .12 LMSS-SCM (Baseline): .15 x .15 x .15 LMSS - Plasma (Baseline): .17 x .15 x .20	24 x .26 x .20	Panoramic(1): Static (4): Electronics:	Sensor: .066 x .047 x .045 Electronics: .14 x .10 x .02 Cable: 0.008 dia Boom: 2 m
Mass Without Contingency (CBE*) kg	25	1	9.4	14.2	6.1	10	9.8	2.0
Mass Contingency (%)	30	30	30	30	30	30	30	30
Mass with Contingency kg	32.5	1.3	12.3	18.5	7.9	13.0	12.7	2.6
Average Payload Power Without Contingency (W)	11	2	4.6	12.0	8.6	0	25.8	1.9
Average Payload Power Contingency (%)	30	30	30	30	30	30	30	30
Average Payload Power with Contingency (W)	14.3	2.6	6.0	15.6	11.2	0.0	33.5	2.5
Average Science Data Rate Without Contingency (kbps)	17	8	9.4	1.5	60	0.0	9.4	1
Average Science Data Rate Contingency (%)	30	30	30	30	30	30	30	30
Average Science Data Rate with Contingency (kbps)	22.1	10.4	12.2	2.0	78	0.0	12.2	1.3
Fields of View (if appropriate) (degrees)	N/A	N/A	N/A	N/A	LMSS-Plasma: 360° x 120°	180	Panoramic: 80 degree diagonal WFOV Static 40 Degree diagonal FOV.	N/A
Pointing Requirements (knowledge) degrees	N/A	N/A	N/A	N/A	N/A	10	N/A	N/A
Pointing Requirements (control) degrees	N/A	N/A	N/A	N/A	N/A	10	N/A	N/A
Pointing Requirements (stability) deg/sec	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A

Short Period Seismometer (SP) (developed by Imperial College of London and Oxford University, UK, and Kinemetrics Inc., US): The InSight SP sensor-head consists of a micromachined silicon sensor 25 mm on a side and front-end electronics (FEE) that have been operating continuously on Mars since November 2018. A three-axis system of these sensor heads and associated magnets can be packaged in a 5 cm X 6 cm X 6 cm box. To achieve a broad frequency response, it also uses a force feedback board (FBB). The FBB demodulates the signal received from the capacitive position-sensors on the seismometer proof mass and uses this output to drive the sensor coils and maintain the proof mass at a null point. This signal also constitutes the velocity outputs from the sensors. The velocity output from the FBB is then recorded in a back end electronics system that can be built to be common between all seismic systems on each LGN Lander. This instrument is currently TRL ~7 for the Moon.

VBB and SP have direct heritage from the InSight SEIS instrument (Lognonné *et al.*, 2019). The two major differences for VBB are a larger proof mass (249g vs. 190g), increasing the pendulum period to 5 sec, and a larger displacement transducer voltage, decreasing its electronic noise. No changes are necessary for other VBB parts, including mechanisms, and integration in a vacuum Earth-sealed sphere, which was necessary for Mars, is not required for the lunar environment. VBB and SP feedback and the back-end electronics are inherited from SEIS-InSight and from the Europa Seismic package ICEE-2 funded effort (Kedar *et al.*, 2016).

Short Period (SP) Silicon Audio Buried Seismometer (SPSAB) to study noise reduction below the lunar surface (developed by the University of Arizona, Silicon Audio, and Honeybee Robotics, US): The instrument deploys a Silicon Audio (SiA) Ultra-Low Noise seismometers in a three-axis configuration housed in a 50-mm diameter borehole sonde to study seismic activity on the Moon. The instrument's deployment mechanism allows for integration flexibility as it can be mounted to the lander belly pan or a leg. The sonde instrument, with seismic sensors, is deployed 0.7 m into the lunar regolith by a tube deployed from a reel system consisting of interlocking stainless steel strips. The sonde advances by a combination of deployment mechanism and pneumatic jets at the nozzle that drill and displace the regolith back up to the surface of the Moon as it advances. To minimize vibrations from the lander, the deployment structure retracts while spooling out an electrical umbilical cable to the sonde. The sensor TRL is currently 5 and the burial system is TRL 4. By the end of the DALI program in 2022, the sensor will be TRL 6 and the burial system will be TRL 5.

Heat Flow Probe (HFP) (developed by Texas Tech University and Honeybee Robotics, US): The HFP is designed to penetrate 3 m into the lunar regolith and measures the thermal gradient and thermal conductivity of the depth interval penetrated. Heat flow is obtained as a product of these two measurements. The instrument uses a pneumatic drill to penetrate into the lunar regolith. The instrument's deployment mechanism allows for integration flexibility as it can be mounted to a Lander leg, spools out a boom, made of Kapton and glass fiber composite in a manner similar to a steel tape measure. Once spooled out, the boom forms a cylinder (for mechanical strength) with a penetrating cone at its leading end. The cone advances by discharging gas jets at its tip and blowing away regolith particles, while the boom actuator pushes the cone down (Zacny *et al.*, 2013). Every site chosen will have loose regolith that will be verified by LROC imagery. Two LISTER (Lunar Instrumentation for Subsurface Thermal Exploration with Rapidity; Nagihara *et al.*, 2019) heat flow probes are deployed per Lander. It has a current TRL for the Moon of ~5 and will be flown on a CLPS Lander to Mare Crisium in late 2022/early 2023.

Lunar Magnetotelluric Sounding Suite (LMSS) (developed by Southwest Research Institute, the Heliospace Corporation, GSFC, and the University of California, Berkeley, US): The LMSS instrument suite is comprised of the Lunar Magnetotelluric Sounder (LMS; **Figure 3**) and two supplementary instruments: a search coil magnetometer (SCM) and an ion electrostatic analyzer (ESA). LMS determines the electrical-conductivity structure of the lunar interior from low-frequency magnetic and electric-field measurements. The fluxgate magnetometer is identical to the MAVEN MAG instrument. The electrometer design derives from the THEMIS EFI instrument. These sensors are integrated into LMSS using electronics based on MSL RAD. From a prior program, the magnetotelluric instrument integrated from these subsystems is at TRL 6 for Europa. LMS will achieve TRL 9 in late 2022/early 2023 during flight operations to the Moon's surface under CLPS. The SCM may improve performance at the highest frequencies in the band and is the recently-flown DSX TASC (TRL 8). The ESA provides particle measurements complementary to the electromagnetic fields. It is a new compact design (TRL 4-5) that draws high heritage components from a range of instruments including Wind 3DP, THEMIS ESA, MAVEN SWIA, and Parker Solar Probe SPAN. Final development of LMSS to TRL 6 prior to LGN CDR would require only demonstration of the ESA and updated electronics to include the SCM.

Next Generation Lunar Retroreflector (NGLR) (developed by the University of Maryland in collaboration with INFN, Italy): The NGLR consists of a large (100 mm), single Corner Cube Retroreflector (CCR), the housing, mounting structure and a pointing actuator (Currie *et al.*, 2013). Each LGN Lander will carry two NGLRs – one mounted on the deck of the Lander and one placed on the lander leg near the lunar surface. Once deployed and the dust cover removed, the reflector must be pointed (two angles) towards Earth at the center of the lunar libration pattern with an accuracy of ~1 degree. This pointing is achieved using a dedicated gimbal holding the mechanical housing of the NGLR. This payload is currently TRL 7 for the Moon and will be flown on a CLPS Lander to Mare Crisium in late 2022/early 2023.

Cameras: Each Lander will have one panoramic camera and four static cameras. These have heritage from the Mars rover missions. The panoramic camera is used to get a context image of each landing site and check LMSS deployment. The four static cameras are used to verify VBB-SP and Si-Audio seismometer deployments, as well as those of the heat flow probes and laser NGLR on the lunar surface.

3.2 Fight System

The LGN flight system consist of an Orbiter and four identical Landers to support the baseline mission requirements. The mission requirements are shown in **Table 6**. Key requirements that drive the mission are the continuous operation of the instruments, minimizing Lander generated vibrations, widely spaced landing sites that include a Lander on the farside and the need to maintain instruments and Lander electronics within operating temperatures while exposed to the thermal environment of the Moon.

Continuous operation of instruments

The continuous operation of the instruments drives the power subsystem and in particular the mass of the batteries. For each watt of power needed during the lunar night, ~10.0 kg of batteries are required. To minimize lunar night power usage, the instruments provide a low power mode for the lunar night. The Lander thermal design uses louvers that trap the daytime heat for use at night in the thermal enclosure to keep electronics boxes above their cold temperature limits using their radiated heat without the need for heaters. Deployed instruments have heaters built into them and the power needed is accounted for in the instrument power usage. The Lander CDH subsystem was designed to have minimal functionality during the lunar night and to throttle the processor used to collect science data. The communication system is in standby during the lunar night and data collected are recorded and stored for transmission at the start of the next lunar day. The Lander subsystems and instruments achieve a total maximum expected value for night power usage of 40.8 W.

Minimizing Lander-generated vibrations

The need to minimize Lander-generated vibrations drives the use of power sources that do not produce vibrations. The solar array concept has oversized fixed arrays to eliminate the need for solar array gimbals. The use of DTE X-band for the nearside Landers minimizes the use of the HGA gimbal during normal operations. The farside Lander needs to use the HGA to talk to the Orbiter. This occurs on average 4 days a month for an average of 40 contacts that last between 10-20 minutes. In addition, the Lander itself will expand and contract due to the changes in the thermal environment. To address these generated vibrations, a SP Seismometer is placed on the Lander deck to measure and calibrate out the vibrations.

Widely spaced landing sites

Widely spaced landing sites that include a Lander on the farside, drive the need to minimize propulsion hardware, provide a communication link, and provide a terrain navigation system for landing. A common Lander design provides flexibility in landing site selection during operations.

Maintaining instruments and Lander electronics within operating temperatures

Maintaining instruments and Lander electronics within operating temperatures while exposed to the thermal environment of the Moon drives the need to place electronics in common compartments, provide a radiator and louvers and vent any remaining propellant immediately after landing.

Table 6: Mission Requirements.

Mission Requirements Top Level	
• Mission Lifetime of 6 years (10 year goal) • Minimize Lander generated vibrations • 4 Landers at 4 widely spaced landing sites including one on the farside 1. PKT (P60) Lat: 20.7°, Lon: -47.4° 2. Schickard basin Lat: -44.3°, Lon: -55.1° 3. Crisium basin Lat: 18.5°, Lon: 61.8° 4. Korolev basin Lat: -2.4°, Lon: -159.3°	• Landed orientation such that NGLRs are oriented within 10 deg azimuth of Earth-Moon vector. • Reliability Category 2, Class B • Operate Instruments Data Collection continuously
Mission Design Requirements	
• Launch 8/30/2030 • Landers land at lunar dawn • Falcon 9 Heavy with 5m fairing	• 250 km lunar circular polar orbit for Orbiter plus Landers • Less than 1 m/s velocity at 1 m above surface • Falcon 9 Heavy provides TLI (C3 $\approx$ -2.5 km^2/s^2)
Orbiter Requirements	
• Provide ΔV of 850 m/s to achieve 250 km circular orbit • 3-axis Attitude control 180 arcsec (3-sigma) • Data Storage 4 Tbits • Provide relay return link services for all science and TLM data for up to 4 Landers from the Lunar Surface back to Earth. • X-band uplink for up to 4 Landers from the Lunar surface. <u>Landers</u> • Antenna pointing control of 1.7 mrad 1-sigma • 1 pps signal with 10^{-6} stability relative to ground station • Provide trickle charge for up to 4 Landers during the cruise to the Moon • Decrypt commands	• Maintain a 250 km Circular Polar Orbit for at least 6 years • Attitude knowledge Roll 60 arcsec, Pitch and Yaw 30 arcsec (3-sigma) • Return 574.8 Gbits total from all 4 Landers per lunar day • X-Band and Ka-Band downlink to Earth • Provide relay forward link services for all Lander Commands for up to 4 Landers • LVDS data interface with magnetometer • Mechanically support a stack configuration at launch for up to 4 Landers • 28 V power System • Encrypt downlink
Lander Requirements	
• ≥1 m/s separation velocity from Orbiter • Provide Delta V of 1864 m/s for landing • Land with a velocity of ≤ 0.5 m/s vertical and ≤ 0.1 m/s horizontal at 5 m above the surface • Land Safely with clearance between surface and lower deck of at least 0.5m boulder • Lander final orientation relative to gravity (nadir) < 5 deg • Landed orientation such that Laser Reflectometers and Heat Flow Probes are oriented within in 10 deg azimuth of Earth Moon vector • Lander final position knowledge within 2 km • Operate Instrument Data Collection continuously • Minimize Lander generated vibrations • Data Storage 1 Tbits • Return 574.8 Gbits total from all 4 Landers per lunar day • Provide thermal control for all instruments attached to Lander • Deliver 100 kg of science instruments to lunar surface • LVDS data interface with instruments	• Provide 90 W electrical power to the science instruments during the Lunar day and 30 W during the Lunar night. • 28 V non regulated power System • Decrypt commands • Encrypt downlink • 0.1 ms timing accuracy with 10^{-6} stability relative to ground station <u>Deploy Instruments</u> • Panoramic Camera with unobstructed FOV • Search Coil Magnetometer • VBB under Lander • Mount Lunar NGLR on Deck and 1 Leg facing Earth • Mount SP Seismometer on deck and deploy 1 on surface • Mount LMSS ESA on deck • Mount LMS Deployment mechanism on Deck on each of 4 sides with 180 deg FOV • Mount Si Audio MEMS Seismometer on one Lander leg • Mount HFP on two opposite legs
Ground System Requirements	
• 34m DSN Antenna • Receive Orbiter and Landers engineering & science data telemetry • Encrypt commands • Decrypt downlink • Provide commanding • Record/Archive science data	• Provide critical event telecom coverage: Launch Sep, S/A Deployment, Instrument Deployments • DDOR • Science Data Center • Science Operations Center • Mission Operations Center
Operations Requirements	
• Implement required DDOR • Manage time correlations • Maneuvers • Support DSN passes • Monitor Orbiter and Landers state of health	• Implement contingency procedures • Implement science sequences • Manage Orbiter and Lander operations • Perform ops sim testing

3.2.1 Lander Concept

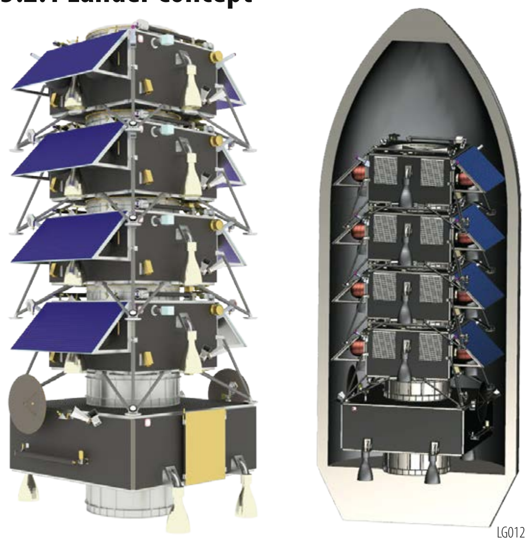


Figure 4: Stowed LGN Orbiter and Landers inside the Falcon 9 Heavy 5m fairing.

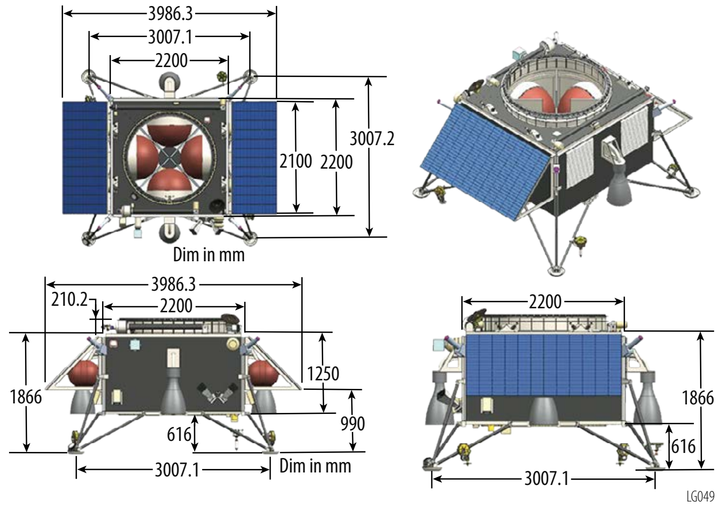


Figure 5: Lander Stowed Views and Dimensions.

Each Lander is 1.8 meters tall and 3.15 meters square. LGN Orbiter and Landers fit within the Falcon 9 heavy 5m fairing as shown in **Figure 4**. The diameter of the fairing is within scope of standard 5m fairings used on the Delta IV, Atlas and baselined for the Block 1 SLS but the height is significantly smaller (see **Appendix B, Figure B-21**). Baselining the smaller fairing ensures that, from a packaging standpoint, a broad spectrum of fairings are viable. **Section 2.1** in **Appendix B** provides details on the Lander design and concept. **Figure 5** shows the Lander concept. **Table 7** provides the mass and power of the Lander. The solar array size was calculated for the highest planned latitude of 44.3° with 30% margin on the loads and 30% margin on the solar array area. **Appendix B**, **Section 2.1.7** discuss the Lander power subsystem.

Table 7: Lander Mass and Power for Day and Night Operations.

	Mass			Average Power (Day)			Average Power (Night)		
Lander	**CBE (kg)**	**% Cont.**	**MEV (kg)**	**CBE (W)**	**% Cont.**	**MEV (W)**	**CBE (W)**	**% Cont.**	**MEV (W)**
Structures & Mechanisms	381.8	20	452.3	0	0	0	0	0	0
Thermal Control	29.9	10	32.9	0	0	0	0	0	0
Propulsion (Dry Mass)	188.1	10	206.9	0	0	0	0	0	0
Attitude Control	12.7	10	13.9	0	10	0	0	0	0
Avionics	23.1	10	25.4	52	10	57.2	15.6	30	20.3
Telecommunications	26.5	10	29.2	5	10	5.5	5	10	5.5
Power System	366.9	10	403.5	5	10	5.5	1	10	1.1
Total Lander	**1,105.5**	**11.4**	**1,263.6**	**62.0**	**10**	**68.2**	**21.6**	**20**	**26.9**

3.2.2 Orbiter Concept

The Orbiter serves as a carrier/delivery system for the Landers, communication relay and a limited science platform. Utilizing the Orbiter as the carrier/delivery system minimizes the mass for each Lander since the Landers propulsion systems can be sized for landing only and do not have to include additional capability to perform the Lunar Orbit Insertion (LOI) individually. The Orbiter performs the necessary LOI for all Landers and controls and executes the release of the Landers for deployment to the surface. The Orbiter carries enough propellant to perform the LOI and orbit maintenance for the duration of the mission. The Orbiter science payload consists of a magnetometer on a 2 m boom that operates continuously. **Section 2.2** in **Appendix B** provides details on the Orbiter concept. **Table 8** shows the mass, power and mission data volume for the magnetometer. **Table 6** provides details on the magnetometer. **Figure 4** shows the Orbiter concept with the Landers. **Figure 6** shows the deployed Orbiter. **Table 9** provides the mass and power of the Orbiter.

Table 8: Orbiter Payload Mass and Power Table.

	Mass			Average Power			Mission Data Volume
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)	6 Years (Gbits)
Magnetometer with Boom	2	30	2.6	1.9	3	2.5	189

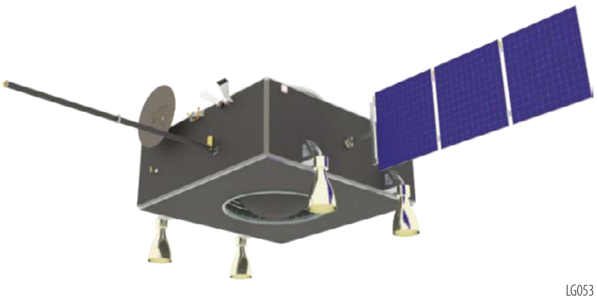


Figure 6: LGN Orbiter communication relay deployed configuration.

Table 9: LGN Orbiter mass and power.

Orbiter	Mass			Average Power (Day)		
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
Structures & Mechanisms	629.6	30	810.9	20	0	20
Thermal Control	34	10	37.4	129.0	10	141.9
Propulsion (Dry Mass)	279.0	10	306.9	0	0	0
Attitude Control	84.8	10	93.2	35.7	10	39.2
Avionics	38.9	10	42.8	58.1	10	63.9
Telecommunications	50.0	10	55.0	256.0	10	281.6
Power System	56.2	10	63.8	5.0	10	5.5
Total Orbiter	**1,174.5**	**20**	**1,386.7**	**503.8**	**10**	**554.1**

Table 10: LGN Flight Element Characteristics.

Flight System Element Parameter	Landers	Orbiter
General		
Design Life	6 years	6 years
Structure		
Structures Material	Aluminum, Composite, Titanium	Aluminum, Composite, Titanium
Number of Articulated Structures	3	2
Number of Deployed Structures	22	17
Aeroshell Diameter, m	N/A	N/A
Thermal Control		
Thermal Control, type	Louvers/Radiators, Heat Pipes	Radiators, Heat Pipes
Propulsion		
Estimated ΔV budget, m/s	1,864 m/s	1,506 m/s
Propulsion Type(s) and Associated Propellant(s)/Oxidizer(s)	Regulated Bipropellant, MMH, NTO	Regulated Bipropellant, MMH, NTO
Number of Thrusters and Tanks	4 Main Engines 8 ACS Engines 2 MMH Tanks 2 NTO Tanks 2 Pressurant Tanks	4 Main Engines 16 ACS Engines 1 MMH Tank 4 NTO Tanks 2 Pressurant Tanks
Specific Impulse of Each Propulsion Mode, seconds	293	293
Attitude Control		
Control Method (3-axis, spinner, grav-gradient, etc.)	3-axis	3-axis
Control Reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	inertial	inertial
Attitude Control Capability, arcseconds	50	50
Attitude Knowledge Limit, arcseconds	6	6
Agility Requirements (maneuvers, scanning, etc.)	Terminal velocity <0.5 m/s	N/A
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	High Gain Antenna	2 High Gain Antennas, 2 single-axis S/A gimbal
Sensor and Actuator Information (precision/errors, torque, momentum storage capabilities, etc.)	0.35 deg sun sensors, 50 arcsec star scanners, 0.005 deg/hr MIMU	0.35 deg sun sensors, 50 arcsec star scanners, 0.005 deg/hr MIMU
Avionics		
Flight Element Housekeeping Data Rate, kbps	2,000	2,000

Data Storage Capacity, Mbits	1,000,000	4,000,000
Maximum Storage Record Rate, kbps	1,000	1,000
Maximum Storage Playback Rate, kbps	100,000	100,000
Power		
Type of Array Structure (rigid, flexible, body mounted, deployed, articulated)	Body mounted	Single axis
Array Size, square meters	7.6	3.5
Solar Cell Type (Si, GaAs, Multi-junction GaAs, concentrators)	TJGaAs	TJGaAs
Expected Power Generation at Beginning of Life (BOL) and End of Life (EOL), Watts	800W BOL, 700W EOL	837W BOL, 744W EOL
On-orbit Average Power Consumption, Watts	N/A	554.1
Lunar Day Power Consumption, Watts	162.4	N/A
Lunar night Power Consumption, Watts	40.8	N/A
Battery Type (NiCd, NiH, Li-ion)	Li-ion	Li-ion
Battery Storage Capacity, amp-hours	1,100	22

3.3 Concept of Operations and Mission Design

A summary of the LGN mission design is shown in **Table 11**. The details of the mission design is provided in **Appendix B**.

Table 11: LGN Mission Design Summary.

Parameter	Orbiter with 3 Landers	Orbiter with 4 Landers	Lander	Unit
Orbit Parameters (apogee, perigee, inclination, etc.)	250 circular polar	250 circular polar	0	km
Mission Lifetime	6	6	6	yrs
Maximum Eclipse Period	66 minutes	66 minutes	14.5 days	
Launch Site	Cape Canaveral, FL	Cape Canaveral, FL	Cape Canaveral, FL	
Total wet Mass with Contingency (includes instruments)	5,629.1	6,683.1	2,551.9	kg
Propellant Mass Without Contingency	3,818.2	4,766.8	1,159.5	kg
Propellant Contingency	10	10	10	%
Propellant Mass with Contingency	4,242.4	5,296.4	1,288.3	kg
Orbiter Launch Adapter Mass with Contingency (sized for 4 landers)	97.4	97.4	N/A	kg
Total dry Launch Mass	5,177.6	6,441.3	1,263.6	kg
Total wet Launch Mass	13,382.3	16,988.3	2,551.9	kg
Launch Vehicle	Falcon 9 Heavy	Falcon 9 Heavy	N/A	Type
Launch Vehicle Lift Capability	15,500	15,500	N/A	kg
Launch Vehicle dry Mass Margin	10,322.4	9,058.7	N/A	kg
Launch Vehicle dry Mass Margin (%)	199.4	140.6	N/A	%
Launch Vehicle wet Mass Margin	2,117.7	-1,488.3	N/A	kg
Launch Vehicle wet Mass Margin (%)	15.8	-8.8	N/A	%

3.3.1 Launch Operations

The LGN mission launches from Cape Canaveral, Florida on a single Falcon 9 Heavy vehicle with 5m fairing as shown in **Figure 7**. The baseline mission is to launch four Landers, however since the Falcon 9 Heavy launch mass performance can only accommodate three Landers based on the current mass properties (**Table 11**), a secondary launch will be required for the fourth Lander. Another current launch vehicle that could accommodate four Landers in the current configuration is the Space Launch System (SLS) Block I Cargo. For now, which approach is pursued for launching the baseline four lander version of the mission will depend on future work. The mission concept assumes that a solution is found and that the launch vehicle upper stage places LGN on a direct transfer to the Moon.The launch vehicle upper stage performs the Trans-Lunar Injection (TLI) maneuver, placing LGN on a direct trajectory to the Moon. The transfer time to the Moon is a function of lunar phase and varies between four and five days. There are two launch opportunities for a minimum energy direct transfer to the Moon per day, namely, a short coast and a long coast. The two solutions achieve the lunar transfer in two different orbit planes and differ in launch time as well as coast time. Additional constraints (*e.g.*, total eclipse time during the transfer phase) lead to an obvious choice between the long and short coast solutions. As an example, the Lunar Reconnaissance Orbiter (LRO) mission, which utilized a direct transfer to achieve its polar orbit,

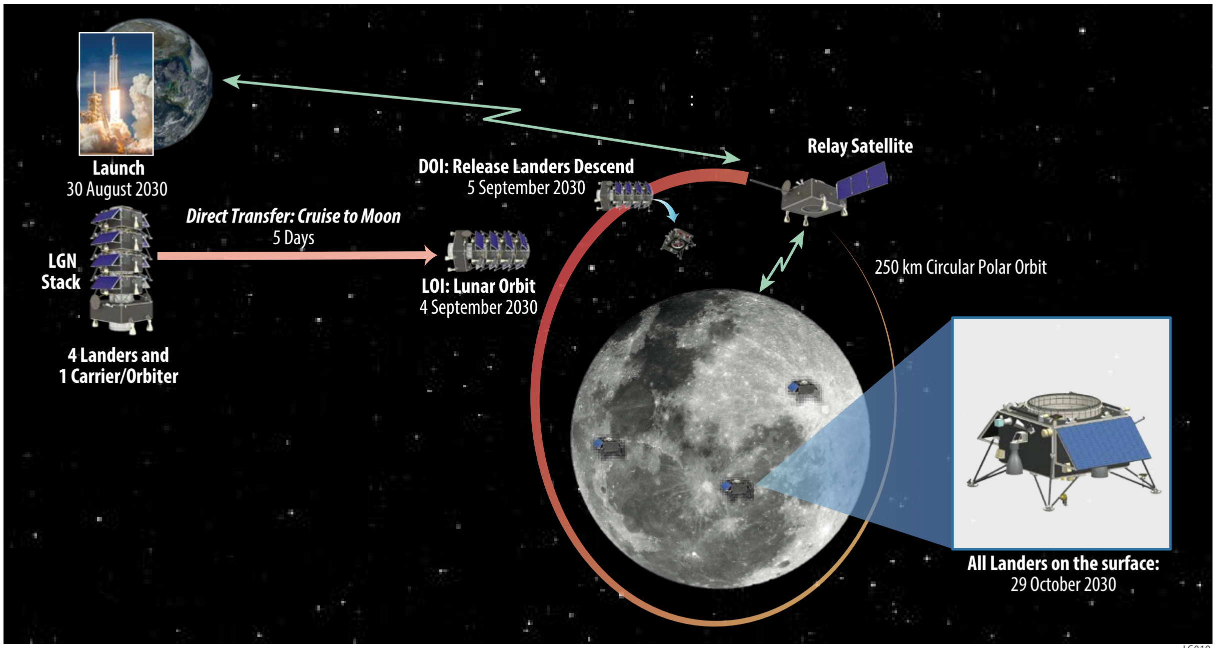


Figure 7: An overview of the LGN mission con-ops including launch, cruise, descent, and landing.

had a launch window of three days every two weeks. The LGN Orbiter provides navigation, power, communication and attitude control during the lunar transfer period.

3.3.2 Lunar Orbit Operations

Upon arrival at the Moon, the Orbiter will perform the LOI maneuver to place LGN into a 250 km circular polar orbit around the Moon. The LOI maneuver costs about 865 m/s in terms of Delta-V, while the circularization burn is approximately 40 m/s, bringing the total transfer cost to 905 m/s. Once in the 250 km circular polar orbit, the Orbiter will be commanded by the flight ops team to release each Lander so that it lands at its site synchronized with the start of the lunar daylight cycle (dawn). Each Lander will be deployed sequentially (spaced out ~30 earth days) and they each will land shortly after lunar dawn (9:00am local time) at the given landing site. Once landing has been verified and the instruments successfully deployed, the next Lander will be sent to the surface. The current baseline will have the Landers deployed in the following order: PKT (P-60 young basalt field) site on the nearside, Schickard site on the nearside, Korolev site on the farside, and Crisium site on the nearside (**Figure 2**).

3.3.3 Landing Operations

Each Lander, will perform a De-Orbit Insertion (DOI) maneuver to lower the Lander orbit periapsis. A Braking Maneuver (BM) followed by Terrain Relative Navigation (TRN) will bring the spacecraft down to an attitude of 100 m above the landing site with a velocity magnitude of 8.2 m/s. This scenario is depicted in **Figure 8**. TRN is discussed in more detail in **Appendix B**, **Section 2.1.5**. At this stage, the spacecraft is expected to perform its final descent (**Figure 9**) to land with a terminal velocity of less than or equal to 0.5 m/s at 1 m above the surface. Once landed, the Lander will vent any remaining propellant through the ACS thrusters to prevent freezing of the propellant. The entire landing operation, from the start of the DOI to touchdown, takes a little over one hour. Additional details are provided in **Appendix B**.

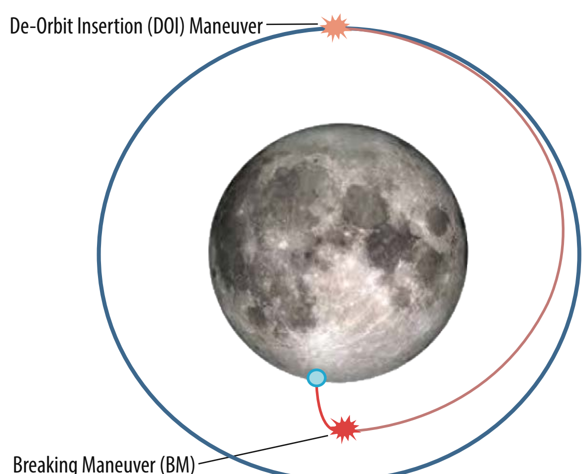


Figure 8: Each Lander performs a BM and DOI to land on the surface of the Moon.

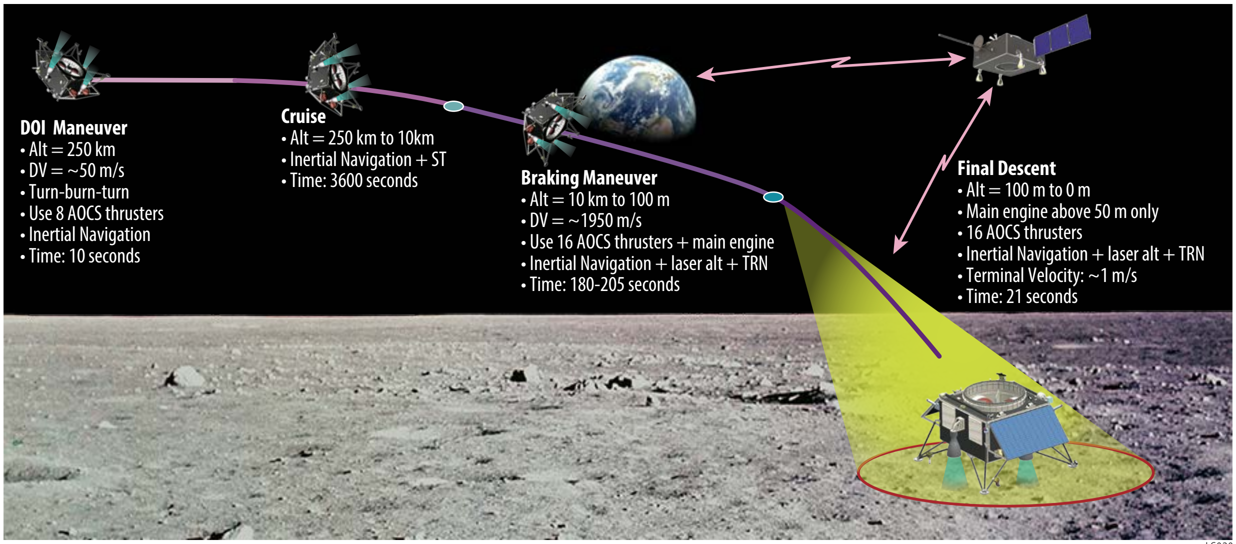


Figure 9: The LGN descent and landing concept is designed to safely deliver the Landers to the lunar surface. Communications are shown between the farside Lander and the Orbiter. For nearside Landers, communication direct to Earth.

3.3.4 Deployment Operations

Once the Lander has landed on the lunar surface, a five-day deployment and checkout period (**Table 12**) will begin. Following completion of the post-landing checkout, instrument commissioning and initial data collection will begin. The Lander conditioning and instrument deployment concept of operations (con-ops) is estimated to take ~50 hours. Estimated daylight at each landing site ~336 hours (14.5 days) providing sufficient margin should any anomalies occur prior to the Lander entering the lunar night. Throughout the duration of the deployment, instrument commissioning and surface operations phases, all Landers will have sufficient contact time with the Earth via DSN and the relay Orbiter. Each nearside Lander will initially have 24-hour coverage by DSN during landing and checkout. Once initial data collection begins, each nearside Lander will have approximately a one-hour duration contact with the DSN each Earth day during the lunar day . After the first 14.5 days have passed, each Lander will enter lunar night for 14.5 days.

The farside Lander uses the HGA to communicate with the Orbiter which in turn communicates with DSN. The Orbiter is in contact with the farside Lander on average 4 days a month for an average of 40 contacts that last between 10-20 minutes. The Orbiter is in contact with DSN on a daily basis with 3-4 contacts per day averaging 128 minutes each.

Table 12: The LGN deployment concept for Instruments and Lander.

	Instrument/Ops	Deployment	Duration
1	SP Seismometer on Deck	• None: Activated on Lander deck	1 min
2	Lander Deployments and Ops	• Deploy the High Gain Antenna – Critical Event • Health and safety checkout	3 hrs
3	Vent Remaining Propellant	• Venting of any excess fuel in the tanks to reduce Lander-induced seismic noise	1 hr
4	Deploy Panoramic mast camera	• Take & transmit context panoramic image • Activate DVR on panoramic imager	1 hr
5	Science Operations Committee (SOC) evaluation of deployment	• Activate Engineering Camera 1 on Lander leg • Take & transmit image of site prior to drilling	1 hr
6	Heat Flow Probe 1 deployment to 3 meters below surface	• Pneumatic Deployment system	12 hrs
7	SOC deployment evaluation	• Activate Engineering Camera 1 on Lander leg • Take & transmit image of site post-deployment	1 hr
8	SOC deployment evaluation	• Activate Engineering Camera 2 on Lander leg • Take & transmit image of site prior to drilling	1 hr
9	Deployment of buried SP seismometer to 0.7 meters	• Pneumatic (deployment 3 axis Silicon Audio sensors and 3 MEMS sensors)	3 hrs

10	SOC deployment evaluation	• Activate Engineering Camera 2 on Lander leg • Take & transmit image of site post-deployment	1hr
11	Science Operations Committee (SOC) evaluation of deployment/take	• Activate Engineering Camera 3 on Lander leg • Take & transmit image of site prior to drilling	1 hr
12	Heat Flow Probe 2 deployment to 3 meters below surface	• Pneumatic Deployment system	12 hrs
13	SOC deployment evaluation	• Activate Engineering Camera 3 on Lander leg • Take & transmit image of site post-deployment	1 hr
14	SOC deployment evaluation	• Activate Camera below landing deck; • Take & transmit image of site pre-VBB–SP deployment	1 hr
15	Deployment of VBB-SP package on the lunar surface beneath the center of Lander	• Lowered on crane cable below the Lander	3 hrs
16	SOC deployment evaluation	• Activate Engineering Camera below Lander deck • Take & transmit image of site post-deployment	1 hr
17	Deployment of LMSS mast magnetometer	• MAG deploy via mast on Lander deck	1 hr
18	Deployment of LMSS electrodes	• EM sensors Deployed to surface via ballistic mechanisms	1 hr
19	SOC deployment evaluation	• Activate PanCam to take & transmit images of Lander deck and lunar surface site post-deployment	1hr
20	Deployment of LMSS-SCM	• Search coil deployment via boom edge of deck	1 hr
21	Deployment of LMSS Electrostatic Analyzer	• Eject protective cover (sensor is mounted directly to the deck)	1 hr
22	Deployment of NGLR on Lander deck	• Deployed on the Lander deck w/ Gimbal cover mechanism release & pointing at Earth	1 hr
23	SOC deployment evaluation	• None. Determined by engineering data.	1 hr
24	SOC deployment evaluation	• Activate Engineering Camera 4 on Lander leg; • Take image of site post-deployment	1 hr
25	Deployment of NGLR on lunar surface	• Deployed on Lander leg near the surface	1 hr

3.3.5 Surface Operations (Day and Night)

During the lunar day, all science instruments are operating at full science modes. All data collected from each Lander on the nearside will be transmitted back to the MOC via the DTE communication links, the farside Lander will uplink all instrument data to the dedicated Orbiter for relay back to Earth. The Lander will utilize solar array power for the majority of the day operations, the batteries will re-charge in parallel during normal operations at some point during the 14-day period before entering the next lunar night cycle. The SOC will command the instruments at the end of the lunar day cycle to the appropriate modes for night operations. The SOC will verify health and safety of all instruments prior to going into the lunar night.

During the lunar night the science instruments operate at reduced power and data collection frequency. The Lander subsystems also enter a reduced power mode. All Landers will be equipped with X-band communication subsystems and will transmit all science data either to the Earth, or in the case of the farside Lander, to the Orbiter during the sunlight portion of the lunar day. By restricting communications to the sunlight portion of the lunar day, the power used during the lunar night is maximized for science data, survival, and thermal needs of the spacecraft and instrument components. Nominal science/surface operations will continue for the next six years. **Figure 10**, depicts the day and night operations for each instrument and key Lander operations.

3.3.6 Orbiter Operations

The Orbiter transitions from carrier/deliver to relay communication satellite after the farside Lander is deployed. The Orbiter will remain in the 250 km circular polar orbit and serve as the primary relay for the farside Lander, and a backup relay for the nearside Landers. Other orbits can be explored to achieve maximum coverage of all four Landers on the surface, and that will guarantee enough bandwidth to support the transfer of all the data before the lunar night approaches. Orbit maintenance will be performed by the Orbiter, mission design analysis shows the Orbiter can maintain its orbit for the life of the mission (details in **Section 1.5.3** in **Appendix B**).

Lunar Day: ~14.5 Earth Days	Instruments/ Data Ops	Lunar Night: ~14.5 Earth Days
Continual Science Measurements	VBB/SP Seismometer	Continual Science Measurements
Continual Science Measurements	SP Seismometer (On Deck)	Continual Science Measurements
Continual Science Measurements	Silicon/Audio MEMS Seis.	Continual Science Measurements
Continual Science Measurements various duty cycles	Heat Flow Probe 1	Night Time Operations Are At Reduced Cycles
Continual Science Measurements various duty cycles	Heat Flow Probe 2	Night Time Operations Are At Reduced Cycles
Continual Science Measurements	LMSS	Night Time Operations Are At Reduced Cycles
Continual Science Measurements (Passive)	NGLR	Continual Science Measurements (Passive)
Continual Science Measurements (Passive)	NGLR	Continual Science Measurements (Passive)
Re-transmit stored night data; Transmit Science Data to Orbiter/DTE (10 Passes per Earth Day)	Transmitter	No Transmission of Science Data
Receive, Process and Store science data from all instruments	C&DH	Receive, Process and Store science data from all instruments

LG021

Figure 10: The surface operations timeline is shown in the diagram above.

3.3.7 Mission Operations

The LGN concept will require mission operations to manage command and control of the Orbiter and four Landers for at least 6 years. The Mission Operations Center (MOC) will manage sustained parallel mission operations, with long-term mission planning, uplink of command loads. Lander automation features, driven by Flight Software (FSW) will allow day-to-day operations to become less complex overtime, upon completion of the deployment phase. This LGN approach simplifies the Flight Ops team operations during lunar transfer, by tracking one S/C vs five individually. The current operational approach is to release one Lander every two weeks (~14 days) from the Orbiter. This will allow operations to focus on one Lander at a time during these mission-critical operations, document lessons learned, update command procedures, and plan for the next Lander release. The mission operations team will utilize DTE communication links to talk to all Landers on the nearside of the Moon. The Orbiter will serve as the primary communication relay for the farside Lander; operations will have the option to communicate with the nearside Landers via the Orbiter relay comm links. **Figure 11** depicts the LGN Ground System architecture.

3.3.8 Communications

The Deep Space Network (DSN) will be used as the primary means for all communications during the cruise to the Moon and for descent/surface operations (**Figure 12**). The nearside Landers will utilize DTE communication links to forward commands to and receive science and housekeeping data from the lunar surface. The nearside Landers are capable of communicating with the Orbiter during normal operations as a backup, contingencies and potentially mission critical events as determined by mission operations. The farside Lander will rely on the Orbiter established relay links throughout the mission for the transmission and receipt of all mission data. Details on the communication subsystem are in **Appendix B**.

Link analyses for the Landers and Orbiter were conducted. Theses link budgets show 10 million symbols per second (msps) margin for Lander and Orbiter links with DSN and an abundance of margin. 32 kilo-bits per second (kbps) Orbiter HGA uplink with ample margin exists. 8 kbps for Orbiter to Lander LGA also is provided. **Table 13** summarizes the communication link analyses.

