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From Earth To Moon: Satellite Constellations For Future Lunar Exploration

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Abstract

Humanity's enduring fascination with space exploration dates back to the dawn of the space age in the 1950s and this curiosity has started a series of captivating visions, supported by individuals, space agencies, and private entities.
The International Space Exploration Coordination Group (ISECG) and its consortium of space agencies anticipate that, by the mid-2020s, the establishment of a Lunar Orbital Space Platform Gateway (LOP-G) will announce the beginning of the human journey into the Moon's land. This station will not only facilitate lunar exploration but also promote commercial ventures into deep space.
Pioneering space programs like NASA's ARTEMIS project, which aims to return humans to the lunar surface and lay the foundation stone for the first stable outpost, have acted as captivating catalysts in recent years for numerous feasibility studies to pave the way for also commercial missions on the Earth's satellite.
Initiatives such as the European Space Agency's Moonlight mission have highlighted the need for supplementary lunar surface capabilities, specifically a Lunar Communications and Navigation Service (LCNS). This will simplify data transmission for upcoming missions, particularly those directed to regions hidden from Earth's view, thereby reduce companies missions complexity and promote the development of a lunar-based economy.
The present work investigates the requirements and capability of a lunar satellite constellation designed to meet communication and coverage needs, connecting Earth and strategically significant lunar regions, such as the South Pole and the Equatorial Near-Earth side. These regions are viewed as potential locations for future human return missions, robotic exploration, and the establishment of stable habitats.
Developing such a communication network poses several remarkable challenges. First, the considerable distance between Moon and Earth results in signal propagation delays and losses, making communication a significant challenge that can lead to the necessity of high-power subsystems for the spacecrafts. The latter are more exposed to cosmic radiation than their Earth-based counterparts, which can substantially impact the operational lifetime. Furthermore, the lunar orbits selection is a complex task due to their instability which introduce additional difficulties in the design. In fact, the closer one gets to the Moon, the greater the perturbations caused by its gravitational lumpiness, while at higher altitudes, the gravitational attraction of the Earth predominates, pulling satellites out of their paths. Finally, the launch costs and complexity for big constellations of large satellites in the lunar environment can be prohibitively expensive and intricate.
Possible solutions for the aforementioned problems include the use of specific stable

orbits that minimize the need for orbital maintenance while enabling spacecraft to achieve their communication goals. This approach can reduce fuel requirements, decreases the overall system size, and extends the operational lifetime. Consequently, it becomes feasible to use smaller satellites deployed in a single launch, utilizing them as piggyback payloads of larger exploration missions, creating a more flexible, agile, and less customized system. Along with this, one can consider the possibility to use the expected lunar outpost as relay terminals to communicate with Earth reducing the workload and power requirements for the constellation.

Two main architectures have been conceived and analysed to achieve the objectives proposed by the study. These include constellations in specific Keplerian orbits, containing the minimum number of satellites capable of ensuring nearly real-time communication with lunar sites of interest, while differing in direct communication with Earth. In one case, larger satellites are employed to cover all the necessary links, while in the other, small CubeSat-sized are used, supported in relaying data to Earth by communication terminals placed on the lunar sites.

The work started from an examination on several orbits through researches that helped making a choice and focused the analysis on particular Keplerian ones, known as 'Frozen Orbits'. Using simulations conducted in AGI System Tool Kit (STK) software, it was evaluated the necessary number, types, and behaviour of satellites in those trajectories under the influence of gravitational fields from the central body Moon, Earth and Sun, as well as the impact of solar radiation pressure. The study proceeded assessing satellites of various sizes, originating from existing Earth-based communication constellations with a special attention to their communication subsystems, frequencies, power requirements, mass, and antenna configurations. These satellite formats represent promising solutions to ensure the anticipated performance. Both the analysis performed, converged in a second simulation that allowed to define the two satellite platforms previously highlighted and so the related architectures or Concept of Operations, able to guarantee the necessary communication requirements. Once the possibility of carrying out the mission with a small-sized satellite was determined, the study focused on its development, assessing the main subsystems in terms of required mass and power. A trade-off was made to define which architecture, with its combination of spacecraft numbers, orbits configuration and communication subsystem represented the best option, considering also the ΔV that should be spent to perform station keeping manoeuvres and the required propulsive system. Having thus defined the best mission architecture, the orbit, and all necessary elements, it is finally possible to refine the preliminary design of the satellite.

Acknowledgements

I would like to express my sincere gratitude to Professor Nicole Viola, whose lectures instilled in me a deep interest and curiosity to delve into the aspects of space systems engineering, passions that found full application in this present thesis work.

Over these months, I've had the opportunity to firsthand explore various facets of space mission design and satellite engineering, significantly enriching my academic knowledge at the conclusion of this educational journey.
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Introduction

1.1 Motivations for the study

The upcoming top global priority in space exploration involves establishing a secure and sustainable human presence on the Moon. At present, there are 14 space agencies endorsing the new Global Exploration Roadmap, which represents a shared vision uniting governments and private companies with the goal of expanding human activities from Earth's orbit to the Moon and eventually to Mars.

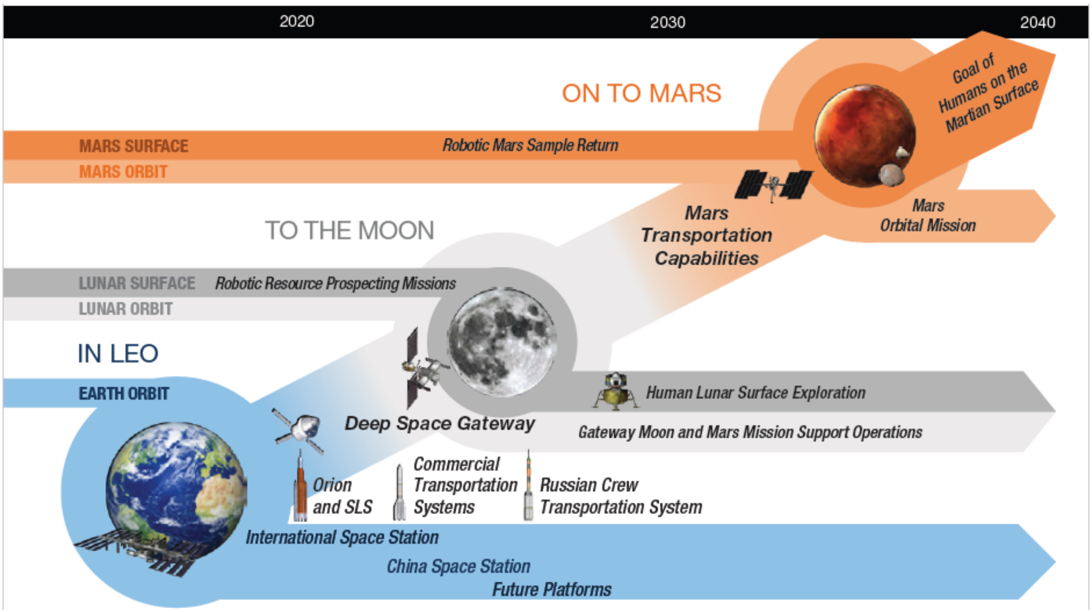


Figure 1.1: The global exploration roadmap: Image Credits [1]

This decade will witness numerous technological demonstrations, unmanned scientific missions, and human Moon landings, all of which will lay the groundwork for the emerging lunar economy in space exploration. The Lunar Gateway, an orbiting space station located in the cis-lunar environment, will provide communication and operational support for activities on the Moon's surface. This support will drive the development of additional infrastructure, including habitats, robots, commercial routes, and transportation, marking the initial stages in the establishment of a lunar colony. Despite the increasing number of missions targeting lunar orbits, there is currently no dedicated communication system that can meet the diverse needs of future lunar surface users. This situation is highly constraining because it necessitates the development of separate, ad-hoc solutions for each mission also limiting the exploration to few large public or private agencies [1].
In order to sustain the increasing demand of operational support with communications and navigation systems, ESA proposed the Moonlight initiative, which consist on the implementation of a single, specialized lunar telecommunication architecture that has the potential to reduce mission design complexities. This is expected to facilitate an increased number of institutional and commercial missions, as it allows them to concentrate exclusively on their primary objectives. With this dedicated system, missions could also reduce significantly their weight and creates additional room for scientific instruments or other payloads on launchers [2].

Figure 1.2: ESA Moonlight mission logo: Image Credits: [2]

The present work focuses on the development of a satellite constellation for lunar orbit communication through a feasibility study of the mission and the preliminary design of the required satellite platform.
In the first chapter, the analysis was devoted to evaluating existing systems within the terrestrial environment to examine their potential adaptability for lunar use. This analysis considered various terrestrial architectures, taking into account the emerging trend in space economy known as "New Space." This approach aims to create smaller-sized satellites using efficient design and production processes, which are competitive compared to traditional construction methods. Additionally, a

detailed analysis of the lunar environment was conducted, exploring the current state of communications and the usable orbits around the Moon.
The second chapter applied the mission design approach to the specific case, outlining requirements, objectives, and constraints through an analysis of potential mission stakeholders. Within this chapter, an in-depth analysis of possible orbital configurations for satellite allocation was conducted, concluding that the "Frozen orbits" are the most suitable choice. Two types of satellite platforms, the 12U cubesat Hemeria IoT and the medium-sized Iridium Next satellite, were considered for preliminary design.
In the third chapter, preliminary analyses were carried out to define the mission statement and identify two feasible architectures. The configuration demonstrating the ability to use a smaller satellite was selected, using the Hemeria cubesat as a reference to define the "Point Design."
The fourth chapter presents the methodologies employed to conduct the analyses and simulations detailed in the subsequent Chapter 5. These analyses cover aspects such as coverage performance, station-keeping, and the trade-off to define the most suitable constellation configuration.
Chapter 5 reports all analyses performed using the Agi STK software, presenting their respective outcomes in multiple steps, from preliminary coverage analyses to orbital perturbation simulations, evaluating gap influences, satellite eclipse times, and finally, station-keeping. The chapter concludes with the trade-off process on constellation configurations to identify the most suitable solution for the mission, and refined the satellite design.

1.2 Telecommunication around Earth

Satellite communication involves utilizing artificial satellites positioned in space to send and receive signals between multiple points on Earth. Among various applications, communications stand out significantly due to the extensive potential and the sheer quantity of spacecraft launched within this category. Telecommunication satellites offer a diverse array of services, notably encompassing television and radio broadcasting, high-speed internet access, mobile broadband, and navigation. Employing satellites for communication offers numerous advantages over ground-based networks, such as broader coverage areas, fast transmission speeds and capacity, consistent service provision, and resilience against natural disasters. Additionally, reliance on terrestrial infrastructure to deliver services to users is reduced [3].
To gain a deeper insight into the significance of communication satellites within the space industry, reference can be made to Fig. 1.3 , which illustrates active satellites and their respective orbits [4].

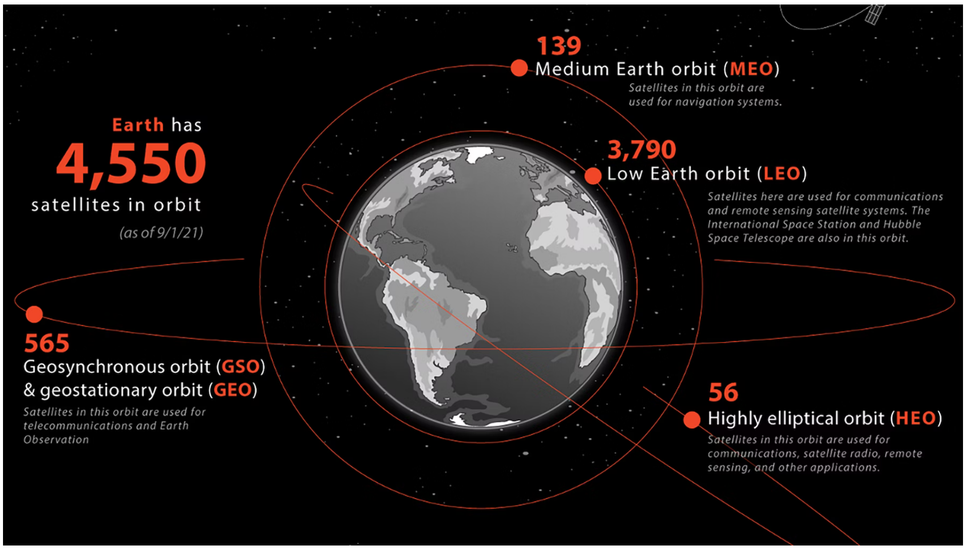


Figure 1.3: Every satellites orbiting Earth. Image Credits: [4]

Within all those 4550 satellites, over 60% serve for communication aim. A breakdown of main purposes is depicted in Fig. 1.4

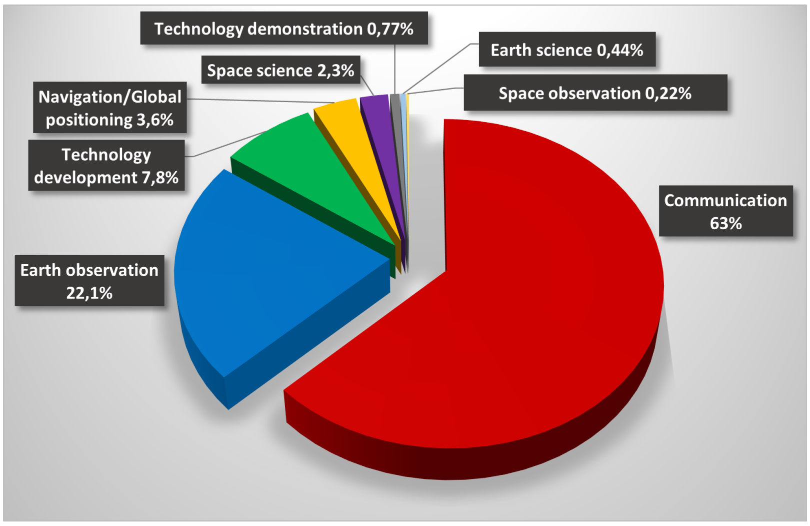


Figure 1.4: Satellite orbiting Earth typology. Data credits: [4]

1.2.1 State of the art on telecommunication constellations around the Earth

To understand the potential benefits and challenges of a lunar satellite constellation for communication, it is important to first examine the current state of the art on

Earth's satellite constellation for telecommunication, with a focus on the orbits used, primary configurations employed and the various types of satellites adopted. Starting from the orbits it can be state that communication spacecrafts are mainly placed in one of three classes based on height above ground and eccentricity [4]:

- Geostationary Orbit (GEO).
- Low Earth orbit (LEO).
- Highly Elliptical Orbit (HEO).

For better clarity, in Fig. 1.5 are depicted all of the mentioned orbits.

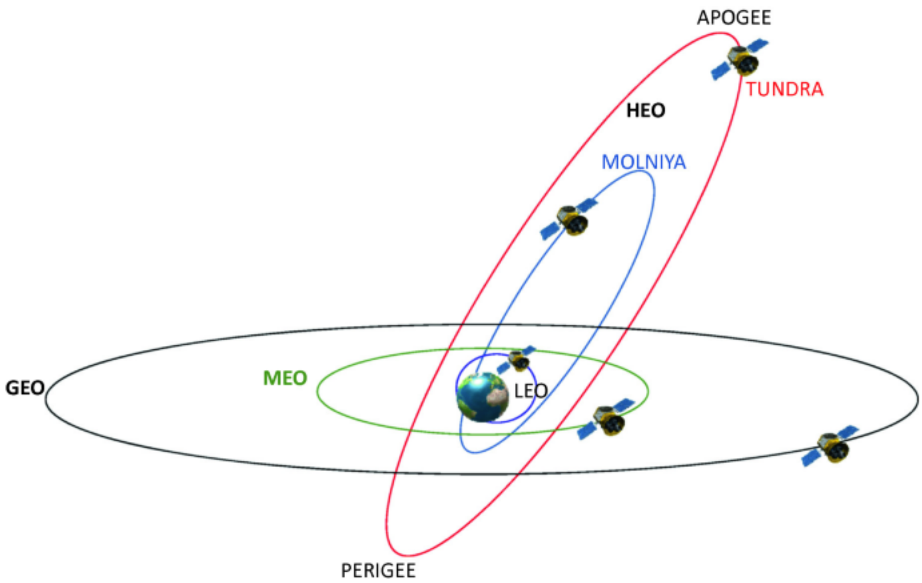


Figure 1.5: Typical satellite orbits for Earth communication: [5]

The decision regarding which orbit is preferable depends on the mission objectives, and once it is determined, it impacts mission costs, lifetime, and payload performance [6].

Geostationary orbits

These have an altitude of approximately 36000 km, letting satellites to moves only above the equator from West to East in a synchronized motion with Earth rotation and seemingly hover above a fixed position. The combination of movement and height ensures that an Earth-based antenna can remain stationary and continuously pointed towards the spacecraft without the need for constant adjustments, furthermore just three space elements can provide near global coverage [7, 8]. The configuration enable to communicate over large distances without the need for intermediate links on ground or with other spacecrafts, however costs for launch and satellite implementation are high, the polar regions can't be covered, and the big distance bring to longer signal delay that is a downside for low-latency communication applications. So those orbits are more likely to be used for meteorology, navigation or one-way data broadcasting, such as television [6, 9]. Within those orbits, notable constellations include:

- **Inmarsat**, with a fleet of 15 spacecraft that deliver a wide range of services, from Internet of Things connectivity to voice and data links for terrestrial, maritime, and airborne communications [10].

- The NASA's **Tracking and Data Relay Satellite (TDRS)** constellation, composed of seven spacecrafts, strategically positioned across the Atlantic, Pacific, and Indian Oceans offer uninterrupted data relay services to more than 25 missions including Hubble Space Telescope, International Space Station, and various Earth-observing missions like Global Precipitation Measurement, Terra, and Aqua [11].

- **Telesat** extensive worldwide constellation offers coverage and connectivity solutions to the requirements of a diverse range of clients, including broadcasting, corporate, telecommunications, and government sectors on a global scale with 14 space elements [12].

Low Earth orbits

Defined by an height ranging from 160 km to 2000 km, in this orbits satellites pass over the surface very fast [7, 13]. The low altitude allow to reduce launch costs, transmitted power required to compensate for propagation path loss and, limiting the Earth view area, to lessen the jamming susceptibility. Unlike GEO they don't need to follow a specified equatorial path, but can be tilted assuming different inclinations, than it is possible to cover all Earth regions. Nevertheless, for ground station facilities is more complex to achieve a good pointing performance, the network control is challenging, and large constellations are inevitable for high link availability. The benefits of a narrower scope of observation and minimal latency, ensuring precise transmission of larger data volumes at significantly faster speeds and with robust signal strength. This versatility makes them particularly suitable for communication purposes, including applications like IoT, and mobile broadband [6, 7, 8].
Since the early '90 several constellations of this category providing global coverage for cellular and personal communications were proposed and those comprises Iridum, Globalstar, Teledesic, Orbcomm, Celestri, Skybridge, Spaceway and so on, but

among them only a few were developed and made operational [14]. Some of the most successful with their peculiar characteristics are depicted in the following:

- **Iridium** constellation was designed to be a digital communication system with a cellular architecture, provides full-Earth coverage with a particular displacement of 77 spacecrafts divided in 7 polar orbits accurately chosen to minimize orbital elements. The 780 km altitude was selected to minimize the need for station keeping due to atmospheric drag experienced at lower heights and the necessity to implement costly radiation hardened electronics at highest elevation [15].

- **Globalstar** is a satellite mobile phone network that enables individuals to make calls from anywhere on the planet within latitudes ranging in ± 70 degrees north to south. It is displaced in a Walker 48-8-1 array, consisting of 48 satellites in low Earth orbit, positioned at an altitude of 1400 km within 8 orbital planes with 6 spacecraft each, at an inclination of 52 degrees relative to the equator [16].

