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Satellite-Enabled Lunar Exploration Network Expansion

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Abstract

The increasing ambition for robotic and human exploration of the Moon necessitates reliable infrastructure to ensure safe and efficient operations. One challenge lies in ensuring precise navigation and continuous communication to enable scientific activities, resource extraction, and exploration. Addressing this gap, this paper focuses on the Phase 0 and Phase A development of a cost-effective Global Navigation Satellite System (GNSS) and telecommunication constellation for the Moon, designed to provide coverage across 95% of the lunar surface by 2037. The constellation's deployment will involve three launches, primarily using SpaceX Starship, with Ariane 6 as a backup option. Each launch will deploy eight satellites along with a lander equipped to serve as a ground station on the lunar surface. Initial simulations validate the system's performance, including coverage, orbital stability, and communication latency, to meet mission objectives. By leveraging modular spacecraft designs, standardized protocols, and cost-efficient deployment strategies, this study outlines a scalable and sustainable approach to establishing essential services for lunar exploration, paving the way for operational deployment within the proposed timeline.

1. Introduction

Lunar exploration has entered a new era, driven by ambitious international efforts to establish a sustained human and robotic presence on the Moon. Missions such as NASA's Artemis program and ESA's Moonlight initiative underline the growing demand for reliable infrastructure to support scientific research, resource extraction, and surface operations. Central to these efforts is the need for precise positioning systems and robust telecommunication networks, which are critical for navigation, real-time data transfer, and operational coordination across the lunar surface.

The harsh lunar environment, characterized by extreme temperature variations, radiation exposure, and gravitational irregularities, presents unique challenges to the development of such systems. Addressing these challenges requires innovative solutions that balance technical performance, sustainability, and cost-effectiveness.

1.1 Mission Statement

1.1.1 Goals

The SELENE mission (Satellite-Enabled Lunar Exploration Network Expansion) aims to support the increasing ambition for robotic and human lunar exploration by establishing a reliable and cost-effective infrastructure for positioning and telecommunication services. This infrastructure will enable precision navigation and continuous communication for scientific research, resource utilization, and long-term exploration activities on the Moon.

The primary goal of the SELENE mission is to achieve 95% surface coverage of the Moon with navigation and communication capabilities, enabling sustainable exploration and supporting the global space exploration community. This mission aligns with international efforts to create scalable and interoperable space-based systems for lunar operations.

1.1.2 Mission and Objectives

To achieve this goal, the SELENE mission will deploy a constellation of 24 satellites in Medium Lunar Orbit (MLO) and three stationary lunar landers. These assets will provide:

- High-precision navigation services with a positioning accuracy of 1 to 5 meters on open terrain, enabling activities such as mining operations, surface navigation, and precise landing near bases.

DOI: 10.13009/EUCASS2025-213

- Reliable Moon-to-Moon communication with low latency (<200 ms) and medium data rates (>20 Mbps), supporting remote rover control and base-to-base communications.
- High data-rate Moon-to-Earth communication (>200 Mbps) for real-time data transfer, including video calls, scientific data, and telemetry.
- Sustainability through robust design and redundancy, ensuring system operation for at least 10 years with 99% uptime.

Additionally, the mission will promote international collaboration by enabling open access to GNSS and communication services using standardized, open-source protocols. By designing for modularity and scalability, SELENE aims to serve as a template for future interplanetary navigation and telecommunication systems.

1.1.3 Stakeholders

The SELENE mission is designed to address the needs of multiple stakeholders, including:

- **Space Agencies:** Supporting international lunar exploration efforts (e.g., NASA Artemis, ESA Moonlight).
- **Private Companies:** Enabling commercial activities such as resource mining and tourism.
- **Scientific Community:** Facilitating high-resolution mapping, surface operations, and real-time data transfer for research purposes.
- **Governments:** Demonstrating technological leadership in space exploration.

The stakeholder network integrates these actors through open protocols, ensuring broad usability and fostering collaboration across missions. A simplified stakeholder value network (SVN) helps to visualize their interactions in figure 1, highlighting the mutual benefits of the SELENE infrastructure.

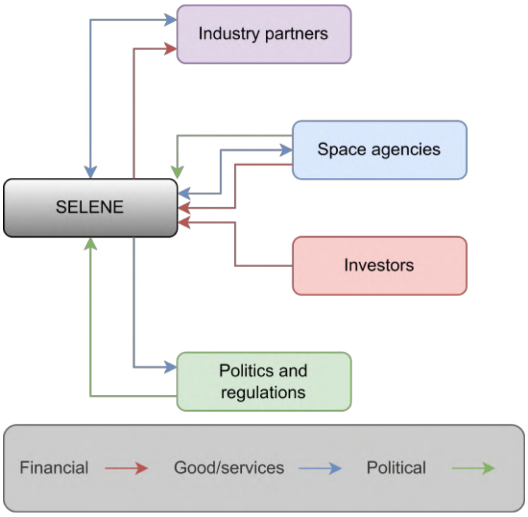

Figure 1: Stakeholder value network for the SELENE project

1.1.4 Comparative Anaysis

The SELENE mission builds on lessons learned from prior missions such as Galileo, Starlink, and NASA's Lunar Reconnaissance Orbiter. While Galileo and Starlink demonstrated the feasibility of large-scale GNSS and communication networks, SELENE extends these capabilities to the lunar environment. Its innovative approach to combining GNSS and telecommunication services with a single constellation offers a cost-effective and scalable solution for future lunar and interplanetary missions.

1.1.5 Flow Down from Requirements

1. **International Demand for Lunar Infrastructure:** The SELENE mission addresses the need for GNSS and communication systems required for safe navigation and real-time data transfer in future lunar exploration missions[7].
2. **Challenges of the Lunar Environment:** The mission's design integrates resilience against extreme lunar conditions, including temperature variations, radiation, and vacuum, to ensure system reliability.
3. **Cost-Effectiveness and Scalability:** Modular designs and commercial off-the-shelf components reduce costs and provide a clear path to scale the system if capacity limits are reached.
4. **Technical Requirements Driving the Design:** The mission is designed to meet critical performance requirements, such as <5 m positioning accuracy, high data rates, and 99% system uptime.
5. **Risk Mitigation through Requirements Flow Down:** Redundant systems and advanced orbit-keeping capabilities are implemented to mitigate risks like orbit instability and hardware failures.