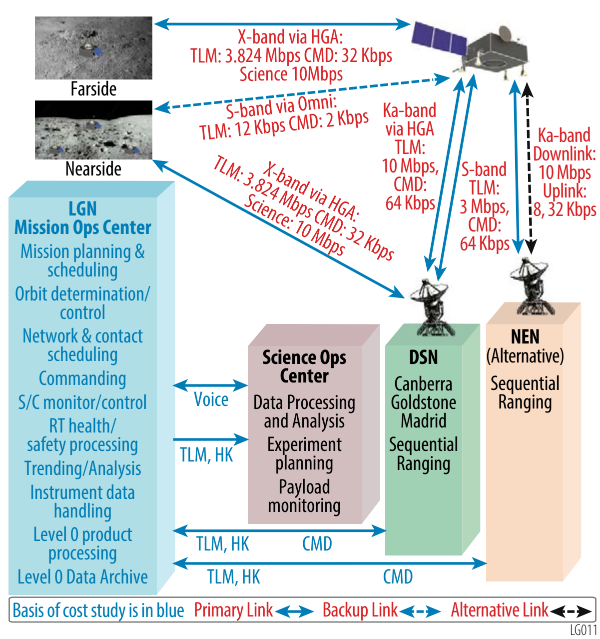


Figure 11: LGN Ground Systems Architecture.

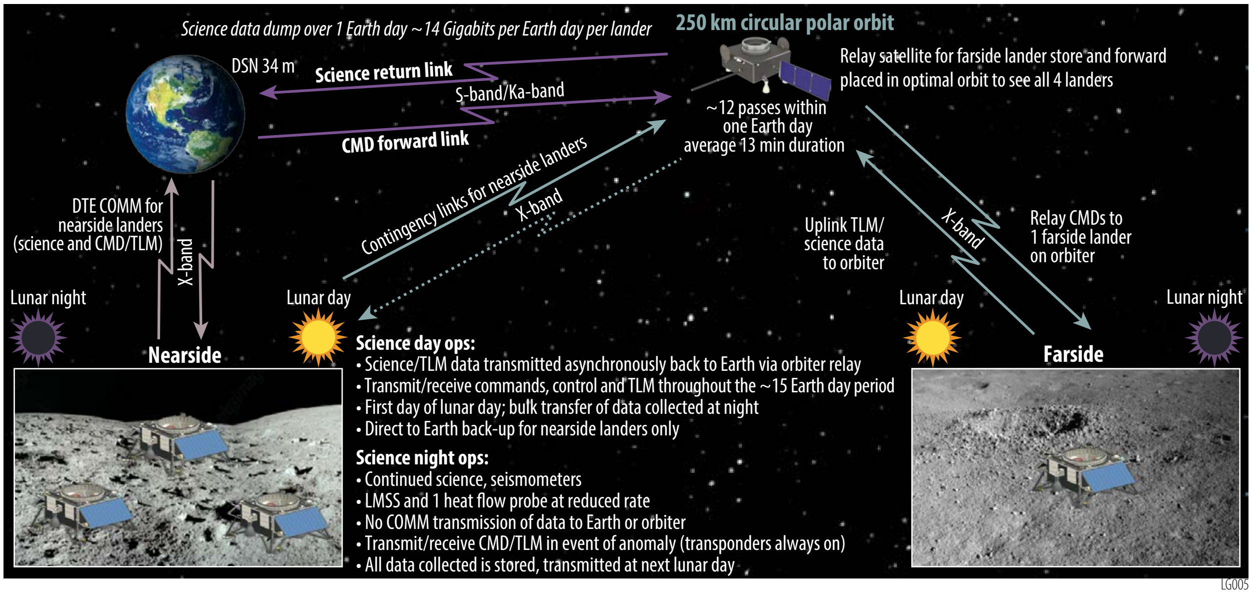


Figure 12: The LGN communication concept provides Direct to Earth communication between the nearside Landers and Lander to Orbiter to Earth for the farside Lander during the Lunar day.

Table 13: LGN Lander RF Communications Subsystem Summary of Link Analyses.

Link Information	Mission Phase 1 Flight Ops	Mission Phase 2 Post-Landing Checkout	Mission Phase 3 Instrument Commissioning	Mission Phase 4 Surface Ops
Number of Weeks for Mission Phase, weeks	2	8	4	311
Downlink Information				
Orbiter HGA to DSN 34m, Ka-Band Link				
Number of Contacts (per we week)	24	24	24	24
Average duration of contacts (minutes)	128	128	128	128
Total Daily Data Volume, per day (required/available)	36 Mb / 76 Gb	6 Gb / 76 Gb	6 Gb / 76 Gb	6 Gb / 76 Gb
Downlink Frequency Band, GHz	26.5	26.5	26.5	26.5
Downlink Telemetry Data Rate, Mbps	10 Mbps	10 Mbps	10 Mbps	10 Mbps
Transmitting Gain(s), dBi	46.27	46.27	46.27	46.27
Transmitting Power Output, Watts	3	3	3	3
Downlink Receiving Antenna Gain, dBi	76.9	76.9	76.9	76.9
Downlink Margin, dB	15.7	15.7	15.7	15.7
Orbiter HGA to DSN 34m, S-Band Link				
Number of Contacts per week	24	24	24	24
Average duration of contacts (minutes)	128	128	128	128
Total Daily Data Volume, per day (required/available)	36 Mb / 23 Gb	6 Gb / 23 Gb	6 Gb / 23 Gb	6 Gb / 23 Gb
Downlink Frequency Band, GHz	2.3	2.3	2.3	2.3
Downlink Telemetry Data Rate, Mbps	3	3	3	3
Transmitting Gain(s), dBi	24.99	24.99	24.99	24.99
Transmitting Power Output, Watts	8	8	8	8
Downlink Receiving Antenna Gain, dBi	34	34	34	34
Downlink Margin, dB	3.9	3.9	3.9	3.9
Orbiter LGA to DSN 34m, S-Band Link				
Number of Contacts per week	24	24	24	24
Average duration of contacts (minutes)	128	128	128	128
Total Daily Data Volume, per day (required/available)	17 Mb / 34 Mb	17 Mb / 34 Mb	17 Mb / 34 Mb	17 Mb / 34 Mb
Downlink Frequency Band, GHz	2.3	2.3	2.3	2.3
Downlink Telemetry Data Rate, Mbps	0.004	0.004	0.004	0.004

Transmitting Gain(s), dBi	-2.0	-2.0	-2.0	-2.0
Transmitting Power Output, Watts	8	8	8	8
Downlink Receiving Antenna Gain, dBi	55.62	55.62	55.62	55.62
Downlink Margin, dB	5.5	5.5	5.5	5.5
Lander HGA to DSN 34m, X-Band Link				
Number of Contacts per week	N/A	7	7	7
Average duration of contacts (hours)	N/A	1	1	1
Total Daily Data Volume, per day (required/available)	N/A	19 Gb / 10 Tb	19 Gb / 10 Tb	19 Gb / 10 Tb lunar day
Downlink Frequency Band, GHz	8.4	8.4	8.4	8.4
Downlink Telemetry Data Rate, Mbps	10	10	10	10
Transmitting Gain(s), dBi	30.3	30.3	30.3	30.3
Transmitting Power Output, Watts	17	17	17	17
Downlink Receiving Antenna Gain, dBi	34	34	34	34
Downlink Margin, dB	6.26	6.26	6.26	6.26
Lander LGA to DSN 34m, X-Band Link				
Number of Contacts per week	N/A	7	7	7
Average duration of contacts (hours)	N/A	1	1	1
Total Daily Data Volume, per day (required/available)	N/A	17 Gb /34 G/B	17 Gb /34 G/B	17 Gb /34 G/B
Downlink Frequency Band, GHz	8.4	8.4	8.4	8.4
Downlink Telemetry Data Rate, Mbps	0.012	0.012	0.012	0.012
Transmitting Gain(s), dBi	-2.0	-2.0	-2.0	-2.0
Transmitting Power Output, Watts	17	17	17	17
Downlink Receiving Antenna Gain, dBi	34	34	34	34
Downlink Margin, dB	3.2	3.2	3.2	3.2
Farside Lander HGA to Orbiter HGA, X-Band Link				
Number of Contacts per week	40	40	40	40
Average duration of contacts	14	14	14	14
Total Daily Data Volume, per day (required/available)	N/A	19 Gb / 336 Gb	19 Gb / 336 Gb	19 Gb / 336 Gb
Downlink Frequency Band, GHz	8.4	8.4	8.4	8.4
Downlink Telemetry Data Rate, Mbps	10	10	10	10
Transmitting Gain(s), dBi	30.3	30.3	30.3	30.3
Transmitting Power Output, Watts	17	17	17	17
Downlink Receiving Antenna Gain, dBi	35.8	35.8	35.8	35.8
Downlink Margin, dB	25.14	25.14	25.14	25.14
Farside Lander LGA to Orbiter LGA, X-Band Link				
Number of Contacts per month	40	40	40	40
Average duration of contacts, minutes	14	14	14	14
Total Daily Data Volume, per day (required/available)	N/A	17 Gb / 34 Gb	17 Gb / 34 Gb	17 Gb / 34 Gb
Downlink Frequency Band, GHz	8.4	8.4	8.4	8.4
Downlink Telemetry Data Rate, Mbps	0.35	0.35	0.35	0.35
Transmitting Gain(s), dBi	-2	-2	-2	-2
Transmitting Power Output, Watts	17	17	17	17
Downlink Receiving Antenna Gain, dBi	35.9	35.9	35.9	35.9
Downlink Margin, dB	7.4	7.4	7.4	7.4
Uplink Information				
DSN 34m to Orbiter HGA, Ka-Band Link				
Number of Uplinks per day	1	1	1	1 during lunar day 0 during lunar night
Uplink Frequency Band, GHz	23.0	23.0	23.0	23.0
Telecommand Data Rate, kbps	64	64	64	64
Receiving Antenna Gain(s), DBi	45.04	45.04	45.04	45.04
DSN 34m to Orbiter HGA, S-Band Link				
Number of Uplinks per day	1	1	1	1
Uplink Frequency Band, GHz	2.1	2.1	2.1	2.1

Telecommand Data Rate, kbps	64	64	64	64
Receiving Antenna Gain(s), DBi	24.3	24.3	24.3	24.3
DSN 34m to Orbiter LGA, S-Band Link				
Number of Uplinks per day	1	1	1	1
Uplink Frequency Band, GHz	2.1	2.1	2.1	2.1
Telecommand Data Rate, kbps	64	64	64	64
Receiving Antenna Gain(s), DBi	-2	-2	-2	-2
DSN 34m to Lander HGA, X-Band Link				
Number of Uplinks per day	1	1	1	1 during lunar day 0 during lunar night
Uplink Frequency Band, GHz	7.2	7.2	7.2	7.2
Telecommand Data Rate, kbps	32	32	32	32
Receiving Antenna Gain(s), DBi	28.9	28.9	28.9	28.9
DSN 34m to Lander LGA, X-Band Link				
Number of Uplinks per day	1	1	1	1 during lunar day 0 during lunar night
Uplink Frequency Band, GHz	7.2	7.2	7.2	7.2
Telecommand Data Rate, kbps	32	32	32	32
Receiving Antenna Type(s) and Gain(s), DBi	-2	-2	-2	-2
Orbiter HGA to Farside Lander HGA, X-Band Link				
Number of Uplinks per day	1	1	1	1
Uplink Frequency Band, GHz	7.2	7.2	7.2	7.2
Telecommand Data Rate, kbps	32	32	32	32
Receiving Antenna Gain(s), DBi	28.93	28.93	28.93	28.93
Orbiter HGA Farside Lander LGA, X-Band Link				
Number of Uplinks per day	1	1	1	1
Uplink Frequency Band, GHz	7.2	7.2	7.2	7.2
Telecommand Data Rate, kbps	8	8	8	8
Receiving Antenna Gain(s), DBi	-2	-2	-2	-2

3.4 Mission Risk List

The LGN team developed a risk register to identify the focus areas for consideration. The graphic in **Figure 13** displays the LGN Top Risk Matrix while all the risks are listed in **Table 14**.

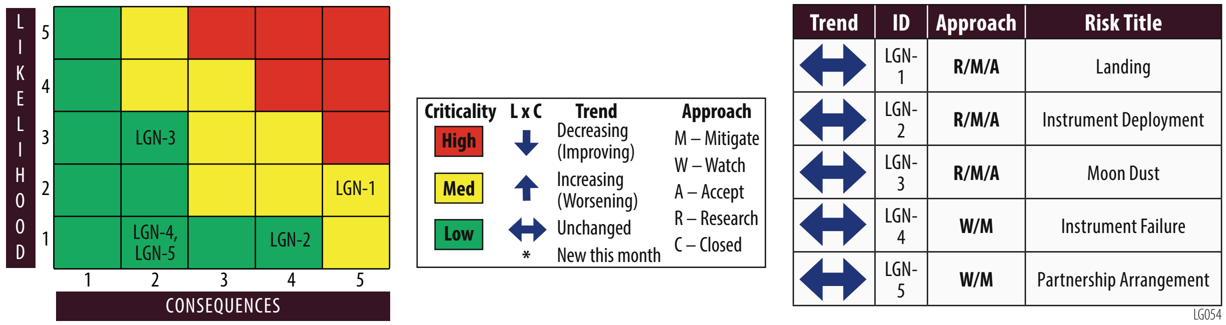


Figure 13: LGN Mission Risk Matrix.

Table 14: LGN Mission and Instrument Risks.

MISSION LEVEL RISKS			
Risk ID & Risk Name	**Risk Statement**	**Approach**	**Status/Comments**
LGN-001 **Landing** 2 x 5 Expected Closure: After Deployment	**Given that:** the LGN mission relies on Landers **There is a possibility of:** • Crash (i.e. Lithobraking) • Hard landing (instrument deployment systems) • Not stable • Wrong orientation (Reflector, radiator efficiency in shade vs sun) • Dust (Antenna, Pan-Cam, SA, retroreflector) • Slope/close to/under Boulder/Crater to deploy an instrument (VBB, LMS) • **Resulting in:** Lander loss, degradation of the science return	**Research/ Mitigate / Accept (R/M/A)**	The LGN project team will: • Carefully select a site with minimum hazard, minimum slope and rock abundance. • Utilize systematic mapping of the moon from LRO images to pre-determine landing sites. • Terrain navigation • Multiple Landers • Use of sun sensor or star tracker • Two Landers provide the threshold mission
LGN-002 **Instrument Deployment** 1 x 4 Expected Closure: After Deployment	**Given that:** LGN assumes complex deployment operations **There is a possibility that:** that not all instruments will be able to deploy on time **Resulting in:** some instruments have to survive the lunar night prior to deployment in the stowed configuration	**Research/ Mitigate/ Accept (R/M/A)**	• Current con-ops have LGN Landers landing at dawn, providing 336 hours to deploy all instruments. • Parallel deployments when possible. • Prioritize deployment operations (Order of operations for deployment). • Two heat probes, if one deploys then the second can wait • No issue for retroreflector provided electronics can be maintained warm. • No issue with the seismometer. • Mechanisms and sensors will survive the lunar night. • Electronics and actuators need to be kept warm. • Ensure rock-free landing sites are chosen (LROC imagery)
LGN-003 **Moon Dust** 3 x 2 (L x C) Expected Closure: After Deployment	**Given that:** the LGN mission employees several Landers on the lunar surface for a long duration network **There is a possibility that:** dust may degrade instrument performance **Resulting in:** degradation of the science return	**Research/ Mitigate/ Accept (R/M/A)**	• LGN Landers deploys one-time use mechanism (no moving parts), solar arrays are fixed • Solar arrays are placed at a fixed angle • Power budget includes dust mitigation
LGN-004 **Instrument Failure Prior the End of the Prime Mission** 1 x 2 (L x C) Expected Closure: After completion of primary mission	**Given that:** the LGN mission is a long duration mission (6 years) **There is a possibility that:** some instrument components may fail degrading performance **Resulting in:** degradation of the science return	**Research/ Mitigate/ Accept (R/M/A)**	• The majority of LGN instruments are high TRL with heritage in space environment • The LGN mission will perform life cycle tests to validate the performance of critical instruments • Instrument test protocols will be developed to minimize the risks of early instrument degradation

LGN-005 Partnership Arrangement (International Team) 1 x 2 (L x C) Expected Closure: By PDR	**Given that:** the LGN project team is an international group consisting of government agencies, academia, and industry partners **There is a possibility that:** miscommunication and miscommunication of requirements and engineering design parameters can occur between instrument and Project teams **Resulting in:** cost and schedule impacts	**Watch/ Mitigate / Learn from previous missions (W/M)**	• The management team will schedule weekly meetings with all partners to minimize the risk of misunderstanding. • A comprehensive integrated schedule with clear milestones and deliverable dates will be developed to help manage the project. • The management team will develop metrics to monitor the timely execution of the LGN project
INSTRUMENT LEVEL RISKS			
LGN-006 VBB Seismometer Changes 2 x 4 (L x C) Expected Closure: By mission PDR	**Given that:** there is significant increase of the VBB mobile mass in order to adapt it to lunar gravity (as compared to Mars for the InSight Mission) there is the need to re-do mechanical analysis and STM tests **There is a possibility that:** failure would imply a significant redesign of the VBB **Resulting in:** significant cost and schedule impacts	**Watch/ Mitigate (W/M)**	• The VBB team is planning to redo the random vibe test using parts left over from the InSight mission • It is planned that the random vibe tests will be completed prior to the end of 2023
LGN-007 VBB Seismometer Performance Verification 2 x 3 Expected Closure: after first month of operations	**Given that:** Seismometer instrument performance cannot be demonstrated during tests due to the Earth environment (gravity), the project will have to launch without successful performances tests and accept the risk **There is a possibility:** to discover what the actual performances of the seismometer once deployed on the Moon is not optimum **Resulting in:** degradation of the science return data	**Research/ Accept (R/A)**	• The VBB team will continue to simulate the seismometer performance based on experimental results, and the performance of the InSight instrument
LGN-008 VBB Seismometer Operational Noise 2 x 3 Expected Closure: By the end of 2023	**Given that:** after landing activities onboard, the Lander may generate noise over expected levels (micro-vibs, thermal, EMC) **There is a possibility of:** performance degradation due to seismometer / Lander interactions **Resulting in**: science return degradation	**Watch/ Mitigate/ Research/ Accept (W/M/R/A)**	• The LGN team will develop thermal (noise) stability model to provide other instruments with requirements such as: – Mechanical isolation requirements – Power stability requirements – Grounding requirements • The LGN team will utilize requirements from the InSight mission.
LGN-009 Heat flow probe burial 2 x 4 (L x C) Expected Closure: after heat flow deployment	**Given that:** during the heat flow probe deployment (burial) encounters buried large rocks **There is a possibility that:** the rock will halt its advance **Resulting in:** science return degradation	**Watch/ Mitigate (W/M)**	• The LGN team will select a landing site that has low likelihood of encountering buried rocks (Arecibo radar returns, LRO rock abundance analysis) • Each Lander will carry two heat flow probes for redundancy • The Apollo samples analysis (Carrier, 2005) suggests that the probably of hitting a rock larger than 4 cm diameter in 1 m of drilling is 0.2%
LGN-0010 Non-commanded Electrodes Deployment 1 x 4 (L x C) Expected Closure: Instrument deployment	**Given that:** the magnetotelluric experiment or LMSS depends on deployed electrodes **There is a possibility of:** Non-commanded deployment of electrodes or mast **Resulting in:** non-ideal electrodes placement and science return degradation	**Watch/ Mitigate (W/M)**	The instrument team will take extra steps in order to avoid non-commanded electrodes deployment such as: • Use flight-proven mechanism • Extra thermal analysis • Blanketing to maintain temperatures <70°C
LGN-0011 Electrode Deployments 2 x 4 (L x C) Expected Closure: Instrument deployment	**Given that:** the magnetotelluric experiment or LMSS deploys 4 electrodes **There is a possibility of:** a number of electrodes may not deploy properly **Resulting in:** placement failure and science return degradation	**Watch/ Mitigate (W/M)**	The instrument team will use flight-proven mechanism. Otherwise: • One electrode failure ->reconstruct 2 spatial components from 3 electrodes • Two electrodes failure -> loss of one spatial component • Three electrodes failure ->loss of one spatial component & loss of data quality due to whole-lander voltage reference • Four electrodes failure -> experiment fails

3.5 Descopes and Resiliency

3.5.1 Instruments

The total mass of instrumentation, deployment mechanisms, and electronics on each Lander is 76.5 kg without margin. This payload represents a risk reduction posture as it includes one very broadband seismometer (VBB), three SP seismometers (one buried, one on the lander deck, and one deployed with the VBB), two heat flow probes, two lunar NGLRs, and Lunar Magnetotelluric Sounding Suite. The LMSS contains two mast-deployed magnetometers, an ion electrostatic analyzer (for plasma measurement), and four electrodes. If non-optimal deployment occurs, science data can still be returned and the mission goal can be achieved. Hence, the Lander instrument complement represents a highly resilient mission profile.

The team has formulated a graceful instrument descope plan that reduces the mass, power, and cost of each Lander while preserving the primary science data that will allow the mission goal, objectives, and investigations to be achieved. This descope strategy will increase the mission risk posture because instrument (and hence data) redundancy is removed (**Table 15**).

Table 15: Instrument Descope Plan.

Descope	Rationale	Savings
Short Period (SP) Silicon Audio Buried Seismometer Experiment	Lowest TRL instrument; other short period seismeters on Lander deck and lunar surface	9.4 kg
LMSS-Plasma Sensor	Needed for a secondary science objective	1.5 kg
LMSS-Search Coil Magnetometer	Improves high frequency data; not mission essential	1.4 kg
Lunar NGLR on Lander Deck	Used to understand effects of Lander expansion and contraction on LRR data	4.5 kg
Heat Flow Probe 1	Data can be obtained from Heat Flow Probe 2 if deployment is normal	5.3 kg
	Total:	**22.1 kg**

** Camera mass is currently bookkept as 28.1 kg and mass reductions are being explored.*

3.5.2 Landers

The **baseline LGN mission** consists of four Landers deployed sequentially to the PKT (P-60 young basalt field) site just south of the Aristarchus Plateau, the Schickard Basin, and the Crisium Basin on the nearside, and in the Korolev Basin on the farside (**Figure 2**). The baseline mission examines seismic signals from known deep moonquake clusters as they pass through the Moon to understand lunar structure, performs magnetotelliruc sounding and measures heat flow from distinct lunar terranes and a variety of crustal thicknesses, and expands the current LRR network. The **threshold LGN mission** requires two Landers: one at the PKT (P-60) site and one in the Schickard basin (**Figure 2**). This preserves deployment in distinct terranes with distinct crustal thicknesses and thermal regimes, uses known deep moonquake clusters (including those on the farside) to explore the deep lunar structure, and expands the current LRR network. Between the baseline and threshold missions a graceful descope trade space exists. The Lander descope scenario presented here places the farside site as a high priority as this maximally enables observation of core-transmitted phases from the known nearside deep moonquake clusters (Yamada *et al.*, 2013).

- Descope the Crisium Lander. <u>Rationale:</u> Minimizes impacts to observations of seismic coverage for core-transmitted phases. Crisium Basin will have preliminary geophysical data from the 2023 CLPS mission.
- Descope two nearside Landers (Schickard and Crisium). <u>Rationale:</u> Minimizes impacts to observations of seismic coverage for core-transmitted phases, and preserves deployment of Landers in the PKT and the farside FHT. The remaining two sites permit using deep moonquake activity to explore the core and mantle of the Moon (the Korolev Basin site is antipodal to the nearside A-1 deep moonquake nest, and the PKT site is offset to the A-33 farside nest to explore mantle structure).
- Descope the communications satellite and deploy four nearside Landers. <u>Rationale:</u> Reduces operational complexity, but has direct to Earth as the only communications option. Place the original farside Lander at a polar region (*e.g.*, in the Wiechert region of the south pole close the observed lobate scarps). This removes any investigation of the thick crust on the farside, but retains the investigation of lobate scarps, heat flow in the FHT, and adds a NGLR station at the south pole of the Moon.

- Descope the Orbiter and farside Lander, deploy three nearside Landers in the sites proposed. Rationale: Reduces operational complexity, but has direct to Earth as the only communications option.
- Descope the Crisium and Korolev Landers and the communications satellite. Rationale: this represents the **threshold LGN mission**. This preserves the ability to understand the internal structure of the Moon (utilizing deep moonquake cluster activity), records heat flow within the PKT and FHT, adds a southern hemisphere node to the NGLR network, and achieves the LGN mission goal.

3.5.3 Descope Optimization

Lander and instrument descopes can be combined into an optimization matrix to assess each configuration's total mission costs, relative risk posture, and anticipated science return (see **Figure 14**). We have developed a methodology to consider eleven different instrument descope options, from all of the baseline mission instruments to just the VBB+SP. This needs to be reconciled, and combined with six mission architectures, from 4 to 2 Landers both with and without an orbiting deployment communications satellite, a total of 66 configurations. The configurations with an Orbiter include a farside station and those without are all on the nearside. The trade space between baseline and threshold missions enables flexibility in the implementation of the LGN mission. The total mission costs are derived from **Section 5.2, LGN Cost Estimates**, **Table 17** and takes into consideration the additional costs of each Lander when an Orbiter is not included in the mission architecture (reference **Section 2.4 Table 3**, **Architecture Trade Studies**). We assume the Lander bus size and subsystems are fixed regardless of the total number of instruments on each Lander. Cost scores are either under (10) or over the cost cap (1). While all options for the baseline mission are currently over the New Frontiers cost cap, this strategy highlights

LGN Descope Optimization Matrix

(Lander-instrument configuration trades, increasing level of instrument science priority)

# of instruments: →

	# of landers:	(decreasing mission complexity)	1 All Instruments	2 minus Silicon Audio Borehole Seismometer (+eng cam)	3 minus Heat Flow Probe 1 (+eng cam)	4 minus Deck NGLR	5 minus Deck SP Seismometer	6 minus Surface retroreflector (+eng cam)	7 minus Heat Flow Probe 1&2 (+eng cam)	8 minus LMSS Plasma & SCM only	9 minus LMSS (+pano cam)	10 minus ~1 instrument per discipline (1 HFP, deck NGLR, Borehole SiA, SCM, ESA) (+ 2 eng cam)	11 Only Instrument: Seismometer (VBB+SP) (+eng cam)
Baseline	4 + comm sat	Cost Score:	1	1	1	1	1	1	1	1	1	1	1
		Risk Score:	9	9	9	9	9	9	9	9	9	9	9
		Science Score:	10	10	10	10	10	9	9	10	8	9	5
		RMS:	8.0	7.8	7.8	7.8	7.8	7.4	7.3	7.8	7.3	7.4	5.9
	4 landers nearside only	Cost Score:	1	1	1	1	1	1	1	1	1	1	1
		Risk Score:	9	9	9	9	9	9	9	9	9	8	8
		Science Score:	10	9	9	9	9	8	8	9	8	9	5
		RMS:	7.5	7.4	7.4	7.4	7.4	7.0	6.9	7.4	6.9	6.9	5.5
	3 + comm sat	Cost Score:	1	1	1	1	1	1	1	1	1	1	10
		Risk Score:	9	9	9	9	9	9	9	9	9	8	8
		Science Score:	8	8	8	8	8	8	7	8	7	8	5
		RMS:	7.0	6.9	6.9	6.9	6.8	6.8	6.4	6.9	6.4	6.5	7.9
	3 landers nearside only	Cost Score:	1	10	1	1	10	1	10	1	10	10	10
		Risk Score:	8	8	8	8	8	8	8	8	8	8	7
		Science Score:	9	8	8	8	8	7	7	8	7	8	4
		RMS:	6.8	8.8	6.7	6.7	8.8	6.3	8.5	6.7	8.5	8.5	7.6
	2 + comm sat	Cost Score:	10	10	10	10	10	10	10	10	10	10	10
		Risk Score:	8	8	8	8	8	8	8	8	8	8	8
		Science Score:	7	7	7	7	7	7	6	7	6	7	4
		RMS:	8.6	8.5	8.5	8.5	8.5	8.4	8.2	8.5	8.2	8.3	7.6
Threshold	2 landers nearside only	Cost Score:	10	10	10	10	10	10	10	10	10	10	10
		Risk Score:	7	7	7	7	7	7	7	7	7	7	6
		Science Score:	7	6	6	7	6	6	6	6	6	6	4
		RMS:	8.1	8.0	8.0	8.0	8.0	8.0	7.8	8.0	7.8	7.8	7.2

Cost:		Risk:			Science:		
10: under cost cap	1: over cost cap	8-10: low mission risk	4-7: medium risk	1-3: high mission risk	8-10: high science return	4-7: medium science return	1-3: low science return

LG099

Figure 14: The LGN descope matrix assesses instrument-Lander configurations in terms of mission level costs, risk, and science return. Color coded as described in keys at bottom of table. While baseline mission provides the highest science return and lowest risk posture, it is also over the cost cap with the current design. The threshold mission fits within the cost cap with a variety of instrument options, and accomplishes the primary mission goal at a higher risk posture.

areas for future work, and we do have configurations within the threshold mission that fit within it.

Risk is assessed to include any action that could impede meeting the stated objectives in three tiers: high (1-3), medium (4-7), and low risk (8-10). This includes technical risk (including engineering and science instruments with components and/or deployments that could fail), and mission level risks such as losing a Lander or not be able to communicate with them. Science instruments with a low TRL increase mission risk score. These are relative risks to the mission assuming the engineering design is a nominal risk posture. Direct to Earth communication for the nearside Landers along with the communications Orbiter adds a redundant system and a lower risk posture than those without the Orbiter. Mission configurations without a communication satellite rely on all communication direct to Earth with no back up. Risk scores range from 6 to 9. Our baseline mission with all of the science instruments represents the lowest mission risk posture.

Science scores are assessed based on how each primary instrument by discipline contributes to the **Science Traceability Matrix**. Additional instruments contribute to the overall science score with each configuration obtaining a low (1-3), medium (4-7), or high (8-10) science return score. Four Landers with one on the farside (the baseline mission) provides the best science mission architecture to address seismic network objectives. A reduced science score results from configurations without an orbiting magnetometer, which precludes the magnetic core deflection analysis, and these will not have a farside station that impacts network science return. Less than three nearside stations result in a reduced NGLR science score. Science scores run from 4 to 10.

Each configuration's cost, risk, and science scores are combined into a root-mean-square (RMS) value. This exercise provides a tool to observe the influence of each science instrument, and independently the Lander-Orbiter mission architecture, on the mission as a whole to fully explore the graceful descope options in terms of cost, risk, and science return. While the baseline mission with all instruments provides the highest science return and lowest risk posture, it is also over the cost cap with the current design, but highlights areas of future work that would enable the baseline mission to fit in the cost cap. However, the 50% reserve requirement has significantly contributed to this. The descoping 1 Lander from the baseline mission fits within the cost cap for a variety of instrument options, and accomplishes the primary mission goal at a slightly higher risk posture.

4. DEVELOPMENT SCHEDULE AND SCHEDULE CONSTRAINTS

4.1 High-Level Mission Schedule

The LGN high-level mission schedule is shown in **Figure 15**. For a 2030 launch example, it was assumed that a NF Announcement of Opportunity (AO) will come during the first quarter of 2023. For

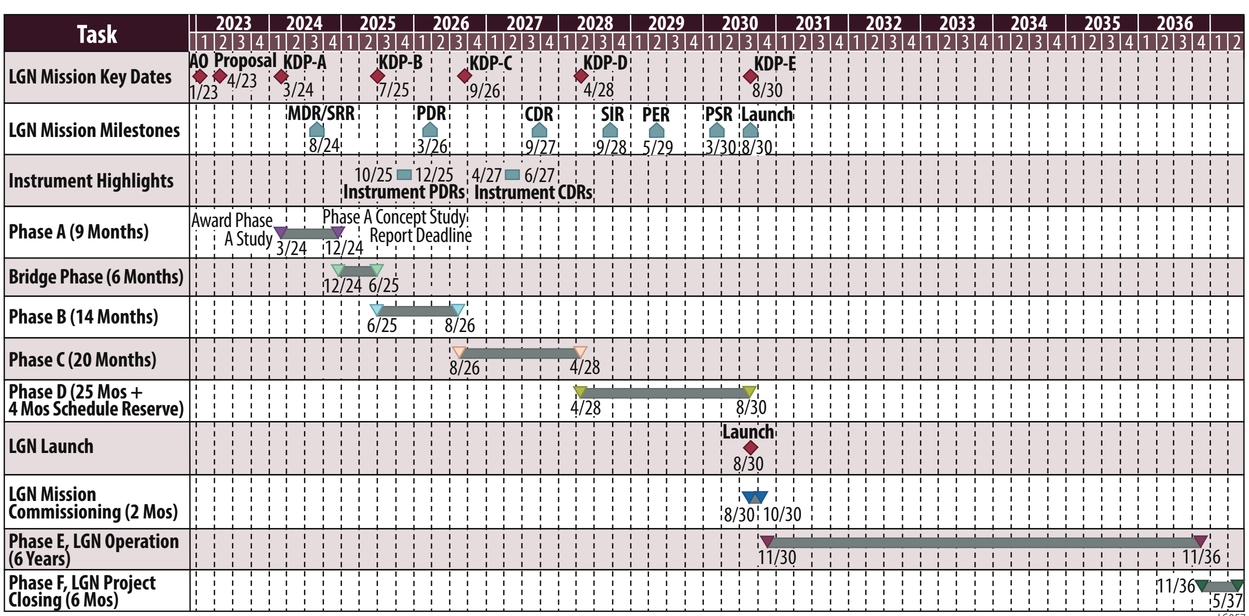


Figure 15: LGN Mission Milestones and Project Phase durations.

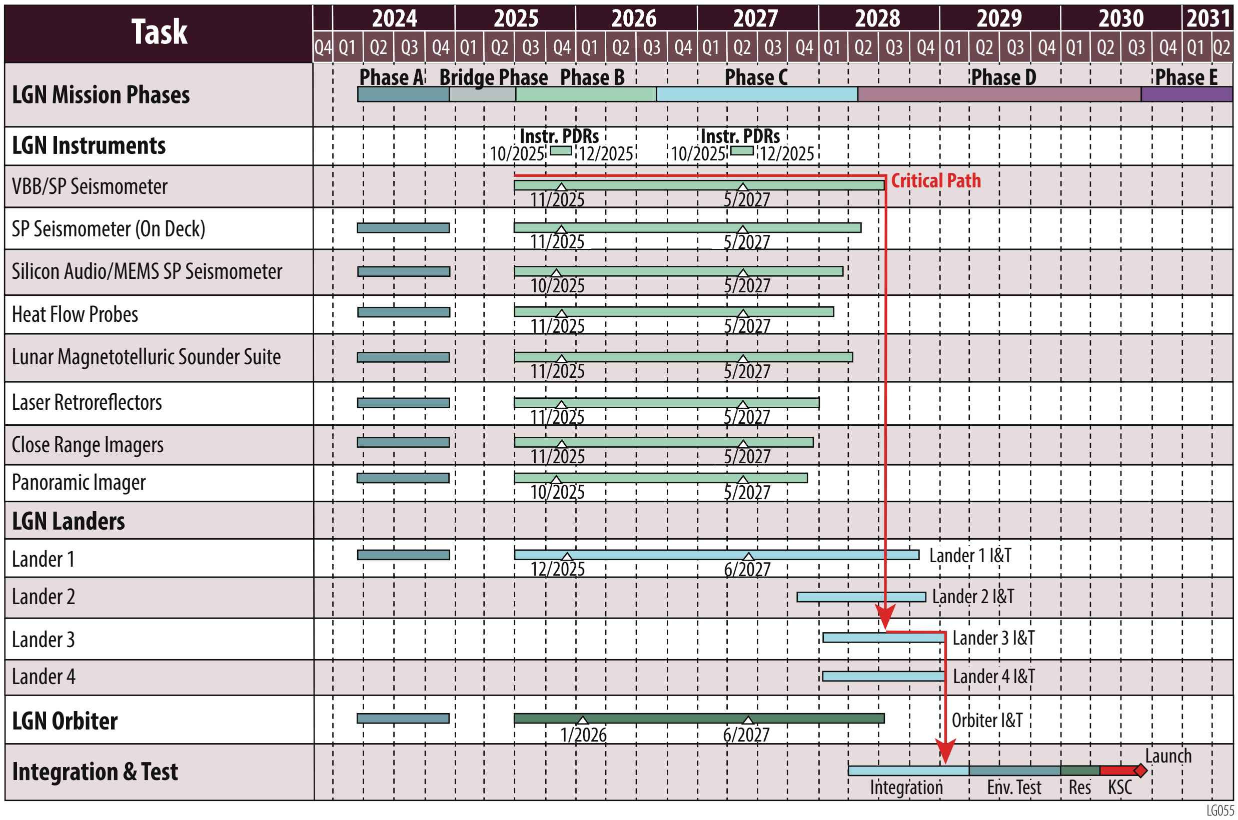


Figure 16: LGN Mission High Level Schedule.

the mid decade NF AO, the schedule would approximately slide to the right accordingly.

The assumptions and consideration made for the schedule development are the following:

1. It is anticipated that the LGN Lander development efforts will likely benefit from lessons learned and cost saving approaches developed under the Commercial Lunar Payload Services (CLPS) program. Assuming the CLPS program continues into the next decade, adding reliability, survival durations and thermal options will allow for lower cost, shorter schedule lunar lander options. The study is taking the approach that some schedule and cost savings will be realized by partnering with CLPS providers using some CLPS approaches. This will provide substantial schedule savings, including utilizing approaches such as a protoflight development.
2. The project assumes that Landers and instruments will take advantage of parallel assembly, integration and test.
3. The schedule includes the recommended reserves during integration and testing activities.

The LGN key phase durations are shown in **Figure 16**. The LGN mission contains multiple Landers with each Lander containing multiple instruments. The input in the rows corresponding to "Start of Phase B to Delivery of Instrument" corresponds to the delivery of the last instrument.

4.2 Technology Development Plan

Development plans for the instruments are shown in **Table 16**.

4.2.1 Orbiter and Lander Components

The structural concepts developed for this study are within the current state of the art and have heritage. It is likely that landing dynamics, kinetic energy absorption and self-stabilizing system will require customization for the specific landing environment, but the technology maturity is high with several planetary landings providing heritage and extensive methodologies.

4.3 Development Schedule and Constraints

The key constraint of the LGN mission is how to efficiently build multiple instruments and Landers. The study team investigated how the hardware for the Magnetospheric Multiscale (MMS) mission was built. MMS had similar requirements since it consisted of four identical spacecraft. With such a demanding hardware build special attention should be given to facility requirements planning and the Assembly, Test and Launch Operations (ATLO) activities. In order to achieve saving in cost and schedule, the LGN team will utilize the possibility of parallel builds whenever possible.

5. MISSION LIFE-CYCLE COST

The LGN mission is considered a high maturity mission concept (CML 5). Most major subsystems have a long heritage and were used in previous missions. The LGN mission concept described in this report is technically feasible and the cost estimate is at the upper limit of the New Frontiers class and in flagship mission territory. Depending on the configuration, the LGN mission can be a New Frontiers class mission, while the fully loaded all options mission enters the flagship mission cost level using the current design, indicating future work is required to fit the baseline mission as defined here in the NF cost cap (see **Section 6**).

The ground rules and assumptions for the LGN estimate were based on the distributed "Ground rules for Mission Concept Studies in Support of Planetary Decadal Survey." Cost estimates are presented in fiscal year 2025 dollars (FY25$). Initial estimates were generated in FY20$ and then inflation adjusted to FY25$ dollars. The cost estimates assume that NASA will fund the development of all LGN instruments, Landers, and Orbiter mission costs. The team used a standard mission WBS and the cost estimates cover activities through the end of Phase F.

Table 16: LGN Key Phase Duration Table.

Project Phase	Duration (Months)
Phase A – Conceptual Design	9 Months
Phase B – Preliminary Design	14 Months
Phase C – Detailed Design	20 Months
Phase D – Integration & Test	29 Months
Phase E – Primary Mission Operations	72 Months
Phase E – Extended Mission Operations	48 Months
Phase F – Mission Closeout	6 Months
Start of Phase B to PDR	9 Months
Start of Phase B to CDR	27 Months
Start of Phase B to Delivery of VBB/SP Seismometer (Instr. #1)	36 Months
Start of Phase B to Delivery of SP Seismometer (Instr. #2)	34 Months
Start of Phase B to Delivery of Silicon Audio/MEMS SP (Instr. #3)	32 Months
Start of Phase B to Delivery of Heat Flow Probe (Instr. #4)	31 Months
Start of Phase B to Delivery of LMSS (Instr. #5)	33 Months
Start of Phase B to Delivery of Laser Retroreflector (Instr. #6)	28 Months
Start of Phase B to Delivery of Close Range Imagers (Instr. #7)	27 Months
Start of Phase B to Delivery of Panoramic Imager (Instr. #8)	26 Months
Start of Phase B to Delivery of Lander #1	39 Months
Start of Phase B to Delivery of Lander #2	39 Months
Start of Phase B to Delivery of Lander #3	42 Months
Start of Phase B to Delivery of Lander #4	42 Months
Start of Phase B to Delivery of Orbiter Spacecraft	36 Months
System Level Integration & Test	16 Months
Project Total Funded Schedule Reserve	4 Months
Total Development Time Phase B - D	63 Months

The cost estimates assume that the Launch Vehicle Services Program will provide the launch vehicle needed to deliver the LGN Landers and Orbiter to the Moon.

The LGN Instrument and Subsystem Ground Rules and Assumptions are:

- The LGN team will select instruments onboard the lunar Landers primarily based on their ability to deliver exceptional science data.
- NASA will fund all instrument development.
- Each LGN Lander includes 6 science instrument suites, 4 close-range imagers, and a panoramic imager.
- All lunar Landers are identical.

LGN cost includes 50% reserves on Phases B-D costs as recommended by the ground rules. Finally, it is anticipated that the LGN team will be able to collaborate (through a competitive Partnering Opportunity Document (POD) or other equivalent process) with one of the commercial lunar Lander providers currently in the Commercial Lunar Payload Services (CLPS) program. This will ensure that the mission will take advantage of a number of the recent and future advances in the area of lunar Landers.

5.1 Costing Methodology and Basis of Estimate

LGN costing methodology for the Orbiter/carrier spacecraft and the lunar Landers is based on a mix of parametric cost modeling, analogies to prior missions, and historic cost wrap factors (to account for "overhead" costs such as program support, facilities, unallocated expenses, etc.). Price H parametric model estimates are driven by preliminary Master Equipment Lists (MELs). The MEL line item masses, types of materials, TRLs, and complexity are combined with mission-level cost wrap factors to derive an initial estimated mission cost. No grassroots estimate was developed for the study. A reserve of 50% on Phases A-D and 30% on Phase E was added to the total derived cost. The 50% reserve equates to an approximate 70% confidence level in the cost certainty in conventional cost risk analysis. No cost or reserve was added to the estimate for the Launch Vehicle. All costs are in Fiscal Year (FY) 2025 dollars.

