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MISSION ANALYSIS WITH EMPHASIS ON ORBITAL ANALYSIS FOR A LUNAR NANOSATELLITE MISSION

ANALÝZA MISE S DŮRAZEM NA ORBITÁLNÍ ANALÝZU PRO MISI LUNÁRNÍHO NANOSATELITU

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The aim of this thesis is to analyse and design orbital configurations for a nanosatellite operating in lunar orbit. The thesis includes a review of existing and planned lunar missions with a focus on their orbital configurations, based on which a suitable orbit is proposed for a selected scientific or technological objective of the nanosatellite mission. The work further focuses on the design and analysis of lunar orbits using analytical and numerical methods to assess their stability, accessibility, and feasibility of transfer from Earth, including the computation of the required Δv for transfer, orbit insertion, and maintenance. The thesis also includes an evaluation of the lunar orbital environment and a comparison of propulsion systems suitable for nanosatellite lunar missions. The outcome of the thesis is a preliminary mission concept and recommendations for further development.

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Abstract

This thesis presents a preliminary mission analysis for a lunar nanosatellite mission focused on the in-situ investigation of dust in the lunar exosphere, motivated by the need to better characterise the lunar dust environment for future robotic and human activities. The thesis defines the mission background, mission objectives, observation and operational requirements, orbit-selection logic, and a concept of operations based on stepwise descent through low lunar science orbits. It further addresses the main environmental conditions relevant to the mission, including eclipse, thermal, and radiation aspects. The work also includes an assessment of manoeuvre capability, station-keeping needs, and propulsion options for the proposed mission concept.

Keywords

Lunar CubeSat, Feasibility Study, Mission Analysis, Orbit Design, Lunar Exosphere, Lunar Dust, Station Keeping, Electric Propulsion.

Abstrakt

Táto diplomová práca predstavuje predbežnú analýzu lunárnej nanosatelitnej misie zameranej na in-situ skúmanie prachu v mesačnej exosfére. Motiváciou práce je potreba lepšie charakterizovať mesačné prachové prostredie vzhľadom na budúce robotické a ľudské aktivity na Mesiaci. Práca definuje kontext misie, jej ciele, pozorovacie a operačné požiadavky, logiku výberu obežnej dráhy a koncepciu operácií založenú na postupnom zostupe cez nízke lunárne vedecké dráhy. Ďalej sa venuje hlavným environmentálnym podmienkam relevantným pre misiu, vrátane zatmení, tepelných a radiačných aspektov. Súčasťou práce je aj posúdenie manévrovacej schopnosti, potreby udržiavania dráhy a možností pohonu pre navrhovaný koncept misie.

Klíčová slova

Lunárny CubeSat, Štúdia uskutočniteľnosti, Analýza misie, Návrh obežnej dráhy, Mesačná exosféra, Mesačný prach, Udržiavanie dráhy, Elektrický pohon.

Extended abstract

Táto diplomová práca sa zaoberá predbežnou analýzou lunárnej nanosatelitnej misie zameranej na in-situ skúmanie prachu v mesačnej exosfére ich výškovú závislosť a možné zmeny spôsobené prirodzenými alebo umelými udalosťami. Práca je motivovaná potrebou lepšie charakterizovať mesačné prachové prostredie, ktoré je dôležité pre budúce robotické a ľudské aktivity na Mesiaci. V úvodnej časti je predstavený širší kontext lunárnych misií, význam nanosatelitov a dôležitosť skúmania mesačného prachu, exosféry a regolitu.

Hlavná časť práce definuje ciele navrhovanej misie, pozorovacie a operačné požiadavky, kritériá výberu obežnej dráhy a základnú koncepciu operácií. Navrhovaný scenár vychádza z postupného zostupu cez nízke lunárne vedecké dráhy, čo umožňuje uvažovať o meraniach v rôznych výškových režimoch nad mesačným povrchom. Práca sa ďalej venuje hlavným environmentálnym podmienkam, najmä zatmením, tepelnému prostrediu a radiačným vplyvom.

Súčasťou práce je aj posúdenie manévrovacej schopnosti, potreby udržiavania dráhy a predbežného delta-v rozpočtu pre zvolený koncept misie. Záver práce sa venuje porovnaniu možností pohonu a výberu vhodného pohonného systému pre navrhované orbitálne operácie.

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Brno, May 24, 2026

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INTRODUCTION

Lunar exploration is entering a new phase in which the Moon is becoming not only a scientific target, but also an operational environment for future robotic missions, surface infrastructure, resource utilisation, and human activities. This increases the need to better understand environmental conditions that may affect spacecraft systems and mission operations. One of these factors is lunar dust, which is relevant for both science and future lunar systems, as it may influence optical surfaces, thermal-control elements, mechanisms, solar panels, and surface operations.

Therefore, this thesis focuses on a preliminary mission analysis for a lunar nanosatellite mission aimed at the in-situ investigation of dust in the lunar exosphere. CubeSat-class spacecraft are attractive for focused lunar investigations due to their smaller size, lower mission cost, and increasing subsystem maturity. However, lunar nanosatellite missions remain technically demanding because of strict limitations in mass, volume, power, communication capability, thermal control, radiation tolerance, and propulsion performance.

The main objective of this thesis is to define and evaluate a preliminary mission concept for a lunar dust-focused nanosatellite. The work includes the definition of mission objectives, observation and operational requirements, orbit-selection criteria, and a concept of operations based on stepwise descent through low lunar science orbits. The thesis also addresses eclipse conditions, thermal and radiation environment, manoeuvre capability, station-keeping needs, and propulsion selection.

These objectives are addressed through literature review, requirement definition, analytical calculations, and numerical orbital analysis. Previous lunar missions and orbit configurations are reviewed to identify relevant heritage and lessons learned. The proposed orbit and operational concept are assessed using orbital calculations, ANSYS System tool kit (STK)-based propagation, eclipse evaluation, preliminary thermal and radiation assessment, station-keeping analysis, and propulsion trade-off.

The main contribution of this thesis is the consolidation of the main mission-driving aspects into one preliminary feasibility-oriented study. The work connects the scientific motivation with orbit design, environmental constraints, manoeuvre strategy, station-keeping assessment, and propulsion selection. The aim is to establish a system-level basis for later detailed design and refinement of the mission architecture.

The thesis is structured into six chapters. Chapter 1 introduces the mission background and the relevance of lunar dust and nanosatellites. Chapter 2 surveys representative lunar missions and orbit configurations. Chapter 3 defines the mission objectives, requirements, orbit-selection logic, and concept of operations. Chapter 4 addresses the environmental analysis. Chapter 5 presents the manoeuvre analysis, station-keeping strategy, and delta-v budget. Chapter 6 discusses propulsion options and the final propulsion selection.

1. MISSION BACKGROUND AND CONTEXT

Lunar exploration has re-emerged as a major focus of international space activity over the past two decades, driven by scientific, technological, and strategic objectives [1]. Programs such as NASA's Artemis and European Space Agency's (ESA) Moonlight initiative highlight the transition of the Moon from a purely scientific target to a platform for sustained exploration and cislunar infrastructure development [1], [2], [3], [4].

A key contributor to this renewed era is China National Space Administration through the Chang'e program, initiated in 2004. The program demonstrates a systematic progression from orbital missions to soft landings, far side exploration, and sample return, including the first far side sample return mission (Chang'e-6) [5], [6], [7].

Scientifically, the Moon provides critical insights into early Solar System evolution, impact history, regolith processes, and volatile transport [6]. Missions such as KAGUYA and Chandrayaan-1 significantly advanced knowledge of lunar topography, mineralogy, and the presence of water-related species [8], [9].

Simultaneously, the Moon is increasingly treated as an operational environment. Artemis aims to enable sustained human presence and deep-space exploration, while Moonlight focuses on lunar communication and navigation services [2], [3], [10]. Future missions such as Chang'e-7 and Chang'e-8 further emphasize in-situ resource utilization and long-term activity, particularly in the lunar south polar region [7], [11].

The current lunar exploration landscape is also characterized by a growing number of actors and mission types. According to ESA, more than 400 lunar missions are expected in the coming two decades, underscoring the need for robust mission architectures, orbital strategies, and supporting infrastructure [12].

1.1 Role of nanosatellites in lunar missions

Nanosatellites are relevant to lunar exploration because they allow narrowly defined mission objectives to be pursued with a smaller spacecraft, lower accommodation demand, and a lower mission entry threshold than is typical for conventional lunar spacecraft. Their main value is therefore not to replace larger lunar missions, but to enable pathfinder missions, focused technology demonstrations, precursor measurements, and other limited-purpose tasks that would be difficult to justify as standalone large missions [13], [14].

As shown in Figure 1.1, the long-term increase in nanosatellite launches reflects the expansion of industrial capability, subsystem availability, and operational experience across the small-spacecraft domain [15]. Although this trend is not specific to lunar exploration, it forms the technological and programmatic background that has made lunar nanosatellite missions increasingly realistic. The use of standardised spacecraft formats, miniaturised avionics, and more mature deployment solutions has created conditions in

which small spacecraft can now be considered for missions beyond Earth orbit, including cislunar and lunar applications [13], [14], [15].

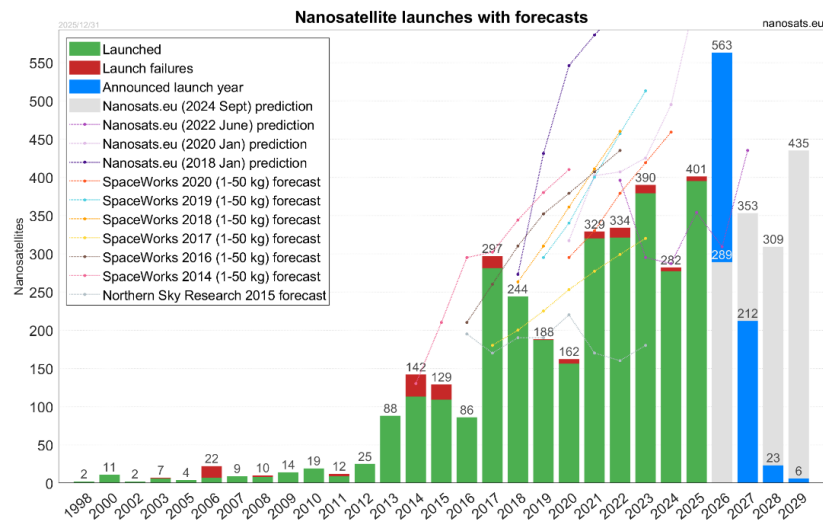


Figure 1.1 Nanosatellite launch forecast [15].

A major advantage in the lunar context is access to flight opportunities through secondary-payload accommodation. As shown in Figure 1.2 and Figure 1.3, the Orion Stage Adapter of the SLS architecture can use otherwise unused internal launch volume for CubeSat deployment. NASA states that this configuration can accommodate CubeSats in combinations of 6U and 12U sizes, with the actual manifested number depending on mission constraints [16]. This is important because it allows nanosatellites to reach cislunar or deep-space trajectories without requiring a dedicated lunar launch, which directly improves cost-efficiency and mission accessibility [14], [17].



Figure 1.2 The Artemis I Orion stage adapter with mounting brackets for secondary[17].

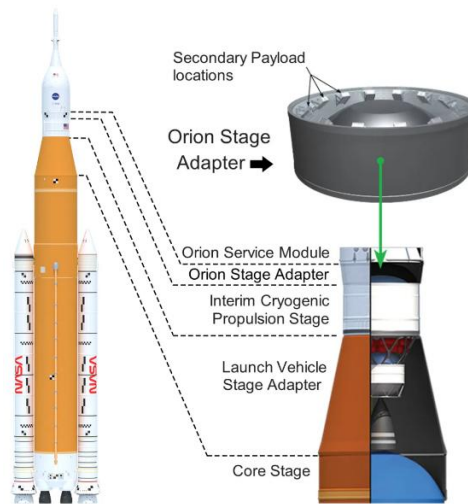


Figure 1.3 The SLS rocket Secondary payload location[16].

This approach has already been demonstrated in flight. Artemis I deployed 10 6U CubeSats: Lunar IceCube, LunaH-Map, OMOTENASHI, LunIR, CuSP, NEA Scout, EQUULEUS, BioSentinel, ArgoMoon, and Team Miles [18]. Artemis II extended this model with four 12U CubeSats from international partners: TACHELES, K-Rad Cube, SHMS, and ATENEA [19]. In both cases, the missions show that nanosatellites can be integrated into large lunar exploration campaigns as secondary elements with their own scientific or technological objectives, rather than being treated only as isolated experimental payloads.

As shown in Figure 1.4, the SLS secondary-payload concept uses standardised 6U and 12U dispensers, which reduces interface complexity and simplifies integration into a shared launch architecture. In practical terms, this means that the spacecraft can be designed within a known geometric and mechanical envelope, which lowers integration effort and supports faster accommodation within rideshare-based mission architectures [16], [17].

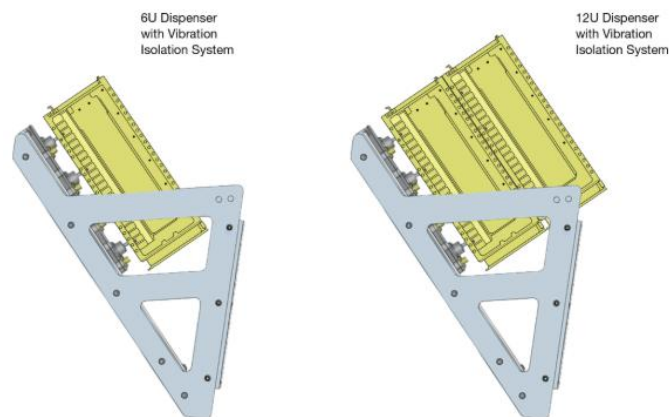


Figure 1.4 The Secondary payload dispenser configurations[16].

A second important advantage is that nanosatellites are well suited to incremental mission development. Instead of waiting for a large and highly integrated mission, selected technologies or operational concepts can be tested earlier on a smaller platform [14]. This is illustrated by CAPSTONE, which validated operation in a Near Rectilinear Halo Orbit relevant to the Gateway architecture and showed that a small spacecraft can contribute directly to cislunar mission preparation. In this sense, nanosatellites reduce not only spacecraft scale, but also development risk at the level of the overall exploration architecture [14], [20].

Their relevance is likely to increase further as lunar exploration becomes more infrastructure based. ESA's Moonlight programme is being developed to provide lunar communications and navigation services, which can reduce the standalone operational burden on individual spacecraft and improve the feasibility of smaller missions. For nanosatellites, this is particularly important because their usefulness in lunar missions depends not only on miniaturisation itself, but also on the availability of a broader operational framework into which they can be inserted [2], [3]. In parallel, NASA's Commercial Lunar Payload Services (CLPS) model supports more frequent and flexible access to the lunar domain for science and technology payloads[21], [22].

1.1.1 Main limitations of nanosatellites in lunar missions

The principal limitation of nanosatellites in lunar missions is the combination of small spacecraft resource budgets with high transfer, orbit, communications, and environmental demands [13], [14], [23]. In practice, feasibility is determined by whether the required mission profile can be closed within strict limits on delta-v, electrical power, thermal rejection, antenna performance, onboard autonomy, and radiation tolerance [23].

Propulsion

A first major limitation is propulsion capability. Lunar missions commonly require non-negligible onboard delta-v for injection correction, phasing, deterministic and statistical trajectory correction manoeuvres, lunar orbit acquisition support, orbit trimming, and station-keeping, depending on the selected architecture [19], [23]. In nanosatellite platforms, propulsion performance is constrained by low propellant volume, limited total impulse, low thrust level, and restricted electrical power for actuation and feed systems. These constraints directly reduce the set of accessible trajectories and target orbits, especially when the spacecraft is deployed as a secondary payload and must accept non-optimal initial conditions after release [23].

Power and thermal control

A second major limitation is electrical power and thermal control. Although solar irradiance near the Moon is of the same order as near Earth, nanosatellites have only limited surface area for power generation and heat rejection. This becomes critical when the mission includes long eclipse intervals, continuous attitude control, deep-space communications, or electric propulsion. Small spacecraft therefore operate with narrow

energy margins and limited peak-power capability, while compact internal packaging makes thermal gradients and local hot spots more difficult to control. In lunar missions, power sizing and thermal design are consequently mission-driving parameters rather than secondary subsystem checks [23].

Communications and navigation

A third major limitation is communications and navigation performance at lunar distance. Compared with low Earth orbit missions, lunar links require higher link margin because of the much larger range and the associated free-space path loss. For nanosatellites, this is difficult to compensate because antenna aperture, transmitter power, and pointing stability are all constrained by spacecraft size. The direct consequence is lower achievable data rate, tighter attitude constraints during communication windows, and stronger sensitivity to communications geometry and ground-station access [23].

A partial future mitigation of this limitation is the development of ESA's Moonlight programme as a dedicated lunar communications and navigation system [2], [3], [12], while Lunar Pathfinder is intended as an early relay element supporting communications services in lunar orbit [24]. If such infrastructure becomes operational and widely accessible, part of the communications and navigation burden can be shifted from the individual spacecraft to the external system. For nanosatellites, this could reduce onboard requirements for antenna performance, operational complexity, and mission-level communications support, although mission feasibility would then become partly dependent on the availability and service architecture of that external infrastructure.

Guidance, navigation, and control

A related limitation is guidance, navigation, and control in cislunar space. Nanosatellites beyond low Earth orbit cannot rely on the same operational assumptions commonly used in Earth-orbiting CubeSat missions. Attitude determination and control must be achieved with higher pointing accuracy, and orbit knowledge must be maintained without routine dependence on low-cost Earth-orbit navigation solution. This increases the need for accurate star trackers, inertial sensing, autonomous onboard estimation, and robust fault handling, all within a very constrained avionics envelope. The limitation is therefore not only sensor performance itself, but the difficulty of implementing precise and reliable GNC functions in a highly constrained spacecraft [23]

Environmental Robustness

Another important limitation is environmental robustness. Lunar and cislunar missions expose spacecraft to radiation, thermal cycling, and mission durations that exceed the conditions for which many CubeSat-class components were originally qualified. At nanosatellite scale, shielding thickness, fault-tolerant design, and redundancy are inherently limited. As a result, the spacecraft is more sensitive to single-event effects, cumulative radiation damage, temperature-driven degradation, and single-

string failures than a larger planetary spacecraft with greater structural, thermal, and avionics margin [23].

Mission architecture dependence

Finally, nanosatellites are strongly constrained by mission architecture dependence. Their achievable performance is often set not only by internal subsystem capability, but also by the launch opportunity, separation conditions, available tracking support, and external communications infrastructure. This is particularly important for rideshare-based lunar missions, where the spacecraft does not control the primary transfer design and must adapt to the trajectory, schedule, and operational constraints imposed by the host mission [13], [14].

1.2 Potential of the lunar missions

The potential of lunar missions is broader than the investigation of the Moon as an isolated planetary body. The Moon is simultaneously a scientific archive of early Solar System evolution, a source of potentially useful material resources, a nearby environment for engineering validation under deep-space conditions, and a support domain for later cislunar and interplanetary exploration architectures [25], [26], [27], [28]. Its proximity allows repeated mission access and progressive capability build-up, while its environment still imposes key non-LEO constraints such as vacuum, thermal extremes, dust, long-distance communications, and increased autonomy requirements [1], [12], [29].

1.2.1 Scientific research potential

The potential of lunar missions extends well beyond the study of the Moon as an isolated planetary body. The Moon is a preserved record of early Solar System evolution, a reference object for comparative planetology, and a uniquely accessible environment for scientific investigations that are difficult to perform elsewhere. Because the lunar surface has not been erased by atmosphere, hydrology, or active plate tectonics, it preserves the bombardment history of the inner Solar System, constrains the chronology of impact basins and volcanic units, and provides information on the geological evolution of a differentiated silicate body through its crust, mantle, core, volcanic products, and basin structures [25], [26], [27].

A particularly important scientific theme is the study of polar volatiles. Permanently shadowed regions near the lunar poles are cold enough to retain water ice and other volatile-bearing materials over geologic timescales, which makes them relevant not only for identifying the presence of volatiles, but also for studying their origin, migration, sequestration, composition, age, and vertical and lateral distribution. These deposits are scientifically important because they preserve information on volatile delivery and processing in the inner Solar System and because they connect the regolith, exosphere, and solar-wind environment into a coupled surface-boundary system [25], [26]. In this

domain, CubeSat-class spacecraft are especially useful where the measurement return depends on regional coverage, repeated revisit, and local orbital access, for example in orbital prospecting above polar terrains, repeat mapping of illumination or thermal conditions, and support to later landed investigations [13], [30].

A second major scientific theme is the surface-boundary environment, including the exosphere, dust, and plasma interaction. The lunar exosphere is the most accessible surface-boundary exosphere in the Solar System, and the Moon therefore provides a natural laboratory for studying sputtering, meteoritic vaporization, gas-surface interaction, exospheric transport, electrostatic charging, and dust motion above an airless body. Open questions remain regarding the density, composition, variability, and source processes of the exosphere, as well as the size, charge, mobility, and deposition of transported dust grains [25], [26]. In this area, CubeSat-class spacecraft are relevant because they can support distributed plasma and field measurements, multi-point observations, and repeated low-altitude orbital passes over selected regions [13], [30].

The Moon also has important scientific value as a platform for astronomy and astrophysics. The radio-shielded far side is widely regarded as one of the best accessible locations in the inner Solar System for low-frequency radio astronomy, because wavelengths blocked or distorted by Earth's ionosphere can be observed there [26]. In addition, the lunar surface may preserve long-term records of astrophysical processes, including solar-wind implantation, solar energetic particle products, galactic cosmic-ray signatures, and possibly material associated with the local interstellar environment [26],[31].

1.2.2 Material and resource potential

The Moon also has clear material and resource potential, although this potential must be treated as a question of technically accessible feedstocks rather than assumed industrial maturity. The two most important resource classes are polar volatiles and regolith-derived materials [6]. Polar deposits are relevant because they may provide water and, after processing, oxygen and hydrogen for life support and propellant production [6], while the regolith is relevant because it contains oxygen bound in oxides as well as silicate, metallic, and ceramic-forming constituents that can support local shielding, construction, and material processing [6], [32], [33].

From a materials perspective, the lunar regolith is important not only as raw feedstock, but also as an engineering medium whose grain-size distribution, mineralogy, cohesion, abrasiveness, beneficiation response, and thermal-processing behaviour directly affect excavation, transport, oxygen extraction, sintering, and additive manufacturing [31], [32], [33]. Current review work on lunar in-situ resource utilization and lunar-based manufacturing therefore treats the Moon as a place where resource characterization, extraction, and manufacturing must be developed together, not as a simple case of mining a well-characterized deposit [31], [32], [33].

The construction potential of the Moon follows from the same logic. Recent reviews classify lunar regolith construction strategies into distinct groups, including reaction-based solidification, sintering or melting, bonding-based solidification, and confinement-based approaches, and evaluate them in terms of strength, energy demand, curing requirements, environmental compatibility, and achievable in-situ material ratio [32], [33]. This makes lunar missions relevant to material science not only because of what may be extracted, but because the Moon provides a realistic environment in which construction-grade material processing, structural fabrication, and local manufacturing from regolith-derived feedstock can be tested under vacuum, dust, thermal cycling, and reduced gravity [32], [33].

1.2.3 Engineering and infrastructure potential

From an engineering point of view, the Moon is valuable because it combines deep-space operational constraints with planetary-surface constraints in a comparatively accessible environment[1], [29]. Lunar missions therefore allow the validation of systems and operations directly relevant to later exploration, including precision landing, hazard detection, surface mobility, excavation, drilling, cryogenic operations, dust mitigation, long-duration power and thermal control, robotic autonomy, and coordinated orbital-surface operations. Subsystem performance must be demonstrated not only in isolation, but under coupled mission conditions involving power, thermal control, communications, navigation, operational timelines, and surface interaction [1], [29].

This is particularly important for technologies that cannot be validated adequately on earth. A landing system must be tested not only for descent guidance and control, but also for plume-surface interaction and navigation under low-angle illumination. A rover must be tested not only for mobility, but also for dust tolerance, thermal survivability, communications geometry, and autonomous operation under long communication delays [1], [29]. An In-situ resource utilization (ISRU) system must be tested not only for extraction efficiency, but also for excavation, feedstock variability, contamination control, thermal management, storage, and integration with downstream systems[6].

The Moon is therefore an effective engineering test environment because it exposes the coupling between hardware, environment, and operations in a way that laboratory demonstrations cannot fully reproduce [32], [33].

1.2.4 Operational and support potential for deep space missions

The Moon also has clear value as a support environment for deeper-space exploration. In international exploration roadmaps, the lunar vicinity and lunar surface are treated as the next major step beyond low Earth orbit and as a pathway toward later robotic and human missions to Mars and other destinations. In this framework, lunar missions are used to validate habitation systems, logistics, autonomy, communications, radiation operations, human-robotic interaction, and long-duration mission techniques under deep-

space conditions while remaining more accessible than Mars. The Moon therefore functions not only as a destination, but also as an intermediate operational domain in which exploration architectures can be assembled, exercised, and progressively extended [1], [29].

This support role is not only conceptual, but also architectural. The ISECG science white paper explicitly treats the lunar vicinity as an enabling environment for science and exploration infrastructure, while the Global Exploration Roadmap identifies the Moon as the near-term focus for sustained exploration activity and as a key step in building capabilities needed for later Mars missions. This includes logistics, orbital-surface coordination, autonomous operations, communications networks, and the validation of exploration concepts under realistic deep-space conditions [1], [29]. CubeSat-class spacecraft are especially relevant as low-mass support elements: deployable relays, distributed sensors, local reconnaissance orbiters, or auxiliary platforms associated with larger campaigns.

A concrete example of this support-oriented approach is the move toward dedicated lunar communications and navigation infrastructure. ESA's Moonlight programme is being developed as a lunar telecommunications and navigation system intended to support autonomous landings, surface mobility, and high-speed Earth–Moon communications while reducing operational burden and user cost [2], [3]. Such infrastructure is directly relevant to future deep-space exploration because it creates a cislunar operational layer on which more complex missions can build. For CubeSat-class spacecraft, this is particularly important because external communications and navigation infrastructure can reduce the burden placed on very small spacecraft and make them more effective as mission support elements within broader lunar campaigns [1], [29], [2], [3].

1.3 Relevance of lunar dust and exosphere studies

1.3.1 Scientific and exploration relevance

Lunar dust and the exosphere form the surface-boundary environment of the Moon, linking the regolith, the surrounding plasma, and the tenuous neutral environment above the surface. Their behaviour is governed by processes such as impact vaporization, sputtering, volatile release and transport, dust mobilization, and plasma surface interaction. Because the Moon is the most accessible airless body, these processes are also used as a reference for interpreting similar environments on Mercury, asteroids, and other regolith-covered bodies directly exposed to space [34], [35], [36].

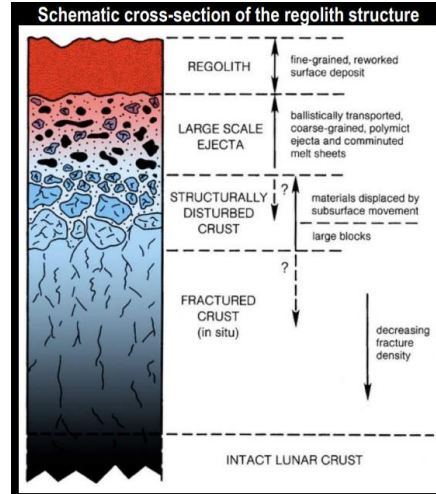


Figure 1.5 Schematic cross section of the regolith structure[37].

The regolith also preserves long-term records of external forcing. In the absence of atmosphere, liquid water, and active plate tectonics, the lunar surface can retain signatures of solar-wind implantation, energetic-particle exposure, and other astrophysical inputs over very long timescales. Dust and exosphere studies are therefore connected not only with present-day surface processes, but also with the Moon as a recorder of Solar System and space-environment history [34], [38].

The same environment defines part of the boundary conditions for future lunar exploration. Dust transport, exospheric variability, and near-surface environmental processes affect the conditions under which landers, instruments, habitats, surface systems, and long-duration operations will function on the Moon. For this reason, characterization of the dust-exosphere system is relevant both to scientific investigation and to the planning of sustained lunar activity [25], [26], [31], [34].

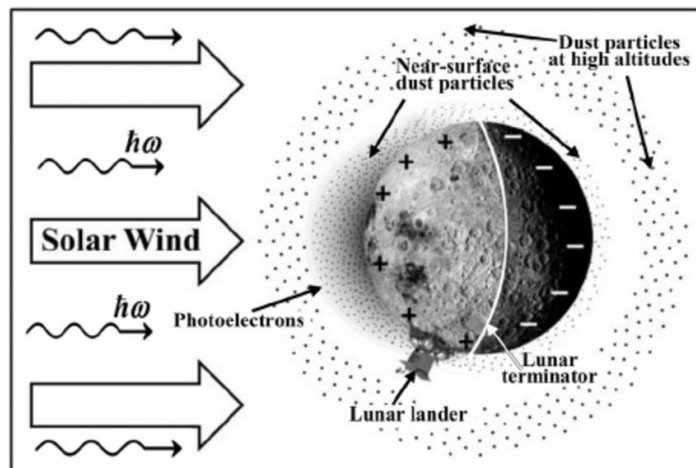


Figure 1.6 The main elements characterizing the dusty plasma system over the Moon[39].