- **Teledesic** was planned to be a system complementary to Earth wireless network, with high data rate, continuous global coverage and fiber-like delay. To achieve those performances, the constellation is composed of 21 circular obits at altitudes between 695 km and 705 km with sun-synchronous inclination of 89.16 degrees, and each plane contains 40 satellites for a total of 840 space elements [17].

In the past few years, there have been new proposals for extensive low Earth orbit constellations aiming to deliver internet access from space. These are driven by enhanced performance derived from the utilization of digital communication payloads, advanced modulation techniques, multi-beam antennas, and frequency reuse strategies [14]. Among those, the most promising architectures are:

- **Iridium NEXT**, with a constellation of 66 components, organized into six orbital planes intersecting at the Earth's poles. Within each plane, there are 11 satellites evenly distributed and these satellites operate in polar orbits at an inclination of 86.4 degrees with an altitude of 780 km [18].

- **Telesat** comprises 117 satellites in two set of orbits. 6 polar at 1000 km altitude with 12 spacecrafts each, 5 circulars inclined at 37.4 degrees and 10 spacecrafts per plane. The two arrangements have different tasks, the first provides general global and polar coverage, while the second covers the most populated regions.

- **OneWeb** comprises 720 orbital elements distributed in 18 circular planes with a height of 1200 km and high inclination of 87 degrees.

- **SpaceX's Starlink** arrangement comprises 4425 elements displaced in several groups of orbits. A core constellation of 1600 spacecrafts equally distributed in 32 orbital planes of 1150 km altitude and 53 degrees inclination. The second include 2825 satellites follows a different distribution having 1600 satellites placed in 32 planes at 1110 km and an inclination of 53.8 degrees, 8 orbital planes contains 50 satellites each at 1130 km and 74 deg, a set of 5 planes every with 75 satellites at 1275 km and an inclination of 81 deg, and the last 450 satellites in 6 orbital planes at 1325 km at 70 deg of inclination.

All the four systems use circular orbits with comparable altitudes, but only OneWeb and Iridium NEXT makes use of a traditional polar configuration, while Telesat and SpaceX utilize a combination of inclined orbits to provide better coverage of most populated areas [14].

Highly elliptical orbits (HEO)

These orbits are designed with large eccentricity and a height fluctuation between perigee and apogee from 1000 km to 42000 km. Those characteristics makes them ideal to provide good coverage on specified regions for long periods and reduced launch cost for satellite insertion. Despite this, several space elements are necessary to reach optimal communication performances, station-keeping and network control are complex and in addition ground stations antennas must be pointed frequently [6, 8, 19]. There are some examples of the utilization of those orbits for communication in high latitude regions like Canada and Russia.

- **Ellipso** represents a network of elliptical orbit satellites designed for voice and data communication. This system was designed to be composed of two distinct constellations, with a total of fourteen satellites. The first constellation, known as Ellipso Borealis, is structured as two rings, each consisting of four satellites following inclined elliptical orbits. The second, Ellipso Concordia, comprises six satellites in a circular equatorial orbit. This arrangement allows for the rapid launch of commercially viable services and permits the allocation of specific satellites to particular regions based on demand, maximizing service efficiency [20].

- **Sirius Satellite Radio** delivers uninterrupted audio, video, and data content to Continental United States and Canada. The original constellation comprises three satellites each in a highly inclined elliptical geosynchronous orbit separated by 120 degree of right ascension of the ascending node. The configuration ensures that there are always two satellites operating above the equator providing services. In this manner the constellation covers users throughout north America service area [21]. The particular HEO uses is called Tundra orbit and it allows satellites to spends most of their time over a chosen target due to a phenomenon known as apogee dwell. This makes them well suited for communications satellites serving high-latitude regions [22]. In addition, those are also used by Japanese **Quasi-Zenith Satellite System** that with four spacecraft enhance the Global Positioning System (GPS) operations over Asia-Oceania regions [23].

- **Molniya** satellites system, whose name is also associated to a particular type of high elliptical orbit. It was specifically engineered to offer communication and remote sensing services in regions characterized by high latitudes. This orbit features a notably high eccentricity, an inclination exceeding 60 degrees, and a period of roughly half a sidereal day [24]. During the early 2000s, a successor to the initial Molniya program emerged in the form of the Meridian constellation, which presently comprises 8 spacecraft positioned in a comparable orbit. These satellites are specifically tailored for military communication purposes with ships and aircraft operating in the Arctic Ocean, as well as ground-based stations in Siberia and the Russian Far East [25].

- **Inmarsat Global Xpress** are two satellites scheduled to be launched in 2023, into a HEO. Once operational, these payloads will mark a pioneering achievement as the world's sole mobile broadband systems exclusively designed for the Arctic region above 60N. They will provide crucial support to sectors like aviation, maritime, and government [8].

From this initial overview of terrestrial constellations for communication, it can be stated that there is no inherently superior configuration to another, but the orbits used vary and primarily differ based on the type of link they need to perform and the goals of the coverage, whether global or in a specific region. Generally, for low-latency communications such as telephony or IoT, low Earth orbits are preferred, as they also help in reducing the size and, consequently, the launch costs. However, they are disadvantaged in terms of coverage as they require a large number of orbiting elements. As it can be seen, the recent trend is to deploy large amount of small spacecrafts in LEO known also as **"Mega-constellations"** [14]. A typical representation is depicted in Fig. 1.6
In general, one can assert that LEO orbits allow for the use of smaller satellites compared to those in GEO. This is primarily due to the shorter distance from the users on the surface, which reduces propagation losses and, therefore, the need for higher power provided by larger bus systems. Furthermore, at lower altitudes, spacecrafts are subject to reduced levels of cosmic radiation, requiring less use of shielded components. LEO satellites are also more affected by atmospheric drag, their lifetime is shorter and so the required fuel for station keeping, while GEO, with an average lifespan of 10 to 15 years need more fuel for position maintenance. Another feature that helps to keep the size of satellites in low orbit smaller, is

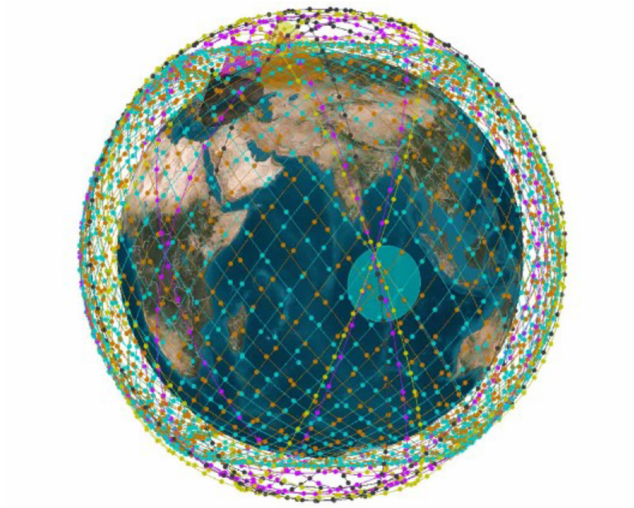


Figure 1.6: Starlink satellites mega constellation pattern. Image Credits: [14]

their decreased coverage footprint, which, due to the lower altitude, needs to be increased using multiple satellites, consequently requiring the use of smaller antennas. However, it's not always the case, as the size of a satellite vary depending on the mission objectives [6].
Focusing on dimension and type of satellite used for communication purposes, the general trend follows a reduction in sizes and masses. This is mainly due to significant technological advancements in satellite design and miniaturization driven by more powerful components, such as integrated circuits and lightweight materials availability, allowing for the creation of more compact and efficient systems. Small satellites have gained prominence and represent a significant element within the emerging **"NewSpace"**.

1.2.2 New space: a new era of exploration and exploitation

This term conveys a shift towards initiatives primarily driven by private businesses and industries, departing from the more conventional model led by government agencies [26].
This represent a shift towards designing smaller satellites characterized by rapid development, modularity, and agility. This approach aims to streamline satellite production, allowing for faster assembly and deployment. These compact spacecraft represent a departure from traditional large-scale models, emphasizing a modular design that facilitates quicker and more flexible construction. By adopting standardized components and assembly techniques akin to automotive manufacturing processes, the objective is to enable mass production on a scale that has previously

been unattainable in the satellite industry. By embracing the principles of efficiency and scalability this trend seeks to revolutionize satellite production, making space exploration and communication more accessible and cost-effective. Several companies today are involved in satellite design following this philosophy, including: Starlink [27], U-Space [28], Loft Orbital [29], Prométhée [30], Endurosat [31], Hemeria [32], and so on. In this direction several constellations has made into orbit for different range of services and applications to satisfy future markets like automotive, entertainment, IoT and Industry 4.0 [33]. The significant expense associated with the design and launch of conventional spacecraft presents a barrier that restricts both the scope of accomplishments and the accessibility of such ventures. small satellites like CubeSats, characterized by their reduced size, modular nature, and cost-effectiveness in construction and launch, are revolutionizing space exploration. Their emergence is transforming the landscape, making space exploration more accessible to a broader spectrum of participants. CubeSats are introducing a realm of fresh opportunities for research and technological advancement to a wide range of stakeholders, including students, institutions of varying sizes, technological innovators, and community-driven initiatives. These miniature satellites hold the potential to operate collectively in constellations, enabling comprehensive observations and detailed analyses of various phenomena. In more intricate missions, clusters of nano-satellites could orbit around a central station, a robust spacecraft capable of managing big computational tasks and transmitting data back to Earth. Simplifying and concentrating on specific functions for each CubeSat could enable more cost-effective deployment, enhanced reliability, and the gradual incorporation of new CubeSats or replacement of malfunctioning units [34]. One can conclude by stating that, in general, new technologies have allowed for the reduction in size and mass of satellites, leading to the development of nanosatellites capable of providing increasingly higher performance. However, the size is primarily determined by the mission requirements that the platform must meet.

1.3 Telecommunication around Moon

1.3.1 General interest on providing telecommunication services around the Moon

In this paragraph will be analysed the key elements that motivates the study and, in the next decades, the realization of a lunar satellite communication constellation. Multiple missions, for various purposes such as observations, scientific analysis and robotic exploration with landers, have been recently launched to the Moon, and many more are scheduled in the next decades [35, 36, 37].
At the current state, all these missions must address challenges not only in terms

of achieving their primary objectives but also in dealing with complications related to the long-distance communication, such as that between Earth and the Moon. This is highly inefficient and limiting since it implies that each mission must rely on a different ad-hoc developed solution. The communication payload plays a crucial role in overall mission success and spacecraft design, influencing power consumption, weight, and consequently dimensions.

The challenges of lunar communication and mission failures

The following presents specific cases of recent missions that highlight how complications in lunar environment communication can lead to a complex design, potentially resulting in mission failures.
On December 2018, the China National Space Administration (CNSA) initiated a mission as part of their Lunar exploration program to achieve the soft landing of the first spacecraft on the far side of the Moon, specifically within the Aitken Basin. The first payload was launched in May and consisted of a relay orbiter named Queqiao, positioned in a designated Earth-Moon L_2 Halo orbit. This was done to ensure a reliable communication link between Earth's command centers and the subsequent elements launched in December 2018: Chang'e 4 and Yutu-2, serving as a lander and a rover, respectively. The mission officially began in January 2019 when the lander deployed Yutu-2 [38]. Consequently, an entire year was required to initiate the exploration, and an additional launch was conducted to deploy the relay satellite, resulting in a more challenging mission and additional costs.
A similar expedition was conducted on July 2023 by Indian Space Research Organization (ISRO), with the Chandrayaan program. The payload was accommodated in a single launcher and included an orbiter connected to the Vikram lander, housing the Pragyan rover. In this scenario, mission data generated by the rover were transmitted to the lander, positioned at the lunar south pole in an area visible to the antennas of the ESA deep space system [39]. Despite the lower complexity compared to the Chinese mission, it should be noted that the payload related to communication with Earth has certainly had its impact on the overall mission.
NASA's mission to map the Moon's south pole water with the Volatiles Investigating Polar Exploration Rover (VIPER) is planned to launch in 2024. The purpose of the mission is to investigate water in shadowed craters with different instruments, so due to its position, keeping connection with Earth is a difficult task. To overcome this problem, NASA have defined a precise position for landing on the edge of a crater, and a specific path for the rover, enabling the maintenance of direct line of sight with Earth [40]. This is another case of a mission that would have benefited and seen a reduction in terms of complexity with the presence of a satellite communication system.
The challenges of providing communication contact with lunar spacecraft can lead

to a complete failure, also on recent missions like Jaxa's Omotenashi and Ispace's Haruto-R, whose crashes on the lunar surface were due to communication problems [41, 42]. Roscosmos too, with its last lunar mission, Luna 25, failed to accomplish the lander touch down due to a maneuvering engine malfunction and a lost of communication contact for over 40 minutes [43].

Future visions for lunar exploration initiatives

The NASA's Artemis program, has set its sights on returning humans to the Moon by 2024. This initiative is designed to accomplish a couple of key objectives. Firstly, it intends to explore new areas of the lunar surface, particularly focusing on the uncharted territory of the South Pole. The primary goal is to conduct in-depth studies of the Moon's surface, resources, and potential for supporting a sustained human presence. Additionally, the Artemis mission is looking to establish a lunar orbiting outpost, the International Deep Space Gateway, in a Near Rectilinear Halo Orbit (NRHO) around Lagrange L_1 point, that serve as a crucial stopover for missions traveling to the lunar surface but also as a scientific laboratory as the International Space Station (ISS) orbiting the Earth.
Moreover, worldwide collaboration is a cornerstone of the Artemis program, emphasize the deep involvement in Moon and deep space exploration of several nations. So NASA is partnering with various international space agencies, such as the European Space Agency (ESA), Canadian Space Agency (CSA), Japan Aerospace Exploration Agency (JAXA) and so on [44].
Crucially, the mission aims not exclusively to land astronauts on the Moon but also to lay the groundwork for a sustainable human presence. This involves utilizing resources found on the Moon and conducting scientific research that will not only benefit future lunar missions but could also inform and support potential prospected expeditions to Mars and beyond [37].

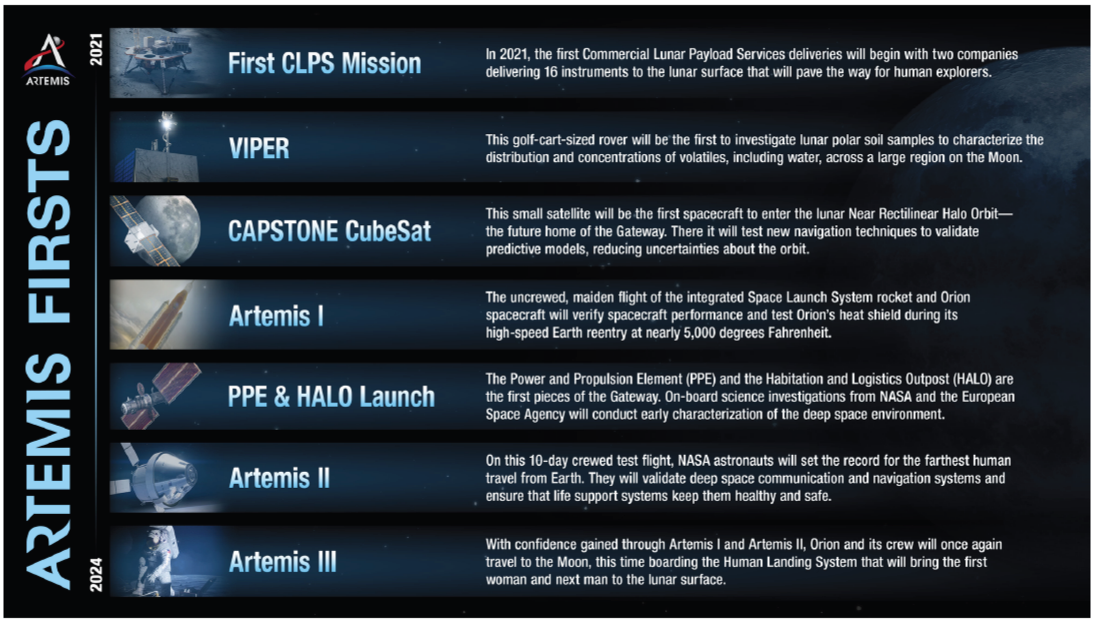


Figure 1.7: Artemis mission program. Image Credits: [37]

The primary components of the initial base camp comprise a Lunar Terrain Vehicle (LTV), Pressurized Rover (PR), Surface Habitat (SH), power systems, and in-situ resource utilization (ISRU) systems. These elements facilitate extended-duration missions, potentially enabling a continuous astronaut presence for prolonged extravehicular activities and robotic operations [45].
Interplanetary communication and navigation stand out as pivotal aspects of lunar exploration. Maintaining constant contact between lunar explorers and mission controllers is essential. Enabling the transmission of not just voice signals but also data packets is critical, allowing for high-definition video streaming, real-time monitoring of structures and equipment via telemetry data, and continuous tracking of astronauts' vital signs. The communication network must extend beyond a mere radio link to Earth, ensuring coverage for all lunar surface operations, including those not visible from Earth. Establishing a comprehensive connection between explorers, stationary equipment, and mobile devices on the lunar terrain becomes imperative, facilitating remote oversight and control of sensors, instruments, and rovers [46].
As previously anticipated, the ESA's Moonlight initiative seeks to establish a unified Lunar Communication and Navigation System (LCNS) to support the present and future missions of both the Artemis partners and private spacecraft. The LCNS will consist of a constellation, the specifics of which are yet to be determined, as well as lunar surface infrastructure. It will be the result of a careful balance between communication requirements, such as maintaining a constant link between

the lunar far side and Earth, and the ongoing investigation into precise navigation needs concerning accuracy, latency, and coverage across the lunar surface. This comprehensive system will enable astronauts to virtually land in any location and navigate while maintaining a continuous connection with Earth's ground control and other lunar users [2, 47].

1.3.2 State of the art on telecommunication constellations around the Moon

After an examination of reasons behind the importance of developing a satellite constellation for communication around the Moon, the need for an in-depth investigation into the current state of the art of such architectures became apparent.
The study's attention was into identifying any already defined satellite platforms, emphasizing their dimensions and subsystem performances, particularly those related to communication payloads, and furthermore on trajectories alternatives to guarantee the coverage capabilities.