Once cost estimates were generated, the team compared them to the cost of past analogous missions. Special emphasis was given to the cost of the Lander and Orbiter (WBS 6). The cost of the LGN Landers was analyzed using data from the Cost Analysis Data Requirement (CADRe) database for the Phoenix Mars Lander and the Interior Exploration using Seismic Investigations, Geodesy and Heat Transport (InSight) missions. The LGN team also reached out to two potential industry Lander providers as another way of verifying estimates. Another point of reference for the Lander cost was recently provided with the contract award to Astrobotic to deliver NASA's Volatiles Investigating Polar Exploration Rover (VIPER) to the Moon's South Pole in late 2023 (https://www.nasa.gov/press-release/nasa-selects-astrobotic-to-fly-water-hunting-rover-to-the-moon). For the LGN spacecraft (Orbiter) the team analyzed the cost of analogous spacecraft used for Lunar Reconnaissance Orbiter (LRO), Gravity Recovery and Interior Laboratory (GRAIL), and Lunar Atmosphere and Dust Environment Explorer (LADEE) missions which represent the high and low ends of spacecraft complexity.

Finally, even though there is a high probability for international collaboration and contributions, the presented cost assumes that NASA will bear all costs for developing the LGN mission.

5.2 Cost Estimates

Based on the Price H model and cost analogies, the LGN mission cost was estimated at a total cost ranging from \$1.0B to \$1.5B. All cost estimates, including 50% reserves, are considered preliminary and are commensurate with concept maturity level of the missions studied. Even though the baseline LGN mission is above the New Frontiers (NF) cost limit (assumed to be \$1.1B FY25\$), there are versions of the mission considered here that can fit in the NF envelope. **Table 17** displays LGN cost for two versions, of the mission, one containing four Landers and an Orbiter and a second version with three Landers. For these two version a 50% reserve was assumed. Also displayed in the table is the cost of the 6-year Phase E assuming 30% reserves. Technology development costs (to bring new technology to TRL 6 level) are included in the mission cost estimate. Uncertainty exists in the technology development cost, but most of them are low risk since the LGN mission uses mature technologies used in previous missions to Mars. Unforeseen development problems will likely cause minor cost increases. The Technology Development Plan is provided in **Section 4.2**. Not included in this estimate is the possibility of likely international contributions.

6. FUTURE WORK

As is seen through the cost and mass estimates, formulated on the basis of this very brief study, there is future work to do in order to get four Landers to the lunar surface with a communication satellite – our baseline mission. The PMCS project details our first attempt at developing a mission architecture, and shows the areas where more work is needed. Here, we highlight some focus areas for this future work that will enable the desired baseline mission within the New Frontiers cost cap.

As the project matures and the design process begins, the following trades should be performed:

- Contact dynamics
- Lowest altitude allowable for of thruster operation
- C&DH memory and speed for TRN
- TRN algorithm (pattern matching vs correlation)
- Map size and resolution
- Lander Attitude requirements
- HA maneuver size

6.1 Power

The high mass of each LGN Lander in this study is driven by the battery mass for a 6-year (goal of 10 years) solar-powered mission. While consideration of NGRTG was permitted under PMCS ground rules, the study schedule and budget did not permit simultaneous development of both solar and NGRTG designs in parallel. Using NGRTG as a power source would certainly reduce the mass of each Lander, but the thermal design may need to be adjusted. However, a detailed study of this trade was not possible under PMCS time and budget constraints.

Table 17: LGN Cost Estimate (FY25$M).

WBS	Description	Phase A	TOTAL Phase B-D (4 Landers & Orbiter)	TOTAL Phase B-D (3 Landers)	Phase E (6 Years)
01	Project Management	4.00	58.77	38.98	4.80
02	System Engineering		44.08	29.24	0.75
03	Safety & Mission Assurance		36.73	24.36	0.60
04	Science		44.08	29.24	33.75
05	Payload Total		250.16	202.25	
05.01	Payload Management, Payload System Engineering, and Payload S&MA		19.75	15.80	
05.02	VBB/SP Seismometer w/ thermal insulation and deployment system		55.14	42.67	
05.03	SP Seismometer (On Deck)		28.60	22.88	
05.04	Silicon Audio/MEMS SP Seismometer w/ burial system		23.45	19.45	
05.05	Heat Flow Probes w/ burial system		39.47	31.46	
05.06	LMSS		45.76	36.61	
05.07	NGLR w/Gimbal		13.73	11.44	
05.08	Close range imagers		21.48	17.37	
05.09	Panoramic imager		19.27	15.58	
06	Flight Systems Total		484.50	285.00	
06.01	Lander		352.50	285.00	
06.02	Orbiter		130.00		
06.03	Orbiter Magnetometer with Boom		2.00		
07	Mission Operations System		22.93	22.93	58.80
08	Launch Vehicle				
09	Ground Data System		20.32	20.32	7.20
10	Integration & Test		35.47	24.36	
	Total:	4.00	998.30	676.68	105.90
	Reserves:		499.15	338.34	31.77
	Total:		**1497.45**	**1015.02**	**137.67**

6.2 Thermal Design

Following on from the power trade, the thermal design for the LGN Landers needs to be robust given the large temperature swings between night and day, coupled with the long mission duration. The change in mass and power supply between solar and NGRTG also requires a detailed study beyond what was possible here.

In addition, thermal thermal analyses must be performed to determine if thermal shields are required for the main engines.

6.3 Lander Design

While current CLPS Landers will not survive the lunar night, we have not been able to explore if the basic commercial Lander design could be used for the LGN mission in a hybrid CLPS model, where a commercial partner becomes part of the team. CEMA costs for NASA-built Landers are higher than the estimates we have received from commercial vendors, which requires further study, as does the need to reduce the mass of the Landers to be within the launch vehicle capability (**Section 6.5**). Separation systems between each Lander and the one between the bottom Lander and Orbiter were evaluated and a COTS separation system that provided a viable solution with a known mass and high TRL was selected. However, it was beyond the scope of the study to look at Orbiter and Landers to launch vehicle coupled dynamics. This will need to be evaluated.

6.4 Communications

Rather than maintaining a communications satellite for the duration of the mission, we undertook preliminary discussions with a commercial vendor, Surrey Satellite Technologies Limited, to explore a commercial option. This company is partnering with European Space Agency to develop a commercial communications network around the Moon. Due to time limitations (and COVID19) these discussions have not progressed to the point of discussing costs for the communication services, and further work is needed in this area. We also have not fully considered the availability of a NASA-provided communication asset (Gateway), which could also be available. Refinement of data loads, orbits of the communication assets, and costs need to be quantified.

6.5 Mass Reduction Initiative

It is evident that the mass of the LGN baseline mission will need to be reduced because this is a direct driver for fuel (which adds to the mass) and thus cost. Each system needs to be looked at individually and in conjunction with others to develop new and novel ways to reduce mass.

7. SUMMARY AND RECOMMENDATIONS

This planetary mission concept study has highlighted the feasibility of conducting a Lunar Geophysical Network mission under the auspices of NASA's New Frontiers program. This study has explored architectures that illustrate what types of mission could fit in a New Frontiers cost cap and shows the limits of each mission. It is evident, though, that the parameter space for such mission architectures has expanded with new commercial companies as potential partners may reduce lander costs.

Recommendation: The next New Frontiers call should open up the parameter space for commercial participation to potentially enable a 4-lander LGN mission that would maximize the science return.

We view the LGN concept study as a first run of implementing such a mission. **Section 6** highlights future work that could potentially fit the 4-lander plus orbiter concept into the New Frontiers budget. The main concept followed here was for a solar-powered mission, resulting in a high lander mass due to the batteries needed for night time survivability and operations. An initial look at using NGRTGs was undertaken, but requires more fidelity to ensure the concept estimates are accurate. What was not studied was the use of a hybrid power solution of half NGRTGs and half solar power.

Recommendation: Although our baseline mission concept fell outside the cost cap of New Frontiers and the fairing size of the Falcon heavy, it should not be discounted as impossible to implement because of the future work to reduce mass, volume, and budget outlined in **Section 6**.

We are baselining a 6-year primary mission to cover one lunar cycle, but have a goal of 10 years. This mission would form the basis of the International Lunar Network and the duration will allow single, more short-lived nodes to be added by international/commercial partners, or via a human presence on the Moon.

Recommendation: The LGN mission represents an opportunity for NASA to strengthen existing and create new international and commercial partnerships.

While it is recognized that the cost of New Frontiers is in the medium-class (Class B), it is important that emerging technologies be considered in mission implementation. There is a fine balance between minimizing risk through use of tried and tested technologies and stifling innovation.

Recommendation/Finding: The next New Frontiers call should seek to find a balance between risk and technological innovation that is carried through from the initial draft AO, through the review process and Phase A study, to mission implementation.

Science Traceability Matrix

The Lunar Geophysical Network (LGN)

Primary Goal	Primary Objectives	Mission Investigations	Measurement Requirements Physical Parameters	Measurement Requirements: Observables	Instrument Requirements	Mission Requirements	Decadal Survey	LEAG Lunar Exploration Roadmap	Scientific Context for the Exploration of the Moon	LEAG Advancing the Science of the Moon
Understand the initial stages of terrestrial planet evolution.	Evaluate the interior structure and dynamics of the Moon. Constrain the interior and bulk composition of the Moon. Delineate the vertical and lateral heterogeneities within the interior of the Moon as they relate to surface features and terranes. Evaluate the current seismo-tectonic activity of the Moon.	Determine the size, state, and composition of the lunar core.	**Seismics:** Radius, seismic velocity and layering of the core. **NGLR:** Determine the size and shape of liquid lunar core and eccentricity of the inner core. **LMSS:** Radius of the core	**Seismics:** Repeated measurement of waves that traverse the core by using known DMQ sources to investigate inner/outer core and core size (PKP arrivals - 25/year with minimum SNR=?). **NGLR:** Repeated ranging to each of the landers with a 1 mm accuracy over the life of the mission and continuing over the next 5 decades. **LMSS:** Magnetotail deflection.	**Seismics:** Sensitivity 1 order of magnitude improvement over Apollo at 0.1Hz ($5x10^{-11}$). (replace with figure). Timing accuracy: 10-50 ms. **NGLR:** The NGLR is required to receive the short laser pulse from the LLRs and send back a well collimated beam with an internal delay of less than a few picoseconds. This requires very precise passive control of the internal temperature gradients. It must also be pointed to the center of the Libration pattern of the Earth to within several degrees. Finally, it must be designed to have a design goal of a lifetime of at least 50 years. **LMSS:** Simultaneous orbital and surface magnetic-field measurements monitored for 1 year	Mission Lifetime of 6 years (10 year goal) Reliability Category 2, Class B Operate Instruments Data Collection continuously Minimize lander generated vibrations Location accuracy (for station): 200 m (= 50 ms* 4 km/s). Location accuracy (for SMQs) = 10-30 km. 4 Landers at 4 widely spaced landing sites including one on the far side PKT (P-60 Young Basalt Field) Lat: 20.7˚, Lon: -47.4˚ Schickard basin Lat: -44.3˚, Lon: -55.1˚ Crisium basin Lat: 18.5˚, Lon: 61.8˚ Korolev basin Lat: -2.4˚, Lon: -159.3˚ Landed orientation such that Laser Reflectometers and Heat Flow Probes are oriented within in 10 deg azimuth of Earth Moon vector	Understand terrestrial planet differentiation by constraining the structure and composition of the Moon. *How are planetary magnetic fields initiated and maintained? [Ch. 5]*	**Objective Sci-A-5:** Understand Lunar Differentiation. **Objective Sci-A-8:** Determine the stratigraphy, structure, and geological history of the Moon. **Objective Sci-A-9:** Understand formation of the Earth-Moon system. **Objective Sci-B-1:** Understand the impact history of the inner Solar System as recorded on the Moon.	**Concept 2:** The structure and composition of the lunar interior provide fundamental information on the evolution of a differentiated planetary body. **Goal 2c -** Determine the size, composition, and state (solid/liquid) of the core of the Moon.	**Concept 2:** The structure and composition of the lunar interior provide fundamental information on the evolution of a differentiated planetary body. The structure and composition of the lunar interior provide fundamental information on the evolution of a differentiated planetary body. Meaningful progress in expanding our knowledge of planetary differentiation can be made by the emplacement of equipment such as a simultaneous, globally distributed seismic and heat flow network and/or an expanded retroreflector network. In order to make significant progress toward addressing the fundamental questions related to the lunar interior as raised in Concept 2, the recommendations for implementation in the 2007 NRC report remain valid. These recommendations include emplacement of instruments such as a simultaneous, globally distributed seismic and heat flow network and/or an expanded retroreflector network, as well as strategic collection of samples from terrains of different ages that can provide constraints on lunar geochemistry and new information on the history of the lunar dynamo.
		Determine the state of as well as the chemical/physical stratification in the lunar mantle. *Is there a partial melt layer in lower mantle? Is the upper and lower mantle compositionally distinct?* *What are the mechanical properties of the major components Moon (e.g. crust, mantle, CMB and core) as this will illuminage the formation of similar bodies like the inner planets?*	**Seismics:** Spatial variation of seismic velocity to the CMB across different terranes. **NGLR:** Evauate the details of regular and irreugular components of the lunar response to the gravitaional influence of the Earth. Compare the tempeoral aspects of this motion with the predicted magnitude of these effects by various lunar aspects. **LMSS:** Mantle electrical conductivity **HF:** Determine the heat flux from the mantle	**Seismics:** Measure waves that traverse the mantle from known DMQ nests as well as shallow and surface sources. **NGLR:** Observatories will repeatedly measure the distance from the LLRO to each of the passive NGLRs with an accuracy of 1 mm. This will be performed during the LGN mission and for at least four decades afterward. **LMSS:** EM sounding using electric and magnetic fields (magnetotelluric method). **HF:** Temperature gradient and thermal conductivity	**Seismics:** Sensitivity 1 order of magnitude improvement over Apollo at 0.1Hz ($5x10^{-11}$). (replace with figure). Timing accuracy: 10-50 ms. **NGLR:** The NGLR is required to receive the short laser pulse from the LLRs and send back a well collimated beam with an internal delay of less than a few picoseconds. This requires very precise passive control of the internal temperature gradients. It must also be pointed to the center of the Libration pattern of the Earth to within several degrees. Finally, it must be designed to have a design goal of a lifetime of at least 50 years. **LMSS:** FIelds 10-5 Hz to 10 Hz, monitor 1 year. **HF:** Heat flow determination within 2 mW/m2. Thermal conductivity and gradient 7% accuracy. Depth: temperature and thermal conductivity measured at a minimum of 3 depths below 1.5 m a spacing of 0.3 m or greater.		Constrain the bulk composition of the terrestrial planets to understand their formation from the solar nebula and controls on their subsequent evolution. *How do the structure and composition of each planetary body vary with respect to location, depth, and time? [Ch. 5]*		**Concept 2:** The structure and composition of the lunar interior provide fundamental information on the evolution of a differentiated planetary body. **Goal 2b** - Characterize the chemical/physical stratification in the mantle, particularly the nature of the putative 500-km discontinuity and the composition of the lower mantle.	
		Determine the thickness of the lunar crust and characterize its vertical and lateral variability.	**Seismics:** Determine crustal thickness at different locations across the Moon. **NGLR:** Determination of local Love numbers at different locations across the Moon. **LMSS:** Determine the electrical conductivity of the crust at different locations **HF:** Determine heat flux from the crust at different locations	**Seismics:** Measure seismic signal from impacts of known timing and locations and measure receiver functions. Without camera: 25 located impacts/year. **NGLR:** Observatories will repeatedly measure the distance from the LLRO to each of the passive NGLRs with an accuracy of 1 mm. This will be performed during the LGN mission and for at least four decades afterward. **LMSS:** EM sounding using electric and magnetic fields (magnetotelluric method). **HF:** Temperature gradient and thermal conductivity	**Seismics:** 1 Hz. Location accuracy (for impact) = 1 km. Timing accuacy (for impact) = 1 s. **NGLR:** The NGLR is required to receive the short laser pulse from the LLRs and send back a well collimated beam with an internal delay of less than a few picoseconds. This requires very precise passive control of the internal temperature gradients. It must also be pointed to the center of the Libration pattern of the Earth to within several degrees. Finally, it must be designed to have a design goal of a lifetime of at least 50 years. **LMSS:** FIelds 10^{-5} Hz to 100 Hz, monitor 1 year. **HF:** Heat flow determination within 2 mW/m^2. Thermal conductivity and gradient 7% accuracy. Depth: temperature and thermal conductivity measured at a minimum of 3 depths below 1.5 m a spacing of 0.3 m or greater.		Characterize planetary interiors to understand how they differentiate and dynamically evolve from their initial state. [Ch. 5]		**Concept 2:** The structure and composition of the lunar interior provide fundamental information on the evolution of a differentiated planetary body. **Goal 2a -** Determine the thickness of the lunar crust (upper and lower) and characterize its lateral variability on regional and global scales.	
		Determine the thermal state of the lunar interior and elucidate the workings of the planetary heat engine. *Key point of the mission where all measurements come together.*	**Seismics:** Determine the seismic velocity gradient and attenuations within the lunar interior. **NGLR:** Determine the size and shape of liquid lunar core and eccentricity of the inner core. **LMSS:** Electrical conductivity **HF:** Heat flow from different geologic terranes.	**Seismics:** body wave arrival times, amplitude. **NGLR:** Repeated ranging to each of the landers with a 1 mm accuracy over the life of the mission and continuing over the next 5 decades. **LMSS:** EM sounding using electric and magnetic fields (magnetotelluric method). **HF:** Temperature gradient and thermal conductivity	**Seismics:** knowledge of instrument response as a function of temperature with a relative accuracy of 5% in the 0.05 - 10 Hz. **NGLR:** The NGLR is required to receive the short laser pulse from the LLRs and send back a well collimated beam with an internal delay of less than a few picoseconds. This requires very precise passive control of the internal temperature gradients. It must also be pointed to the center of the Libration pattern of the Earth to within several degrees. Finally, it must be designed to have a design goal of a lifetime of at least 50 years. **LMSS:** FIelds 10^{-5} Hz to 10 Hz, monitor 1 year. **HF:** Heat flow determination within 2 mW/m^2. Thermal conductivity and gradient 7% accuracy. Depth: temperature and thermal conductivity measured at a minimum of 3 depths below 1.5 m a spacing of 0.3 m or greater.		*What are the proportions and compositions of the major components (e.g. crust, mantle, core, atmosphere/exosphere) of the inner planets? [Ch. 5]* *What are the major heat-loss mechanisms and associated dynamics of their cores and mantles? [Ch. 5]*		**Concept 2:** The structure and composition of the lunar interior provide fundamental information on the evolution of a differentiated planetary body. **Goal 2d -** Characterize the thermal state of the interior and elucidate the workings of the planetary heat engine.	
		Determine the locations and source of shallow moonquakes and characterize their relationship with young tectonic landforms.	Determine the recurrence frequency of shallow moonquakes, source magnitudes, slip pattern, and depth	**Seismics:** Identifying shallow moonquake/high-frequency teleseismic signals across multiple stations. 3-5 events per year (per Apollo).	**Seismics:** 10 Hz. Timing accuacy (for SMQ) = 1 s.					
		Monitor impacts on to the lunar surface as an aid to exploring the lunar interior.	**Seismics:** size-frequency distribution of impact signals.	**Seismics:** Identifying impact seismic signals. Constrain locations using camera observations. 100 near-side, 25 far-side imaged impacts per year.	**Seismics:** 0.1 - 5 Hz.		Characterize Planetary Surfaces to Understand How They Are Modified by Geologic Processes.			

Science Traceability Matrix

Secondary Goal	Secondary Mission Objectives	Mission Investigations	Measurement Requirements	Decadal Survey	LEAG Lunar Exploration Roadmap	Scientific Context for the Exploration of the Moon
Understand the current space environment.	Characterize the present impact flux, and variations therein, on the Moon	Monitor impacts on to the lunar surface as an aid to exploring the lunar interior		*What were the sources and timing of the early and recent impact flux of the inner solar system? [Ch. 5]*	**Objective Sci-A-7:** Understand the impact process.	Concept 1: The bombardment history of the inner solar system is uniquely revealed on the Moon
				Characterize planetary surfaces to understand how they are modified by geologic processes.		The Moon is an accessible laboratory for studying the impact process on planetary scales
	Obtain fundamental data about the current lunar surface plasma environment	Characterize and monitor lunar surface plasma and its spatiotemporal input processes.	Surface electron and ion plasma energy including densities and temperatures.	Characterize planetary surface processes to understand how they are modified by goelogic processes. [pg. 5-3]	**Objective C-2:** The lunar plasma environment is a unique laboratory for studying fundamental plasma physics processes.	The Moon is a natural laboratory for regolith processes and weathering on anhydrous airless bodies
				Characterize plasma-surface-volatile effects including transport mechanisms. [pg. 5-8]		
				How do magnetosphehres interact with the solar wind? How is surface material modified? [pg. 7-27]		
Provide the most sensitive available tests of current gravitational theories and General Relativity.	Characterize the fundamental aspects of gravitation and General Relativity i.e., the (Weak Equivalence Principle (WEP), the Strong Equivalence Principle (SEP), GravioMagnetics, etc.) as they relate to the fundamental aspects of Cosmology (Dark Matter, Dark Energy and/or the conflict between General Relativity and Quantum Mechanics).	Evaluate the deviations of motion of the center of mass of the Moon as compared to the predictions of General Relativity. Test of the possibile rate of change of the gravitational constant G	The NGLR is required to receive the short laser pulses from the LLROs and send back, with the variation of the internal temporal delay of less than a few picoseconds, a well-collimated beam with a divergence of less than a few arc-seconds. The latter requires very precise passive control of the internal temperature gradients, at a level of a few tenths of a degree Kelvin. This is achieved by very strong thermal isolation between the hot housing and the cold Cube Corner Retroreflector (CCR).The NGLR must also be pointed to the center of the libration pattern of the Earth to within several degrees. The pointing will be accomplished with a system of gimbals and a camera. After successfully pointing, the gimbals will be locked for the rest of the mission. Finally, the NGLRs must be designed to have a design goal of a lifetime of at least 50 years.		**Objective Sci-C-1:** Astrophysical and Basic Physics Investigations using the Moon. **Investigation-Sci-C-1C:** Key Tests of the Strong Equivalence Principle in Gravitational Field Theory.	LLR retroreflectors provide important data on the lunar orbit and are now challenging (our) understanding of basic physics.

Exploration Goal	Exploration Objectives	Mission Investigations	Measurement Requirements: Physical Parameters	Measurement Requirements: Observables	Instrument Requirements	Decadal Survey
Evaluate the risk to humans living and working on the Moon .	Characterize the present impact flux, and variations therein, on the Moon.	Monitor impacts on to the lunar surface as an aid to exploring the lunar interior.				The Human Exploration Program [Page 26] To reduce the cost and risk for future human exploration, robotic precursor missions would be needed to acquire information concerning potential resources and hazards, to perform technology and flight system demonstrations, and to deploy infrastructure to support future human exploration activities. [Ch 6]
						What were the sources and timing of the early and recent impact flux of the inner solar system? [Ch. 5] Characterize the present impact flux, and variations therein, on the Moon.
	Obtain fundamental data about the current lunar surface plasma environment.	Characterize and monitor lunar surface plasma and its spatiotemporal input processes.	Surface electron and ion plasma energy.	Electron ion plasma moments including densities and temperatures.	~Near continuous measurements of electron and ion plasma densities and temperatures including during the lunar night, and within magnetotail crossings. Monitor for at least 1 year.	Measuring and modeling the characteristics and timescales of planetary magnetic fields and their influence on planetary volatile losses, surface charging, and radiation environments. [Ch 5]
Establish infrastructure to support missions to the Moon.	Support precision navigation for crewed and robotic missions on route to the Moon and in the lunar vicinity.	**Navigation:** The ME (mean Earth/polar axis) based upon the JPL ephemeris DE421 or later and the LLR program is the current navigation frame that addresses passage from Earth to the Moon. The NGLRs will result is very significant improvement in the ME frame (i.e. approaching a factor of at least 5 compared the current ME frame which is based upon laser ranging to the Apollo retroreflectors). **Lunar Cartography:** The current Apollo retroreflector arrays for the tie points for the Principal Axis (PA) reference frame. Thus the the NGLRs will provide additional tie points that are the basis for systems like LRO to define the locations of various lunar features. **Earth Orientation:** The LLR program currently provides a rapid determination of the Universal Time Zero (UT0 - that is, the Earth rotation as observed at a given LLRO. The LGN set of NGLRs will significantly improve the accuracy of the measurements of UT0.	The NGLR is required to receive the short laser pulses from the LLROs and send back, with the variation of the internal temporal delay of less than a few picoseconds, a well-collimated beam with a divergence of less than a few arc-seconds. The latter requires very precise passive control of the internal temperature gradients, at a level of a few tenths of a degree Kelvin. This is achieved by very strong thermal isolation between the hot housing and the cold Cube Corner Retroreflector (CCR).The NGLR must also be pointed to the center of the libration pattern of the Earth to within several degrees. The pointing will be accomplished with a system of gimbals and a camera. After successfully pointing, the gimbals will be locked for the rest of the mission. Finally, the NGLRs must be designed to have a design goal of a lifetime of at least 50 years.			

APPENDIX A: ACRONYMS

ACS......................Attitude Control System
ADC.....................Analog Digital Converter
AH.......................Amp Hour
ALSEP.................Apollo Lunar Surface Experiment Package
AO.......................Announcement of Opportunity
AOS.....................Acquisition of Signal
APE.....................Attitude and Position Estimate
ARM....................Advanced RISC Machines
ASI......................Agenzia Spaziale Italiana
ASRG..................Advanced Stirling Radioisotope Generator
ATC.....................Analog Telemetry Card
ATLO...................Assembly, Test, and Launch Operations
ATP.....................Authority To Proceed
B.........................Billion
BCM....................Battery Charge Module
BM......................Breaking Maneuver
BOL.....................Beginning of Life
C.........................Celsius
C3.......................launch energy
CBE.....................Current Best Estimate
CAD.....................Computer Aided Design
CADRe................Cost Analysis Data Requirement
CCD.....................Charge Coupled Device
CCSDS.................Consultative Committee for Space Data Systems
CCR.....................Corner Cube Retroreflector
C&DH.................Command and Data Handling
CDR.....................Critical Design Review
CEMA..................Cost Estimating, Modeling & Analysis
cg........................center-of-gravity
CLPS...................Commercial Lunar Payload Services
cm.......................centimeter
CM......................center of mass
CM......................Configuration Management
CMD...................Command
CML....................Concept Maturity Level

CNES....................National Centre for Space Studies
CoM....................Center of Mass
COMSEC..............Communications Security
ConOPS ....Concept of Operations
COTS....................Commercial off the Shelf
CSR......................Concept Study Report
CSS......................Coarse Sun Sensors
CTE......................Controlled Thermal Expansion, Coefficient of Thermal Expansion
DAVINCI+...........Deep Atmosphere of Venus Investigation of Noble gases, Chemistry, & Imaging, Plus
dB ....Decibel
DALI ....Development and Advancement of Lunar Instrumentation
DCS ....DTE Communication Subsystem
DDOR..................Delta Differential One-way Ranging
DEA ....Digital Electronics Assembly
Dec.....................Declination
DV ....Delta Velocity, Change in Velocity
ΔV ....Delta Velocity, Change in Velocity
DEM ....Digital Elevation Model
DLA ....Declination of Launch Asymptote
DOD....................Depth Of Discharge
DOF ....Degrees of Freedom
DOI.....................De-Orbit Insertion
DPDT ....Double Pull Double Throw
DPU....................Data Processing Unit
DSM ....Deep Space Manuever
DSN....................Deep Space Network
DSX ....Demonstration and Science Experiments
DTE.....................Direct To Earth
EDL.....................Entry, Descent and Landing
EFI......................Electric Fields Instrument
EM......................Engineering Model
EMTG..................Evolutionary Mission Trajectory Generator
EOL.....................End of Life
ESA.....................European Space Agency
ESA.....................Electrostatic Analyzer - Plasma Instrument
FBB ....Force Feedback Board
FDC.....................Fault Detection Correction

FDIR ....Fault detection, isolation, and recovery
FGM ....Fluxgate Magnetometer
FEC ....Forward Error Correction
FEE ....Front-End Electronics
FETS ....Field Effect Transistors
FHT......................Feldspathic Highlands Terrane
FOT......................Flight Operations Team
FOV ....Field of View
FSW.....................Flight Software
FT ....Fault Tolerance
FTE ....Full Time Equivalent
FWHM.................Full Width at Half Maximum
FY........................Fiscal Year
g ....measurement of Earth gravitational acceleration (9.81 m/s^2)
GaAs....................Gallium Arsenide
Gbits ....Gigabits
Gbps....................Gigabits per second
GFIC.....................Goddard Fellow Innovation Challenge
GMSK ....Gaussian Minimum Shift Keying
GN&C ....Guidance Navigation and Control
GOI ....Goals, Objectives, and Investigations
GPR ....Goddard Procedural Requirements
GSDR...................Global Slope Data Record
GSFC....................Goddard Space Flight Center
GSM ....Generic Switch Module
GRAIL ....Gravity Recovery and Interior Laboratory
HA ....Hazard Avoidance
H&S.....................Health and Status
HD&A ....Hazard Detection and Avoidance
HFP ....Heat Flow Probe
HGA.....................High Gain Antenna
HiPAT...................High Performance Apogee Thruster
HK ....Housekeeping
HPSC ....High Performance Spacecraft Computing
H&S.....................Health and Status
Hz........................Hertz
ICDR ....Instrument Critical Design Review

ICEE....................Instrument Concepts for Europa Exploration
ICRF....................Inertial Centered Reference Frame
IFOV ....Instantaneous Field of View
ILN ....International Lunar Network
IMU ....Inertial Measurement Unit
INFN...................National Institute for Nuclear Physics
I&T ....Integration and Test
InSight ....Interior Exploration using Seismic Investigations, Geodesy and Heat Transport Mars mission
IOAG...................Interagency Operations Advisory Group
IPDR...................Instrument Preliminary Design Review
IPGP ....Paris Institute of Earth Physics
IR ....Infrared
JWST ....James Webb Space Telescope
K ....Kelvin
KE.......................Kinetic Energy
kg.......................kilogram
KISS....................Keck Institute for Space Studies
km/s...................kilometers per second
kN ....Kilo Newtons
kpbs ....kilobits per second
KSC.....................Kennedy Space Center
kW ....kilowatt
LADEE.................Lunar Atmosphere and Dust Environment Explorer
LDPC ....low-density parity-check
LGA ....Low Gain Antenna
LGN ....Lunar Geophysical Network
LHA ....Landing Hazard Avoidance
LHCP ....Left hand circular polarization
LiDAR ....Light Detection And Ranging
LISTER ....Lunar Instrumentation for Subsurface Thermal Exploration with Rapidity
LLR.....................Lunar Laser Retroreflector
LMS....................Lunar Magnetotelluric Sounder
LMSS..................Lunar Magnetotelluric Sounding Suite
LMSS-MT...........Lunar Magnetotelluric Sounder Magnetometer
LOI......................Lunar Orbit Insertion
LOS.....................Line Of Sight
LRD ....Launch Readiness Date

LRO ....Lunar Reconnaissance Orbiter
LROC ....Lunar Reconnaissance Orbiter Camera
LRR.....................Launch Readiness Review
LRR.....................Lunar Radio-phase Ranging
LV ....Launch Vehicle
LVH.....................Local Vertical Horizon
LVDS...................Low Voltage Differential Signaling
LVLH...................Local Vertical, Local Horizontal
LVPS...................Low Voltage Power Supply
m ....meter
M ....million
m/s ....meters per second
Mag....................Magnetometer
MAVEN ....Mars Atmosphere and Volatile EvolutioN
Mbits..................Megabits
Mbps..................Megabits per second
MCC....................Motor Controller Card
MCDR ....Mission Critical Design Review
MCU ....Mechanism Control Unit
MDL ....Goddard Space Flight Center's Mission Design Lab
MEL....................Master Equipment List
MEMS.................micro-electro-mechanical systems
MET....................Mission Elapsed Timer
MEV ....Maximum Expected Value
MGA...................Medium Gain Antenna
MGS ....Mars Global Surveyor
MIC.....................Multi-Interface Cards
mJ ....millijoules
MLI.....................Multi-Layer Insulation
mm ....millimeter
MMH..................Monomethylhydrazine
MMRTG ....Multi-Mission Radioisotope Thermoelectric Generator
MMS...................Magnetospheric Multiscale mission
MOC ....Mission Operations Center
MOCET................Mission Operations Cost Estimating Tool
MOSFETS ....Metal–Oxide–Semiconductor Field-Effect Transistor
MPDR ....Mission Preliminary Design Review

MPU ....Mechanism and Propulsion Unit
MRC ....Mechanism Release Card
MSC/NASTRAN ...MacNeal-Schwendler Corp (mechanical analysis software)
MSL.....................Mars Science Laboratory
MSL RAD ....Mars Science Laboratory Radiation Assessment Detector
MSPS..................Million Symbols Per Second
MSR ....Mars Sample Return
NASA..................National Aeronautics and Space Administration
NEA ....Non-Explosive Actuator
NESZ...................noise-equivalent-sigma-nought
NF ....New Frontiers
NFT.....................Natural Feature Tracker
NGLR ....Next Generation Lunar Retroreflector
NGRTG................Next Generation Radioisotope Thermoelectric Generator
NICM ....NASA Instrument Cost Model
NIR ....Near Infrared
nm ....nanometer
ns.......................nanosecond
NTO ....Dinitrogen tetroxide
OD......................Orbit Determination
OSIRIS-Rex ....Origins, Spectral Interpretation, Resource Identification, Security, Regolith Explorer mission
OSAM-1..............On-orbit Servicing, Assembly, and Manufacturing Mission 1
OSR ....Optical Solar Reflectors
P ....Pressure
PAF.....................Payload Attach Fitting
PC.......................Panoramic Camera
PDC ....Propulsion Drive Card
PDR ....Preliminary Design Review
PI ....Principal Investigator
PKP ....The phase travelling as a P-wave from the source through the mantle again in its path back to the surface. P-waves in the outer core labelled K.
PKT.....................Procellarum KREEP Terrane
PM ....Project Management
PMCS..................Planetary Mission Concept Studies
POD....................Partnering Opportunity Document
PPS.....................Pulse Per Second
PSE.....................Power System Electronics

PSP.....................Parker Solar Probe
RAAN..................Right Ascension of the Ascending Node
RAD....................Radiation Assessment Detector
RADAR................Radio Azimuth Direction and Ranging
RAM...................Random Access Memory
RAO....................Resource Analysis Office
RF.......................Radio Frequency
RHCP..................Right hand circular polarization
RMS...................Root Mean Square
RO......................Radio Occultation
ROM...................Rough Order of Magnitude
rpm....................rotations per minute
RTG.....................Radioisotope thermoelectric generator
RWA...................Reaction Wheel Assembly
SA, S/A...............Solar Array
SARM.................Solar Array Regulation Module
S-band...............2 to 4 GHz (15 to 7.5 cm wavelength) Communications Band
SBC.....................Single Board Computer
SCM....................Search Coil Magnetometer
SEIS....................Seismic Experiment for Interior Structure, on the InSight mission
SEU.....................Single Event Upset
SiA.....................Silicon Audio
SIMPLEx............Small Innovative Missions for Planetary Exploration
SINDA.................Systems Improved Numerical Differencing Analyzer
S:N.....................Signal-to-Noise
S/C.....................spacecraft
SLS.....................Space Launch System
SNR....................Signal to Noise Ratio
SOC....................Science Operations Center
SOI.....................Silicon on Insulator
SP.......................Short Period
SPAN..................Solar Probe Analyzer
SPDT...................single-pole double-throw
SPENVIS............. Space Environmental Effects and Education System
SPSAB................Short Period Silicon Audio Buried (Seismometer)
SQPSK................Staggered quadrature phase-shift keying
S/W....................Software

SRR ....System Requirements Review
SSN ....Sunspot Number
SSPA...................Solid State Power Amplifier
SSR.....................Solid-State Recorder
STM....................Science Traceability Matrix
SWaP..................Size, Weight and Power
SWIA ....Solar Wind Ion Analyzer
T.........................Temperature
TASC...................Tri-axial Search Coil
TASK...................Tool for Analysis of Surface Cracks
Tbits ....Terabits
TCM....................Trajectory Correction Maneuver
THEMIS...............Time History of Events and Macroscale Interactions during Substorms mission
TJGaAs................Triple Junction Gallium Arsendie
TLI ....Trans Lunar Injection
TLM....................Telemetry
TMR....................Triple Modular Redundant
TRL.....................Technology Readiness Levels
TRN ....Terrain Relative Navigation
TWCP ....Tape Wrapped Carbon Phenolic
TWTA ....Traveling Wave Tube Amplifier
ULA ....United Launch Alliance
USN....................Universal Space Network
VBB ....Very Broad Band
V ....Volt
VIPER ....Volatiles Investigating Polar Exploration Rover
W ....Watt
WBS ....Work Breakdown Structure
WG ....waveguide
X-band ....2.5 - 3.5 cm; 7.0 to 11.2 GHz communications band

1. APPENDIX B: LGN ENGINEERING PAPER

1.1 Mission Overview / Instruction

The Lunar Geophysical Network mission will deploy an Orbiter and four solar-powered Landers with instrumentation as described by the International Lunar Network report (ILN, 2009). Based on the LGN Concept Study conducted as part of the previous decadal survey (Shearer & Tahu, 2013), four Landers are baselined. Deploying four Landers enables global distribution (including the farside) and allows for redundancy, as a threshold of two Landers still achieves the threshold mission. The four Landers are long-lived (6 years, with a goal of 10 years) to maximize science and allow other nodes to be added by international and commercial partners during the lifetime of the mission, thus increasing data fidelity.

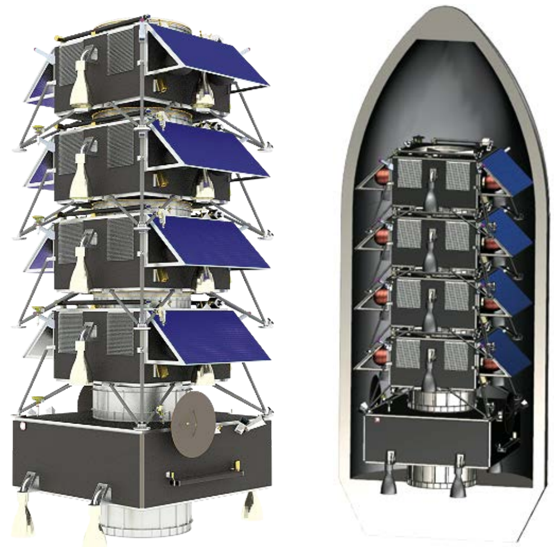


Figure B-1: Lander Launch Configuration.

The Orbiter and four Landers will be launched together on the Falcon 9 heavy expendable launch vehicle (**Figure B-1**) and the Orbiter will provide the delta V need to place the Orbiter and four Landers into lunar orbit, where the Landers will be deployed sequentially. Each Lander will contain a very broad band and short period seismometer package (beneath the Lander), one buried short period seismometer (deployed from one of the legs), two heat flow probes (deployed in two other legs), and two laser retroreflectors (one on the Lander deck and one on the lunar surface).

The Orbiter also carries a communications payload that provides a telecommunications relay with all Landers. This allows a Lander to be placed on the farside of the Moon by providing instrument deployment monitoring and data relay back to Earth. Because each Lander will be capable of sending data directly to Earth, the Orbiter also provides redundancy for the near side Landers.

Landing sites have been chosen to reflect a global distribution around the Moon with each Lander being in distinct terranes. Four Landers will be deployed sequentially to the PKT (P-60) site just south of the Aristarchus, the Schickard Basin, and the Crisium Basin on the nearside, and in the Korolev Basin on the farside. In addition, the Orbiter carries a magnetometer on a two-meter boom to supplement surface magnetotelluric measurements. The threshold LGN mission requires two Landers: one at the PKT (P-60) site and one in the Schickard basin. The baseline and threshold missions will explore distinct lunar terranes, a variety of crustal thicknesses, examine seismic signals from known deep moonquake nests as they pass through the Moon to understand lunar structure, and expand the current LRR network. Rationale for these sites are as follows:

PKT (P60 basalt field) site just south of the Aristarchus Plateau (latitude = 20.7; longitude = -47.4): Relatively young and flat volcanic terrain, with few boulder fields. This landing site is well within the boundaries of the PKT and just south of the thorium anomaly at the Aristarchus Plateau.

Schickard Basin (latitude = -44.3; longitude = -55.1º): This site is in the southern hemisphere of the Moon and the floor is partially flooded with basaltic lava flows that form a relatively flat landing site, with few craters and boulder fields. This landing site is well within the FHT.

Crisium Basin (latitude = 18.5; longitude = 61.8º): This basalt-covered basin has been visited by Landers before in the 1970s. Luna 23 (November 1974) failed with the spacecraft damaged upon landing, whereas the Luna 24 robotic sample return was successful (see Plescia, 2012). The lavas on the floor of the basin for a relatively flat terrain and have an age of 3.6-3.7 Ga (Nyquist *et al.,* 1978). The lavas contain secondary crater populations that will need to be avoided. According to the latest crustal thickness maps, the primary crust is essentially absent (Wieczorek *et al.,* 2013) allowing mantle heat flow to be directly measured. In late 2022/early 2023, a CLPS mission is scheduled to land in the Crisium Basin that will be carrying three instruments proposed for the LGN mission: the lunar magnetotelluric sounder, the LISTER heat flow probe, and the next generation laser retroreflector.

Korolev Basin (latitude = -2.4; longitude = -159.3º): This site will allow the first surface geophysical measurements to be made on the farside of the Moon. The Korolev Basin affords a relatively flat and boulder-free landing area that is in the vicinity of a lobate scarp. It is situated in the highest topographic area of the Moon, which represents the thickest crust.

1.2 High Level Mission Concept

1.3 Mission Requirements Traceability

The LGN flight system consist of an Orbiter and four identical Landers to support the baseline mission. The mission requirements are shown in **Table B-1**. Key requirements that drive the mission are the continuous operation of the instruments, minimizing Lander generated vibrations, widely spaced landing sites that include a Lander on the far side and the need to maintain instruments and Lander electronics within operating temperatures while exposed to the thermal environment of the Moon.

Table B-1: Mission Requirements.