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2. Methods

2.1 Mission design

2.1.1 CONOPS

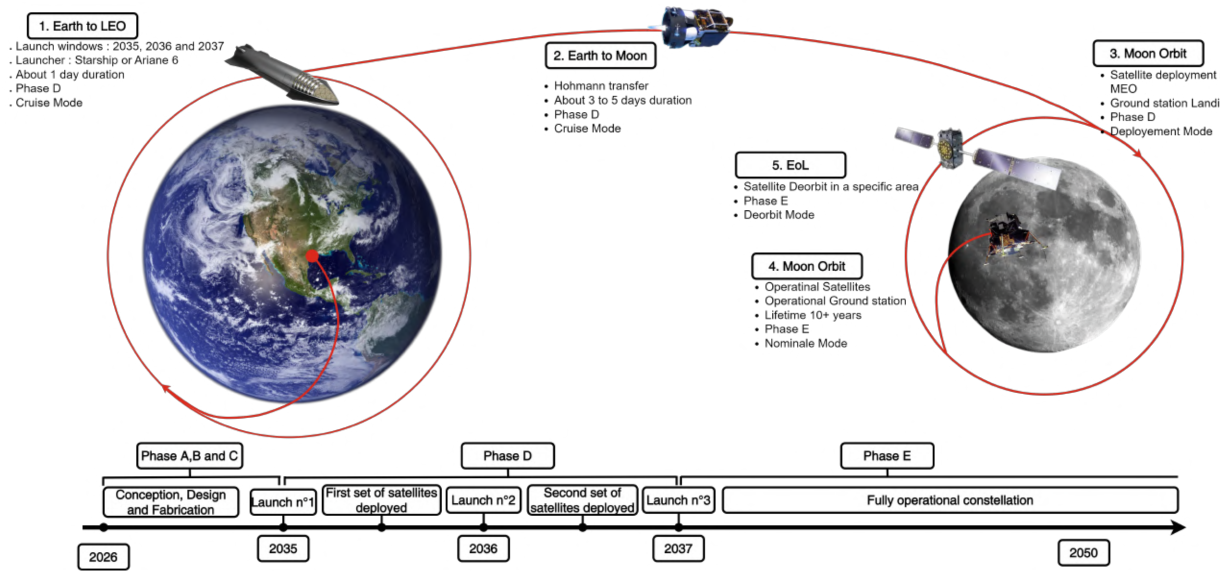


Figure 2: Conops with phases, modes and timeline

Mission Phases

The SELENE mission is divided into distinct operational phases, as outlined below:

1. **Phase A :** Concept Development and Preliminary Design
   Define high-level mission requirements, objectives, and system functionalities.
   Conduct initial feasibility studies, including simulations for orbital coverage, latency, and system stability.
2. **Phase B :** Detailed Design and Prototyping
   Develop detailed designs for the satellite constellation, lander, and associated subsystems.
   Conduct subsystem-level testing, including environmental and functionality validation.
3. **Phase C :** Manufacturing and Assembly
   Manufacture and integrate satellite and lander components.
   Validate system functionality through rigorous end-to-end testing in simulated environments.
4. **Phase D :** Deployment
   Launch three batches of eight satellites each, using SpaceX Starship (or Ariane 6 as a backup).
   Deploy the satellites into MLO and ensure proper alignment. Land the ground station on the lunar surface, establishing initial GNSS and communication capabilities.
5. **Phase E :** Operational Mission
   Operate the constellation for a minimum of 10 years.
   Perform regular software maintenance and updates to ensure high system availability (99% reliability).
   Deorbit satellites at the end of their operational lifetime into designated lunar impact zones.

System Modes

The SELENE mission includes several operational modes tailored to meet the mission requirements throughout its lifecycle.

1. **Cruise Mode**: Activated during the journey from Earth to the Moon. Operates with minimal subsystem activity to conserve energy, while maintaining essential functions like thermal management, propulsion for trajectory corrections, and communication with Earth.
2. **Nominal Mode**: Normal operational mode for GNSS and telecommunication services. Continuous data relay between satellites, the Moon, and Earth.

DOI: 10.13009/EUCASS2025-213

3. **Safe Mode**: Activated in the event of subsystem anomalies or system failures. Minimal functionality to preserve energy and enable diagnostics.
4. **Deployment Mode**: Specific to the initial phases of satellite deployment. Includes activation of propulsion systems for orbit insertion and alignment.
5. **Maintenance Mode**: Used for system updates or repairs during operations. Incorporates software updates and periodic recalibration of navigation payloads.
6. **Deorbit Mode**: Final operational mode for retiring satellites. Executes controlled deorbit maneuvers to avoid creating lunar orbital debris.

Project Timeline: The project timeline aligns with the typical system engineering lifecycle, from concept development to mission deployment and end-of-life.

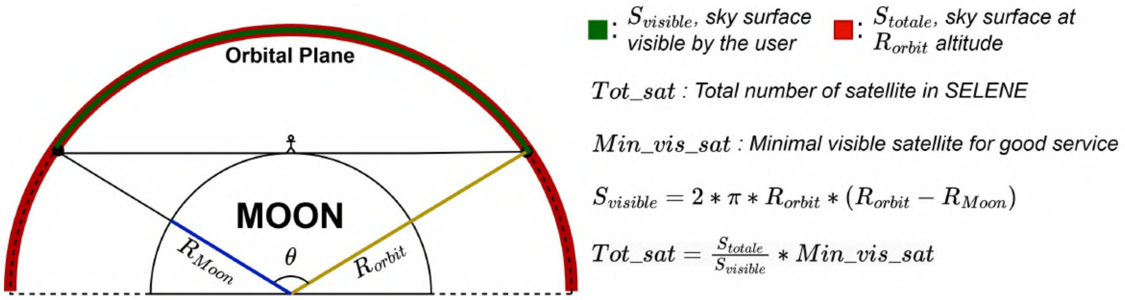
1. **Phase A (2023-2025):** Concept definition and preliminary simulations.
2. **Phase B (2026-2029):** Detailed design and subsystem prototyping.
3. **Phase C (2030-2034):** Manufacturing and integration of satellites and landers.
4. **Phase D (2035-2037):** Deployment of the constellation in three launches.
5. **Phase E (2038-20xx):** Full operation and system maintenance, followed by deorbit. The constellation shall be replaced every 10 years for a continuous operation

Integration of the Lander: The three lunar landers play a crucial role in the SELENE mission, serving as stationary ground stations on the Moon. They provide a stable GNSS reference point and support communication between the satellite constellation and lunar surface users. Each lander is equipped with high-gain antennas, power systems, and thermal management to operate in the harsh lunar environment. It should be noted that most of the work presented in this paper focuses on the design of the satellites, while the preliminary and detailed analyses required for the landers are left for future work.

2.1.2 Orbital configuration

The configuration was chosen to provide full coverage of the lunar surface and low-latency communication with a minimal number of satellites. The orbits were defined to emphasize coverage at the Moon's poles, as these are preferred areas for future lunar landings and bases[17]. However, full coverage capabilities is maintained at the equator. A total of 24 satellites are distributed across 3 distinct orbits. Their semi-major axis is 3600 [km], approximately 2000 [km] above the surface. The inclination is 80 [°] to increase coverage of the polar regions, and each plane has a RAAN difference of 120 [°], ensuring they are evenly spread around the Moon.

The number of spacecraft was determined by basic geometric approximations, as can be seen on fig 3



🟩: $S_{visible}$, sky surface visible by the user 🟥: S_{totale}, sky surface at R_{orbit} altitude

Tot_sat : Total number of satellite in SELENE

Min_vis_sat : Minimal visible satellite for good service

$$S_{visible} = 2 * \pi * R_{orbit} * (R_{orbit} - R_{Moon})$$

$$Tot\_sat = \frac{S_{totale}}{S_{visible}} * Min\_vis\_sat$$

Figure 3: Geometric calculation for the number of satellites required

With $R_{orbit} = 3600[km], R_{moon} = 1734[km]$ and $Min\_vis\_sat = 5$, we obtain a total of 23.15 satellite. This number is rounded upward to 24. Min_vis_sat could theoretically be set to 4, but this simple computation doesn't take into account valley, mountain or the imperfect distribution of satellites in the sky.

One ground station is positioned near the south pole to maximize contact time with the constellation for orbit and GNSS corrections, while the other two are placed on the Earth-facing side of the Moon at the equator to maintain a permanent view of Earth and serve as the main relay stations between Earth and the SELENE system.