ESA's ambitious Moonlight vision to create a network of communications and data relay satellites serving users worldwide, lay the groundwork for Lunar Pathfinder satellite feasibility studies. This satellite hold the potential to furnish navigation data for lunar exploration, similar to Earth based systems like Galileo and GPS. Within the Lunar Pathfinder mission, ESA is hosting two distinct experiments. The first explores the viability of using existing navigation satellites to determine positioning on the Moon. The second experiment involves a space weather monitor, vital for understanding radiation levels around the Moon, a crucial aspect for future human explorations. The spacecraft is designed to offer cost-effective communication services to lunar missions. Weighing 280 kg, this satellite serves as a mission enabler for polar and far-side explorations, which, lacking direct Earth line of sight, would otherwise need their individual communication relay spacecraft. This solution proves more economically viable compared to Direct-to-Earth alternatives and provides a credible substitute for institutional deep-space ground stations. It significantly enhances availability, safety, and data rates for orbiters and missions on the near side.
Under the Moonlight initiative, Surrey Satellite Technology Ltd (SSTL) has been tasked by ESA to lead a Phase A/B1 Study. The initial phase encompasses the Lunar Pathfinder, a singular spacecraft operating in an Elliptical Frozen Orbit (ELFO). Set for launch by the end of 2024 and operational by 2025, this spacecraft is designed for an 8-year service duration. Facilitating various data exchanges between customer assets and ground stations like tracking telemetry and command (TT&C) and payload data, the satellite also enables direct data transfers between lunar assets without involving Earth stations. Employing a Store and Forward

Architecture, the payload stores data until suitable links are available. The spacecraft offers two simultaneous communication channels to lunar assets: S-band and Ultra-High Frequency (UHF), with data relayed back to Earth ground stations in X-band. Operating within an Elliptical Lunar Frozen Orbit, this satellite ensures coverage of the entire lunar surface with extended coverage of the South Polar Region. This orbit not only optimizes servicing missions in this region of geological significance for the presence of permanently shadowed areas containing volatiles, but it also allows for prolonged access to Earth and offers stability for the spacecraft over the mission's duration. The orbit's natural evolution will be monitored and maintained within acceptable parameters to fulfill the objectives effectively [48, 49, 50].
Various studies have been conducted by NASA's Space Communication Architecture Working Group (SCAWG) to provide the necessary Communication and Navigation (C&N) services for space exploration and science missions out to the 2030 time frame operating anywhere in the solar system, with a network service at Earth, Moon and Mars. Focusing on lunar side, SCAWG proposed architectures designed for flexible implementation, capable of adapting to future changes in lunar exploration strategies. These framework aim to meet the evolving exploration and scientific requirements that support missions involving human and robotic exploration on the moon.
SCAWG has outlined various Lunar Relay Elements, which consist of three segments: the space segment, the lunar surface segment, and the supporting Earth ground segment. The lunar surface segment is responsible for providing two-way connectivity and navigation assistance to users near the Lunar Outpost through the Lunar Communications Terminal (LCT). This terminal offers Wide Area Network (WAN) services and serves as an access point to the Lunar Relay System (LRS). On the other hand, the supporting Earth ground segment ensures two-way space-to-ground connectivity with the LRS via the Ground-based Earth Elements (GEE). The evolutionary process is set to commence with the Robotic Lunar Exploration Program (RLEP). This initial phase aims to mitigate technological risks, space-qualify Communication and Navigation (C&N) components, and accumulate operational experience. This valuable experience will be applied to subsequent phases, such as human sorties and missions to establish a human outpost on the Moon. The Lunar Relay architecture is designed to support missions landing anywhere on the Moon. Relay satellites will provide coverage for individual missions and remain available for future scientific endeavors.
The following tables give a brief description of the solutions proposed by SCAWG studies. These spanning circular, polar circular, elliptical, hybrid configuration of circular and halo orbits, hybrid architectures with communication tower at south pole Malapert Mountain [51, 52]. Among these studies there are also analyses performed to define frequency allocations and data rates for the various segments,

both on the lunar surface, towards the relay satellites, and towards Earth. These will be discussed in more detail in the following chapters, as they have been used as mission requirements in order to define the communication system.
Nasa carry out also a design of a lunar relay satellite to orbit in an elliptical lunar polar orbit to provide communications between lunar South Pole assets and the Earth. The project included a complete equipment list, power requirement and configuration design. The satellite configuration follows a conventional box-shaped spacecraft model, featuring solar arrays spanning 1.7 meters for generating 1 kilowatt of power and a mass of more than 1100 kg. It includes a Ka/S band dish with a 1-meter diameter for relay communications to the Lunar Communications Terminal (LCT) and a Q-band dish for establishing communications between the satellite and assets located on Earth [53].

Other research explores methods for optimizing the satellite constellation to meet the future requirements of lunar exploration. This involves examining communication architectures of terrestrial satellites as models to eventually create a comprehensive network covering the entire lunar surface. This particular study focused on developing and assessing potential architecture solutions for lunar communication and navigation. The findings identified two main lunar orbit setups deemed optimal for designing a satellite constellation: lunar frozen orbits and lunar halo orbits. Various configurations of two and three-satellite constellations were devised within these orbit patterns. These included constellations solely in lunar frozen orbits, constellations orbiting around the L1 and L2 Lagrangian points in lunar halo orbits, as well as hybrid constellations combining satellites from both orbit types [54].
The Korea Aerospace Research Institute (KARI) collaborated with NASA on a shared investigation aimed at establishing a space communications architecture. This framework aims to facilitate the integration of communication and navigation services offered by various entities, including the Deep Space Network (DSN), the Korea DSN (KDSN), a possible lunar relay, and the Korea Pathfinder Lunar Orbiter (KPLO). The paper identify a constellation of circular and elliptical frozen orbits, provides Moon global coverage and reduce the need for station keeping maneuvers. Furthermore by assuming specific parameters for the satellite communication payload, a thorough link analysis was conducted across S-, X-, and UHF bands between the lunar relay orbiter and the lunar surface. Additionally, connectivity with the Earth DSN using Ka-band was examined. The evaluation involved assessing the data rates for both forward and return links, taking into account distances and system performances [55].
The Queqiao relay satellite, part of the Chang'e-4 Chinese mission, is the first satellite specifically designed to facilitate data relay from a lander situated on the far side of the Moon. Positioned in a halo orbit around the Earth-Moon libration

point L_2, it ensures continuous communication coverage. This orbit choice allows constant communication but involves longer transmission distances, necessitating a high-power system and large antenna aperture. The spacecraft utilized a 4.2-meter antenna for various links in S-, X-, and Ka-bands, boasting an average power of 780 Watts and weighing 450 kg. Due to the unstable dynamics inherent in orbits around Earth-Moon libration points, potential navigation maneuvers and execution errors could cause Queqiao's actual trajectory to deviate from its intended path over time. As a countermeasure, a large hydrazine blow-down propulsion system was implemented. This system highlights the challenges associated with these orbits, despite their advantageous coverage capability [56]. Queqiao-1 served as a precursor for Queqiao-2, expected to launch in 2024, to support China's Chang'e 6, 7, and 8 lunar missions. The upgraded Queqiao-2 will operate within an elliptical frozen orbit, instead of the L2 halo orbit. Additionally, it will carry two smaller communication satellites, Tiandu-1 and Tiandu-2, intended to validate the technical aspects of the lunar communication and navigation constellation based on Queqiao technology [57].

1.3.3 Behaviour of artificial satellite trajectories around the Moon

The latest research conducted to appropriately start the subsequent analyses pertains to the types of orbits that can be employed for a lunar communication constellation. The attention was on identifying the proposed orbits considering their behavior under the influence of gravitational perturbations and the guaranteed performances in communication coverage.
Numerous analyses have been conducted to study the orbital dynamics of artificial satellites around the Moon, revealing the unique behavior exhibited by lunar orbits. These are notably unstable due to the Moon's irregular gravitational field as expressed in [58, 59]. If a satellite orbits too close to the lunar surface, the highly non-spherical gravitational field caused by the Moon's uneven mass distribution may lead to an increase in the satellite's eccentricity. Over a few months, this could imply the satellite's orbit become highly elliptical, potentially resulting in an impact on the lunar surface [60, 61].
Conversely, if the satellite maintains a distance from the Moon greater than 750 km, the gravitational influence from Earth prevails and pull the spacecraft away, causing it to veer off from the Moon's orbit entirely [62].

These complexities pose significant challenges in maintaining a satellite in an orbit resembling those typically used for Earth's communication constellations. Because of this, the primary solutions found in the literature for communication purposes, focuses on two main types:

1. Keplerian elliptical **Frozen Orbits**.

2. Non-Keplerian **Halo Orbits**.

1.3.4 Frozen orbits about the Moon

A frozen orbit is described as an orbital path where there is minimal change in the orbital eccentricity and argument of periapsis, achieved by meticulously choosing specific orbital parameters. This orbit maintains constant average values for the semi-major axis (a), eccentricity (e), and inclination (i). Ideally these orbits exhibits a periodic oscillation of the orbital elements, makes them perfect to minimize the need for maintenance [63]. Satellites placed in frozen orbits have apsidal lines that remain constant vectors within planet-centered inertial frames, ensuring a stable argument of periapsis (ω) [64, 65]. In [62, 66] studies have been made to find frozen condition for lunar orbits, where eccentricity and periapsis rate are both set to zero thus minimizing the need for station-keeping maneuvers. In this case, significant solutions can be find with Eq. 1.1

$$e = \left(1 - \frac{5}{3}\cos^2 i\right)^{\frac{1}{2}} \tag{1.1}$$

Only for $\omega = 90°; 270°$. Real solutions exist for $i > 39.23°$, known as critical inclination value. Imposed inclinations between 39.23° and 140.77°, there exists an eccentricity which can be used to set both ω and e rates to zero, when ω is 90° or 270°. Conversely, for any $0 < e < 1$, there is an inclination number that satisfy the frozen condition, as depicted in Fig. 1.8

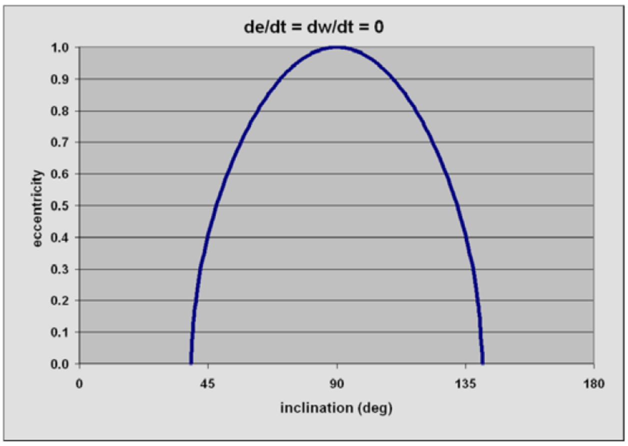


Figure 1.8: Eccentricity and inclination for zero periapsis and eccentricity rate. Image Credits: [62]

The peculiar behaviour exhibited from frozen orbits, inspire some works to investigate the possibility of using them to make a lunar satellite constellation providing coverage for communication purposes.
In the analysis of [66], it is designed a constellation of three satellites in a single inclined elliptical frozen orbit, where two are always in line of sight of the lunar surface for the south polar region, highlighted how the spacecrafts doesn't require orbit control under the effect of gravitational influences. In [67], the previous work is further developed to extend the coverage to a global scale, also considering solar radiation pressure. Two constellation comprises six satellites each were developed that provide near 100% of coverage for a ten year period, additionally demonstrate the need for low to no orbital maintenance. Another study, proposed by [62], designed two constellations, respectively of eight ad twelve satellites, evenly distributed in four frozen orbits for more redundancy and global coverage.

1.3.5 Halo orbits in the Earth-Moon system

A halo orbit is a three-dimensional path situated near one of five Lagrange points in the three-body problem, where three of them are aligned along the line connecting the primary bodies, while the other two form an equilateral triangle with the bodies. Although a Lagrange point appears as an isolated spot in space, its distinctive feature lies in its potential to host either a Lissajous orbit or a Halo orbit. These arises due to the interplay between the gravitational pull of two celestial bodies and the Coriolis and centrifugal forces acting on a spacecraft. Halo orbits can be found within any three-body system, such as a satellite revolving around the Earth-Moon system. Each Lagrange point supports continuous sets of both northern and southern halo orbits [68].
The three aligned points are unstable, and the equilateral-triangle points are only somewhat stable. Hence, some form of positional adjustment is usually necessary to keep the satellite near a libration point. This station-keeping problem, becomes intricate due to perturbative accelerations and additional control requirements imposed by mission constraints [69]. Despite this, Halo orbits configurations have been studied for communication and navigation applications in the lunar environment, because their particular shape allows to have continuous line of sight with Moon and Earth as can be seen in Fig. 1.9

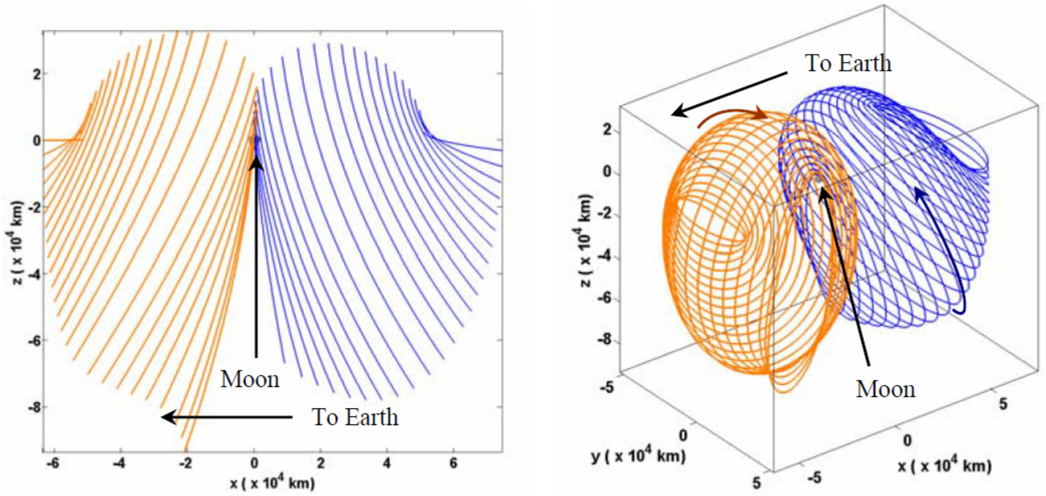


Figure 1.9: Southern Halo Orbit Families: Earth-Moon L1 (Orange) and L2 (Blue). Image Credits: [70]

Thorough examinations of the coverage capacities of Halo orbits and Distant Retrograde Orbits, considering different numbers of satellites per orbit, were conducted in the studies of [71, 72]. These investigations highlight that even with a reduced number of spacecraft, complete lunar coverage is attainable. This is primarily due to the substantial altitude at which these trajectories are positioned and their distinct shapes. Moreover, their shape and their ability to reach high altitudes, up to over 70000 km, make them particularly interesting to provide long-term coverage over specific areas of the lunar surface, specifically focusing on the regions of the South Pole.

Chapter 2

Designing for space: approach to mission design

2.1 Mission life cycle

As previously highlighted, it is evident the necessity and goal to develop a satellite constellation for lunar communication in the coming years. Defining a system of such complexity requires progressing through various phases of study, as outlined in the **ESA Project Life Cycle Model**.
This framework consists of several phases, as outlined in Fig. 2.1, from 0 or pre-phase A to F, that projects typically progress through to help manage space missions, satellite launches, and other space-related endeavors.

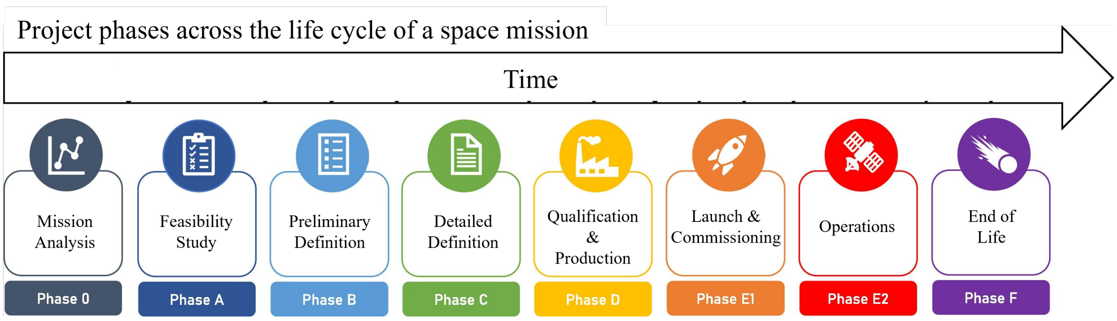


Figure 2.1: Life cycle of a space mission. Image Credits: [73]

Phases 0, A, and B primarily focus on defining system functional and technical requirements, identifying system concepts, detailing all required activities and resources, conducting initial risk assessments, and initiating pre-development activities. Moving into Phases C and D, the focus shifts to executing all tasks necessary for developing and qualifying both the space and ground segments. Phase E encompasses the comprehensive set of activities required for the launch, commissioning, utilization, and maintenance of orbital elements and associated ground segment elements. Lastly, Phase F encompasses the suite of activities aimed at the safe disposal of all products launched into space and the ground segment [74].

2.2 Charting the steps for mission design process

The current work falls within phases 0-A, specifically this chapter contributes to the **mission design process**. The design encompass three primary steps:

- Objectives definition.
- Mission characterization
- System characterization

These involve a series of iterative activities essential to finalize the design, as resumed in Fig. 2.2

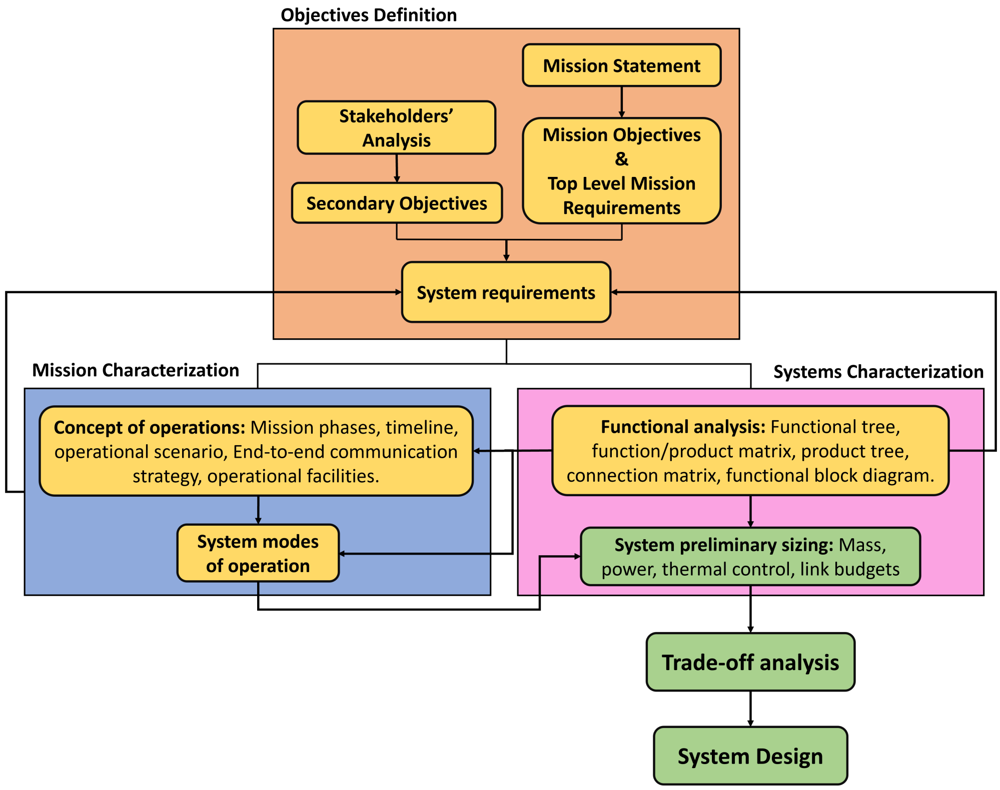


Figure 2.2: Mission design process workflow

Initially, the first include outlining top-level **mission objectives** and **requirements**, which are derived from the **mission statement**. A mission or project statement is a comprehensive, and clear representation of the mission's purpose for existence, often expressed concisely in a single phrase. It is essential to note that mission statement and derived objectives are fixed in this process, and do not actively participate in the subsequent iterative process. They represent the foundation of the mission.
Simultaneously, a critical parallel activity comprises analyzing **stakeholders' needs** by identifying all mission actors and their expectations. This analysis helps in deriving additional or secondary objectives vital to the success of the mission. Assessing the requirements of main stakeholders involved is an essential step on mission design. These can be categorized based on their roles in [75]:

- Sponsors often define the mission statement, set constraints, and allocate funding resources. It is essential to compare the needs of sponsors with those of developers to meet conceptual ideas with practical and technological necessities.

- Operators, usually engineering organizations, oversee the management and maintenance of both space and ground assets.

- End-users are typically scientists or engineers, the consumer of the products and capabilities delivered by the space mission.
- Customers and/or developers, distinct from end-users, are entities or individuals who pay for access to specific space mission products or services.