Mission Requirements Top Level	
• Mission Lifetime of 6 years (10 year goal) • Minimize Lander generated vibrations • 4 Landers at 4 widely spaced landing sites including one on the farside 1. PKT (P60) Lat: 20.7°, Lon: -47.4° 2. Schickard basin Lat: -44.3°, Lon: -55.1° 3. Crisium basin Lat: 18.5°, Lon: 61.8° 4. Korolev basin Lat: -2.4°, Lon: -159.3°	• Landed orientation such that NGLRs are oriented within 10 deg azimuth of Earth-Moon vector. • Reliability Category 2, Class B • Operate Instruments Data Collection continuously
Mission Design Requirements	
• Launch 8/30/2030 • Landers land at lunar dawn • Falcon 9 Heavy with 5m fairing	• 250 km lunar circular polar orbit for Orbiter plus Landers • Less than 1 m/s velocity at 1 m above surface • Falcon 9 Heavy provides TLI (C3 $\approx$ -2.5 km^2/s^2)
Orbiter Requirements	
• Provide ΔV of 850 m/s to achieve 250 km circular orbit • 3-axis Attitude control 180 arcsec (3-sigma) • Data Storage 4 Tbits • Provide relay return link services for all science and TLM data for up to 4 Landers from the Lunar Surface back to Earth. • X-band uplink for up to 4 Landers from the Lunar surface. <u>Landers</u> • Antenna pointing control of 1.7 mrad 1-sigma • 1 pps signal with 10^{-6} stability relative to ground station • Provide trickle charge for up to 4 Landers during the cruise to the Moon • Decrypt commands	• Maintain a 250 km Circular Polar Orbit for at least 6 years • Attitude knowledge Roll 60 arcsec, Pitch and Yaw 30 arcsec (3-sigma) • Return 574.8 Gbits total from all 4 Landers per lunar day • X-Band and Ka-Band downlink to Earth • Provide relay forward link services for all Lander Commands for up to 4 Landers • LVDS data interface with magnetometer • Mechanically support a stack configuration at launch for up to 4 Landers • 28 V power System • Encrypt downlink
Lander Requirements	
• ≥1 m/s separation velocity from Orbiter • Provide Delta V of 1864 m/s for landing • Land with a velocity of ≤ 0.5 m/s vertical and ≤ 0.1 m/s horizontal at 5 m above the surface • Land Safely with clearance between surface and lower deck of at least 0.5m boulder • Lander final orientation relative to gravity (nadir) < 5 deg • Landed orientation such that Laser Reflectometers and Heat Flow Probes are oriented within in 10 deg azimuth of Earth Moon vector • Lander final position knowledge within 2 km • Operate Instrument Data Collection continuously • Minimize Lander generated vibrations • Data Storage 1 Tbits • Return 574.8 Gbits total from all 4 Landers per lunar day • Provide thermal control for all instruments attached to Lander • Deliver 100 kg of science instruments to lunar surface • LVDS data interface with instruments	• Provide 90 W electrical power to the science instruments during the Lunar day and 30 W during the Lunar night. • 28 V non regulated power System • Decrypt commands • Encrypt downlink • 0.1 ms timing accuracy with 10^{-6} stability relative to ground station <u>Deploy Instruments</u> Panoramic Camera with unobstructed FOV Search Coil Magnetometer VBB under Lander • Mount Lunar NGLR on Deck and 1 Leg facing Earth • Mount SP Seismometer on deck and deploy 1 on surface • Mount LMS EDA on deck • Mount LMS Deployment mechanism on Deck on each of 4 sides with 180 deg FOV • Mount Si Audio MEMS Seismometer on one Lander leg • Mount HFP on two opposite legs

Ground System Requirements	
• 34m DSN Antenna • Receive Orbiter and Landers engineering & science data telemetry • Encrypt commands • Decrypt downlink • Provide commanding • Record/Archive science data	• Provide critical event telecom coverage: Launch Sep, S/A Deployment, Instrument Deployments • DDOR • Science Data Center • Science Operations Center • Mission Operations Center
Operations Requirements	
• Implement required DDOR • Manage time correlations • Maneuvers • Support DSN passes • Monitor Orbiter and Landers state of health	• Implement contingency procedures • Implement science sequences • Manage Orbiter and Lander operations • Perform ops sim testing

Continuous operation of instruments

The continuous operation of the instruments drives the power subsystem and in particular the mass of the batteries. For each W of power needed during the lunar night ~10.0 kg of batteries are required. To minimize lunar night power usage the instruments provide a low power mode for the lunar night. The Lander thermal design uses louvers that trap the daytime heat for use at night in the thermal enclosure to keep electronics boxes above their cold temperature limits using their radiated heat without the need for heaters. Deployed instruments have heaters built into them and the power needed is accounted for in the instrument power usage. The Lander CDH subsystem was designed to have minimal functionality during the lunar night and to throttle the processor used to collect science data. The communication system is off during the lunar night. The Lander subsystems and instruments achieve a total maximum expected value for night power usage of 40.8 W.

Minimizing Lander generated vibrations

The need to minimize Lander-generated vibrations drives the use of power sources that do not produce vibrations. The solar array concept has oversized fixed arrays to eliminate the need for solar array gimbals. The use of DTE X-band for the near side Landers minimizes the use of the HGA gimbal during normal operations. The far side Lander needs to use the HGA to talk to the Orbiter. This occurs on average 4 days a month for an average of 40 contacts that last between 10-20 minutes. In addition, the Lander itself will expand and contract due to the changes in the thermal environment. To address these generated vibrations a SP Seismometer is placed on the Lander deck to measure and calibrate out the vibrations.

Widely spaced landing sites

Widely spaced landing sites that include a Lander on the far side, drive the need to minimize propulsion hardware, provide a communication link, and provide a terrain navigation system for landing.

Maintaining instruments and Lander electronics within operating temperatures

Maintaining instruments and Lander electronics within operating temperatures while exposed to the thermal environment of the Moon drives the need to place electronics in common compartments, provide a radiator and louvers and vent any remaining propellant after landing.

1.4 Trades

1.4.1 Key Mission Trades

This study builds upon the LGN Concept Study conducted as part of the previous decadal survey (Shearer & Tahu, 2013). The previous decadal study baselined the use of an Advanced Stirling Radioisotope Generator (ASRG) in order to meet lunar night power needs. The current effort focused on developing a solar powered architecture concept and then comparing it to a concept that uses a Next Generation Radioisotope Thermoelectric Generator (NGRTG) to determine the optimal design that would fit within a New Frontiers cost cap. **Table B-2** shows the comparison. Included in the effort was the exploration of using a carrier to act as an Orbiter that provides a platform for a magnetometer and a communication link to a far side Lander.

Table B-2: The LGN team performed multiple architecture trade studies to refine the mission design, revealing the optimal use of resources is a single Orbiter/carrier that enables eliminating Lander propulsion mass.

Trade	Launch Mass (kg)	Launch Margin Falcon 9 Heavy Expendable	Launch Margin SLS Block 1	Number of Landers	Lander Dry Mass (kg)	Lander Mass (wet) (kg)	Orbiter Dry Mass (kg)	Orbiter Mass (wet) (kg)	Fits within Falcon 9 Heavy Expendable 5m fairing	Fits within SLS Block 1 5.1 m Fairing	Regulatory Approval Difficutly	Launch Site Integration and Test Complexity	Landing Operations Complexity	Landing Site Flexibility	Thermal Design Complexity	Farside Comm
Carrier, Comm, Battery and Solar Array Power	14,240.0	8.8%	89.6%	3	1,263.6	2,550.0	1,386.7	6,590.1	Yes	Yes	Low	Low	Low	High	Low	Yes
Carrier, Comm, Battery and Solar Array Power	16,879.8	-8.2%	60.0%	4	1,263.6	2,550.0	1,386.7	6,679.9	Yes	Yes	Low	Low	Low	High	Low	Yes
No Comm w/Carrier, Battery and Solar Array Power*	12,826.8	20.8%	110.5%	3	1,263.6	2,550.0	1,322.6	5,176.9	Yes	Yes	Low	Low	Low	High	Low	No
No Comm w/Carrier, Battery and Solar Array Power*	16,430.0	-5.7%	64.3%	4	1,263.6	2,550.0	1,322.6	6,230.1	No	Yes	Low	Low	Low	High	Low	No
No Comm, No Carrier w/Battery and Solar Array Power	13,991.9	10.8%	93.0%	3	1,658.5	4,664.0	N/A	N/A	Yes	Yes	Low	Low	High	Low	Low	No
No Comm, No Carrier w/Battery and Solar Array Power	18,655.9	-16.9%	44.7%	4	1,658.5	4,664.0	N/A	N/A	No	Yes	Low	Low	High	Low	Low	No
Carrier, Comm, RTG Power	9,589.3	61.6%	181.6%	3	892.3	1800.6	1,389.3	4,187.5	Yes	Yes	High	Medium	Low	High	High	Yes
Carrier, Comm, RTG Power	12,094.5	28.2%	123.2%	4	892.3	1800.6	1,389.3	4,892.1	Yes	Yes	High	Medium	Low	High	High	Yes
No Comm, w/Carrier, RTG Power*	9,500.1	63.2%	184.2%	3	892.3	1800.6	1,325.2	4,098.3	Yes	Yes	High	Medium	Low	High	High	No
No Comm, w/Carrier, RTG Power*	12,005.4	29.1%	124.9%	4	892.3	1800.6	1,325.2	4,802.9	Yes	Yes	High	Medium	Low	High	High	No
No Comm, No Carrier w/RTG Power	9,403.0	64.8%	187.1%	3	1,237.2	3134.3	N/A	N/A	Yes	Yes	High	Medium	High	Low	High	No
No Comm, No Carrier w/RTG Power	12,537.3	23.6%	115.4%	4	1,237.2	3134.3	N/A	N/A	Yes	Yes	High	Medium	High	Low	High	No

** No Comm w/Carrier options use one of the Landers to communicate with Earth during Lunar Transfer and Deployment of Landers.*

The architecture trade study revealed the pairing and combination of the various options (Solar Arrays, Batteries, NGRTG and Orbiter) have significant impact to the mass of the Landers and marginal impact on cost. The cost is somewhat normalized by the increased pricing and launch costs of using an NGRTG or increased cost of the Landers with higher mass due to batteries.

A key factor in the architecture trade is the approach of using the Orbiter as a carrier to perform the Lunar Orbit Insertion (LOI) maneuver that places the Landers into orbit about the Moon. This eliminates additional propulsion masses that would have been required for each Lander to perform its own maneuver. Using one single carrier yields a mass/propellant efficient Lander design and provides a communication link to enable a far side Lander. Additional factors evaluated included the complexity introduced by both regulatory and integration and test of a NGRTG, thermal design complexity, landing site flexibility and landing operations complexity. The solar array concepts offered the lowest regulatory and integration and test complexity, and thermal design complexity. As expected concepts with the Orbiter offered far side Lander communication, and the lowest landing operations complexity as the Landers do not need to land at the same time allowing them to be spaced out over months to minimize the operations team size and allow them to focus on one Lander at a time and apply lessons learned to each subsequent Lander. The Orbiter also enabled the ability to retry landing at a site with another Lander if a landing failed providing redundancy. **Table B-2** shows the results of the architecture trade study. Overall, the solar array Lander option with Orbiter was the least complex and most flexible but at the expense of a 4th Lander for the Falcon 9 Heavy launch vehicle or the need to use the Space Launch System (SLS) Block 1 launch vehicle to include the 4th Lander in one launch. Options exist to launch the 4th solar powered Lander separately on a second launch vehicle, but these were not explored in this study.

The overall technical result that is best suited for launching 4 Landers is the NGRTG with Orbiter concept. However, 3 Landers are acceptable to meet science objectives. The 3 Lander solar array concept provides excellent performance with less complexity and more flexibility at a lower cost than the NGRTG.

1.4.2 Subsystem Trades

The study team conducted additional engineering trades at the subsystem level of the Lander as shown in **Table B-3**.

Table B-3: Lander Subsystem Trades.

	Options	Mass	Cost	Complexity	Performance	Summary/Results
Electrical	Solar Array and Battery Solar Array and NGRTGs	SA/battery less mass efficient	SA/battery more cost efficient NGRTGs have high procurement and launch costs	Use of NGRTGs introduces regulatory and integration and test complexity as well as driving the thermal subsystem design during lunar noon.	Solar arrays and batteries can supply the needed lunar nightime power. NGRTGs can supply the needed power during lunar nightime.	Solar arrays and batteries were chosen. Low temperature battery chemistry could survive night temperatures and offset the development, cost, schedule. NGRTGs had the advantage of producing heat to offset lunar night temps and providing night power with the least mass but created issues with heat rejection during the lunar day.
Attitude Control	TRN Passive	TRN requires mass for LiDAR, Laser Altimeter and Camera	TRN is more costly	TRN is more complex	Passive attitude control during landing does not meet safe landing requirements	TRN was chosen as it meets safe landing requirements.
Thermal	Louvers Heat Pipes Warm electronics thermal cavity Heaters NGRTGs	Mass was not a discriminator between the options.	Louvers, heat pipes, heaters and a warm electronics thermal cavity are inexpensive. NGRTGs have high procurement and launch costs	Louvers, heat pipes, heaters and MLI blanketing of a cavity are common passive thermal control methods. TRN is more complex.	A system of Louvers, heat pipes, heaters and MLI blanketing of a warm thermal cavity meets requirements. NGRTGs meet nightime requirements, but are problematic during the lunar day at noon.	Louvers where chosen in combination with heat pipes for the battery and a warm thermal cavity for the electronics. The Louvers allow the ability to store heat from the daytime in combination with the electronics radiation to keep the warm electronics thermal cavity warm without the use of heaters. This provides a substantial power savings.
Structures	Three legs Four legs	Four legs are more massive.	Four legs cost more than three legs	Not a discriminator	Four-leg design increase the tip-over distance & stability at the cost of mass & packaging complexity.	Four legs were selected because of the net gain in stability compared at minimal mass and cost impact.
	Fixed Legs Deployable Leg	Fixed legs are less massive since deployment mechanism are not needed.	Fixed legs are less costly since its just structure. Deployable legs can provide a larger stance for more stability.	Fixed legs are less complex.	Both fixed and deployable leg designs meet mission requirements.	The tripod leg concept could be implemented without requiring deployment. A wider stance using a deployable leg was considered, hazard avoidance system obviated the need for a wider leg stance for the preliminary concept.

Structures	Fixed SA Deployable SA	Fixed SA more mass efficient	Fixed SA more cost effective	Fixed SA less complex	Both options will provide required power. Fixed arrays are larger and may not fit in launch vehicle fairing.	Fixed solar arrays were determined to be able to fit within the launch vehicle and were chosen as the better option due to simplicity, and expected mass and cost savings.
Communications	Ka-band X-band S-Band Antennas	Not a discriminator	Not a discriminator	Not a discriminator	Ka-band has higher data rate performance but not by a significant amount over x-band. X-band more power efficient.	Ka-Band for the Orbiter to DSN link was chosen as well as X-Band for redundancy. Landers use X-band since it is more power efficient. S-Band not chosen for landers due to restrictions on using S-band on the Moon. S-band is used for Orbiter for DDOR.
	DTE near-side Landers Orbiter Relay for near-side Landers	Not a discriminator	Not a discriminator	Not a discriminator	DTE provides maximum comm availability. If orbiter relay is available, can use it as a backup.	DTE for all near side landers was chosen for maximum performance.
	Gimbal No Gimbal	Gimbal is more mass.	Gimbal costs more.	Gimbal is more complex.	Gimbal allows landing to be less precise and provides longer contact times with Earth and Orbiter.	Gimbal selected for maximum performance.

1.4.3 Launch Operations

The LGN mission launches from Cape Canaveral, Florida on a single Falcon 9 Heavy vehicle with 5m fairing as shown in **Figure B-2**. The baseline mission is to launch four Landers, however since the Falcon 9 Heavy launch mass performance can only accommodate three Landers based on the current mass properties (**Table B-4**), a secondary launch will be required for the fourth Lander. Another current launch vehicle that could accommodate four Landers is the Space Launch System (SLS) Block I Cargo. For now, which approach is pursued for launching the baseline four lander version of the mission will depend on future work. The mission concept assumes that a solution is found and that the launch vehicle upper stage places LGN on a direct transfer to the moon. The launch vehicle upper stage performs the Trans-Lunar Injection (TLI) maneuver, placing LGN on a direct trajectory to the moon. The transfer time to the moon

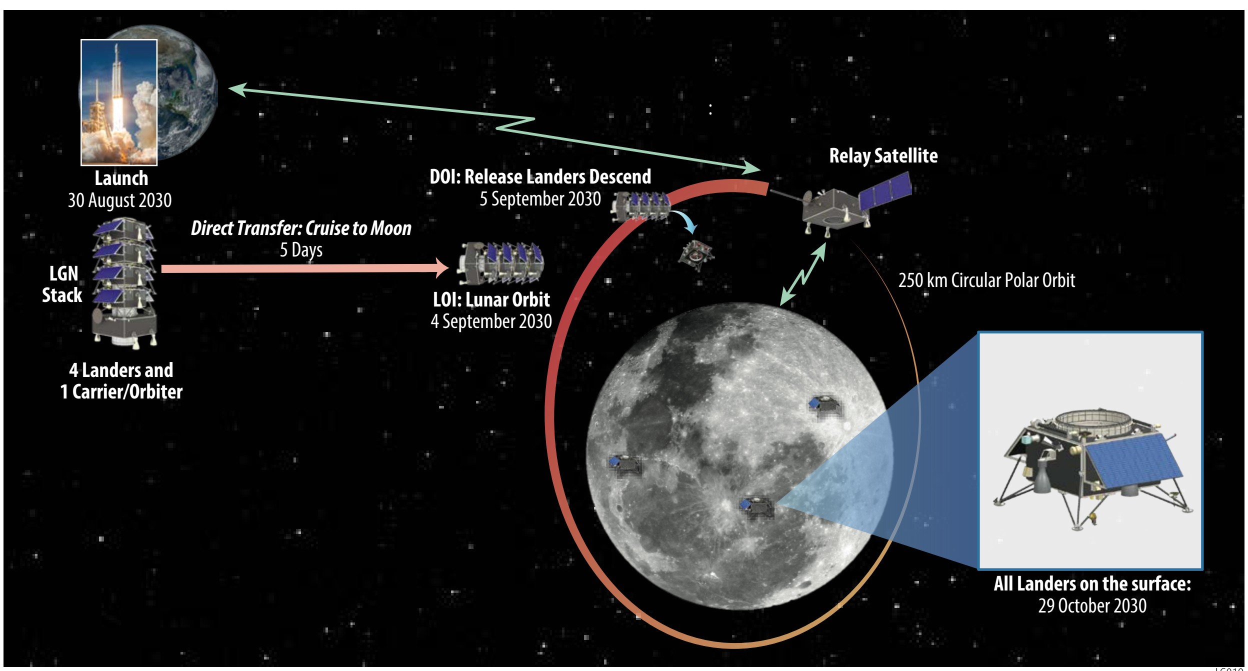


Figure B-2: LGN Mission Concept. An overview of the LGN mission con-ops including launch, cruise, descent, and landing.

Table B-4: LGN Mission Design Summary.

Parameter	Orbiter with 3 Landers	Orbiter with 4 Landers	Lander	Unit
Orbit Parameters (apogee, perigee, inclination, etc.)	250 circular polar	250 circular polar	0	km
Mission Lifetime	6	6	6	yrs
Maximum Eclipse Period	66 minutes	66 minutes	14.5 days	
Launch Site	Cape Canaveral, FL	Cape Canaveral, FL	Cape Canaveral, FL	
Total Wet Mass with Contingency (includes instruments)	5,629.1	6,683.1	2,551.9	kg
Propellant Mass Without Contingency	3,818.2	4,766.8	1,159.5	kg
Propellant Contingency	10	10	10	%
Propellant Mass with Contingency	4,242.4	5,296.4	1,288.3	kg
Orbiter Launch Adapter Mass with Contingency (sized for 4 landers)	97.4	97.4	N/A	kg
Total dry Launch Mass	5,177.6	6,441.3	1,263.6	kg
Total wet Launch Mass	13,382.3	16,988.3	2,551.9	kg
Launch Vehicle	Falcon 9 Heavy	Falcon 9 Heavy	N/A	Type
Launch Vehicle Lift Capability	15,500	15,500	N/A	kg
Launch Vehicle dry Mass Margin	10,322.4	9,058.7	N/A	kg
Launch Vehicle dry Mass Margin (%)	199.4	140.6	N/A	%
Launch Vehicle wet Mass Margin	2,117.7	-1,488.3	N/A	kg
Launch Vehicle wet Mass Margin (%)	15.8	-8.8	N/A	%

is a function of lunar phase and varies between four and five days. There are two launch opportunities for a minimum energy direct transfer to the moon per day, namely, a short coast and a long coast. The two solutions achieve the lunar transfer in two different orbit planes and differ in launch time as well as coast time. Additional constraints (*e.g.*, total eclipse time during the transfer phase) lead to an obvious choice between the long and short coast solutions. As an example, the Lunar Reconnaissance Orbiter (LRO) mission, which utilized a direct transfer to achieve its polar orbit, had a launch window of three days every two weeks. LGN has this same launch window. The LGN orbiter provides navigation, power, communication and attitude control during the lunar transfer period.

1.4.4 Lunar Orbit Operations

Upon arrival at the Moon the Orbiter will perform the LOI (Lunar Orbit Insertion) maneuver to place LGN into a 250 km circular polar orbit around the Moon. Once in the 250 km lunar orbit, the Orbiter will be commanded by the flight ops team to release each Lander so that it lands at its site synchronized with the start of the lunar daylight cycle (9:00am local time). Each Lander will be deployed sequentially and each will land shortly after lunar dawn at the given landing site. Only after landing has been verified and the instruments successfully deployed will the next Lander be sent to the surface. The first Lander will be deployed at the PKT (P-6 basalt field) site, followed by the Schickard site, the farside Korolev site, with the Crisium site being last.

1.4.5 Landing Operations

Each Lander, will perform a De-Orbit Insertion (DOI) maneuver to lower the Lander orbit periapsis as shown in **Figure B-3**. A Braking Maneuver (BM) followed by Terrain Relative Navigation (TRN) will bring the spacecraft down to an attitude of 100 m above the landing site with a velocity magnitude of 8.2 m/s. TRN is discussed in more detail in **Appendix B**, **Section 2.1.5**. At this stage, the Lander is expected to perform its vertical descent to land with a terminal velocity of less than or equal to 0.5 m/s at 1 m above the surface. Once landed, the Lander will vent any remaining propellant through the ACS thrusters to prevent freezing of the propellant. The entire landing operation, from the start of the DOI to touchdown, takes a little over one hour.

1.4.6 Surface Operations and Timeline

Once the Lander is deployed to the lunar surface, a five-day Lander checkout will be performed. Following completion of the post-landing checkout, instrument commissioning and initial data collection will begin. The Lander conditioning and instrument deployment concept of operations (con-ops) is estimated to take ~50 hours. Estimated daylight at each landing site ~ 336 hours (14.5 days) providing sufficient margin should any anomalies occur prior to the Lander entering the lunar night. Throughout the

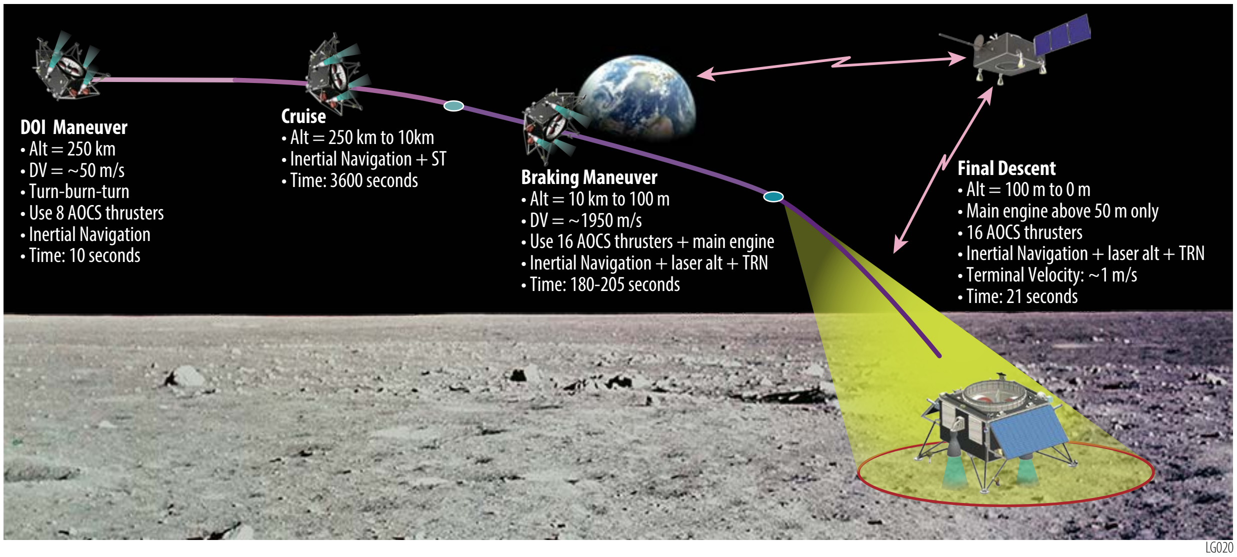


Figure B-3: The LGN Descent and Landing concept is designed to safely deliver the Landers to the lunar surface. Communications are shown between the farside Lander and the Orbiter. For nearside Landers, communication would be direct to Earth.

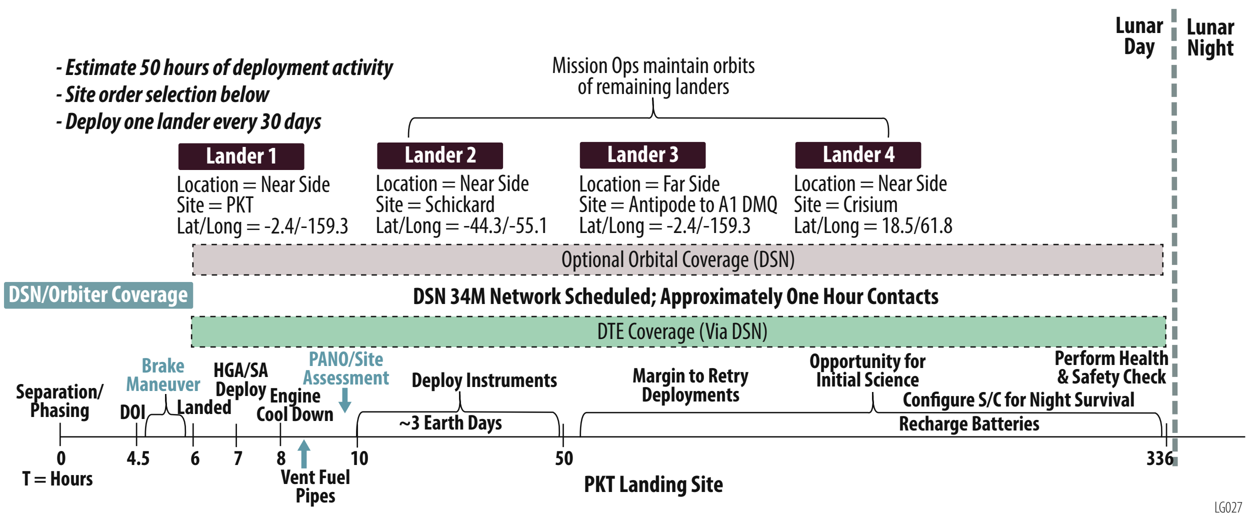


Figure B-4: Descent/Deployment/Surface Ops Timeline (Nearside Landers).

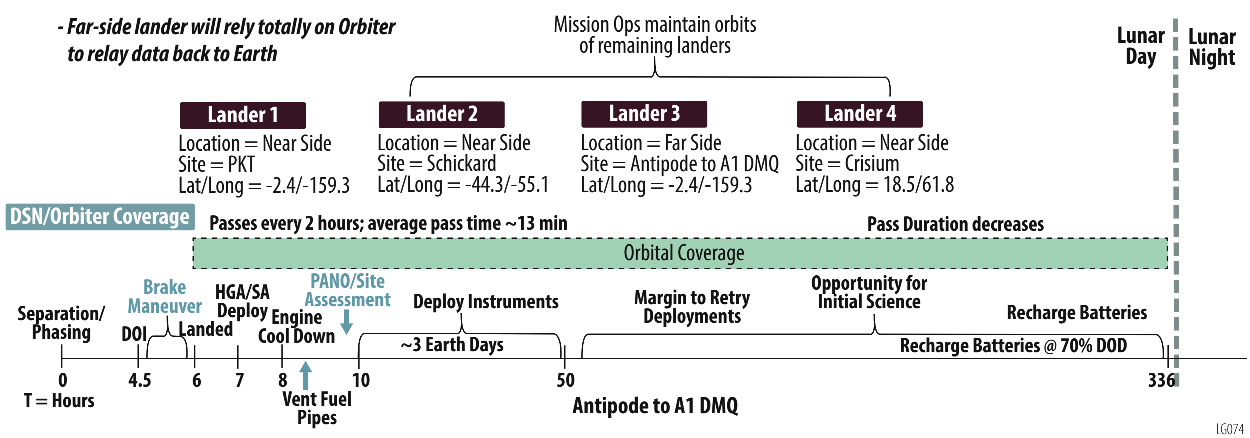


Figure B-5: Descent/Deployment/Surface Ops Timeline (Farside Lander).

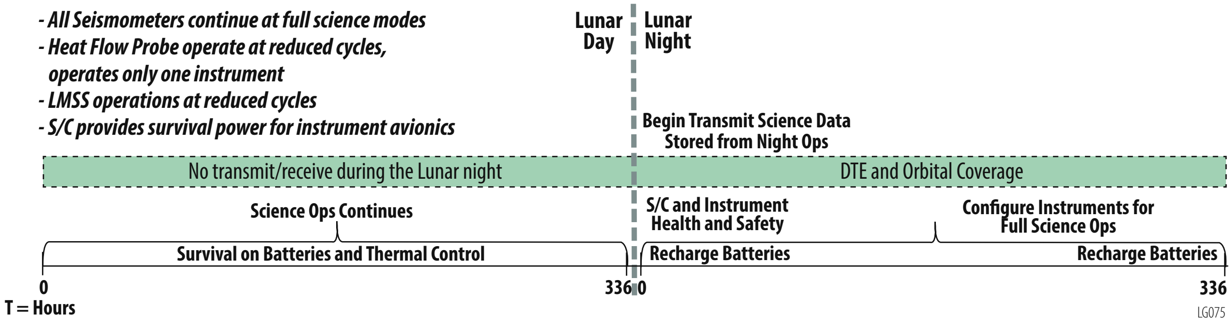


Figure B-6: Surface Night Ops Timeline (Farside Lander).

duration of the deployment, instrument commissioning and surface operations phases, all near side Landers (**Figure B-4**) will have sufficient contact time with the Earth via DSN and the farside Lander (**Figure B-5**) with the Orbiter. Each nearside Lander will initially have 24-hour coverage by DSN during landing and checkout. Once initial data collection begins, each near side Lander will require at least a one-hour duration contact with the DSN each Earth day during the lunar day. The farside Lander uses the HGA to communicate with the Orbiter which in turn communicates with DSN. The Orbiter is in contact with the farside Lander on average 4 days a month for an average of 40 contacts that last between 10-20 minutes. The Orbiter is in contact with DSN on a daily basis with 3-4 contacts per day averaging 128 minutes each. After the first 14.5 days have passed, each Lander will enter lunar night for 14.5 days (**Figure B-6**).

1.5 Mission Design

1.5.1 Transfer

The Falcon 9 Heavy Expendable Launch vehicle performs the Trans-Lunar Injection (TLI) maneuver, achieving a C3 value of -2.051 km^2/s^2, which puts the Orbiter and Landers onto a five-day direct transfer to the moon as shown in **Figure B-7**. Upon arrival, the Orbiter spacecraft performs the Lunar Orbit Insertion (LOI) maneuver to get into a slightly elliptical polar orbit. A small circularization burn is performed to achieve the desired 250 km circular polar orbit (**Figure B-8**). The LOI maneuver costs about 850 m/s in terms of Delta-V, while the circularization burn is approximately 22 m/s, bringing the total transfer cost to 872 m/s.

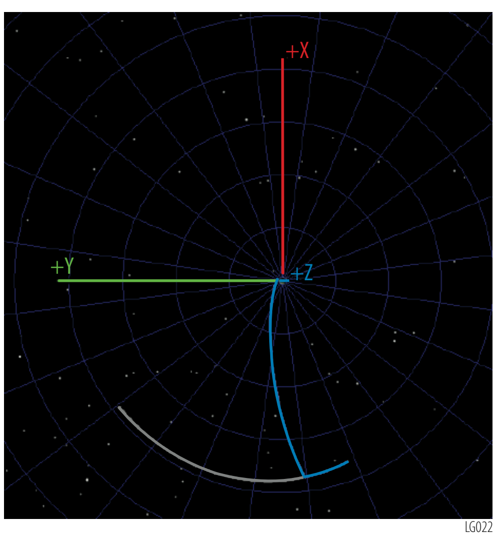


Figure B-7: Five-Day Direct Transfer Trajectory.

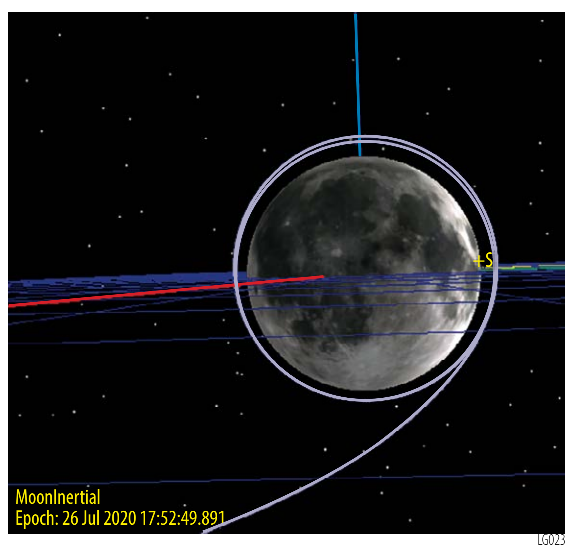


Figure B-8: Lunar Capture and Orbit Circularization.

1.5.2 Landing

Each Lander, upon separating from the Orbiter spacecraft, will await in its orbit until the landing site rotates beneath its orbital plane. At this time, a De-Orbit Insertion (DOI) maneuver is performed

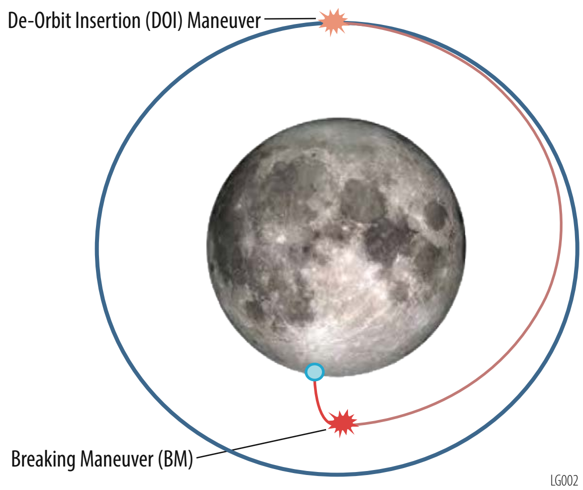


Figure B-9: Landing Strategy (Not to Scale).

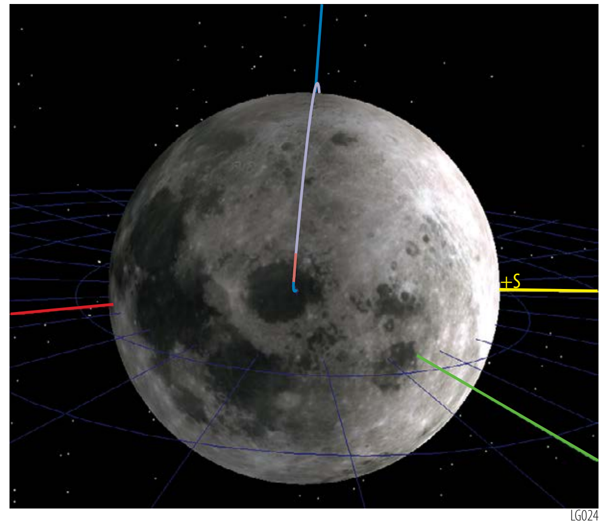


Figure B-10: Landing on Crisium - Red Portion Shows the Breaking Maneuver.

Table B-5: Delta-V Cost per Landing Site.

Site Number	ΔVPhasing	ΔVDOI	ΔVBM	ΔVDescent	ΔVTotal [m/s]
1	7.15	48.70	1,741.97	41.00	**1,838.82**
2	4.18	52.45	1,746.35	41.00	**1,843.98**
3	10.42	58.48	1,753.54	41.00	**1,863.44**
4	9.50	50.05	1,738.92	41.00	**1,839.47**

to lower the orbit periapsis. A Braking Maneuver (BM) is subsequently performed to bring the spacecraft down to an attitude of 100 m above the site with a velocity magnitude of 8.2 m/s. This scenario is depicted in **Figure B-9**. At this stage, the Lander is expected to perform its vertical descent to land with a terminal velocity of less than or equal to 0.5 m/s at 1 m. Prior to the DOI maneuver, a small phasing maneuver is performed (**Figure B-9**) that puts the Lander at the correct location in orbit to perform the DOI maneuver. The entire landing operation, from the start of the DOI to touchdown, takes a little over one hour. A high-fidelity simulation, including finite burns and a full force model, is developed to simulate the entire landing sequence in an end-to-end manner, excluding the vertical descent phase. **Table B-5** summarizes the corresponding Delta-V required for each landing site. **Figure B-10** illustrates the landing trajectory for the landing site Crisium.

1.5.3 Orbit maintenance

The orbit maintenance strategy comprises a two-burn sequence to closely maintain the periapsis and apoapsis of the 250 km polar orbit. The two burns are performed one-orbit apart and are repeated every 45 days (Earth days). As illustrated in **Figure B-11**, the semi-major axis of the orbit is tightly kept at 250 km desired value. There is periodic variation in inclination, but its mean value is maintained at the desired 90 degrees. The total Delta-V cost, for 50 pairs of maintenance maneuvers during the six-year period, is 225 m/s. Using this strategy, the orbit is maintained without directly controlling the expensive out-of-plane variations. As evident by **Figure B-11**, the Right Ascension of the Ascending Node (RAAN) experiences secular growth in the absence of out-of-plane control. The variation in RAAN has no effect on the

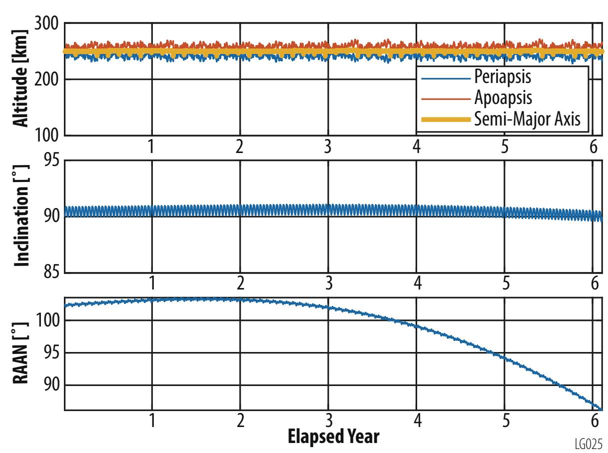


Figure B-11: Orbital Elements Variations during the Six-Year Maintenance Period.

Orbiter's communication with the Landers. However, if there is a direct constraint on the inertial orientation of the orbit for any reason, the RAAN must be maintained directly. Therefore, a Delta-V budget of 140 m/s is allocated for correcting 5 degrees of RAAN, which allows the Orbiter to maintain its RAAN within ± 10 degrees of its nominal value.

1.6 Operations Concept

1.6.1 Deployments

Each Lander will be deployed sequentially and each will land shortly after local lunar dawn (9:00 am) at the given landing site. Only after a soft landing has been verified and the instruments successfully deployed with the next Lander be sent to the surface. The first Lander will be deployed at the PKT (P-60)

Table B-6: The LGN deployment concept for Instruments and Lander.

	Instrument/Ops	Deployment	Duration
1	SP Seismometer on Deck	• None: Activated on Lander deck	1 min
2	Lander Deployments and Ops	• Deploy the High Gain Antenna – Critical Event • Deploy Solar Arrays – Critical Event • Health and safety checkout	3 hours
3	Vent Remaining Propellant	• Venting of any excess fuel in the tanks to reduce Lander-induced seismic noise	1 hour
4	Deploy Panoramic mast camera	• Take & transmit context panoramic image • Activate DVR on panoramic imager	1 hour
5	Science Operations Committee (SOC) evaluation of deployment	• Activate Engineering Camera 1 on Lander leg • Take & transmit image of site prior to drilling	1 hour
6	Heat Flow Probe 1 deployment to 3 meters below surface	• Pneumatic Deployment system	12hrs
7	SOC deployment evaluation	• Activate Engineering Camera 1 on Lander leg • Take & transmit image of site post-deployment	1 hr
8	SOC deployment evaluation	• Activate Engineering Camera 2 on Lander leg • Take & transmit image of site prior to drilling	1 hr
9	Deployment of buried SP seismometer to 0.7 meters	• Pneumatic (deployment 3 axis Silicon Audio sensors and 3 MEMs sensors)	3hrs
10	SOC deployment evaluation	• Activate Engineering Camera 2 on Lander leg • take & transmit image of site post-deployment	1hr
11	Science Operations Committee (SOC) evaluation of deployment/take	• Activate Engineering Camera 3 on Lander leg • Take & transmit image of site prior to drilling	1 hr
12	Heat Flow Probe 2 deployment to 3 meters below surface	• Pneumatic Deployment system	12hrs
13	SOC deployment evaluation	• Activate Engineering Camera 3 on Lander leg • Take & transmit image of site post-deployment	1 hr
14	SOC deployment evaluation	• Activate Camera below landing deck; • Take & transmit image of site pre-VBB–SP deployment	1 hr
15	Deployment of VBB-SP package on the lunar surface beneath the center of Lander	• Lowered on crane cable below the Lander	3 hrs
16	SOC deployment evaluation	• Activate Engineering Camera below Lander deck • Take & transmit image of site post-deployment	1 hr
17	Deployment of LMSS mast magnetometer	• MAG deploy via mast on Lander deck	1hr
18	Deployment of LMSS electrodes	• EM sensors Deployed to surface via ballistic mechanisms	1 hr
19	SOC deployment evaluation	• Activate PanCam to take & transmit images of Lander deck and lunar surface site post-deployment	1hr
20	Deployment of LMSS-SCM	• Search coil deployment via boom edge of deck	1 hr
21	Deployment of LMSS Electrostatic Analyzer	• Eject protective cover (sensor is mounted directly to the deck)	1hr
22	Deployment of NGLR on Lander deck	• Deployed on the Lander deck w/ Gimbal cover mechanism release & pointing at Earth	1 hr
23	SOC deployment evaluation	• None. Determined by engineering data.	1 hr
24	SOC deployment evaluation	• Activate Engineering Camera 4 on Lander leg; • Take image of site post-deployment	1 hr
25	Deployment of NGLR on lunar surface	• Deployed on Lander leg near the surface	1 hr

site, followed by the Schickard site, the farside Korolev site, with the Crisium site being last. The Lander conditioning and instrument deployment concept of operations (con-ops) is estimated to take 50 hours. Estimated daylight at each landing site will exceed 240 hours. The instrument deployment sequence is shown in **Table B-6**. Details of each individual instrument deployment are shown in **Section 2.1.1.1**.