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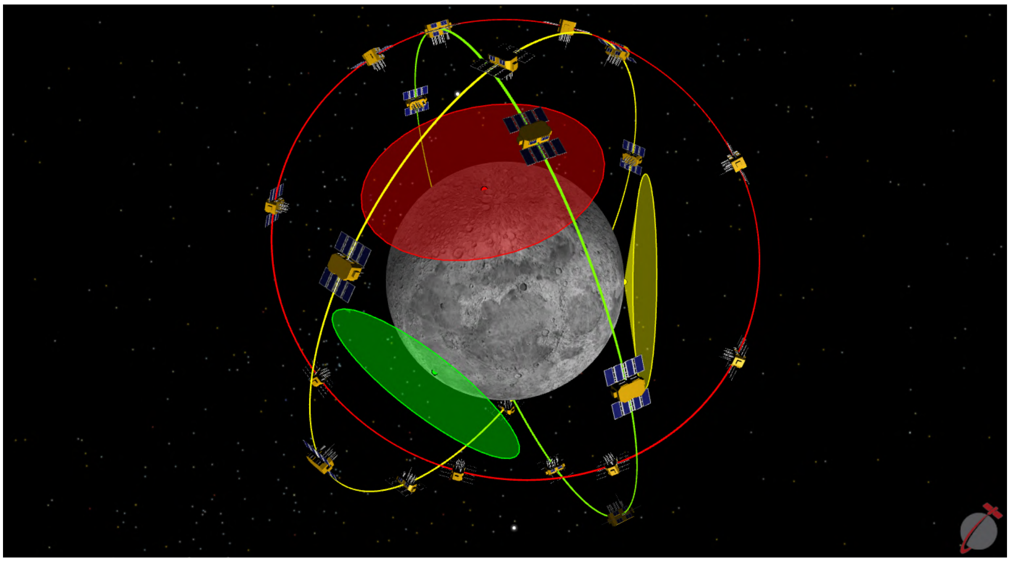


Figure 4: Scaled representation of the orbital configuration of the constellation generated using the Freeflyer software. The polyhedron represents the visibility cone of each ground station.

Using Freeflyer, it was estimated that on average, any given satellites was in contact with the polar ground station around 22.64 [%] of the time along a full 27 [days] lunar orbit, while the contact with a equatorial station was lower at 17.74 [%]. In average, any given satellites is in contact with at least one of the ground station 56 [%] of the time (compared to 52.8 [%] with an equatorial-only configuration. The values drop down to 35.26 [%] if only the earth-facing ones are considered, which is an ideal configuration where a moon-earth communication could happen directly through the ground station with only one relay through a satellite.

2.2 System engineering

2.2.1 Functionality Analysis

The functionalities of the SELENE mission stem directly from its mission statement, which focuses on robotic and human lunar exploration through precise positioning and robust telecommunication services. These functionalities are based on an in-depth functional decomposition and are linked to the mission CONOPS, customer requirements and operational context.

Positioning: Positioning is a key functionality of SELENE, designed to offer sub-5-meter accuracy over 95% of the lunar surface. This accuracy enables precision landing, surface navigation for rovers and astronauts, and real-time tracking of equipment. The positioning system relies on data acquisition via satellite signal transmission and user detection, enabling the system to identify active objects on the surface. The acquired signals are processed to calculate distance using time synchronization provided by atomic clocks.

Setting up a telecommunications network: SELENE is also setting up a telecommunications network to ensure high-speed, low-latency communications. The signal transmission/reception system ensures the continuous transfer of telemetry, scientific data and video streams between the Moon and Earth, as well as Moon-Moon communication. The telecommunication system includes communication management functions such as bandwidth allocation, ensuring that critical data, such as telemetry or emergency signals, are given priority over less urgent tasks. The system supports collaborative operations between lunar bases, rovers and mission control on Earth, meeting customer requirements for seamless integration between platforms.

2.2.2 Requirements

High-Level Requirements

- The system shall provide sub-5-meters positioning capabilities on at least 95% of the moon surface
- The telecommunication system shall ensure data rates exceeding 200 Mbps for Moon-to-Earth communication and 20 Mbps for Moon-to-Moon links, with latency below 200 milliseconds.

DOI: 10.13009/EUCASS2025-213

- The system shall operate with greater than 99% uptime for a minimum lifespan of 10 years under lunar environmental conditions.
- The satellite constellation shall include at least 24 satellites, deployed across three orbital planes by 2037, to ensure consistent coverage and redundancy.

Derived Requirements

- The propulsion system shall provide sufficient delta-v to perform orbital station-keeping and mitigate lunar gravitational perturbations.
- The satellites shall be equipped with radiation-hardened electronics to withstand prolonged exposure to cosmic and solar radiation.
- The system shall include redundant subsystems for critical components, such as atomic clocks and communication modules, to ensure operational reliability.
- The system shall operate in a temperature range from -150[°C] to 150[°C].

Relation Between Requirements: The high-level requirement for navigation accuracy drives the need for precise time synchronization and robust GNSS payload design, while the data rate requirement informs the use of high-bandwidth communication modules and efficient bandwidth allocation. The requirement for a 10-year lifespan flows into the thermal management, radiation shielding, and redundancy measures. Each derived requirement addresses a specific risk, such as radiation-induced failures or thermal stress, ensuring that the design is resilient under lunar conditions.

2.2.3 Mission constraints and limitations

Thermal Variations: The Moon's surface experiences extreme temperature variations between -150[°C] during lunar nights and +150[°C] during lunar days. These variations present risks for the satellite's subsystems, such as thermal stress and failure of electronic components.

Radiation Exposure: The absence of a protective atmosphere on the Moon exposes SELENE's satellites and landers to cosmic and solar radiation, increasing the risk of radiation-induced failure of electronic components. To mitigate this risk, SELENE uses shielding for critical systems and redundant designs to ensure operational resilience.

Vacuum and Lunar Dust: The lunar environment is a vacuum, and its regolith is highly abrasive, presenting contamination risks for optical systems and mechanical parts. SELENE solves this problem by using protective coatings, dustproof materials and sealed housings for sensitive components.

Operational Constraints: Orbital Instability: The Moon's irregular gravitational field creates disturbances that can destabilize satellite orbits, necessitating regular station-keeping maneuvers. This constraint has repercussions on propulsion system requirements, notably a Delta V sufficient to maintain orbital stability.

Flow into Requirements and Risks: The environmental and operational constraints directly influence the mission requirements and associated risks. For instance, thermal and radiation challenges drive the need for robust thermal control and radiation shielding, while orbital instabilities shape the propulsion system's Delta V requirements. Risks such as loss of satellite functionality due to radiation or communication failure are mitigated through redundant systems, advanced materials, and optimized network design.

2.2.4 Mission success criteria

Primary Success Criteria

- A minimum of 19 out of 24 satellites must be successfully deployed in Medium Lunar Orbit (MLO) by 2037 to ensure 95% lunar surface coverage.
- The GNSS system must deliver sub-5-meter positional accuracy across 95% of the lunar surface to support surface navigation, precision landings, and asset tracking.
- The system must achieve data rates of >200 Mbps for Moon-to-Earth links and >20 Mbps for Moon-to-Moon communication, with a latency of less than 200 milliseconds. This ensures effective real-time coordination during the operational phase.
- Continuous service with >99% uptime over a minimum operational lifespan of 10 years

DOI: 10.13009/EUCASS2025-213

Secondary Success Criteria: If primary objectives are not fully met, secondary criteria ensure partial mission success. These include deploying a minimum of 16 satellites, achieving 85% lunar surface coverage, and maintaining functional navigation and communication services. The telecommunication system should deliver reduced data rates of at least 100 Mbps for Moon-to-Earth links and 10 Mbps for Moon-to-Moon communication. Additionally, the system must operate for a minimum of 7 years with sustained performance, providing essential support for ongoing and planned lunar exploration missions.