1.3.2 Measurement approaches

Lunar dust and the exosphere are measured using a combination of direct particle detection, optical observations, and neutral or ion measurements, because no single method captures the full range of particle sizes, altitudes, spatial scales, and source processes relevant to the lunar surface-boundary environment. In practice, the available methods constrain different parts of the environment and are therefore complementary rather than interchangeable [34], [35].

A first group of methods is based on in-situ dust detection. These instruments register direct particle impacts on the spacecraft and are used to infer dust presence, impact rate, approximate particle properties, and variability along the instrument trajectory [35]. This is the most direct way to confirm that a dust population exists in a given region of space, because the result does not depend on optical scattering assumptions. The main limitation is geometric: the measurement is local to the spacecraft path, so it samples only the dust environment encountered along the orbit or trajectory rather than providing a complete global distribution [34], [35].

A second group consists of optical and remote observations. In this case, dust is constrained indirectly through scattered light, extinction, or brightness structure near the surface and limb [34], [40], [41]. The classical example is the observation of horizon glow near the lunar terminator (Figure 1.7), interpreted as forward-scattered sunlight from dust-related near-surface structure [38].

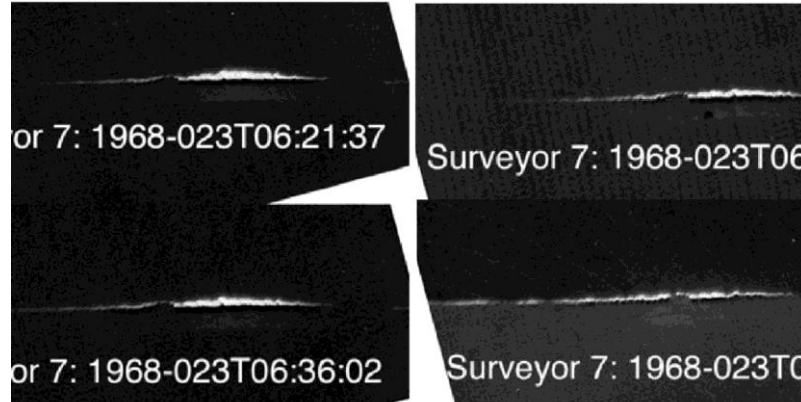


Figure 1.7 Lunar horizon-glow observations from the Surveyor missions[42].

Later orbital ultraviolet–visible observations were also used to search for or constrain low-altitude dust populations [36], [41]. The advantage of these approaches is that they can sample broader geometry than an in-situ detector and may provide information over a larger field of view. Their limitation is that the interpretation depends on assumed particle size, scattering behaviour, vertical distribution, and sometimes the temporal stability of the observed feature [34], [41].

A third group consists of exosphere measurements, most commonly through neutral mass spectrometry and related ion or plasma observations. These methods are used to

determine the composition and variability of the tenuous lunar atmosphere and to connect surface release processes, micrometeoroid effects, sputtering, and plasma interaction to the near-surface environment [34], [43], [44]. Although these measurements do not directly count dust grains in the same way as a dust detector, they are essential for describing the coupled dust–exosphere system, because some of the same processes that mobilize or release dust also contribute to neutral and ion populations above the surface [34].

In current lunar exploration, these approaches are often combined within the same mission architecture. A representative example is LADEE, which carried a direct dust detector, an ultraviolet–visible spectrometer, and a neutral mass spectrometer (Figure 1.8) [43], [45], [46]. This combination made it possible to compare local dust detections, optical constraints, and exospheric composition within one orbital dataset. Even so, the resulting picture remained incomplete, because each instrument was sensitive to a different regime of the environment [34], [45], [47].



Figure 1.8 LADEE science and technology payload instruments[42].

The present measurement framework is therefore best understood as a multi-method constraint problem. Direct dust detections provide the strongest evidence for specific populations encountered in orbit, optical observations constrain scattering-related structure and upper limits, and neutral or ion measurements describe the surrounding exosphere and its variability [34], [43], [45], [47]. Taken together, these methods define the current observational basis of lunar dust and exosphere studies, but they still leave significant uncertainty in the near-surface concentration field, especially where altitude, particle size, local time, and source mechanism interact [34], [47].

1.3.3 Factors influencing lunar dust concentration and current distribution uncertainty

Lunar dust concentration is not expected to be constant in space or time. The available evidence indicates that the dust environment is controlled by several coupled processes,

electrostatic charging and lofting, illumination-dependent transport, micrometeoroid impact ejecta, and local disturbances produced by landings or other surface activity. At higher altitude, direct measurements support the existence of a persistent but tenuous ejecta cloud, while at lower altitude and near the surface the environment is constrained mainly by upper limits, optical observations, and transport models rather than by one definitive concentration profile [34], [40], [41], [47]. The present uncertainty is therefore not only a lack of data volume, but also a consequence of the fact that different observations constrain different particle sizes, altitude ranges, and physical regimes rather than one single global concentration field [34], [39], [44].

1.3.3.1 Electrostatic levitation and surface charging

Electrostatic charging modifies dust behaviour after grains are released from the surface. Under solar ultraviolet irradiation and ambient plasma exposure, both the regolith and individual grains acquire charge, and the resulting near-surface electric fields can influence particle detachment, lofting, retention, and lateral transport. This effect is strongest for the finest grains, for which charge-to-mass ratio is high, and it depends on local sheath structure, solar-wind conditions, and surface topography. Electrostatic transport is therefore treated as a candidate mechanism that can redistribute dust even without direct mechanical disturbance [34], [35], [36], [38].

1.3.3.2 Illumination and local-time dependence

Illumination directly affects dust mobility because it controls photoemission and therefore the charging state of the surface and the near-surface sheath. The terminator is particularly relevant, since it separates illuminated and shadowed charging regimes and can generate strong local electric-field gradients. This is one reason why terminator regions have long been associated with possible dust lofting and redistribution. Early Surveyor observations of horizon glow after local sunset are consistent with illumination-dependent dust-related structure near sunlight–shadow boundaries, and later models likewise treat local time as a controlling parameter rather than a secondary correction [35], [36], [38], [40].

1.3.3.3 Micrometeoroid impacts

Micrometeoroid bombardment is the most robust natural source of lunar dust because it continuously injects regolith material into the near-surface environment. In the absence of an atmosphere, incoming particles strike the surface at hypervelocity, generate ejecta, and release vaporized material that contributes both to the dust cloud and to the exosphere [34], [41], [48]. LADEE/LDEX measurements showed that the Moon is surrounded by a permanent asymmetric ejecta cloud and that this cloud becomes denser during meteoroid streams such as the Geminids [41]. Micrometeoroid forcing therefore sets a background source term for lunar dust and also introduces measurable temporal variability tied to the external meteoroid flux [34], [41], [48].

1.3.3.4 Surface disturbances from landings and plume effects

Surface disturbances produced by landings and plume–surface interaction represent a different class of dust source, because they inject particles through a short-duration but locally extreme mechanical forcing event.

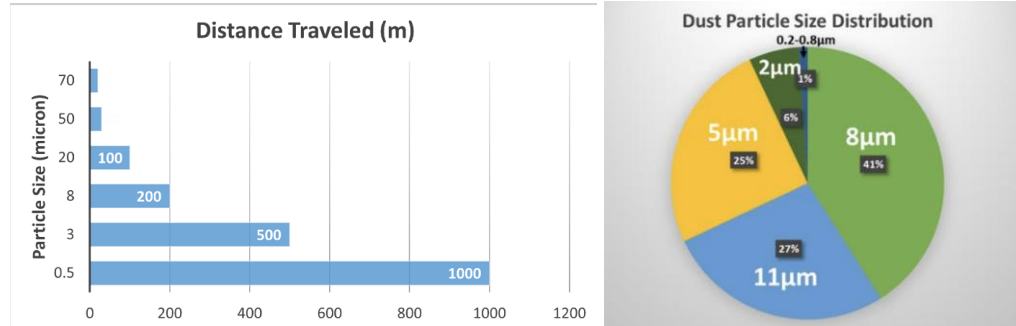


Figure 1.9 LEFT: Estimated transport distance of lunar dust particles during lander operations, RIGHT: Lunar dust particle-size distribution[49].

Rocket exhaust impinging on the regolith excavates, accelerates, and redistributes particles, creating a dense local dust cloud and high-velocity ejecta [46], [47], [48]. Within 50m of landing site, particle velocity are estimated to be 300 – 2000m/s. Apollo-based analyses and more recent review work show that plume effects can reduce visibility during landing, erode the surface, and generate ejecta with velocities high enough to damage nearby hardware or persist beyond the immediate landing zone [46], [47], [48]. For future lunar activity, plume-driven dust transport is therefore best treated as an event-driven disturbance superimposed on the natural background environment [49].

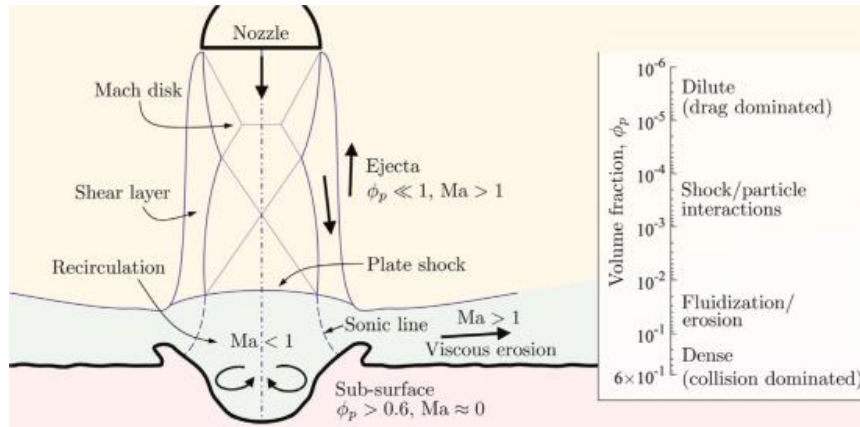


Figure 1.10 Plume–surface interaction during lunar landing or engine firing near the surface[54].

1.3.3.5 Expected variation with altitude, local time, and region

The available observations indicate that dust concentration varies with altitude, local time, and region rather than following a single uniform profile [34], [40], [41], [47]. At higher altitude, the strongest direct in-situ evidence comes from LDEX, which sampled

the dust cloud over roughly 3 to 250 km altitude and identified a persistent population of small ejecta particles with number densities on the order of 0.4 to $4 \times 10^{-3} \text{ m}^{-3}$ [41]. These measurements show that a tenuous dust exosphere exists well above the surface and that the observed cloud is asymmetric, with enhanced density near the morning side [41].

Closer to the surface, the environment is much less well constrained [34], [47]. LADEE/UVS observations in the approximate altitude range of 1 to 10 km produced upper limits rather than a definitive detection of a dense low-altitude dust layer [44]. For an assumed differential size distribution proportional to s^{-3} and a scale height of 1 km, the derived upper limit at the surface was 142 m^{-3} [44]. This indicates that the low-altitude environment cannot be inferred directly by extrapolating the higher-altitude ejecta cloud [41], [47].

Regional and local-time dependence are expected for the same physical reasons. Terminator zones experience strong illumination transitions and therefore changing charging conditions, while meteoroid-stream geometry produces hemisphere- and time-dependent enhancement of impact ejecta [35], [40], [41]. Surveyor horizon-glow observations after sunset support the presence of illumination-dependent dust-related structure near the terminator, and the LADEE ejecta cloud shows a clear local-time asymmetry [40], [47]. Taken together, these results imply that contamination severity and degradation risk may differ significantly for otherwise similar hardware operating at different altitudes, different local times, or in different regional contexts [35], [40], [41]. For measurement-oriented missions, this directly defines the key observables: concentration should ideally be resolved as a function of altitude, local time, and region, with at least partial sensitivity to particle size or process type where possible [34], [47].

1.3.4 Importance for lunar activities

Lunar dust is not only a scientific subject but also an operational constraint for future lunar activity. Once regular landings, repeated EVA, rover traffic, cargo transfer, and long-duration infrastructure are introduced, dust is no longer only a background environmental parameter; it becomes part of the operational system itself through transport, re-mobilization, accumulation, and repeated contamination of interfaces and habitable volumes [55], [56], [57], [58].

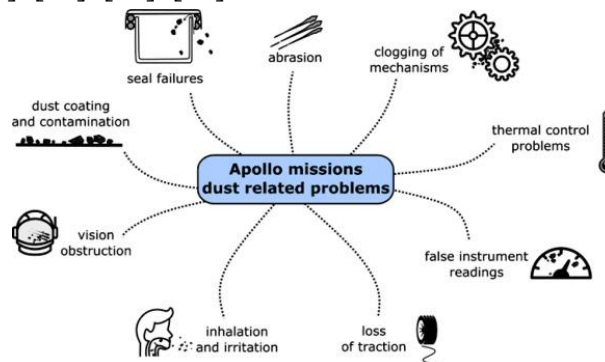


Figure 1.11 Dust-related problems reported during Apollo lunar surface operations [51].

1.3.4.1 Surface operations and infrastructure

For surface operations, lunar dust acts as a persistent contamination and wear source at the interfaces through which lunar activity is performed. During landing, EVA, rover motion, cargo handling, tool use, and repeated opening or closing of interfaces, dust can be mobilized, transported, deposited, and re-mobilized again [55], [56], [57], [58]. The operational issue is therefore not limited to one component or one event. Instead, dust becomes part of the local operating environment around landing zones, work areas, access points, and repeatedly used hardware, where it can accumulate over time and progressively increase maintenance and performance risk.

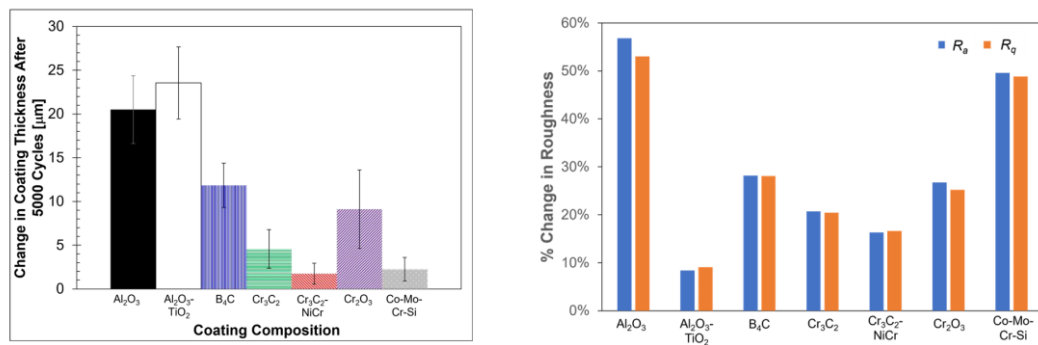


Figure 1.12 Left: Change in protective coating thickness after abrasion testing, Right: Change in surface roughness of protective coatings after abrasion testing [49].

This affects a broad range of surface systems. Repeated dust exposure can degrade airlocks, suitports, hatches, rover interfaces, tools, vents, filters, and other external or semi-exposed hardware. The same applies to power and support systems: dust deposition on exposed photovoltaic surfaces can reduce available power, while dust entering moving interfaces can increase friction and wear in joints, seals, hinges, bearings, and other mechanisms [50], [52], [53]. Dust accumulation on optical surfaces, thermal-control elements, and exposed coatings can further reduce performance margin and increase uncertainty in long-duration operation [58], [59]. Increase in dust deposition on exposed photovoltaic surfaces, reduces the incident solar flux reaching the cells and lowers available power (Figure 1.13), while the deposited layer can also modify thermal behaviour of the array surface [58], [59]. Since surface systems often depend on stable local power generation, dust-related degradation of solar panels directly affects operational margin, battery charging, payload duty cycle and communications availability.

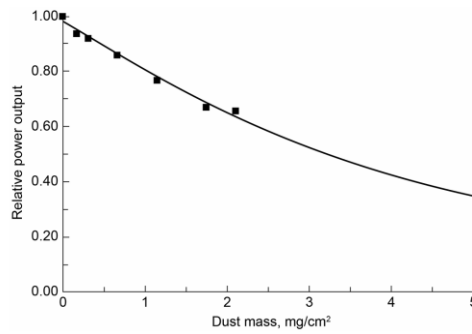


Figure 1.13 Reduction of photovoltaic power output due to dust[59].

In this sense, dust is not only a contamination issue, but also a driver of reduced operational efficiency and progressive hardware degradation across the surface architecture.

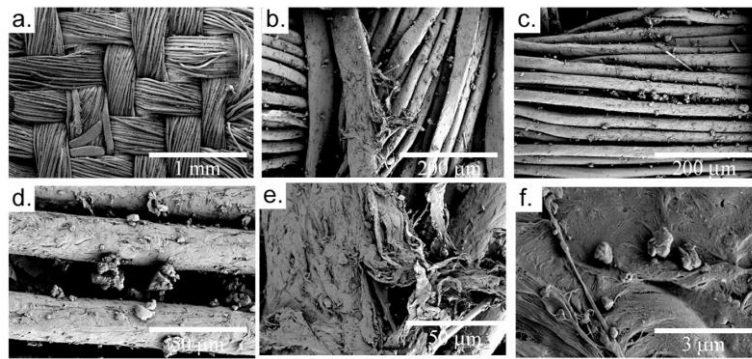


Figure 1.14 Apollo spacesuit contamination microscopic fabric damage caused by lunar dust[60].

The severity of the problem increases with activity density. Repeated landings, plume-disturbed landing areas, frequent rover traverses, cargo unloading, and routine field work all introduce additional disturbance on top of the natural background dust environment [50], [52]. For future lunar infrastructure, this means that performance will depend not only on component-level robustness, but also on landing-zone management, traffic organization, interface protection, cleaning strategy, and the spacing of sensitive assets from disturbed areas [55], [56]. Surface systems on the Moon therefore have to be treated as operating within an evolving dust environment, not a static one.

The main consequence is that dust control becomes part of operational architecture rather than a secondary cleanliness measure. Site layout, access concepts, shielding of sensitive interfaces, allowable contamination limits, and maintenance planning all depend on how dust is expected to be mobilized and redistributed during nominal activity. This is particularly relevant for long-duration missions, where repeated exposure can transform a manageable short-term nuisance into a cumulative infrastructure and reliability problem [55], [56], [57], [58].

1.3.4.2 Human presence and habitable systems

For human lunar missions, dust is both a contamination-control problem and a habitability problem [55], [56], [58], [61]. Surface dust is transported on suits, tools, samples, and external hardware into pressurized systems, where it can settle on work surfaces, seals, filters, optics, connectors, and other internal interfaces[50], [51]. Once it enters the habitat, the problem is no longer limited to external contamination: dust becomes part of the internal operational environment and can persist through repeated deposition, cleaning, and re-suspension cycles [50], [51], [53].

Apollo experience already showed that dust ingress affected cabin cleanliness and crew comfort [50]. Dust brought into the cabin was associated with eye and throat irritation and short-lived respiratory discomfort, while later reviews concluded that the main health concern is linked to the finer, respirable fraction and that the consequences of repeated or long-duration exposure remain insufficiently constrained for future sustained missions. In that sense, dust control in habitable systems is connected directly to both contamination management and crew health protection [50], [53].



Figure 1.15 Dust contamination transported into the lunar module after EVA[61].

The same issue propagates through the habitat as a system. Dust loading can increase the burden on air handling, filtration, cleaning, inspection, and maintenance, while contamination of suit-handling areas, airlock interfaces, and internal equipment can reduce operational efficiency over time [56], [58], [62]. This is why habitat dust control is treated not as a secondary housekeeping issue, but as part of the design of ingress pathways, suit management, internal airflow, filtration architecture, and cleaning procedures [51], [56].

Lunar dust is also a potential toxicant for humans. The finer fraction is respirable, and the hazard is linked to small particle size, angular morphology, surface reactivity, and the presence of reactive mineral phases, which can promote irritation, inflammation, and oxidative stress after inhalation or tissue contact [50], [55], [57]. Current reviews identify the pulmonary pathway as the main concern and note that repeated or long-duration exposure remains insufficiently constrained for future sustained missions [50], [57].

Apollo experience further showed that dust carried into the cabin caused eye, throat, and respiratory irritation, linking the issue directly to habitat operations and crew exposure control [55]. Useful for the Apollo heritage observations, including eye, throat, and respiratory irritation after cabin dust ingress.

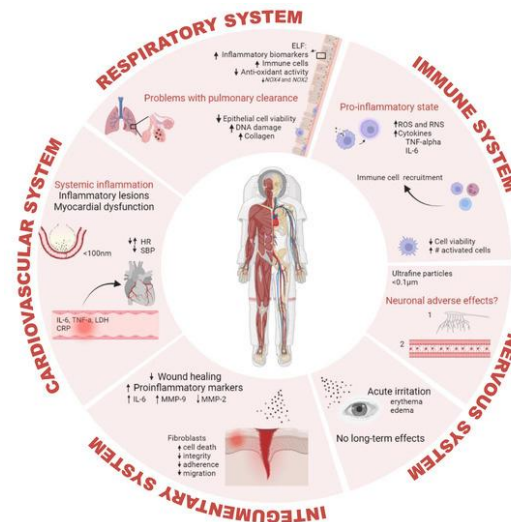


Figure 1.16 Potential human-health effects of lunar dust exposure[65].

For long-duration lunar presence, habitable systems must be designed around repeated contamination events rather than around one-time exposure. Airlock and suitport concepts, dust-removal steps before ingress, internal particulate control, and allowable dust-loading limits all become part of operational design. Human presence on the Moon therefore depends not only on keeping dust out of hardware, but also on keeping dust under control inside the living and working environment [56], [62].

1.3.4.3 Orbital platforms and cislunar logistics

For orbital platforms and cislunar logistics, lunar dust is less relevant as a continuous natural background load than on the surface, but it remains operationally relevant through the transfer of contamination across the surface–orbit architecture. Future lunar stations, logistics platforms, transfer vehicles, and docking nodes may repeatedly receive crews, cargo, samples, tools, and surface-exposed hardware returning from the Moon, so dust can be introduced indirectly through transport and handling rather than through direct exposure to regolith at the platform location itself [56], [57], [58], [66]. In this sense, orbital infrastructure can become part of the same contamination chain that begins at the surface and continues through ascent vehicles, airlocks, docking interfaces, cargo transfer, and internal handling operations [52], [53], [59].

This is relevant first at the level of interfaces and logistics hardware. If dust is carried on suit outer layers, sample containers, external payloads, or reusable surface systems, then contamination can accumulate at docking mechanisms, hatches, transfer bags,

stowage zones, filters, and repeatedly handled contact surfaces [56], [57], [58], [66]. The problem is therefore not limited to one contaminated item, but to repeated transfer of particulate matter through the logistics system. In architectures with regular surface sorties and cargo return, dust control becomes part of operational design for docking, cargo handling, quarantine or cleaning steps, and interface protection [52], [53], [59].

A second issue is that contamination introduced through logistics can affect exposed support hardware on orbital platforms as well. Dust deposited on photovoltaic surfaces can reduce available power, while contamination of optical or thermal-control surfaces can degrade the performance of sensors, cameras, and radiators [59], [67].

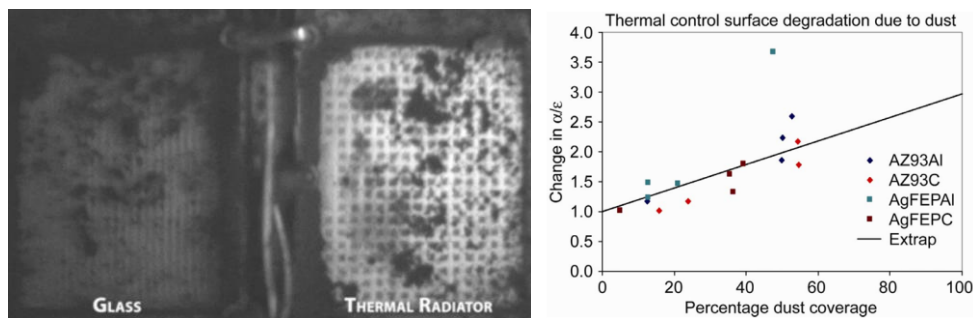


Figure 1.17 Left: Dust-covered glass and thermal radiator surfaces during lunar dust mitigation testing, Right: Thermal-control surface degradation as a function of dust coverage [67], [68].

For thermal hardware in particular, even partial dust coverage can modify absorptivity and emissivity and therefore reduce the predictability of thermal control (Figure 1.17) [67]. This matters because orbital platforms and transfer vehicles rely on stable power generation and thermal balance for long-duration operations, docking support, cargo preservation, and crewed habitation periods. Although the contamination source is indirect, the system consequence can still propagate into reduced power margin, increased thermal uncertainty, and higher maintenance demand [36], [58], [59].

The same logic applies to reusable orbital–surface transportation chains. If ascent and descent vehicles, cargo carriers, and crew systems are reused over multiple cycles, then dust can accumulate not only on the surface segment but across the whole architecture [56], [57], [66].

This increases the relevance of contamination-control procedures between surface return and orbital operations, especially for hardware that must maintain sealing quality, optical cleanliness, thermal performance, or repeated docking reliability. In that sense, orbital platforms do not only receive dust-contaminated elements; they may also become redistribution points within a larger lunar logistics network [56], [57], [66].

2. SURVEY OF LUNAR MISSIONS AND ORBIT CONFIGURATIONS

2.1 Main Lunar Programs

2.1.1 Artemis, CLPS, and Gateway

The NASA-led Artemis programme represents the main United States framework for renewed lunar exploration. The programme started operationally with Artemis I, an uncrewed flight test of the Space Launch System and Orion spacecraft, launched on 16 November 2022 and recovered on 11 December 2022 after a 25.5-day mission around the Moon [69]. The first crewed Artemis flight, Artemis II, launched on 1 April 2026 and completed a nearly ten-day lunar flyby mission with splashdown on 10 April 2026 [70]. According to the current NASA Artemis architecture, Artemis III is planned as a low-Earth-orbit demonstration of Orion docking with commercial lunar lander systems, while Artemis IV is targeted as the first Artemis lunar landing mission of the updated architecture.



Figure 2.1 Overview of the updated Artemis campaign from Artemis I to Artemis V[71], [72].

In parallel with crewed exploration, NASA uses the Commercial Lunar Payload Services programme, abbreviated CLPS, to deliver scientific and technology payloads to the lunar surface using commercial landers. CLPS is structured as a commercial service model: NASA purchases end-to-end delivery services, including payload integration, launch from Earth, mission operations, and landing on the Moon [21]. This approach is important because it increases the number of lunar access opportunities and allows instruments to be tested directly on the lunar surface before more complex crewed operations. Early CLPS missions already demonstrated both the potential and the risk of this approach: Astrobotic's Peregrine lander launched on 8 January 2024 but did not land, while Intuitive Machines' Nova-C lander launched on 15 February 2024 and landed near Malapert A on 22 February 2024 [21], [73]. Firefly Aerospace's Blue Ghost Mission 1 later delivered NASA payloads to Mare Crisium in March 2025 [74].

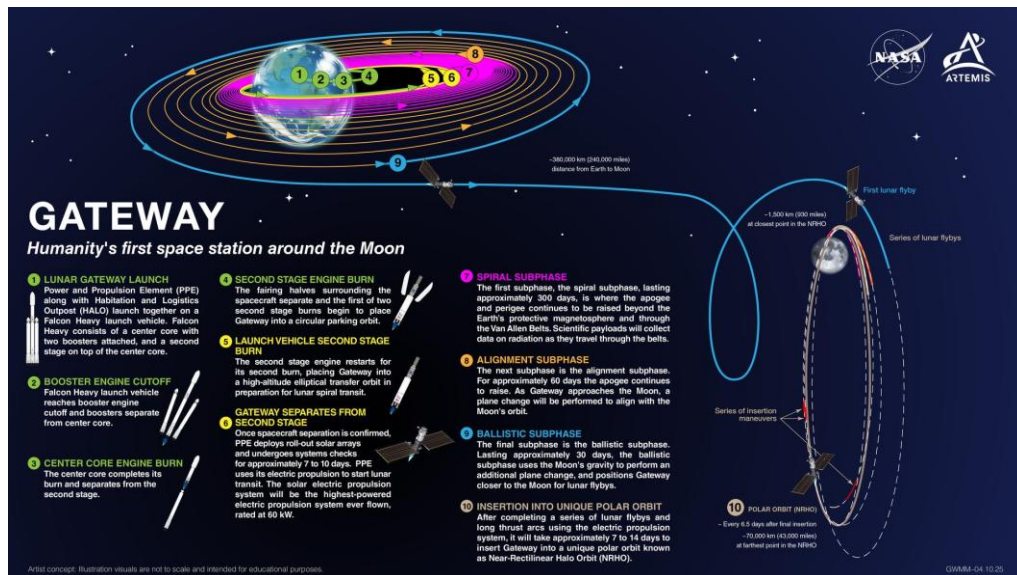


Figure 2.2 Lunar Gateway mission Concept of Operations[75].