1.6.2 Mission Operations:

The LGN mission will require physical staffing at the MOC for critical events and issues autonomous staffing of MOC for non-critical events. In order to support the mission needs, the mission operations plan calls for a Mission Operations Centers (MOCs) and a Science Operation Center (SOC) as shown in **Figure B-12**. Both will contribute during the mission development phases and will be used to support mission simulations. The MOC and SOC personnel will work closely together, especially during the deployment phase of the mission to ensure Command Procedures and Contingency scripts are coordinated with both the Lander, Orbiter and each individual instruments. The mission operations will operate under the following assumptions:

- Real-time Communication with nearside Landers directly via DSN, for farside lander via the Orbiter and then DSN
- Assume Orbiter has Comm link with the Earth ground station (DSN)
- Assume that no ground system reconfiguration is needed to send commands to Orbiter or directly to the Landers (can send a sequence of commands that will automatically be routed to the correct destination without having to change anything on the ground)
- Assume deployment for 1 Lander can be accomplished in 4-day command window
- During Lander Descents: the Flight Operations Team (FOT) is monitoring 1 Orbiter + Stacked Landers and 1 Descending Lander and any operational Landers on Moon surface.
- Full coverage during deployments and critical activities, Real-time science data from other Landers is considered best effort during this time
- The farside Lander must be updated with the Orbiter's maneuver plans before Orbiter executes maneuver

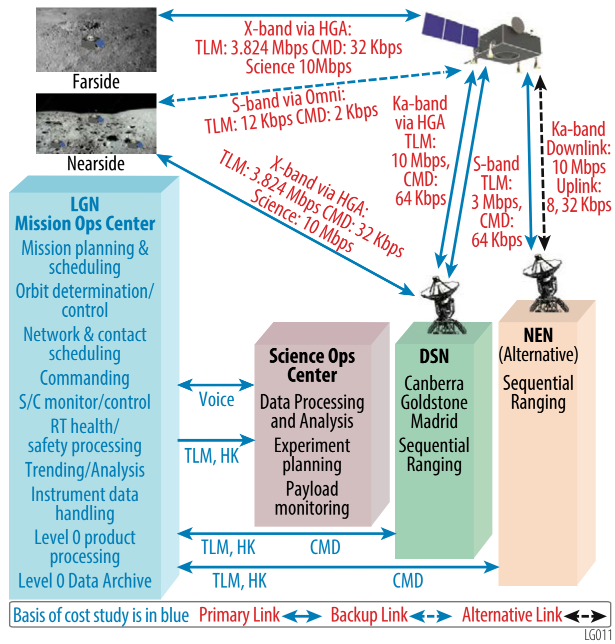


Figure B-12: Ground Systems Architecture.

The mission will utilize DSN 34m Antenna network to establish link control between the Earth and the relay Orbiter, and DTE between the near side Landers for contingency operations. The MOC will manage the following activities:

- Overall management of Ground Stations uplink/downlinks to ensure all contacts and passes are successful and no data is loss
- Receive and process of science housekeeping & Science data/TLM from the dedicated relay Orbiter
- Commanding during the daylight for uploading command loads to the lunar surface to all Landers
- Archival and distribution of science data
- Perform Lander Health and Safety checkout, then monitor SOH
- 0.1 ms timing accuracy with 10^{-6} stability relative to ground station
- Ensure mission planning command loads are processed and uplinked to the Orbiter/Lander
- Manage time correlations
- Maneuvers Support DSN passes
- Monitor Landers state of health
- Implement contingency procedures
- Implement science sequences Inventory data & re-transmit if needed
- Perform ops sim testing

The Flight Operations Team (FOT) will provide coverage for all critical events which includes, Launch and separation, Lunar Cruise, all Lander and Orbiter maneuvers (TLI, LOI, DOI, BM), Descent and Landing, deployment phase (HGA, SA and Instruments).

The MOC will initially be integrated to support mission operations development, simulations, and Integration and Test (I&T) of the Landers. The mission operations team will be integrated with the I&T process and science operations personnel to support staffing efficiencies and continuity between the I&T, launch readiness and post launch phases. In order to reduce overall workforce for the multi-Lander I&T effort, personnel will be shared between the mission operations team and I&T team. A temporary MOC will be re-deployed to the Kennedy Space Center once the Landers are shipped to the launch site where it will support final I&T and the flight operations phase of the mission. Both personnel from mission operations and science operations teams will remain there up to thirty days beyond completion of post-landing checkout and will then be returned to the I&T team.

The SOC will be utilized to operate the Landers and instruments for the twenty-day instrument commissioning phase and the six years of surface operations. The SOC only needs to accommodate a small surface operations team plus sustaining engineering, but will need to be in service for at least six years beyond launch. It will interface with the remotely-located instrument science teams allowing for distribution of health and status (H&S) telemetry and raw science data from their instruments as well as coordinating mission planning.

Table B-7: LGN Lander RF Communications Subsystem Summary of Link Analyses.

Link Information	Mission Phase 1 Flight Ops	Mission Phase 2 Post-Landing Checkout	Mission Phase 3 Instrument Commissioning	Mission Phase 4 Surface Ops
Number of Weeks for Mission Phase, weeks	2	8	4	311
Downlink Information				
Orbiter HGA to DSN 34m, Ka-Band Link				
Number of Contacts (per we week)	24	24	24	24
Average duration of contacts (minutes)	128	128	128	128
Total Daily Data Volume, per day (required/available)	36 Mb / 76 Gb	6 Gb / 76 Gb	6 Gb / 76 Gb	6 Gb / 76 Gb
Downlink Frequency Band, GHz	26.5	26.5	26.5	26.5
Downlink Telemetry Data Rate, Mbps	10 Mbps	10 Mbps	10 Mbps	10 Mbps
Transmitting Gain(s), dBi	46.27	46.27	46.27	46.27
Transmitting Power Output, Watts	3	3	3	3
Downlink Receiving Antenna Gain, dBi	76.9	76.9	76.9	76.9
Downlink Margin, dB	15.7	15.7	15.7	15.7
Orbiter HGA to DSN 34m, S-Band Link				
Number of Contacts per week	24	24	24	24
Average duration of contacts (minutes)	128	128	128	128
Total Daily Data Volume, per day (required/available)	36 Mb / 23 Gb	6 Gb / 23 Gb	6 Gb / 23 Gb	6 Gb / 23 Gb
Downlink Frequency Band, GHz	2.3	2.3	2.3	2.3
Downlink Telemetry Data Rate, Mbps	3	3	3	3
Transmitting Gain(s), dBi	24.99	24.99	24.99	24.99
Transmitting Power Output, Watts	8	8	8	8
Downlink Receiving Antenna Gain, dBi	34	34	34	34
Downlink Margin, dB	3.9	3.9	3.9	3.9
Orbiter LGA to DSN 34m, S-Band Link				
Number of Contacts per week	24	24	24	24
Average duration of contacts (minutes)	128	128	128	128
Total Daily Data Volume, per day (required/available)	17 Mb / 34 Mb	17 Mb / 34 Mb	17 Mb / 34 Mb	17 Mb / 34 Mb
Downlink Frequency Band, GHz	2.3	2.3	2.3	2.3
Downlink Telemetry Data Rate, Mbps	0.004	0.004	0.004	0.004
Transmitting Gain(s), dBi	-2.0	-2.0	-2.0	-2.0
Transmitting Power Output, Watts	8	8	8	8
Downlink Receiving Antenna Gain, dBi	55.62	55.62	55.62	55.62
Downlink Margin, dB	5.5	5.5	5.5	5.5

Lander HGA to DSN 34m, X-Band Link				
Number of Contacts per week	N/A	7	7	7
Average duration of contacts (hours)	N/A	1	1	1
Total Daily Data Volume, per day (required/available)	N/A	19 Gb / 10 Tb	19 Gb / 10 Tb	19 Gb / 10 Tb lunar day
Downlink Frequency Band, GHz	8.4	8.4	8.4	8.4
Downlink Telemetry Data Rate, Mbps	10	10	10	10
Transmitting Gain(s), dBi	30.3	30.3	30.3	30.3
Transmitting Power Output, Watts	17	17	17	17
Downlink Receiving Antenna Gain, dBi	34	34	34	34
Downlink Margin, dB	6.26	6.26	6.26	6.26
Lander LGA to DSN 34m, X-Band Link				
Number of Contacts per week	N/A	7	7	7
Average duration of contacts (hours)	N/A	1	1	1
Total Daily Data Volume, per day (required/available)	N/A	17 Gb /34 G/B	17 Gb /34 G/B	17 Gb /34 G/B
Downlink Frequency Band, GHz	8.4	8.4	8.4	8.4
Downlink Telemetry Data Rate, Mbps	0.012	0.012	0.012	0.012
Transmitting Gain(s), dBi	-2.0	-2.0	-2.0	-2.0
Transmitting Power Output, Watts	17	17	17	17
Downlink Receiving Antenna Gain, dBi	34	34	34	34
Downlink Margin, dB	3.2	3.2	3.2	3.2
Farside Lander HGA to Orbiter HGA, X-Band Link				
Number of Contacts per week	40	40	40	40
Average duration of contacts	14	14	14	14
Total Daily Data Volume, per day (required/available)	N/A	19 Gb / 336 Gb	19 Gb / 336 Gb	19 Gb / 336 Gb
Downlink Frequency Band, GHz	8.4	8.4	8.4	8.4
Downlink Telemetry Data Rate, Mbps	10	10	10	10
Transmitting Gain(s), dBi	30.3	30.3	30.3	30.3
Transmitting Power Output, Watts	17	17	17	17
Downlink Receiving Antenna Gain, dBi	35.8	35.8	35.8	35.8
Downlink Margin, dB	25.14	25.14	25.14	25.14
Farside Lander LGA to Orbiter LGA, X-Band Link				
Number of Contacts per month	40	40	40	40
Average duration of contacts, minutes	14	14	14	14
Total Daily Data Volume, per day (required/available)	N/A	17 Gb / 34 Gb	17 Gb / 34 Gb	17 Gb / 34 Gb
Downlink Frequency Band, GHz	8.4	8.4	8.4	8.4
Downlink Telemetry Data Rate, Mbps	0.35	0.35	0.35	0.35
Transmitting Gain(s), dBi	-2	-2	-2	-2
Transmitting Power Output, Watts	17	17	17	17
Downlink Receiving Antenna Gain, dBi	35.9	35.9	35.9	35.9
Downlink Margin, dB	7.4	7.4	7.4	7.4
Uplink Information				
DSN 34m to Orbiter HGA, Ka-Band Link				
Number of Uplinks per day	1	1	1	1 during lunar day 0 during lunar night
Uplink Frequency Band, GHz	23.0	23.0	23.0	23.0
Telecommand Data Rate, kbps	64	64	64	64
Receiving Antenna Gain(s), DBi	45.04	45.04	45.04	45.04
DSN 34m to Orbiter HGA, S-Band Link				
Number of Uplinks per day	1	1	1	1
Uplink Frequency Band, GHz	2.1	2.1	2.1	2.1
Telecommand Data Rate, kbps	64	64	64	64
Receiving Antenna Gain(s), DBi	24.3	24.3	24.3	24.3
DSN 34m to Orbiter LGA, S-Band Link				
Number of Uplinks per day	1	1	1	1
Uplink Frequency Band, GHz	2.1	2.1	2.1	2.1

Telecommand Data Rate, kbps	64	64	64	64
Receiving Antenna Gain(s), DBi	-2	-2	-2	-2
DSN 34m to Lander HGA, X-Band Link				
Number of Uplinks per day	1	1	1	1 during lunar day 0 during lunar night
Uplink Frequency Band, GHz	7.2	7.2	7.2	7.2
Telecommand Data Rate, kbps	32	32	32	32
Receiving Antenna Gain(s), DBi	28.9	28.9	28.9	28.9
DSN 34m to Lander LGA, X-Band Link				
Number of Uplinks per day	1	1	1	1 during lunar day 0 during lunar night
Uplink Frequency Band, GHz	7.2	7.2	7.2	7.2
Telecommand Data Rate, kbps	32	32	32	32
Receiving Antenna Type(s) and Gain(s), DBi	-2	-2	-2	-2
Orbiter HGA to Farside Lander HGA, X-Band Link				
Number of Uplinks per day	1	1	1	1
Uplink Frequency Band, GHz	7.2	7.2	7.2	7.2
Telecommand Data Rate, kbps	32	32	32	32
Receiving Antenna Gain(s), DBi	28.93	28.93	28.93	28.93
Orbiter HGA Farside Lander LGA, X-Band Link				
Number of Uplinks per day	1	1	1	1
Uplink Frequency Band, GHz	7.2	7.2	7.2	7.2
Telecommand Data Rate, kbps	8	8	8	8
Receiving Antenna Gain(s), DBi	-2	-2	-2	-2

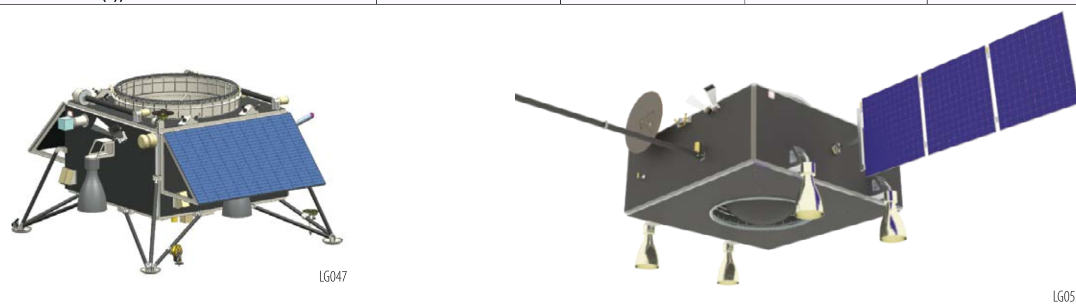


Figure B-13: The LGN Flight system.

A science data landing zone will be established and used for the distribution of data. The surface operations team will place H&S data and level 0 (raw) science data onto the server and the science teams will in turn place the level 1 (calibrated) science 23 data onto the server. The SOC will maintain an archive of all level 0 & 1 science data as well as all H&S data. Level 2 & 3 science data will be placed onto the Planetary Data System server by the science teams.

2. FLIGHT SYSTEM

The LGN flight system consist of an Orbiter and four identical Landers to support the baseline mission as shown in **Figure B-13**. The mission requirements are shown in **Table B-8** and **Table B-9**. Key requirements that drive the mission are the continuous operation of the instru-

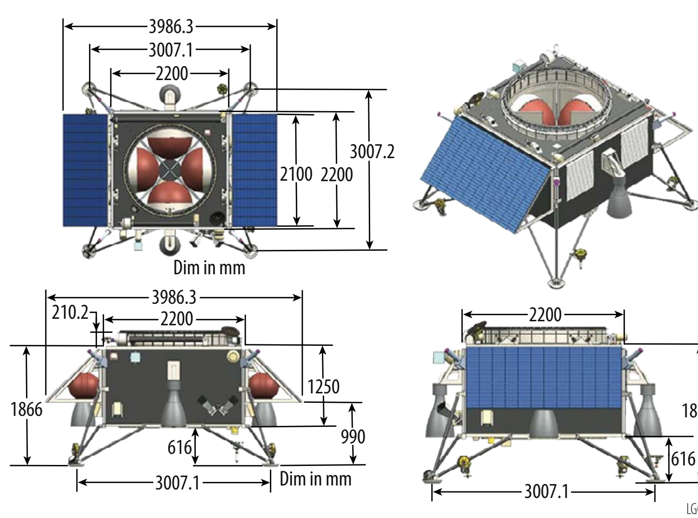


Figure B-14: Lander Dimensions Side View ments, minimizing Lander-generated vibrations, widely spaced landing sites that include a Lander on the far side and the need to maintain instruments and Lander electronics within operating temperatures while exposed to the thermal environment of the Moon.

2.1 Lander Design

Each Lander is 1.8 meters tall and 3.15 meters square. LGN Orbiter and Landers fit with the Falcon 9 heavy 5m fairing as shown in **Figure B-1**. The diameter of the fairing is within scope of standard 5m fairings used on the Delta IV, Atlas and baselined for the Block 1 SLS but the height is significantly smaller (**Figure B-21**). Baselining the smaller fairing ensures that, from a packaging standpoint, a broad spectrum of fairings are viable. **Figure B-14** shows the Lander concept. Each Lander accommodates and maintains all instruments in the harsh lunar environment.

2.1.1 Instrument Accommodations

The LGN Lander was design to host the instruments shown in **Table B-10** (except the Orbiter Magnetometer which is on the Orbiter). The mass and power of the instruments is shown in **Table B-11**. **Figure B-15** shows the location of the instruments on the Lander. On the upper deck the Fluxgate Magnetometer needs to be deployed 2m vertically. The deck is wide enough that a single 2 m boom can be stowed on top of the deck and deployed using a single drive, hinge, and latch system. This is a very simple solution with extensive heritage. The Panoramic Camera needs to be deployed vertically 1 m and can also be accommodated with a single hinged boom. The search coil magnetometer needed to be deployed horizontally 1m and a Stacer boom was chosen to avoid using up too much of the valuable side panel space. A hinge boom could likely be used as well if space permits.

Table B-9: LGN Mass Properties for Orbiter and 4 Landers on Falcon 9 Heavy Expendable.

Launch Vehicle	MEV (Kg)
Orbiter Dry Mass (MEV kg)	1,386.7
Lander Dry Mass (MEV kg)	1,263.6
Lander Propellant (kg)	1,288.3
Orbiter Wet Mass (kg)	6,683.1
Lander Wet Mass (kg)	2,551.9
Number of Landers	4.0
Total Wet Mass (kg)	10,207.7
Orbiter Propellant (kg)	5,296.4
Total DRY Mass (kg)	6,441.3
Payload Adapter (kg)	97.4
Total Launch Mass (ks)	16,988.3
Launch Vehicle Capability (kg)	15,500.0
Launch Vehicle Mass Margin (kg)	-1,488.3
Launch Dry Mass Margin (kg)	9,058.7
Dry Mass Launch Margin (kg)	140.6%
Wet Mass Launch Margin (kg)	-8.8%

Table B-10: LGN Mass Properties for Orbiter and 3 Landers on Falcon 9 Heavy Expendable.

Launch Vehicle	MEV (Kg)
Orbiter Dry Mass (MEV kg)	1,386.7
Lander Dry Mass (MEV kg)	1,263.6
Lander Propellant (kg)	1,288.3
Orbiter Wet Mass (kg)	5,629.1
Lander Wet Mass (kg)	2,551.9
Number of Landers	3.0
Total Wet Mass (kg)	7,655.8
Orbiter Propellant (kg)	4,242.4
Total DRY Mass (kg)	5,177.6
Payload Adapter (kg)	97.4
Total Launch Mass (ks)	13,382.3
Launch Vehicle Capability (kg)	15,500.0
Launch Vehicle Mass Margin (kg)	2,117.7
Launch Dry Mass Margin (kg)	10,322.4
Dry Mass Launch Margin (kg)	199.4%
Wet Mass Launch Margin (kg)	15.8%

Table B-8: Instrument details.

Item *Instrument Type*	**VBB/SP Seismometer** *Seismometer*	**SP Seismometer** *Seismometer*	**Silicon Audio/ MEMS Seis** *Seismometer*	**Heat Flow Probe** *Subsurface Deployment*	**LMSS** *Electromagnetic / Plasma*	**Lunar Reflector** *Retroreflector*	**Cameras**	**Orbiter Magnetometer**
Number of Channels								
Size/Dimensions (m x m x m)	Deployed: 0.42 x .448 On Lander: 0.42 x 0.42 x .10	.03 x (.05 x .05 x .05)	05 diameter x .20 long	Deployment: .33 x .33 x .15 Avionics: .064 x .16 x.24	LMSS-MT (Threshold): Electronics .15 x .12 x .12; Electrodes (4) .15 x .08 x .08; Magnetometer .12 x .06 x .12 LMSS-SCM (Baseline): .15 x .15 x .15 LMSS - Plasma (Baseline): .17 x .15 x .20	24 x .26 x .20	Panoramic(1): Static (4): Electronics:	Sensor: .066 x .047 x .045 Electronics: .14 x .10 x .02 Cable: 0.008 dia Boom: 2 m

Mass Without Contingency (CBE*) kg	25	1	9.4	14.2	6.1	10	9.8	2.0
Mass Contingency (%)	30	30	30	30	30	30	30	30
Mass with Contingency kg	32.5	1.3	12.3	18.5	7.9	13.0	12.7	2.6
Average Payload Power Without Contingency (W)	11	2	4.6	12.0	8.6	0	25.8	1.9
Average Payload Power Contingency (%)	30	30	30	30	30	30	30	30
Average Payload Power with Contingency (W)	14.3	2.6	6.0	15.6	11.2	0.0	33.5	2.5
Average Science Data Rate Without Contingency (kbps)	17	8	9.4	1.5	60	0.0	9.4	1
Average Science Data Rate Contingency (%)	30	30	30	30	30	30	30	30
Average Science Data Rate with Contingency (kbps)	22.1	10.4	12.2	2.0	78	0.0	12.2	1.3
Fields of View (if appropriate) (degrees)	N/A	N/A	N/A	N/A	LMSS-Plasma: 360° x 120°	180	Panoramic: 80 degree diagonal WFOV Static 40 Degree diagonal FOV.	N/A
Pointing Requirements (knowledge) degrees	N/A	N/A	N/A	N/A	N/A	10	N/A	N/A
Pointing Requirements (control) degrees	N/A	N/A	N/A	N/A	N/A	10	N/A	N/A
Pointing Requirements (stability) deg/sec	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A

Table B-11: Lander Payload Mass, Power and Mission Data Table.

	Mass			Average Power			Mission Data Volume
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)	For 6 Years (Gbits)
VBB/SP Seismometer	25	30	32.5	11	30	14.3	3,217
SP Seismometer (On Deck)	1	30	1.3	2	30	2.6	1,514
Silicon Audio/MEMS Seismometer	9.4	30	12.3	4.6	30	6.0	1,779
Heat Flow Probe (2)	14.2	30	18.5	12	30	15.6	284
Lunar Magnetotelluric Sounder Suite	6.1	30	7.9	8.6	30	11.2	11,353
Next Generation Lunar Retroreflector (2)	10	30	13.0	0.0	30	0.0	0
Close Range Imager (4) Panoramic Imager	9.8	30	12.7	25.8	30	33.5	1,779
Payload Totals	**76.5**	**30**	**99.5**	**64**	**30**	**83.2**	**19,926**
Mass and power estimates include instrument electronics and deployment systems as appropriate.							

On the lower deck is the VBB Seismometer which deploys from its mount tucked into the central cylinder down 50cm to the ground. One of the SP Seismometers is deployed along with the VBB. Instruments are also mounted on all four landing leg systems. Two heat flow probes are mounted on two radially opposite legs. The Silicon Audio MEMS Seismometer is mounted to another leg. The last leg supports one of the two NGLR. The other NGLR is mounted to the top deck. Cameras have been mounted so that each instrument that will be interacting with the ground will be within view during operation.

On the bottom deck are mounted hazard avoidance and landing sensors such as a LIDAR and Laser Rangefinder. On the top deck a gimballed antenna dish provides links to the LGN Orbiter. A small Omni antenna is also mounted the top deck and a second one is mounted on a side panel. Most heat generating boxes are mounted to a panel that includes radiators with louvers. Inside the structure the four large batteries are mounted on radials and other low power boxes throughout the various equipment panels. The four main engines are mounted to exterior panels to maintain clearance to the ground. The large radial distance of the engines from the CM of the Lander reduces the need for ACS thrusters on the bottom deck. Additional 22N thrusters are mounted to the top deck. Some shielding to protect the instruments from the main engines plume will likely be needed but this is expected to be a simple, light weight, structure such as sheet metal.

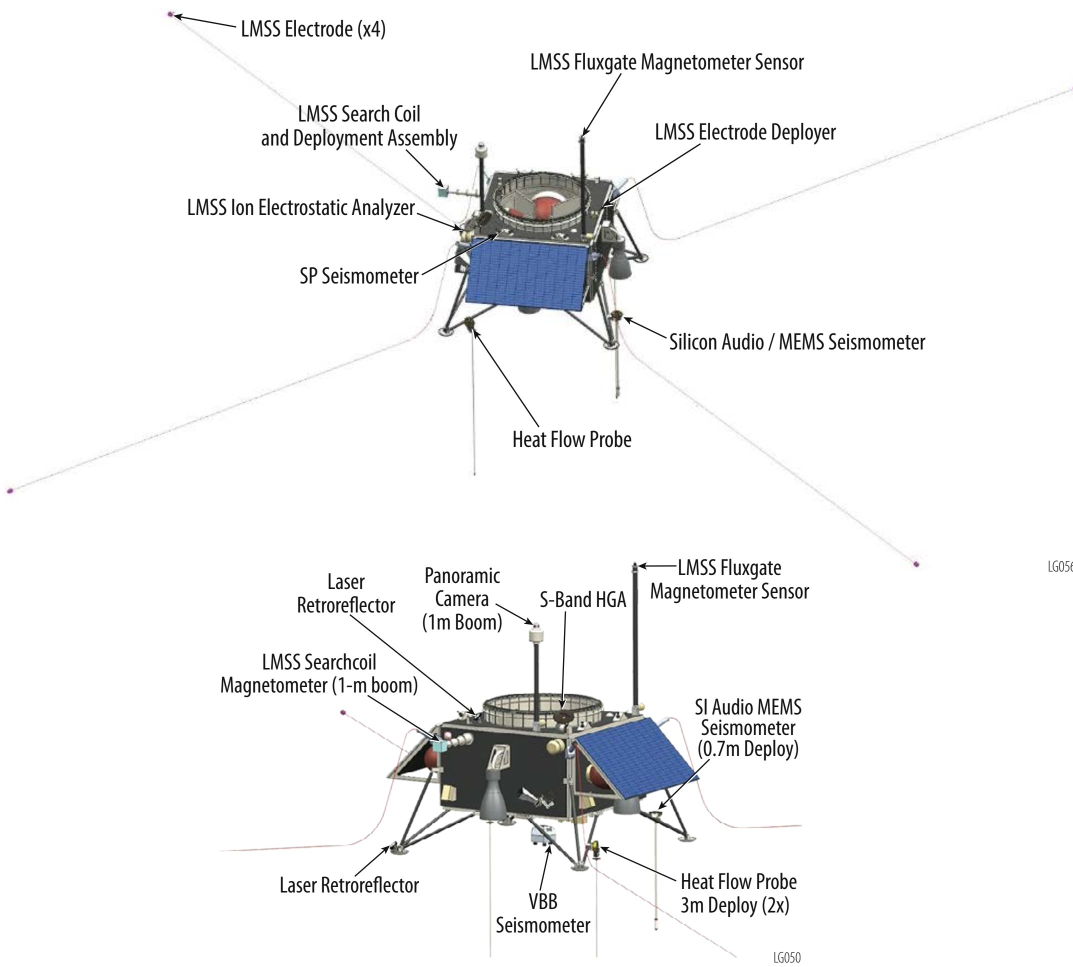


Figure B-15: LGN Lander deployed.

Very Broad Band Seismometer (VBB) (developed by CNES and IPGP in France): The proposed instrument is based on the InSight SEIS VBB seismometer (Lognonné *et al.*, 2019), with some adaptation to the Moon. The VBB on each Lander will contain 4 sensors in an oblique configuration that measures ground acceleration in the 1mHz /25 Hz bandwidth. This seismometer requires a thermal regime between -55˚C to +125˚C and will be deployed on the surface under the Lander, complete with a thermal sheath to ensure thermal stability. With InSight heritage, it is currently TRL ~7 for the Moon. The seismometer will be lowered onto the lunar surface by a "crane" developed by Honeybee Robotics, which has a single actuator, single spool, and two cables (similar to the MSL Descent Brake Deployment in spool winding). The VBB instrument will be positioned at the center of the Lander, which will provide additional thermal protection from the lunar heat by taking advantage of the Lander shading. The VBB system is lowered using stainless-steel cables (1/8-inch diameter, 24 inches) on a bridle, with two cables connected to the bridle to stabilize rotational motion as it is lowered. The cables are retracted after cable separation from the VBB is complete, and the deployment mechanism is mounted directly above the VBB on the spacecraft, with pulleys on opposite sides of the cylinder. Cable deployment heritage comes from MSL SkyCrane, Mars2020 SkyCrane, and JWST sun-shield deployment.

Short Period Seismometer (SP) (developed by Imperial College London and Oxford Instruments, England): The InSight SP sensor-head consists of a micromachined silicon sensor 25 mm on a side and front-end electronics (FEE) that have been operating continuously on Mars since November 2018. A three-axis system of these sensor heads and associated magnets can be packaged in a 5 cm X 6 cm X 6 cm box. To achieve a broad frequency response, it also uses a force feedback board (FBB). The FBB demodulates the signal received from the capacitive position-sensors on the seismometer proof mass and uses this output to drive the sensor coils and maintain the proof mass at a null point. This signal also constitutes the velocity outputs from the sensors. The velocity output from the FBB is then recorded in a backend electronics system that can be built to be common between all seismic systems in the LGN Lander. This instrument is currently TRL ~7 for the Moon.

VBB and SP have direct heritage from the InSight SEIS instrument (Lognonné *et al.*, 2019). The two major differences for VBB are a larger proof mass (249g vs. 190g), increasing the pendulum period to 5 sec, and a larger displacement transducer voltage, decreasing its electronic noise. No changes are necessary for other VBB parts, including mechanisms, and integration in a vacuum Earth-sealed sphere, which was necessary for Mars, is not required for the lunar environment. VBB and SP feedback and the back-end electronics are inherited from SEIS-InSight and from the Europa Seismic package ICEE-2 funded effort (Kedar *et al.*, 2016).

Short Period (SP) Silicon Audio Buried Seismometer (SPSAB) to study noise reduction below the lunar surface (developed by the University of Arizona, Silicon Audio, and Honeybee Robotics, US): The instrument deploys a Silicon Audio (SiA) Ultra-Low Noise seismometers in a three-axis configuration housed in a 50-mm diameter borehole sonde to study seismic activity on the Moon. The instrument's deployment mechanism allows for integration flexibility as it can be mounted to the Lander belly pan or a leg. The sonde instrument, with seismic sensors, is deployed 0.7 m into the lunar regolith by a tube deployed from a reel system consisting of interlocking stainless steel strips. The sonde advances by a combination of deployment mechanism and pneumatic jets at the nozzle that drill and displace the regolith back up to the surface of the Moon as it advances. To minimize vibrations from the Lander, the deployment structure retracts while spooling out an electrical umbilical cable to the sonde. The sensor TRL is currently 5 and the burial system is TRL 4. By the end of the DALI program in 2022, the sensor will be TRL 6 and the burial system will be TRL 5.

Heat Flow Probe (HFP) (developed by Texas Tech University and Honeybee Robotics, US): The HFP is designed to penetrate 3 m into the lunar regolith and measures the thermal gradient and thermal conductivity of the depth interval penetrated. Heat flow is obtained as a product of these two measurements. The instrument uses a pneumatic drill to penetrate into the lunar regolith. The probe deployment mechanism, mounted on the Lander's leg, spools out a boom, made of Kapton and glass fiber composite in a manner similar to a steel tape measure. Once spooled out, the boom forms a cylinder (for mechanical strength) with a penetrating cone at its leading end. The cone advances by discharging gas jets at its tip and blowing away regolith particles, while the boom actuator pushes the cone down (Zacny *et al.*, 2013). Every site chosen will have loose regolith that will be verified by LROC imagery. Two LISTER (Lunar Instrumentation for Subsurface Thermal Exploration with Rapidity; Nagihara *et al.*, 2019) heat flow probes are deployed per Lander. It has a current TRL for the Moon of ~5 and will be flown on a CLPS Lander to Mare Crisium in late 2022/early 2023.

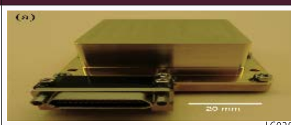
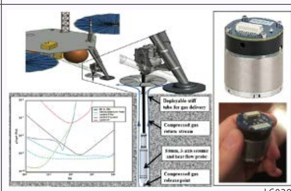
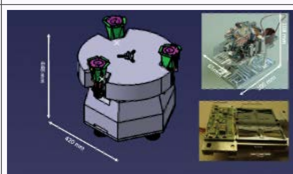
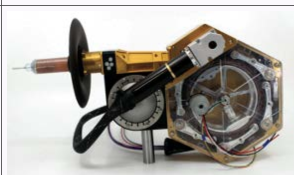
Lunar Magnetotelluric Sounding Suite (LMSS) (developed by Southwest Research Institute, the Heliospace Corporation, GSFC, and the University of California, Berkeley, US): The LMSS instrument suite is comprosied of the Lunar Magnetotelluric Sounder (LMS; **Figure B-15**) and two supplementary instruments: a search coil magnetometer (SCM) and an ion electrostatic analyzer (ESA). LMS determines the electrical-conductivity structure of the lunar interior from low-frequency magnetic and electric-field measurements. The fluxgate magnetometer is identical to the MAVEN MAG instrument. The electrometer design derives from the THEMIS EFI instrument. These sensors are integrated into LMSS using electronics based on MSL RAD. From a prior program, the magnetotelluric instrument integrated from these subsystems is at TRL 6 for Europa. LMS will achieve TRL 9 in late 2022/early 2023 during flight operations to the Moon's surface under CLPS. The SCM may improve performance at the highest frequencies in the band and is the recently-flown DSX TASC (TRL 8). The ESA provides particle measurements complementary to the electromagnetic fields. It is a new compact design (TRL 4-5) that draws high heritage components from a range of instruments including Wind 3DP, THEMIS ESA, MAVEN SWIA, and Parker Solar Probe SPAN. Final development of LMSS to TRL 6 prior to LGN CDR would require only demonstration of the ESA and updated electronics to include the SCM.

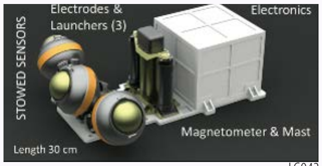
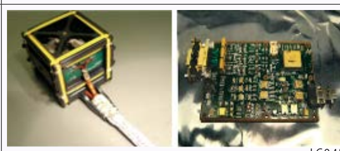
Next Generation Lunar Retroreflector (NGLR) (developed by the University of Maryland in collaboration with INFN, Italy): The NGLR consists of a large (100 mm), single Corner Cube Retroreflector (CCR), the housing, mounting structure and a pointing actuator (Currie *et al.*, 2013). Each LGN Lander will carry two retroreflectors – one mounted on the deck of the Lander and one placed on the lunar surface. Once deployed and the dust cover removed, the reflector must be pointed (two angles) towards Earth at the center of the lunar libration pattern with an accuracy of ~1 degree. This pointing is achieved using a dedicated gimbal holding the mechanical housing of the retroreflector. This payload is currently TRL 8 for the Moon and will be flown on a CLPS Lander to Mare Crisium in late 2022/early 2023.

Cameras: Each Lander will have one panoramic camera and four static cameras. These have heritage from the Mars rover missions. The panoramic camera is used to get a context image of each landing site and check LMSS deployment. The three static cameras are used to verify VBB-SP and Si-Audio seismometer deployments, as well as those of the heat flow probes and laser retroreflector on the lunar surface.

2.1.1.1 Instrument Deployments

Table B-12: Instrument Development Plans : LGN instrument development plans actively mitigate risk by advancing all instruments to TRL 6 or higher by the end of Phase A.

Name	Photo	Current TRL for the Moon	TRL Development Plan or Heritage	Deployment Risk	Mitigation / Comments
Short Period Seismometer (on the deck and on the lunar surface)	 LG038	~7	InSight mission: Currently operating on Mars.	Low - deployment entails switching on the seismometer.	N/A
Si-Audio Seismometer	 LG039	5 (sensor) 4 (burial system)	For Si-Audio, its funded by DALI to advance the TRL, we are expecting to have TRL 6 hardware by September 2022, the end of the DALI contract.	Moderate - pneumatic burial may encounter subsurface rocks.	Ensure landing sites are rock-free and devoid of young craters.
VBB-SP Seismometer Package: - 4 VBB sensors in oblique conf. - 3 SP sensors in Z-H conf.	 LG040	~7	InSight mission: Currently operating on Mars. The lunar VBB will need a larger proof mass (249g), increasing the pendulum period to 5 sec, and a larger voltage of the displacement transducer, decreasing its electronic noise.	Low - deployment is relatively simple. Need to ensure that the surface under the Lander is rock-free and relatively flat.	Ensure landing sites are rock-free and devoid of young craters. Examination of surface under previous landers (inc. Apollo) shows rocket exhaust is not an issue.
Heat Flow Probe	 LG041	~5	Will fly on the CLPS lander to Mare Crisium in late 2022/early 2023.	Low to Moderate - pneumatic burial may encounter subsurface rocks. Deployment and all other systems will have been tested on the lunar surface before LGN.	Moderate - pneumatic burial may encounter subsurface rocks. Two heat flow probes are deployed to reduce this risk.
Next Generation Lunar Retroreflector (NGLR)	 LG042	>6	Will fly on the CLPS lander to Mare Crisium in late 2022/early 2023.	Low - deployment and all other systems will have been tested on the lunar surface before LGN. Surrounding rocks may inhibit full deployment of gimbal.	Ensure landing sites are rock-free and devoid of young craters.

LMSS-MT (Threshold)	 LG043	5	Will fly on the CLPS lander to Mare Crisium in late 2022/early 2023.	Low - deployment and all other systems will have been tested on the lunar surface before LGN. Surrounding rock may inhibit full deployment of electrodes.	Ensure landing sites are rock-free and devoid of young craters.
LMSS-SCM (Baseline)	 LG044	8	Heritage: the recently flown DSX TASC.	Low - boom deployment has high heritage.	N/A
LMSS - Plasma (Baseline)		4-5	Heritage: this is a new compact design based on the MAVEN SWIA.	Low - deployment entails switching on the instrument.	N/A
Orbiter Magnetometer	 LG045	9	A magnetomater that has been proven by previous space missions will be used.	Low - high heritage instrument and deployment mechanisms will be used.	N/A

2.1.1.1.1 Heat Flow Probe Deployment Sequence

Current concept will deploy one Heat Flow Probe at a time. The deployment mechanism will target to reach 3-m depth. The desired depth will be reached by drilling in increments of 0.5-m drilling steps. After each 0.5-m drilling, the deployment sensors will perform thermal measurements of the regolith at that depth. The measurements will begin immediately after the probe completes the 0.5-m descent, and will last for 2 hours. Below is the Heat Flow probe deployment sequence which consists of the following sets of operations.

1. The probe comes out of the instrument housing and reaches the surface regolith (5 minutes)
2. Thermal measurements of the surface soil (2 hours).
3. Drill down to 0.5-m depth (5 minutes)
4. Thermal measurements of the regolith at that depth (2 hours)
5. Drill down to 1-m depth (5 minutes)
6. Thermal measurements of the regolith at that depth (2 hours)
7. Drill down to 1.5-m depth (5 minutes)
8. Thermal measurements of the regolith at that depth (2 hours)
9. Drill down to 2-m depth (5 minutes)
10. Thermal measurements of the regolith at that depth (2 hours)
11. Drill down to 2.5-m depth (5 minutes)
12. Thermal measurements of the regolith at that depth (2 hours)
13. Drill down to 3-m depth (5 minutes)
14. Thermal measurements of the regolith at that depth (2 hours)

Once the probe completes Step 14, it transitions to the post-deployment monitoring phase, in which the thermal sensors are logged once every hour. The actual data acquisition will take only 5 minutes each time, for the remaining 55 minutes the electronics will be idling.

Once Probe #1 has been fully deployed and has transitioned to the post-deployment monitoring phase, we begin deployment of Probe #2. Probe #2 will repeat the same steps (1 through 14) and then transition to the post-deployment monitoring phase.

When a night comes, we shut down Probe #2 to conserve power. After dawn of the following lunar day, Probe #2 is woken up and continues the 1-hour monitoring cycle.

2.1.1.1.2 LMSS Deployment Sequence

1. Post-Landing
2. Power on LMS
3. Deploy FGM
4. Deploy electrode 1
5. Deploy electrode 3 (yes, opposite side)
6. Deploy electrode 2
7. Deploy electrode 4
8. Deploy SCM
9. Power on ESA
10. Deploy ESA cover
11. Both instruments are receiving data
12. Downlink every 24 hrs (nearside) or every available opportunity (farside)
13. Every Lunar Sunset
14. LMS and ESA to sleep mode
15. Wake up every 5 hours
16. Acquire data for 1 hour
17. Transfer data to Lander for next available downlink
18. Sleep
19. Every Lunar Sunrise
20. LMS and ESA to normal ops

2.1.1.1.3 Deployment Mechanisms

Silicon Audio/MEMS Seismometer Drill (developed by the University of Arizona and Honeybee Robotics): The instrument deploys SiA Ultra Low Noise seismometers in 3-C configuration housed in a 50 mm diameter borehole sonde to study seismic activity on the Moon. The deployment mechanism for the instrument allows for integration flexibility as it can be mounted either to the Lander leg or belly pan of the Lander. The sonde instrument, with seismic sensors, is deployed 0.7m into the lunar regolith by rigid tubular mast that consists of interlocking stainless steel strips. The sonde advances by a combination of deployment mechanism and pneumatic jets at the nozzle that drills and displaces the regolith back up to surface of the moon as it advances. To minimize vibrations from the Lander, deployment structure retracts while spooling out an electrical umbilical cable to the sonde. Two instruments will be deployed per Lander. This has a current TRL for the Moon of ~4 and will be developed for TRL 6 by the end of the current DALI effort by 2022.

2.1.1.1.4 VBB crane lowering mechanism:

The deployment mechanism for placing the VBB on the lunar surface uses a single actuator, two cables and a spool similar to MSL Descent Brake winding. An electrical service loop maintains the electrical connection between the VBB and Lander. Stainless-steel cables on a bridal gently lower the system and release the VBB through two pin puller releases once successful surface placement is confirmed. The cables are retracted after cable separation from the VBB is complete. Deployment mechanism is mounted directly above the VBB on the space craft with pulleys on opposite sides of the cylinder.

2.1.2 Communications

The LGN Lander RF Communications subsystem utilizes transceivers compatible with the Deep Space Network (DSN) earth terminals and the user modems being designed for the LGN orbiting store and forward communication spacecraft. X-band was selected because it provides sufficient bandwidth to offload the mission data volume within the allocated contact time with the orbiting spacecraft and meets the bandwidth allocations for direct to Earth allocations. X-band 17W RF amplifiers have been incorporated with a 0.5m High Gain Antenna (HGA) to provide the necessary radiated power for direct to Earth contacts. A single +Z Low Gain Antenna (LGA) is also included allowing for contingency or emergency communication contacts. The 0.5m HGA is gimballed to permit horizon to horizon contacts with the Orbiter. When not in active support of Orbiter communications, the HGA will likely be parked in the direction of Earth. A functional block diagram of the Lander RF communications subsystem is included as **Figure B-16**.