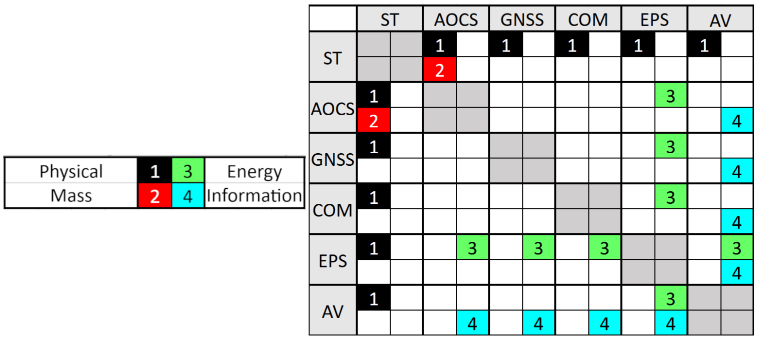
Integration with CONOPS: The success criteria are tightly mapped to the CONOPS phases. During the Deployment Phase, the focus is on the successful placement and activation of satellites, ensuring initial GNSS and communication capabilities. The Operational Phase evaluates sustained navigation accuracy, communication performance, and uptime. Finally, the End-of-Life Phase ensures compliance with deorbiting strategies to minimize orbital debris and maintain long-term sustainability.

2.2.5 Interface analysis

Subsystem Interfaces: One of the most critical internal interfaces is between the GNSS payload and the navigation system, where precise synchronization between atomic clocks and GNSS antennas is essential to maintain the mission's <5 m positioning accuracy requirement. Misalignments or time drifts in this interface could lead to a degradation in positioning accuracy, directly affecting the mission's core functionality.

Propulsion and attitude control systems (AOCS) also form a vital interface, where electric propulsion is coordinated with reaction wheels to maintain orbital stability and support station-keeping maneuvers. The interaction between these systems is essential to ensure that the constellation remains operational throughout its ten-year lifespan. Interface failures could lead to excessive fuel consumption or loss of orbital position.

In addition, the power system interfaces with the thermal management subsystem to regulate power distribution and prevent overheating during prolonged exposure to sunlight, or freezing during eclipses. This interface requires efficient thermal coatings and battery systems capable of maintaining operational temperatures.



Physical	1	3	Energy
Mass	2	4	Information

	ST	AOCS	GNSS	COM	EPS	AV
ST		1 2	1	1	1	1
AOCS	1 2				3	4
GNSS	1				3	4
COM	1				3	4
EPS	1	3	3	3		3 4
AV	1	4	4	4	3 4	

Figure 5: Design structure matrix (DSM) for the satellite

Satellite to Satellite Interfaces: The SELENE constellation relies on laser communication modules for satellite-to-satellite links, ensuring high-speed data transfer and GNSS signal synchronization. This interface enables redundancy in the event of a satellite failure, as neighboring satellites can reroute data.

Satellite to Lander Interfaces: The lander serves as a GNSS reference point and relay station for communications between the Moon and Earth. High-gain antennas enable continuous data exchange between the lander and the orbiting satellites, guaranteeing uninterrupted distribution of GNSS signals and telecommunications.

User Interfaces: Users on the Moon (scientists, rovers, etc.) employ GNSS receivers and communication devices to interact with the constellation and landers. On the lunar surface, GNSS-enabled devices receive navigation signals directly from the constellation, using standard GNSS protocols to ensure compatibility with SELENE and other missions (e.g., Artemis, Moonlight).

Terrestrial users, including mission controllers and scientists, access SELENE via ground stations. The Command and Data Handling (CDH) subsystem manages command inputs, telemetry, and GNSS data distribution. SpaceFibre-based flow control enables access to high-resolution telemetry and scientific data without bottlenecks. Encrypted communication (AES-256) ensures data security.

DOI: 10.13009/EUCASS2025-213

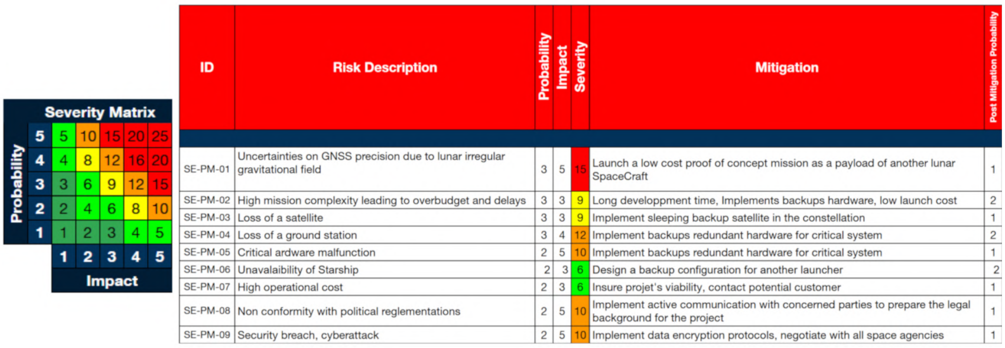
Effect of Interfaces on Requirements: The analysis of these interfaces has directly influenced key mission requirements. For example, the precision demanded by the GNSS payload interface led to the inclusion of triple-redundant atomic clocks to mitigate synchronization risks. Similarly, the vulnerability of laser communication to lunar dust informed the requirement for protective optical coatings. The dependency of thermal management on power distribution resulted in stringent temperature range requirements (-150°C to +150°C), ensuring subsystem survivability in the lunar environment.

2.3 Alternative architectures

Alternative solutions for the SELENE system were examined to address technical, operational, and financial challenges. A high-density, low-altitude constellation was considered to reduce ΔV for station-keeping and enhance GNSS accuracy, but this increased launch demands and complexity. Another approach combined GNSS and telecom on single satellites, reducing unit numbers but complicating payload integration and increasing single-point failure risks. As a backup to Starship, Ariane 6 was proposed for smaller payloads, requiring adjustments to the mission timeline, architecture, and ground station configuration. These options reflect trade-offs in cost, reliability, and complexity.

2.4 Risk analysis

To minimize the probability of failure, a detailed risk assessment was conducted using an asymmetric severity matrix that combines risk impact and probability. Preliminary risk mitigation measures have been proposed, and their consequences analyzed. The main identified risks include uncertainties in GNSS precision caused by the Moon's irregular gravitational field, delays and budget overruns due to mission complexity, potential failures of satellites or critical hardware requiring backup solutions, and the unavailability of the Starship launcher, with Ariane 6 considered a plausible alternative.