The third important element is Gateway (Figure 2.3), a planned lunar-orbiting station intended to support Artemis surface missions, science in lunar orbit, and future deep-space exploration. Gateway is planned to operate in a near-rectilinear halo orbit, or NRHO, which is a highly elliptical polar orbit in the Earth–Moon system (Figure 2.2). NASA describes this orbit as having a period of approximately 6.5 days, with the station passing as close as about 1,500 km from the lunar surface and as far as about 70,000 km from the Moon [76], [77]. The first Gateway elements, the Power and Propulsion Element and HALO, are planned to launch together on a Falcon Heavy before Artemis IV and then transfer to lunar NRHO using solar-electric propulsion [76], [77].

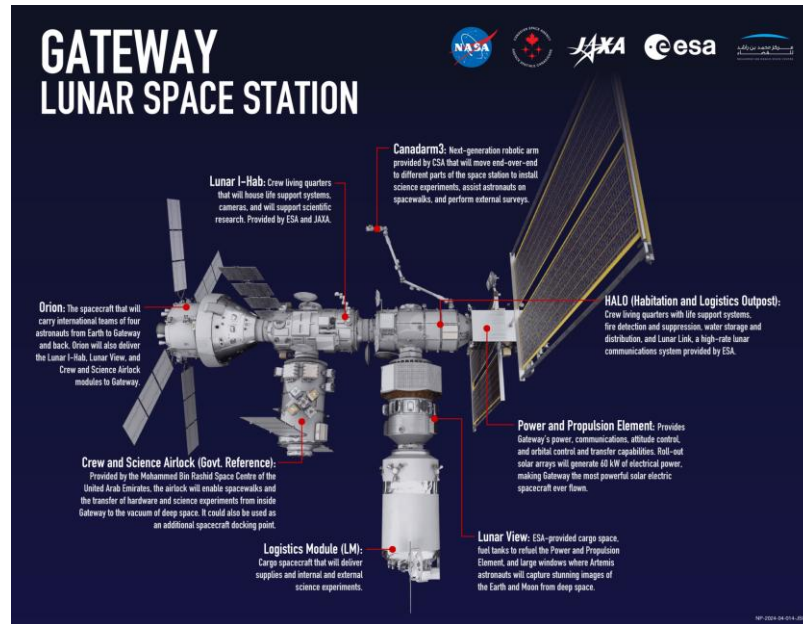


Figure 2.3 Configuration of the Gateway lunar space station[75].

2.1.2 Chinese Lunar Exploration Programme and ILRS

The Chinese Lunar Exploration Programme (Figure 2.4), also known as the Chang'e programme, is China's long-term robotic lunar exploration framework. It was formally approved in January 2004 and developed in a step-by-step sequence: first lunar orbiters, then lunar landing and rover missions, then sample-return missions, and finally south-pole and lunar research-station precursor missions [7].

Flown missions 2004–2020						Missions in development 2020–2030			
CE-1	CE-2	CE-3	CE-5T	CE-4	CE-5	CE-6	CE-7	CE-8	
2007	2010	2013	2014	2018	2019	?	?	?	Robotic and manned exploration
Orbiter	Orbiter	Lander/rover	Orbiter/Earth returner	Relay/lander rover	Sampling return	Sampling return	Lander/station	Lander/station	
200-km-high orbit	100-/15-km high orbit	Mare Imbrium	Earth-Moon orbit	Von Kármán crater	Oceanus Procellarum	South Pole	South Pole	?	
Global survey	High-resolution global survey	Lunar surface survey	High-speed Earth return	Far-side surface survey	Lunar sample	Lunar polar samples	Lunar scientific research	Lunar scientific research	

Figure 2.4 Development roadmap of China's Chang'e lunar exploration programme[7].

The first phase of the programme focused on lunar orbiting missions. Chang'e-1, launched in 2007, and Chang'e-2, launched in 2010, were used for lunar mapping, technology demonstration, and preparation for later landing missions. The second phase moved from orbit to surface operations with Chang'e-3, which landed on the near side of the Moon in 2013, and Chang'e-4, which achieved the first soft landing on the lunar far side in 2019. Because direct radio communication with Earth is not possible from the lunar far side, Chang'e-4 required the Queqiao relay satellite, which was placed in a halo orbit around the Earth–Moon L2 region [78].

The third phase introduced sample-return capability. Chang’e-5 returned approximately 1,731 g of lunar samples to Earth in 2020, making China the third country to return samples from the Moon [79]. This was followed by Chang’e-6, launched in May 2024, which landed in the South Pole–Aitken Basin on the far side of the Moon and returned the first lunar far-side samples to Earth [5]. Chang’e-5 and Chang’e-6 used complex multi-element mission profiles, including lunar orbit operations, surface sampling, ascent from the Moon, lunar orbit rendezvous, and Earth return.

The current phase of the programme is increasingly connected to the International Lunar Research Station, or ILRS, concept. This is planned as a long-term lunar research infrastructure with surface and orbital elements, with a basic model intended for the lunar south-pole region by around 2035 [80]. Chang’e-7 is planned for the lunar south pole to investigate the environment and search for water ice, while Chang’e-8 is planned as a later technology-verification mission connected to resource utilization and ILRS development [11], [81].

2.1.3 European lunar communication and navigation infrastructure (Moonlight)

European lunar exploration activities are increasingly focused not only on individual scientific missions, but also on communication and navigation infrastructure that could support future lunar users. The key programme in this area is ESA’s Moonlight Lunar Communications and Navigation Services, or Moonlight LCNS, which was officially launched by ESA on 15 October 2024. The programme is intended to create a lunar satellite constellation providing telecommunications and navigation services for missions on the lunar surface and in lunar orbit [2], [3], [12].

The motivation for Moonlight is the expected growth of lunar activity over the coming decades. ESA states that more than 400 lunar missions by space agencies and private companies are planned over the next two decades, increasing the need for reliable communication, data transfer, positioning, navigation, and timing services around the Moon [2], [3], [12]. Instead of each mission relying only on its own direct-to-Earth communication system, Moonlight aims to provide shared infrastructure that can reduce mission complexity and improve operational reliability.

The planned Moonlight architecture consists of five satellites, including one communication satellite and four navigation satellites, supported by three dedicated ground stations on Earth. The constellation is intended to prioritize coverage of the lunar south pole, which is important for future exploration because of its favourable illumination regions and permanently shadowed craters that may contain water ice [2], [3], [12].

A near-term precursor to this infrastructure is Lunar Pathfinder, a communications relay satellite developed by Surrey Satellite Technology Ltd. in partnership with ESA and with investment from the UK Space Agency. Lunar Pathfinder is planned as an early service and technology demonstrator for lunar relay communication. It is intended to

support lunar orbiters, landers, and rovers by relaying user data through the spacecraft and then back to Earth in X-band [24]. The mission is listed as planned for launch not earlier than November 2026, together with Firefly Aerospace's Blue Ghost vehicle, and is currently described as being in assembly, integration, and testing [24], [63].

Lunar Pathfinder is planned to operate in an Elliptical Lunar Frozen Orbit, or ELFO, for an operational lifetime of about eight years [24]. This orbit is selected to provide useful communication coverage for lunar users while limiting orbit-maintenance requirements. The spacecraft is designed to support simultaneous communication channels with lunar assets in S-band and UHF, with relay to Earth through X-band [24]. It also hosts experimental payloads, including an ESA GNSS receiver intended to test the use of weak Earth-based GNSS signals at lunar distances, a NASA laser retroreflector, and an ESA radiation monitor [24].

2.2 Representative historical and current lunar missions

The following section surveys selected lunar missions that are relevant to the orbital and architectural design of a future nanosatellite mission. The aim is not to provide a complete chronological history of lunar exploration, but to identify missions that demonstrate important design choices, such as low lunar orbit operations, polar mapping, exosphere and dust measurements, low-energy transfers, electric propulsion, relay-supported operations, CubeSat-class lunar access, and future infrastructure-oriented concepts.

Each mission is discussed with emphasis on its mission objective, spacecraft class, transfer or access strategy, operational orbit, concept of operations, and lessons learned for nanosatellite mission design.

2.2.1 Lunar Reconnaissance Orbiter

NASA's Lunar Reconnaissance Orbiter (LRO) is a robotic lunar orbiter developed by NASA Goddard Space Flight Center to map the Moon and characterize the lunar environment for future robotic and human exploration. It was launched together with LCROSS on an Atlas V launch vehicle on 18 June 2009 and entered lunar orbit on 23 June 2009 after a four-and-a-half-day transfer from Earth. After lunar orbit insertion, several engine burns refined the orbit, the instruments were commissioned, and LRO began its primary one-year mapping mission on 15 September 2009 from a low lunar science orbit about 31 miles, or approximately 50 km, above the Moon [82], [83], [84], [85].

Table 2.1 Lunar Reconnaissance Orbiter mission overview

Parameter	Description
Agency / operator	NASA / Goddard Space Flight Center
Spacecraft class	Large lunar orbiter
Launch date	18-Jun-09
Lunar orbit insertion	23-Jun-09
Mission status	Active mission; longest-lived lunar orbiting mission
Launch vehicle	Atlas V 401
Spacecraft mass	Approximately 1,850 kg
Primary propulsion	Chemical propulsion for lunar orbit insertion and orbit maintenance
Main objective	Map the lunar surface from polar orbit, identify potential landing sites and resources, investigate the radiation environment, and demonstrate technologies for future robotic and human missions
Operational orbit	Initially approximately circular 50 km polar mapping orbit; later changed to an elliptical polar orbit of about 20x165 km with perilune near the lunar south pole

LRO was designed to collect detailed information about the lunar surface, polar regions, possible resources, radiation environment, and potential landing sites. Its mission objectives included identifying safe landing areas, locating possible resources, characterizing the radiation environment, and testing new technologies. The spacecraft instruments measured radiation, surface and subsurface temperatures, far-ultraviolet illumination, hydrogen distribution, topography, high-resolution surface imagery, and radar properties of the lunar surface [82], [85].

The mission used a direct transfer trajectory to the Moon followed by a sequence of lunar orbit insertion and orbit-lowering manoeuvres. During commissioning, LRO was planned to operate first in a quasi-frozen orbit of about 30x216 km, before entering the final low-altitude mapping orbit. The main mapping orbit was a near-polar, approximately 50 km mean-altitude orbit, selected to provide high-resolution global observations of the lunar surface. This low polar orbit allowed LRO to repeatedly pass over both lunar polar regions and gradually build up global coverage as the Moon rotated beneath the spacecraft[82], [83], [84], [85].

The concept of operations (Figure 2.5) was based on a nadir-pointing, three-axis-stabilized spacecraft in low lunar orbit. During the primary mission, the instruments observed the lunar surface and environment from the low polar mapping orbit, while the spacecraft returned a large volume of data to Earth. After the initial low-orbit mapping phase, LRO continued into extended science operations. NASA notes that LRO spent its first three years in low polar orbit and later transitioned to a more stable elliptical orbit that passes low over the lunar south pole [82], [83], [84], [85].

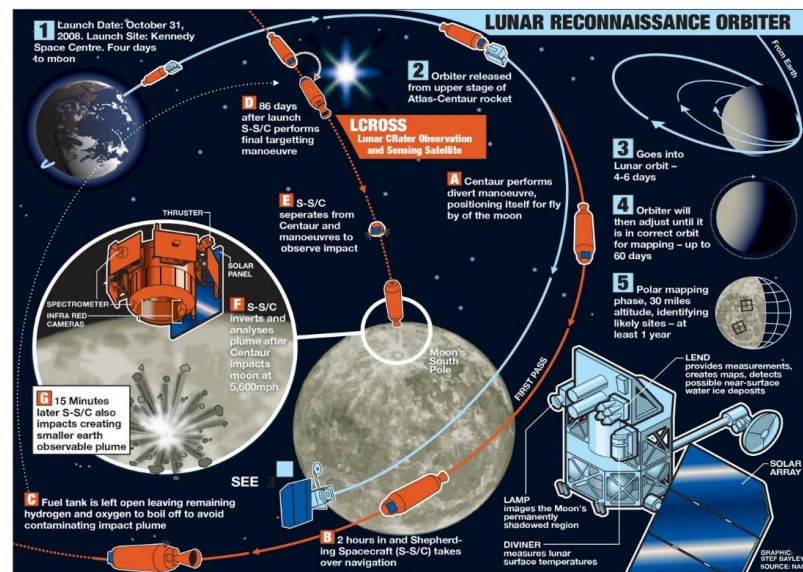


Figure 2.5 NASA's LRO and LCROSS missions Concept of Operations[86].

For LRO mission planning, lunar non-spherical gravity was identified as the dominant perturbing force below 100 km altitude. A technical station-keeping study showed that an initially circular 50 km polar orbit would naturally develop large altitude variations and could eventually impact the lunar surface without correction manoeuvres [84].

LRO therefore provides a strong reference case for low lunar orbit operations. It shows that a low polar orbit can deliver very high scientific and operational value, especially for surface mapping, polar-region analysis, landing-site assessment, and resource investigation. At the same time, the mission illustrates that low lunar orbit requires careful orbit maintenance, accurate gravity-field modelling, regular orbit determination, and long-term operations planning. For a nanosatellite mission, the main lesson is that low lunar orbit can be scientifically attractive, but the orbit-maintenance burden and operational complexity must be treated as part of the mission design from the beginning.

2.2.2 GRAIL

NASA's Gravity Recovery and Interior Laboratory, known as GRAIL, was a twin-spacecraft lunar mission designed to map the Moon's gravitational field in very high resolution. The two spacecraft were launched together on 10 September 2011 and were later named Ebb and Flow. Unlike imaging orbiters, GRAIL did not study the Moon mainly by taking surface images. Its main measurement principle was based on tracking very small changes in the distance and relative velocity between the two spacecraft as they flew over regions with slightly different gravitational attraction [87], [88].

Table 2.2 GRAIL mission overview

Parameter	Description
Agency / operator	NASA / Jet Propulsion Laboratory
Spacecraft class	Twin small lunar orbiters
Launch date	10-Sep-11
Lunar orbit insertion	GRAIL-A: 31-Dec-11; GRAIL-B: 1/2-Jan-12
Mission end	Controlled lunar impact on 17 December 2012
Launch vehicle	Delta II 7920H-10C
Spacecraft mass	Approximately 307 kg each at launch, about 201 kg dry mass
Primary propulsion	Hydrazine propulsion system with MR-106L 22 N main thruster
Main objective	Map the Moon's gravitational field and determine the structure of the lunar interior from crust to core
Operational orbit	Low-altitude, near-circular, near-polar lunar orbit, about 55 km altitude during the primary science phase

Unlike imaging orbiters, GRAIL did not study the Moon mainly by taking surface images. Its main measurement principle was based on tracking very small changes in the distance and relative velocity between the two spacecraft as they flew over regions with slightly different gravitational attraction [88].

The mission was inspired by the Earth-orbiting GRACE gravity mission. In the GRAIL concept, the two spacecraft flew in the same low lunar orbit, one behind the other. When the leading spacecraft passed over a region of stronger gravity, it accelerated slightly before the second spacecraft experienced the same effect. The Lunar Gravity Ranging System measured these small changes in separation between the two spacecraft, allowing the lunar gravity field to be reconstructed. The resulting gravity data were then combined with topography to study the Moon's crust, lithosphere, internal structure, thermal evolution, and mass concentrations, or mascons [87], [88]

GRAIL used a low-energy translunar cruise rather than a short direct transfer. After launch, the spacecraft followed a trajectory that passed near the Sun–Earth L1 region before returning toward the Moon. This extended cruise lasted several months and allowed the two spacecraft to arrive at the Moon slowly and separately, with lunar orbit insertion occurring for GRAIL-A on 31 December 2011 and for GRAIL-B about one day later. After lunar capture, the orbit was adjusted and the spacecraft were placed into the required science formation.[88], [89], [90]

The primary science orbit was a low-altitude, near-circular, near-polar lunar orbit with an altitude of about 55 km. This low altitude was selected because gravity-field sensitivity increases when the spacecraft fly closer to the surface. The orbit was also designed so that natural lunar gravity perturbations could be used without requiring continuous orbit-maintenance manoeuvres during the science phase. During the primary science phase, the Moon rotated beneath the two spacecraft, allowing global gravity coverage over several mapping cycles [88], [89], [90].

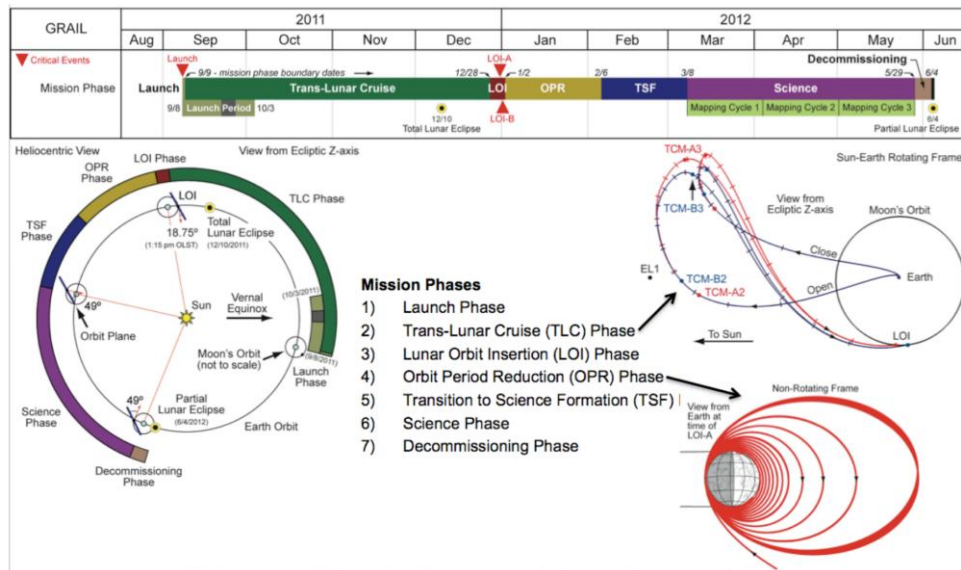


Figure 2.6 GRAIL mission Concept of Operations[87].

The concept of operations (Figure 2.6) was based on formation flying and precise inter-spacecraft tracking. During the science phase, the two spacecraft flew facing each other so that their Ka-band antennas could remain pointed toward one another. Their separation changed during the mapping cycles, from roughly 100–225 km and later down to about 65 km, because different spacecraft separations supported different gravity-field measurement scales. The primary science phase lasted from March to May 2012, and an extended mission phase began in August 2012 [87], [88].

GRAIL ended with a planned disposal by lunar impact. After completing the primary and extended science missions, the two spacecraft no longer had enough altitude and fuel to continue science operations. They were therefore intentionally guided into a mountain near the lunar north pole on 17 December 2012[87].

The main lesson from GRAIL is that accurate lunar gravity knowledge is essential for low lunar orbit design. The same gravitational irregularities that the mission mapped are also responsible for perturbing low lunar spacecraft orbits. GRAIL also showed that low-altitude lunar missions can provide high scientific return, but require precise orbit determination, careful formation control, and a strong understanding of the lunar gravity field.

2.2.3 LADEE (lunar atmosphere, dust, and exosphere mission)

NASA's Lunar Atmosphere and Dust Environment Explorer, abbreviated LADEE, was a robotic lunar orbiter designed to investigate the Moon's extremely tenuous atmosphere, near-surface environment, and the influence of that environment on lunar dust. The spacecraft was launched on 6 September 2013 from NASA's Wallops Flight Facility on a Minotaur V launch vehicle. After launch, LADEE entered a series of Earth phasing orbits, which allowed it to arrive at the Moon with the required timing and

geometry for lunar orbit insertion . NASA states that LADEE reached lunar orbit through a three-minute lunar orbit insertion manoeuvre, which initially placed the spacecraft into an elliptical retrograde equatorial orbit with an orbital period of about 24 hours[91], [92].

Table 2.3 LADEE mission overview

Parameter	Description
Agency / operator	NASA / Ames Research Center
Spacecraft class	Small lunar orbiter
Launch date	06-Sep-13
Lunar orbit insertion	06-Oct-13
Mission end	Planned lunar impact on 18 April 2014
Launch vehicle	Minotaur V
Spacecraft mass	Approximately 383 kg
Primary propulsion	Chemical propulsion for lunar transfer correction, lunar orbit insertion, orbit lowering, and disposal
Main objective	Study the lunar atmosphere, near-surface environment, and environmental effects on lunar dust
Operational orbit	Retrograde near-equatorial lunar orbit; science phase approximately 20–60 km altitude

LADEE’s scientific objective was to gather detailed information about the lunar atmosphere, the conditions near the lunar surface, and the environmental influences on lunar dust. The mission was designed to study the lunar exosphere before it could be significantly disturbed by increased human and robotic activity at the Moon. Its payload included three science instruments and one technology demonstration: the Ultraviolet and Visible Light Spectrometer (UVS), the Neutral Mass Spectrometer (NMS), the Lunar Dust Experiment (LDEX), and the Lunar Laser Communications Demonstration (LLCD) [28]. The UVS determined atmospheric composition using light signatures, the NMS measured variations in the lunar atmosphere over multiple lunar orbits, and LDEX collected and analysed dust particles in the tenuous lunar environment [28].

The mission used a low-altitude, near-equatorial lunar orbit rather than a polar mapping orbit. After lunar capture, a sequence of manoeuvres reduced the orbit to an approximately circular 250 km commissioning orbit. The main science phase was then performed at much lower altitude, with the orbit varying between about 20 and 60 km above the lunar surface because of the Moon’s irregular gravity field. A technical description of the LDEX instrument further describes LADEE as following a small-eccentricity, retrograde, equatorial orbit with a characteristic altitude of about 50 km, with typical altitude excursions between approximately 15 and 100 km [45], [91], [92].

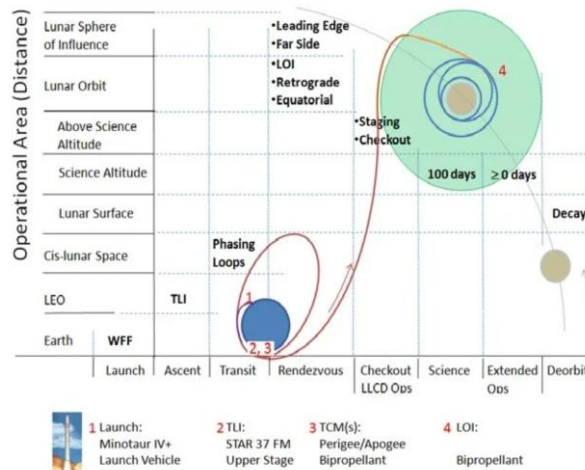


Figure 2.7 LADEE mission Concept of Operations[92]

The concept of operations (Figure 2.7) was closely connected to the orientation and altitude requirements of the science instruments. During dust measurements, LDEX was pointed close to the spacecraft ram direction so that it could detect dust impacts as the spacecraft moved through the lunar environment. The low-altitude orbit allowed the spacecraft to sample the exosphere and dust environment close to the Moon while the lunar surface rotated underneath the orbit. NASA states that during the 100-day science phase, the Moon rotated more than three times beneath LADEE's orbit, supporting measurements over different local times and surface regions [45], [91], [92].

After the science phase, LADEE entered a decommissioning period in which the spacecraft was moved to lower altitudes. The mission ended with a planned impact on the lunar surface on 18 April 2014 UTC [92]. This disposal method was consistent with the very low operational orbit and the limited remaining propellant after science operations. The mission provided direct in-situ measurements of the lunar exosphere and dust environment, including measurements of dust particles over a broad altitude range. LDEX documentation describes the instrument as measuring dust during LADEE's low-altitude lunar operations and being able to observe impact ejecta and possible lofted dust populations [45].

LADEE shows that low-altitude lunar measurements can require an orbit very different from global mapping missions. Its near-equatorial, retrograde orbit was selected for direct sampling of the lunar exosphere and dust environment rather than for polar surface coverage. The mission also shows that lunar dust and exosphere studies depend strongly on altitude, local time, spacecraft pointing, and the interaction between the spacecraft trajectory and the surrounding environment. For future small lunar missions, LADEE is therefore a key reference for designing in-situ measurements close to the Moon.

2.2.4 Kaguya / SELENE

JAXA's Kaguya, also known as SELENE (Selenological and Engineering Explorer) was one of the largest lunar orbiters launched after the Apollo era. The spacecraft was launched on 14 September 2007 by an H-IIA launch vehicle and entered lunar orbit on 4 October 2007. The mission consisted of the main orbiter, Kaguya, and two small subsatellites: Okina, used as a relay satellite, and Ouna, used for VLBI radio science measurements [9], [93].

Table 2.4 Kaguya/SELENE mission overview

Parameter	Description
Agency / operator	JAXA
Spacecraft class	Large lunar orbiter with two small subsatellites
Launch date	14-Sep-07
Lunar orbit insertion	04-Oct-07
Mission end	Controlled lunar impact of main orbiter on 10-Jun-09
Launch vehicle	H-IIA
Spacecraft mass	Main orbiter approximately 3 t; Okina and Ouna approximately 50 kg each
Primary propulsion	Chemical propulsion for lunar orbit insertion, orbit control, and disposal
Main objective	Study the origin and evolution of the Moon and obtain engineering data for future lunar utilization
Operational orbit	Main orbiter: circular polar orbit at approximately 100 km altitude; later lowered to 50 km and then lower-altitude extended operations
Subsatellite orbits	Okina: elliptical polar orbit approximately 100x2400 km; Ouna: elliptical polar orbit approximately 100x800 km

After lunar arrival, the main spacecraft released the two subsatellites and then entered its nominal observation orbit. The main orbiter operated in a nearly circular 100 km polar lunar orbit, passing over both lunar poles. Okina was placed in an elliptical polar orbit of approximately 100x2400 km, while Ouna used an elliptical polar orbit of approximately 100x800 km [9], [93]. This architecture allowed the mission to combine global mapping from the main orbiter with radio-science measurements supported by the subsatellites.

The main scientific objective of Kaguya was to study the origin and evolution of the Moon through global observations. The payload included instruments for topography, mineralogy, elemental composition, magnetic-field measurements, plasma observations, radar sounding, gravity investigation, and high-definition imaging. The Terrain Camera, Multiband Imager, Spectral Profiler, Lunar Radar Sounder, Laser Altimeter, X-ray and gamma-ray spectrometers, and plasma instruments were used to produce a detailed global dataset of the lunar surface and environment [9], [93].

The mission concept of operations (Figure 2.8) was based on continuous global observation from polar orbit. As the Moon rotated beneath the spacecraft, Kaguya repeatedly passed over different surface regions, allowing global coverage from the 100 km orbit. The relay satellite Okina was used to support radio-science measurements, especially for the lunar far side, where direct Earth tracking of the main orbiter is not

possible. Ouna supported differential VLBI measurements, improving orbit determination and gravity-field analysis [9], [93], [94].

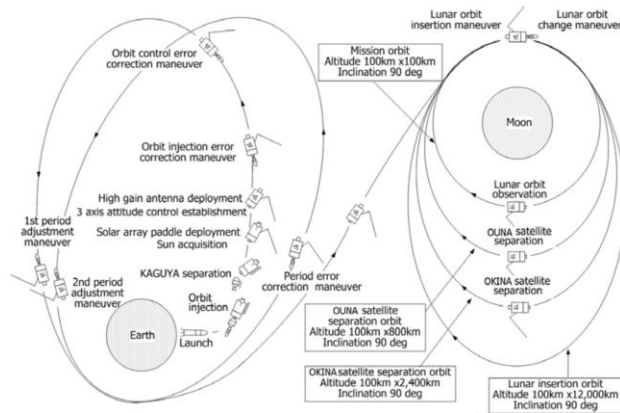


Figure 2.8 Kaguya / SELENE mission Concept of Operations[95].

After completing its nominal operation phase, Kaguya entered an extended mission. The main orbiter was lowered from the 100 km nominal observation orbit to about 50 km in February 2009, and later to lower-altitude operations with perilune around 10–30 km from April 2009. These lower-altitude operations allowed additional observations, especially for magnetic-field and gravity-related studies. The mission ended with a controlled impact near Gill crater on 10 June 2009 UTC [9], [93], [94].

During the extended mission, one of the reaction wheels developed increased friction torque, and the mission plan had to be modified. The reduced attitude-control capability affected low-altitude operations, increased the use of thrusters for attitude control, and complicated some data processing and orbit determination during the extended phase [94].

2.2.5 Chandrayaan-2

Chandrayaan-2 was India's second lunar exploration mission and consisted of an orbiter, the Vikram lander, and the Pragyan rover. The mission was launched on 22 July 2019 by GSLV MkIII-M1 from Satish Dhawan Space Centre. After a sequence of Earth-bound manoeuvres, the spacecraft entered its lunar transfer trajectory on 14 August 2019 and performed lunar orbit insertion on 20 August 2019. The orbit was then reduced through lunar-bound manoeuvres until the orbiter reached a 100 x 100 km circular polar orbit around the Moon [96], [97].