2.1.2.1 LGN Lander RF Communications Link Analysis

The LGN Lander RF Communications subsystem would use transceivers compatible with the Deep Space Network (DSN) Earth. X-band was selected because it provides sufficient bandwidth to offload the mission data volume within the allocated contact time with DSN and meets the bandwidth allocations for direct to Earth allocations. X-band 17 W RF amplifiers would be incorporated with a 0.5m High Gain Antenna (HGA) to provide the necessary radiated power for direct to Earth contacts. A single +Z Low Gain Antenna (LGA) would be also included allowing for contingency or emergency communication contacts. The 0.5m HGA would be gimballed to permit contact with DSN during the 5 day transfer from Earth to the Moon and during the lunar surface mission. A functional block diagram of the Lander RF communications subsystem is included as **Figure B-16**.

Link budgets show 10 Mbps with 6.3 dB margin with DSN. The use of the Universal Space Network 11m class systems was evaluated and showed it provides 1.0 Mbps with 5.7 dB of margin. With the daily downlink volume at 17 Gbits/day the USN would be a viable alternative to DSN. **Table B-7** summarizes the communication link analyses. **Table B-13** provides the link analysis for DSN links. A table summarizing the communication link analyses is included as **Table B-13**. Details on each link are shown in **Figure B-17, Figure B-18, Figure B-19,** and **Figure B-20**.

The communication subsystem design would use high-TRL class transceivers and power amplifiers. The HGA is available as a commercial product and the gimbals would be similar to the ones developed for Euclid (Launch 2022). No delivery risks would be expected for a 2030 expected launch date. A double pull double throw (DPDT) switch would be included after the diplexers allowing one remaining comm slice access to both antenna systems. Communication subsystem mass, power, and cost estimates were provided elsewhere for project tracking. A great deal of international lunar communication coordination is still in development. For example, the NASA IOAG "The Future Lunar Communications Architecture", 10/2019 study presently limits X-band downlink symbol rates to 4 Mbps with GMSK

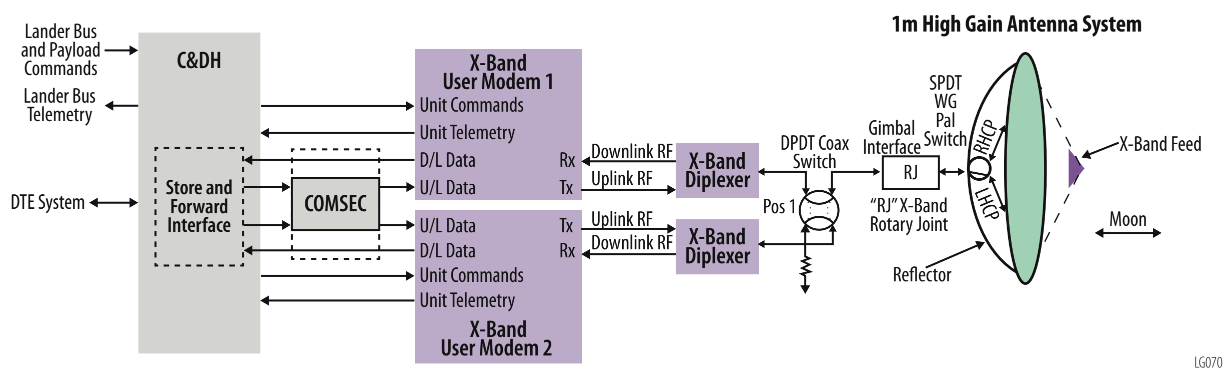


Figure B-16: LGN Lander RF Communications Subsystem Functional Block Diagram.

Table B-13: LGN Lander RF Communications Subsystem Summary of Link Analyses.

Link	Direction	From	Power (W)	Antenna	To	Antenna	Rate (kbps)	Margin (db)
Near Side Lander Links								
1	Downlink	Lander	17	LGA	DSN	34 m	12.0	3.2
2	Uplink	DSN	1000	34 m	Lander	LGA	32.0	8.9
3	Downlink	Lander	17	HGA	DSN	34 m	10,000	6.3
4	Uplink	DSN	200	34 m	Lander	HGA	32	33.5
5	Downlink	Lander	17	LGA	USN	11 m	1.0	3.5
6	Uplink	USN	1000 W	11 m	Lander	LGA	32.0	8.9
7	Downlink	Lander	17	HGA	USN	11	10,000	5.7
8	Uplink	USN	200 W	11 m	Lander	HGA	32.0	23.7
Farside Lander Links								
9	Downlink	Lander	17	HGA	Orbiter	HGA	10,000	25.1
10	Uplink	Orbiter	0.5	HGA	Lander	HGA	32	29.2
11	Downlink	Lander	17	LGA	Orbiter	HGA	350	7.4
12	Uplink	Orbiter	0.5	HGA	Lander	LGA	8	3.7

Lander X-band Downlinks | **Lander X-band Uplinks**

8.4 GHz Downlink Lander LGA to DSN 34m
410416 km Range
LDPC 7/8

17 W	12.30	HPA pwr (dBW)	7.30 EIRP
	-3.00	pass loss (dB)	
	-2.00	Lander ant gain (dBi)	
	-223.19	path loss (dB)	
	-0.50	pointing + pol loss (dB)	
34.00 m	66.92	DSN ant gain (dBi)	38.63 G/T
1.0 dB NF	-1.00	pass loss (dB)	
26.9 dBK	-27.29	Ant temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
12 ksps	-40.79	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	3.20	Margin (dB)	

5

7.2 GHz Uplink DSN 34m to Lander LGA
410416 km Range
No FEC

1000 W	30.00	HPA pwr (dBW)	94.08 EIRP
	-1.50	pass loss (dB)	
34.00 m	65.58	DSN ant gain (dBi)	
	-221.85	path loss (dB)	
	-0.50	pointing + pol (dB)	
	-2.00	Lander ant gain (dBi)	-33.76 G/T
2.0 dB NF	-3.00	pass loss (dB)	
24.7 dBK	-28.76	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
32.0 kbps	-45.05	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	8.92	Margin (dB)	

6

8.4 GHz Downlink Lander HGA to DSN 34m
410416 km Range
LDPC 7/8

17 W	12.30	HPA pwr (dBW)	39.57 EIRP
	-3.00	pass loss (dB)	
0.50 m	30.27	Lander ant gain (dBi)	
	-223.19	path loss (dB)	
	-0.50	pointing + pol (dB)	
34.00 m	66.92	DSN ant gain (dBi)	38.63 G/T
1.0 dB NF	-1.00	pass loss (dB)	
26.9 dBK	-27.29	Ant temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
10.0 Msps	-70.00	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	6.26	Margin (dB)	

7

7.2 GHz Uplink DSN 34m to Lander HGA
410416 km Range
No FEC

200 W	23.01	HPA pwr (dBW)	87.09 EIRP
	-1.50	pass loss (dB)	
34.00 m	65.58	DSN ant gain (dBi)	
	-221.85	path loss (dB)	
	-0.50	pointing + pol (dB)	
0.50 m	28.93	Lander ant gain (dBi)	-2.23 G/T
2.0 dB NF	-3.00	pass loss (dB)	
20.0 dBK	-28.16	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
32.0 kbps	-45.05	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	33.47	Margin (dB)	

8

LG077

Figure B-17: Lunar Lander to DSN Communication Link Analyses.

Lander X-band Downlinks | **Lander X-band Uplinks**

8.4 GHz Downlink Lander HGA to USN 11m
410416 km Range
LDPC 7/8

17 W	12.30	HPA pwr (dBW)	39.57 EIRP
	-3.00	pass loss (dB)	
0.50 m	30.27	Lander ant gain (dBi)	
	-223.19	path loss (dB)	
	-0.50	pointing + pol (dB)	
11.00 m	57.12	DSN ant gain (dBi)	28.09 G/T
2.0 dB NF	-2.00	pass loss (dB)	
26.9 dBK	-27.03	Ant temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
1.0 Msps	-60.00	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	5.72	Margin (dB)	

7a

7.2 GHz Uplink USN 11m to Lander HGA
410416 km Range
No FEC

200 W	23.01	HPA pwr (dBW)	77.29 EIRP
	-1.50	pass loss (dB)	
11.00 m	55.78	DSN ant gain (dBi)	
	-221.85	path loss (dB)	
	-0.50	pointing + pol (dB)	
0.50 m	28.93	Lander ant gain (dBi)	-2.23 G/T
2.0 dB NF	-3.00	pass loss (dB)	
20.0 dBK	-28.16	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
32.0 kbps	-45.05	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	23.67	Margin (dB)	

8a

8.4 GHz Downlink Lander LGA to USN 11m
410416 km Range
LDPC 7/8

17 W	12.30	HPA pwr (dBW)	7.30 EIRP
	-3.00	pass loss (dB)	
	-2.00	Lander ant gain (dBi)	
	-223.19	path loss (dB)	
	-0.50	pointing + pol loss (dB)	
11.00 m	57.12	DSN ant gain (dBi)	28.09 G/T
2.0 dB NF	-2.00	pass loss (dB)	
26.9 dBK	-27.03	Ant temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
1 ksps	-30.00	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	3.45	Margin (dB)	

5a

7.2 GHz Uplink DSN 34m to Lander LGA
410416 km Range
No FEC

1000 W	30.00	HPA pwr (dBW)	94.08 EIRP
	-1.50	pass loss (dB)	
34.00 m	65.58	DSN ant gain (dBi)	
	-221.85	path loss (dB)	
	-0.50	pointing + pol (dB)	
2.0 dB NF	-2.00	Lander ant gain (dBi)	-33.76 G/T
24.7 dBK	-3.00	pass loss (dB)	
290K sys	-28.76	Ant temp (dB)	
	228.60	Boltz (dBW/Hz-K)	
32.0 kbps	-45.05	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	8.92	Margin (dB)	

6a

LG090

Figure B-18: Lunar Lander to USN Communication Link Analyses.

Lander X-band Downlinks / **Lander X-band Uplinks**

8.4 GHz Downlink Lander LGA to DSN 34m
410416 km Range
LDPC 7/8

17 W	12.30	HPA pwr (dBW)	7.30 EIRP
	-3.00	pass loss (dB)	
	-2.00	Lander ant gain (dBi)	
	-223.19	path loss (dB)	
	-0.50	pointing + pol loss (dB)	
34.00 m	66.92	DSN ant gain (dBi)	38.63 G/T
1.0 dB NF	-1.00	pass loss (dB)	
26.9 dBK	-27.29	Ant temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
12 ksps	-40.79	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	3.20	Margin (dB)	

5

7.2 GHz Uplink DSN 34m to Lander LGA
410416 km Range
No FEC

1000 W	30.00	HPA pwr (dBW)	94.08 EIRP
	-1.50	pass loss (dB)	
34.00 m	65.58	DSN ant gain (dBi)	
	-221.85	path loss (dB)	
	-0.50	pointing + pol (dB)	
	-2.00	Lander ant gain (dBi)	-33.76 G/T
2.0 dB NF	-3.00	pass loss (dB)	
24.7 dBK	-28.76	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
32.0 kbps	-45.05	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	8.92	Margin (dB)	

6

8.4 GHz Downlink Lander HGA to DSN 34m
410416 km Range
LDPC 7/8

17 W	12.30	HPA pwr (dBW)	39.57 EIRP
	-3.00	pass loss (dB)	
0.50 m	30.27	Lander ant gain (dBi)	
	-223.19	path loss (dB)	
	-0.50	pointing + pol (dB)	
34.00 m	66.92	DSN ant gain (dBi)	38.63 G/T
1.0 dB NF	-1.00	pass loss (dB)	
26.9 dBK	-27.29	Ant temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
10.0 Msps	-70.00	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	6.26	Margin (dB)	

7

7.2 GHz Uplink DSN 34m to Lander HGA
410416 km Range
No FEC

200 W	23.01	HPA pwr (dBW)	87.09 EIRP
	-1.50	pass loss (dB)	
34.00 m	65.58	DSN ant gain (dBi)	
	-221.85	path loss (dB)	
	-0.50	pointing + pol (dB)	
0.50 m	28.93	Lander ant gain (dBi)	-2.23 G/T
2.0 dB NF	-3.00	pass loss (dB)	
20.0 dBK	-28.16	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
32.0 kbps	-45.05	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	33.47	Margin (dB)	

8

LG077

Figure B-19: LGN Lander to DSN Communication Link Analyses.

Lander X-band Downlinks / **Lander X-band Uplinks**

8.4 GHz Downlink Lander HGA to Orbiter HGA
250 km Orbit = 825 km Range
LDPC 7/8

17 W	12.30	HPA pwr (dBW)	39.57 EIRP
	-3.00	pass loss (dB)	
0.50 m	30.27	Lander ant gain (dBi)	
	-169.26	path loss (dB)	
	-0.50	pointing + pol loss (dB)	-130.2 prec
1.00 m	35.88	Orbiter ant gain (dB)	3.58 G/T
2.0 dB NF	-3.00	pass loss (dB)	
26.9 dBK	-29.29	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
10.0 Msps	-70.00	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	25.14	Margin (dB)	

1

7.2 GHz Uplink Orbiter HGA to Lander HGA
250 km Orbit = 825 km Range
No FEC

0.5 W	-3.01	HPA pwr (dBW)	28.94 EIRP
	-3.00	pass loss (dB)	
1.00 m	34.95	Orbiter ant gain (dBi)	
	-167.92	path loss (dB)	
	-0.50	pointing + pol (dB)	
0.50 m	28.93	Lander ant gain (dBi)	-2.23 G/T
2.0 dB NF	-3.00	pass loss (dB)	
20.0 dBK	-28.16	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
32.0 kbps	-45.05	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	29.25	Margin (dB)	

2

8.4 GHz Downlink Lander LGA to Orbiter HGA
250 km Orbit = 825 km Range
LDPC 7/8

17 W	12.30	HPA pwr (dBW)	7.30 EIRP
	-3.00	pass loss (dB)	
	-2.00	Lander ant gain (dBi)	
	-169.26	path loss (dB)	
	-0.50	pointing + pol loss (dB)	
1.00 m	35.88	Orbiter ant gain (dB)	3.58 G/T
2.0 dB NF	-3.00	pass loss (dB)	
26.9 dBK	-29.29	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
350 ksps	-55.44	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	7.43	Margin (dB)	

3

7.2 GHz Uplink Orbiter HGA to Lander LGA
250 km Orbit = 825 km Range
No FEC

0.5 W	-3.01	HPA pwr (dBW)	28.94 EIRP
	-3.00	pass loss (dB)	
1.00 m	34.95	Orbiter ant gain (dBi)	
	-167.92	path loss (dB)	
	-0.50	pointing + pol (dB)	
	-2.00	Lander ant gain (dBi)	-33.76 G/T
2.0 dB NF	-3.00	pass loss (dB)	
24.7 dBK	-28.76	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
8.0 kbps	-39.03	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	3.73	Margin (dB)	

4

LG078

Figure B-20: LGN Lander to Orbiter Communication Link Analyses.

modulation formats. The Lander currently baselines 10 Mbps, Staggered quadrature phase-shift keying (SQPSK), and low-density parity-check (LDPC) 7/8 formats. Coordination with the Lunar Community would be required to use the higher data rate and LDPC to avoid a DSN contact time of 22.5 hours.

2.1.3 Structures

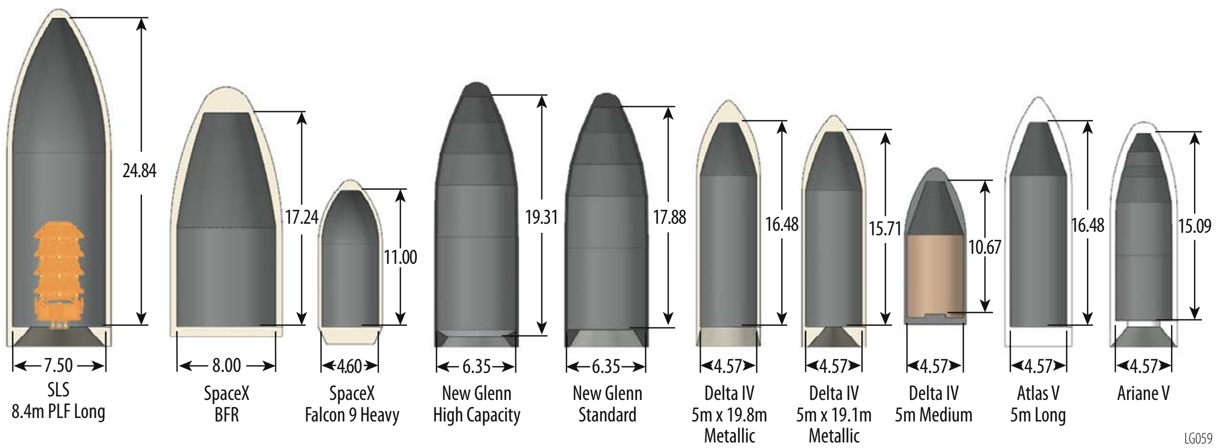


Figure B-21: Launch Vehicle Fairings.

The packaging of all the elements of the LGN launch stack was driven by the Falcon 9 heavy fairing size. The diameter of the fairing is within scope of standard 5m fairings used on the Delta IV, Atlas and baselined for the Block 1 SLS but the height is significantly smaller. Baselining the smaller fairing ensures that, from a packaging standpoint, a broad spectrum of fairings are viable. All current launch vehicles have a limited center of mass offset. Although this was not a specific design driver, it was a consideration that nudged the design to as squat a height as possible. This had the additional advantage of a more efficient mass design. **Figure B-22** show the 3 lander and 4 lander configuration fit within the fairing.

The multi-Lander stack is built around a series of co-axial central cylinder segments with a 1666mm bolt diameter interconnected with notional separation rings such as Marmon clamp bands or light band. These are considered notional since no attempt was made to look at other mount/release systems. They were merely used to provide a mass within range of viable options with the expectation that detailed design of these would be future work all well within the state-of-the-art. This concept was based on MMS heritage. The MMS central cylinder diameter of 1666mm does not match the 1575mm standard Falcon 9 interface but is consistent with other launch vehicles in the same class. The slightly larger diameter was chosen to maintain similarity with MMS design. It seemed reasonable that the standard 1666mm interface or transition adapters will be available for the Falcon 9 in the future. Switching to a 1575mm diameter should not have a significant effect on the design if that is required.

Reducing the overall stack height, and thereby reducing the Center of Mass (CM) height, and increasing the number of Landers that could fit in the smaller Falcon 9 fairing, required the outboard placement of the main engines. This had the add-

Figure B-22: LGN Three Lander Launch Configuration (left), LGN Four Lander Launch Configuration (right).

ed benefit of providing significant ACS capability. Packaging the propulsion tank sizes drove many aspects of the mechanical design along with trying to avoid deployable systems where possible. The concept developed has fixed solar arrays and non-deploying legs on the Lander. This greatly simplifies the design but does make packaging a challenge particularly because of the requirement to have 50cm clearance from the bottom of the landing legs to the bottom deck of the Lander. The resulting stack has the legs slightly nested over the Lander below.

The goal of each Lander is to safely deliver the instrument suite to the moon's surface. The Landers are intended to be identical. **Figure B-23** show the Lander in its landed configuration prior to deployments. The landing sites are assumed to be well defined but will have variability. Therefore the design must be robust enough to accommodate different landing environments with a single design. From a packaging standpoint, the structure is heavily driven by the four primary propulsion tanks (two fuel and two oxidizers) which consume a substantial portion of the Lander volume. Smaller pressurant tanks are also required but easily accommodated. The overall height constraints of the launch stack drove the decision to mount the tanks around the central cylinder in the bus structure plane. By imbedding the tanks into the central cylinder the center of mass for the tanks and propellant/oxidizer was moved to directly over the primary load path of the central cylinder walls. The additional mass of the central cylinder tank mounting rings is likely offset by the mass savings of the Lander bus structure not needing to carry the propellant loads directly. This should resulting in an overall mass savings, but further analysis is needed. The other advantage of moving the tanks inward toward the centerline is a significant reduction in the primary Lander structure footprint. This allows the mounting of fixed solar arrays and eliminates the need to deploy the legs. The savings in complexity and cost of not deploying either should be significant.

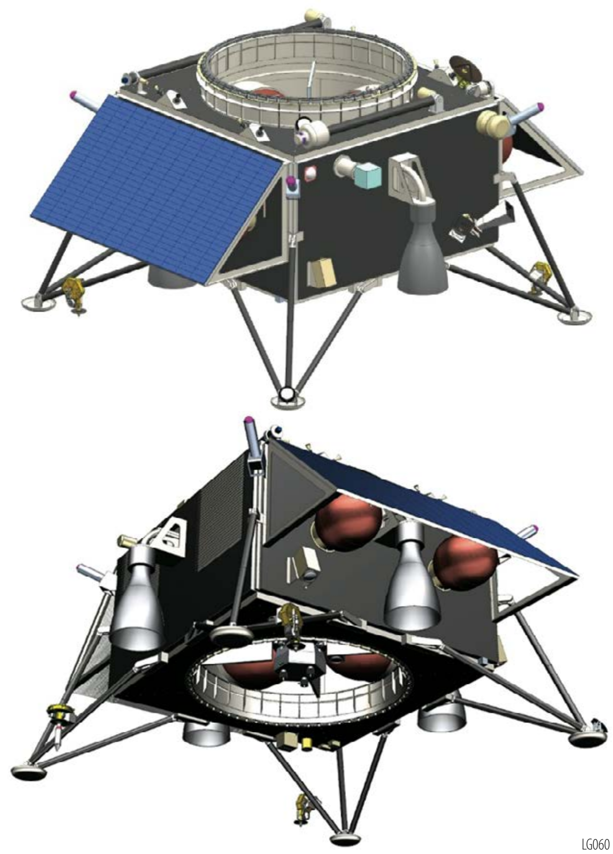


Figure B-23: Lander Stowed View.

The legs are simple tripod design similar to Viking or Surveyor. The upper leg tube consists of two telescoping tubes and a crushable material cartridge which provides energy absorption on impact if needed. No leveling system has been added to this design but a stabilizing solution will likely be required. There is the possibility of a small amount of rocking if one leg is off the ground and the CM is near the tip line of remaining legs. It is assumed that site selection and hazard avoidance capability will make it possible to choose a very benign landing area obviating the need for active leveling. A well characterized landing surface cannot guarantee that all potential for rocking has been eliminate. Rocking could be a factor when the drilling probes are deployed or when the antenna gimbal is moving. A simple solution may be a spring/detent, with a damper, to allow controlled, discrete, compression of the telescoping upper leg or as part of the foot pad. The damper reacts the initial landing loads and then relaxes to allow the mass of Lander to engage the spring/detent system. If the spring/detent settling set-mass is one-third of the landing mass, it will continue to settle until all legs are below the set mass, which will only happen when the fourth leg is engaged with the ground. Although not completely passive, it is a non-actuated stabilizing solution allowing the legs to essentially sag a small distance into a final resting position after landing ensuring all landing pads are engaged with the ground. There are many other self-stabilizing concepts for four legs on uneven surfaces which could be adapted so it is not considered a particularly difficult challenge to solve, and as such was not pursued beyond recognition of the need.

The Lander engine cut-off is expected to occur 5m above the moon's surface with a decent velocity at cut-off of 0.5m/s. This results in an impact velocity of approximately 4.5m/s. At this velocity the stroke of the legs during impact must be at least 87mm to keep the landing g-loads at or below axial loads requirement of 12g. This exceeds the Falcon 9 maximum launch loads of 6g axial and becomes the driving case. The leg geometry requires the telescoping upper legs to compress a minimum of 58mm to achieve the minimum vertical stroke distance desired. The worst case scenario for keeping landing loads down is if all four legs were to hit at the same time. The crush force of the energy absorbing material must be low enough to maintain the minimum vertical stroke distance in that situation. Assuming a dry mass of 1257.4kg and 200kg of residual propellant for a total landing mass of 1457.4kg, the expected mass kinetic energy (KE) in the system 14.9kJ. This energy must be absorbed upon impact to obviate any unwanted landing dynamics. To achieve this goal, the crushable material cartridge in the leg must be tuned to a crush strength of 31.8MPa for the current leg geometry. This can be achieved with material selection and crush cartridge geometry. Many options are available for flight qualified crushable material. The other landing scenario is only three legs absorbing the impact KE. The crush strength of the energy absorbing material is set by the four leg case. The resulting compression of the crushable material in the three leg case results in a vertical stroke distance of 116mm and softer landing load of 9gs. This defines the location of the pre-landing foot pads below the lower deck 616mm to achieve a minimum clearance of 500mm after landing. These calculations ignore any post-landing, self-stabilizing additions to the leg design as mentioned above.

Like the Orbiter, the Lander structure was designed as a structural box with composite face sheet/aluminum honeycomb core panels using the clip and post method, a central cylinder, radial and equipment panels. Unlike the Orbiter, the propellant tanks (both fuel and oxidizer) are imbedded in four radially spaced rings in the central cylinder providing a direct load path for the fuel/oxidizer mass to the launch vehicle. Mounting the tanks in the central cylinder was driven by the need to accommodate a significant amount of tankage while keeping the Lander profile small in both height and width. Preliminary hand calculations of the structure in the bottom most Lander, which is carrying the most mass, suggests that the central cylinder axial loading and first mode is adequate without considering the rest of the box structure. The lateral stiffness is adequate only when including the surrounding box structure. This suggests that the structural design, while very preliminary, is reasonably conservative. A detailed structural analysis is left for future work.

The structural concepts developed for this study are within the current state of the art and have heritage. It is likely that landing dynamics, KE absorption and self-stabilizing system will require customization for the specific landing environment, but the technology maturity is high with numerous planetary landings providing heritage and extensive methodologies.

2.1.4 Propulsion

The LGN Lander propulsion subsystem would be a large regulated bi-propellant system. The propellant would be stored in COTS tanks, and the main engines would be a set of four 4,000 N engines (AJ PN R-40B, see **Figure B-24**). Three redundant sets of 2 x 450 N engines (AJ PN R-4D-15 HIPAT, see **Figure B-24**) would be used for attitude control during maneuvers. Separate pressurization manifolds would be used to provide regulated pressure to both the fuel and oxidizer tanks. The system would be single fault tolerant. Each pressurization string would be fully redundant, and there would be separate strings of redundant attitude control thrusters. A schematic of the subsystem is shown in **Figure B-25**.

Figure B-24: R-40B 4,000N and R-4D-15 450N (HiPAT) rocket engines (not to scale).

The pressurant tanks would be isolated by redundant pyro valves during launch. The system would be pressurized during the transfer to lunar orbit insertion and a calibration maneuver would be performed. All maneuvers would be performed with the main engines, except for smaller orbit maintenance maneuvers.

All of the components would be COTS. All of the components in the system are TRL-9. Several of the components are long-lead items, in particular the engines, pressure transducers, regulators, and tanks.

The system is single fault tolerant, with the exception of the main engine. Each pressurization string is fully redundant, and there are separate strings of redundant attitude control thrusters.

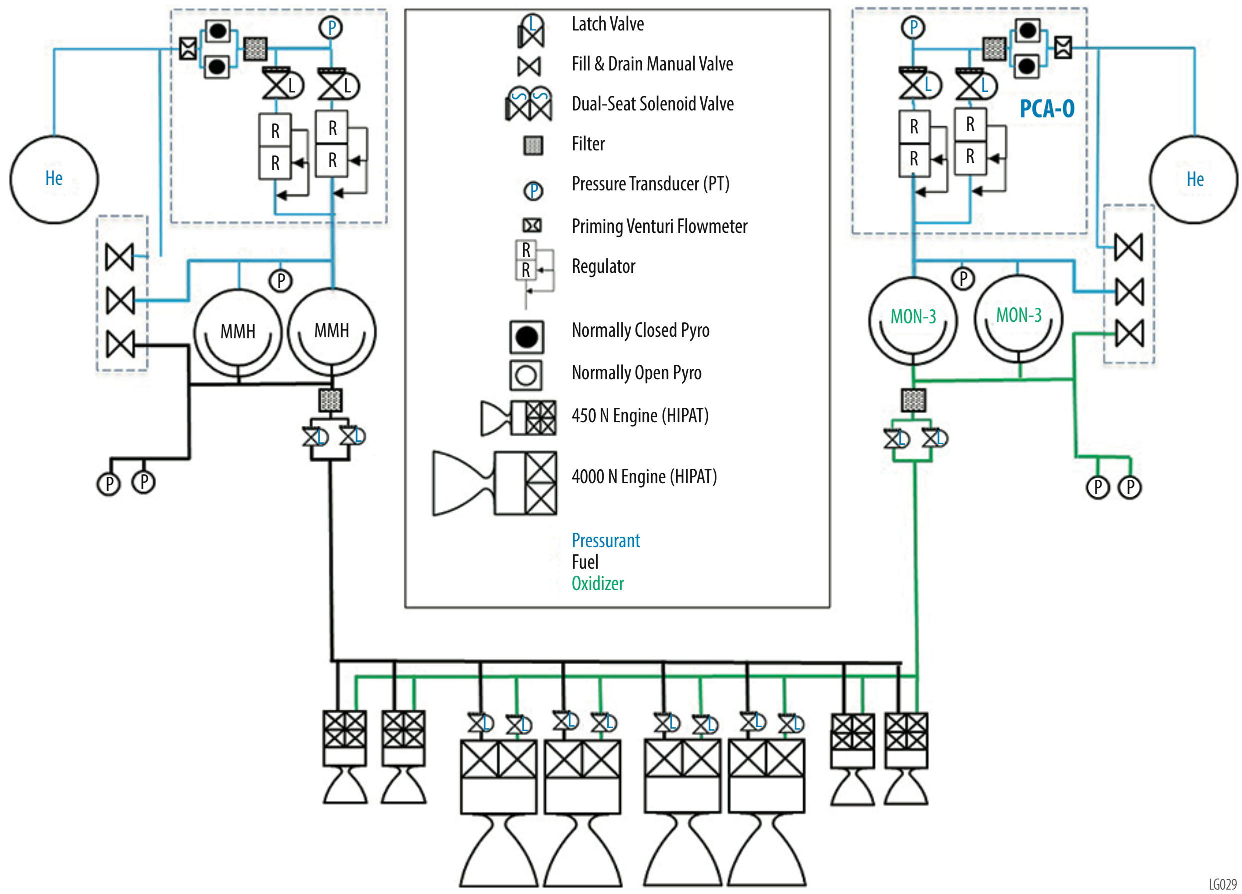


Figure B-25: Schematic of the Propulsion Subsystem.

2.1.5 Attitude Control System (ACS)

Figure B-26 shows the LGN release of landers and their descent and landing maneuvers. The navigation portion of the Lander Flight Software (FSW) would be required to reduce the errors within the decent corridor. This part of the FSW contains the Terrain Relative Navigation (TRN) used during the final portion of decent. Since there would not be ground in the loop, the navigation algorithm must be robust to account for uncertainties and disturbances.

The Lander ACS would have three modes (**Figure B-27**): Stowed, ΔV, and Sun Safe (Sun Acq, Rate Null). The ΔV mode would receive attitude and positions targets from the navigation algorithm. This would be a deviation from most heritage ΔV modes and will require some new flight software. Flight software costs were accounted for using standard "wraps" in the costing exercise. The other modes would be based on heritage algorithms.

The LGN Lander transports the science instrument platform to surface target ellipse with a required clocking angle and orientation relative to the local gravity field. After safely landing, stabilizing, and initializing, the Lander ACS is no lon-

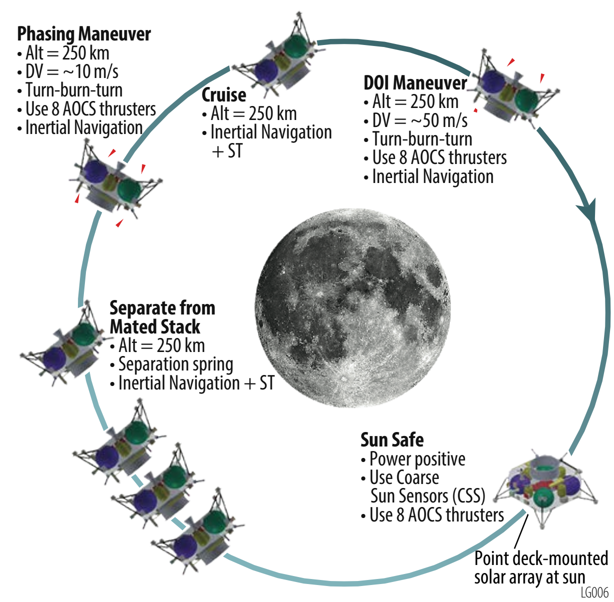


Figure B-26: LGN Stack and Release Maneuvers.

ger needed. The Lander's Guidance Navigation and Control (GN&C) system, which works in conjunction with ACS, is significantly more complicated than the Orbiter and has a shorter life span. The driving attitude requirements for the Lander are looser (two orders of magnitude) than the Orbiter. The navigation requirements are:

- Lander final actual position within 1 km of target site
- Lander final position knowledge within 2 km
- Lander final orientation relative to gravity (nadir) < 5 deg
- Lander velocity at touchdown < 0.5 m/s @5m vertical, < 0.1 m/s horizontal

All the components of the ACS system have flight heritage and have been used in algorithms that are similar to the proposed algorithms for this mission. The laser rangefinder and the Optical/IR Camera would be used in the navigation algorithms to provide positional knowledge of the lander. The position information would be used in closed-loop navigation and the Terrain Navigation (TRN) controller.

Lander concept of operations would be based on heritage and new algorithms. The new algorithms would be developed based on earth reentry vehicles and proposed landers from missions such as Mars 2020 and applied to the lunar case. Prior to separation, the lander would be initialized with the stack configuration states. After separation and rate null, the lander ACS would start the deorbit maneuver the vehicle while maintaining the desired attitude. The Lander ACS block diagram is provided in **Figure B-28**.

The DeltaV mode, which is the main mode of operation, receives navigation commands associated with the ground commands and uses the navigation algorithm to provide navigation estimates (Terrain Relative Navigation) of the position and attitude relative to the local frame. This is a deviation from most heritage DeltaV modes. The other modes are based on heritage algorithms. The primary purpose the Lander is to descend from orbit and land on the Lunar surface. The decent and landing maneuver is described in **Figure B-29**.

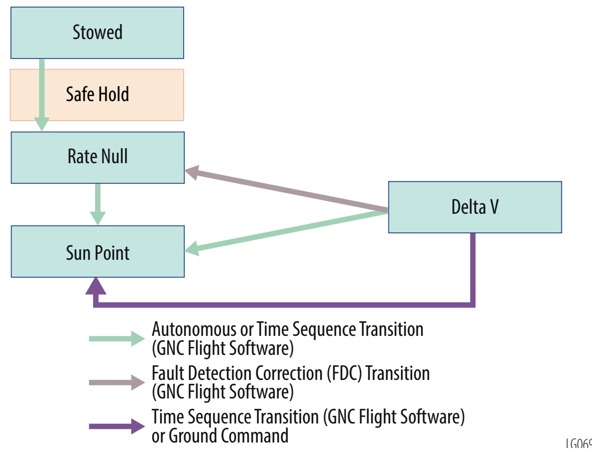


Figure B-27: Lander Modes.

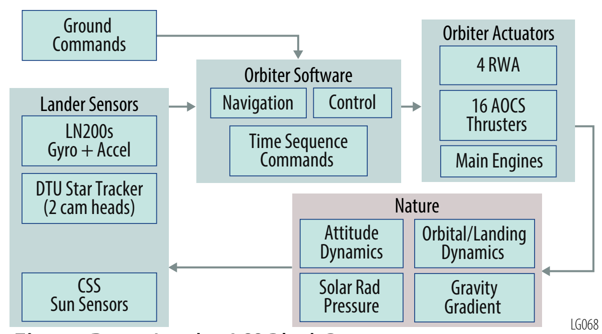


Figure B-28: Lander ACS Block Diagram.

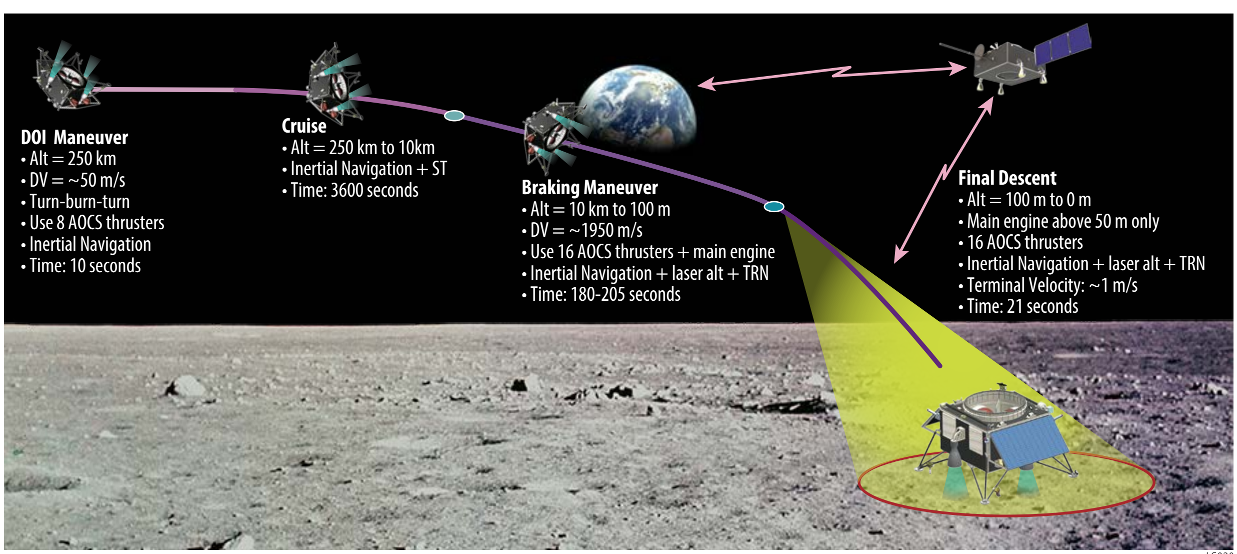


Figure B-29: LGN descent profile.

The DOI is the first maneuver the Lander performs. The turn and burn maneuver results in a 50m/s DV. This DV result in an altitude change of 240km during the Cruise phase. During this phase the Lander maintains a power positive or communication attitude. After the Cruise phase, a Braking maneuver is performed to produce a DV of 1950 m/s, which reduces the altitude down to 100m. During this maneuver, the Lander maintains a LVH attitude to place the DV in the opposite direction of the ram direction. The final decent starts at 100m and the ACS thrusters are used to maintain a vertical attitude (parallel to the gravity direction). At a certain high above the surface, the thruster are turned off and the Lander is allowed to fall the rest of the way.

TRN estimates the vehicle local relative position by comparing terrain maps, which would be loaded into memory prior to separation, with terrain measurements from navigation sensors (Lidar, Optical camera, Altimeter). There would be three different approaches TRN estimates (Global position using a global reference, Local position and velocity). In addition, there would be two primary algorithms for TRN (correlation and pattern Matching). In the correlation approach, the TRN Correlation algorithms place the sensor generated terrain image, the patch, at every location in the map and then measure the similarity between the patch and the map values. The location in the map with the highest correlation would be the best estimate of the current position of the vehicle. The other TRN algorithm would be pattern matching. This algorithm would use predefined landmarks and their define characteristics (lighting, shape, location,…) to match with those in the sensor generated images. The TRN architecture for the mission is shown in **Figure B-30**.

The synthetic image generated images are compared via the correlation or matching algorithms. For the TRN algorithm to be useful a significant amount of processing speed and memory is needed to obtain estimates at a useful cadence to ensure the successful landing with hazard avoidance.

Most of the algorithms and technology associated with the Lander would be mature and have flight heritage. The separation mode would be an extension of existing launch vehicle modes that would be modified to meet the needs of the mission. This would also be true for the Lander technology and algorithms. The key technology development for the Lander would be the interface between the HA and the TRN. There are no long-lead time components for the ACS that require more than 18 months. However, the ACS design and algorithm development would require a significant amount of development time. Flight software costs were accounted for using standard "wraps" in the costing exercise. As the project matures and the design process begins, trades should be performed to optimize lowest altitude allowable altitude for of thruster operation, C&DH memory and speed for TRN, the TRN algorithm (pattern matching vs correlation) and map size and resolution, and HA maneuver size.

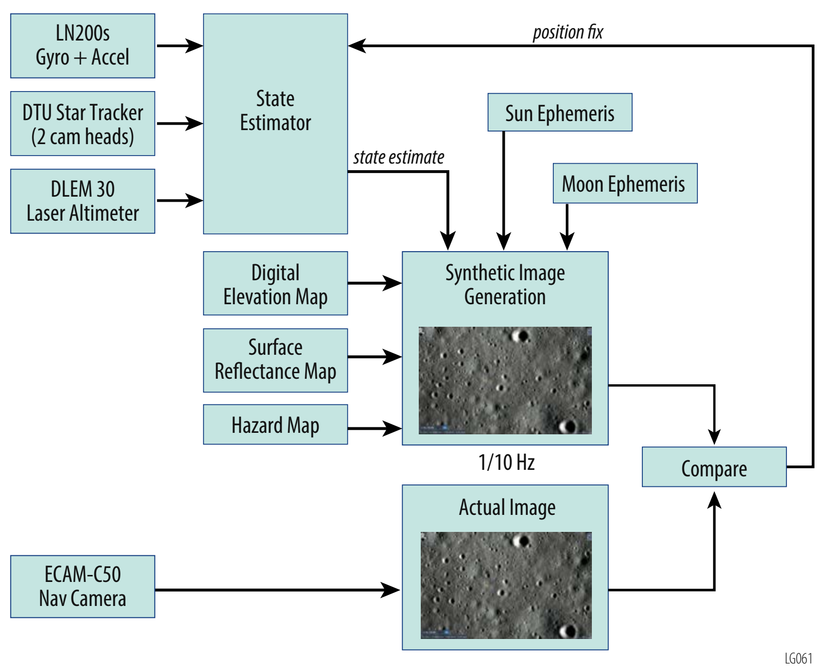


Figure B-30: Lunar TRN Architecture.

2.1.6 Avionics

The lander avionics consists of a block redundant system for Command and Data Handling (C&DH), attitude control sensors, power conditioning and distribution, mechanisms for launch locks, deployments and motors, and control of the main engine propulsion. The avionics implementation in **Figure B-31** consists of three enclosures, C&DH Unit, the Power System Electronics (PSE) and the Mechanism, and Propulsion Unit (MPU).