Severity Matrix

Probability					
5	5	10	15	20	25
4	4	8	12	16	20
3	3	6	9	12	15
2	2	4	6	8	10
1	1	2	3	4	5
Impact	1	2	3	4	5

ID	Risk Description	Probability	Impact	Severity	Mitigation	Post Mitigation Probability
SE-PM-01	Uncertainties on GNSS precision due to lunar irregular gravitational field	3	5	15	Launch a low cost proof of concept mission as a payload of another lunar SpaceCraft	1
SE-PM-02	High mission complexity leading to overbudget and delays	3	3	9	Long developpment time, Implements backups hardware, low launch cost	2
SE-PM-03	Loss of a satellite	3	3	9	Implement sleeping backup satellite in the constellation	1
SE-PM-04	Loss of a ground station	3	4	12	Implement backups redundant hardware for critical system	2
SE-PM-05	Critical ardware malfunction	2	5	10	Implement backups redundant hardware for critical system	1
SE-PM-06	Unavalaibility of Starship	2	3	6	Design a backup configuration for another launcher	2
SE-PM-07	High operational cost	2	3	6	Insure projet's viability, contact potential customer	1
SE-PM-08	Non conformity with political reglementations	2	5	10	Implement active communication with concerned parties to prepare the legal background for the project	1
SE-PM-09	Security breach, cyberattack	2	5	10	Implement data encryption protocols, negotiate with all space agencies	1

Figure 6: Snapshot of the risk analysis tool with the more important risks, with the corresponding severity matrix

2.5 Budgets

Preliminary budgets have been defined for the SELENE system based on analysis and comparisons with existing missions.

- **Cost budgets :** Assuming a cost of approximately 10-15M$ per Starship launch, 10M$ per spacecraft, and 50-100M$ per kickstage/lander, the total mission cost is estimated to be around 600M$ for a fully deployed SELENE constellation. This estimation does not account for operational costs, which still need to be evaluated.[11]
- **Power budget :** The instantaneous power required to operate all equipment is estimated at 1924 [W], with an average power consumption of 678 [W]. This has led to the selection of a 1065 [W] solar panel for each satellite. Further details are provided in Section 3.5.
- **Mass budgets :** Each satellite is expected to weigh 600 [kg], while the lander has a dry mass of 11.25 [T] and a wet mass of 85.2 [T], resulting in a total mass of 90 [T] per Starship launch when accounting for the eight satellites. In the A6 configuration, only six spacecraft are transported, along with a 4.4 [T] lunar injection stage.

DOI: 10.13009/EUCASS2025-213

3. Results and subsystems design

3.1 Spacecraft configuration

The spacecraft features a trapezoidal form factor with dimensions 2 x 1.2 x 0.5 [m]. The orientable solar panel is mounted on the largest side and faces outward during the mission. The telecommunication equipment is located on the smaller, Moon-facing side, while the inter-satellite laser communication module is positioned on one of the lateral sides.

On the lander, the 8 satellites are arranged around a central pole and are radially deployed using suitable mechanisms once in lunar orbit. The central pole also hosts the lander's communication payload. The landing legs are designed to be fixed rather than deployable, as the Starship provides sufficient payload volume to accommodate them in this configuration.

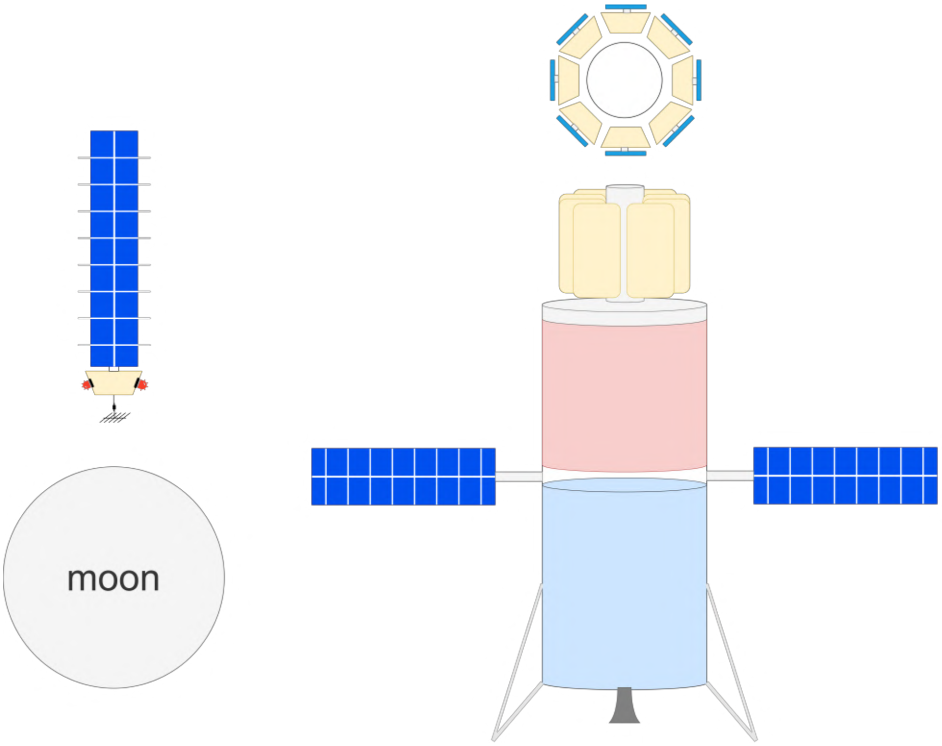


Figure 7: Lander and satellites configuration. Red represents the methane tank, while blue is the LOX tank. The antenna, laser diodes and moon are only schematic and are not scaled correctly

3.2 Propulsion and AOCS

Two main propulsion systems are necessary for the SELENE system:

- A kickstage/lander that provides all the necessary ΔV from low Earth orbit after separation from the launcher, through lunar injection, satellite deployment, and ultimately lunar landing. The design prioritizes operational simplicity by using the same propellant as the primary launcher (SpaceX Starship), which is methalox. This decision is based on an existing mission, NOVA-C by Intuitive Machines, that successfully performed payload fueling operations on the launch pad with the same provider, as well as a lunar landing with a methalox engine [10] [14]. Since the propellant is cryogenic, fueling of the lander must occur shortly before launch, as it cannot be loaded in advance. This operation will require a dedicated custom system developed in collaboration with the launch provider. The system must deliver sufficient thrust and ΔV for all phases, with the lunar landing being the most demanding in terms of thrust, while the lunar transfer consumes the most ΔV.
- A propulsion system for orbit keeping, reaction wheel desaturation, and de-orbiting at the end of life (EOL) of each individual spacecraft in the constellation.

DOI: 10.13009/EUCASS2025-213

3.2.1 Kickstage/lander propulsion system

For the kickstage/lander system, no specific engine has been chosen yet. However, an estimation of the kickstage size based on plausible engine parameters has been conducted. The estimated ΔV requirements for the transfer from LEO to the lunar surface are as follows: approximately 3.2 [km/s] for trans-lunar injection, 1 [km/s] for lunar orbit insertion, and 2 [km/s] for landing from the deployment orbit [13]. A conservative I_{sp} of 350 [s] was assumed for the methalox propulsion system, based on the performance of existing engines. The mass fraction of the lander was considered to be 0.125, accounting for the payload and landing hardware. Eight satellites, each weighing 600 [kg], are expected to be launched per kickstage, resulting in a total payload mass of 4.8 [t]. Based on these assumptions, the ΔV was calculated using the Tsiolkovsky rocket equation:

$$\Delta V = I_{\text{sp}} \cdot g_0 \cdot \ln \frac{m_0}{m_f},$$

where I_{sp} is the specific impulse of the engine, g_0 is the gravitational acceleration at Earth's surface, and m_0 and m_f are the initial and final masses of the spacecraft.

The reduction in mass due to the deployment of satellites in lunar orbit was taken into account, which decreased the propellant required for the final landing. Additionally, the volume of propellant needed to achieve the required ΔV was estimated using propellant densities of 1141 [kg/m^3] for LOX and 442 [kg/m^3] for methane, along with an oxidizer-to-fuel (O:F) ratio of 3.6. This calculation was essential to verify whether the payload could fit within the typical payload volume of the launcher.