Once the broader mission objectives are established, the design process progresses to the systems design by specifying the necessary requirements essential to accomplish the mission.
Functional Analysis, a key aspect of this part, is employed to identify the primary functions crucial for mission success and the corresponding physical components. This phase also includes outlining the system's scale by defining parameters such as mass, power, thermal considerations, and link budget.
With the systems defined, design methodology focuses on evaluating the **Concept of Operations (ConOps)**. This falls within the Mission Definition phase and aims to outline how the system will operate throughout its life-cycle phases to meet stakeholders' expectations effectively.
The system definition and mission definition are interconnected, and subsequent to this phase, a **trade-off analysis** is conducted. This rigorously assesses the advantages and disadvantages of each solution based on predetermined and measurable criteria. Parameters such as mass, complexity, and maturity are evaluated against a scoring system, enabling designers to select the most optimal solution based on its specific merits.
This systematic approach ensures that the mission's objectives, stakeholder expectations, operational procedures, and system functionalities are intricately linked and carefully evaluated to achieve the most viable and effective solution.
The following presents the studies conducted in order to identify the expected mission requirements under the guide of space mission design steps.

2.3 Mission objectives and constraints identification

2.3.1 Regions of interest for lunar outpost

Analysis have been conducted to identify key Regions Of Interest (ROI) on the lunar surface intended for future missions. Specifically, considerable attention was devoted to identifying the most suitable locations for fixed habitats that necessitate for the adequate communication, in order to define coverage objectives. Several studies have been conducted to identify the best positions, assessing multiple factors including: surface conditions for landing, significance for scientific research

purposes such as the presence of frozen volatiles, and optimal lighting conditions [76] [77] [78]. Further research also highlights the feasibility of establishing fixed communication systems, known as **Lunar Communication Terminals (LCT)**, to be placed in areas constantly visible from Earth's network antennas, such as the Malapert Mountains in the south pole or equatorial regions within Earth's constant line of sight [51] [78].
It can be concluded that the primary locations identified for fixed outposts are:

- Shakleton Carter at the South Pole.
- Mare Tranquillitatis in Equatorial near Earth side.

Additionally, other scientific important regions have been determined, primarily situated on the far side near the **Von Karman crater** and at the **North Pole** [79].

2.3.2 Stakeholders' identification and secondary objectives

Potential stakeholders have been identified and placed to the belong categories. ESA and Surrey Satellite Technology (SSTL) or other spacecraft manufacturer can be classified as the sponsors. The main operators are SSTL, Telespazio, NASA's Deep Space Network operators.End-users can be scientists, astronauts, ground segment engineers, international space agencies. Customers, in this case, private companies that might potentially leverage the system could be involved.

Secondary mission objectives have been formulated by assessing the requirements of the identified stakeholders:

- To ensure coverage of precise locations on the lunar surface: Mare Tranquillitatis and South Pole Shackleton Crater.
- To enhance reliability and lifetime of constellation satellites.
- To reduce future lunar missions complexity.
- To increase data volume transmission from the lunar environment.

Within the stakeholders analysis, specific **mission constraints** arise:

- To ensure timely connection with Earth.
- To ensure connection among lunar user for more than 90% of constellation lifetime.
- To reduce the complexity of constellation by promoting modular small sized satellites.

2.4 Requirements, system and mission definition

The direct outcome of defining the mission's objectives is the establishment of requirements. These can be of various types: mission, functional, configuration, interface, environmental, operational, and logistic support. The first directly come from the mission objectives, while the others are a result of the functional analysis and the concept of operations.
There are strong interactions between requirements, concept of operations and functional analysis. After establishing the mission objectives, the high-level mission requirements can be delineated. Subsequently, the conceptual design process progresses, encompassing both the system and mission definition.

2.4.1 Functional analysis elements

To effectively identify the foundational elements crucial for the mission, employing functional analysis becomes imperative. Within the conceptual design, this approach operates across varying tiers, including subsystem, system, and system-of-systems levels. Depending on the chosen tier, the components constituting the future product manifest as equipment, subsystems, or system.
At the core of Functional Analysis lies a collection of key tools: the functional tree, functions/products matrix, product tree, connection matrix, and functional/physical block diagrams. The functional tree serves as the starting point, facilitating the division of high-level functions originating from the mission objectives or top-level system requirements into progressively lower tiers. This process ultimately pinpoints the fundamental functions essential for the envisioned product. Higher-level functions represent tasks that necessitate decomposition into simpler lower-level functions to facilitate comprehensive analysis. Thus, starting from the so-called top-level function, the functional tree extends branches that comes from intricate functions to foundational ones, situated at the base.
After determining the fundamental functions, selecting the components responsible for executing them becomes feasible through the utilization of the functions/products matrix.
This matrix serves the purpose of associating functions with their corresponding physical components. Its construction involves a straightforward process of aligning the foundational functions, located at the bottom of the functional tree, with rows of components capable of fulfilling them.

2.4.2 Concept of operations

The Concept of Operations (ConOps) serves to detail the practical implementation of a mission to align with stakeholder expectations. It articulates the system's functional aspects from an operational standpoint, encompassing elements such as data acquisition methods, mission control strategies, and feasible orbital configurations. The ConOps comprehensively addresses all operational phases, spanning integration, testing, launch procedures, and eventual disposal. Key components of the mission concept outlined encompass a breakdown of significant steps, timelines for operations, end-to-end communication strategy, and details regarding operational infrastructure. This also include the operational scenario, providing a dynamic view of system operations, defining how the system is envisioned to work across various modes or scenarios [80].
The objective of this study is to define a lunar communication constellation, specifically concentrating on satellites, conducting preliminary sizing and orbital analysis to meet the expected requirements. To have a comprehensive overview of the entire mission, all the potential phases have been identified, but the subsequent sections will focus into the lunar stage for further analysis. ConOps starting from the definition of the following mission phases:

1. Launch operations.
2. Trans-lunar injection.
3. Satellites inserction into lunar orbits.
4. Detumbling, deployment of solar panels, commissioning, validation of S/C subsystems, calibration of communication payload and bus system.
5. Communications activities with designated targets.
6. Station keeping maneuvers to maintain constellation performances.
7. End of life and disposal strategy.

As previously stated, focusing on the mission operational phase, the following concept of operation was considered in order to satisfy requirements:

"*To provide communication relay to lunar users and Earth's control facility with at least one satellites always in line of sight with Moon surface's elements.*"

2.4.3 Surface elements in the lunar scenario

Identify the key elements that may be located within the scenario and explain their interactions, is essential to support the delineated end-to-end communication

strategy and potential mission architectures.
In [81] NASA express the plan to transport astronauts and cargo to lunar orbit and the lunar surface. Landers and rovers will be used to explore the lunar surface and conduct scientific investigations, including experiments related to in-situ resource utilization (ISRU) to generate detailed information on the availability and extraction of usable resources such as water and oxygen. The Lunar Communications Terminal (LCT) and Lunar Relay Satellites (LRS) will serve as major network access points for multiplexing and routing data within the lunar vicinity and between the moon and earth.

Figure 2.3: Artist's concept of Artemis Base Camp. Credits: [37]

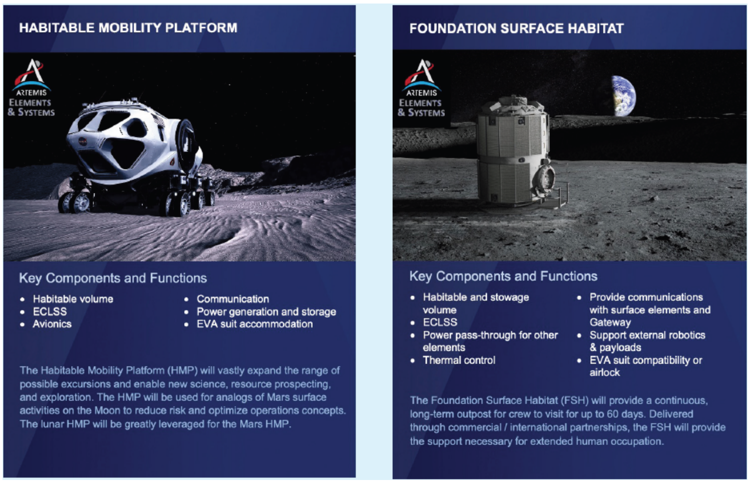


Figure 2.4: Artist's concept of lunar elements for exploration. Credits: [37]

The LRS and LCT will need to provide service to multiple surface assets simultaneously, each having different data volume requirements depending on the operational scenario [82]. Other papers as [51] [78] [83] highlights the possibility to use lunar terminals or towers to have a direct link with Earth while communication between surface elements use satellite's relay. A concept of these architectures is represented in fig. 2.5

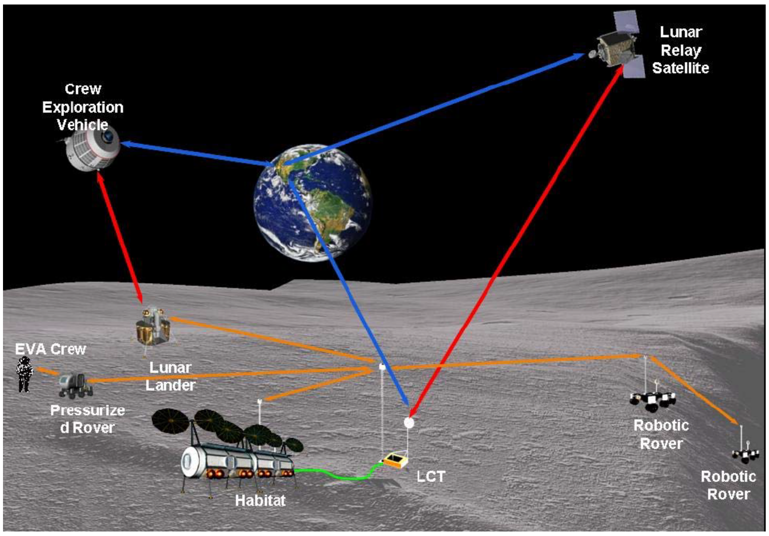


Figure 2.5: Lunar mission communication links. Credits: [78]

2.4.4 Frequency and data rate allocation: End-to-end communication architecture

Once the primary elements involved in the communication network are identified, another crucial aspect to characterize the **end-to-end communication architecture** is the evaluation of potential frequencies allocation for the links. Referring to some studies like in [84] [85] [86], specified frequency bands and links to be performed have been identified.
The links comprises:

- Forward and backward communication from Earth Deep Space Network (DSN) to lunar relay satellites (LRS).
- Forward and backward communication from Earth Deep Space Network (DSN) to lunar communicarion terminals (LCT).
- Forward and backward communication from LCT to LRS.
- Inter-satellite links (ISL).

In addition to interfaces and bands, the end to end communications strategy provides with type and amount of exchanged data. These information were extracted mainly from [51] [85] [87]. All the data are summarised in Tab. 2.1 for better clarity.
Once the necessary data for establishing the various links required by the mission have been identified, it is then possible to define the End-to-End communication architecture of interest, as schematically depicted in the Fig. 2.6

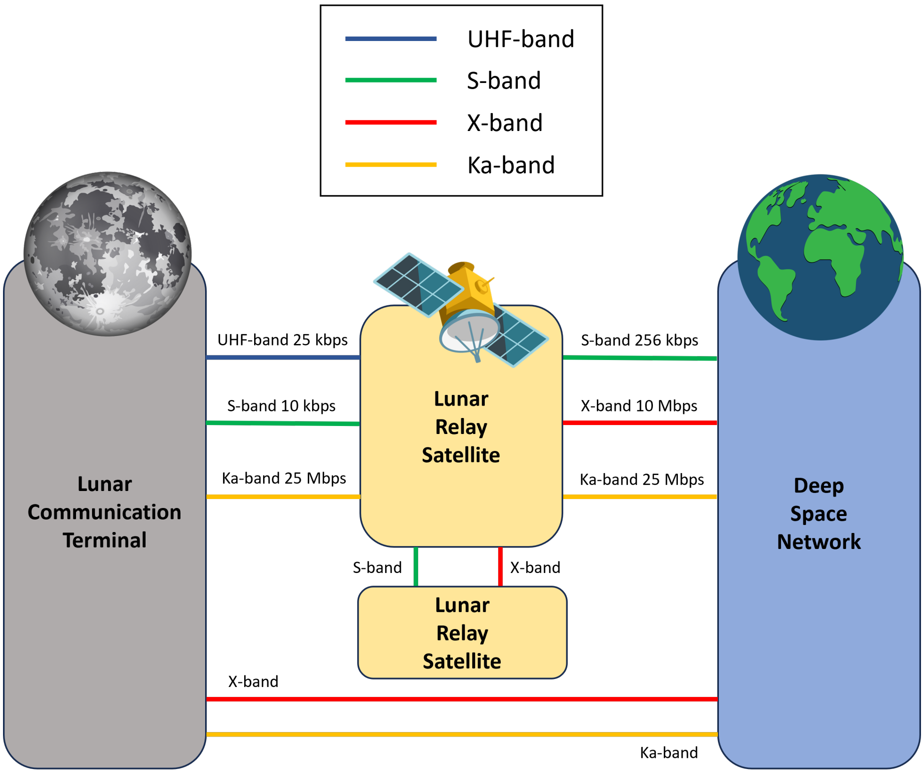


Figure 2.6: End-to-end communication architecture

2.5 Satellite trajectories impact on design: Frozen orbits vs Halo

To meet the stakeholders' needs and fulfill the expected performance of the mission, a preliminary assessment was carried out on the characteristics of the orbits most extensively studied in this field, in order to definr the most appropriate.
In the search for the most suitable orbits for the mission, the primary considerations were stability and the ability to cover sites of interest with the fewest satellites possible while ensuring high-quality communication in terms of reduced data transmission delay and required power. These criteria were evaluated to facilitate the

Frequency bands and data rate allocations						
Link type	Rates	Band	fmax [MHz]	fmin [Mhz]	Service	Data Rates [kbps]
Earth DSN to LRS	Low	S-band	2110	2025	Voice/Command	72
		X-band	7235	7190	Command/Ranging	1000
	Medium	Ka-band	23150	22550	Voice/Data/Video	10000
LRS to Earth DSN	Low	S-band	2290	2200	Voice/data	256
	Medium	X-band	8500	8450	Telemetry/Ranging	10000
		Ka-band	27000	25500	Voice/Data/Video	25000
LRS to LCT	Low	UHF-band	405	390	Command	1
		S-band	2500	2483.5	Voice/TT&C	10
	Medium	S-band	2110	2025	Voice/TT&C/ Health status	1000
	High	Ka-band	23150	22550	Voice/Data/Video	25000
LCT to LRS	Low	UHF-band	450	435	Telemetry	10
		UHF-band	1626.5	1610	Voice/TT&C/Command	1000
	Medium	S-band	2290	2200	Voice/TT&C	3000
	High	Ka-band	27000	25500	Voice/Data	25000
ISL	High	Ka-band	23550	23150	User data	2500

Table 2.1: Communication data allocation for DSN and LCT link

achievement of mission objectives and, simultaneously, to minimize the size of the satellite's main subsystems, namely the propulsion and communication.
Keeping this concept in mind, the decision made in favor of Frozen Keplerian orbits over Halo orbits, stem from several considerations. Although certain families of Halo orbits might exhibit greater stability than Frozen orbits, the latter offer distinct advantages in specific operational aspects.
One crucial factor influencing the preference for Frozen orbits lies in their significantly lower operational distance from the lunar surface. While a higher distance can be beneficial in some aspects, such as scientific observations or specific mission objectives, the high altitude of Halo trajectories poses challenges in establishing low-latency communications. Indeed the Halo orbits spans a range from approximately 2000 km at periapsis to over 60000 km at the apoapsis which is also the region where satellites spend most of their orbital time. Therefore, the use of these orbits appears to be more suitable for surface navigation on the moon, even while accepting signal delays, or for the navigation of satellites in lower orbits [71] [88].
Moreover, Halo orbits appear better suited for polar coverage compared to the specific needs of the mission. This is because their shape allows the apoapsis to extend mainly into these regions [89].
On the other hand, while certain families of Halo orbits exist closer to the lunar surface, these tend to be more unstable compared to Frozen ones. This instability could compromise the predictability and reliability required for sustained lunar missions, potentially impacting the safety and success of spacecraft operations [71].
Therefore, despite potential stability advantages in some Halo orbit families, the combination of good stability and lower distance offered by Frozen orbits poses fewer communication hurdles compared to Halo.
The enhanced communication reliability and manageable operational aspects of Frozen orbits make them a more suitable choice for ensuring consistent and effective communication and control during lunar missions.

2.6 How to size the constellation satellites

This section examines potential methodologies assessed for sizing the satellite concerning its constituent subsystems. The objective of this work is not to develop a detailed design but rather to define the subsystems in terms of mass and required power. Emphasis has been placed on the communication subsystem, serving as the primary payload of the spacecraft, and the propulsion system. These elements serve as main drivers for sizing and are influenced respectively by the mission architecture and the chosen orbits.

2.6.1 Earth satellites database: parametric design approach

The first sizing method followed the logical flow outlined in the diagram in Fig. 2.7

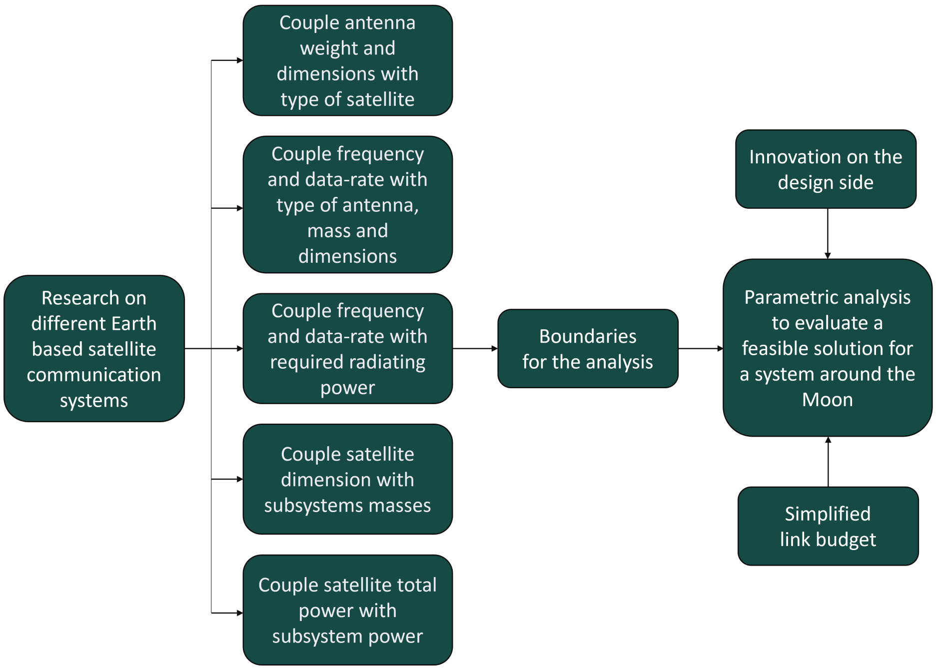


Figure 2.7: Parametric analysis workflow

Initially an extensive analysis has been conducted to identify Earth-based communication satellites and their subsystems. The research phase focuses on different types of communication spacecraft, evaluating:

- Total mass and power.
- Power allocation for the subsystems.
- Mass allocation for the subsystems.
- Communication payload in therms of antenna types, dimensions, mass, frequencies and data-rate.

Data have been searched on several documents [16] [17] [18] [20] [90] [91] [92] [93] [94]. The research on the satellite's characteristics was unsuccessful due to the lack of data, attributed to the fact that the information is managed by private companies, making it challenging to retrieve.

Further investigations led to the study of [95], where a compilation of mass and power data on numerous satellites is available. Utilizing this database, depicted in

Fig. 2.8, correlations between total masses and subsystem masses, total powers and subsystem powers were established through linear regressions.