Table 2.5 Chandrayaan mission overview

Parameter	Description
Agency / operator	ISRO
Spacecraft class	Orbiter, lander, and rover mission
Launch date	22-Jul-19
Lunar orbit insertion	20-Aug-19
Mission status	Orbiter operational after loss of lander communication
Launch vehicle	GSLV MkIII-M1
Spacecraft mass	Orbiter: 2,379 kg; Vikram lander: 1,471 kg; Pragyan rover: 27 kg
Primary propulsion	Chemical propulsion for Earth-bound manoeuvres, lunar transfer, lunar orbit insertion, orbit lowering, and lander descent
Main objective	Study lunar topography, mineralogy, elemental composition, lunar exosphere, and polar-region water/ice indicators; demonstrate soft landing and rover operation
Operational orbit	Orbiter in approximately 100x100 km circular lunar polar orbit

The mission objectives combined lunar science with technology demonstration. The orbiter was intended to study the Moon from polar orbit, while the lander and rover were intended to demonstrate soft landing and surface mobility near the lunar south-polar region. The orbiter carried eight scientific payloads: CLASS for X-ray fluorescence measurements of major elements, XSM for solar X-ray monitoring, CHACE-2 for in-situ lunar neutral exosphere measurements, DFSAR for dual-frequency radar observations of polar regions and possible water ice, IIRS for mineral and volatile mapping, TMC-2 for terrain mapping, OHRC for high-resolution landing-site imaging, and DFRS for radio-science measurements [96], [98].

The orbiter operated in an approximately 100x100 km lunar polar orbit. This orbit supported global and polar-region observations while also allowing the orbiter to support the lander segment before separation. ISRO states that the orbiter remained healthy after the lander loss and that all orbiter payloads were operational. The orbiter mass is listed as 2,379 kg; the mission design and launch performance also extended the expected orbiter lifetime to almost seven years instead of the originally planned one year .[96]

The Vikram lander separated from the orbiter on 2 September 2019. After separation, de-orbit manoeuvres lowered the lander trajectory to approximately 35x101 km before the landing attempt. On 7 September 2019, Vikram followed the planned descent trajectory from this orbit down to around 2 km above the lunar surface, after which communication between the lander and the ground station was lost.

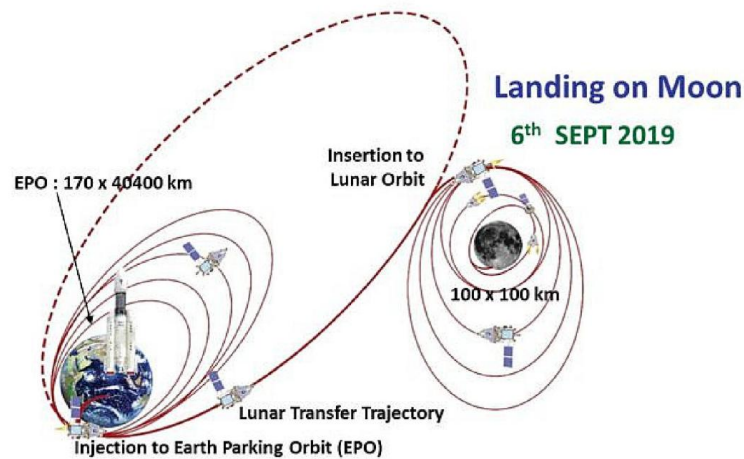


Figure 2.9 Chandrayaan-2 Concept of Operations[96].

The concept of operations (Figure 2.9) therefore separated the mission into an orbital science segment and a surface-demonstration segment. The orbiter was responsible for long-duration remote sensing from polar orbit, while Vikram was intended to perform autonomous powered descent, soft landing, and deployment of the Pragyan rover for approximately one lunar day. Although the lander and rover did not complete their planned surface phase, the orbiter continued the scientific mission from lunar orbit [96], [97].

The mission also shows the criticality of the final descent phase: even after successful launch, transfer, lunar orbit insertion, lander separation, and descent down to low altitude, the terminal landing phase remained mission critical. Chandrayaan-2 also demonstrates the value of separating orbiter and lander functions because the orbiter remained available for long-duration lunar science after the loss of the surface segment.

2.2.6 CAPSTONE

NASA's CAPSTONE, the Cislunar Autonomous Positioning System Technology Operations and Navigation Experiment, is a 12U CubeSat developed as a pathfinder for the lunar near-rectilinear halo orbit planned for Gateway. The spacecraft is owned and operated by Advanced Space and supported by NASA's Small Spacecraft Technology programme. CAPSTONE was launched on 28 June 2022 and entered its target lunar orbit on 13 November 2022. NASA describes it as the first spacecraft to operate in this NRHO and verify its dynamics[20], [99], [100] .

Table 2.6 CAPSTONE mission overview

Parameter	Description
Agency / operator	NASA / Advanced Space
Spacecraft class	12U CubeSat
Launch date	28-Jun-22
Lunar orbit insertion	13-Nov-22
Mission status	Active / extended mission after completion of primary objectives
Launch vehicle	Rocket Lab Electron with Lunar Photon upper stage
Spacecraft mass	Approximately 25 kg / 55 lb
Primary propulsion	Hydrazine propulsion system with eight small thrusters
Main objective	Demonstrate operations in near-rectilinear halo orbit and test autonomous cislunar navigation
Transfer strategy	Ballistic Lunar Transfer
Operational orbit	Near-rectilinear halo orbit around the Moon, similar to the orbit planned for Gateway

CAPSTONE used a Ballistic Lunar Transfer (Figure 2.10). After deployment, the spacecraft followed a long, fuel-efficient route that used solar gravity and trajectory-correction manoeuvres to target lunar arrival. Advanced Space describes the transfer as reaching approximately 1.5 million km from Earth before returning toward the Moon for NRHO insertion. This transfer strategy reduced the propulsion demand, but required several months of cruise operations, navigation updates, and trajectory-correction manoeuvres.

The target orbit is a near-rectilinear halo orbit, a highly elongated three-body orbit in the Earth–Moon system. NASA describes the orbit as bringing CAPSTONE within about 1,000 miles of one lunar pole at close approach and about 43,500 miles from the opposite pole at its farthest point, with an orbital period of approximately seven days. This orbit was selected for testing because Gateway is planned to use the same orbit family, and CAPSTONE could provide real operational data on navigation, communication, station keeping, and orbit maintenance in this environment [20], [99], [100].

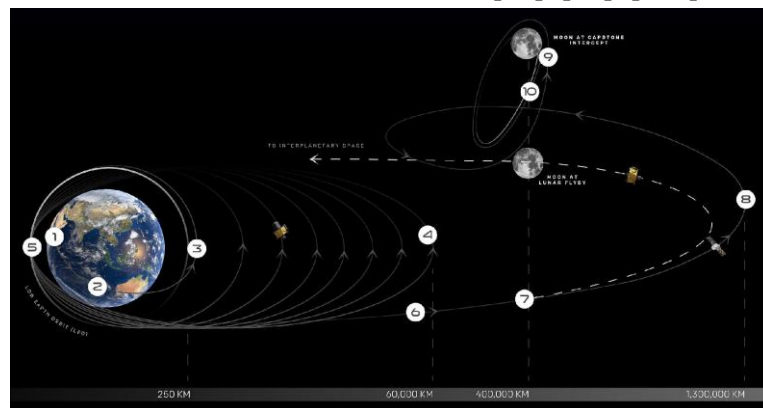


Figure 2.10 CAPSTONE mission Concept of Operations[99].

The concept of operations focused on demonstrating that a small spacecraft can enter, maintain, and operate in NRHO. After orbit insertion, the mission team performed clean-

up manoeuvres to refine the trajectory, then used the spacecraft to validate power and propulsion requirements for maintaining the orbit. NASA states that CAPSTONE completed its primary objectives within the planned six-month period and then continued into an extended mission for further orbit characterization, communication, navigation, and networking demonstrations [20], [99].

A major technology demonstration was the Cislunar Autonomous Positioning System. CAPSTONE communicates with NASA's Lunar Reconnaissance Orbiter and uses the crosslink data to estimate relative distance and velocity between the spacecraft. During a May 2023 experiment, CAPSTONE successfully tested this spacecraft-to-spacecraft navigation approach, which is intended to reduce dependence on Earth-based tracking for future lunar missions [20].

The mission also experienced operational anomalies during cruise. After a trajectory-correction manoeuvre on 8 September 2022, CAPSTONE began tumbling beyond the capability of its reaction wheels, experienced periodic computer resets, and was using more power than it generated from its solar panels. NASA later reported that the likely cause was a valve-related issue in one of the spacecraft's eight thrusters, which produced thrust when the propulsion system was pressurized. The team stopped the spin and recovered three-axis attitude control in October 2022 [20].

For CubeSat and nanosatellite missions, CAPSTONE is mainly relevant because it shows that very small spacecraft can perform complex cislunar navigation and orbit-maintenance operations, but only with careful trajectory design, strong ground operations, and anomaly recovery capability. The mission also shows that propulsion, attitude control, communication, power generation, and navigation are tightly coupled during deep-space CubeSat operations.

2.2.7 Artemis I Secondary Payload CubeSats

Artemis I carried several CubeSat-class secondary payloads intended to demonstrate low-cost deep-space exploration technologies and perform focused lunar science investigations. Among the most relevant missions for this thesis are LunaH-Map, and Lunar IceCube, which targeted lunar polar regions, water and volatile detection, and advanced low-energy transfer concepts. These missions are particularly important because they demonstrate modern approaches to miniaturized propulsion, autonomous operations, weak-stability-boundary trajectories, and low-altitude lunar science orbits relevant to future lunar nanosatellite missions.

Although several missions encountered significant technical difficulties after deployment, they still provided valuable lessons for future lunar CubeSat missions, especially in propulsion reliability, communication robustness, autonomous operations, and trajectory correction during early mission phases.

Their mission concepts showed that advanced lunar science investigations can be performed using small and relatively low-cost spacecraft. At the same time, the missions

highlighted the importance of reliable propulsion systems, robust communication architectures, autonomous spacecraft operations, and early trajectory-correction capabilities for future lunar nanosatellite missions [101], [102], [103].

Table 2.7 Selected Artemis I secondary payload CubeSats overview

Mission	Main objective	Transfer strategy	Planned operational orbit	Propulsion system	Main issue
LunaH-Map	Hydrogen mapping near the lunar south pole	Lunar flyby and low-thrust correction manoeuvres	Highly elliptical polar orbit with passes as low as ~10 km	Busek BIT-3 iodine RF propulsion	Propulsion valve issue
Lunar Flashlight	Detection of surface water ice	Low-energy transfer toward NRHO	NRHO with close south-pole passes	ASCENT green monopropellant propulsion	Thruster underperformance
Lunar IceCube	Investigation of lunar water and volatiles	Weak capture and low-thrust spiral-down	Elliptical lunar orbit (~100 kmx5000 km)	Busek BIT-3 iodine RF propulsion	Communication issues

2.2.7.1 LunaH-Map

LunaH-Map (Lunar Polar Hydrogen Mapper) was a 6U CubeSat mission led by Arizona State University and launched as an Artemis I secondary payload on 16 November 2022. The spacecraft was designed to measure neutron flux variations above the lunar south pole in order to identify hydrogen-rich regions potentially associated with water ice deposits inside permanently shadowed craters.

The mission planned to use the Busek BIT-3 iodine RF electric propulsion system to achieve a highly elliptical polar orbit with extremely low-altitude passes over the lunar south pole. Planned science measurements were expected from altitudes as low as approximately 10 km above the lunar surface[101], [104]. Such low-altitude passes are particularly relevant for lunar observations because they allow high spatial-resolution observations close to the surface.

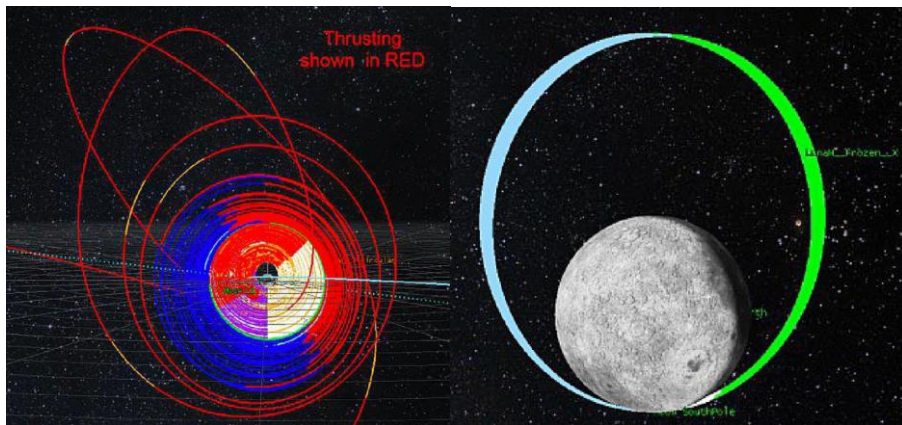


Figure 2.11 LunaH-Map lunar orbit trajectory

After deployment from Artemis I, several spacecraft systems were successfully commissioned. However, the propulsion system was unable to generate the required

thrust for the planned trajectory-correction manoeuvres. NASA later reported that a propulsion-system valve was likely partially stuck, preventing proper operation of the propulsion system. Because the required manoeuvres could not be completed, LunaH-Map failed to enter its planned lunar orbit and mission operations ended in 2023 [88].

Despite the unsuccessful orbit insertion, the mission still demonstrated several important deep-space CubeSat technologies and returned useful neutron detector measurements during the lunar flyby phase. For future lunar nanosatellite missions, LunaH-Map highlights the importance of propulsion-system reliability and early post-deployment commissioning.

2.2.7.2 Lunar Flashlight

Lunar Flashlight was a NASA/JPL-led 6U CubeSat intended to launch on the ARTEMIS 1 mission as the secondary payload, but missed the integration window [105]. It launched on 11 December 2022 as a rideshare payload together with the HAKUTO-R Mission 1 lander. [106]

The mission aimed to detect exposed water ice in permanently shadowed regions near the lunar south pole using active laser reflectometry. The spacecraft was designed to follow a low-energy trajectory toward a near-rectilinear halo orbit (NRHO). The planned orbit would provide repeated close passes approximately 15 km above the lunar south pole while minimizing propulsion requirements compared to traditional low lunar orbits.[102], [107]

Lunar Flashlight used the ASCENT green monopropellant propulsion system, representing one of the first interplanetary demonstrations of this propulsion technology. Shortly after launch, however, the spacecraft experienced severe thruster underperformance, likely caused by obstructions in the fuel lines. Multiple recovery attempts and modified manoeuvre concepts were investigated, but the spacecraft was ultimately unable to achieve the required trajectory corrections or enter the planned NRHO science orbit [108].

Although the primary science objectives were not achieved, the mission still demonstrated several technologies related to deep-space CubeSat operations, precision navigation, and miniaturized spacecraft systems. The mission also provided important lessons regarding propulsion-system robustness and contingency operations for lunar CubeSats.

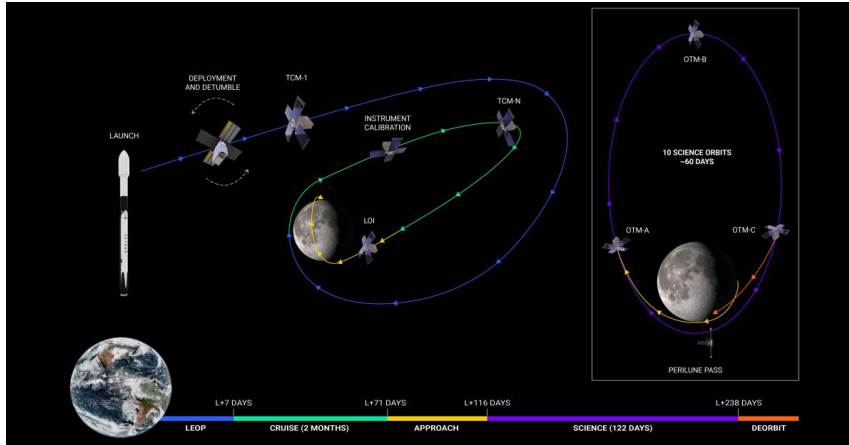


Figure 2.12 Lunar Flashlight mission Concept of Operations[109].

2.2.7.3 Lunar IceCube

Lunar IceCube was a NASA-supported 6U CubeSat led by Morehead State University and launched aboard Artemis I on 16 November 2022. The spacecraft was designed to investigate the distribution of lunar water and volatiles using the infrared spectrometer [103], [110].

The planned mission architecture relied on a low-energy multibody transfer using lunar flybys, solar-gravity phasing, weak capture, and subsequent low-thrust spiral-down manoeuvres. After capture into the Earth–Moon system, the Busek BIT-3 iodine electric propulsion system was intended to gradually reduce the orbital altitude until reaching the final science orbit [110], [111].

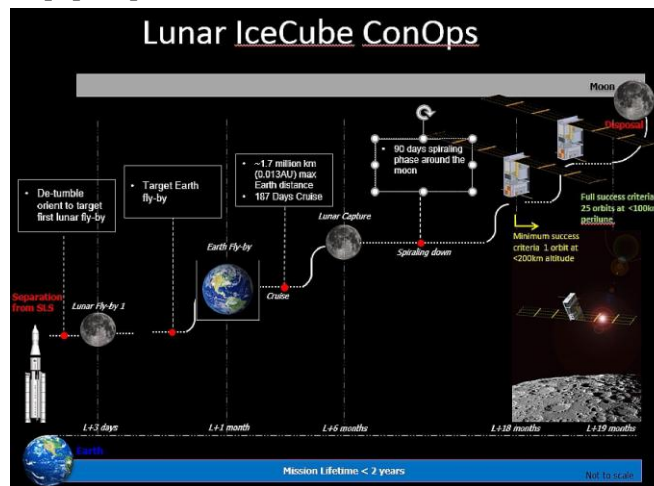


Figure 2.13 Lunar IceCube Concept of Operations [103].

The planned operational orbit was a highly inclined elliptical lunar orbit with an approximate 100 km perilune and 5000 km apolune. This orbit would enable repeated spectroscopic observations of the lunar surface while maintaining feasible communication and propulsion requirements for a CubeSat platform. The mission

concept also required a high level of spacecraft autonomy because ground contact opportunities were limited to approximately every third orbit [90], [97].

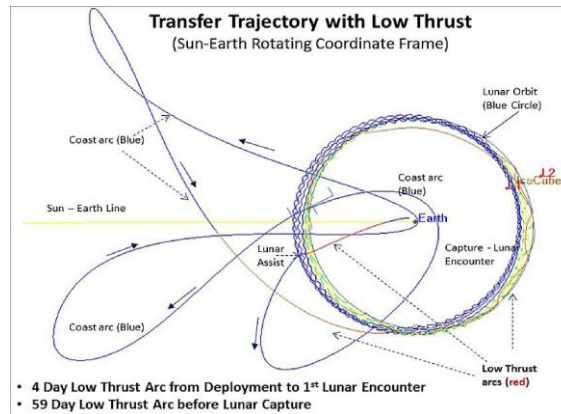


Figure 2.14 Low-thrust Lunar IceCube transfer trajectory [103].

Shortly after deployment, Lunar IceCube established initial contact with the ground segment, but communication problems soon occurred. Reports indicate that the spacecraft carrier signal was detected for approximately one hour with a significant frequency offset before the signal was lost. Continued tracking attempts over the following weeks were unsuccessful, preventing reliable spacecraft commanding and completion of the planned trajectory-correction manoeuvres and lunar science orbit insertion. [112].

Nevertheless, the mission remains highly relevant for lunar nanosatellite mission analysis because it demonstrates the complexity of combining low-energy transfers, electric propulsion, autonomous operations, and deep-space CubeSat constraints within a small spacecraft platform.

2.3 Planned and future lunar missions

Future lunar missions show a clear shift toward mission-specific orbit design, south-pole access, shared communication/navigation infrastructure, and smaller spacecraft used as targeted science or technology demonstrators. The most relevant planned missions for this survey are therefore not only large human-exploration missions, but also relay spacecraft, CubeSats, SmallSats, and robotic missions testing volatile detection, low-altitude observations, autonomous operations, and lunar surface access.

First important mission planend is Lunar Gateway mission, which orbital is planned to operate in a near-rectilinear halo orbit, passing as close as about 1,500 km and as far as about 70,000 km from the Moon [76], [77].

Commercial lunar access is also becoming more important. Griffin-1 is planned as a CLPS delivery to the lunar south pole, while NASA has selected Blue Origin to prepare delivery of the VIPER rover (Figure 2.15) to the south pole in late 2027, with the final delivery option depending on review of the task order and Blue Origin's first Blue Moon

MK1 flight [113], [114]. These missions are surface missions, to test south-pole access, landing operations, volatile prospecting, and commercial delivery models.



Figure 2.15 NASA's VIPER rover and Blue Origin's Blue Moon Mark 1 lander [114].

Lunar Pathfinder is planned as a lunar relay satellite in an Elliptical Lunar Frozen Orbit with an operational lifetime of about eight years, providing S-band and UHF links to lunar users and X-band relay to Earth [115]. LUMIO is a planned 12U CubeSat that will operate in a halo orbit around the Earth–Moon L2 point to observe meteoroid impacts on the lunar far side [116]. VMMO is a planned 16U CubeSat for mapping water ice and ilmenite near the lunar south pole using a polar frozen orbit and active lidar measurements [117]. These missions show that CubeSats are moving from simple technology demonstrations toward operational roles in relay, environmental monitoring, resource mapping, and cislunar situational awareness.

Future Chinese missions continue the south-pole and infrastructure trend. Chang'e-7 is planned to target the lunar south pole, search for water ice, and use a multi-element architecture including an orbiter, lander, rover, and hopping probe for difficult polar terrain and permanently shadowed regions [118]. Chang'e-8 is planned to target the Leibnitz-Beta Plateau near the south pole and conduct scientific exploration and in-situ resource utilization experiments as a precursor to the International Lunar Research Station [81].

India's next lunar missions also move toward more complex architectures. Chandrayaan-4 is planned as a lunar sample-return mission from the southern polar region, including surface sampling, ascent from the Moon, docking in lunar orbit, sample transfer, Earth return, and re-entry [119]. LUPEX / Chandrayaan-5, developed with JAXA, is planned to investigate the quantity and quality of lunar water resources in the south polar region using a rover equipped with instruments for water, volatile, subsurface, and regolith analysis [120], [121].

Europe is also preparing independent surface-access capability through Argonaut. The proposed first Argonaut mission would launch on Ariane 6 around the end of 2030 and demonstrate Europe's end-to-end capability to land on the lunar south-pole region, with possible European payloads such as surface robotic assets or navigation/telecommunication modules [122].

3. MISSION OBJECTIVES AND ORBIT-SELECTION LOGIC FOR THE PROPOSED NANOSATELLITE MISSION

3.1 Mission objective of the proposed nanosatellite

The primary objective of the mission is to characterize dust in the lunar exosphere and assess its impact on spacecraft systems and future human activities. The mission focuses on in-situ measurements of dust properties, composition, and temporal behaviour, while demonstrating relevant sensing technologies. Derived mission objectives are:

Objective 1: Characterization of Lunar Exospheric Dust

- Generate a spatial dust distribution map of the lunar exosphere
- Determine the elemental composition and relative abundance of particles (verification of PLD capability)
- Estimate particle size distribution (validation of PLD performance)

Objective 2: Technology Demonstration

- Demonstrate the functionality and performance of the HANKA mass spectrometer for in-situ particle detection in the lunar exosphere

Objective 3: Observation of Dust-Generating Events

- Detect transient increases in dust density associated with external events
- Correlate measured particle flux, size distribution, and composition with dust-generating events
- Identify temporal signatures (e.g., short-term spikes) in exospheric dust measurements
- Distinguish between different dust sources, including:
 - natural events (e.g., micrometeoroid impacts)
 - artificial events (e.g., lander/plume interactions, surface operations)
- Support the localization and timing of events based on measurement data and contribute to understanding the relative contribution of different sources to the lunar exosphere

3.2 Derived observation and operational requirements

Based on the mission objectives, a set of derived observation and operational requirements was defined to describe what the mission must achieve during lunar exosphere operations (Table 3.1). These requirements cover the core science measurements, payload performance, data association, measurement cadence, event detection, contamination control, mission phasing, and end-of-life considerations. They provide a link between the high-level mission objectives and the later concept of

operations, orbit selection, manoeuvre analysis, and station-keeping strategy. The requirements are preliminary and include several TBD values, which is common practice at this early stage of mission design. These values should be refined in later design phases once the payload performance, final orbit design, and operational constraints are better defined.

Table 3.1 Observation and operational requirements

ID	Requirement	Category	Parent objective
MOR-01	The mission shall perform in-situ detection of dust particles in the lunar exosphere.	Observation	Obj. 1, Obj. 2
MOR-02	The mission shall collect dust measurements suitable for generating a spatial distribution map of lunar exospheric dust.	Science / observation	Obj. 1
MOR-03	Each dust measurement shall be associated with time, spacecraft position, altitude, attitude, and lunar surface region below the spacecraft.	Data / operations	Obj. 1, Obj. 3
MOR-04	The mission shall remain at each observation altitude long enough to collect dust measurements over different lunar surface regions, with the required observation duration per altitude phase (30 days).	Observation / operations	Obj. 1
MOR-05	The mission shall support comparison of dust measurements at TBD altitude regimes above the lunar surface.	Observation / orbit	Obj. 1
MOR-06	The mission shall include an initial observation phase at a safe initial mapping altitude suitable for commissioning, payload validation, and background dust characterization, before lower-altitude science operations are attempted.	Mission phasing	Obj. 1, Obj. 3
MOR-07	The mission shall establish a nominal background dust-density level during quiet observation periods.	Observation / data analysis	Obj. 1, Obj. 3
MOR-08	The mission shall be able to detect short-term increases in dust density or particle flux above the nominal background level.	Event detection	Obj. 3
MOR-09	The measurement cadence shall be sufficient to identify short-duration temporal signatures, such as spikes or rapid increases in particle flux (TBD s).	Observation / data	Obj. 3
MOR-10	The mission shall support correlation of dust measurements with possible natural dust-generating events, such as micrometeoroid impacts.	Science / event analysis	Obj. 3
MOR-11	The mission shall support correlation of dust measurements with possible artificial dust-generating events, such as lander plume interactions or surface operations.	Science / event analysis	Obj. 3
MOR-12	The payload shall support estimation of particle elemental composition and relative abundance within TBD measurable range.	Payload / science	Obj. 1, Obj. 2
MOR-13	The payload shall support estimation of particle size distribution within TBD measurable range.	Payload / science	Obj. 1, Obj. 2
MOR-14	The mission shall demonstrate the functionality and performance of the HANKA mass spectrometer during lunar exosphere operations.	Technology demonstration	Obj. 2
MOR-15	The mission shall collect sufficient housekeeping and calibration data to distinguish between true lunar dust detections, instrument background, and spacecraft-generated contamination.	Data quality / operations	Obj. 1, Obj. 2, Obj. 3

MOR-16	The spacecraft operations concept shall minimize contamination of dust measurements from thruster firings, outgassing, attitude-control activity, or spacecraft-generated particles.	Operations / measurement quality	Obj. 1, Obj. 2, Obj. 3
MOR-17	The mission shall define no-measurement or flagged-data windows around propulsive manoeuvres if thruster activity may affect particle measurements (flagged-data window TBD min/h before and after manoeuvre).	Data quality	Obj. 1, Obj. 2, Obj. 3
MOR-18	The mission shall support reconstruction of the approximate timing and source region of detected dust-generating events, with localization accuracy (TBD km / TBD deg).	Data analysis / operations	Obj. 3
MOR-19	The mission shall define a feasible end-of-life strategy compatible with applicable space debris mitigation requirements.	End-of-life	

3.3 Orbit-selection criteria

Based on the observation and operational requirements, a set of derived science orbit requirements was defined to support the selection of a suitable lunar orbit concept (Table 3.2). These requirements describe the main orbit-related needs of the mission, including surface coverage, repeated regional sampling, altitude selection, illumination conditions, communication opportunities, orbit maintenance, and correlation with dust-generating events.

The requirements provide the link between the mission objectives and the later orbit trade-off, manoeuvre analysis, station-keeping assessment, and ConOps definition. Some parameters remain preliminary and are marked as TBD, since they depend on the final payload performance, communication architecture, and detailed mission design.