For a sub-optimal mission using the Ariane 6 launcher, a similar procedure was followed. A translunar injection limit of 8 [t] was imposed [5], and the kickstage's role was restricted to lunar orbital insertion, requiring approximately 1 [km/s] of ΔV. In both configurations, the tank diameter was iteratively adjusted to achieve an optimal design. The final tank diameters were determined to be 3.5 [m] for the Starship configuration and 1.5 [m] for the Ariane 6 configuration.

The final dimensions of the propellant tank sections for the Kickstage/lander (Starship) and the Kickstage-only (Ariane 6) configurations are summarized below:

Variable	Calculated value	Unit
LOX mass	50050	kg
Methane mass	13900	kg
LOX volume	43.86	m^3
Methane volume	32.94	m^3
LOX tank length	4.56	m
Methane tank length	3.43	m
Payload section length	2.5	m
Dry mass	11250	kg
Payload mass	4800	kg
Total length	10.5	m
Total mass	90000	kg

Table 1: Starship configuration

Variable	Calculated value	Unit
LOX mass	2660	kg
Methane mass	730	kg
LOX volume	2.33	m^3
Methane volume	1.75	m^3
LOX tank length	1.32	m
Methane tank length	1	m
Payload section length	2.5	m
Dry mass	1000	kg
Payload mass	3600	kg
Total length	4.8	m
Total mass	8000	kg

Table 2: Ariane 6 configuration

3.2.2 Satellites propulsion and AOCS

The satellite propulsion system must be powerful enough to compensate for environmental perturbations, such as the Moon's irregular gravitational field and solar radiation. The ΔV required to maintain an orbit at $a = 2000$ [km] was estimated to be about 1-2 [m/s] per week, based on the Lunar Reconnaissance Orbiter, which requires approximately 2.8 [m/s] but operates in a much lower orbit, increasing station-keeping needs [4]. The decision was made to adopt an electric propulsion system due to its high ISP and low thrust requirements, specifically leveraging Hall Effect thruster technology for its efficient balance of power consumption, thrust, and operational efficiency. Given that the operational lifetime of individual satellites is set to 10 [years], this implies a total ΔV requirement of at least 1 [km/s] for station-keeping. Additional reserves must account for de-orbit maneuvers at EOL and AOCS desaturation. Using FreeFlyer software, it was determined that ΔV requirements are 350 [m/s] for de-orbiting and about 50 [m/s] for reaction wheel desaturation [9].

DOI: 10.13009/EUCASS2025-213

Furthermore, maintaining a ΔV of 2 [m/s] per week for a 600 [kg] spacecraft requires a minimum thrust, which can be calculated as follows:

$$F = \frac{m \cdot \Delta v}{t} \tag{1}$$

where F is the thrust, m is the spacecraft mass (600 [kg]), ΔV is the required velocity correction (2 [m/s]), and t is the burn duration (1 [week]). This calculation results in a minimum thrust of approximately 2 [mN].

With a 10-year lifetime and an additional 500 [m/s], the total ΔV requirement is 1.5 [km/s]. Based on these values, a suitable electric engine was identified: the Halo12 Spacecraft Thruster by ExoTerra Resource, LLC [8]. This thruster provides approximately 30 [mN] of thrust at 450 [W], with an adequate expected lifetime. To achieve a ΔV of 1.35 [km/s] with an ISP of 1300 [s], the spacecraft would require approximately 60 [kg] of xenon propellant. A corresponding propellant tank, the S-XTA-60 from MT Aerospace [12], with a volume of 60 [L] and a maximum rated pressure of 187 [bar], meets the requirements. It is important to note that these calculations were performed under the assumption of 2 [m/s] per week for the entire mission duration, representing an absolute worst-case scenario.

Concerning the AOCS, a suitable reaction wheel model was identified and used to estimate the mass and power draw of such a system. The selected item [3] is rated for a lifetime of 5 [years]. Consequently, a "cold redundancy" approach was adopted, requiring two reaction wheel systems for the nominal mission duration, plus an additional one for redundancy, resulting in a total mass of 18 [kg].

3.3 Launch segment

As previously mentioned, the main launcher selected is SpaceX Starship, which allows the launch of 8 satellites and one lander simultaneously. A suboptimal configuration, containing only 6 satellites, can be launched on Ariane 6. The primary drivers for launcher selection were the maximum payload volume and mass requirements specified by the launcher manufacturers. The Starship is advertised to have a payload capacity of 150 [T] in LEO [15], while Ariane 6 offers a capacity of 8 [T] [5] for a lunar injection trajectory.

A detailed analysis, presented in Table 1 and Table 3, confirms that the mass requirements are met for both configurations. Regarding the payload volume, two scaled schematic representations of each configuration were created to verify their fit within the available volumes provided by the manufacturers [16] [2].

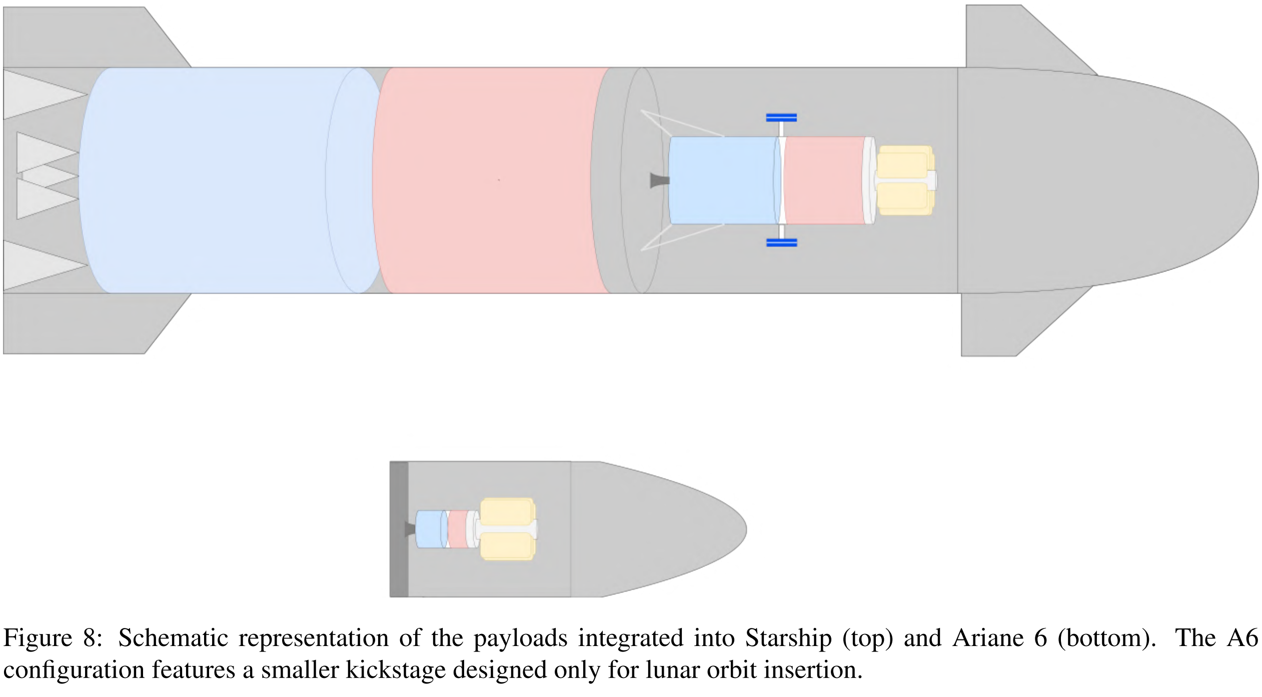


Figure 8: Schematic representation of the payloads integrated into Starship (top) and Ariane 6 (bottom). The A6 configuration features a smaller kickstage designed only for lunar orbit insertion.