System	Filing date	No. sats	T_{life}, years	Alt., km	Downlink BW, MHz[a]	Downlink CFreq, GHz[b]	M_{dry}, kg	M_{prop}, kg	M_{wet}, kg	M_{pl}, kg	P_{PL}, W	P_{bol}, W	P_{eol}, W
@contact	May 99	20	12	10,400	1,100	19.55	2,542	870	3,412	583	6,264	—	—
AMSC NGSO	Nov. 94	10	10	10,355	16.5	2.49	2,450	600	3,050	950	4,500	—	4,900
Boeing NGSO	Jan. 99	20	12	20,182	1,000	12.20	2,118	1,743	3,861	1,217	9,500	14,201	10,678
Celestri	June 97	63	8	1,400	1,000	29.30	2,500	600	3,100	—	—	13,600	4,600
Constellation	June 91	48	5	1,018.6	16.5	2.49	113.4	11.3	124.7	34.5	49	250	49
Ellipso	Nov. 90	24	5	875	16	2.49	68	0	68	—	—	360	120
E-Sat	Nov. 94	6	10	1,261	1	0.14	100	14	114	—	—	—	200
Final Analysis	Nov. 94	24	7	1,000	0.225	0.14	98.5	0	98.5	—	29.5	59	47
GE LEO	Nov. 94	24	5	800	0.034	0.40	15	0	15	—	11	10.56	9.1
GEMnet	Nov. 94	38	5	1,000	1	0.14	—	—	45	—	—	—	—
Globalstar	June 91	48	7.5	1,389	16.6	1.62	222	40	262	60	50	875	150
Globalstar 2GHz	Sept. 97	64	7.5	1,420	40	2.18	676	156	832	300	1,200	3,000	1,520
Globalstar GS40	Sept. 97	80	7.5	1,440	1,000	38.00	992	234	1,226	—	—	4,500	2,280
HughesLINK	Jan. 99	22	12	15,000	1,000	11.73	2,540	400	2,940	1,000	6,000	10,500	9,100
HughesNET	Jan. 99	70	10	1,490	500	11.73	1,650	350	2,050	600	4,000	8,200	7,500
ICO	Sept. 97	10	12	10,355	30	2.19	2,413.8	336.2	2,750	898	5,994	—	9,000
Iridium	Dec. 90	77	5	765	15.5	1.62	299.4	41.3	340.7	165.1	686	—	1,429
Iridium Mcell	Sept. 97	96	7.5	853	40	2.18	1,442	271	1,713	670	1,105	7,300	—
Leo One	Sept. 94	48	5	950	0.2	0.14	154	0	154	26	158	—	290
LM MEO	Dec. 97	32	10	10,352	3,000	40.00	2,133	38	2,171	840	6,610	—	8,760
M Star	Sept. 96	72	8	1,350	3,000	39.00	2,210	323	2,535	—	—	3,100	1,530
Odyssey	May 91	12	10	10,371	16.5	2.49	1,620	880	2,500	450	1,200	—	1,800
Orbcomm	Feb. 90	20	7	970	0.27	0.14	145	5	150	—	325	450	360
Orblink	Sept. 97	7	7	9,000	1,000	38.00	1,268	742	2,010	615	3,650	—	4,010
Pentriad	Sept. 97	13	—	HEO	2,000	39.50	1,455	684	2,139	592	100	10,247	7,684
SkyBridge	Feb. 97	64	8	1,457	1,050	11.73	—	—	800	300	2,150	—	3,000
SkyBridge II	Dec. 97	96	10	1,468	1,250	19.00	—	—	2,650	1,000	5,000	—	9,000
Spaceway	Dec. 97	20	12	10,352	500	19.05	2,500	350	2,850	—	7,500	13,800	10,000
StarLynx	Sept. 97	20	12	10,352	1,100	38.05	3,050	450	3,500	—	—	17,000	15,000
StarSys	May 90	24	5	1,300	1	0.14	112	0	112	24	84	—	125
Teledesic	Mar. 94	840	10	700	400	19.00	747	48	795	173	3,600	11,595	6,626
Teledesic Ku	Jan. 99	30	7	10,320	2,000	11.70	1,132	192	1,324	—	1,200	6,500	1,500
Teledesic V	Sept. 97	72	7	1,375	1,000	38.00	566	48	614	—	600	5,000	800
TRW EHF	Sept. 97	15	15	10,355	3,000	39.00	2,707	3,227	5,934	926	15,000	—	15,500
Virgo	Jan. 99	15	12	20,281	1,500	11.95	2,778	252	3,030	1,058	9,900	—	10,593
VITA	Sept. 90	2	5	805.5	0.06	0.14	43	2.5	45.5	—	—	42	25.3

Figure 2.8: Communication satellites system data. Credits: [95]

In order to assess the quality of the found models, results were then compared with those from [96] [97] [98] where a similar approach was adopted on a larger sample of satellites.
The findings of [96] are shown in the Tab. 2.2 2.3 2.4 2.5

Satellite mass class	Mass model for communication subsystem to total satellite mass
Large	$M_{comsub} = 0.2764 \times M_{total} + 156.56$
Other categories	$M_{comsub} = 0.062 \times M_{total} + 0.1179$

Table 2.2: Model for communication subsystem mass to total mass. Credits: [96]

Mission	Mass model of power subsystem
Telecommunication	$M_{powersub} = 0.0971 \times M_{total} + 100.13$
Other	$M_{powersub} = 0.1012 \times M_{total} + 3.255$

Table 2.3: Model for power subsystem mass to total mass. Credits: [96]

Satellite mass class	Power model for communication subsystem to communication subsystem mass
Large	$P_{comsub} = 3.5143 \times M_{comsub} - 666.38$
Other categories	$P_{comsub} = 3.4256 \times M_{comsub} - 0.2487$

Table 2.4: Model for communication subsystem power to communication subsystem mass. Credits: [96]

Satellite class	Power model to satellite class
Large	$P = 2.927 \times M_{total} - 3206$
Small	$P = 1.6746 \times M_{total} - 419.1$
Mini	$P = 3.1079 \times M_{total} + 359.43$
Micro	$P = 1.1344 \times M_{total} - 11.384$
Nano	$P = 4 \times M_{total} - 2$

Table 2.5: Model for total mass to total power Credits: [96]

The comparison between the models generated conflicting results. Moreover, applying the relationships found in the literature and those defined by linear regression to the available satellite data in the databases, revealed significant discrepancies between the estimated and actual results. This led to the exclusion of this approach from the design process of the satellite platform and its subsystems.
The cause of these results can be attributed to the need for a more accurate classification of communication satellite classes. Specifically, it would have been appropriate to categorize spacecraft based on size and orbital belonging. Subsequently, within these subgroups, differentiating satellites utilizing high-frequency systems from those using medium to low-frequency systems would have been beneficial. At this point, models could have been defined for each subgroup to relate subsystem masses and powers. However, in this case, such a process was not feasible due to the lack of specific data as explained above.

2.6.2 Scaling method

Scale-up and scale-down sizing methods involve adjusting the size or capacity of a system, in this case, a satellite, either upwards (scale-up) or downwards (scale-down) while maintaining its fundamental functionalities and characteristics. In satellite dimensioning, this process can be used to modify the size, weight, power, and other parameters of the satellite based on specific needs or limitations, referring to one or more satellites that have already been operational.

This could involve increasing or decreasing the dimensions of the satellite components, altering power requirements, adjusting payload capacities, etc., while ensuring that the satellite still fulfills its intended purpose and performance criteria. These methods require careful consideration and analysis to ensure that any modifications do not negatively impact the satellite's functionality, performance, or reliability.

Before proceeding with this approach, it is essential to identify one or more reference satellite platforms and establish the interrelationships among their subsystems.To support this process, reference was also made to Space System Engineering (SMAD) [6], particularly utilizing the mass and power distribution tables depicted in the Fig. 2.9

Subsystem (% of Dry Mass)	No Prop (%)	LEO Prop (%)	High Earth (%)	Planetary (%)
Payload	41%	31%	32%	15%
Structure and Mechanisms	20%	27%	24%	25%
Thermal Control	2%	2%	4%	6%
Power (including harness)	19%	21%	17%	21%
TT&C	2%	2%	4%	7%
On-Board Processing	5%	5%	3%	4%
Attitude Determination and Control	8%	6%	6%	6%
Propulsion	0%	3%	7%	13%
Other (balance + launch)	3%	3%	3%	3%
Total	100%	100%	100%	100%
Average Dry Mass (kg)	1497	2344	1258	888

Subsystem (% of Total Power)	No Prop (%)	LEO Prop (%)	High Earth (%)	Planetary (%)
Payload	43%	46%	35%	22%
Structure and Mechanisms	0%	1%	0%	1%
Thermal Control	5%	10%	14%	15%
Power (including harness)	10%	9%	7%	10%
TT&C	11%	12%	16%	18%
On-Board Processing	13%	12%	10%	11%
Attitude Determination and Control	18%	10%	16%	12%
Propulsion	0%	0%	2%	11%
Average Power (W)	299	794	691	749

Figure 2.9: Subsystem percentage of dry mass and power. Credits: [6]

Platform data

Considering the proposed architectures, two distinct satellite platforms were selected as potential feasible solutions for the mission and for the application of the scaling method.
The first is a CubeSat from Hemeria company, designed as a small satellite for Internet of Things (IoT) operating in Low Earth Orbit (LEO) [99].

Figure 2.10: Hemeria HPIOT satellite. Credits: [99]

Main characteristics of the platform are resumed in the following Tab. 2.6

Hemeria HP-IoT satellite		
Platform Size [mm]	Regular	Boosted
	220x230x500	
Platform Mass [kg]	20	22
Max Payload Mass [kg]	15	
Max Payload Volume	8 U	
Battery Capacity [Wh]	172	
Payload avg. Power [W]	50-80	80-120
Payload Peak Power [W]	200	
Uplink Rate [kbps]	64 (S-band)	
Downlink Rate [kbps]	1000 (S-band)	1000 (S-band) 150-300 Mbps (X-band)
DeltaV [m/s]	>150	

Table 2.6: Hemeria satellite Data sheet. Credits: [99]

The second option is the larger Iridium NEXT satellite developed by Thales and based on ELiTeBus-1000 (Extended LifeTime Bus) [100], also positioned in low Earth orbit [101].

Figure 2.11: Iridium NEXT satellite. Credits: [102]

Tab. 2.7 summarizes the characteristics of Thales' satellite.

Thales Iridium NEXT satellite	
Platform Size [mm]	3057x1040x2135
Platform Mass [kg]	750
Max Payload Mass [kg]	350
Payload avg. Power [W]	955
Peak Power [W]	2000
Downlink Rate	128 kbps (L-band) 1.5 Mbps (L-band) 8 Mbps (Ka-band)
Fuel Mass [kg]	164 (hydrazine)
DeltaV [m/s]	>380

Table 2.7: Iridium NEXT satellite Data sheet. Credits: [94] [100]

2.6.3 Link budget

Before proceeding with the analyses, it is important to define the parameters that most significantly influence communication and therefore affect the power required to ensure the link closure, allowing the considered elements to exchange the required data. The process is known as Link Budget, and it is the accounting of all of the gains and losses from the transmitter, through the medium (free space, atmosphere, cables and connectors), to the receiver in a telecommunication system [6].
In the performed simulations, the parameters necessary to define the link budget were obtained using the E_b/N_0 method, which is based on Eq. 2.1:

$$\frac{E_b}{N_0} = \frac{P_t \cdot L_t \cdot G_t \cdot L_{TSL} \cdot G_r}{k_B \cdot T_s \cdot R} \tag{2.1}$$

For better clarity on the parameters introduced is useful to refer to the Fig. 2.12 representing a communication system architecture with the elements of transmitter and receiver.

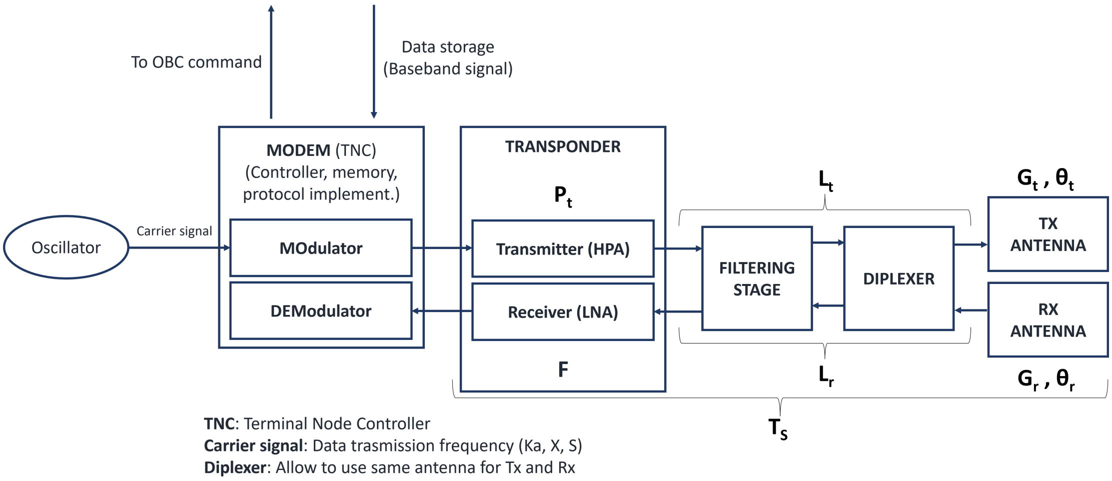


Figure 2.12: Communication system architecture

The parameters with subscript 't' represent the transmitter path, while those with subscript 'r' represent the receiver.
The E_b/N_0 value is the energy required to transmit a bit (Eb) in presence of white noise (N0). It depends on:

- P_t is the power output of the High Power Amplifier (HPA).
- L_t represent losses along the transmission line. Loss is estimated knowing the length of the line and the attenuation/meter value typical of coaxial cables.
- G_t is the transmitting antenna gain.
- L_{TSL} is the total space loss, and comprises antenna pointing loss (L_{pr}), path loss (L_p) and atmospheric loss (L_a).
- G_r is the receiving antenna gain.
- k_B is the Stefan-Boltzmann constant.
- T_s is the system noise temperature.
- R represent the data rate of the considered link.

G is strictly related to the type of antenna and frequency used, in particular the higher the frequency the greater the gain. As for the propagation losses, those are related to the distance from transmitter to receiver and frequency by the Eq. 2.2

$$L_s = 20 \log_{10} \left(\frac{\lambda}{4\pi S} \right) \tag{2.2}$$

Where λ is the wave length and S represent the Slant Range, that is to say the distance in line of sight from transmitter to receiver. Now is possible to express the E_b/N_0 in decibels by the Eq. 2.3

$$\begin{aligned} \frac{E_b}{N_0}[\text{dB}] &= (P_t + L_t + G_t)[\text{dB}] + L_{TLS}[\text{dB}] + G_r[\text{dB}] - k_B[\text{dB}] - T_S[\text{dB}] - R[\text{dB}] \\ &= EIRP + L_{TLS} + G_r - k_B - T_S - R \end{aligned} \tag{2.3}$$

The value will be greater the higher G_t, G_r, and transmission power are, while it will be penalized by the propagation distance of the signal, receiver noise temperature, and the amount of data transmitted.
To understand the real capability of the system to accomplish the communication, this Eb/N0 shall be compared with a required E_b/N_0 for specific choices of:

- Bit Error Rate (BER), defines the probability that a wrong transmission of a bit occurs. Small BERs require a better system because a better accuracy is required.

- Signal modulation.

- Coding.

The required value of E_b/N_0 is provided through tables and graphs like Fig. 2.13

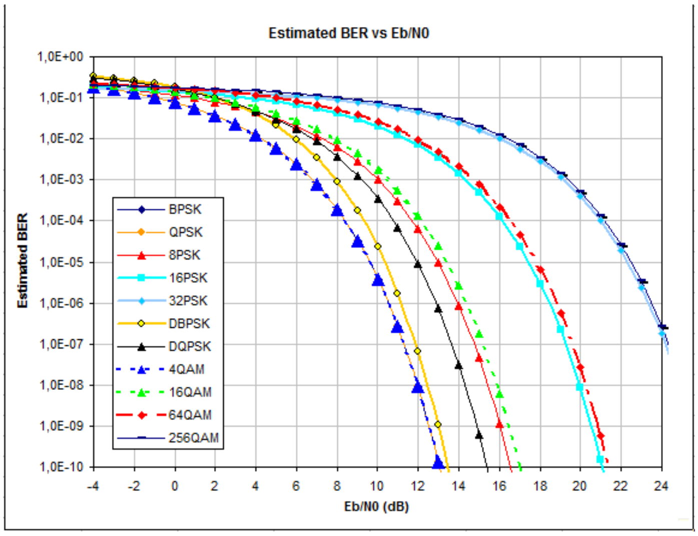


Figure 2.13: Bit Error Rate vs E_b/N_0

Comparing the E_b/N_0 of the system and the E_b/N_0 required, the link margin (LM) is defined by Eq. 2.4

$$LM = \left(\frac{E_b}{N_0}\right)_{calc} - \left(\frac{E_b}{N_0}\right)_{req} \tag{2.4}$$

Link margin value express the effective capability of the communication system:

- If $LM < 0$, link is not present between transmitter and receiver.
- If $0 < LM < 6$, link between transmitter and receiver is marginal.
- If $LM > 6$, link between transmitter and receiver is closed and communication is guaranteed.

Chapter 3

Mission design: preliminary analysis and results

3.1 Mission analysis for the case study

After outlining the mission design process and establishing the objectives, requirements, and constraints gathered from various studies in the literature, the mission design process can be finalized by making assumptions, conducting analyses, and defining the missing parts.

3.1.1 Mission statement

For the lunar constellation, the following statement at system of systems level have been defined:

"*To develop a satellite constellation around the Moon able to ensure continuous relay communication among various users on the lunar surface, and connecting them to Earth for the support of human, scientific and commercial missions that will explore Earth's satellite, and to aid, over time, the development of stable colonies and a Lunar economy.*"

As a consequence of the statement, the primary mission objectives have been identified as extensive goals for the mission:

- To enhance lunar communication capabilities.
- To support increased human activities on the lunar surface.
- To extend space economy to commercial ventures.

- To support targeted missions on the lunar surface.
- To improve safety for human activities.

3.1.2 Mission building blocks definition

The scheme in fig. 3.1 is the result of the first step of the functional analysis, that is to say, the functional tree.

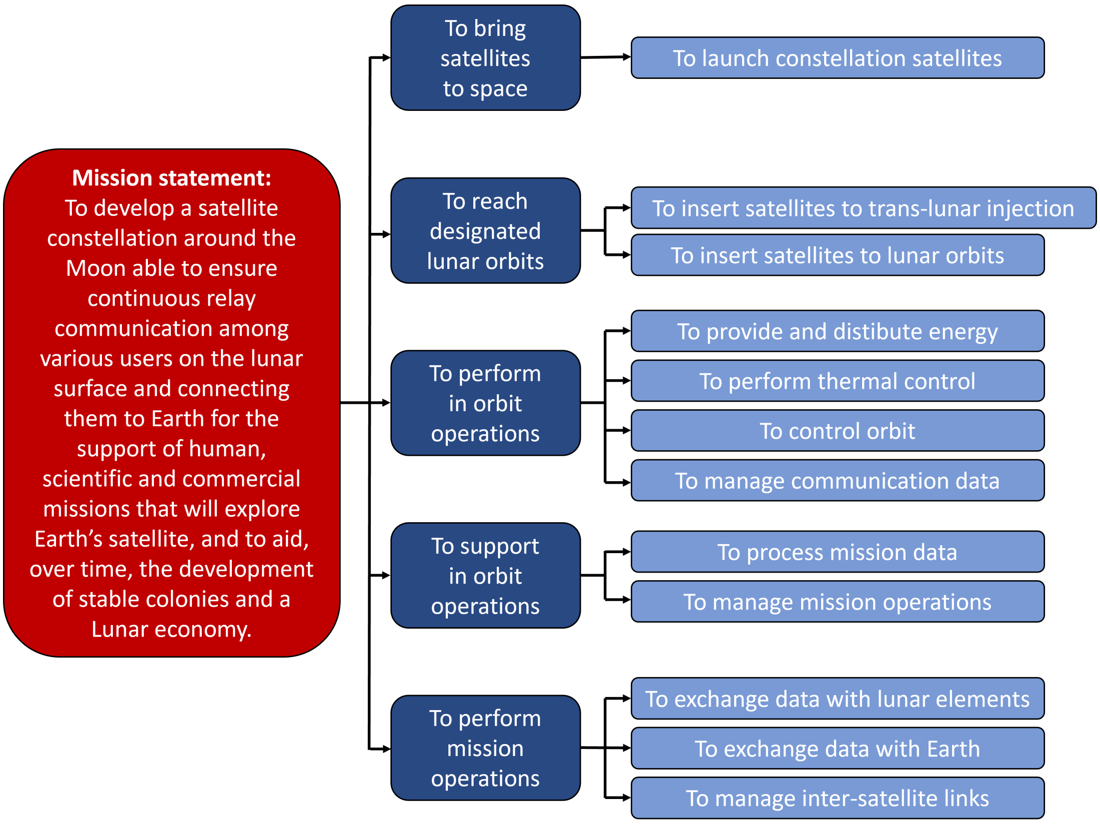


Figure 3.1: Functional tree

At this stage, we can extract functional requirements by referencing the previously established functional tree:

- Constellation satellites shall be launched into space.
- Constellation satellites shall be inserted into trans-lunar injection.
- Constellation satellites shall be placed into lunar orbits.
- Energy shall be provided and distributed.