Table 3.2 Orbit selection requirements

ID	Requirement	Category	Parent requirement
SOR-01	The selected orbit shall provide access to a broad range of lunar latitudes and surface regions.	Coverage	MOR-02, MOR-04
SOR-02	The selected orbit shall allow repeated measurements over different lunar regions at a comparable altitude.	Regional sampling	MOR-04
SOR-03	The selected orbit shall support measurements under different local-time and illumination conditions.	Temporal / illumination sampling	MOR-03, MOR-08, MOR-09
SOR-04	The selected orbit shall provide sufficiently low measurement altitude to improve sensitivity to weak near-surface dust populations, with candidate science altitudes to be evaluated during orbit analysis (initial altitude ~100 km TBC ; lowest altitudes 10 km TBC).	Science geometry	MOR-01, MOR-05
SOR-05	The selected orbit shall allow comparison between 10 TBC altitude regimes above the lunar surface.	Altitude sampling	MOR-05
SOR-06	The selected orbit shall maintain altitude variation within an acceptable range during each observation phase, with the allowable variation (± 2 TBC km).	Orbit maintenance	MOR-02, MOR-04, MOR-05
SOR-07	The selected orbit shall provide sufficient communication opportunities or compatibility with relay support for TT&C and science-data return.	Communications	

SOR-08	The selected orbit shall provide sufficient solar illumination throughout the mission to support spacecraft power generation and thermal constraints.	Power & Thermal	
SOR-09	The selected orbit shall support correlation of measurements with predicted or observed dust-generating events.	Event correlation	MOR-08, MOR-09, MOR-10, MOR-11, MOR-18

3.4 Orbits comparison

Lunar orbits can be classified using several independent parameters. A mission orbit is usually defined by a combination of these parameters, rather than by one label only. The overview of different orbital configurations can be seen in the Table 3.3.

Table 3.3 Orbit types comparison

Orbit option	Classification parameter	Main advantage	Main limitation	Suitability
Very low lunar orbit	Altitude	Maximum proximity to the lunar surface	High impact risk and strong station-keeping demand	Very high science value, high risk
Low lunar orbit	Altitude	Good surface access and measurement resolution	Strong perturbations	High
High lunar orbit	Altitude	Lower perturbation sensitivity, useful for staging/relay	Poor surface resolution	Low to medium
Equatorial / low-inclination orbit	Inclination	Repeated coverage of low-latitude regions	Limited polar and high-latitude coverage	Low to medium
Inclined non-polar orbit	Inclination	Covers selected latitude bands	Does not provide full global/polar coverage	Medium
Polar orbit	Inclination	Broad latitude coverage over time	Infrequent coverage of low level regions	High
Circular to Near-circular orbit	Eccentricity	Uniform altitude and measurement geometry	Higher station keeping demand	High
Elliptical orbit	Eccentricity	Useful for capture, staging, or special observation geometry	Non-uniform altitude and science conditions	Medium
Frozen orbit	Stability / dynamical family	Improved natural stability	Limited feasible configurations	Medium
NRHO / DRO	Stability / dynamical family	Stable, useful for infrastructure and staging	Far from lunar surface	Low for this mission

3.5 Final orbit and rationale for the selection

The preliminary orbit concept assumes a low, nearly circular, high-inclination lunar orbit. After an initial science and calibration phase, the spacecraft would lower its orbit to an initial scientific altitude of approximately 100 km. The scientific phase consists of several altitudes, the altitude changes in controlled steps of approximately 10 km per phase. Each altitude phase would be maintained for a sufficient observation period to sample different lunar surface regions before the next lowering manoeuvre is executed.

This strategy allows comparison of dust concentration as a function of altitude and lunar region. The right ascension of the ascending node is assumed to be driven mainly

by launch availability during the early mission design stage, while the resulting local-time evolution should be evaluated later because dust activity may depend on illumination and terminator conditions. Since low lunar orbits are significantly affected by lunar gravity-field perturbations, the altitude reduction should be achieved through planned propulsive manoeuvres rather than uncontrolled orbital evolution. Station keeping is also required to maintain the spacecraft within the desired altitude and timing window during each observation phase.

3.6 Mission concept of operations

The concept of operations defines the high-level mission sequence from launch to end-of-life. For this preliminary feasibility study, the ConOps is focused on the lunar mission phases after delivery, commissioning, orbit lowering, science operations, station-keeping, and final disposal. The exact launch vehicle and lunar delivery architecture are not fixed at this stage; therefore, the early mission phase is treated in a generic way.

The mission is divided into five main phases: launch, delivery and deployment, early spacecraft operations and commissioning, transfer to the first science orbit, science operations with stepwise descent, and end-of-life disposal. This structure provides the operational framework for the manoeuvre capability and station-keeping analyses presented in the following sections.

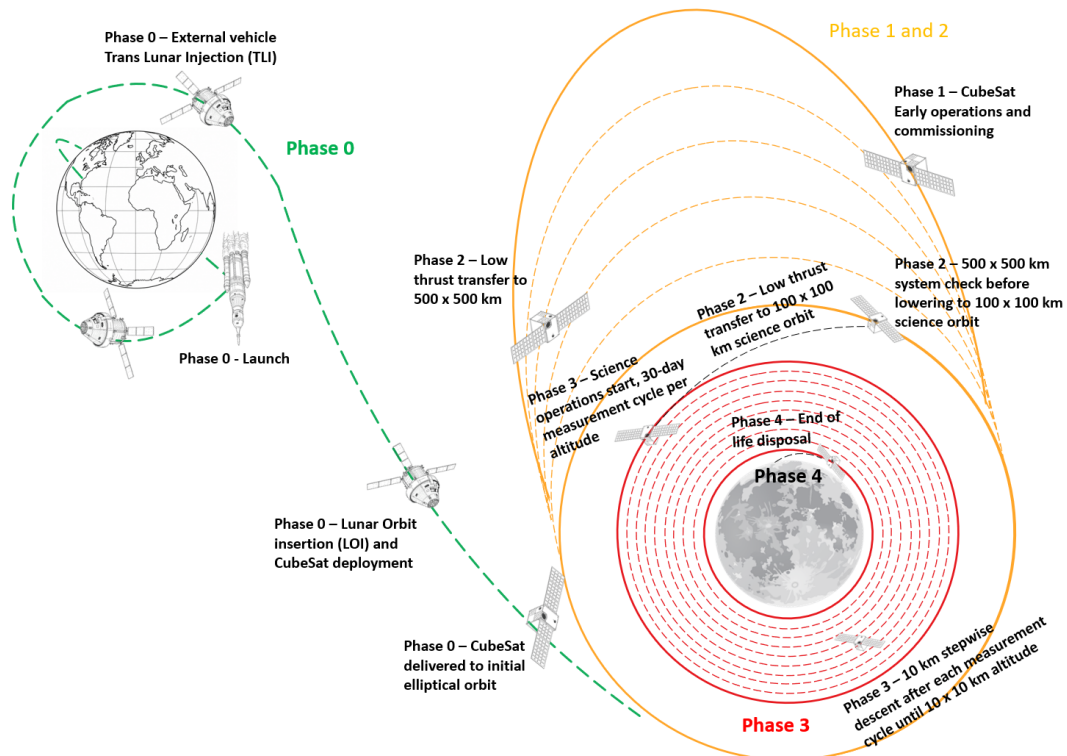


Figure 3.1 MAIA Mission proposed Concept of operations

3.6.1 Phase 0 - Launch, delivery, and deployment

The launch, delivery, and deployment phase covers the period from launch until spacecraft separation from the launch vehicle, carrier spacecraft, or delivery system. For this feasibility study, the detailed lunar transfer architecture has not yet been selected. Therefore, this phase is treated as a generic mission access phase and works as the interface between the external launch or delivery provider and the spacecraft's independent lunar operations.

The spacecraft is assumed to remain integrated with the launcher, deployer, transfer vehicle, or host spacecraft during this phase. After successful delivery and deployment, the spacecraft enters autonomous early operations and commissioning. The analysis assumes that the spacecraft is delivered into a lunar arrival condition or captured lunar orbit compatible with the 5000 x 500 km staging orbit used as the starting point for the manoeuvre assessment.

Launch and delivery configuration

- The spacecraft is integrated into the launch vehicle, deployer, transfer vehicle, or host spacecraft.
- The exact lunar access architecture is not fixed at this stage.
- Possible delivery options include direct lunar transfer, rideshare delivery, Artemis-class secondary payload opportunity, orbital transfer vehicle, or mothercraft-assisted delivery.
- The spacecraft is assumed to be deployed into a lunar arrival condition or into a captured lunar orbit compatible with the mission analysis assumptions.

Deployment sequence

1. **Launch and transfer** - The spacecraft is transported toward the Moon by the selected launch or delivery architecture.
2. **Lunar delivery / capture** - The delivery architecture provides access to the lunar environment or to an initial lunar orbit.
3. **Spacecraft separation** - The spacecraft is released from the launcher, carrier, or deployer.
4. **Initial autonomous state** - After deployment, the spacecraft enters a safe autonomous mode and prepares for first contact and commissioning.

Expected outcomes

- Spacecraft successfully delivered to the lunar mission environment.
- Spacecraft safely separated from the delivery system.

3.6.2 Phase 1 - Early spacecraft operations and commissioning

The early spacecraft operations and commissioning phase starts after spacecraft deployment from the launch vehicle, transfer vehicle, or carrier system. During this phase, the spacecraft begins independent operation, and the main objective is to confirm that all critical subsystems are functional before starting orbit-lowering manoeuvres.

Early operations

1. **Initial autonomous mode** - After separation, the spacecraft enters a predefined initial mode.
2. **Power system activation** - The EPS is checked to confirm power-positive operation.
3. **Attitude stabilisation** - The AOCS is activated and the spacecraft attitude is stabilised.
4. **First telemetry acquisition** and link verification- Initial telemetry is received and analysed by the operations team, Telecommand and telemetry links are verified.
5. **Subsystem health check** - OBC, EPS, TT&C, AOCS, thermal, and propulsion subsystem status are verified.
6. **Propulsion system check** - The propulsion system is checked before beginning orbit-lowering operations.
7. **Navigation/orbit determination** - The spacecraft orbit is estimated and compared with the expected post-deployment condition.

Commissioning activities

- Verification of spacecraft telemetry and housekeeping data.
- Confirmation of stable power and thermal conditions.
- Functional check of communication system.
- Initial attitude determination and control verification.
- Propulsion system readiness assessment.
- Confirmation of safe operational state.
- Preparation of the first orbit-lowering manoeuvre plan.

Expected outcomes

- Spacecraft is power-positive and thermally stable.
- Communication with the spacecraft is established.
- Spacecraft health status is confirmed.
- Attitude rates and orbit determination are within acceptable limits.
- Propulsion system is confirmed ready for manoeuvre operations.

3.6.3 Phase 2 - Transfer to first science orbit

The transfer to first science orbit phase starts after the spacecraft has completed early operations and commissioning. The objective of this phase is to transfer the spacecraft from the assumed initial lunar staging orbit of 5000 x 500 km to the first science orbit of 100 x 100 km.

Orbit-lowering activities

1. **Initial orbit confirmation** - The actual lunar orbit is determined and compared with the expected staging orbit.

2. **Manoeuvre planning** - The orbit-lowering sequence is prepared based on the current spacecraft state, navigation solution, propulsion status, and available propellant.
3. **Apolune reduction** - The spacecraft gradually lowers the apolune from 5000 km toward 500 km, while maintaining the perilune close to 500 km. The orbit is adjusted to reach approximately 500 x 500 km.
4. **Intermediate orbit checkout at 500 x 500 km** - Before continuing to lower altitudes, the spacecraft performs a final system and orbit verification in the 500 x 500 km orbit.
5. **Transfer to 100 km orbit** - After successful checkout, the spacecraft lowers the orbit from **500 x 500 km** to **100 x 100 km**. The achieved first science orbit is checked before entering the science operations phase.
6. **Readiness review** - Spacecraft health, orbit parameters, power, communication, thermal status, AOCS performance, and propulsion status are verified.

Intermediate 500 x 500 km checkout

The 500 x 500 km orbit is used as an intermediate verification point before entering the low lunar orbit regime. At this altitude, the spacecraft is less affected by the stronger perturbations present at lower lunar altitudes, which makes it suitable for a final operational checkout. This step reduces mission risk by allowing the operations team to verify spacecraft performance, navigation accuracy, and propulsion behaviour before committing to the transfer toward 100 x 100 km.

Expected outcomes

- Final 100 x 100 km orbit achieved.
- Spacecraft is verified and ready to start the science operations phase.

3.6.4 Phase 3 - Science operations with stepwise descent

Phase 3 starts after the spacecraft reaches the first science orbit of **100 x 100 km**. The objective of this phase is to perform in-situ measurements of the lunar dust environment at progressively lower altitudes.

At each science orbit, the spacecraft performs a **30-day measurement interval**. During this time, the payload collects data on dust particle flux, particle properties, temporal variations, and possible short-term increases caused by dust-generating events. After each measurement interval is completed, the spacecraft descends by **10 km** and repeats the same measurement cycle at the next lower orbit.

Science measurement cycle

1. **Target orbit reached** - The spacecraft reaches the selected science altitude.
2. **Payload activation** - The dust measurement payload is activated and checked.
3. **Nominal measurements** - Regular in-situ dust measurements are performed during the 30-day interval.
4. **Event monitoring** - The payload monitors for short-term increases in dust density

or particle flux.

5. **Data storage and downlink** - Science data are stored and downlinked during available communication windows.
6. **Station keeping** - Maintaining the spacecraft altitude within the predefined control band during each science phase to ensure stable measurement conditions.
7. **End of measurement interval** - The 30-day science phase is completed.
8. **Descent to next orbit** - The spacecraft descends by 10 km and the cycle is repeated, until 10 x 10 km orbit is reached

Expected outcome

The spacecraft completes dust measurements at all selected science altitudes from 100 x 100 km down to 10 x 10 km, enabling comparison of the lunar dust environment as a function of orbital altitude.

3.6.5 Phase 4 - End-of-life and disposal

Phase 4 starts after the final science measurements are completed, or earlier if the spacecraft can no longer safely continue nominal operations. The objective of this phase is to terminate science operations and place the spacecraft into a safe end-of-life condition.

End-of-life activities

1. **Final data downlink** - Remaining science data and housekeeping telemetry are downlinked.
2. **Final spacecraft health assessment** - Spacecraft status, remaining power, thermal state, communication capability, and propellant are evaluated.
3. **End-of-mission decision** - The operations team confirms whether the mission can be extended or should be terminated.
4. **Passivation** - Spacecraft systems are placed into a safe final configuration.
5. **Disposal execution** - Depending on the remaining spacecraft capability and need, a disposal strategy is performed.

4. ENVIRONMENTAL ANALYSIS

4.1 Lunar orbital dynamics and perturbation environment

The motion of a spacecraft in lunar orbit is first approximated by the classical two-body problem, where the spacecraft is influenced only by the central gravitational attraction of the Moon. In this ideal case, the Moon is assumed to be spherical and homogeneous, and all other perturbing effects are neglected. The two-body equation of motion can be written as [123]:

$$\ddot{\mathbf{r}} = -\frac{\mu_M}{r^3}\mathbf{\vec{r}} \quad (4.1)$$

where μ_M is the lunar gravitational parameter and $\mathbf{\vec{r}}$ is the spacecraft position vector relative to the Moon. This model is useful for first-order orbit sizing, but it is not sufficient for long-duration lunar orbit analysis because the real lunar environment includes non-spherical lunar gravity, Earth and Sun third-body gravity, and solar radiation pressure .

For this thesis, the spacecraft acceleration can be written in simplified form as:

$$\ddot{\mathbf{r}} = -\frac{\mu_M}{r^3}\mathbf{\vec{r}} + \mathbf{a}_{LG} + \mathbf{a}_{3B} + \mathbf{a}_{SRP} \quad (4.2)$$

where $\mathbf{a}_{LG}$ is the perturbing acceleration due to the non-spherical lunar gravity field, $\mathbf{a}_{3B}$ represents third-body perturbations from the Earth and Sun, and $\mathbf{a}_{SRP}$ is solar radiation pressure [123], [124]. The relative importance of these terms changes with altitude and orbital geometry. Below approximately 750 km, the lunar non-spherical gravity field becomes the main driver of orbit dynamics, while above this altitude Earth third-body perturbations become increasingly important [125].

4.1.1 Lunar non-spherical gravity field

The Moon has a highly non-uniform gravity field, which cannot be represented accurately by a simple spherical model, especially for low lunar orbits. Its internal mass distribution, large impact basins, and mascons create local gravity anomalies that disturb spacecraft motion. The GRAIL mission substantially improved lunar gravity knowledge, initial GRAIL models reached spherical harmonic degree and order 420, while the later GRGM900C solution extended the model to degree and order 900, resolving spatial blocks of approximately 6 km [126].

An example of this effect is shown in radial surface acceleration maps computed at a constant reference radius in Figure 4.1. A solution truncated to degrees 2 through 40 captures the dominant long-wavelength features of the lunar gravity field, but only the broadest anomalies remain visible. When the expansion is extended to much higher degree, such as degrees 2 through 1800, the gravity field becomes substantially more irregular and reveals dense fine-scale structure associated with local and regional mass variations.

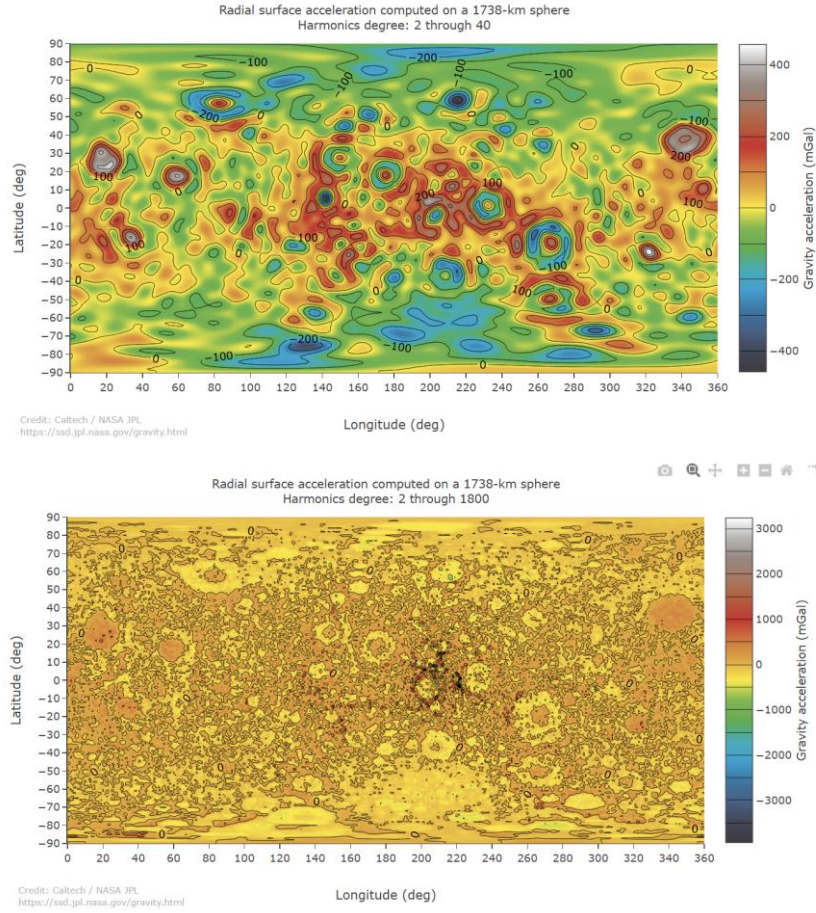


Figure 4.1 Lunar radial gravity acceleration maps for different harmonic resolutions
Top: Harmonics degree - 2 through 40 Bottom: Harmonics degree - 2 through 1800[127].

The gravitational field is strongly affected by non-uniform mass distribution, including mascons, which causes perturbations in the spacecraft orbit. For this reason, the lunar gravitational potential is commonly expressed using a spherical-harmonic expansion:

$$U = \frac{\mu_M}{r} \left[1 + \sum_{l=2}^{\infty} \sum_{m=0}^l \left(\frac{R_{Moon}}{r} \right)^l P_{l,m}(\sin \phi) (C_{l,m} \cos(m\lambda) + S_{l,m} \sin(m\lambda)) \right] \quad (4.3)$$

here U is the lunar gravitational potential, μ is the lunar gravitational parameter, r is the distance of the spacecraft from the centre of the Moon, and R_{Moon} is the lunar reference radius. The indices l and m represent the degree and order of the spherical-harmonic expansion. The terms $P_{l,m}$, $C_{l,m}$, and $S_{l,m}$ are the associated Legendre functions and lunar gravity-field coefficients, while ϕ and λ describe the spacecraft latitude and longitude in the lunar-fixed frame[124], [128].

This perturbation mainly affects the periapsis altitude, apoapsis altitude, eccentricity, inclination, RAAN, and argument of periapsis. In low lunar orbit, variations of several kilometres in altitude are considered typical, and high-order gravity models are required for reliable orbit evolution prediction. The effect becomes stronger as altitude decreases, because the spacecraft becomes more sensitive to local gravity anomalies [128], [129].

For the proposed mission, this is one of the most important perturbations because the spacecraft descends from higher lunar orbits toward very low circular science orbits. The lower the orbit, the more the mission depends on station-keeping to prevent excessive periapsis decay, eccentricity raise or impact risk [130].

4.1.2 Third-body perturbations

Third-body perturbations are caused by the gravitational attraction of external bodies, mainly the Earth and the Sun. In a Moon-centred reference frame, this acceleration contains a direct term acting on the spacecraft and an indirect term caused by the acceleration of the Moon-centred frame itself [124], [128], [131].

For a perturbing body B , the third-body acceleration can be written as:

$$a_{3B} = \mu_B \left[\frac{\mathbf{r}_B - \mathbf{r}}{|\mathbf{r}_B - \mathbf{r}|^3} - \frac{\mathbf{r}_B}{|\mathbf{r}_B|^3} \right] \quad (4.4)$$

where μ_B is the gravitational parameter of the perturbing body, $\mathbf{r}$ is the spacecraft position vector relative to the Moon, and $\mathbf{r}_B$ is the position vector of the perturbing body relative to the Moon. The first term represents the direct gravitational attraction of the perturbing body on the spacecraft, while the second term represents the indirect acceleration of the Moon-centred frame [131].

Their relative importance increases at higher lunar altitudes because the spacecraft is farther from the Moon, making the central lunar gravity and high-order gravity-field terms weaker, while the gravitational influence of external bodies such as the Earth and the Sun becomes comparatively more significant. This is relevant for the initial 500×5000 km orbit phase, where Earth and Sun perturbations are expected to be more visible than in the final very low circular orbit phases. A NASA lunar orbit lifetime study found that, for the analysed cases, Earth perturbation effects were about 170 times larger than Sun perturbation effects, and that third-body effects mainly influence eccentricity, which is a key parameter for orbit lifetime [132].

For a near-circular polar orbit with an initial perilune altitude of 300 km, including Earth-Sun perturbations resulted in a perilune altitude about 20 km higher after 180 days compared with the case where these perturbations were neglected. For a 100 km initial perilune-altitude polar orbit, the lifetime changed from 144 days without Earth-Sun perturbations to 157 days when they were included [132].

4.1.3 Solar radiation pressure

Solar radiation pressure is caused by momentum transfer from solar photons to the spacecraft surface. Its magnitude depends on the spacecraft area-to-mass ratio,

reflectivity, attitude, Sun-spacecraft distance, and whether the spacecraft is illuminated or in eclipse [124], [132], [133].

For preliminary mission analysis, a simplified cannonball SRP model can be written as:

$$\mathbf{a}_{SRP} = P_{SRP} C_R \frac{A}{m} \mathbf{r}_s \quad (4.5)$$

where P_{SRP} is the solar radiation pressure at the spacecraft distance from the Sun, C_R is the reflectivity coefficient, $\frac{A}{m}$ is the spacecraft area-to-mass ratio, $\mathbf{r}_s$ and is the Sun-spacecraft direction unit vector[133].

For compact spacecraft, SRP is usually smaller than lunar gravity harmonics and Earth third-body gravity. However, it can become more important for nanosatellites because they often have high area-to-mass ratios, especially when deployable solar panels, antennas, or large reflective surfaces are used [134]. The NASA lunar orbit lifetime study found that, for a near-circular 300 km lunar orbit, SRP changed the perilune altitude by less than 1 km over 180 days for the analysed spacecraft case; it also notes that SRP becomes more relevant for spacecraft with large cross-sectional area and low mass [132].

For this thesis, SRP is treated as a secondary perturbation for the baseline orbit evolution, but it remains relevant for long-duration propagation and nanosatellite design because the spacecraft area-to-mass ratio can be relatively high.

4.2 Illumination and eclipse analysis

The illumination environment was analysed in STK using the propagated lunar mission trajectory. The objective of this analysis was to identify the periods during which the spacecraft is exposed to direct sunlight, partial shadow (penumbra), or full eclipse (umbra).

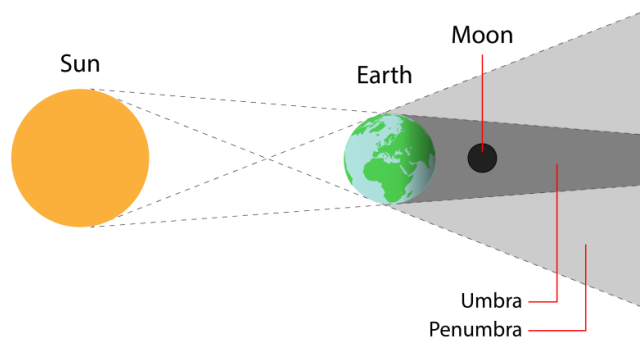


Figure 4.2 Umbra and Penumbra definition

4.2.1 Start-month sensitivity analysis for full mission sequence

The first part of the illumination analysis evaluates the sensitivity of eclipse conditions to the mission start month. The same lunar full mission sequence was propagated for twelve different initial epochs, from January to December. For each case, the STK Eclipse Times output was post-processed by separating events caused by Moon obstruction and Earth obstruction. Moon-obstruction events correspond to the nominal eclipse passages caused by the spacecraft entering the lunar shadow, while Earth-obstruction events represent rarer Sun–Earth–spacecraft alignment cases.

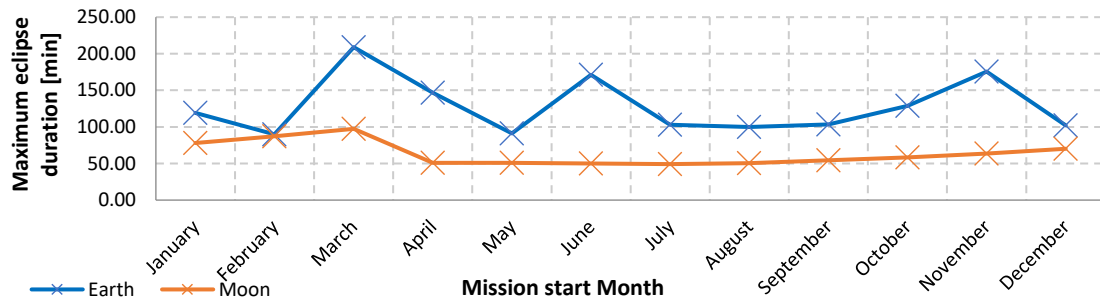


Figure 4.3 Maximum eclipse duration changes with the change of mission start month

As shown in Figure 4.3, the maximum duration of an individual eclipse depends strongly on the selected mission start month. The longest Moon-obstruction eclipse occurs in the March-start case, reaching approximately 97 min, whereas the shortest worst-case Moon-obstruction eclipse occurs in the July-start case, with a duration of approximately 49 min. This confirms that the worst-case lunar eclipse duration is not constant throughout the year but depends on the Sun–Moon–spacecraft geometry at the selected mission epoch.

The Earth-obstruction events show larger maximum durations than the Moon-obstruction events in several start-month cases. The longest Earth-obstruction event occurs for the March-start case, reaching approximately 209 min. However, these events are much less frequent than Moon-obstruction events and should therefore be treated as rare alignment cases rather than as part of the nominal eclipse environment. This trend is shown in Figure 4.6, where the average number of Earth-obstruction events per day is significantly lower than for Moon-obstruction events. Instead, they represent rare long-duration alignment cases that may be relevant for contingency power and thermal margins.

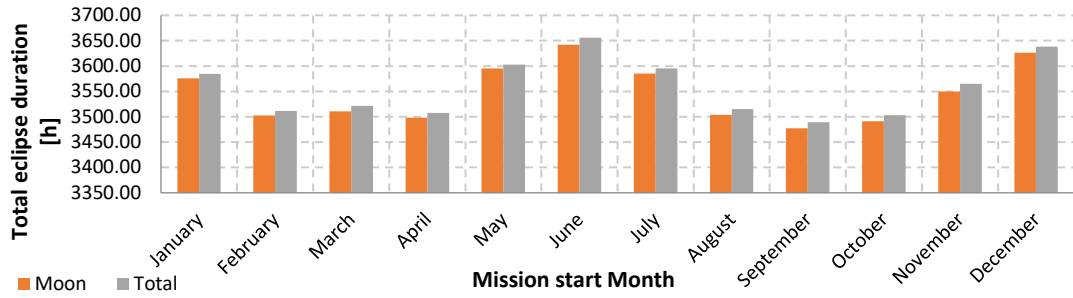


Figure 4.4 Total eclipse duration changes with the change of mission start month

Figure 4.4 shows the cumulative eclipse duration over the analysed mission sequence for each start month. The total eclipse exposure is dominated by Moon-obstruction events. This indicates that, although Earth-obstruction events can produce long individual eclipses, their contribution to the total eclipse time is small compared with the repeated Moon-shadow passages.