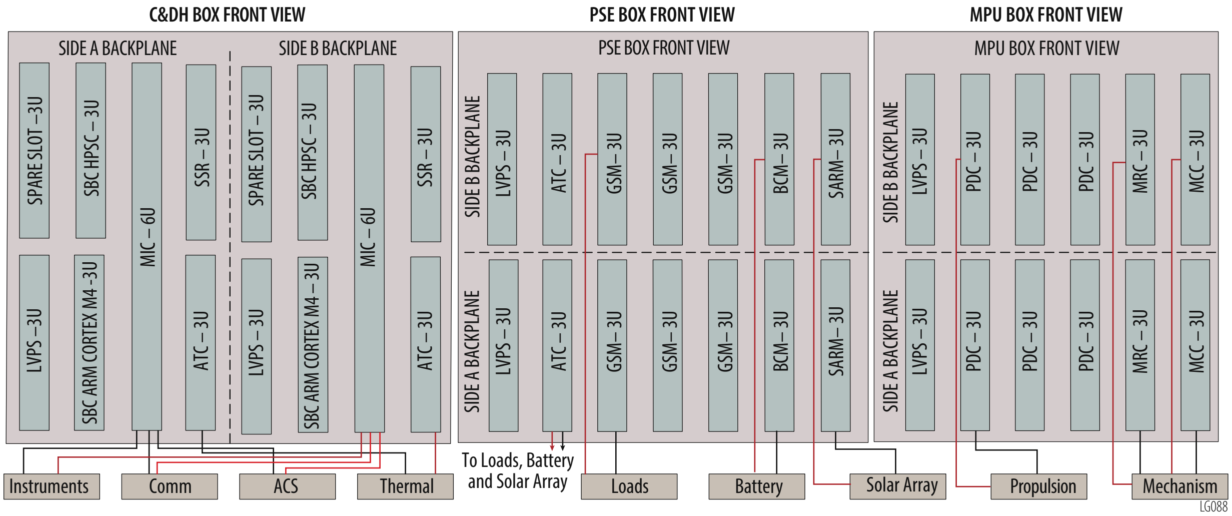


Figure B-31: Avionics implementation.

All units would be block redundant and internally redundant within the mechanical enclosure. Only one block side would be hot (powered) at a time and the other block side would be cold (un-powered). The mechanism for switch over from one block side to the other occurs by two different mechanisms. These include autonomous switch-over or switch-over via hardware ground command. The autonomous switch-over would occur based upon missing heartbeats from a processor to the Multi-Interface Cards (MICs), which perform the decision between them. Note: The MIC implements the hardware command decoder, Forward Error Correction (FEC) encoding of telemetry transfer frames and would have the communication interface to the transponder or transmitter among other functions.

The functions of the C&DH portion of the avionics performs basic command and control of the Lander including Entry, Descent and Landing (EDL) to the lunar surface. This phase is the highest risk phase because a fault of some portions of the system during this phase may result in a destruction of the Lander. Because of the short duration of the EDL phase of the mission and therefore the low probability for a fault in the portions of the system that could end the mission, the avionics is not designed to fail operate, *i.e.,* fly-through fault operation.

The C&DH would be comprised of a low-power processor based upon the VORAGO ARM Cortex-M4, the processor chip would have a power of less than 1W (the board peripherals would be about 3W additional). This would be important to reduce the power during the eclipse period on the lunar surface (about 2 weeks) to reduce the capacity of the battery required. During this phase of the mission, the processor gathers science data and stores it to the Solid-State Recorder (SRR) card. The processor would execute stored commands and gathers telemetry during the science phase of the mission. It would gather house-keeping telemetry and perform Fault Detection, Isolation, and Recovery (FDIR). The processor would perform ***Consultative Committee for Space Data Systems*** (CCSDS) Acquisition of Signal (AOS) transfer frame generation of telemetry data and pass it to the Multi Interface Card (MIC), which would have the communication interface function and performs Forward Error Correction (FEC) encoding (Reed Solomon or Turbo) and would have the interface to the transponder (S-band and X-band).

For the EDL portion of the mission, a second Single Board Computer (SBC) is used which is based upon the High Performance Spacecraft Computing (HPSC) chip that has several ARM Cortex A53 64 bit processors or other rad hard high performance processor for the Terrain Relative Navigation (TRN) processing. This SBC is higher power and therefore only used during the EDL portion of the mission. After landing it is powered down.

The MIC is a multi-function card. It would have the high-speed interfaces, Mil-Std 1553B interface and well as RS-422 interfaces to support the nominal interfaces found on a spacecraft. It also would have the communication functions (transfer frame FEC encoding, hardware command decoding and execution as well transponder interface and transmitter interface for the different RF bands). It would have GN&C interfaces for sun sensor, Inertial Measurement Unit (IMU) and reaction wheels. It would have non-volatile memory as well that may be used to supplement or replace the SSR depending upon the memory size requirements. The MIC also would have the Mission Elapsed Timer (MET) and provides the One Pulse Per Second interface (1 PPS) to spacecraft sub-systems as well as the watch-dog timers and logic to determine block switch-over. The MIC would be the only card that would be powered on both sides for this reason. Both MICs communicate to help determine the switch-over condition using Triple Modular Redundant (TMR) logic. Note the Low Voltage Power Supply (LVPS) would be powered on both sides as well to support powering of the redundant side MIC. There would be two for redundancy.

The Analog Telemetry Card (ATC) would have the analog digital converter (ADC) and analog multiplexers to gather temperature and other analog telemetry for the spacecraft. There would be two for redundancy. The Solid State Recorder (SSR) would have the non-volatile memory (Flash) 4 Tbits of data for the DTN relay communication storage. There would be two for redundancy. Lastly, the C&DH enclosure would have the Low Voltage Power Supply (LVPS) to power the cards in the C&DH. There would be two for redundancy.

The functions of the Power System Electronics (PSE) portion of the avionics are to perform solar array current regulation for the spacecraft loads and control the battery charging, the power distribution switches to various spacecraft loads and telemetry for power system functions. Note that the PSE functions would be redundant within the same enclosure. The Solar Array Regulation Module (SARM) would have the Field Effect Transistors (FETS) to switch the solar array strings for current regulation. There would be two SARM cards in the PSE for redundancy.

The Battery Charge Module (BCM) card would perform battery charge control (one per battery, though there are examples where redundant systems use a single battery, *i.e.,* Lucy (Discovery class mission)). There would be two Analog Telemetry Card (ATC) cards in the PSE, one primary and one redundant to read analog telemetry for the PSE. The power switch function would use six Generic Switch Module (GSM) cards, three for primary side and three for redundant side. Lastly, the PSE would have one house-keeping power supply, which would be internally redundant.

The MPU would have the functions for mechanisms, including launch locks, motors and propulsion valve drive. The pressure sensor telemetry would also go to the C&DH Unit. All cards have I2C interfaces. The Motor Controller Card (MCC) would have the H-bridge circuits and the relays to control 3 phase stepper motors. Each MCC would have the ability to drive four motors. Currently two MCCs would be baselined for redundancy. The Mechanism Release Card (MRC) would have the ability to control eight mechanisms switching both high and low side with an arm switch. Two MRCs would be currently baselined for redundancy. The Propulsion Drive Card (PDC) would have the ability to control eight valves switching both high and low side with an arm switch. Six PDC would be currently baselined for redundancy. Lastly, the MPU would have a LVPS card to provide secondary voltages to the MPU cards, two for redundancy.

Hardware currently exists from system integrators that meet the requirements within the MEL and PEL of this proposal. Typical schedules would be one year from System Requirements Review (SRR) to Preliminary Design Review (PDR), one year from PDR to Critical Design Review (CDR), one year from CDR to start of spacecraft integration and one year from integration start to spacecraft launch.

2.1.7 Electrical Power

The LGN lander power system consists of solar arrays, a secondary battery, and supporting power electronics. The power system configuration is primarily driven by the lunar night (**Figure B-32**), the length of which drives thermal loads and energy storage requirements. **Figure B-32** shows 42 days of Lander operations. The first 14.5 days (20,880 minutes) are during the lunar daylight and the so-

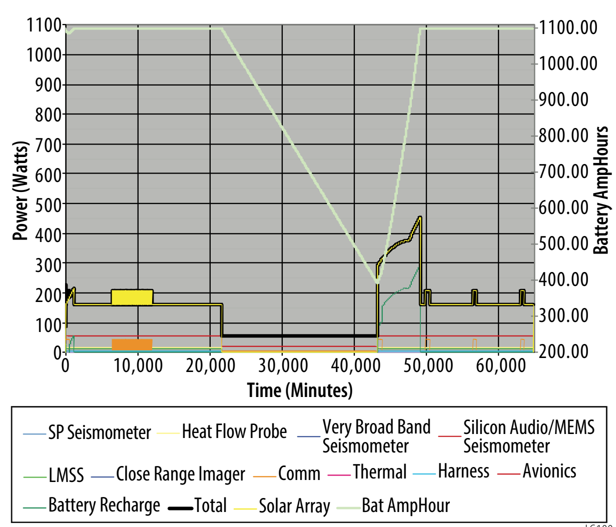


Figure B-32: 42 days of Lander operatios.

lar arrays provide all the needed power. The second 14.5 days are during the lunar night and the battery is used resulting in a depth of discharge of 60 % before the next lunar daylight begins to recharge the battery. Two standard fixed solar array panels are baselined, one mounted on the east and one on the west side of the lander. TJGaAs solar cells with bare cell efficiency of 29.5%, a solar constant of 1353 w/m^2, array operating temp at 100 deg C, Space Environmental Effects and Education System (SPENVIS) solar array degradation factors, and an assumed latitude of -44 degrees were used to derive the array area requirement. This yielded two panels with 2.95 m^2 active area each (3.8 m^2 total substrate area each). The arrays are mounted at a 45 degree angle on either side of the lander and will produce a daylight peak power of 700W EOL, 800W BOL at lunar noon. An 1100AH battery is used to support night loads. This is expected to be Li Ion chemistry optimized for low temperature operation. The Power System Electronics (PSE) will be a heritage 28VDC battery dominated bus included as cards in the avionics package. The PSE will control battery charging and power distribution. The battery, solar array and harness are all TRL 7. The PSE, 28 vDC battery dominated bus is TRL 6 for this application.

2.1.8 Thermal

The LGN Thermal overview can be summarized in 1 sentence: Minimize heater power! The Moon is a very hard place to stay within temperature limits; the daytime has regolith temperatures nearing 110°C during local noon (plus ~1420 W/m^2 of solar load), and the nighttime side has regolith temperatures nearing -180°C (with zero solar input). **Figure B-33** shows a plot of lunar regolith temperature vs. time during the daytime.

In order to make the thermal design work (dissipate 162.4 watts of electronics power during the daytime, stay above survival temperatures during the night while only dissipating 40.8 watts), the following are the thermal highlights of the LGN design:

1. Have a dedicated 2.8 m^2 of radiator on the anti-sun side of the Lander used to dissipate the Avionics and Comm. heat loads:
   - Pointing “North” on north hemispherical Landers
   - Pointing “South” on south hemispherical Landers
2. Use louvers over the radiator to reduce heater power during nighttime and colder “shoulders” of the daytime hours
   - Morning and evening hours
   - 14.5 days of nighttime (no solar load and extremely cold regolith temperature)
3. The Battery is coupled to the avionics radiator with a heat pipe
   - Radiator does not have room for the battery, but battery needs to stay above survival limits.
4. Li-Ion battery has very wide temperature limits
   - -40°C to +80°C
5. Propulsion system will be vented after landing
   - Heaters on tanks, lines, and valves can be turned off, saving heater power
6. The Solar Arrays will never be “open circuit” during hot daytime operation
   - This lowers their maximum temperature
7. The Lander takes care of Instrument thermal control
   - Heaters for nighttime, radiators if needed, for operational case

Figure B-34 and **Figure B-35** show the external and internal thermal models of the Lander.

In order to save heater power, louvers are utilized on the Lander so that the full radiator area can dissipate its 162.4 watts of power to the environment during mid-daytime hours, but the louvers can par-

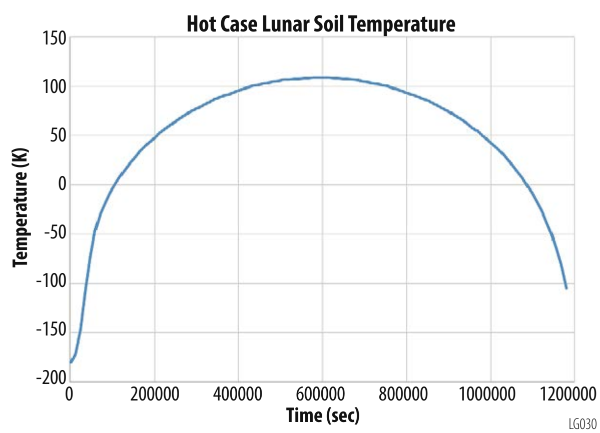


Figure B-33: Lunar Regolith Daytime Temperature vs. Time.

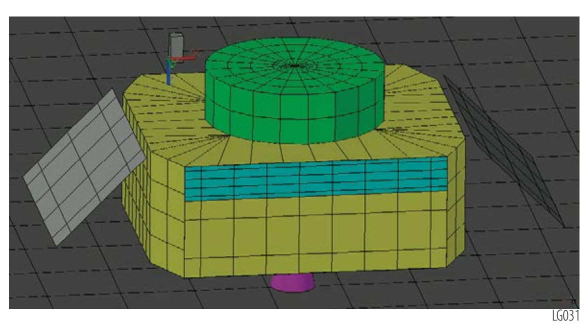


Figure B-34: External Thermal Model.

tially close during morning and afternoon portions of the lunar day. The louvers will fully close during the night time when power on the radiator is minimal (40.8 watts).

The louvers use bi-metallic springs on each of their vanes, so no power or electronic circuitry is needed for its operation. The springs open and close the louver's vanes automatically as the temperature changes.

The second thermal characteristic of the Lander is that once its lands onto the lunar surface, the Propellant is vented from the tanks, valves, etc… so that no propulsion heaters are needed to be used once on the Lunar surface. This is a required mode of operation, as there is no heater power available to keep the propulsion system warm for the 14.5 days of lunar night time.

With no propellant to stay warm, and the louvers totally closed the Lander stays above its avionics and comm survival limits with very little power being dissipated.

The radiators need to be pointing anti-sun to minimize radiator area; so for landing sites north of the lunar equator, the radiator needs to be on the North side of the Lander. For Southern landing sites, the radiator needs to point South.

Results of the Thermal model for hot and cold biased conditions are shown in **Figure B-36, Figure B-37,** and **Figure B-38**.

All thermal hardware has a high TRL level. Louvers have been used on spacecraft since the 1950's. And have been used as recently as PSP (Parker Solar Probe), New Horizons, Messenger, and are currently being used on LUCY. Although each louver will have different length and height (number of vanes) between spacecrafts (mechanical dimensions change), the basic design does not change.

Thermal hardware on the Moon can be effected by the small dust particle sizes. Dust needs to be prevented from working its way into the louver blades; preventing their operation (open/shut). In addition, the extremely low required emissivity's of the louvers (gold plating or polished aluminum) can be increased by dust. Secondly, the Lander needs to survive the nighttime hours (14.5 Earth days) with almost no power being dissipated. While louvers keep the avionics boxes warm, and the propulsion system is vented, the mechanical structure of the Lander will be extremely cold. Graphite honeycomb structures need to be analyzed for the extreme cold.

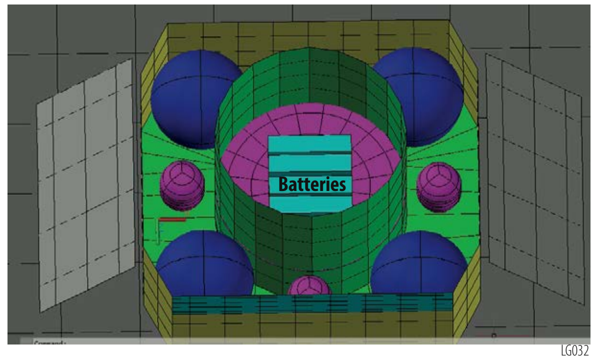


Figure B-35: Internal Thermal Model.

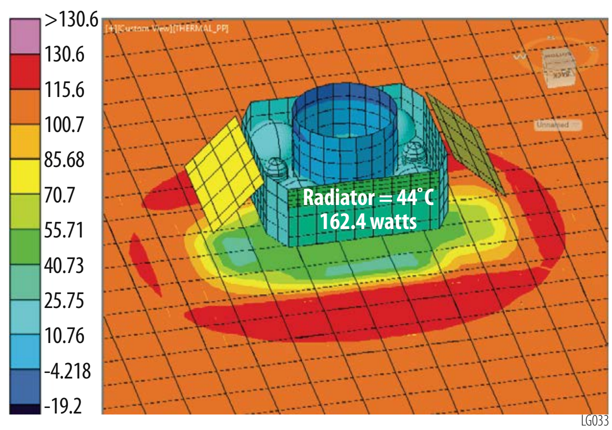


Figure B-36: Hot Biased Temperatures at Lunar Noon.

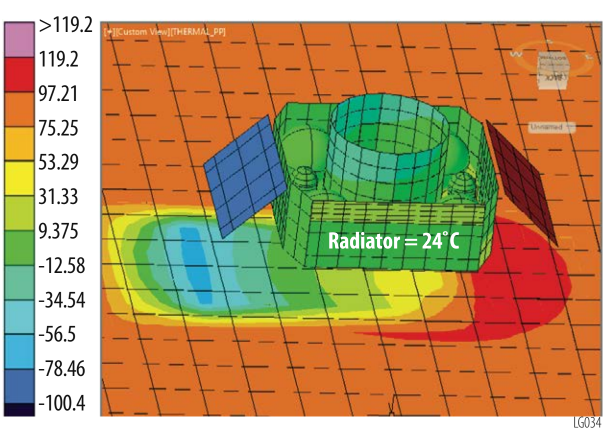


Figure B-37: Hot Biased Temperatures at Lunar 3PM.

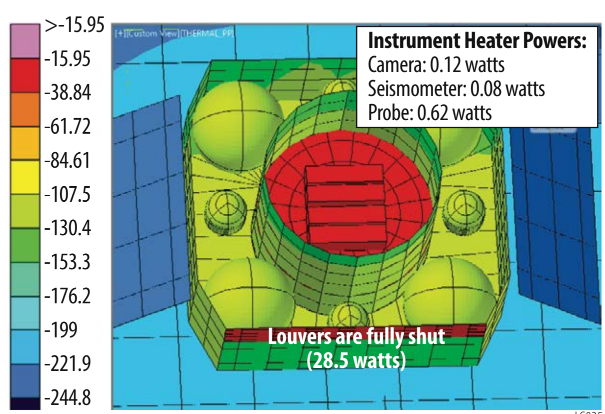


Figure B-38: Cold Biased Nighttime Temperatures.

2.2 Orbiter Design

The Orbiter serves as a carrier/delivery system for the Landers, communication relay and a limited science platform. Utilizing the Orbiter as the carrier/delivery system minimizes the mass for each Lander since the Landers propulsions systems can be sized for landing only and do not have to include additional capability to perform the Lunar Orbit Insertion (LOI) individually. The Orbiter performs the necessary LOI for all Landers and controls and executes the release of the Landers for deployment to the surface. The Orbiter carries enough propellant to perform the LOI and orbit maintenance for the duration of the mission. The Orbiter science payload consists of a magnetometer on a 2 m boom. **Figure B-40** shows the Orbiter concept with the Landers. **Figure B-41** shows the deployed Orbiter. **Table B-14** provides the mass and power of the Orbiter.

2.2.1 Instrument Accommodations

The Orbiter has only one instrument as shown in **Table B-10**, the Magnetometer which is deployed on a 2m boom. The mass and power of the magnetometer is shown in **Table B-15**.

The LGN Orbiter serves as the vehicle carrying the suite of Landers into a stable lunar orbit and the platform initiating each Lander descent to the moon's surface. One of the LGN Landers is planned to be deployed onto the far side of the moon. This mandates the Orbiter is also needed in support of the RF communications with the far side Lander. The Orbiter carries two distinct communications subsystems. The X-band Lunar Interface system is designed to provide RF comm connectivity with every Lander on the moon surface. X-band for the moon's surface was also selected because it can be operated simultaneously, without interference, with the Orbiter's DTE S-band and Ka-band systems. The Direct to Earth (DTE) interface is a parallel set of S-band and Ka-band channels sharing a dual feed 1m HGA system.

The design of the Orbiter's lunar communication interface is fairly simple. Two X-band user modems share a 1 m HGA system through a DPDT switch. On the back of the HGA is a single pull double throw switch allowing the access to .the feed's left (LHCP) or right (RHCP) hand circular polarization ports. The Lander HGA is RHCP and the LGA is LHCP. A functional block diagram of the Lander RF communications subsystem is included as **Figure B-42**.

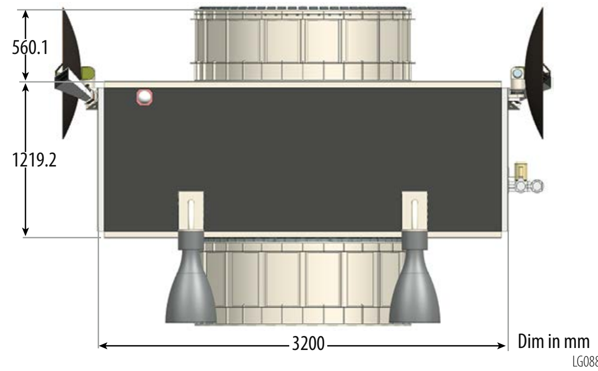


Figure B-39: LGN Orbitor.

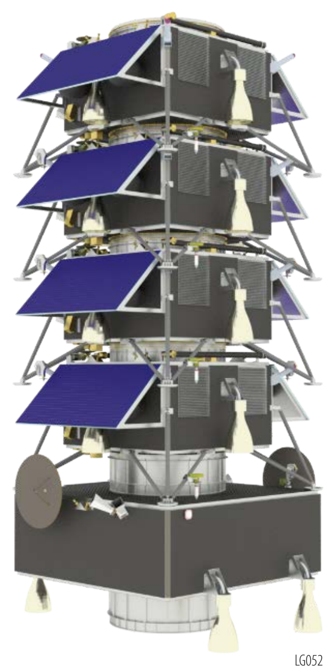


Figure B-40: LGN Orbiter with Lander.

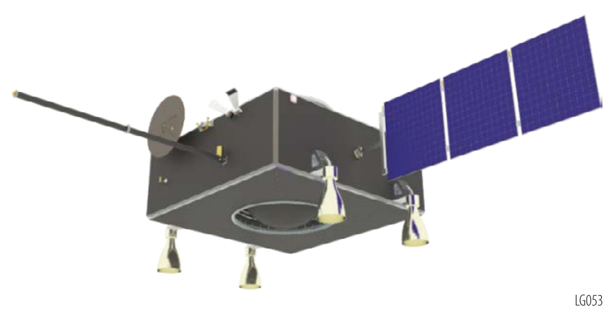


Figure B-41: LGN Orbiter communication relay deployed configuration.

Table B-14: LGN Orbiter Mass and Power.

	Mass			Average Power		
Orbiter	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
Structures & Mechanisms	629.6	30	810.9	20	10	20
Thermal Control	34	10	37.4	129.0	10	141.9
Propulsion (Dry Mass)	279.0	10	306.9	0	0	0
Attitude Control	84.8	10	93.2	35.7	10	39.2
Avionics	38.9	10	42.8	58.1	10	63.9
Telecommunications	50.0	10	55.0	256.0	10	281.6
Power System	56.2	10	63.8	5.0	10	5.5
Total Orbiter	1,174.5	20	1,386.7	503.8	10	554.1

Table B-15: Orbiter Payload Mass and Power Table

	Mass			Average Power		
Orbiter	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
Magnetometer with Boom	2.0	30	2.6	1.9	30	2.5

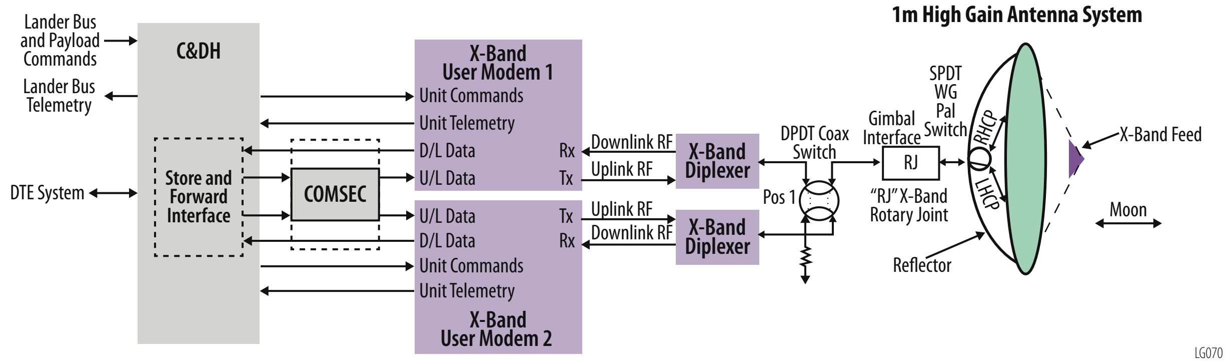


Figure B-42: LGN Orbiter Lunar User RF Communications Subsystem Functional Block Diagram.

LGN Orbiter Lunar Communications User Link Analyses

The Lander link budgets show an abundance of downlink margin at 10 msps. Of note, because each Lander is using a 0.5m antenna and the Orbiter baselines a 1m HGA, the RF power required from the modem is relatively low. Up links have sufficient margin with only a 0.5W RF modem output. If trade studies show enthusiasm for a redefinable X-band Lunar User Modem, adding (or specifying) a 3W class SSPA will support forward links for landed HGAs into the msps range.

The S-band half of the Orbiter DTE Communications system is based on two "standard" DSN compatible NASA 8W transponders. Each transponder shares the 1m HGA or LGA path through a DPDT switch. A SPDT switch down selects one LGA. The down link analyses were problematic due to the insertion losses encountered from a previous hybrid which allowed access to both LGAs simultaneously.

The Ka-band side of the DTE Communications system has two 3W Ka-band transceivers sharing the dingle port to the 1m HGA. Ka-band transceivers were selected because they are increasingly in development for Cubesat class applications. If finding available Ka-band transceivers becomes problematic, down selecting to Ka-band transmitters may occur. The DTE S-band uplinks have a great deal of available margin for much wider bandwidth towards the moon's surface. A functional block diagram of the Orbiter DTE communications subsystem is included as **Figure B-43**.

LGN Orbiter DTE Communications Link Analyses

The LGN DTE Link analyses are summarized in **Table B-16**. Adequate margin exists for all links listed.

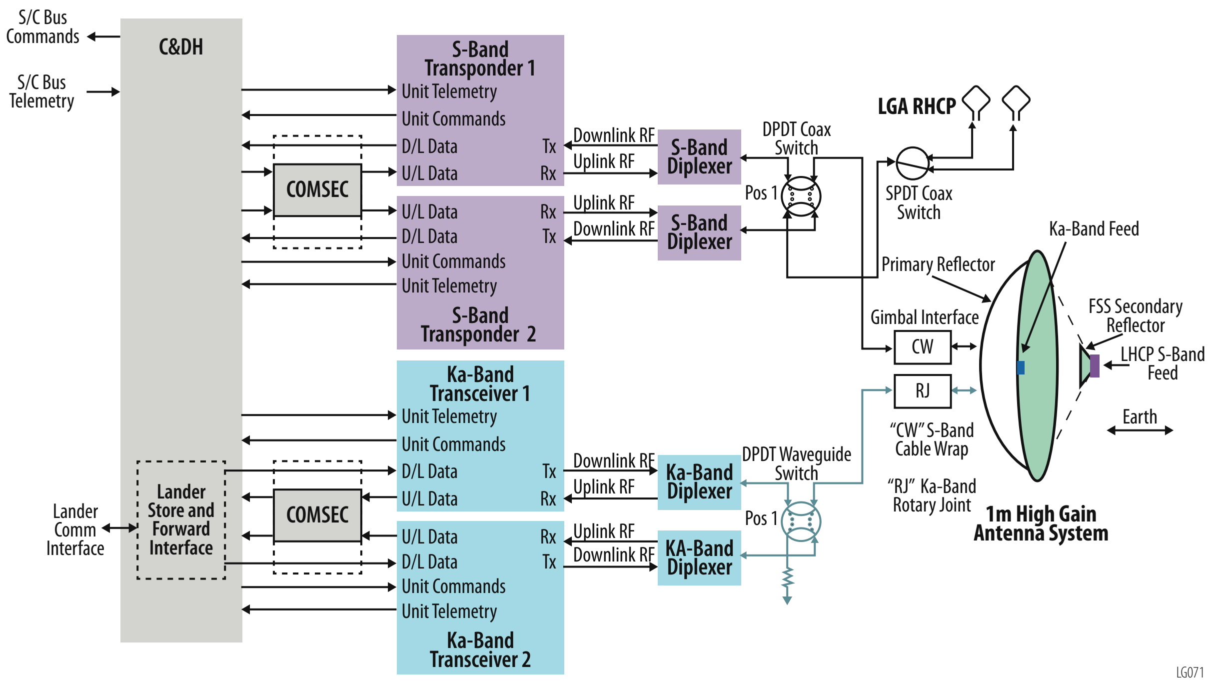


Figure B-43: LGN Orbiter DTE Communications Subsystem Functional Block Diagram.

Table B-16: LGN Orbiter DTE RF Communications Subsystem Summary of Link Analyses.

Link	Band	Direction	From	Power (W)	Antenna	To	Antenna	Rate (kbps)	Margin (db)
13	Ka-band	Downlink	Orbiter	3	HGA	DSN	34 m	10,000	15.7
14	Ka-band	Uplink	DSN	200	34 m	Orbiter	HGA	64	48.3
15	S-band	Downlink	Orbiter	8	HGA	DSN	34 m	3,000	3.9
16	S-band	Uplink	DSN	200	34 m	Orbiter	HGA	64	27.2
17	S-band	Downlink	Orbiter	8	LGA	DSN	34 m	4	5.5
18	S-band	Uplink	DSN	2,000	34 m	Orbiter	LGA	64	10.7

Orbiter DTE Downlinks

26.5 GHz DTE Downlink Orbiter HGA to DSN 34m
410416 km Range
LDPC 7/8

3 W	4.77	HPA pwr (dBW)	49.04 EIRP
	-2.00	pass loss (dB)	
1.00 m	46.27	Orbiter ant gain (dBi)	
	-233.17	path loss (dB)	
	-0.50	pointing + pol (dB)	
34.00 m	76.90	DSN ant gain (dBi)	48.61 G/T
1.0 dB NF	-1.00	pass loss (dB)	
26.9 dBK	-27.29	sys temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
10.0 Msps	-70.00	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	15.73	Margin (dB)	

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Orbiter DTE Uplinks

23.0 GHz DTE Uplink DSN 34m to Orbiter HGA
410416 km Range
No FEC

200 W	23.01	HPA pwr (dBW)	96.18 EIRP
	-2.50	pass loss (dB)	
34.00 m	75.67	DSN ant gain (dBi)	
	-231.94	path loss (dB)	
	-0.50	pointing + pol (dB)	
1.00 m	45.04	Orbiter ant gain (dBi)	16.65 G/T
3.0 dB NF	-2.00	pass loss (dB)	
24.6 dBK	-26.39	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
64.0 kbps	-48.06	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	48.34	Margin (dB)	

10

2.3 GHz DTE Downlink Orbiter HGA to DSN 34m
410416 km Range
LDPC 7/8

8 W	9.03	HPA pwr (dBW)	32.02 EIRP
	-2.00	pass loss (dB)	
1.00 m	24.99	Orbiter ant gain (dBi)	
	-211.89	path loss (dB)	
	-0.50	pointing + pol (dB)	
34.00 m	55.62	DSN ant gain (dBi)	27.33 G/T
1.0 dB NF	-1.00	pass loss (dB)	
26.9 dBK	-27.29	sys temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
3.0 Msps	-64.77	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	3.94	Margin (dB)	

11

2.1 GHz DTE Uplink DSN 34m to Orbiter HGA
410416 km Range
No FEC

200 W	23.01	HPA pwr (dBW)	75.41 EIRP
	-2.50	pass loss (dB)	
34.00 m	54.90	DSN ant gain (dBi)	
	-211.18	path loss (dB)	
	-0.50	pointing + pol (dB)	
1.00 m	24.27	Orbiter ant gain (dBi)	-4.48 G/T
3.0 dB NF	-2.00	pass loss (dB)	
24.6 dBK	-26.75	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
64.0 kbps	-48.06	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	27.21	Margin (dB)	

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2.3 GHz DTE Down Link Orbiter LGA to DSN 34m
410416 km Range
LDPC 7/8

8 W	9.03	HPA pwr (dBW)	4.83 EIRP
	-2.20	pass loss (dB)	
	-2.00	Orbiter ant gain (dBi)	
	-211.89	path loss (dB)	
	-0.50	pointing + pol (dB)	
34.00 m	55.62	DSN ant gain (dBi)	27.33 G/T
1.0 dB NF	-1.00	pass loss (dB)	
26.9 dBK	-27.29	sys temp (dB)	
100K sys			
	228.60	Boltz (dBW/Hz-K)	
4 ksps	-36.02	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	5.50	Margin (dB)	

13

2.1 GHz DTE Up Link DSN 34m to Orbiter LGA
410416 km Range
No FEC

2000 W	33.01	HPA pwr (dBW)	85.41 EIRP
	-2.50	pass loss (dB)	
34.00 m	54.90	DSN ant gain (dBi)	
	-211.18	path loss (dB)	
	-0.50	pointing + pol (dB)	
	-2.00	Orbiter ant gain (dBi)	-30.95 G/T
2.0 dB NF	-2.20	pass loss (dB)	
24.6 dBK	-26.75	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
64.0 kbps	-48.06	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	10.74	Margin (dB)	

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LG083

Figure B-44: LGN Orbiter DTE to DSN Communication Link Analyses.

Orbiter DTE Downlinks

26.5 GHz DTE Downlink Orbiter HGA to NEN 18m
410416 km Range
LDPC 7/8

3 W	4.77	HPA pwr (dBW)	49.04 EIRP
	-2.00	pass loss (dB)	
1.00 m	46.27	Orbiter ant gain (dBi)	
	-233.17	path loss (dB)	
	-0.50	pointing + pol (dB)	
18.00 m	71.37	NEN ant gain (dBi)	38.80 G/T
2.5 dB NF	-3.00	pass loss (dB)	
26.9 dBK	-29.57	sys temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
10.0 Msps	-70.00	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	5.92	Margin (dB)	

9a

2.3 GHz DTE Downlink Orbiter HGA to NEN 18m
410416 km Range
LDPC 7/8

8 W	9.03	HPA pwr (dBW)	32.02 EIRP
	-2.00	pass loss (dB)	
1.00 m	24.99	Orbiter ant gain (dBi)	
	-211.89	path loss (dB)	
	-0.50	pointing + pol (dB)	
18.00 m	50.09	NEN ant gain (dBi)	18.67 G/T
2.0 dB NF	-2.00	pass loss (dB)	
26.9 dBK	-29.42	sys temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
500.0 ksps	-56.99	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	3.06	Margin (dB)	

11a

2.3 GHz DTE Down Link Orbiter LGA to NEN 18m
410416 km Range
LDPC 7/8

8 W	9.03	HPA pwr (dBW)	4.83 EIRP
	-2.20	pass loss (dB)	
	-2.00	Orbiter ant gain (dBi)	
	-211.89	path loss (dB)	
	-0.50	pointing + pol (dB)	
18.00 m	50.09	NEN ant gain (dBi)	18.67 G/T
2.0 dB NF	-2.00	pass loss (dB)	
26.9 dBK	-29.42	sys temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
1 ksps	-30.00	data rate (dB-Hz)	
	-3.85	Min Es/No (dB)	
	-3.00	imp (dB)	
	2.86	Margin (dB)	

13a

Orbiter DTE Uplinks

23.0 GHz DTE Uplink NEN 18m to Orbiter HGA
410416 km Range
No FEC

200 W	23.01	HPA pwr (dBW)	90.65 EIRP
	-2.50	pass loss (dB)	
18.00 m	70.14	NEN ant gain (dBi)	
	-231.94	path loss (dB)	
	-0.50	pointing + pol (dB)	
1.00 m	45.04	Orbiter ant gain (dBi)	16.65 G/T
3.0 dB NF	-2.00	pass loss (dB)	
24.6 dBK	-26.39	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
64.0 kbps	-48.06	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	42.81	Margin (dB)	

10a

2.1 GHz DTE Uplink NEN 18m to Orbiter HGA
410416 km Range
No FEC

200 W	23.01	HPA pwr (dBW)	69.89 EIRP
	-2.50	pass loss (dB)	
18.00 m	49.38	NEN ant gain (dBi)	
	-211.18	path loss (dB)	
	-0.50	pointing + pol (dB)	
1.00 m	24.27	Orbiter ant gain (dBi)	-4.48 G/T
3.0 dB NF	-2.00	pass loss (dB)	
24.6 dBK	-26.75	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
64.0 kbps	-48.06	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	21.69	Margin (dB)	

12a

2.1 GHz DTE Up Link NEN 18m to Orbiter LGA
410416 km Range
No FEC

2000 W	33.01	HPA pwr (dBW)	79.89 EIRP
	-2.50	pass loss (dB)	
18.00 m	49.38	NEN ant gain (dBi)	
	-211.18	path loss (dB)	
	-0.50	pointing + pol (dB)	
	-2.00	Orbiter ant gain (dBi)	-30.95 G/T
2.0 dB NF	-2.20	pass loss (dB)	
24.6 dBK	-26.75	Ant temp (dB)	
290K sys			
	228.60	Boltz (dBW/Hz-K)	
64.0 kbps	-48.06	data rate (dB-Hz)	
	-9.59	Min Eb/No (dB)	
	-3.00	imp (dB)	
	5.21	Margin (dB)	

14a

LG084

Figure B-45: LGN Orbiter DTE to NEN Communication Link Analyses.

2.2.2 Structures

The Orbiter serves as a carrier/delivery system for the Lander stack, communication relay and a limited science platform. The substantial propulsion system on the Orbiter provides orbit insertion braking and positioning for the release of the Landers. The communication relay system consists of a Lunar Communication Subsystem (DCS) gimballed dish antenna and a DTE Communication Subsystem (DCS) gimballed dish antenna. The only instrument on the Orbiter is the magnetometer which is deployed on a simple boom. A single axis, three-panel, deployable solar array wing provides power. The Orbiter is 3-axis stabilized with a four-reaction wheel cluster.

The Orbiter structural design was driven by carrying the launch loads of the other mission elements through to the launch vehicle. The Orbiter structure was designed as a structural box with composite face sheet/aluminum honeycomb core panels using the clip and post method employed on other composite structures such as LRO. The basic structure of a central cylinder, upper and lower decks, radials and equipment panels is very common and well understood. The Orbiter acts, in some ways, like an upper stage with propulsion dominating the design. The central cylinder stack concept provides the primary load path for the Landers. The Orbiter structural design is focused primarily on accommodating the large oxidizer tanks, which are not all being carried directly through the central cylinder as they are on the Landers due to the additional propellant volume required by the Orbiter. A single, large fuel tank is mounted in the central cylinder with an equatorial ring. The four oxidizer tanks are mounted radially imbedded in stiffened cut-outs the radial panels using the tank's equatorial rings. This mounting concept may not be optimal for integration and assembly and other, more modular, mounting methods should be studied in future work. Pressurant tanks, which are much smaller, are easily accommodated in the remaining volume of the structure.

2.2.3 Propulsion

The LGN Orbiter propulsion subsystem is a large regulated bipropellant system. It carries a large amount of propellant to perform three principal functions: Lander Entry Targeting, Orbit Insertion (into a 250 km circular polar orbit), and Orbit Maintenance. The propellant is stored in COTS tanks, and the main engines are a set of four 4,000 N engines (AJ PN R-40B in **Figure B-46**). A set of 4 x 450 N engines (AJ PN R-4D-15 in **Figure B-46**) are used for roll control during maneuvers. Separate pressurization manifolds are used to provide regulated pressure to both the fuel and oxidizer tanks. All of the components are COTS.

Figure B-46: R-40B 4,000N and R-4D-15 450N (HiPAT) rocket engines (not to scale).

As shown in **Figure B-47**, the system is single fault tolerant, with the exception of the main engine. Each pressurization string is fully redundant, and there are separate strings of redundant attitude control thrusters.

Redundant pyro valves isolate the pressuring tanks during launch. The system is pressurized during the transfer to lunar orbit insertion and a calibration maneuver is performed. All maneuvers are performed with the main engine, except for smaller orbit maintenance maneuvers.

2.2.4 ACS

The roles that the Orbiter must fullfill drives the Orbiter towards a three-axis stabilized ACS with significant momentum and torque capabilities to account for the large inertia associated with the stacked configuration. The Orbiter ACS hardware is chosen to meet the needed torque and momentum capability when transporting the four Landers to the moon and when serving as a communication Orbiter. The ACS requirements are:

Control: 180/180/180 arcsec, Roll/Pitch/Yaw (3-sigma)
Knowledge: 60/30.0/30.0 arcsec, Roll/Pitch/Yaw (3-sigma)
Attitude targets: Nadir, inertial, quaternion profile

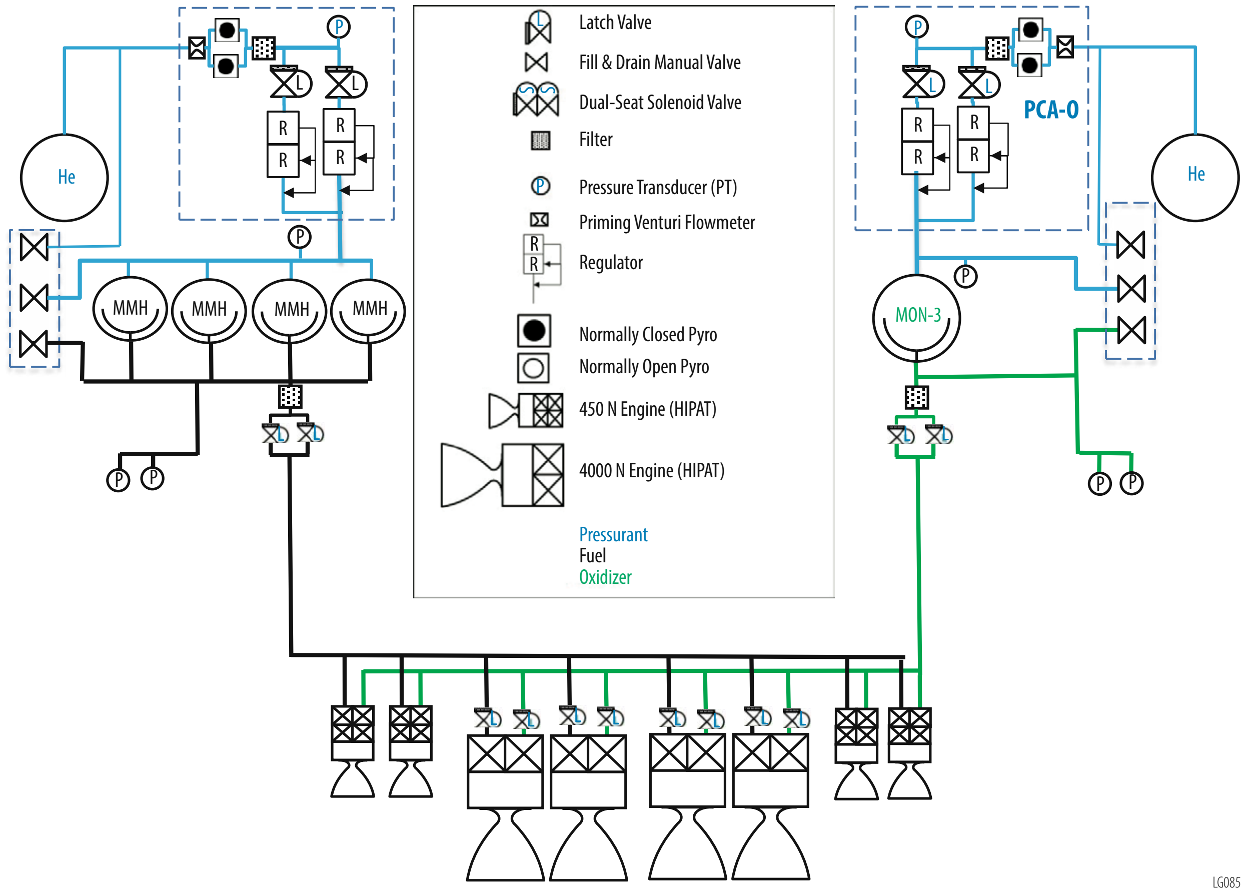

Figure B-47: Schematic of the Propulsion Subsystem.