- Temperature shall be controlled.
- Orbital elements shall be controlled.
- Communication data shall be managed.
- Mission data shall be processed.
- Mission operations shall be managed.
- Data shall be exchanged with lunar elements.
- Data shall be exchanged with Earth.
- Inter-satellite link data shall be managed.

In the fig. 3.2 is shown the first functions/products matrix, associated to higher lever functions of functional tree.

		PRODUCTS		
		Launch segment	Space segments	Ground Segments
FUNCTIONS	To bring satellites to orbit space	X		
	To reach designated lunar orbits		X	
	To perform in orbit operations		X	
	To support in orbit operations			X
	To perform mission operations		X	

Figure 3.2: Higher level functions-products matrix

While the fig. 3.3 represent the matrix related to lower level functions.

		PRODUCTS				
		LS	SS		GS	
		Launcher	Satellites	SpaceTug	Earth stations	Lunar elements
FUNCTIONS	To launch constellation satellites	x				
	To insert satellites to trans-lunar injection			x		
	To insert satellites to lunar orbits			x		
	To provide and distribute energy		x			
	To perform thermal control		x			
	To control orbit		x			
	To manage communication data		x			
	To process mission data				x	
	To manage mission operations				x	
	To exchange data with lunar elements		x			
	To exchange data with Earth		x			x
	To manage inter-satellite link		x			

Figure 3.3: Lower level functions-products matrix

By establishing the lower-level functions-products matrix, it is possible to understand the core components, namely the **mission building blocks**, forming the final product. These elements hold paramount importance in defining the mission scenario and facilitating an initial system sizing.

3.2 Mission architectures: definition and study

3.2.1 Feasible architectures

The possibility of data transfer between LCT and DSN has been highlighted, so the primary determinant shaping the framework is the communication link between the Moon and Earth. Based on previous analysis and considerations, all mission scenario elements and their interactions have been identified, leading to the establishment of two possible mission architectures:

1. The constellation's satellites manage all links and are directly connected with Earth's ground facilities.

2. The constellation's satellites solely communicate with lunar elements, and the Earth link depends on the Lunar Communication Terminals.

In the first scenario, a larger spacecraft class is expected to maintain the necessary performance. However, in the second condition, a smaller CubeSat-like satellite could suffice, meeting the requirements on platform dimensions more effectively.
Other advantages in using the LCT for direct link with Earth may be the use of optical communication, reduced latency and improved reliability and data-rate compared to using a satellite as a relay [78]. Furthermore, studies from [83] highlighted that lunar terminals can be used for short range communication and navigation in the surface, thus reducing the necessity of a large constellation of satellites.
In the Fig. 3.4, the two identified architectures are schematically depicted, which will be the basis for preliminary definition analysis of the satellite platform.

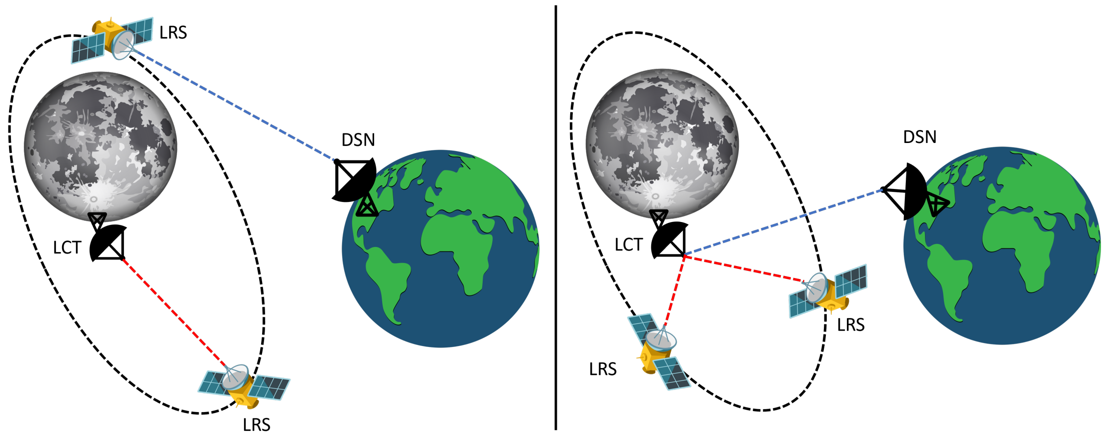


Figure 3.4: Feasible mission architectures

In order to proceed with the study, it is necessary to define which of the proposed architectures is most suitable to meet the requirements and constraints imposed by the mission. Once the necessary analyses have been carried out, it will be possible to determine which of the satellite platforms can adapt to the scenario and proceed with the sizing.

3.2.2 Trade-off on architectures and satellite platforms

To meet the imposed requirements, it would be desirable to have the possibility of using small satellites capable of satisfying the constraints in terms of transmitted data volume, especially among Moon's elements. For this reason, as previously mentioned, two mission architectures have been proposed based on systems that could potentially be placed on the lunar surface, like LCT.
In the scope of the analyses conducted, the two scenarios were defined using AGI's software: System Tool Kit (STK). During these initial simulations, the focus was

solely on determining the transmission powers required to meet communication performances.

Communication with lunar surface elements

For communication with the Moon's elements, a single satellite was placed in a circular inclined orbit, at an altitude of 10000 km, to encompass even the most disadvantageous case in terms of lunar surface distance for the selected trajectories. The scenario is depicted in Fig. 3.5 and represents the communication satellite and one of the two possible lunar outposts, in this case, the one located in the equatorial region in the area known as Mare Tranquillitatis.

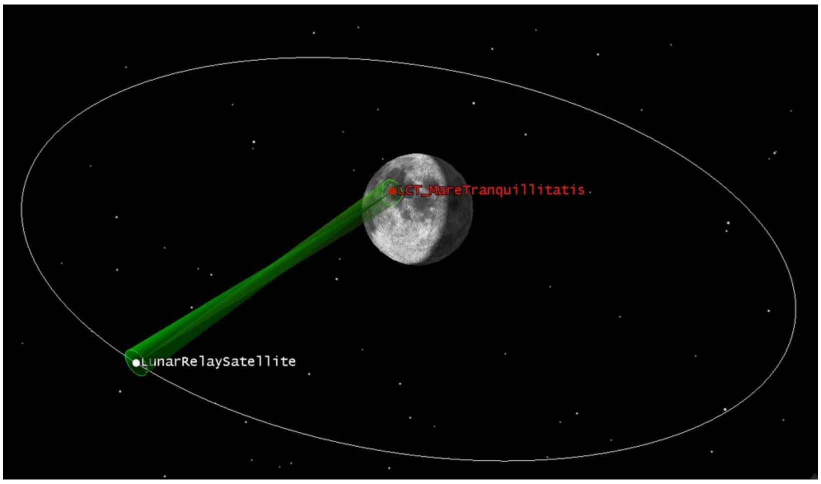


Figure 3.5: Lunar relay satellite communication scenario with LCT

Proceeding with the analyses, it was initially conducted a collection of performance data, concerning gain, transmission power and data rate, provided by antennas using the required frequency bands in the lunar environment. These data are summarized in the Tab. 3.1 and derive from [103]

Communication system solutions		
UHF-band		
	min	Max
Gain [dB]	0	5
Tx Power [dB]	0.25	4
Data Rate [kbps]	0.1	42000
S-band		
Gain [dB]	3	8
Tx Power [dB]	0.5	10
Data Rate [kbps]	4	100000
Ka-band		
Gain [dB]	20	42
Tx Power [W]	0.3	4
Data Rate [Mbps]		up to 150

Table 3.1: Gain, power and data rate for the lunar region frequencies. Credits: [103]

The Tab. 3.2 contains values that have been assumed to remain constant across all communication systems, and are used to perform the simulations.

Communication system constants	
Polarization	RHC
Modulation	QPSK
LNA noise figure [dB]	1.2
Max BER	10^{-6}
Link Margin Threshold [dB]	10.5

Table 3.2: Constant values of communication system

QPSK modulation type was considered to assess the required link margin for the BER, while a typical value was chosen for the loss.
Ultimately, the Tab. 3.3 shows the minimum data rates to be met in the different communication links at their respective frequencies.

At this stage, it was possible to perform a series of simulations, conducting a trade-off among the power and gain values of the three antennas: UHF, S, and Ka-bands, to achieve the required data transfer.
In the following Tab. 3.4 are reported the results of the analysis.

Data requirements	
Frequency band	Data Rate [kbps]
UHF	1-10
S	10-1000
Ka	25000

Table 3.3: Data rate requirements for lunar users

Satellite to LCT communication system			
TRANSMITTER			
Frequency band	UHF-band	S-band	Ka-band
Frequency [MHz]	397.5	2067.5	22850
Signal bandwidth [MHz]	15	85	600
Transmitter power [W]	0.25	10	4
Gain [dB]	2.5	7	25
Data rate [kbps]	25	1000	25000
RECEIVER			
Frequency band	UHF-band	S-band	Ka-band
Frequency [MHz]	442.5	2245	26250
Signal bandwidth [MHz]	15	85	600
Gain [dB]	2.5	7	25

Table 3.4: Satellite communication system data for LCT link

From this analysis, it is concluded that the power required for the communication payload to transmit data to lunar users is 18.25 W, to which an increment of 20% is applied according to ESA margin philosophy for science assessment studies [104], resulting in a transmission power requirement of 22 W.
From the obtained results, it can be initially stated that concerning the payload power, the mission architecture involving communication solely with lunar sites allows the use of a small-sized satellite, comparable to Hemeria's spacecraft. So a 50 W payload could suffice, leading to a total power of about 220 W using SMAD relationships.

Communication with Earth DSN and lunar surface elements

For the analysis of the second mission architecture, the same simulation scenario as the previous one was used. However, in this case, the satellite's antennas were directed towards the Earth's Deep Space Network (DSN), as one can see in Fig. 3.6

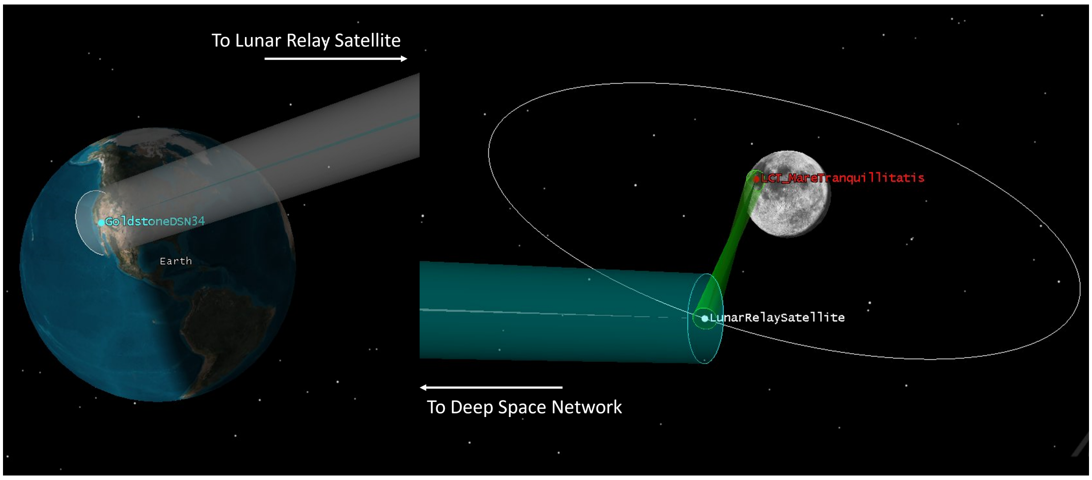


Figure 3.6: Lunar relay satellite communication scenario with LCT and DSN

Before conducting the simulations, a brief investigation was necessary regarding the performance of the antennas in the terrestrial complex. Among the possible antennas within the DSN, the 34-meter Beam Waveguide (BWG) antenna was selected due to its transmission and reception capabilities in the frequency bands of interest, namely S, X, and Ka bands [105], and its characteristics are summarised in Tab. 3.5.

DSN 34m BWG Antenna			
Frequency band	S-band	X-band	Ka-band
Gain [dB]	56.2	66.98	79.58
Transmitter power [kW]	20	20	20
Central frequency [MHz]	2245	8475	26250

Table 3.5: 34 meters DSN antenna characteristics. Credits: [105]

Following the previously described procedures, an initial collection of satellite communication data was compiled in Tab. 3.6, including gain and power for each of the 3 available frequencies [6] [106].

Satellite antenna solutions			
Frequency band	S-band	X-band	Ka-band
Gain range [dB]	2-5	10-20	30-40
Antenna type	Dipole	Horn-Helix	Parabolic-Phased array

Table 3.6: satellite to DSN communication characteristics

The mean gain values for each antenna have been chosen for the analysis. The data considered constant for the simulations are the same as those in the previous scenario. Meanwhile, Tab. 3.7 summarizes the data rate requirements for each band that the spacecraft platform must meet.

Data requirements	
Frequency band	Data Rate [kbps]
S	256
X	10000
Ka	25000

Table 3.7: Data rate requirements for DSN link

In order to satisfy the required performances, during the simulations was made a trade-off on the satellite powers for the various communication links.
The analysis results are summarised in Tab. 3.8

Satellite to DSN communication system			
TRANSMITTER			
Frequency band	S-band	X-band	Ka-band
Frequency [MHz]	2245	8475	26250
Signal bandwidth [MHz]	90	40	1500
Transmitter power [W]	75	200	10
Gain [dB]	3	15	35
Data rate [kbps]	256	10000	25000
RECEIVER			
Frequency band	S-band	X-band	Ka-band
Frequency [MHz]	2070	7212.5	22850
Signal bandwidth [MHz]	90	40	1500
Gain [dB]	3	15	35

Table 3.8: Satellite communication system data for DSN link

From the results of the analysis of this second architecture, it was possible to evaluate that satellites, in the case of communication towards the DSN antennas, require a transmission power of about 300 W, which, as previously done, must be increased by 20%, reaching a value of 360 W. In this scenario, the spacecraft must also communicate with lunar terminals; therefore, using the outcome of the previous analysis, 20 W are added resulting in a total of 380 W for data transmission. By applying the relationships between masses and powers of the subsystems present in the SMAD, an estimate of the overall power of approximately 830 W is obtained. This result is also in line with what was defined by [53], where an estimated power budget for the lunar satellite of approximately 700 W is provided. In this case, using an Iridium like platform could be a good choice

3.2.3 Trade-off on the constellation

Established the mission architecture and the satellite platform to adopt from previous analyses, this paragraph delves into the methodologies that will be presented in the subsequent chapter to select the best orbits to place the constellation's satellites. Simulations were conducted in multiple phases:

1. Evaluation of coverage capacity with a variable number of satellites in different orbits towards the identified lunar sites of interest. The mission duration time was set at 5 years, considering a CubeSat lifespan in LEO of about 8 as a reference [99]. This allowed the identification of potential alternatives in terms of orbits number and satellites per orbit, ensuring communication for at least 90% of the analysis time and meeting the specified requirements.

2. Orbit propagation to determine their stability throughout the mission's duration in terms of primary orbital elements excursions. This enabled a preliminary selection of orbits for further in-depth analysis.

The outcome of these simulations consists of several potential solutions represented by a varying number of satellites and orbits.
Among the coverage, the possibility of extending communication to other sites of potential scientific interest was evaluated. Furthermore, the ability of Lunar Communication Terminals (LCT) to communicate with the terrestrial network was assessed to emphasize the feasibility of the architecture for the intended purposes.
Upon concluding these preliminary analyses, a more detailed study of coverage was undertaken. Specifically, this involved examining the distribution of communication gaps in terms of duration compared to access times and how these gaps affect the efficacy of the constellation under examination.
Subsequently, the simulations also focused on verifying the station-keeping maneuvers necessary to maintain the orbits in their configuration. These latter analyses particularly encompassed determining the required DeltaV, estimating the amount of fuel, maneuver times, and so defining the necessary propulsion system: chemical or electrical. This allowed for the completion of the satellite design.

3.3 Point design definition

After completing the preliminary analyses that allowed the selection of the mission architecture and a feasibility verification of using a small satellite in accordance with the requirements, it is now possible to outline a base design for the satellite

platform. This layout will be further refined based on the results of subsequent analyses.
The aim is to detail more precisely the power and mass requirements for the communication system (payload), electrical power system (EPS), and propulsion system, in order to evaluate the overall mass and power needed for the satellite.
To achieve this, one must refer to the results of the analyses conducted on the communication system, which highlighted a power requirement of approximately 20 W for data transmission.
This allows to take Hemeria's CubeSat as indication, which anticipates a total mass of about 20-22 kg, a payload power of 50 W and a volume ranging between 12U and 16U. However, it is important to emphasize that these are preliminary solutions.
For instance, regarding the propulsion system, further analyses will allow to define the necessary engine and fuel quantity, contributing to the overall mass and power calculations and eventually allowing for the definition of a feasible solution.

Chapter 4

Mission analysis framework

4.1 STK analysis workflow

To effectively present the simulations conducted across various scenarios and the subsequent results, it is crucial to include preliminary evaluations and considerations. These are essential not only to justify the made choices, but also to facilitate a better understanding.
The simulations followed a process that can be outlined through Fig. 4.1

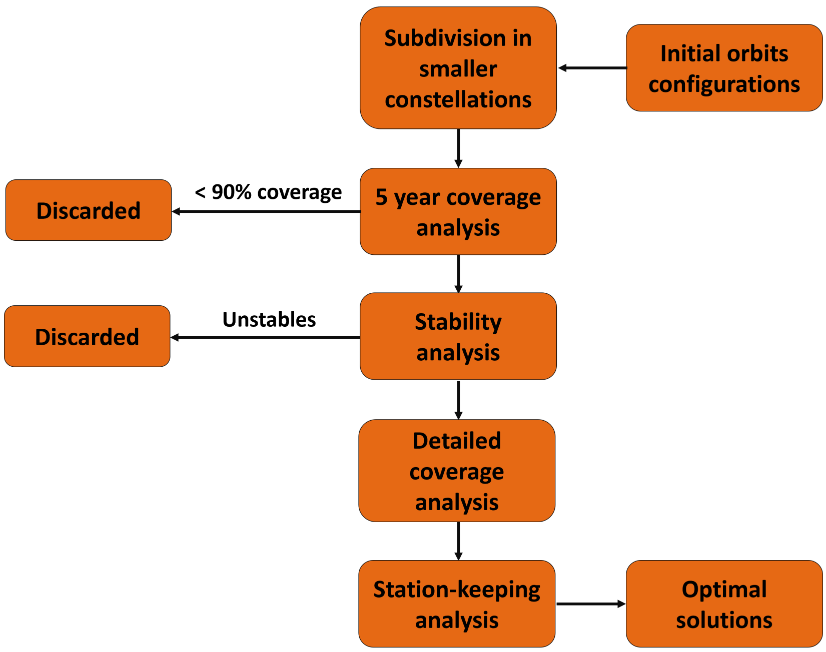


Figure 4.1: Workflow of STK simulations

Initially, the orbits taken from literature were considered individually or grouped in pairs, each containing a varying number of satellites, as some of the identified constellations appeared oversized concerning the analyzed coverage objectives. These configurations were assessed for coverage of the sites of interest throughout the entire mission duration, which spans 5 years. Combinations that didn't achieve at least 90% coverage for the entire duration were discarded.
Subsequent simulations aimed to assess stability by analyzing variations in the primary orbital elements. Unstable trajectories were eliminated, while the others were further analysed in detail regarding their coverage performances. Once the analyses were refined, the final simulations focused on evaluating the necessary station-keeping to maintain the orbits as optimally operational as possible, thus preventing potential collisions with satellites already in orbit around the Moon.