The cumulative full mission sequence Moon eclipse time varies only moderately between the different start months, remaining approximately in the range of 3500 to 3650 h over the analysed mission sequences. The highest cumulative values occur for the June and December start cases, while lower values are observed around April and September. This shows that the start month affects not only the maximum single eclipse duration, but also the accumulated eclipse exposure over the mission.

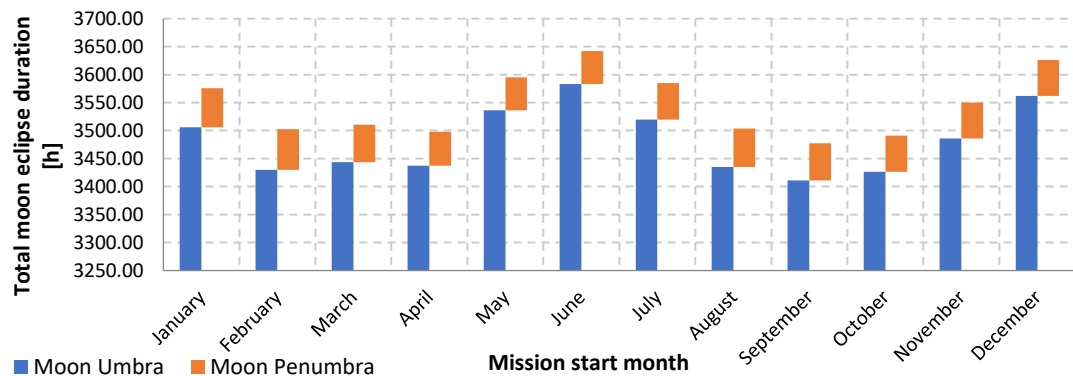


Figure 4.5 Total lunar eclipse duration divided into umbra and penumbra components.

The umbra and penumbra comparison in Figure 4.5 shows that most of the cumulative Moon eclipse time is spent in umbra, while the penumbra contribution is smaller. This is important for the thermal and power analysis because umbra corresponds to complete loss of direct solar illumination, representing the most critical eclipse condition for battery discharge and cold-case thermal behaviour. Penumbra periods are still relevant, but they represent partial solar obstruction rather than full eclipse.

Overall, the start-month sensitivity analysis shows that the selected mission epoch has a significant influence on the worst-case eclipse duration. For nominal lunar-orbit eclipse sizing, the March-start Moon-obstruction case provides the most conservative single-

event duration. Earth-obstruction events should be kept as a separate special case because they can be longer, but they are not representative of the regular lunar eclipse cycle.

4.2.2 Altitude and orbit-phase comparison

The second part of the eclipse analysis evaluates how the eclipse conditions change between different orbit-altitude phases of the mission. Since the Astrogator sequence gradually descends from a higher elliptical lunar orbit to lower circular science orbits, the eclipse events were grouped into orbit phases. This allows the illumination environment to be compared not only by start month, but also by mission phase and altitude.

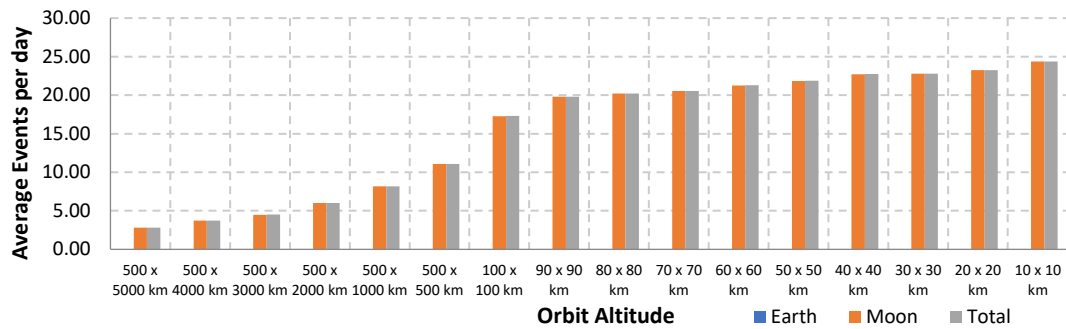


Figure 4.6 Average number of eclipse events per day for different lunar orbit altitudes

As shown in Figure 4.6, the average number of eclipse events per day increases as the spacecraft descends to lower lunar orbits. In the high elliptical phases, such as the 500 x 5000 km and 500 x 4000 km orbits, the spacecraft experiences only a few eclipse events per day. In contrast, the low circular orbit phases below approximately 100 x 100 km show a much higher eclipse frequency, reaching more than 20 eclipse events per day in the lowest orbit buckets.

This behaviour is expected because lower circular orbits have shorter orbital periods and therefore the spacecraft completes more revolutions around the Moon per day. As a result, it has more frequent opportunities to pass through the lunar shadow. The Earth-obstruction contribution remains almost negligible in this comparison, so the total number of eclipse events is dominated by Moon-obstruction events.

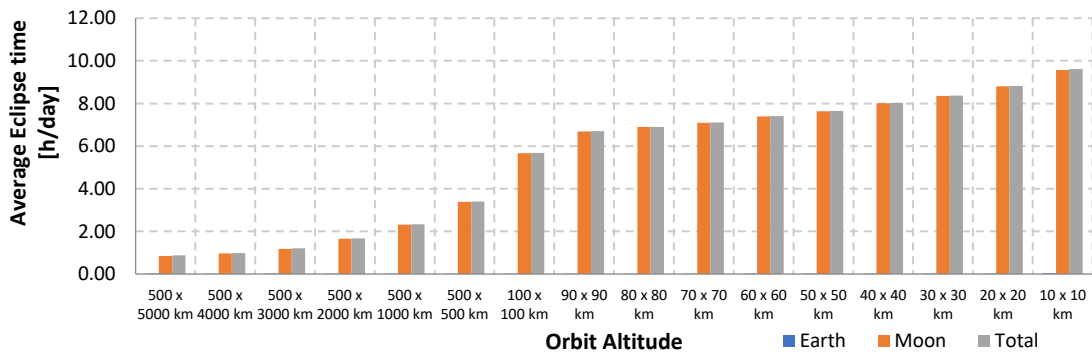


Figure 4.7 Average eclipse time per day for different lunar orbit altitudes.

Figure 4.7 shows the average eclipse time per day for each orbit phase. The same general trend is visible: the eclipse exposure per day increases as the spacecraft descends to lower lunar orbits. In the higher elliptical orbit phases, the average eclipse time is below approximately 1–3 h/day, while in the lowest circular orbit phases it increases to approximately 8–10 h/day.

This result is important for the power and thermal analysis. Although a single eclipse in a low circular orbit may not be the longest eclipse case, the spacecraft experiences eclipse passages much more frequently. Therefore, low-altitude science operations are more demanding in terms of repeated battery cycling and thermal cycling.

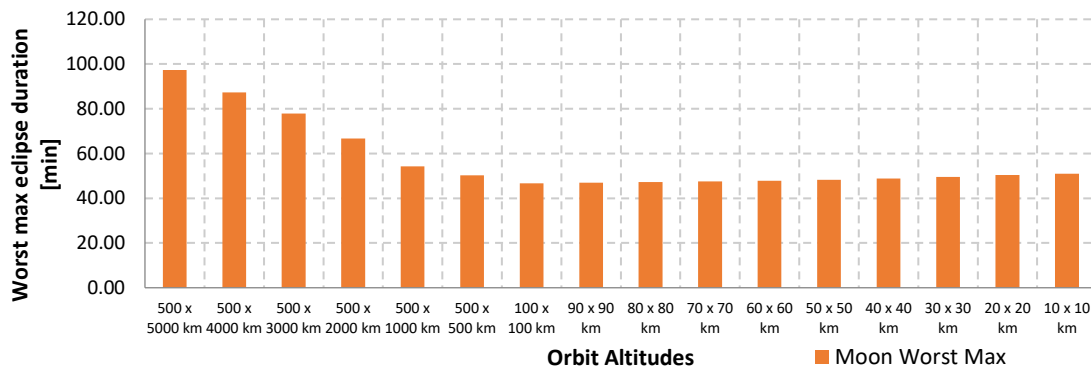


Figure 4.8 Worst-case maximum lunar eclipse duration for different orbit altitudes.

The third graph shows the worst maximum Moon-obstruction eclipse duration for each orbit bucket. Unlike the previous two graphs, the longest individual eclipses occur in the higher elliptical orbit phases. The 500 x 5000 km orbit bucket reaches the largest Moon eclipse duration, approximately 97 min, while the worst-case Moon eclipse duration in the low circular orbit phases remains around 46–51 min.

This means that the eclipse environment has two different worst-case aspects. The higher elliptical orbits are more critical for maximum single-event eclipse duration, while the lower circular orbits are more critical for eclipse frequency and accumulated daily eclipse time. Therefore, both quantities must be considered in the spacecraft power and thermal sizing. The mission therefore cannot be assessed using only one eclipse metric. The maximum eclipse duration is needed for worst-case survival sizing, while the events per day and eclipse time per day are needed to evaluate operational cycling during the science phases.

In addition to the orbit-phases summary, the April-start case was analysed as a full mission time-history. Since the Astrogator sequence includes orbit-lowering phases, the eclipse duration shown in Figure 4.9 reflects both the changing Sun–Moon–spacecraft geometry and the transition between different orbit-altitude phases. The graph therefore provides a mission-level view of eclipse exposure rather than a single-altitude analysis.

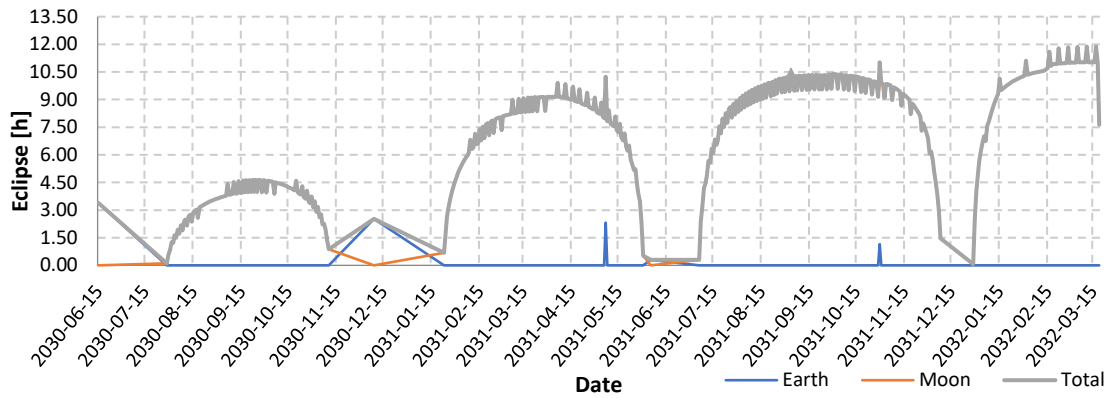


Figure 4.9 Eclipse duration during the analysed mission interval.

The result shows that the eclipse exposure is not constant during the mission. Periods with low eclipse duration are followed by intervals where the accumulated daily eclipse duration increases significantly. This behaviour is caused by the combined effect of the evolving orbital geometry and the descent into lower lunar orbits, where eclipse passages become more frequent. This result is therefore useful for identifying mission phases with increased power and thermal cycling demand.

4.3 Thermal Environment analysis

4.3.1 Thermal environment in lunar orbit

The temperature of a lunar-orbiting spacecraft is affected by both the external lunar environment and the spacecraft configuration. The main drivers are direct solar radiation, lunar-reflected solar radiation, lunar infrared emission, eclipse duration, spacecraft attitude, altitude, exposed surface area, surface optical properties, and internal power dissipation. In vacuum, heat exchange with the external environment occurs mainly through radiation, while heat transfer inside an enclosed small satellite is mainly through conduction between components and structure. Surface coatings, multilayer insulation, radiators, thermal straps, and heaters are therefore commonly used in SmallSat thermal-control design [23].

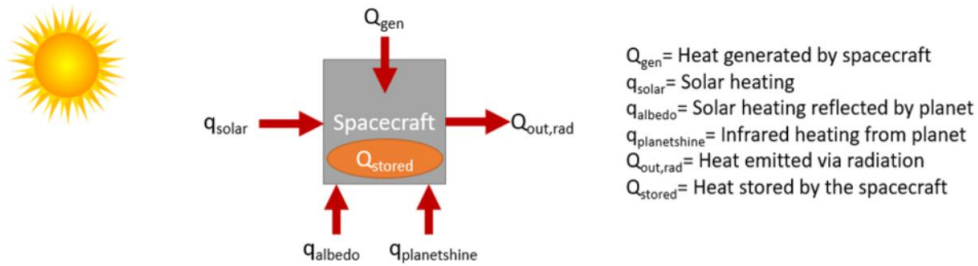


Figure 4.10 Simplified spacecraft thermal-balance model[23].

The dominant external heat input during sunlight is direct solar radiation. At approximately 1 AU, the total solar irradiance is around 1361 W/m², although the exact value varies slightly with solar activity [135]. The absorbed solar heat depends on the projected spacecraft area, attitude, and surface absorptivity. During eclipse, direct solar input is removed, which creates the main cold-case condition for the spacecraft.

The Moon also contributes to the thermal environment through lunar albedo and lunar infrared radiation. Lunar albedo is reflected solar radiation from the surface, while lunar infrared radiation is thermal emission from the heated regolith. The Diviner Lunar Radiometer Experiment on board LRO measures reflected solar and emitted infrared radiation from the Moon in nine spectral channels from 0.3 to 400 μm , which makes it highly relevant for defining the lunar thermal environment [136].

The lunar surface thermal environment is highly variable because the Moon has no dense atmosphere to redistribute heat. Diviner-based global temperature maps show equatorial daytime maximum temperatures of approximately 387–397 K, while pre-sunrise temperatures can decrease to approximately 95 K [T4]. This means that the thermal boundary condition for a lunar orbiter depends strongly on local time, surface properties, latitude, and whether the spacecraft is above illuminated terrain, nightside terrain, or in eclipse.

Table 4.1 Temperatures comparison Moon vs Earth

Object	Moon	Earth
Equator Average Temperature (K)	~206K (390K at noon; ~95 K at midnight)	~299K (303K at noon, ~295K at midnight)
Polar Average Temperature (K)	~98K (outside of shadow)	~256K (North Pole) ~230K (South Pole)
Minimum Temperature (K)	~25K (Hermite Crater)	~184K (Vostok Station, Antarctica)
Maximum Temperature (K)	~410K (small equatorial craters)	~331K (El Aziziya, Libya)

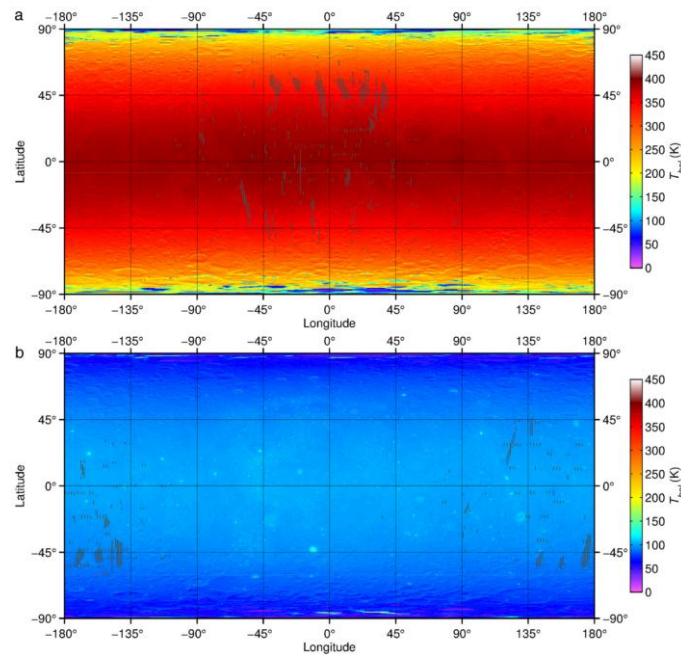


Figure 4.11 Global maximum and minimum surface temperatures of the Moon measured by LRO Diviner[137].

Local thermal anomalies are also important. Permanently shadowed regions near the lunar poles can remain below 110 K, which is cold enough for long-term water-ice stability, and some areas may remain below 30 K [138]. In contrast, shaded regions inside lunar pits can remain near 17 °C, much warmer and more stable than the exposed lunar surface[139]. These examples show that the Moon does not provide a uniform thermal background; instead, the spacecraft may observe very different infrared environments depending on orbit geometry and ground track.

The thermal environment can be divided into hot and cold case. The hot case occurs when the spacecraft is illuminated by the Sun and may also receive heat from lunar albedo, lunar infrared emission, and internal power dissipation. The cold case occurs during eclipse, when direct solar radiation is removed and the spacecraft continues radiating heat to space.

Table 4.2 Hot and Cold case analysis

Case	Representative condition	Environmental values	Main thermal concern
Hot case	Spacecraft in sunlight above illuminated lunar terrain	Solar irradiance at 1 AU: $\approx 1361.6 \text{ W/m}^2$; equatorial lunar daytime surface temperature: $\approx 387\text{--}397 \text{ K}$	Maximum component temperature, radiator sizing, coating selection
Cold case	Spacecraft in eclipse or above cold nightside terrain	Direct solar input: 0 W/m^2 during umbra; pre-sunrise lunar surface temperature: $\approx 95 \text{ K}$; polar permanently shadowed regions can be $<110 \text{ K}$	Battery survival, heater demand, electronics survival temperatures

The STK eclipse results from Section 4.2 are therefore useful as mission-level thermal boundary information. The longest eclipse duration defines the conservative cold-case interval, while the eclipse frequency indicates how often thermal cycling occurs. However, these results do not directly provide spacecraft temperatures; they only define illumination conditions that would be used as inputs to a later thermal model.

4.4 Radiation environment analysis

4.4.1 Lunar radiation sources

The lunar radiation environment is defined by the absence of a dense atmosphere and a global magnetic field. Unlike spacecraft in low Earth orbit, a lunar spacecraft is not continuously protected by Earth's magnetosphere and is therefore exposed to the interplanetary radiation environment. For lunar missions, the dominant primary radiation sources are galactic cosmic rays (GCRs) and solar energetic particles (SEPs). These sources are relevant both for total ionising dose (TID) and for single event effects (SEE) in spacecraft electronics [140], [141], [142].

Table 4.3 Main radiation sources overview

Radiation source	Origin	Typical behaviour	Main relevance for MAIA
Galactic cosmic rays (GCRs)	Outside the Solar System	Continuous background, modulated by solar cycle	SEE, high-LET effects, long-term dose
Solar energetic particles (SEPs)	Solar flares and coronal mass ejections	Sporadic, event-driven, can be intense	Short-term dose increase, proton effects, SEE
Secondary albedo particles	Interaction of GCR/SEP particles with lunar regolith	Depends on altitude, geometry, and surface interaction	Additional local radiation component, especially near surface / LLO
Solar wind particles	Continuous plasma flow from the Sun	Mostly low-energy particles	More relevant for surface charging and space weathering
Earth trapped radiation	Van Allen belts around Earth	Important near Earth, not dominant at the Moon	Mainly relevant during transfer, not the main lunar-orbit source

GCRs are a continuous background radiation source originating outside the Solar System. They consist mainly of high-energy protons, alpha particles, and heavier ions. Although heavy ions represent only a small fraction of the total GCR population, they are important for electronic reliability because their high linear energy transfer (LET) can directly trigger single event upsets in sensitive semiconductor volumes. For this reason, GCRs are especially relevant for long-term SEE analysis of lunar nanosatellite missions.

SEPs are produced by solar activity, such as solar flares and coronal mass ejections. In contrast to GCRs, SEP events are not continuous but can produce short duration increases in particle fluence. These events are often dominated by energetic protons, but heavy ions may also be present. For spacecraft design, SEP events are important because they can significantly increase the accumulated dose over a short time and can also contribute to single event effects [140], [141], [142].

In addition to primary GCR and SEP particles, the lunar surface itself can contribute to the local radiation environment. When energetic particles interact with the lunar regolith, they generate secondary particles, including neutrons, photons, and charged secondaries. This effect is especially relevant for surface assets and low-altitude lunar orbiters, where upward-moving albedo particles from the lunar surface may contribute to the local radiation field [143].

For the MAIA mission, the most relevant radiation sources are therefore GCRs, SEPs, and secondary particles generated by interactions with the lunar surface. Trapped radiation belts, which are important for Earth-orbiting spacecraft, are not treated as a dominant long-term source in lunar orbit. However, the spacecraft may still experience time-dependent variations caused by solar activity and the changing heliospheric environment.

4.4.2 GCR and SEP environment

The GCR and SEP radiation environment for the MAIA mission was analysed using SPENVIS radiation outputs. The galactic cosmic ray environment was evaluated using the ISO 15390 model, while the solar energetic particle environment was evaluated using the SAPPHERE total fluence model. The analysed mission period corresponds to a one-year mission duration from 10 April 2030 to 10 April 2031. The selected trajectory represents the lunar-distance radiation environment, where the spacecraft is outside the main protection of Earth's magnetosphere.

GCRs represent a continuous background radiation source during the mission. The SPENVIS results show that the GCR environment is dominated by protons, followed by helium ions. The proton spectrum remains significant over a wide energy range, confirming that protons form the main continuous particle background. Heavier ion species, such as carbon, oxygen, silicon, and iron, have much lower fluxes than hydrogen and helium. However, despite their lower abundance, these particles remain important because their higher ionisation capability can contribute to single event effects in spacecraft electronics. The detailed GCR proton spectrum and selected ion spectra are provided in Appendix A -.

SEPs represent the accumulated solar-particle exposure over the analysed mission duration. The SAPPHERE total fluence results show that the SEP environment is mainly dominated by lower-energy protons, which are relevant for cumulative radiation effects such as total ionising dose, surface dose, and material degradation. At the same time, the

higher-energy part of the SEP spectrum remains important because these particles can penetrate deeper through spacecraft shielding. The SEP heavy-ion results show lower fluence values compared with protons, but these ions are still relevant for radiation-induced electronic effects due to their stronger ionisation potential. The detailed SEP proton and heavy-ion spectra are provided in Appendix B -.

Overall, the SPENVIS results indicate that the lunar-distance radiation environment is proton-dominated for both the GCR and SEP cases. Protons and helium ions define the main particle background and accumulated fluence, while heavier ions are less frequent but more relevant for high-ionisation effects. For this reason, the detailed particle spectra are used mainly as supporting data for the shielding, TID, LET, and SEU analyses presented in the following sections.

4.4.3 Shielding and TID considerations

The shielding and total ionising dose analysis was performed to evaluate how the accumulated radiation dose decreases with increasing spacecraft shielding thickness. The shielding thickness was expressed as aluminium-equivalent shielding, while the corresponding areal density was calculated in g/cm^2 . The resulting total dose is shown as a function of shielding thickness in Figure 4.12 Total ionising dose and areal density as a function of aluminium shielding thickness.

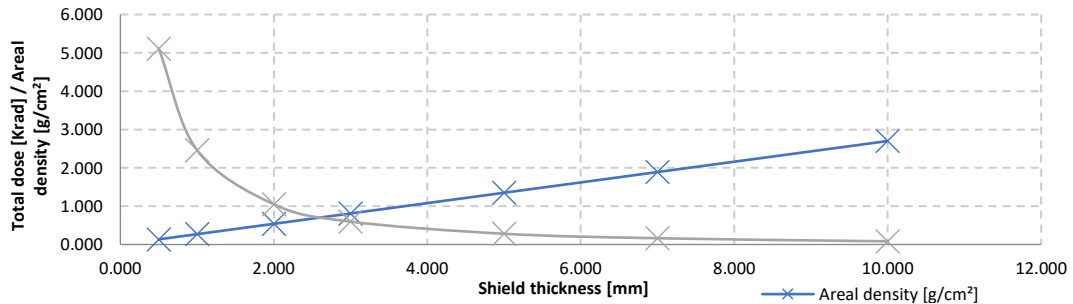


Figure 4.12 Total ionising dose and areal density as a function of aluminium shielding thickness

The total ionising dose decreases strongly as the shielding thickness increases. The reduction is especially significant at low shielding thicknesses. At very small shielding values, the total dose is several krad(Si), while increasing the shielding to around 1–2 mm already reduces the dose substantially. Beyond this point, the total dose continues to decrease, but the improvement becomes more gradual.

The result illustrates the typical spacecraft shielding trade-off. Increasing the aluminium-equivalent shielding thickness reduces the total ionising dose, especially within the first few millimetres. After this initial reduction, the benefit of additional shielding becomes smaller, while the areal density and therefore mass continue to increase. This indicates that moderate shielding can significantly reduce the accumulated dose, but applying thick shielding uniformly across the spacecraft may not be mass-

efficient. A more suitable approach is to combine structural shielding with local protection of radiation-sensitive components, such as the on-board computer, memory devices, payload electronics, and power-management units.

The final shielding selection should therefore be assessed together with the radiation tolerance of the selected components. If the predicted dose remains below the component tolerance with sufficient margin, the shielding level can be considered acceptable. If not, additional local shielding, component relocation, or more radiation-tolerant parts should be considered.

4.4.4 Long term SEU and LET

The long-term single event upset (SEU) and linear energy transfer (LET) analysis was performed to evaluate the sensitivity of the MAIA electronics to high-ionisation particle events. The SPENVIS analysis considered the combined contribution of solar particles and galactic cosmic rays, using SAPHIRE worst-event solar particle fluence and ISO 15390 GCR particles for the ion range from H to U. The spacecraft shielding thickness was set to 0.20 cm aluminium equivalent, and one default silicon device was analysed.

The LET spectrum in Figure 4.13 the spacecraft shielded LET spectrum in silicon. The integral fluence decreases with increasing LET, which means that particles with very high LET are less frequent. However, the high-LET part of the spectrum is still important because these particles can deposit a large amount of charge in sensitive electronic volumes and directly trigger single event effects.

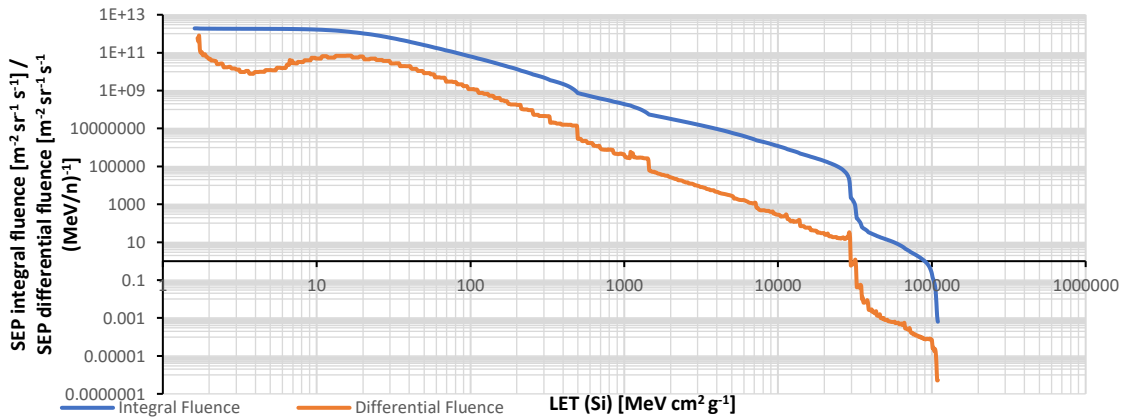


Figure 4.13 Linear energy transfer spectrum for solar energetic particles

The analysed device was a default silicon component with a rectangular sensitive volume of $38.70 \times 38.70 \times 2.00 \mu\text{m}^3$. The heavy-ion upset calculation used the CREME algorithm with a critical charge of 1.13×10^{-2} pC. For proton-induced ionisation, the model parameters were $A = 4.88$ MeV and $B = 7.09$ MeV.

Table 4.4 SEU Spenvis analysis output

Device	Effect	(bit ⁻¹)
DEFAULT	Direct ionization	1.11E+02
	Proton induced ionization	9.52E-02
	Total	1.11E+02

The result (Table 4.4) shows that the calculated SEU contribution is almost completely dominated by direct ionization. The proton-induced contribution is several orders of magnitude lower and represents less than 0.1 % of the total upset result. This is consistent with the LET spectrum, where high-LET ions, although less frequent, are capable of producing direct ionisation in the sensitive volume of the device.

This result indicates that heavy-ion-induced single event effects should be considered an important design driver for the electronics. Mitigation should therefore include measures such as error detection and correction, watchdog timers, memory scrubbing, safe-mode recovery, and careful selection of radiation-tolerant components.

However, the absolute SEU value should be treated as a preliminary screening result, because the analysed component is a generic DEFAULT device and not a confirmed flight component. Final SEU assessment should be repeated using the actual selected electronic components and their component-specific SEU cross-section or radiation-test data.

5. MANOEUVRE ANALYSIS, VALIDATION, AND DELTA-V BUDGET

5.1 Lunar transfer

At the current stage of the mission design, the exact launch vehicle, rideshare opportunity, and lunar delivery conditions are not yet defined. Because of this uncertainty, the Earth-to-Moon transfer is not modelled as a complete launch-to-final-orbit optimisation in this thesis. Instead, the manoeuvre analysis begins from a representative initial lunar orbit of 5000 x 500 km, where the first value represents the apolune altitude and the second value represents the perilune altitude above the lunar surface.