3.4 Thermal management

An estimation of the steady-state temperature of the satellites was computed for different orientation scenarios and mission phases using the following formula, which represents the radiative thermal equilibrium of a satellite. In this equation, $A_\alpha \cdot J_{\text{Sun}}$ is the absorbed solar flux, $A_a \cdot J_a$ accounts for planetary albedo, $A_p \cdot J_p$ represents the planetary infrared flux, and P_{internal} corresponds to the satellite's internal heat dissipation. The input values and results are summarized in Table 3.

$$T^4 = \frac{(A_\alpha \cdot J_{\text{Sun}} + A_a \cdot J_a)}{A_\epsilon \cdot \sigma} \cdot \left(\frac{\alpha}{\epsilon}\right) + \frac{A_p \cdot J_p}{A_\epsilon \cdot \sigma} + \frac{P_{\text{internal}}}{\sigma \cdot A_\epsilon \cdot \epsilon} \tag{2}$$

Variable	Earth orbit	Earth-moon transfer	Moon shadow	Moon orbit #1	Moon orbit #2
Jsun [w/m^2]	1361	1361	1361	1361	1361
Ja [w/m^2]	16.33	0	0	16.33	16.33
Jp [w/m^2]	365.45	0	5.38	210	210
alpha/epsilon [-]	2.8	2.8	2.8	2.8	2.8
Epsilon [-]	0.05	0.05	0.05	0.05	0.05
Aepsilon [m^2]	19.2	19.2	19.2	19.2	19.2
Aalpha [m^2]	2.4	0.46	0	2.4	10.46
Aa [m^2]	2.4	0	2.4	2.4	2.4
Ap [m^2]	2.4	0	2.4	2.4	2.4
Pinternal [W]	300	300	300	300	300
Temperature [K]	347	290	273	346	454

Table 3: Parameters and steady-state temperature per configuration. Moon orbit #1 corresponds to a Sun-Moon-satellite alignment, while #2 forms a 90° Sun-satellite-Moon angle.

In most cases, the temperature is kept within the reasonable bounds of 0 to 75 [°C], as expected in Configuration 2, where the solar panel could induce a significant thermal transfer. However, these calculations are very preliminary, do not account for internal thermal flux, and assume the temperature is in steady state. The internal power dissipated in heat inside of the spacecraft is considered to be about 300 [W] for this preliminary calculation.

The chosen coating for the satellite is aluminized Kapton, selected by iteratively adjusting the calculations presented above until a suitable temperature range for all configurations was achieved.

3.5 EPS

The spacecraft will use a combination of solar panel and battery. The FreeFlyer simulation allowed the computation of the Eclipse time (52.1 [min]) and the Sunlight time (270.92 [min]) in the worst case scenario.

The result of the computation are as follow :

- **Solar array** :
  - **Technology** : Ga-As
  - **Power output** : 1065[W]
  - **Surface** : 5.4[m2]
  - **Mass** : 42.6[kg]
- **Batteries** :
  - **Technology** : Li-ion
  - **Number of cell** : 3
  - **Capacity per cell** : 730[Wh]
  - **Mass** : 11[kg]

Power Budget	P instant (W)	Ton/Toff	P average (W)
Propulsion	450	0.2	90
AOCS	120	0.1	12
Atomic Clock	140	1	140
CDH (OBC+RTU)	40	1	40
Thermal	0	0.5	0
Telecom : X-Band	560	0.125	62.5
Telecom : Laser	100	1	100
Telecom : UHF	30	1	30
Electronic/Loss	60	1	60
Result (with added 30% margin)	1924[W]	-	678.5[W]

Table 4: Power usage of the spacecraft subsystem

DOI: 10.13009/EUCASS2025-213

3.6 Telecom

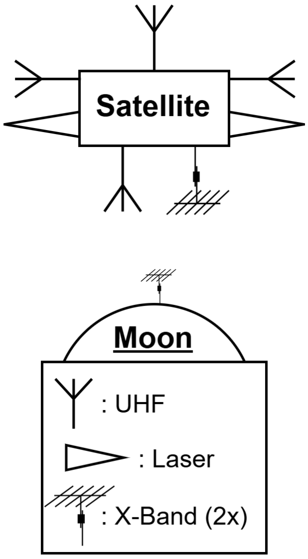


Figure 9: Schematic representation of the antennas position on the satellites

Telecom is an important subsystem, because it serves as a payload and as the C&T [a] system. It is split in three main components :

- **X-Band (2x)** : Two 1[GB/s] systems with antenna pointing to the Moon are working in parallel. It ensures operations with reduced data rate in case of failure. The antenna orientation to a moon GS allows to reduce their size compared to a direct link to earth.
- **UHF** : 4 antennas are spread around the satellite, to have an "almost" omnidirectional antenna. The data rate of 256[KB/s] serves for the C&T.
- **Laser** : Two laser pointing in the orbit axis allows communications between satellites on the same orbit. The technology is promising, but more development is needed.

A complete link margin analysis was done for the UHF and the X-Band. The result can be seen on the table 5.

---

[a] **C&T** : Command and Telemetry

	Data-rate	Antenna Gain	Bit Error Rate	Protocol	Link Margin
X-Band	2*1[GB/s]	9Bi	10^-5	TurboCode, R=0.5	4.77
UHF	256[KB/s]	5dBi	10^-7	Uncoded	6.7

Table 5: Table recapitulating link margin analysis

3.6.1 Command and data handling

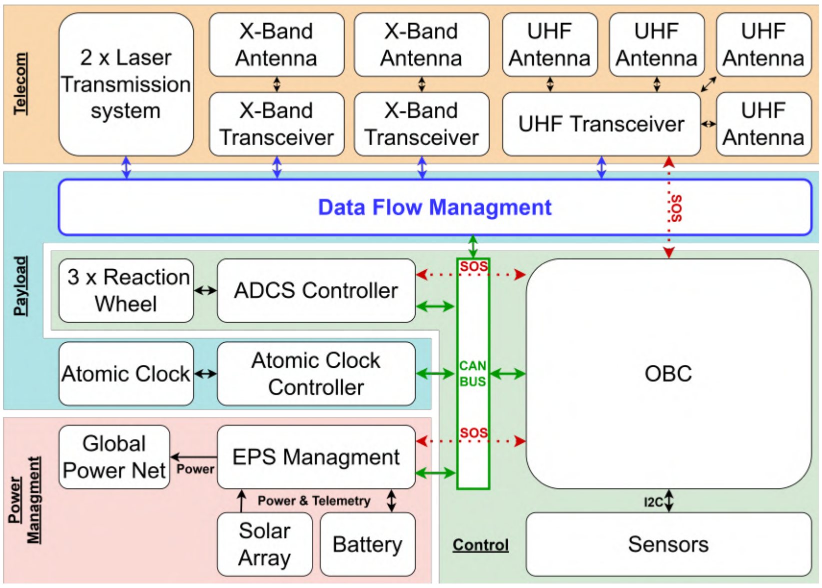


Figure 10: Schematic of the communications links between subsystems

DOI: 10.13009/EUCASS2025-213

The primary communication links between the spacecraft subsystems are illustrated in Figure 10. The architecture is designed with a clear separation between C&T and the high data rate requirements of the telecommunication payload.