4.2 Coverage analysis: considerations and methods

4.2.1 Simulation constraints

In all simulations, to ensure greater accuracy, a minimum elevation angle of 8 degrees was set for the terminal sensors on the lunar surface. This is defined as the angle between the horizon's plane and the connecting line between facility and satellite as indicated in Fig. 4.2 with ε_0.

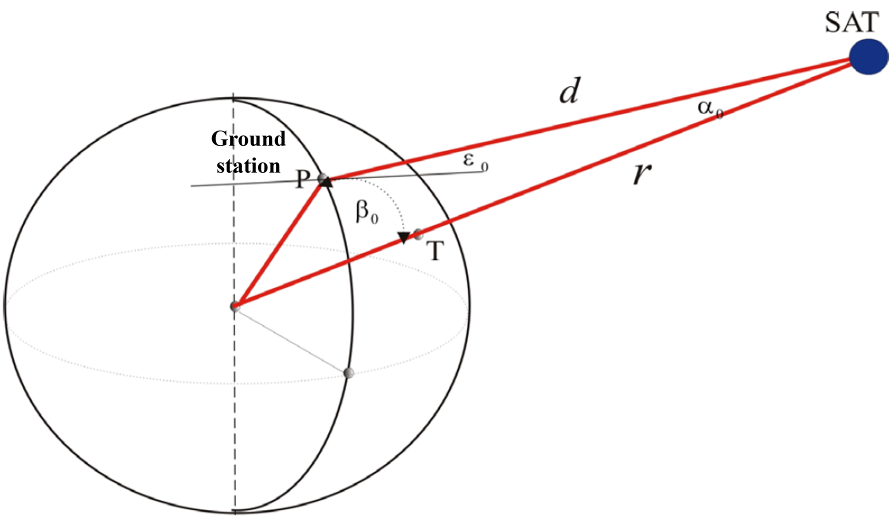


Figure 4.2: Elevation angle between ground station and satellite. Image Credits: [107]

The elevation angle represents a physical constraint for line-of-sight visibility, determined by both the characteristics of the ground antenna and the geological configuration of the location where it is positioned [107].

4.2.2 Assessing the impact of communication gaps on coverage

After an initial assessment of the coverage provided by individual orbits, these were paired in various combinations. Simulations were then conducted with an increasing number of satellites strategically positioned, until the required access duration was achieved. Specifically, efforts were made to minimize both the number of orbits and the spacecraft per orbit.
Evaluated the stability of the orbits that met the coverage requirements, a more thorough review of communication was carried out. Specifically, having selected orbits with similar coverage characteristics for the entire mission duration, subsequent simulations were conducted over a shorter period to quantify the occurrence of communication interruptions. So simulations were performed throughout an entire lunar rotation, because a more detailed examination of communication in the different relative positions between the satellites and lunar terminals was possible. The chosen simulation interval was 30 days, considering that the time required for a complete lunar rotation is about 27 days 7 hours and 43 minutes [108].
Having verified that within the shortened 30-day period, the coverage percentage on the relevant sites remains above 90%, it is necessary to evaluate the impact of the existing gaps. Specifically, the aim is to determine whether, during the periods of interruptions, these outweigh the duration of communication in that time frame. It is important to assess that, in the case of long-duration gaps, there is at least a brief daily interval available for communication. To achieve this, the average value of the gap duration within the occurring interval will be calculated. This is sufficient to quantify their impact once it has been confirmed that indeed there is at least a small daily communication window available.
In periods where continuous communication is not possible, high-risk missions will not be scheduled in order to avoid issues.

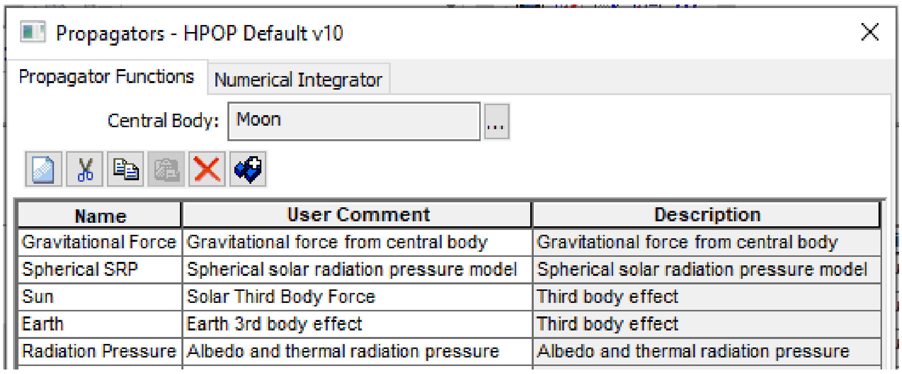
4.2.3 Additional considerations

A final coverage analyses also encompassed the secondary areas of interest, identified as 'exploration sites'; however, these findings did not significantly contribute to the final process of determining the optimal constellation. It was, nonetheless, intriguing to assess the intervals throughout the mission where coverage could be ensured in these secondary sites, proving beneficial for the planning of exploration missions.

4.3 Orbit perturbations and control mechanisms

4.3.1 Orbital propagator

In the stability analysis, in order to achieve the most accurate results, main disturbances affected the orbit were taken into account, using the High-Precision Orbit Propagator (HPOP) provided by STK. The HPOP employs numerical integration of motion's differential equations to produce ephemeris data. It can encompass various force modeling factors within its analysis, such as a full gravitational field model (utilizing spherical harmonics), third-body gravity, atmospheric drag, and solar radiation pressure [109]. The elements used in the propagator are depicted in the Fig. 4.3 taken from STK scenario.



Name	User Comment	Description
Gravitational Force	Gravitational force from central body	Gravitational force from central body
Spherical SRP	Spherical solar radiation pressure model	Spherical solar radiation pressure model
Sun	Solar Third Body Force	Third body effect
Earth	Earth 3rd body effect	Third body effect
Radiation Pressure	Albedo and thermal radiation pressure	Albedo and thermal radiation pressure

Figure 4.3: Functions of the STK HPOP propagator used

Referring to Fig. 4.4, the main orbital parameters used to assess the stability method will be defined [108].

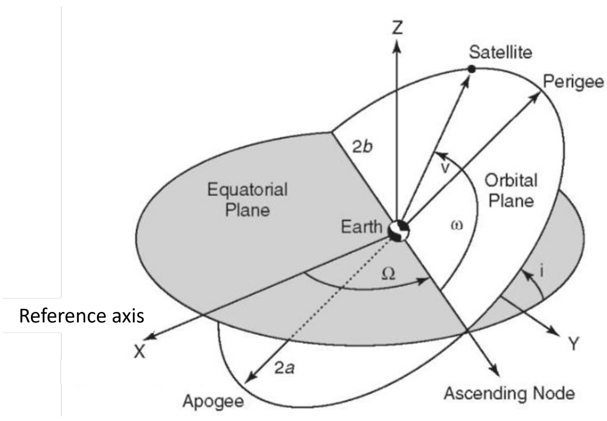


Figure 4.4: Classic orbital elements. Image Credits [108]

The figure identifies the parameters in the Earth's inertial reference frame, but they are the same in any central body.

- **Periapsis Altitude** represents the satellite's altitude at its closest point to the lunar surface.
- **Apoapsis Altitude** is the satellite's altitude at its farthest point from the lunar surface.
- **Right Ascension of the Ascending Node (RAAN or** Ω**)** indicate the angle measured on the equatorial plane between a direction in the reference system of the central body and the **Ascending Node**. The Ascending Node is where the orbit intersects the equatorial plane as the satellite moves from the southern hemisphere to the northern.
- **Argument of Periapsis (**ω allows defining the periapsis position and is the angle formed between the vector pointing to the ascending node and the one indicating the periapsis direction.
- **Inclination (i)** is the angle formed between the equatorial plane and the orbital plane.

The stability of the orbits was primarily assessed through two distinct approaches. In the first method, the time evolution of the main orbital parameters was considered, referring to the information presented in Chapter 1. It was observed that Frozen Orbits exhibit variations in orbital elements while maintaining constant on average. This ideally periodic phenomenon is seen as a desirable behavior that the system strives to approach. In the second method, the relative value of the Right Ascension of the Ascending Node (RAAN) between orbits was examined. This evaluation using Ω determines the relative phasing of orbits, to exclude those with undesired overlaps.

4.3.2 Orbital mechanics: how to change classical elements

After these analyses, comprehensive simulations were conducted on the coverage, as previously mentioned. Consequently, the final orbits ensuring the desired coverage for the sites of interest and adequate stability have been derived. To finalize the constellation selection and gather all the necessary components for completing the preliminary satellite design, an analysis on orbital control maneuvers has been carried out.
In order to justify the adopted station-keeping strategies, it's necessary to recall the possible maneuvers to be carried out, how they are executed, under what

circumstances their propulsion cost is minimal, and which ones are the most expensive. Only impulsive maneuvers will be considered, meaning those in which the thruster with infinitesimal burn made the required velocity change (DeltaV) instantaneously. All these concepts are defined through orbital mechanics [108] [110].
It is anticipated that, generally, maneuvers carried out within the starting orbital plane are cheaper in terms of DeltaV and thus require less propellant.

Altitude changing

Regarding the change in altitude, this is accomplished by making a velocity variation with a thrust in the orbital plane itself, particularly in a direction tangential to the trajectory. In the case of elliptical orbits, thrusting at periapsis will affect the apoapsis, and vice versa for periapsis with the minimum amount of ΔV. For circular orbits, pushing at any point will alter the altitude at the diametrically opposite point. When considering an increase in altitude between two elliptical trajectories, like that in Fig. 4.5, the Eq. 4.1 apply, where the mechanical energy of the two orbits has been equated.

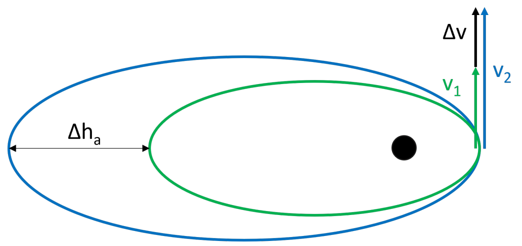


Figure 4.5: Single impulse maneuver for apoapsis altitude increase

$$\Delta E = E_2 - E_1 = \frac{v_2^2}{2} - \frac{\mu}{2a_2} = \frac{v_1^2}{2} - \frac{\mu}{2a_1} \tag{4.1}$$

Following a ΔV defined by Eq. 4.2

$$\Delta V = v_2 - v_1 = \sqrt{v_1^2 - \frac{\mu}{a_2} + \frac{\mu}{a_1}} - v_1 \tag{4.2}$$

Where v_1 is the initial velocity at the periapsis of the first orbit, and v_2 the velocity after the impulse in the same point. μ represent the gravitational constant of the central body and a the semi-major axis of the orbit. The same approach apply to a reduction of the height, in that case the velocity variation will be negative. This method is the most cost-effective in terms of ΔV, but requires the longest time to be executed. It can be considered as a particular case of a Hohmann transfer, that is the cheapest two-impulse maneuvers used to modify both the altitude at Apoapsis and Periapsis.

Argument of periapsis changing

This maneuver is particularly inefficient among those that can be performed with a single in-plane impulse. It involves changing the argument of periapsis and can be outlined using Fig. 4.6

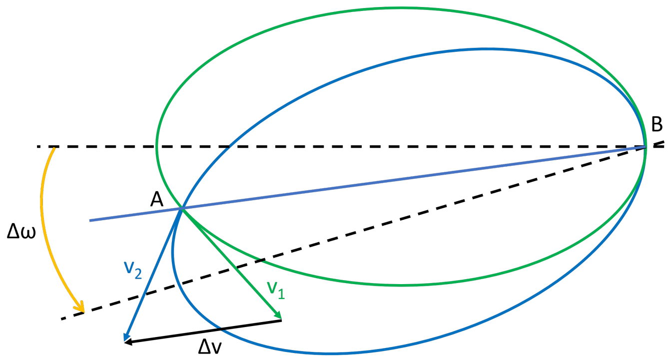


Figure 4.6: Single impulse maneuver for argument of periapsis variation

In this case, there is no alteration in altitudes, so there are no changes in the orbital energy. However, the maneuver will incur a propulsion cost, which remains the same whether conducted at point A or B, and it can be expressed with Eq. 4.3.

$$\Delta V = 2v_r = 2\frac{\mu}{h}e\left|sin\left(\frac{\Delta\omega}{2}\right)\right| \tag{4.3}$$

Where h is the orbit momentum vector and e is the eccentricity. Particularly, the thrust direction must be parallel to the line connecting the two intersection points of the initial and final trajectories.

Plane changing: inclination control

A change in velocity, occurring within the orbit's plane, can modify either its size, shape, or rotate the line of Apsides. Altering the orientation of the orbital plane in space necessitates a ΔV component perpendicular to the plane of the orbit. If, following the application of a finite DeltaV, the satellite's speed and flight-path angle remain unaltered, it indicates a mere adjustment in the orbit's plane. This is called a simple plane change. An instance of a simple plane change is transitioning from an inclined orbit to an equatorial one, illustrated in Fig. 4.7. The alteration in the orbit's plane occurs at an angle, i. The initial and final velocities are identical in magnitude and, along with the required ΔV, constitute an isosceles vector triangle as seen in right of Fig. 4.7. Assuming knowledge of the actual orbit's velocity and desired final i, DeltaV can be calculated using Eq.

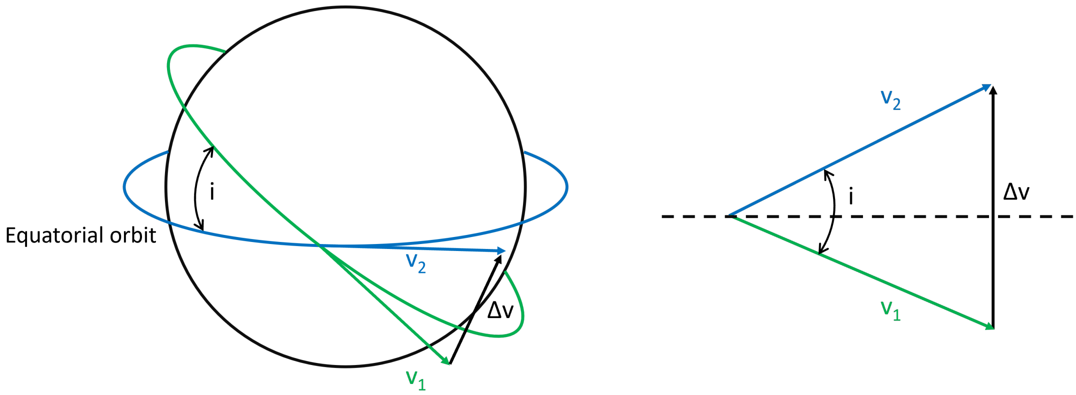


Figure 4.7: Simple plane change through an angle i

$$\Delta V = 2v\, sin\frac{i}{2} \tag{4.4}$$

If the object of the maneuver is exclusively a plane change, the most effective way to do so is at one of the node, ascending or descending. If the maneuver is performed at any point along the trajectory between the nodes and the apoapsis or periapsis, in addition to a Δi variation, there will also be a $\Delta\Omega$, which indicates a change in the right ascension of the ascending node, as can be seen in Fig. 4.8

Where Ψ, indicate the satellites flight path angle. Therefore, if one intends to execute an inclination change in this specific case, it will necessitate accepting a modification of the RAAN, requiring a larger DeltaV. Furthermore, the maneuver will be carried out based on the variation of the flight path angle according to Eq.4.5

$$\Delta v = 2v_\theta\, sin\left(\frac{\Delta\Psi}{2}\right) \tag{4.5}$$

The velcity v_{theta}, is the component tangent to trajectory. And the Eq. 4.6 correlates i and Ψ.

$$\Delta\Psi = \Psi_2 - \Psi_1 = arcsin\left(\frac{cos(i_1 + \Delta i)}{cos\delta_M}\right) - \left(\frac{cos\, i_1}{cos\delta_M}\right) \tag{4.6}$$

Where δ_M is the latitude of the maneuver point. So when the satellite is at a node, the flight path angle and δ_M are zero so the maneuver induces a change solely in inclination.

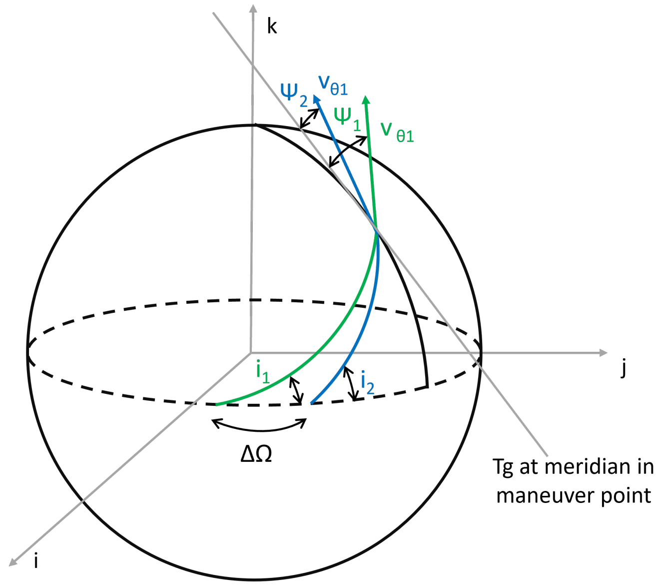


Figure 4.8: Plane change through flight path angle variation

Plane changing: RAAN control

The control of RAAN is an inclination changing. Actually, as previously discussed, when a Δv is impressed in a point that is not a Node, Δi comes with $\Delta\Omega$. So if one wants to perform a control on the right ascension of the ascending node, reducing the propulsion cost and avoiding inclination changes, this should be conducted at the Apoapsis or Periapsis, as shown in the Fig. 4.9

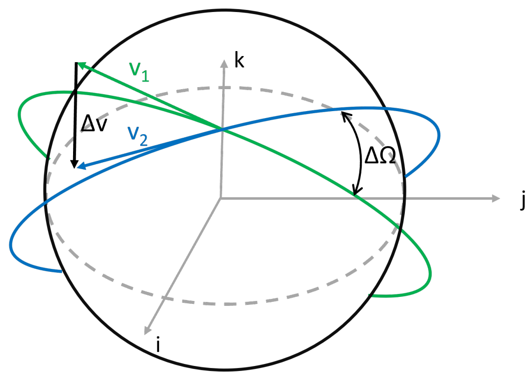


Figure 4.9: Simple plane change through an angle Ω

4.4 Station keeping strategies and application

The briefly outlined methodologies refer to ideal impulsive maneuvers, which have been adopted as a basis for defining the station-keeping process aimed at evaluating strategies for controlling orbital parameters. However, in reality, impulsive maneuvers do not exist; therefore, correction phases are characterized by a finite time. Additionally, even if small, there are always alignment errors between the thrust direction and the ideal direction for the maneuver. These two factors result in the displacement of the maneuver point throughout the entire combustion phase, which combined with the misalignment of the thrust, causes also minimal change in other orbital elements.