Table 5.1 Initial orbit parameters

Parameter	Perilune altitude	Apolune altitude	Moon mean radius	Perilune radius	Apolune radius	Semi-major axis	Eccentricity	Orbital period
Value	500 km	5000 km	1737.4 km	2237.4 km	6737.4 km	4487.4 km	0.501	7.49 h

The selected orbit is therefore used as a preliminary post-lunar-orbit-insertion staging condition. It does not represent a confirmed injection orbit from a specific launcher, nor a final science orbit. Its purpose is to provide a realistic starting point from which the proposed lunar orbit concept of operations can be analysed. This approach allows the uncertain Earth-to-Moon access problem to be separated from the local lunar manoeuvre problem. The analysis can then focus on the delta-v and time required to transfer from an initially captured lunar orbit to the lower circular and low-altitude operational orbits considered in this thesis.

The use of an elliptical orbit after lunar capture is consistent with several historical lunar missions. Spacecraft are often not inserted directly into their final operational orbit. Instead, they first enter an elliptical capture or commissioning orbit and then perform additional manoeuvres to reduce the orbit altitude. For example, Chandrayaan-1 was initially inserted into an elliptical lunar orbit of approximately 7502x504 km, after which its orbit was gradually reduced to a 100 km polar orbit.

Other lunar missions also demonstrate staged lunar orbit acquisition. Kaguya/SELENE entered an initial polar lunar orbit of approximately 101 x 11,741 km before being lowered to its operational orbit. Lunar Prospector also used an initial capture orbit and subsequent orbit corrections before reaching its 100 km polar mapping orbit. These examples support the use of a representative elliptical staging orbit for preliminary mission analysis.

For this reason, the 5000 x 500 km orbit is treated as a mission analysis boundary condition. It represents the assumed state after lunar capture or external delivery, from

which the spacecraft begins its own orbit-shaping sequence. Once a specific launch opportunity, rideshare interface, or transfer provider is identified, the initial orbit can be updated and the delta-v budget recalculated. Several access architectures could lead to an orbit similar to the assumed 5000 x 500 km staging orbit.

Table 5.2 Possible lunar access architectures

Access route	Category	Basic principle	Positives	Risks / limitations	Heritage / examples
Low-thrust spiral transfer	Spacecraft-led transfer	Spacecraft uses electric propulsion to gradually raise its orbit from Earth orbit/GTO toward lunar distance.	Very propellant-efficient	Long transfer; power-demanding; radiation-belt exposure; complex operations.	SMART-1
Low energy / ballistic transfer	Spacecraft-led transfer	Spacecraft follows a longer multi-body trajectory with lower lunar arrival energy.	Lower capture energy	Long transfer; complex navigation; sensitive trajectory design.	CAPSTONE
Earth phasing-loop transfer	Spacecraft-led / launcher-assisted	Spacecraft is launched into highly elliptical Earth orbits and performs corrections before lunar encounter.	More flexible than direct transfer; useful when launch energy is limited.	Longer than direct transfer; needs multiple manoeuvres and precise navigation.	Chandrayaan-1, LADEE
Direct TLI + spacecraft LOI	Launcher-assisted delivery	Launcher injects spacecraft toward the Moon; spacecraft performs LOI to enter lunar orbit.	Fast and simple transfer architecture.	LOI is critical and can require high instantaneous ΔV .	Lunar Prospector
Artemis / SLS secondary payload	Externally assisted delivery	CubeSat is deployed into deep space/cislunar trajectory and must complete targeting and capture independently.	Provides CubeSat access beyond LEO.	Deployment trajectory may not match target orbit	Artemis CubeSat secondary payloads.
Commercial lunar delivery	Externally assisted delivery	Commercial provider transports payload toward the Moon, as part of a larger lunar mission.	Potentially more frequent opportunities.	Independent lunar orbiter deployment is not guaranteed.	CLPS
OTV / mothercraft delivery	Externally assisted delivery	Larger spacecraft or transfer vehicle carries the CubeSat toward the Moon or releases it near/in lunar orbit.	Reduces CubeSat propulsion burden.	Strong dependence on provider; release orbit may be constrained.	HERA CubeSat concepts

5.2 Manoeuvre capability assessment

The purpose of the manoeuvre capability assessment is to evaluate whether the proposed spacecraft propulsion concept can perform the required lunar orbit-lowering sequence within the assumed operational constraints. The analysis focuses on the transfer from the initial post-capture staging orbit to the lower circular lunar orbits. Station-keeping during the science phases is not included in this chapter and is treated separately in section 5.3.

The theoretical calculation is used as a lower-bound reference, while the STK simulations are used to assess the effect of finite-duration burns, duty cycle limits, and propagation model fidelity.

5.2.1 Analysis inputs and assumptions

The manoeuvre capability assessment was performed from the assumed post-capture lunar staging orbit of 5000 x 500 km. The objective of this analysis is to evaluate the

spacecraft capability to lower its orbit from the initial staging orbit to the selected low-altitude lunar orbits using the assumed propulsion duty-cycle constraints. The analysis compares the theoretical two-body impulsive ΔV estimate with STK simulations performed using the Point Mass and High precision numerical orbit propagator (HPOP) propagators.

Table 5.3 Analysis inputs

Parameter	Assumption / value
Mission start epoch	10.4.2030 10:00:00
Initial analysis orbit	5000 x 500 km
Intermediate orbit	500 x 500 km
Low lunar Science orbit start	100 x 100 km
Final Science analysed orbit	10 x 10 km
Low-altitude descent step	10 km
Theoretical model	Two-body impulsive calculation
Numerical models	STK Point Mass and HPOP
Manoeuvre model in STK	Finite-duration manoeuvres
Spacecraft dry mass	26 kg
Initial fuel mass in STK	1.5 kg
Propulsion Thrust	1.1 mN
Propulsion Specific Impulse	2150 s

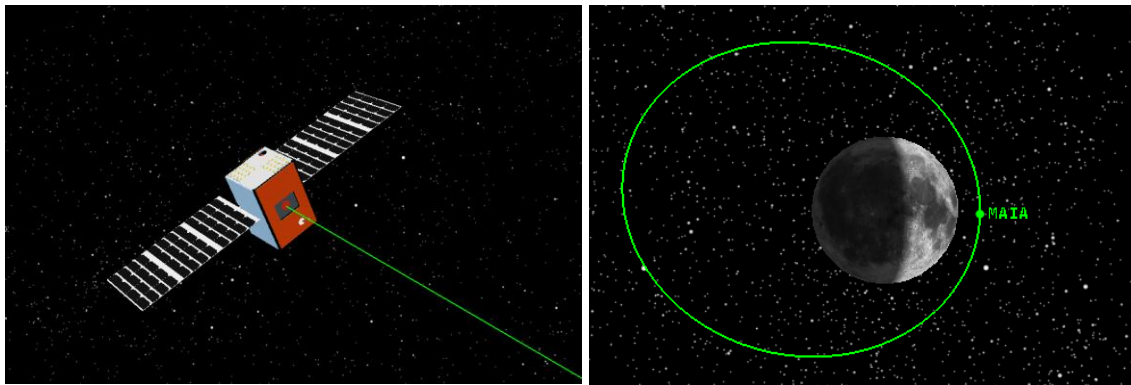


Figure 5.1 MAIA spacecraft visualisation in lunar orbit

5.2.2 Theoretical reference ΔV calculation

The theoretical ΔV reference was calculated using a simplified two-body impulsive manoeuvre model. The Moon was assumed to be the central body, and perturbations, finite burn duration, thrust direction losses, and operational constraints were neglected. Therefore, the calculated values represent an ideal minimum ΔV , which is later compared with the STK Point Mass and HPOP results.

The basic equation used for the theoretical calculation is the vis-viva equation:

$$v = \sqrt{\mu \left(\frac{2}{r} - \frac{1}{a} \right)} \quad (5.1)$$

where v is the orbital velocity at a given point, μ is gravitational parameter of the Moon, r is the distance from the Moon centre at the burn location and a is a semi major axis of the orbit.

For the transfer from the initial elliptical staging orbit toward the 500 x 500 km circular orbit, the apolune is reduced while the perilune remains approximately constant. In this case, the burn is performed near perilune.

For a transfer between two circular orbits, a two-impulse Hohmann transfer was used as the theoretical reference. For orbit lowering, both manoeuvres are retrograde. The first burn lowers the opposite side of the orbit, and the second burn circularises the spacecraft at the lower altitude.

For the initial transfer from 5000 x 500 km to 500 x 500 km, the apolune is reduced while the perilune remains approximately constant. The burn is therefore applied near perilune. For each apolune-lowering step, the required ΔV is calculated as the difference between the perilune velocity before and after the manoeuvre:

$$\Delta V = \left| \sqrt{2\mu \left(\frac{1}{r_p} - \frac{1}{2a_2} \right)} - \sqrt{2\mu \left(\frac{1}{r_p} - \frac{1}{2a_1} \right)} \right| \quad (5.2)$$

where r_p is the perilune radius, a_1 is the semi-major axis before the burn, and a_2 is the semi-major axis after the burn.

For transfers between circular orbits, a two-impulse Hohmann transfer was used. The semi-major axis of the transfer ellipse is:

$$a_t = \frac{r_1 + r_2}{2} \quad (5.3)$$

The two impulses are calculated as:

$$\Delta V_1 = \left| \sqrt{2\mu \left(\frac{1}{r_1} - \frac{1}{2a_t} \right)} - \sqrt{\frac{\mu}{r_1}} \right| \quad (5.4)$$

$$\Delta V_2 = \left| \sqrt{\frac{\mu}{r_2}} - \sqrt{2\mu \left(\frac{1}{r_2} - \frac{1}{2a_t} \right)} \right| \quad (5.5)$$

The total Hohmann transfer ΔV is then:

$$\Delta V_{Hohmann} = \Delta V_1 + \Delta V_2 \quad (5.6)$$

5.2.3 Manoeuvre strategy

The manoeuvre sequence was designed to transfer the spacecraft from the initial lunar orbit to the selected low-altitude orbits. The sequence was divided into three parts: apolune reduction to 500 x 500 km, circular lowering to 100 x 100 km, and stepwise descent from 100 km to 10 km.

The theoretical calculation assumes instantaneous impulsive burns. However, in STK the manoeuvres were modelled as finite-duration burns, which better represents the behaviour of a low-thrust propulsion system. To approximate the ideal impulsive manoeuvre location, the burns were implemented as centre burns, meaning that the midpoint of the finite burn was placed at the desired apsis location. This approach reduces the error caused by long burn durations and keeps the manoeuvre centred around the most efficient point in the orbit.

For orbit lowering, the manoeuvres were performed in the retrograde direction. During the initial elliptical phase, burns near perilune were used to reduce apolune, while burns near apolune were used to correct and maintain the target perilune altitude. This allowed the spacecraft to gradually reduce the orbit from 5000 x 500 km to 500 x 500 km while keeping the perilune close to 500 km.

Table 5.4 Manoeuvres to access the lunar circular scientific orbit

Phase	AB Burn [s]	AB Approx Orbit %	Periapsis B [s]	PB Approx Orbit %	Max Prop Duty cycle %	Orbit Period EOP [s]	Apoapsis EOP [km]	Periapsis EOP [km]
HPOP								
Phase 1	4400	19.46	13800	61.04	80.50	22608	4000	500
Phase 2	4000	21.66	10800	58.48	80.14	18468	3000	500
Phase3	2300	15.70	8000	54.60	70.30	14652	2000	500
Phase 4	1200	10.79	5000	44.95	55.74	11124	1000	500
Phase 5	1000	10.52	5000	52.61	63.13	9504	500	500
POINT MASS								
Phase 1	4600	20.35	13700	60.60	80.94	22608	4000	500
Phase 2	3500	18.95	11400	61.73	80.68	18468	3000	500
Phase3	2300	15.70	8200	55.97	71.66	14652	2000	500
Phase 4	1000	8.99	5600	50.34	59.33	11124	1000	500
Phase 5	1000	10.52	5000	52.61	63.13	9504	500	500

After reaching the 500 x 500 km circular orbit, the spacecraft performed a circular lowering manoeuvre toward 100 x 100 km. This phase used a fixed propulsion duty cycle of 40% of the orbit. The final descent from 100 x 100 km to 10 x 10 km was performed in 10 km steps, with a fixed duty cycle of 25% for each step.

Table 5.5 Summary of the low-thrust descent phases from a 500 km circular lunar orbit to a 100 km circular lunar orbit and stepwise scientific descent

Phase	Burn duration [s]	Orbit prop duty cycle [%]	Orbit Period End [s]	Apoapsis EOP [km]	Periapsis EOP [km]
Phase 6	2826.80	40.00	7067	100	100
Phase 7	1752.50	25.00	7010	90	90
Phase 8	1738.00	25.00	6952	80	80
Phase 9	1723.75	25.00	6895	70	70
Phase 10	1709.50	25.00	6838	60	60
Phase 11	1695.25	25.00	6781	50	50
Phase 12	1681.00	25.00	6724	40	40
Phase 13	1666.75	25.00	6667	30	30
Phase 14	1652.75	25.00	6611	20	20
Phase 15	1638.75	25.00	6555	10	10

The duty cycle was included because the assumed electric propulsion system cannot be operated continuously during the whole orbit. The duty-cycle limit represents operational constraints such as available power, thermal limitations, attitude-control requirements, communication periods, and general spacecraft safety margins.

The same manoeuvre logic was applied in both the Point Mass and HPOP STK cases. The Point Mass model provides a simplified reference propagation, while the HPOP model includes higher-fidelity perturbations including SRP and third body perturbations. Differences between the two results therefore show how the manoeuvre sequence is affected by the dynamical model and finite-burn implementation.

5.2.4 Manoeuvre strategy results

The transfer to the 100 x 100 km orbit represents the most demanding part of the analysed manoeuvre sequence. In the HPOP case, the total ΔV from 5000 x 500 km to 100 x 100 km is approximately 762.33 m/s, while the Point Mass case gives approximately 738.35 m/s. The corresponding transfer durations are approximately 377 days for HPOP and 363 days for Point Mass. The higher ΔV and longer duration in the HPOP case are caused by the higher-fidelity dynamical model and the finite-duration manoeuvre implementation.

Table 5.6 Maneuver strategy results to achieve scientific orbit

Phase	Start orbit [km]	Target orbit [km]	Duration [days]	Burn count	ΔV [m/s]	Final periapsis [km]	Final apoapsis [km]
HPOP							
Phase 1	5000 x 500	4000 x 500	33.74	236	85.87	499.00	4000.86
Phase 2	4000 x 500	3000 x 500	44.50	376	111.87	513.03	3000.90
Phase 3	3000 x 500	2000 x 500	59.28	622	129.57	502.54	2000.90

Phase 4	2000 x 500	1000 x 500	91.28	1228	155.16	552.75	1000.97
Phase 5	1000 x 500	500 x 500	53.02	883	127.06	500.19	500.99
Phase 6	500 x 500	100 x 100	95.18	1	152.81	101.00	101.63
Point Mass							
Phase 1	5000 x 500	4000 x 500	34.51	242	88.37	497.58	4000.98
Phase 2	4000 x 500	3000 x 500	42.32	359	107.06	496.46	3000.95
Phase 3	3000 x 500	2000 x 500	56.51	594	126.05	509.70	2000.95
Phase 4	2000 x 500	1000 x 500	82.59	1119	150.66	503.10	1000.98
Phase 5	1000 x 500	500 x 500	51.68	867	113.21	500.69	500.99
Phase 6	500 x 500	100 x 100	95.36	1	152.99	101.00	101.20

The second part of the sequence lowers the spacecraft from 100 x 100 km to 10 x 10 km in 10 km steps. Compared with the previous transfer, this phase requires a much smaller ΔV . The total ΔV for the low-altitude descent is 38.73 m/s in HPOP and 41.5 m/s in the Point Mass model. The active lowering time is approximately 43.14 days for HPOP and 46.36 days for Point Mass. In addition to this, each altitude step is followed by a 30-day science propagation interval, resulting in 270 days of science propagation over the full low-altitude descent sequence.

Table 5.7 Science orbit maneuver strategy results

Phase	Start orbit	Target orbit	Lowering duration [days]	Science[days]	Phase duration [days]	ΔV [m/s]	Final periapsis [km]	Final apoapsis [km]
HPOP								
Phase 7	100 x 100	90 x 90	4.08	30	34.08	3.65	90.10	92.66
Phase 8	90 x 90	80 x 80	5.06		35.06	4.54	80.10	81.63
Phase 9	80 x 80	70 x 70	4.17		34.17	3.74	70.10	72.89
Phase 10	70 x 70	60 x 60	5.13		35.13	4.6	60.10	60.29
Phase 11	60 x 60	50 x 50	4.95		34.95	4.44	50.10	51.11
Phase 12	50 x 50	40 x 40	5.20		35.20	4.67	40.10	40.43
Phase 13	40 x 40	30 x 30	4.27		34.27	3.84	30.10	32.13
Phase 14	30 x 30	20 x 20	5.26		35.26	4.73	20.10	20.68
Phase 15	20 x 20	10 x 10	5.02		35.02	4.52	10.10	11.29
Point mass								
Phase 7	100 x 100	90 x 90	4.99	30	34.99	4.46	91.00	91.19
Phase 8	90 x 90	80 x 80	5.03		35.03	4.51	81.00	81.15
Phase 9	80 x 80	70 x 70	5.06		35.06	4.53	71.00	71.17
Phase 10	70 x 70	60 x 60	5.11		35.11	4.57	61.00	61.16
Phase 11	60 x 60	50 x 50	5.15		35.15	4.61	51.00	51.16
Phase 12	50 x 50	40 x 40	5.18		35.18	4.64	41.00	41.21
Phase 13	40 x 40	30 x 30	5.25		35.25	4.70	31.00	31.14
Phase 14	30 x 30	20 x 20	5.29		35.29	4.74	21.00	21.10
Phase 15	20 x 20	10 x 10	5.30		35.30	4.75	11.00	11.19

The results show that the final low-altitude descent is not the dominant contributor to the ΔV budget. Most of the required ΔV is used before reaching the 100 x 100 km orbit, especially during the reduction of the initial high elliptical staging orbit. However, the low-altitude phase is operationally important because it contains long science propagation intervals and repeated targeting of progressively lower orbits.

The final achieved periapsis and apoapsis values remain close to the intended target altitudes. The Point Mass case gives more regular final orbit values, as expected from the simplified central-body model. The HPOP case shows slightly larger differences between the target and achieved apoapsis values, especially at lower altitudes, because it includes higher-fidelity lunar dynamics. These deviations are acceptable for the preliminary manoeuvre capability assessment and demonstrate the importance of using HPOP for realistic lunar orbit propagation.

Overall, the complete post-capture orbit-lowering sequence from 5000 x 500 km to 10 x 10 km requires approximately 787.35 m/s in HPOP and 774.08 m/s in Point Mass, excluding lunar orbit insertion and station-keeping.

5.3 Station-keeping strategy

The station-keeping analysis was performed to evaluate whether the selected low lunar science orbits can be maintained for the required 30-day propagation interval after each descent step.

Two cases were analysed. First, each circular science orbit was propagated without station-keeping to evaluate the natural orbital drift. Second, a station-keeping strategy was applied using altitude control bands around each target science orbit.

5.3.1 Natural orbit evolution without station-keeping

The first analysis propagated each science orbit for 30 days without corrective manoeuvres. The purpose was to identify how strongly the orbit evolves under the selected propagation model and whether the spacecraft would remain above the lunar surface.

The uncontrolled propagation results show that the dominant effect is not a uniform decrease of the whole orbit altitude, but an increase in orbital eccentricity. During the 30-day propagation interval, the periapsis decreases while the apoapsis increases. As a result, the initially circular orbit evolves into a more eccentric orbit with a larger altitude spread.

This behaviour is visible in all analysed science orbits. For example, the 100 x 100 km orbit evolves to a minimum periapsis of 78.895 km and a maximum apoapsis of 120.638 km, producing an altitude spread of approximately 40.877 km. At lower altitudes, the same trend becomes more critical. The 30 x 30 km case reaches a minimum periapsis of -3.702 km, while the apoapsis increases to 63.586 km, resulting in surface impact after approximately 14.414 days.

Therefore, the main station-keeping objective is not only to compensate for altitude loss, but to control the growth of eccentricity by preventing the periapsis from decreasing and the apoapsis from increasing beyond the allowed control band.

Table 5.8 Natural drift assessment

Science orbit [km]	Propagation	Min periaps [km]	Max periaps drift [km]	Max apoapsis [km]	Max apopsis drift [km]	Max spread [km]	Impact after [days]
100x100	30 days	78.895	-21.105	120.638	20.638	40.877	NO impact
90 x 90		67.944	-22.056	111.635	21.635	42.772	NO impact
80 x 80		56.896	-23.104	102.698	22.698	44.864	NO impact
70 x 70		45.748	-24.252	93.885	23.885	47.150	NO impact
60 x 60		34.007	-25.993	85.669	25.669	50.817	NO impact
50 x 50		21.741	-28.259	77.995	27.995	55.350	NO impact
40 x 40		9.187	-30.813	70.625	30.625	60.459	NO impact
30 x 30		-3.702	-33.702	63.586	33.586	66.243	14.41
20 x 20		-17.004	-37.004	56.960	36.960	72.828	6.86
10 x 10		-25.460	-35.460	46.996	36.996	71.513	4.14

5.3.2 Station-keeping control strategy

The station-keeping strategy was based on maintaining the spacecraft within a predefined altitude control band around each nominal science orbit. A control band of approximately ± 2 km around the target altitude was used to maintain the spacecraft within the required scientific observation region and to ensure that the measurement conditions remained consistent throughout the mission phase.

When the spacecraft approached the lower boundary of the control band, a raise manoeuvre was applied. When it approached the upper boundary, a lowering manoeuvre was applied. This approach keeps the orbit bounded while allowing small natural oscillations inside the control band.

The manoeuvres were modelled as finite-duration station-keeping burns. The station-keeping performance was evaluated using the total number of burns, the burn fraction per orbit, the achieved minimum and maximum altitudes, and the total station-keeping ΔV over the 30-day interval.

Table 5.9 Station keeping strategy results

Science orbit [km]	Orbit period [s]	Burn/orbit [%]	Control Band	Raise burns	Lower burns	Total burns	SK ΔV [m/s]	Min per [km]	Max apo [km]	Max spread [km]
100 x 100	7067.00	5.66	98 x 102	210	207	417	7.23	98.10	101.90	3.80
90 x 90	7010.00	5.71	88 x 92	220	231	451	7.79	88.09	91.90	3.81
80 x 80	6952.00	5.75	78 x 82	221	222	443	7.76	78.11	81.90	3.79
70 x 70	6895.00	5.80	68 x 72	255	263	518	9.07	68.10	71.90	3.80
60 x 60	6838.00	5.85	58 x 62	265	254	519	9.02	58.10	61.90	3.80
50 x 50	6781.00	5.90	48 x 52	279	333	612	10.70	48.07	51.90	3.83
40 x 40	6724.00	5.95	38 x 42	289	334	623	10.91	38.07	41.90	3.82
30 x 30	6667.00	6.00	28 x 32	340	373	713	12.53	28.07	31.90	3.83
20 x 20	6611.00	6.05	18 x 22	375	494	869	15.43	18.08	21.90	3.82
10 x 10	6555.00	6.10	8 x 12	424	482	906	16.47	8.10	11.90	3.80

The station-keeping results show that the selected control strategy is able to maintain all analysed science orbits within the required altitude band for the full 30-day interval. In each case, the final altitude spread remains close to 3.8 km, corresponding to the selected approximately ± 2 km control band.

The required station-keeping ΔV increases as the orbital altitude decreases. At 100 x 100 km, the 30-day station-keeping requirement is 7.23 m/s, while at 10 x 10 km it increases to 16.47 m/s. This trend is expected because lower lunar orbits are more sensitive to gravitational perturbations and require more frequent correction manoeuvres.

The number of station-keeping burns also increases with decreasing altitude. The 100 x 100 km case requires 417 burns, while the 10 x 10 km case requires 906 burns over 30 days. This indicates that the lowest orbits are not only more demanding in terms of ΔV , but also more operationally intensive.

The total station-keeping ΔV for maintaining all analysed science orbits for one 30-day interval each is $\Delta V_{SK, total} = 106.91 \text{ m/s}$

This value represents the station-keeping contribution for the analysed low-altitude science sequence in the consolidated mission ΔV budget.

5.4 Consolidated mission delta-v budget

The consolidated delta-v budget combines the orbit-lowering manoeuvres from section 5.2 with the station-keeping requirement from section 5.3. The budget is evaluated from the assumed post-capture lunar staging orbit of **5000 x 500 km**. Lunar orbit insertion is not included, because the final launch and lunar access architecture are not yet defined.

The theoretical calculation represents the ideal impulsive lower-bound case. The Moon Point Mass and HPOP results are obtained from STK simulations using finite-duration manoeuvres. The HPOP case is used as the final reference because it includes the higher-fidelity lunar propagation environment.

Table 5.10 Delta-V results comparison table

Manoeuvre phase	Theoretical ΔV [m/s]	Moon Point Mass ΔV [m/s]	HPOP ΔV [m/s]
5000 x 500 km $\rightarrow$ 100 x 100 km	486.37	738.35	762.33
100 x 100 km $\rightarrow$ 10 x 10 km	41.54	41.50	38.73
Subtotal orbit lowering	527.91	779.85	801.06

The comparison shows that the theoretical calculation gives the lowest ΔV because it assumes ideal impulsive manoeuvres. The difference between the theoretical and STK results is mainly caused by the finite-duration low-thrust implementation, especially during the transfer from the initial 5000 x 500 km staging orbit to the 100 x 100 km science orbit. In contrast, the stepwise descent from 100 x 100 km to 10 x 10 km shows good agreement between the theoretical, Point Mass, and HPOP results.

For the final mission budget, the HPOP orbit-lowering result is combined with the station-keeping ΔV from Section 5.3. A 20% preliminary design margin is then applied to the total post-capture ΔV .

Table 5.11 Consolidated final Delta-V budget

Manoeuvre	HPOP [m/s]
Orbit lowering: 5000 x 500 km $\rightarrow$ 100 x 100 km	762.33
Stepwise descent: 100 x 100 km $\rightarrow$ 10 x 10 km	38.73
Station-keeping for science sequence	106.89
Total ΔV	907.94
Total ΔV w 20% margin	1089.529

The final post-capture delta-v requirement is therefore **907.94 m/s** without margin and approximately **1.09 km/s** with a 20% preliminary design margin.

6. PROPULSION TRADE-OFF AND FINAL MISSION DESIGN

6.1 Propulsion requirements

Based on the selected mission concept, derived propulsion requirements were defined to capture the main propulsion-related needs of the spacecraft. These requirements cover lunar access support, orbit-lowering manoeuvres, station-keeping, altitude maintenance, manoeuvre timing, delta-v margin, end-of-life strategy, and compatibility with spacecraft power, thermal, AOCS, and communication constraints. They provide the link between the science orbit requirements and the later propulsion trade-off, manoeuvre capability assessment, station-keeping analysis, and consolidated delta-v budget. Several values remain marked as TBD because they depend on the final launch scenario, detailed duty-cycle constraints, and end-of-life strategy.

Table 6.1 Propulsion requirements

ID	Requirement	Category	Parent requirement
PR-01	The propulsion system shall provide sufficient ΔV for post-deployment trajectory correction and lunar orbit acquisition according to the selected access scenario (transfer correction ΔV TBD; lunar capture / LOI ΔV TBD).	Lunar access	SOR-10
PR-02	The propulsion system shall support planned orbit-lowering manoeuvres between observation phases (ΔV for lowering phases 38.73 TBC m/s; number of phases 10 TBC).	Orbit phasing	MOR-05, SOR-05
PR-03	The propulsion system shall support station keeping during low lunar orbit operations (station-keeping ΔV 106.89 TBC m/s).	Station keeping	SOR-06
PR-04	The propulsion system shall maintain the orbit within defined altitude during each science phase (altitude control ± 2 TBC km).	Orbit maintenance	MOR-04, SOR-02, SOR-06
PR-05	The propulsion system shall allow orbit changes to be timed to support measurements over selected lunar regions and local-time conditions (manoeuvre timing accuracy TBD h).	Observation timing	SOR-02, SOR-03
PR-06	The propulsion budget shall include margin for navigation errors, manoeuvre execution errors, lower-altitude phases, and missed or delayed correction opportunities (ΔV margin ≥ 20 % for phase 0)	ΔV margin	PR-01, PR-02, PR-03, PR-04, PR-05
PR-07	The propulsion system shall support a feasible end-of-life strategy (end-of-life ΔV TBD m/s).	End-of-life	MOR-19
PR-08	The propulsion system shall be compatible with spacecraft power, thermal, attitude-control, and communication constraints during manoeuvre execution (maximum burn duration TBD; maximum power demand 80 TBC W).	System compatibility	SOR07, SOR-08
PR-09	The propulsion system shall support repeated small correction manoeuvres (minimum useful impulse bit TBD N·s).	Manoeuvre flexibility	PR-01, PR-02, PR-03, PR-04

PR-10	The propulsion system shall support manoeuvres without exceeding allowable spacecraft pointing disturbance during science operations (maximum attitude disturbance TBD deg or settling time TBD min).	AOCS compatibility	MOR-16, MOR-17
PR-11	The propulsion system shall provide sufficient total impulse for all planned trajectory correction, orbit insertion, orbit lowering, station keeping, contingency, and disposal manoeuvres (total impulse 29662 TBC N·s).	Total impulse	

6.2 Comparison of candidate propulsion systems

6.2.1 Propulsion Trade-Off analysis

In order to select a suitable propulsion concept for the proposed lunar nanosatellite mission, a trade-off analysis was performed between chemical propulsion and electric propulsion. The purpose of the trade-off is to compare both technologies with respect to their suitability for orbital manoeuvres, station keeping, orbit lowering phases, and long-duration mission operations.