All of the ACS components have significant flight heritage and can meet the life requirement for the mission. In addition, the ACS has selective Fault Tolerance (FT). This FT is based on the risk of failure and the impact to mission success.

After Launch vehicle (LV) separation, the Orbiter must detumble and slew to a power/communication positive attitude. After Orbiter checkout, the LOI is performed to place the Orbiter and its Landers into a lunar orbit. While in orbit about the Moon (**Figure B-48**), the Orbiter will align the body axis with the Local Vertical Local Horizontal (LVLH) Frame. Prior to the spin-up, the top Lander will be power on, initialized with the appropriate navigation states and checked out. At the appropriate time and position, the Orbiter will spin-up and eject the top Lander. After Lander separation, the Orbiter will despin and return to tracking the LVLH frame (approx. Lunar Nadir point). After settling into the nominal orbital attitude, the Orbiter will perform a checkout and determine the appropriate scale factors for the observatory actuation. This process is repeated until all Landers are separated.

The needed spin rates for the Orbiter is provided in **Table B-17**.

After all Landers are separated, the Orbiter can act as a communication relay and a plat-

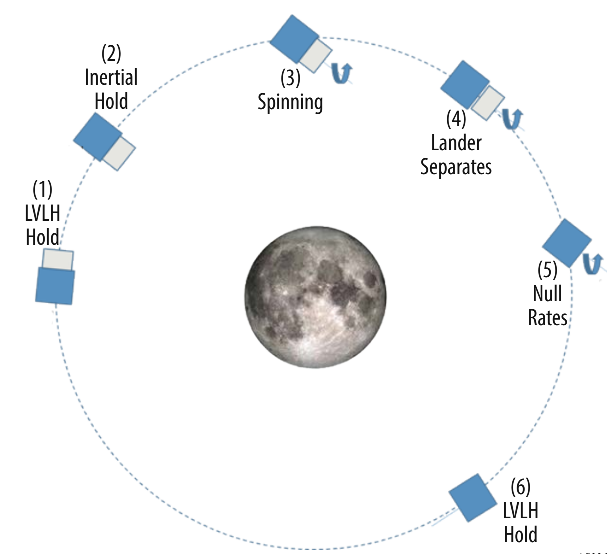

Figure B-48: Lander separation sequence.

form for other orbital instruments. The Orbiter alone concept of operation has some flight heritage (LRO). With the major difference between LGN and previous missions being the needed control authority four the 4 Landers and Orbiter (significantly higher inertia ->tipoff momentum, slew torques). The Orbiter ACS block diagram is based on previous missions and shown below in **Figure B-49**.

The navigation portion of the Orbiter Flight Software (FS) is required to estimate and control the spacecraft position, which reduces the needed for additional ground processing to ensure a precise Lander separation point/time. The Orbiter's ACS has five modes: Mission (Nadir, and inertial point), Separation (spin-up/spin-down), Delta V, Delta H, Sun Safe (Sun Acq, Rate Null). The new mode in this architecture is the separation mode. The other modes are based on heritage algorithms/ modes. The separation mode is designed to facilitate the safe and accurate ejection of the Landers. This mode places the momentum vector in the correct direction and then increases it's momentum magnitude. After Lander separation, the Orbiter will perform a post separation checkout, which includes sensor and actuator calibrations. These calibrations are used to determine scale factors for the thruster commanding.

Table B-17: Needed Spin rates.

Number of Landers	Imparted Momentum on Observatory/ Stack (Kg-m^2/s)	Rates Imparted on Observatory/ Stac k (deg/sec)	Conservative Observatory Spin Rates (deg/s)
1	135.5	1.465	2
2	135.5	1.021	1.5
3	135.5	0.781	1

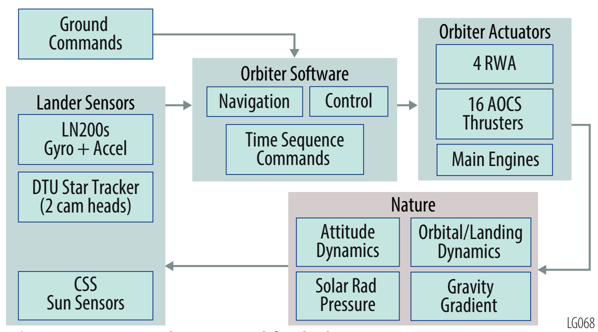

Figure B-49: Orbiter ACS block diagram.

2.2.5 Avionics

The avionics for the Lunar Geophysical Network (LGN) Orbiter is a block redundant system to meet the reliability for the New Frontiers class mission. The avionics consists of the following functions: Command and Data Handling (C&DH), attitude control sensors, power conditioning and distribution, mechanisms for launch locks, deployments and motors, and control of main engine propulsion. The avionics implementation consists of thee enclosures, C&DH Unit, the Power System Electronics (PSE) and the Mechanism, and Propulsion Unit (MPU). All units are block redundant and internally redundant within the mechanical enclosure. Only one block side is hot (powered) at a time and the other block side is cold (un-powered). The mechanism for switch over from one block side to the other occurs by two different mechanisms. These include autonomous switch-over or switch-over via hardware ground command. The autonomous switch-over occurs based upon missing heartbeats from a processor to the Multi-Interface Cards (MICs), which perform the decision between them. Note: The MIC implements the hardware command decoder, Forward Error Correction (FEC) encoding of telemetry transfer frames and has the communication interface to the transponder or transmitter among other functions. See MIC description below.

The Orbiter for the most of the mission, functions as a communication relay between Earth and the Landers. More on other functions in Concept of Operation section. The Orbiter utilizes a store and forward protocol called Delayed Tolerant Network (DTN), which has the ability to store packets from the Lander or Orbiter and forward them to their respective destination (Earth or Lander). DTN is a protocol implemented by the C&DH system independent of the implementation of the RF system. The Orbiter receives communication from Earth and stores the data until it has line of sight (LOS) to the Lander at which time it transmits the data to the Lander in a reliable manner (with acknowledgments from the Lander and possible retransmissions from the Orbiter until all the data is acknowledged as being received correctly). The same happens in the reverse direction from the Lander to the Orbiter to Earth, where each hop in the communication is reliability transferred and stored before forwarded to the next destination until it reaches the final destination based upon communication availability, *i.e.,* LOS. If the next hop in the communication is unavailable, delay in communication occurs until LOS is established. For LGN there are only two hops (Earth to Orbiter and Orbiter to Lander, and the reverse direction).

2.2.5.1 C&DH Unit:

The functions of the C&DH portion of the avionics performs basic command and control of the Orbiter. The C&DH implements the RAD750 v3 processor. The processor runs the flight software to command GN&C effectors based upon the GN&C sensors and algorithms. It also gathers house-keeping telemetry and performs fault detection isolation and recovery (FDIR) autonomously. The Orbiter runs on a scheduled timeline uploaded in advance to perform autonomous operations. The spacecraft processor works as the file manager for the Solid-State Recorder (SRR) card that stores the packets from Earth and the Landers for reliable forwarding to their respective destination (Lander or Earth), *i.e.,* DTN bundle protocol operations for reliable communication. The functions of the various cards in the C&DH system are internally redundant and are as follows:

The processor based upon the RAD750 v3 or next generation runs the flight software to control the Orbiter and has the cPCI interface to communicate with the MIC and SSR.

The MIC is a multi-function card. It has the high-speed interfaces, Mil-Std 1553B interface and well as RS-422 interfaces to support the nominal interfaces found on a spacecraft. It also has the communication functions (transfer frame FEC encoding, hardware command decoding and execution as well transponder interface and transmitter interface for the different RF bands). It has GN&C interfaces for sun sensor, IMU and reaction wheels. It has non-volatile memory as well that may be used to supplement or replace the SSR depending upon the memory size requirements. The MIC also has the Mission Elapsed Timer (MET) and provides the One Pulse Per Second interface (1 PPS) to spacecraft sub-systems as well as the watch-dog timers and logic to determine block switch-over. The MIC is the only card that is powered on both sides for this reason. Both MICs communicate to help determine the switch-over condition using Triple Modular Redundant (TMR) logic. Note the LVPS is powered on both sides as well to support powering of the redundant side MIC.

The Analog Telemetry Card (ATC) has the analog digital converter (ADC) and analog multiplexers to gather temperature and other analog telemetry for the spacecraft.

The Solid State Recorder (SSR) has the non-volatile memory (Flash) 4 TBytes of data for the DTN relay communication storage.

Lastly, the C&DH enclosure has the Low Voltage Power Supply (LVPS) to power the cards in the C&DH.

2.2.5.2 PSE Unit:

The functions of the Power System Electronics (PSE) portion of the avionics performs solar array current regulation for the spacecraft loads and control the battery charging, the power distribution switches to various spacecraft loads and telemetry for power system functions. Note that the PSE functions are redundant within the same enclosure.

The Solar Array Regulation Module (SARM) has the Field Effect Transistors (FETS) to switch the solar array strings for current regulation. There are two SARM cards in the PSE for redundancy.

The Battery Charge Module (BCM) card performs battery charge control. Depending if there is one or two batteries determines how many BCMs are present (one per battery). There are examples where a redundant systems use a single battery, *i.e.,* Lucy (Discovery class mission).

There are two Analog Telemetry Card (ATC) cards in the PSE, one primary and one redundant to read analog telemetry for the PSE.

The power switch function uses six Generic Switch Module (GSM) cards. Three for primary side and three for redundant side.

Lastly, the PSE has one house-keeping power supply, which is internally redundant.

2.2.5.3 MPU:

The MPU has the functions for mechanisms, which include the launch locks, motors and propulsion valve drive. Please note that the pressure sensor telemetry goes to the C&DH Unit. All cards have I2C interfaces.

The Motor Controller Card (MCC) has the H-bridge circuits and the relays to control 3 phase stepper motors. Each MCC has the ability to drive four motors. Currently one MCC is baselined for antenna and solar array gimbals. Two MCCs are baselined for redundancy.

The Mechanism Release Card (MRC) has the ability to control eight mechanisms switching both high and low side with an arm switch. Two MRC are currently baselined for redundancy.

The Propulsion Drive Card (PDC) has the ability to control eight valves switching both high and low side with an arm switch. Six PDCs are currently baselined for redundancy.

Lastly, the MPU has two LVPS cards to provide secondary voltages to the MPU cards. Two for redundancy.

All the parts are readily available. The board designs are based upon the Eurocard form factor 3U and 6U. For the C&DH Unit, the backplane is cPCI. For the PSE and MPU the backplane is based upon an I2C interface. The board level products exist that could be used to piece together a system. The Orbiter processor is a COTS product.

2.2.6 Electrical Power

The LGN mission consists of two distinct segments – an orbiter and a lander. The orbiter power system consists of solar arrays, a secondary battery, and supporting power electronics. The LGN orbital period is 2.2 hrs. with .8 hrs. of night. TJGaAs solar cells with bare cell efficiency of 29.5%, a solar constant of 1353 w/m^2, array operating temp at 130 deg C, and Space Environmental Effects and Education System (SPENVIS) solar array degradation factors were used to derive the array area requirement. A single two axis tracking panel with 3.1m^2 active area (3.45 m^2 total substrate area) will provide 744W EOL, 837W BOL of power to support loads and battery recharge. A high energy density 22AH Li Ion battery is used to support night loads. The Power System Electronics (PSE) will be a heritage 28VDC battery dominated bus included as cards in the avionics package. The PSE will control battery charging and power distribution.

2.2.7 Thermal

The Lunar Orbiter thermal design consists of placing radiators on the Orbiter surface to dissipate electronics heat while keeping the radiators out of the Sun (and view of the hot daytime Lunar surface). Approximately 455 watts of heat need to be dissipated from the Instruments, Avionics, and Comm system with radiator patches on the Orbiter's two side radiator surfaces. A warm cavity approach eliminates Propulsion system heaters on the Prop tanks and lines, but adding heaters to the Torque tube and painting the inner surfaces of the Orbiter black. All thermal hardware has a high TRL level.

1. Heater patches are used on the Orbiter's sides
   - Approximately 1.25m^2 of radiator area is needed to dissipate the 455 watts (orbit average) of Orbiter's Electronics heat. 133 watts of heat is dissipated in the safe hold case.
   - Small patch radiators can be used for every box, or spreader heat pipes can be embedded in the Orbiter's lower deck to help spread heat from the Deck to Radiator Patches.
   - Radiators can be placed on one side of the Orbiter or placed on opposite Orbiter sides; thermally, they are equivalent.
2. The thrust tube is painted black (internal and external surfaces) to radiate heat into the Propulsion tanks. The internal walls of the Orbiter are painted black as well to keep the tanks warm.
3. White MLI blanketing covers the entire external surface of the Orbiter, except for the radiator patches which are made up of Z93C55 White paint.

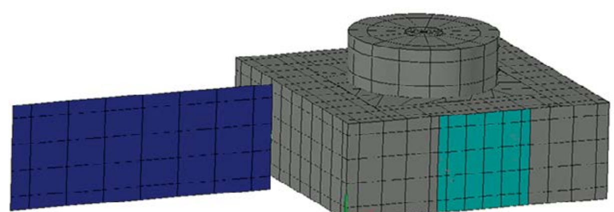

Figure B-50: External Thermal Model.

Figure B-50 shows the external thermal model of the Orbiter.

The Orbiter must keep the electronics boxes (instruments, Comm system, and Avionics) below its operating temperature limits during the Lunar orbit. To do this, the two radiator surfaces remain parallel to the Sun (beta angles 0° through 90°). The single axis solar array will rotate to keep it perpendicular to the Sun.

Figure B-51 shows the orbit profile for a Beta 0 orbit. Orbit Altitude is assumed to be 250 km for the hot biased case.

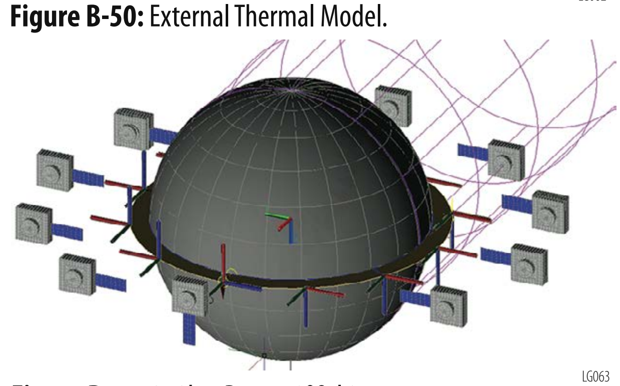

Figure B-51: 250km Beta =0°Orbit.

Figure B-52 shows a plot of the radiator temperatures as a function of time throughout the orbit.

Figure B-53 and **Figure B-54** show the Orbiter hot case and cold case. The propulsion system (tanks and lines) stay warm by 70 watts of heater power on the Torque Tube, keeping them above the hydrazine freezing point at all times.

It should be noted that to keep the electronics within operating temperature range while orbiting the Moon (hot biased conditions), there needs to be 32 watts of heater power (in addition 133 watts of electronics power + heater power) to keep the boxes above survival limits during the cold safe hold case.

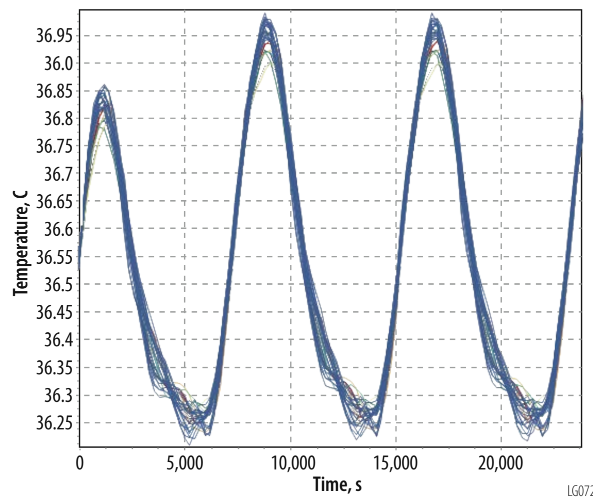

Figure B-52: Radiator Temperature, Beta=0°Hot Case.

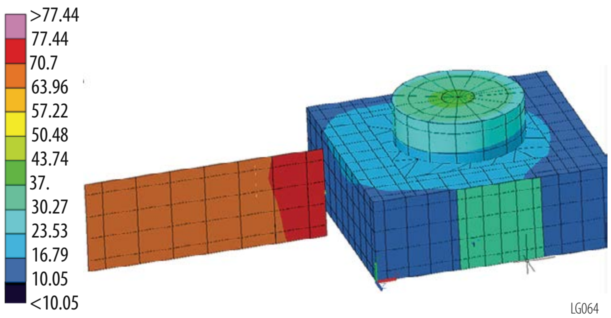

Figure B-53: Orbital Temperatures during Hot Beta=90°

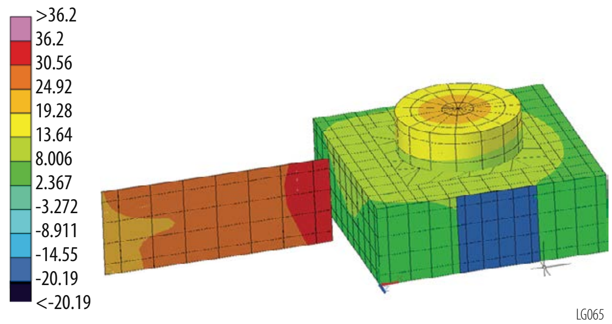

Figure B-54: Cold Survival Temperatures

3. LGN AND APOLLO LANDING SITES AND MOONQUAKE LOCATIONS

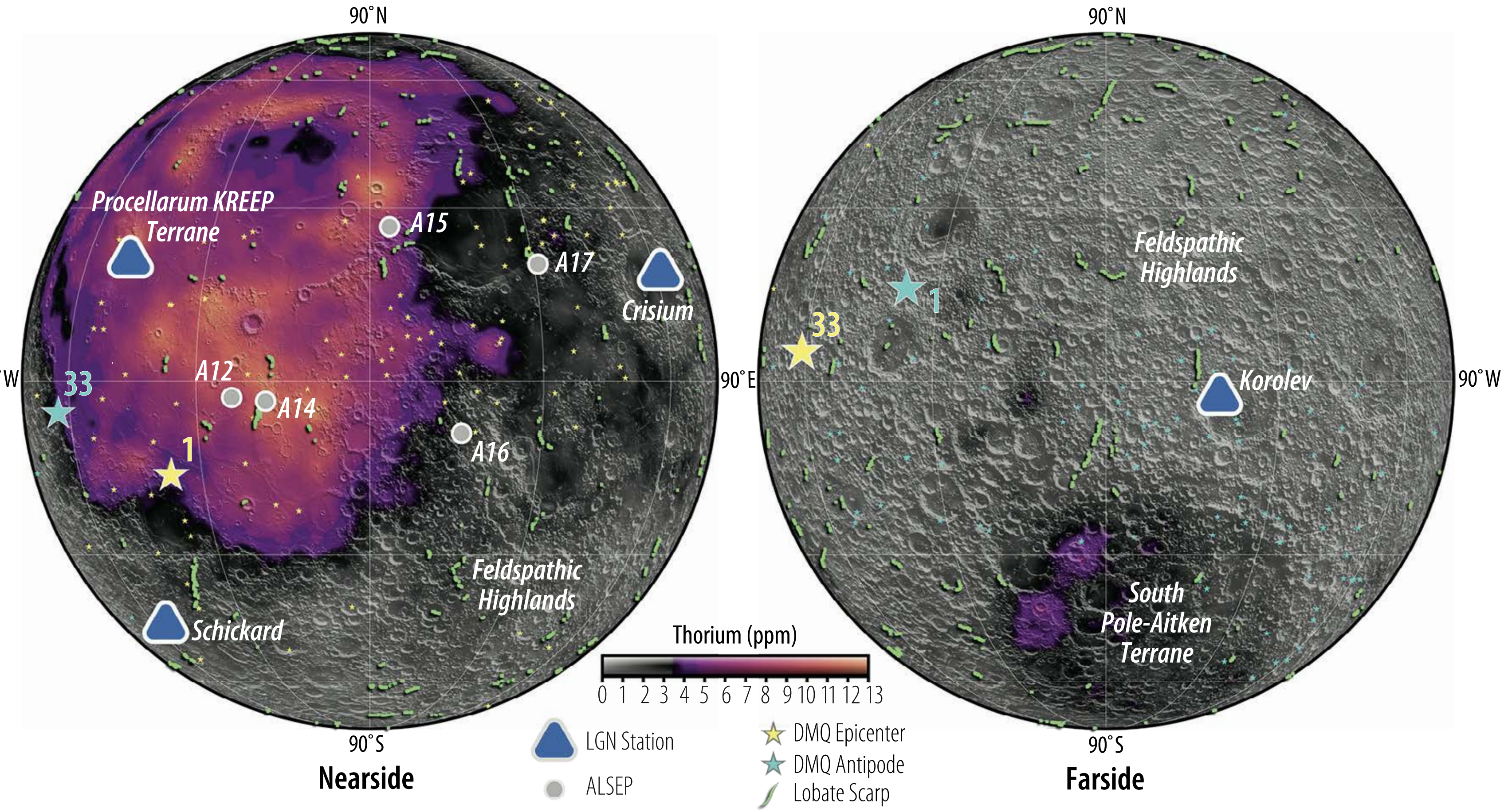

Figure B-55: The Lunar Geophysical Network will place stations that sample across major lunar terranes and enable new investigation of the deep interior and tectonic evolution. The proposed LGN stations (blue triangles) are positioned to take advantage recent lobate scarp seismicity (green lines, Watters *et al.*, 2019) and deep moonquake (DMQ) clusters (yellow stars/dots) and their antipodal locations (cyan stars/dots), both types of seismicity detected by the Apollo ALSEP stations (gray circles). Two deep moonquake clusters (01 and 33) are highlighted (see text). The LGN is designed to geophysically interrogate the internal structure, temperature, composition, and tectonics/seismicity both in the Feldspathic Highlands (unsampled by Apollo) and within the Procellarum KREEP terranes, outlined by the Lunar Prospector thorium abundance *e.g.*, Lawrence *et al.*, 2000. Shaded surface relief is derived from LOLA topography (Smith *et al.*, 2017).

APPENDIX C: THE SCIENTIFIC RATIONALE FOR DEPLOYMENT OF A LONG-LIVED GEOPHYSICAL NETWORK ON THE MOON

A White Paper to be Submitted to the 2023 Planetary Science Decadal Survey.

THE SCIENTIFIC RATIONALE FOR DEPLOYMENT OF A LONG-LIVED GEOPHYSICAL NETWORK ON THE MOON

A White Paper to be Submitted to the 2023 Planetary Science Decadal Survey

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Devanshu Jha, MVJ College of Engineering
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Brigitte Knapmeyer-Endrun, U. of Cologne
Krishan Khurana, UCLA
Juan Lorenzo, LSU
Angela Marusiak, JPL
Patrick McGovern, LPI
Laurent Montesi, UMD
Francis Nimmo, UCSC
Deanna Phillips, UAH
Jacob Richardson, UMD
Charles Shearer, UNM/LPI
Krista Soderlund, UT Austin
Sean C. Solomon, Columbia University
Tilman Spohn, ISSI
Eleonore Stutzmann, IPGP
Sonia Tikoo, Stanford University
Slava Turyshev, JPL
Dany Waller, JHU
Ryuhei Yamada, University of Aizu
Maria Zuber, MIT

Introduction

This white paper focuses on the scientific rationale for deploying a long-lived, global network of geophysical instruments on the surface of the Moon to understand the nature and evolution of the lunar interior. The acquired data will allow the examination of initial planetary differentiation processes that are preserved on the Moon. Evidence for such preservation comes from mare basalt samples derived from source regions consistent with having been emplaced from an early lunar magma ocean [1]. Geophysical data are critical to understanding terrestrial planet formation and early differentiation processes, and also for understanding the collision process that generated our unique Earth-Moon system. These geophysical observations of the Moon will yield a wealth of Solar System-level knowledge that builds on the Apollo geophysical experiments and exploits data from the Lunar Prospector, Kaguya, LRO, and GRAIL missions.

Over a minimum of 6 years (covering one primary tidal cycle), new data collected will characterize the nature and evolution of the lunar interior using a combination of seismic, heat flow, laser ranging, and electromagnetic sounding data. Furthermore, these data will help to constrain the Moon's current electrostatic charging environment and meteoroid impact flux, including hazard assessment. Data from a modern geophysical network will expand upon pioneering measurements made by the Apollo Lunar Surface Experiments Packages or ALSEPs (at Apollo 12, 14, 15, 16, and 17) and the two Lunokhod retroreflectors (Fig. 1). The ALSEPs returned essential data on the lunar surface environment and the lunar interior. However, fundamental questions remain unresolved, in part because of the sensitivity of the instruments but also because the ALSEP stations were clustered in a small equatorial region on the nearside of the Moon (Fig. 1). The Apollo landing sites were later discovered to straddle a geological province that differs from the rest of the Moon by its enrichment in heat producing elements, now referred to as the Procellarum KREEP Terrane (PKT) [2]. Unbeknownst to researchers at the time, much of the data acquired from the ALSEP stations are now thought to be unrepresentative of the Moon as a whole.

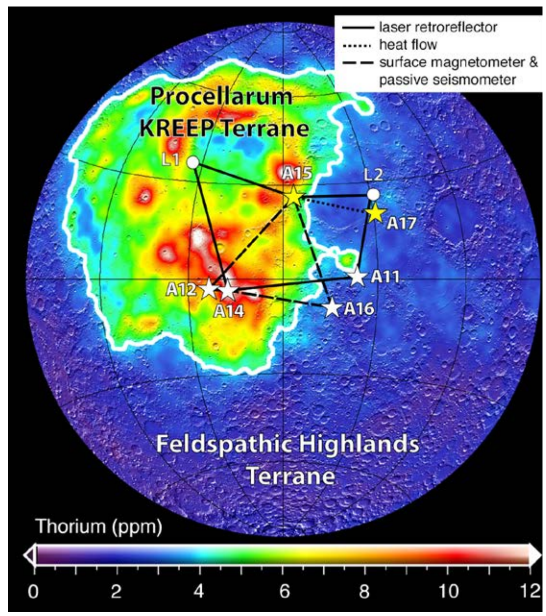

Figure 1: Locations of ALSEP and Lunokhod stations in relation to the boundary of the PKT.

At the time of writing, 43 years have passed since the ALSEP stations stopped returning data from the lunar surface. In the intervening period, a wealth of observations from later orbital missions, new analyses of Apollo data and samples, and improved modeling techniques have advanced our scientific understanding of the Moon, sometimes offering conflicting hypotheses for some of the most fundamental processes that shaped the lunar interior. Taking these advances into account, it is clear that a more nuanced view of the lunar interior drives new questions that can be answered only by a Lunar Geophysical Network.

Summary of lunar internal structure

The Moon, like other terrestrial planets, is differentiated into a crust, mantle, and core. This structure is a consequence of the accretion of the Moon from a circum-terrestrial impact-generated debris disk, and its subsequent differentiation from an initial magma ocean. To first order, the Moon's moment of inertia is roughly approximated by that of a homogeneous sphere. However, the modern Moon exhibits strong departures from the simple spherically symmetric stratified interior expected at the end of magma ocean crystallization, with hemispherical heterogeneities in crustal thickness, volcanism, magnetism, and the distribution of heat-producing elements.

Knowledge of the lunar interior stems from a long line of missions stretching back more than 50 years. Lunar Laser Ranging (LLR) has precisely monitored the Moon's solid-body motions since 1969, using retroreflector arrays deployed during Apollo and the Russian Luna missions. Changes in the round-trip laser travel time provide information on the rotation of the Moon which can be analyzed to separate orbital motion from geodynamic effects. Dissipation inferred from LLR data provided the first evidence for a fluid core, with a radius dependent upon composition.

Electromagnetic sounding of the Moon performed during and after the Apollo program provided broad constraints on core size, mantle composition, and interior temperature. Electrical conductivity and mantle temperatures were constrained at the Apollo 12 site, using concurrent surface and orbital magnetometer measurements. The observed lateral heterogeneity in electrical conductivity is consistent with the presence of the PKT. The Lunar Prospector and Kaguya magnetometers also detected an induced moment within the Moon, observed in Earth's geomagnetic tail. Under the assumption that the induced field is caused by electrical currents near the surface of a conductive metallic core, the core radius was estimated.

Seismometers were deployed on the lunar nearside at five Apollo sites, and operated continuously at four of those sites from 1969 to 1977. Many different types of naturally occurring seismic events were recorded, including deep, shallow, and thermal moonquakes, and meteorite impacts. These events continue to be analyzed to produce seismic structure models. Early models based on arrival time inversion alone were supplanted by newer models using maximum likelihood estimates, joint seismic and pre-GRAIL gravity inversion, and free oscillations. Crustal thickness estimates have decreased over the years as newer and more computationally intensive techniques were applied. The newer models mostly agree that the only major discernible discontinuity in the lunar interior is the crust-mantle boundary located at about 30 km depth. Although the small aperture of the Apollo passive array limited initial constraints of average radial structure below ~1000 km depth, recent re-analyses have found evidence for a core from reflected phases.

The recent GRAIL mission mapped the Moon's gravity field in extreme detail. Shallow crustal structure is tightly constrained, but still tied to ground-truth seismic estimates at the Apollo landing sites and hence carries the associated uncertainty. Although the GRAIL mission produced a family of core models consistent with geodetic parameters and seismic constraints, perspectives differ on whether a partial melt layer in the lowermost mantle is required to satisfy available constraints. A melt layer is consistent with inversions of multiple geophysical data in combination with phase-equilibrium computations, but not required in viscoelastic dissipation models derived from laboratory measurements of deformation of melt-free polycrystalline olivine.

Heat flow measurements were performed on Apollo 15 and 17. Characterization of the lunar global heat loss is important in understanding the thermal evolution of the Moon. In the crust-mantle differentiation process, greater concentrations of radiogenic heat-producing elements (U, Th, and K) likely ended up in the crust. It is essential to quantify the crust's radiogenic contribution to the heat flow released through the lunar surface, which we are able to measure, in constraining the thermal structure of the deeper interior. Would heat production vary among the crust of different terranes or between maria and highlands? Is there any geographic variation in the heat flow out of the mantle? Such knowledge would also help us answer questions on the history of a possible lunar core dynamo, by which the Moon may have generated and maintained its own global magnetic field in the past.

Magnetism is ubiquitous in the Solar System. Both deep structure and magnetism have bearing on the now-extinct lunar dynamo, which in turn has implications for lunar thermal history and core state. Paleomagnetism and crustal magnetism studies can inform understanding of the dynamo, but the precise origin of lunar magnetic anomalies is still unclear. In addition, a magnetic low has been observed beneath the PKT, which invites questions as to the depth, history, and extent of magnetic carriers. The full nature of the extinct lunar dynamo is unknown, and even its existence on such a small body is surprising. Recent modeling shows that core convection driven by a single mechanism, in particular thermochemical convection, cannot explain either the dynamo duration or the inferred magnitude intensity of the paleofield.

The Moon's complex present-day surface record has been driven by the time-integrated history of its internal processes. The Lunar Geophysical Network mission will allow us to develop a structure model for the Moon that is consistent with all observations and provides logical context for the Moon's early history and insight into broader terrestrial planet formation and evolution.

Limitations of existing data

Apollo Passive Seismic Experiment: These seismometers were deployed at every Apollo site except Apollo 17 (the instrument at Apollo 11 provided data for only 21 days) [3]. A network of four seismometers was completed in April 1972 (Fig. 1) and operated until 30 September 1977. The experiment clearly demonstrated that the Moon exhibits seismic activity at a similar level to that of an intraplate setting on Earth [4,5]. Multiple types of lunar seismic events were identified and used to infer global 1D structure [6], but lunar seismograms suffer complications compared with their terrestrial counterparts. The lunar megaregolith intensely scatters seismic energy. Secondary phases, which contain information on deep structure such as reflections of seismic energy from the crust-mantle (Moho) or core-mantle boundaries, are therefore masked by codas.

Heat Flow Experiment: Four heat flow measurements were made on the Moon during Apollo: two at the Apollo 15 site and two at Apollo 17 [7]. Unfortunately, these measurements were made in the crustal transition zones between terranes (Apollo 15) and between maria and highlands (Apollo 17), and therefore provided an ambiguous mixed signal of these geologic provinces [8,9].

Lunar Surface Magnetometers: Static magnetometers were deployed at the Apollo 12, 15 and 16 landing sites, where data were collected until 14 June 1974 [10,11]. In addition, portable magnetometers were employed as part of the Apollo 14 and 16 missions. These measurements quantified the strength and direction of the remanent crustal fields at the Apollo landing sites, and how they varied over kilometer length scales with the portable measurements. Apollo 15 and 16 also deployed orbiting magnetometers on subsatellites. For Apollo 12, concurrent measurements with the orbiting Explorer 35 satellite enabled probing of the lunar interior using electromagnetic induction. However, this magnetic transfer function approach was band limited by plasma effects, such that the minimum investigation depth was a few hundred kilometers. At greater depths, the temperature profiles inferred from the recovered electrical conductivity structure were in broad agreement with thermal models, but significant uncertainty remained due to a combination of measurement error and constitutive relations.

Lunar Laser Ranging (LLR) [12,13]: Retroreflectors were deployed by Apollo 11, 14, and 15 astronauts, and were also fitted to the Soviet rovers Lunokhod 1 and 2 (Luna 17 and 21 landers, respectively; Fig. 1). There are now 50 years of increasingly accurate Earth-based laser ranges. LLR analysis allows an evaluation of the deep lunar interior that extends to interactions at the core-mantle boundary as well as the Moon's deepest mantle. Dissipation at the core-mantle boundary indicates the presence of a fluid core with a radius about 20% of the Moon [14,15], but an inner core has not yet been independently constrained with LLR analysis, in part because the restricted geographical extent of the existing retroreflector network limits the accuracy of lunar rotation and tide determinations [16].

Unanswered questions on the lunar interior

Despite 50 years of Apollo data analysis and more recent orbital constraints on lunar internal structure, we still do not have unambiguous observations of a mid-mantle discontinuity, the mineralogy and temperature profile of the upper mantle, the nature of the lower mantle, the presence of a partial melt layer above the outer core, or the nature of the inner core [6,17-22]. There is no consensus regarding the presence of a mid-mantle seismic discontinuity, which has been used to suggest the lower bound of an ancient lunar magma ocean (LMO). If a discontinuity is not present, or present but not global, the LMO model may need to be revised. If it is global, it suggests that the Moon did not completely melt, which has implications for its thermal evolution. Seismic data have additionally been interpreted to indicate the presence of garnet in the lower lunar mantle [23-25]. However, the same data were also interpreted to represent an increased proportion of Mg-rich olivine [18,26]. Such interpretations have implications for the bulk Moon composition.

Core constraints include [27] but do not require [28] the presence of a partial melt layer above the liquid outer core, and other analyses both support [29] and discount [30] the likelihood of its existence. The details of this deep structure are needed because they fundamentally affect the origin, extent, and duration of the lunar dynamo and the resulting record of crustal magnetic anomalies, including those at swirls. Over 50 years of laser ranging data have markedly contributed to our understanding of the lunar core and mantle, but the current network is not sufficiently distributed to provide conclusive answers regarding the size and density of the lunar core, the presence and properties of a solid inner core, and the nature of the free nutation (analogous to the Chandler wobble). Furthermore, new retroreflectors have been designed to provide more accurate ranges [31,32]; using these to expand the current network on the lunar surface would substantially improve the determination of 3D rotation and tides (and the geophysical quantities derived from them), our understanding of the deep mantle environment, and our constraints on the presence/absence of a solid inner core and fluid core/solid mantle boundary conditions. This expanded and enhanced LRR network will also address fundamental physics questions [33,34].

More than 7,000 deep moonquakes were recorded by Apollo, clustered in 318 source regions or nests, but <10 nests are undisputedly on the farside [35]. The attenuating core may prohibit detection of seismic energy from nests that could exist on the farside. Although tides are known to influence the occurrence times of deep moonquakes, the full mechanism remains unknown [36]. The precise locations and origin(s) of the rare shallow moonquakes are likewise unknown. They were initially suggested to be associated with boundaries between dissimilar surface features (e.g., impact basin rims [37]), later attributed to the interaction of the Moon with nuggets of high-energy particles ("strange quark matter") originating outside the Solar System [38,39], and most recently suggested to represent slip on tectonically active faults that underlie lobate scarps [40]. Because these are the largest lunar seismic events, they are interesting not only scientifically but also for exploration initiatives, as seismic shaking may have implications for any future infrastructure supporting a sustained human presence on the Moon.

Variations in the lunar crust (mineralogy and thickness) have been difficult to constrain far from the Apollo network sites using seismic data. Markov chain Monte Carlo algorithms have been applied to seismic wave arrival times from artificial and meteoroid impacts to estimate crustal thickness variations [41], but studies of this type are limited because the seismic arrivals from such impacts are highly uncertain. New seismic constraints are needed to provide ground truth for GRAIL's global constraints on crustal thickness [42].

Because most of the Apollo sites are located in crustal transition zones, new geophysical data from well inside areas of relatively uniform geology are needed to contextualize the two existing heat flow measurement pairs [7]. Even though we now have maps of the surface abundance of radiogenic heat-producing elements (Fig. 1), we do not know their vertical distribution through the crustal layer. Therefore, we do not have tight constraints on the total heat generated within the PKT crust as a whole and the heat production in the mantle beneath it. We also lack knowledge of the base-level heat flux outside the PKT. These represent significant knowledge gaps in defining the Moon's global heat flow budget [8,43] and the bulk composition of the Moon in terms of radioactive heat-producing elements [9,44].

The electrical conductivity structure of the outermost 500 km of the Moon, and its lateral variation, is not well understood. This zone is important as it may contain a transition from upper-mantle melt residuum to pristine lower mantle, as well as differences in crustal composition and lithospheric thickness and heat flow associated with the primary geological provinces of the Moon. Similarly, the deep conductivity structure of the Moon is under-constrained [45]. A tighter average mantle conductivity profile will better constrain temperature and composition. Furthermore, very long-period measurements could distinguish a molten silicate from an iron core.

Proposed instruments for a next generation geophysical network

Four primary instruments can jointly define the interior structure of the Moon, constrain its interior and bulk composition, delineate the vertical and lateral heterogeneities within the interior as they relate to surface features and terranes, and evaluate its current seismic and tectonic activity.

Seismometer: The recent Mars InSight mission has demonstrated that seismology continues to serve as the key tool for assessing planetary interiors. Like the SEIS package on InSight [46], a future Lunar Geophysical Network should carry a seismometer with a broad bandwidth, low noise floor, and improved sensitivity over those deployed during Apollo.

Heat flow probe: The heat flow is obtained as a product of two separate measurements of the thermal gradient and the thermal conductivity of the regolith over the depth interval penetrated by a probe [7]. The probe should reach a depth below the influence of the diurnal and annual insolation cycles (2 to 3 m). Sites of measurements should be distributed among the various types and thicknesses of crust.

Magnetotelluric (MT) sounder: The MT method can greatly improve imaging of electrical conductivity structure, as it is largely insensitive to plasma effects and can achieve higher bandwidth and better depth resolution. MT measures both electric and magnetic fields and does not require a reference orbiter. Constitutive relations have been much better determined since Apollo so that, together with heat flow, composition and temperature can be robustly separated. A combination of surface measurements and orbital reference may also improve core size constraints.

Laser retroreflector: Modern retroreflectors consist of a single corner cube, eliminating temporal spreading of the laser pulse intrinsic to previous retroreflector arrays. The resulting improvements in range accuracy improve accuracy of science results by factors of 3 to more than 100, depending upon the chosen parameter. These passive retroreflectors will have a lifetime that extends to many decades, as has been the case for the Apollo retroreflector arrays [31, 32].

Supporting measurements and technology:

1. Impact flash observation: Meteoroid impacts can be localized spatially and temporally using ground-based telescopic and orbital observations [47,48]. These impact events provide seismic sources that can be used to constrain and refine structure models.
2. Surface plasma physics: Characterization of surface plasma properties provides context for induced electromagnetic field analyses and improves understanding of the spatiotemporal input processes that influence volatile transport, surface weathering, and surface charging.
3. Farside communications relay: An optimally deployed geophysical network would be widely distributed and include one or more stations deployed on the farside. A widespread distribution would require a communications relay satellite to deliver data back to Earth.
4. Long-lived night survival and operations: To permit continuous observations, each station would require power and thermal systems capable of surviving the harsh extremes of the lunar environment, for a minimum of 6 years (~76 lunations).

Relevance of a Lunar Geophysical Network to Solar System science

This white paper demonstrates that a globally distributed, long-lived geophysical network on the Moon, with each station containing a sensitive broadband seismometer, heat flow probe, magnetotelluric sounder, and laser retroreflector, will address many fundamental lunar science hypotheses that remain to be tested, including the magma ocean hypothesis, the stagnant lid hypothesis, and the early lunar dynamo hypothesis. Furthermore, the Moon provides a nearly pristine compositional and temporal record of formation and evolution through time, which can be extrapolated to model the evolution of other terrestrial planets. It is the only readily accessible body to study the relationship between parent bodies and their satellites. The volatile history of the Earth-Moon system is preserved on the Moon; electromagnetic sounding and to a lesser extent seismology can address if the lunar interior is dry and degassed, or volatile enriched. Early crustal evolution and the effects of giant impacts can inform models by which increasing fracture density enables plate recycling on larger bodies. The risk to future human exploration from moonquakes and meteoroid impacts can be addressed by long-term monitoring of the seismicity and impact rates of the Moon. Establishing a geophysical network on the Moon is therefore critical to gain a better understanding of lunar and inner Solar System science and facilitate future lunar exploration.

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APPENDIX D: REFERENCES

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