The C&T data is transmitted via two Controller Area Network (CAN) buses. One serving as the main bus (in green on figure 10 and the other as a redundant backup (not displayed in figure 10). The. CAN is a serial, differential interface bus, where complementary signals are transmitted over two wires, and the logical state of the bus is determined by the voltage difference between the wires [1]. It employs Non-Return-to-Zero (NRZ)[2] encoding, enabling a compact message format with high robustness to external disturbances. The effective data rate of the CAN bus ranges from 100 to 200 [KB/s][6], which is well-suited for the low-bandwidth requirements of C&T communication.

The redundancy in the CAN bus enhances spacecraft reliability by ensuring an alternate path for C&T data in case of failures. For additional low-data-rate sensors, such as thermometers, the I2C protocol is employed due to its simplicity and low pin count.

The high data rate communication requirements for the telecommunication payload are managed by the Data Flow Management System, as shown in blue in Figure 10. This subsystem uses the SpaceFibre protocol, capable of achieving data rates exceeding 1 Gbit/s. The payload data is routed to a dedicated processor that directs data packets from specific transceivers to their appropriate destinations based on packet descriptors.

The processor operates in two distinct modes:

- **Priority Mode :** Optimized for low-latency Moon-to-Moon communications.
- **Normal Mode :** Designed for high-data-rate Moon-to-Earth communications.

When the spacecraft is not in its nominal mode (Mode 2), subsystem communication is handled by "SOS" links, which are illustrated in red in Figure 10. These links utilize direct RS-422 communication lines between the Onboard Computer (OBC) and critical subsystems such as UHF, ADCS, and POWER. RS-422 is a standard differential protocol using four wires (two for RX and two for TX), offering a simple and reliable method for ensuring communication during critical scenarios.

3.6.2 Ground segment

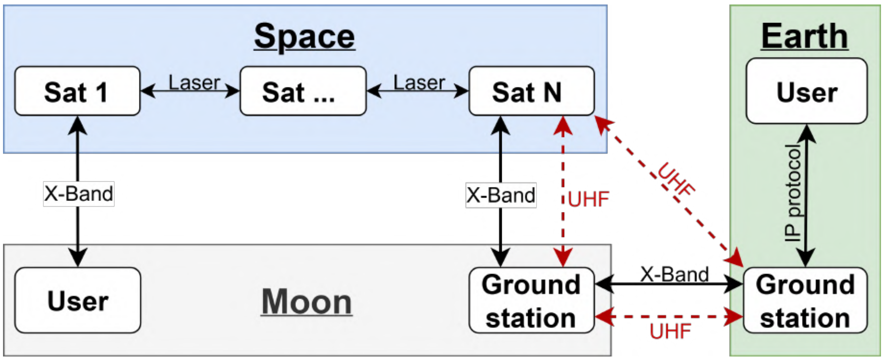


Figure 11: Communications links between two users

The SELENE network requires two types of Ground Stations (GS): one on Earth and another on the Moon, each serving distinct roles in the overall architecture.

The Earth-based control center serves as the interface between the SELENE network and the World Wide Web. As illustrated in Figure 11, it communicates with the Moon GS through X-band channels. To meet system performance requirements, the X-band link supports a total data rate of 12 GB/s, enabling efficient data transfer between the Earth and Moon.

On the Moon, the three kickstages, act as lunar ground stations. Following their landing, each kickstage deploys multiple parabolic antennas. A fixed X-band antenna system is used to establish a high-data-rate link to the Earth GS, providing a data rate of 4[GB/s]. Additionally, a secondary X-band beam is utilized to track and communicate with the satellites. This tracking enhances the overall system gain. This second system requires a data rate of 2 GB/s. The increase (2x) of bandwidth can be attributed to additional encoding needed for reliable communications[6].

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[1] Differential signaling improves noise immunity by reducing susceptibility to external disturbances.

[2] **NRZ**: Non-Return-to-Zero encoding minimizes signal transitions to achieve compact messages and increased resilience to interference [1].

DOI: 10.13009/EUCASS2025-213

Beyond communication, the lunar kickstages also function as reference points for the GNSS, significantly improving the precision of SELENE positioning system.

To enhance the robustness of telemetry and command operations, both lunar surface ground stations and lunar-orbiting satellites are equipped with omni-directional UHF antennas. These systems serve as a redundant communication pathway, ensuring continued connectivity in scenarios where the primary X-band system becomes compromised, such as during satellite tumbling or antenna misalignment.

4. Discussion

The results obtained confirm the overall feasibility of the SELENE GNSS mission, with the current design meeting the majority of the requirements for propulsion, thermal management, and power systems. The kickstage's methalox propulsion system demonstrates sufficient ΔV capabilities for both Starship and Ariane 6 configurations, though the Ariane 6 scenario imposes stricter constraints on payload and tank sizing and prohibits the use of ground stations, which would reduce the overall system performance, particularly in terms of GNSS precision and telecom bandwidth. For the satellites, the selected propulsion system meets the ΔV and operational lifetime requirements, but additional analysis could refine the margins for orbit keeping, de-orbit, and AOCS desaturation maneuvers, as the current values are only rough estimations. In all cases, embarking additional propellant is feasible.

Thermal simulations indicate that the satellite temperatures remain within acceptable bounds (0 to 75 [°C]), with aluminized Kapton providing an effective coating. However, the calculations assume steady-state conditions and do not account for transient scenarios or internal heat dissipation variability. At this stage of design, no active cooling solutions are envisioned, but if they are later found to be necessary, a rework of the EPS would be required to account for the additional electrical power demands. A key source of uncertainty is the solar panels, which present a large surface area exposed to radiation but have a small cross-section link to the main body of the spacecraft, limiting thermal transfer. Additionally, the selected coating (aluminized Kapton) cannot be applied to the active side of the solar panels.

The link margin computations for the UHF and the X-Band shows that the data rate objectives are achievable with enough margin to ensure communications. A detailed design will be necessary to ensure the low-latency.

Finally, the electrical power system appears robust for nominal operations, but further evaluation is needed to ensure redundancy under eclipse and peak load conditions. The current design is primarily based on a general average use-case scenario.

Only preliminary work has been conducted on the lander vehicle, and further detailed analysis is necessary. This is very notable and highlights the need of a significant amount of work missing to assess the feasibility of the mission.

5. Conclusion and Future Work

The SELENE GNSS mission demonstrates the feasibility of deploying a robust lunar infrastructure for high-precision navigation and continuous communication, addressing key challenges of future lunar exploration. The design satisfies critical requirements for propulsion, thermal management, and power systems, while maintaining scalability and cost-effectiveness. The current configuration shows that it is capable to achieve the desired >95% lunar coverage and the planned data rate, making it a valuable contribution to future lunar operations.

However, certain limitations must be addressed in subsequent phases. Firstly, some initial estimations about thermal management, link margin. ΔV and power requirement shall be improved by more advanced computation and simulation tool to gain confidence in the system's feasibility. Additionally, only very preliminary work has been performed on the lander vehicle.

Future work will focus on detailed design of the satellite and lander system. Beyond SELENE, this mission architecture could serve as a template for interplanetary GNSS and communication constellations, enabling sustainable exploration of other celestial bodies.

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