The propulsion system selection has a direct impact on the overall spacecraft design, including mass budget, volume allocation, power system sizing, thermal control, operational planning, and mission flexibility. Therefore, both options were evaluated using several key decision parameters: operations complexity, technological complexity, technological maturity, direct cost, and mass/size constraints. Each parameter was assigned a priority value according to its relevance for the mission and its expected influence on the spacecraft design.

Table 6.2 Propulsion system trade-Off

Priority	Parameter	Chemical	Electric
4	Operations complexity	High	Medium
5	Technological complexity	Medium	Medium
3	tech maturity	High	Medium
1	Direct cost	Low	High
3	Mass/size	High	Low

6.2.1.1 Operations complexity

Operations complexity was assigned a priority of 4, as propulsion manoeuvres are expected to be important during several phases of the mission, including orbit correction, station keeping, and controlled orbit lowering. Chemical propulsion was evaluated as having high operational complexity. This is mainly because chemical systems typically perform short and impulsive burns, which require accurate timing, precise attitude control, and careful manoeuvre planning. In a lunar orbit environment, where

perturbations and station keeping windows can be demanding, limited correction capability after an inaccurate burn may increase operational risk.

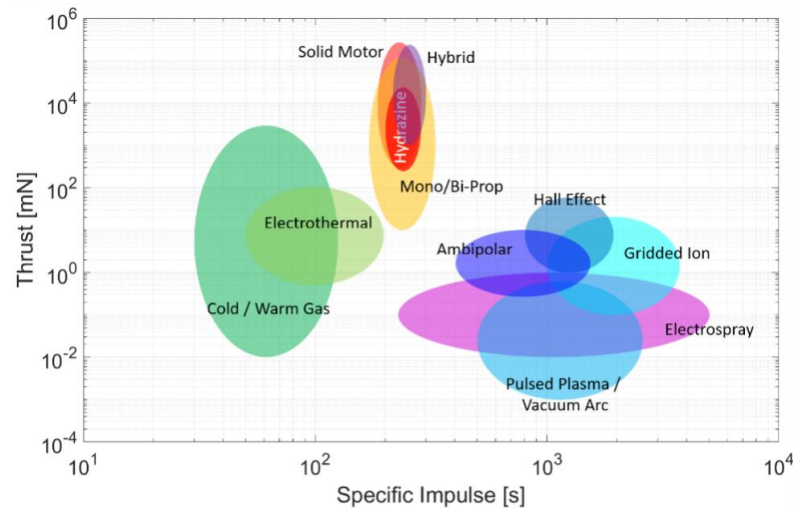


Figure 6.1 Typical thrust and specific-impulse ranges of small-spacecraft propulsion technologies[23].

Electric propulsion was evaluated as having medium operational complexity. Although electric propulsion requires longer burn durations and more careful power management, the low-thrust nature of the system allows manoeuvres to be performed more gradually. This can provide higher flexibility during orbit shaping and station keeping, as manoeuvres may be adjusted or corrected over time. However, long-duration thrusting requires more detailed operational planning and monitoring, especially when combined with power, communication, and attitude constraints.

6.2.1.2 Technological complexity

Technological complexity received the highest priority of 5, because the propulsion system strongly influences several spacecraft subsystems, especially the structure, electrical power subsystem, thermal control, and attitude control system. Both chemical and electric propulsion were evaluated as having medium technological complexity, although for different reasons.

Chemical propulsion introduces challenges related to propellant storage, valves, pressure regulation, leak-tightness, safety, and thermal protection. Depending on the selected architecture, pressurized tanks or hazardous propellants may require additional qualification, handling procedures, and safety measures. The system must also be structurally compatible with short-duration thrust loads and must provide reliable ignition or actuation during critical manoeuvres.

Electric propulsion introduces a different set of design challenges. It usually requires high-voltage electronics, stable power supply, careful grounding, and mitigation of potential electromagnetic interference. The low thrust level also means that orbital manoeuvres are performed over longer time periods, which creates stronger coupling

between propulsion, attitude control, power generation, and mission operations. For a small spacecraft, the available power may become one of the main limiting factors for electric propulsion performance.

6.2.1.3 Technological maturity

Technological maturity was assigned a priority of **3**. While maturity is important for mission reliability, the mission may still accept a medium-maturity technology if it provides significant benefits in terms of delta-V capability, mass efficiency, and mission flexibility.

Chemical propulsion was rated as having high technological maturity. Chemical systems have a long history of use in spacecraft applications and are available in many flight-proven configurations. Their behaviour is generally well understood, and their high-thrust capability makes them suitable for fast manoeuvres.

Electric propulsion was rated as having medium technological maturity in the CubeSat and nanosatellite domain. Electric propulsion has already been demonstrated on small spacecraft, but it is still less standardized than conventional chemical propulsion. Its maturity is increasing, especially for missions requiring efficient orbit raising, station keeping, or long-duration manoeuvring. Nevertheless, the integration of electric propulsion into a small spacecraft still requires careful consideration of power availability, thermal dissipation, and electromagnetic compatibility.

6.2.1.4 Direct cost

Direct cost was assigned the lowest priority of 1, because hardware price alone is not the main driver of the propulsion selection. For this mission, the propulsion system must primarily satisfy the required orbital manoeuvre capability and fit within the spacecraft-level constraints.

Chemical propulsion was evaluated as having low direct cost, mainly due to its wider availability, simpler power interface, and higher level of standardization. Electric propulsion was evaluated as having high direct cost, because electric thrusters often include more complex power processing units, high-voltage electronics, and more specialized components. Although the initial procurement cost of electric propulsion may be higher, this must be balanced against its potential mass efficiency and higher total delta-V capability.

6.2.1.5 Mass and size

Mass and size were assigned a priority of 3, as the spacecraft is strongly volume- and mass-constrained. Chemical propulsion was evaluated as having high mass/size impact. This is mainly due to the need for tanks, valves, feed lines, pressure regulation components, and possible thermal or structural protection. These elements can consume a significant fraction of the available CubeSat volume and may limit the accommodation of other subsystems.

Electric propulsion was evaluated as having low mass/size impact for the propulsion hardware itself. Electric propulsion systems are often compact and can provide high total impulse with relatively low propellant mass due to their high specific impulse. However, this advantage must be considered together with the additional requirements imposed on the electrical power subsystem, solar arrays, batteries, and thermal design. Even so, from a propulsion hardware accommodation perspective, electric propulsion is generally more suitable for a constrained nanosatellite platform.

6.2.1.6 Trade-off analysis conclusion

Based on the trade-off analysis, electric propulsion appears to be the more suitable option for the proposed lunar nanosatellite mission. Chemical propulsion offers clear advantages in terms of maturity, lower direct cost, and high-thrust capability. However, it also introduces higher operational complexity and more demanding mass and volume constraints, which are critical disadvantages for a small spacecraft operating in lunar orbit.

Electric propulsion has disadvantages mainly in terms of higher direct cost, medium technological maturity, and increased dependence on the power subsystem. Nevertheless, it provides important benefits for this mission, especially lower propulsion hardware volume, better propellant efficiency, and greater flexibility for gradual orbit correction, station keeping, and controlled orbit lowering. These characteristics are especially valuable for a mission that may require repeated manoeuvres and careful long-duration orbital management.

Therefore, although chemical propulsion remains attractive for missions requiring fast impulsive manoeuvres, the overall trade-off favours electric propulsion for this mission. Its higher efficiency and better compatibility with long-duration orbital operations make it a more appropriate candidate for the proposed lunar nanosatellite mission, provided that the spacecraft power budget, thermal control, and electromagnetic compatibility are properly addressed during the detailed design phase.

6.3 Final propulsion selection

Following the propulsion technology trade-off, a preliminary market study was performed to identify commercially available or flight-demonstrated electric propulsion systems suitable for a nanosatellite-class lunar mission. The purpose of this market study is not to select the final propulsion unit, but to define realistic candidate systems and compare their approximate performance range, integration needs, and maturity.

Table 6.3 Electric propulsion system market study

Name	Type	Thrust [mN]	Isp [s]	Total Impulse [Ns]	Mass [kg]	Size	Nominal power [W]	Flight heritage	Ref
Exotrail ExoMG™ nano	Hall Effect	1,8	800	<5000	~2	1-2U	60	TRL 9	[144]
Busek BIT-3	RF ION	<1,1	<2150	33 460	<3	180 x 88 x 102mm (1,6U)	75	TRL 9	[145]
ThrustMe NPT30-I2	Gridded Ion / Iodine	1,10	<2400	<5500/ 9500	~1.1 / 1.7	1U/1,5U	65	TRL 9	[146]
ENPULSION NANO	FEED	0.035	1600	4000	~1.5	1U	45	TRL 9	[147]
T4i REGULUS Small/medium	Hall Effect	0.25–0.65	650	3000/ 7000	2.9/4.1	93.8x 95.0x151.0/ 200.0 mm	50	TRL9/ TRL8	[148]
IENAI-Space ATHENA	Electrospray	0,26 per modul	2000	1500	<1per modul	94x 104 x72,6 mm	20	TRL7	[149]
Neumann Drive	Pulsed Arc	236	2000	1000-4500	1,40	96x96x100	50	TRL9	[150]

Based on the propulsion market study and the consolidated delta-v budget, the Busek BIT-3 RF ion propulsion system was selected as the most suitable candidate for the mission. The main selection driver was the required total impulse. The final HPOP delta-v budget with a 20% margin is approximately 1089.53 m/s, which corresponds to a required total impulse for a spacecraft wet mass of approximately 27.5 kg:

$$I_{tot} = m_{s/c} \cdot \Delta V$$

$$I_{tot} = 27.5 \cdot 1089.53 \approx 29962 \text{ Ns}$$

The Busek BIT-3 provides a total impulse of approximately 33 460 Ns, which gives sufficient margin for the analysed post-capture manoeuvre sequence. Other units in the market study provide lower total impulse and are therefore less suitable for the full orbit-lowering and station-keeping concept.

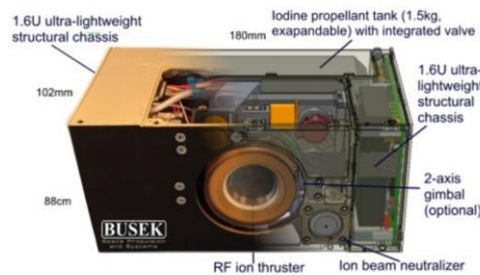


Figure 6.2 Busek BIT-3 propulsion system[145].

The BIT-3 also provides a good balance between thrust, specific impulse, mass, size, and maturity. Its high specific impulse supports efficient long-duration orbit-lowering and station-keeping operations. The selection is also supported by the propulsion system heritage on previous lunar CubeSat missions. The BIT-3 iodine RF ion propulsion system was selected for Lunar IceCube and LunaH-Map, both Artemis I 6U CubeSat missions targeting lunar/cislunar operations. This makes it a relevant reference propulsion option for a nanosatellite mission requiring compact, high-efficiency electric propulsion for lunar orbit manoeuvres.

7. CONCLUSION

This thesis investigated the feasibility of a nanosatellite mission for in-situ measurement of lunar exospheric dust. The work focused on defining the mission objectives, deriving observation and operational requirements, selecting a suitable lunar orbit concept, analysing the lunar environment, assessing the required manoeuvre capability, and performing a propulsion trade-off for the proposed mission.

The selected mission concept is based on a low, nearly circular, high-inclination lunar orbit with science measurements performed at several altitude levels. The analysed post-capture manoeuvre sequence starts from a representative 5000 x 500 km lunar staging orbit, continues to 100 x 100 km, and then descends stepwise down to 10 x 10 km in 10 km altitude increments. The HPOP manoeuvre analysis showed that the orbit-lowering sequence requires approximately 801.06 m/s. In addition, the station-keeping analysis showed that active orbit maintenance is necessary during the low-altitude science phases. Using an approximately ± 2 km control band, all analysed science orbits could be maintained for 30 days each, with a total station-keeping requirement of approximately 106.89 m/s. The consolidated post-capture delta-v budget is therefore 907.94 m/s, or approximately 1.09 km/s with a 20% preliminary margin.

The environmental analysis identified several important constraints for the mission design. The eclipse analysis showed that the maximum lunar eclipse duration depends on the mission start month and orbit phase, with the worst lunar eclipse lasting approximately 97 minutes. The lower circular science orbits produce more frequent eclipses and higher cumulative daily eclipse time, reaching approximately 8–10 hours per day in the lowest orbit phases. This makes the power and thermal design important mission drivers. The radiation analysis showed that the lunar environment is mainly influenced by galactic cosmic rays, solar energetic particles, and secondary particles from lunar surface interactions. The shielding analysis indicated that moderate aluminium shielding can reduce total ionising dose, while single event effects caused by heavy ions remain an important risk for spacecraft electronics.

Several limitations remain in the analysis. The full Earth-to-Moon transfer and lunar orbit insertion were not modelled as the lunar access architecture has not been selected yet, and the study begins from an assumed lunar staging orbit. The thermal analysis was limited to environmental boundary conditions and eclipse evaluation, without a detailed spacecraft thermal model. The power budget was also not closed in detail, even though eclipse duration and electric propulsion operation indicate that power availability may be a major constraint. The radiation analysis used preliminary shielding assumptions and a generic silicon device, so the SEU results should be treated only as a screening-level estimate.

Overall, the results indicate that the proposed mission concept is feasible at a preliminary mission-analysis level, provided that the spacecraft can be delivered to a

suitable lunar staging orbit and that a high-efficiency electric propulsion system is used. The propulsion trade-off showed that electric propulsion is more suitable than chemical propulsion for this concept due to its high specific impulse, compact integration potential, and suitability for repeated low-thrust manoeuvres. Based on the market study, the Busek BIT-3 RF ion propulsion system was selected as the preliminary candidate. Its total impulse of approximately 33 460 Ns provides margin over the estimated required impulse of approximately 29 962 Ns.

Future work related to this thesis should focus on refining the assumptions used in the preliminary feasibility assessment. The lunar access architecture should be defined so that the initial staging orbit and total mission delta-v budget can be updated. A more detailed power and thermal analysis should be performed using the eclipse results, spacecraft attitude assumptions, solar-array sizing, battery capacity, heater demand, and propulsion duty cycle. Finally, the radiation analysis should be updated using the actual selected avionics and payload electronics once component-level radiation data are available.

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SYMBOLS AND ABBREVIATIONS

Abbreviations:

FEEC	Faculty of Electrical Engineering and Communications
BUT	Brno University of Technology
STK	Systems Tool Kit
ESA	European Space Agency
NASA	National Aeronautics and Space Administration
CNSA	China National Space Administration
SLS	Space Launch System
LEO	Low Earth Orbit
GNC	Guidance, Navigation and Control
GNSS	Global Navigation Satellite System
CLPS	Commercial Lunar Payload Services
ISRU	In-Situ Resource Utilization
EVA	Extravehicular Activity
UHF	Ultra High Frequency
X-band	Microwave frequency band used for space communication
ILRS	International Lunar Research Station
LCNS	Lunar Communications and Navigation Services
ELFO	Elliptical Lunar Frozen Orbit
LRO	Lunar Reconnaissance Orbiter
LCROSS	Lunar Crater Observation and Sensing Satellite
GRAIL	Gravity Recovery and Interior Laboratory
LADEE	Lunar Atmosphere and Dust Environment Explorer
SELENE	Selenological and Engineering Explorer
JAXA	Japan Aerospace Exploration Agency
ISRO	Indian Space Research Organisation
CAPSTONE	Cislunar Autonomous Positioning System Technology Operations and Navigation Experiment
NRHO	Near Rectilinear Halo Orbit
DRO	Distant Retrograde Orbit
BLT	Ballistic Lunar Transfer
LDEX	Lunar Dust Experiment
UVS	Ultraviolet and Visible Light Spectrometer
NMS	Neutral Mass Spectrometer
LLCD	Lunar Laser Communications Demonstration
VMMO	Volatile and Mineralogy Mapping Orbiter
LUMIO	Lunar Meteoroid Impacts Observer
LUPEX	Lunar Polar Exploration Mission
MAIA	Multi-Altitude Investigation of the lunar Atmosphere and dust environment
PLD	Payload

MOR	Mission Observation Requirement
SOR	Science Orbit Requirement
ConOps	Concept of Operations
TT&C	Telemetry, Tracking and Command
TBD	To Be Determined
TBC	To Be Confirmed
EPS	Electrical Power System
AOCS	Attitude and Orbit Control System
OBC	On-Board Computer
EOL	End of Life
HPOP	High Precision Orbit Propagator
SRP	Solar Radiation Pressure
RAAN	Right Ascension of the Ascending Node
GCR	Galactic Cosmic Ray
SEP	Solar Energetic Particle
TID	Total Ionising Dose
SEE	Single Event Effect
SEU	Single Event Upset
LET	Linear Energy Transfer
SPENVIS	Space Environment Information System
CREME	Cosmic Ray Effects on MicroElectronics
SAPPHIRE	Solar Accumulated and Peak Proton and Heavy Ion Radiation Environment
ISO	International Organization for Standardization
LLO	Low Lunar Orbit
LOI	Lunar Orbit Insertion
TLI	Trans-Lunar Injection
GTO	Geostationary Transfer Orbit
OTV	Orbital Transfer Vehicle
SK	Station Keeping
EOP	End of Phase
AB	Apoapsis Burn
PB	Periapsis Burn
PR	Propulsion Requirement
RF	Radion Frequency
FEEP	Field-emission electric propulsion
TRL	Technology Readiness Level
EMC	Electromagnetic Compatibility

Symbols:

$\mathbf{r}$	spacecraft position vector relative to the Moon	[m]
r	distance from the Moon centre	[m]
$\ddot{\mathbf{r}}$	spacecraft acceleration vector	[m/s ²]

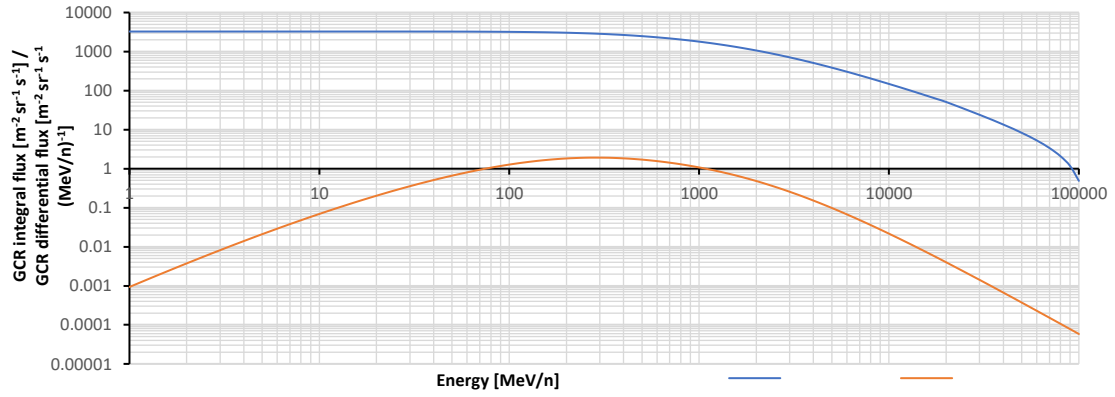
μ_M	lunar gravitational parameter	$[\text{m}^3/\text{s}^2]$
a_{LG}	lunar non-spherical gravity perturbation acceleration	$[\text{m}/\text{s}^2]$
a_{3B}	third-body perturbing acceleration	$[\text{m}/\text{s}^2]$
a_{SRP}	solar radiation pressure acceleration	$[\text{m}/\text{s}^2]$
U	gravitational potential	$[\text{m}^2/\text{s}^2]$
R_{Moon}	lunar reference radius	$[\text{km}]$
l	degree of spherical-harmonic expansion	$[-]$
m	order of spherical-harmonic expansion	$[-]$
$P_{l,m}$	associated Legendre function	$[-]$
$C_{l,m}$	lunar gravity-field cosine coefficient	$[-]$
$S_{l,m}$	lunar gravity-field sine coefficient	$[-]$
ϕ	latitude in the lunar-fixed frame	$[\text{deg}]$
λ	longitude in the lunar-fixed frame	$[\text{deg}]$
μ_B	gravitational parameter of perturbing body	$[\text{m}^3/\text{s}^2]$
r_B	position vector of perturbing body relative to the Moon	$[\text{m}]$
P_{SRP}	solar radiation pressure	$[\text{N}/\text{m}^2]$
C_R	reflectivity coefficient	$[-]$
A/m	spacecraft area-to-mass ratio	$[\text{m}^2/\text{kg}]$
r_s	Sun-spacecraft direction unit vector	$[-]$
v	orbital velocity	$[\text{m}/\text{s}]$
μ	gravitational parameter of the central body	$[\text{m}^3/\text{s}^2]$
r	radius from the central body to spacecraft	$[\text{km}]$
a	semi-major axis	$[\text{km}]$
ΔV	delta-v	$[\text{m}/\text{s}]$
r_p	perilune radius	$[\text{km}]$
a_1	semi-major axis before manoeuvre	$[\text{km}]$
a_2	semi-major axis after manoeuvre	$[\text{km}]$
a_t	semi-major axis of transfer ellipse	$[\text{km}]$
r_1	initial circular orbit radius	$[\text{km}]$
r_2	final circular orbit radius	$[\text{km}]$
ΔV_1	first Hohmann transfer impulse	$[\text{m}/\text{s}]$
ΔV_2	second Hohmann transfer impulse	$[\text{m}/\text{s}]$
ΔV_{SK}	total station-keeping delta-v	$[\text{m}/\text{s}]$
I_{tot}	total impulse	$[\text{N}\cdot\text{s}]$
m	spacecraft mass	$[\text{kg}]$
I_{sp}	specific impulse	$[\text{s}]$
P	power	$[\text{W}]$

LIST OF APPENDICES

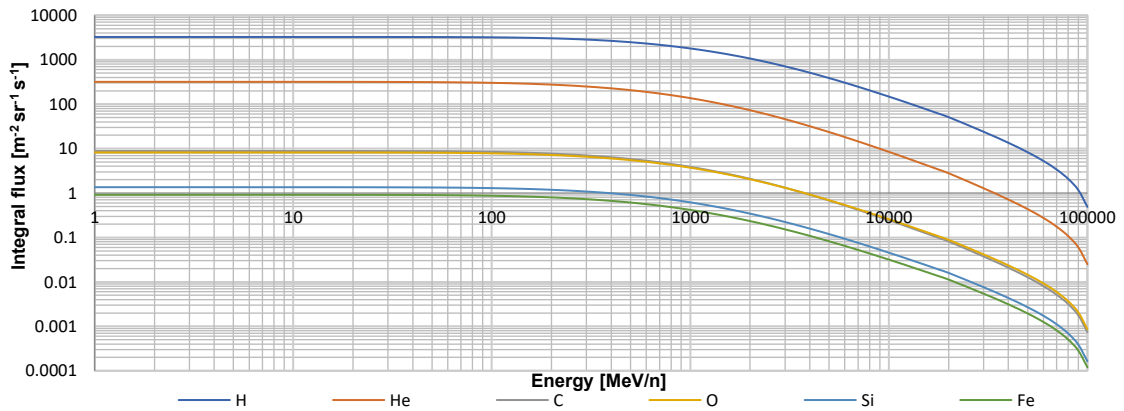
APPENDIX A - SPENVIS RADIATION ANALYSIS GALACTIC COSMIC RAY (GCR) OUTPUT.....	116
APPENDIX B - SPENVIS RADIATION ANALYSIS SOLAR ENERGETIC PARTICLE (SEP) OUTPUT.....	117

Appendix A - Spenvis radiation analysis galactic cosmic ray (GCR) output

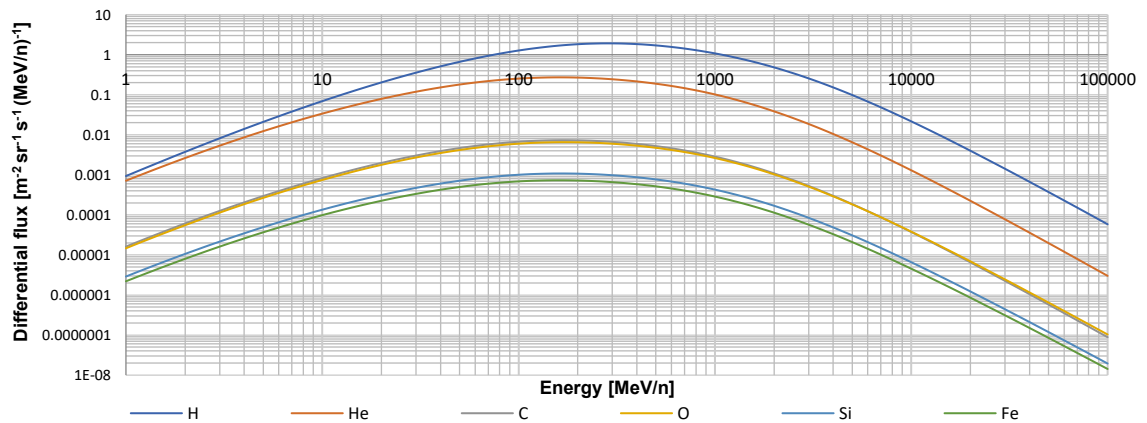
A.1 Mission-averaged galactic cosmic ray proton spectrum.



A.2 Galactic cosmic ray ion spectra for selected particle species - Integral Flux.

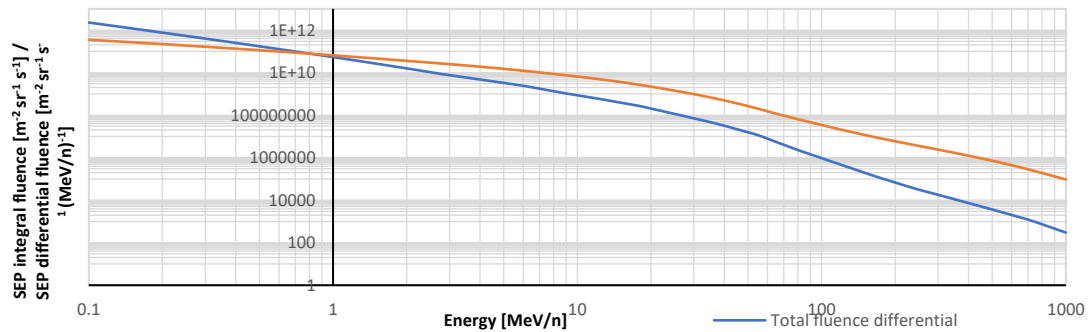


A.3 Galactic cosmic ray ion spectra for selected particle species - Differential Flux.

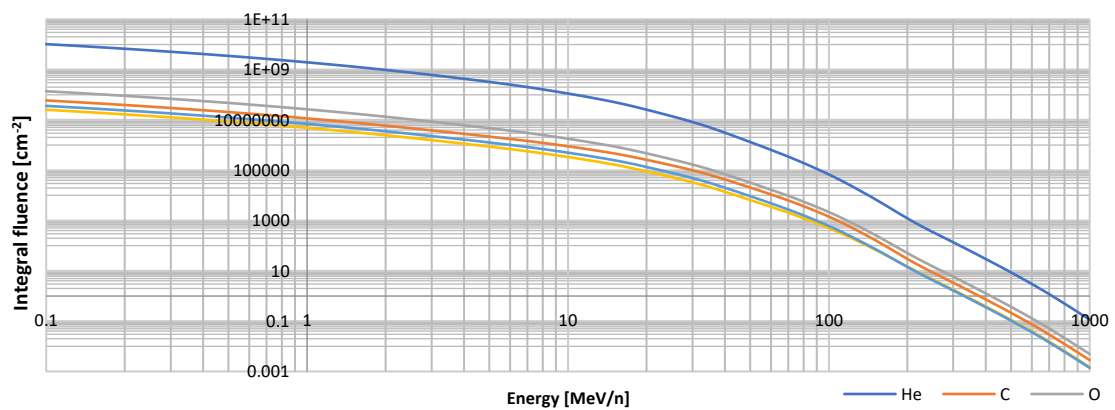


Appendix B - Spenvis radiation analysis solar energetic particle (SEP) output

B.1 Mission-averaged solar energetic particle spectrum.



B.2 Solar energetic particle ion spectra for selected particle species - Integral fluence



B.3 Solar energetic particle ion spectra for selected particle species - Differential fluence