4.4.1 Absolute and relative station keeping

Station-keeping is crucial to maintain satellite intended position and orbital parameters. These maneuvers ensure that spacecrafts remain within specified orbital slots, enabling uninterrupted communication, accurate data collection, and effective operation of satellite-based systems like telecommunications. Perform station-keeping is essential in order to avoid orbital drift, potentially causing collisions or rendering the satellite ineffective for its intended purpose. Thus, station-keeping maneuvers are vital to ensure the longevity and functionality of satellites in orbit [111]. Initially, it's important to differentiate between absolute station-keeping and relative station-keeping. Absolute station-keeping ensures that each satellite remains within a predefined mathematical boundary concerning the central body, whether it's the Moon or inertial space. On the other hand, relative station-keeping focuses solely on maintaining the positions of the satellites in relation to each other without considering absolute positions. The control box denotes the permissible range in orbital parameters that are managed through maneuvers [112].

4.4.2 Ongoing activities around the Moon

For station keeping purposes, it may be interesting an evaluation of what is actually orbiting the Moon, in order to compute preliminary control strategies and avoid collisions.
There are six active spacecraft orbiting the Moon, as illustrated in Fig. 4.10. Within these, NASA's THEMIS mission has repurposed two out of its five probes into ARTEMIS (Acceleration, Reconnection, Turbulence, and Electrodynamics of the Moon's Interaction with the Sun), namely, ARTEMIS P1 and ARTEMIS P2. Both of these spacecraft maneuver in eccentric orbits with low inclinations. Meanwhile, NASA's Lunar Reconnaissance Orbiter (LRO) follows a nearly polar, slightly elliptical orbit. Additionally, India's Chandrayaan-2 and Korea Pathfinder

Lunar Orbiter (KPLO) also navigate in polar orbits at an altitude of 100 km. NASA's Capstone operates within a 9 : 2 resonant southern L2 Near Rectilinear Halo Orbit (NRHO). Its Perilune travels over the lunar North pole at an altitude of 1500 to 1600 km, while the Apolune extends over the South pole, approximately 70000 km away. China's Chang'e 4 mission's data relay satellite Queqiao, was later relocated to a halo orbit near the Earth-Moon Lagrange point L2 [113].

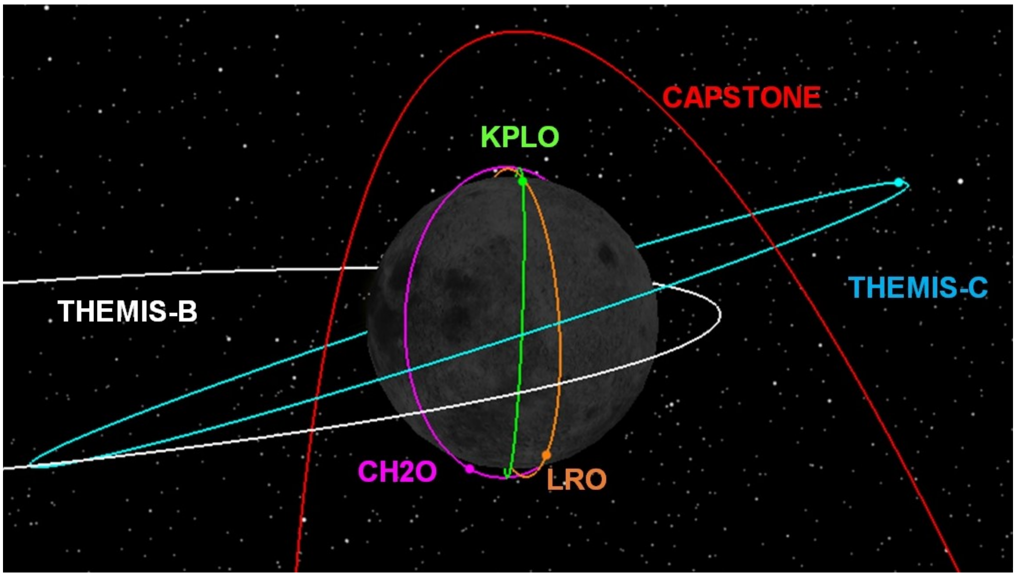
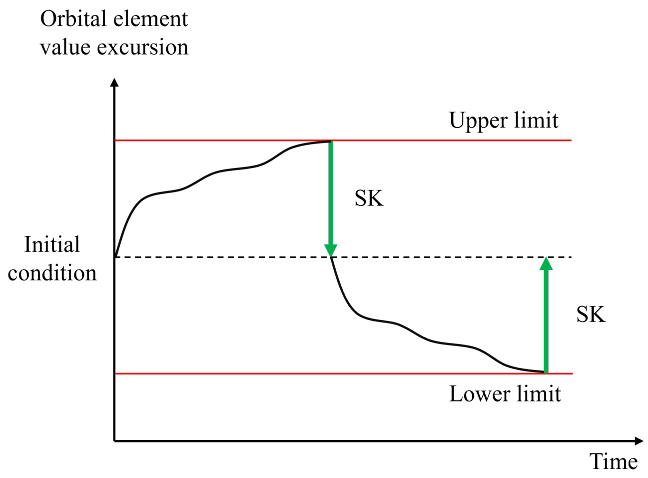


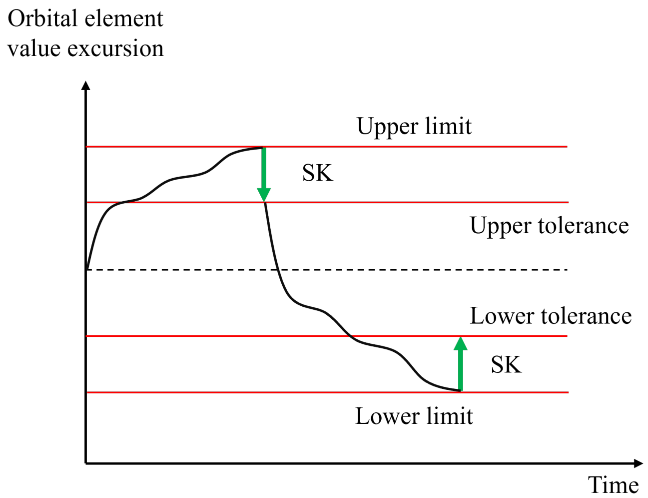
Figure 4.10: Satellites orbiting the Moon as of July 2023. Image Credits: [113]

4.4.3 STK implementation

During the conducted analyses, the station-keeping was exclusively carried out in an absolute manner. Specifically, the adopted strategy can be seen as aimed at maintaining communication performance for the mission while seeking to minimize propulsion costs.
Two strategies have been evaluated, schematically represented in images *a* and *b* of Fig. 4.11.
In both cases, upper and lower threshold values have been established, considering the excursions exhibited by uncontrolled propagated orbits. In particular, in the first case, an attempt was made to leverage the natural periodic variations typical of "frozen orbits" to reduce the number of maneuvers to be performed and consequently the propulsion cost, by restoring the orbital parameters to their initial values at opportune moments. In the second case, in order to further reduce the propulsion cost, albeit at the expense of a greater number of required maneuvers, tolerances with reduced values compared to the upper and lower limits were defined. These



(a) *Station keeping maneuver to initial value*



(b) *Station keeping maneuver to tolerance values*

Figure 4.11: Station keeping strategies implemented in STK

tolerances aimed to bring back the orbital element within these limits with the control maneuver, still maintaining its mean value constant throughout the entire mission.
The two strategies just described have been applied to satellites in the studied STK scenarios. Constraints have been imposed on the maximum and minimum values assumed by parameters to be controlled during the propagation of spacecraft along their orbits. Throughout their orbital paths, satellites traverse specific positions on the trajectory where it is advantageous to perform maneuvers, as discussed in

chapter 4.3, namely the ascending node, periapsis, descending node, and apoapsis. Whenever a satellite reaches one of these positions, an assessment is made to determine if the monitored parameter has exceeded the predetermined threshold. If so, the adequate maneuver is executed.

4.4.4 Selected propulsion technology for satellites control

When deliberating over the choice between chemical and electrical propulsion for the satellite station keeping, several crucial factors came into play. Given the small size of the satellite and the specific goal of maneuvers, it was imperative to consider a propulsion system that aligns with these parameters, so it is obvious that choice falls on small thrusters. The primary objective revolved around selecting propulsion mechanisms tailored explicitly for station keeping. Hence, the focus was on identifying and adopting compact propulsion systems that have demonstrated efficiency in similar satellite-scale applications. These selected propulsion systems are intended to drive the simulations, allowing for a comprehensive evaluation of their effectiveness across multiple criteria.
The evaluation encompass an assessment, examining various performance metrics including mass optimization, power efficiency, fuel utilization, and the time required to execute the necessary maneuvers. As the satellite primarily requires periodic adjustments to maintain its intended orbit, the emphasis lay in identifying propulsion mechanisms that could deliver the requisite thrust efficiently within the constraints of the satellite's size and power availability.
For the intended purposes, two propulsion systems have been chosen, specifically one chemical propulsor and one electric from the available market solutions designed for cubesat applications. The characteristics of both engines are summarized in the table below, which represents their data sheets and parameters used for conducting simulations, derived from [114] [115].

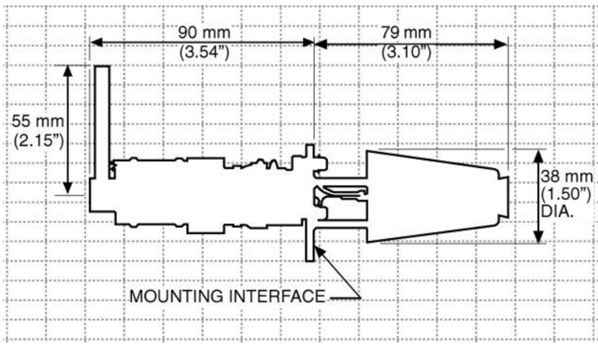


Figure 4.12: Chemical propulsion system MR-111C from Aerojet Rocketdyne. Image Credits: [114]

Aerojet Rocketdyne MR-11C	
Propellant	Hydrazine
Thrust [N]	4
Power [W]	16
$Mass_{dry}$ [kg]	0.33
Specific Impulse [s]	220
Dimensions [mm]	170x55x38

Table 4.1: Aerojet Rocketdyne chemical propulsion data

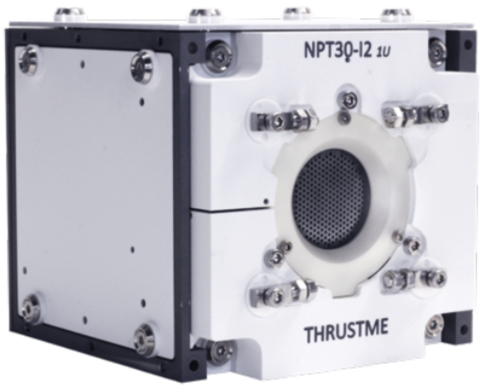


Figure 4.13: Electrical propulsion system NPT30-I2 from ThrustMe. Image Credits: [115]

ThrustMe NPT30-I2	
Propellant	Solid Iodine
Thrust [mN]	0.3-1.1
$Power_{avg}$ [W]	45
Massdry [kg]	1.2
Specific Impulse [s]	1200
Dimensions	1 U

Table 4.2: ThrustMe electrical propulsion data

4.5 Trade-off methodology: Analytical Hierarchy Process

In the process of determining the optimal orbit for mission purposes, a comprehensive **Trade-off** with **Analytical Hierarchy Process** has been undertaken. This process started by defining specific Figures of Merit (FOM), each carrying an assigned weight crucial to the assessment.
Every orbit under analysis underwent evaluation, wherein all associated FOMs were attributed values basing o their importance over mission objective. Subsequently, by multiplying the weight of each FOM by the respective score of the orbit and summing these products, a comparative analysis of results was conducted to discern and select the better orbit for the mission's objectives [116].

The identified Figures of Merit were as follows:

- **Total duration of communication**: assessment of the communication coverage each configuration provided over specific regions.
- **Gap distribution**: how the frequency and distribution of communication gaps influences the goodness of coverage.
- **Eclipse time**: quantification of the time during which satellites remain in umbra, impacting power availability and storage.
- **Station-keeping ΔV**: the required change in velocity for satellite orbital control, express the ease of maneuvering and assess the application of different propulsion systems.
- **Station-keeping maneuver time**: estimation of the time needed for maneuvers in each orbit, influencing operational efficiency. Fuel for station-keeping: assessment of fuel consumption required for maintaining satellite positions within each orbit.
- **Number of maneuvers**: frequency of maneuvers necessary for orbital maintenance and stability within each orbit.
- **Number of satellites**: encompass the ability to reach the designated goals with minimum number of spacecrafts.
- **Number of orbits**: define the number of orbits required for each constellation, as a mean of evaluating complexity.

These Figures of Merit collectively formed a comprehensive framework for systematically assessing and comparing the various orbits under consideration. To examine the FoMs relative importance, it has been made a **Prioritization Matrix**,

defining a normalized final value. In this manner, all FOM are compared to each other, defining a characteristic **weight** used to evaluate the configurations.
The hierarchical process progresses by assessing scores for individual configurations, similar to the methodology used for Figures of Merit (FOMs). Consequently, at the conclusion of this procedure, a final ranking for all constellations is derived, each associated with scores for all the figures of merit. To conclude the process, multiply the scores of each configuration by the corresponding FOM weight figures. This calculation yields a scale where the maximum value is linked to the superior configuration, while the minimum corresponds to the inferior one.

Chapter 5

Mission analysis: simulations and results

5.1 Constellations deployment: preliminary coverage and stability analysis

As previously mentioned, the following chapter will examine the main simulations conducted in STK. The aim is to identify the optimal orbits, fulfilling coverage requirements for the sites of interest while ensuring stability against major disturbances. These analyses will also determine the required number of satellites and define the most suitable propulsion system in terms of maneuver times, mass, power, and necessary fuel, thus completing the preliminary design of the satellites.

5.1.1 Lunar communication terminals to Earth deep space network

The first conducted analysis focused on evaluating the coverage performance provided by the lunar terminals towards the Deep Space Network antennas, in order to reaffirm the feasibility of the chosen architecture. Specifically, the scenario was set up positioning the two LCTs at designated locations on the lunar surface, with the three main DSN antennas respectively located at Goldstone, Madrid, and Canberra. The scenario is depicted for clarity in Fig., while the obtained results are summarized in Tab.

5.1.2 Frozen configuration 1

In Fig. 5.1 is depicted the first analyzed configuration taken from [62] while Tab. 5.1 contain the orbital elements. To simplify the visualization, all the orbits will be represented with only one satellite.

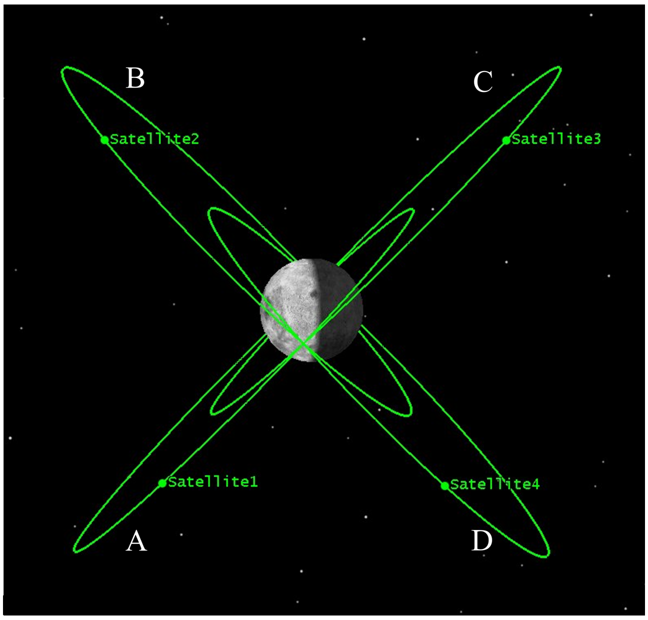


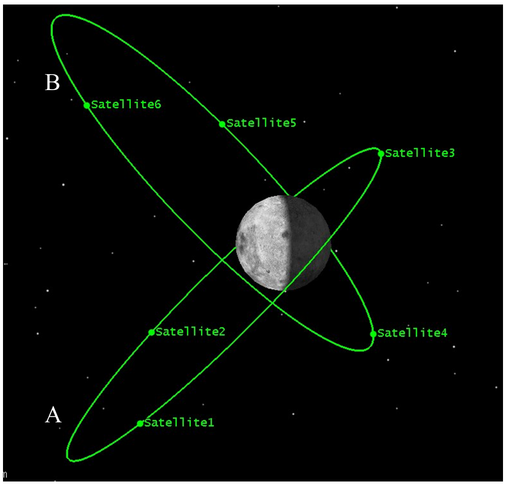
Figure 5.1: Frozen configuration 1

Frozen Configuration 1								
S/C	Plane	Orbit	Period [hr]	a [km]	e	i [deg]	RAAN [deg]	AgP [deg]
1	1	A	18	8049	0.4082	45	0	90
2	1	B	18	8049	0.4082	45	180	270
3	2	C	18	8049	0.4082	45	0	270
4	2	D	18	8049	0.4082	45	180	90

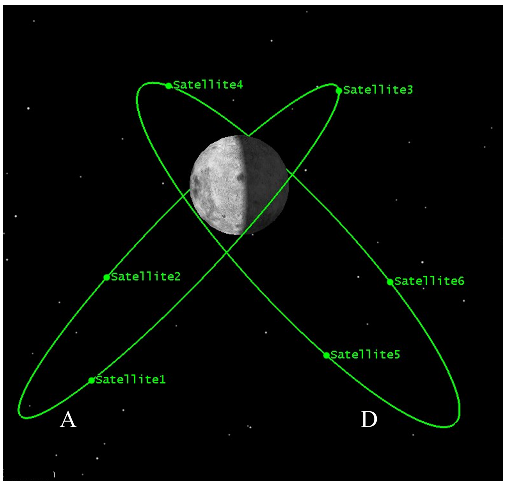
Table 5.1: Frozen configuration 1 orbital elements

The original constellation comprise eight or twelve satellites, equally spaced in the four orbits.
The initial setup was divided into three constellations pairing the orbits **AB**, **AD**, and **BD**. Initially, there were 2 satellites per orbit in the analysis. Eventually, to achieve the required coverage, the final solution consisted of 3 satellites for each orbit, totaling 6 across all constellations, as shown in Fig. 5.2

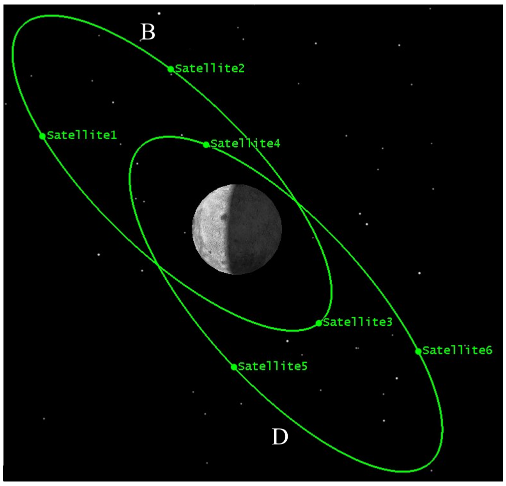
The results of the analysis conducted for the *Frozen configuration 1* constellations, are summarized in Tab. 5.2



(a) *Frozen configuration 1: AB orbits*



(b) *Frozen configuration 1: AD orbits*



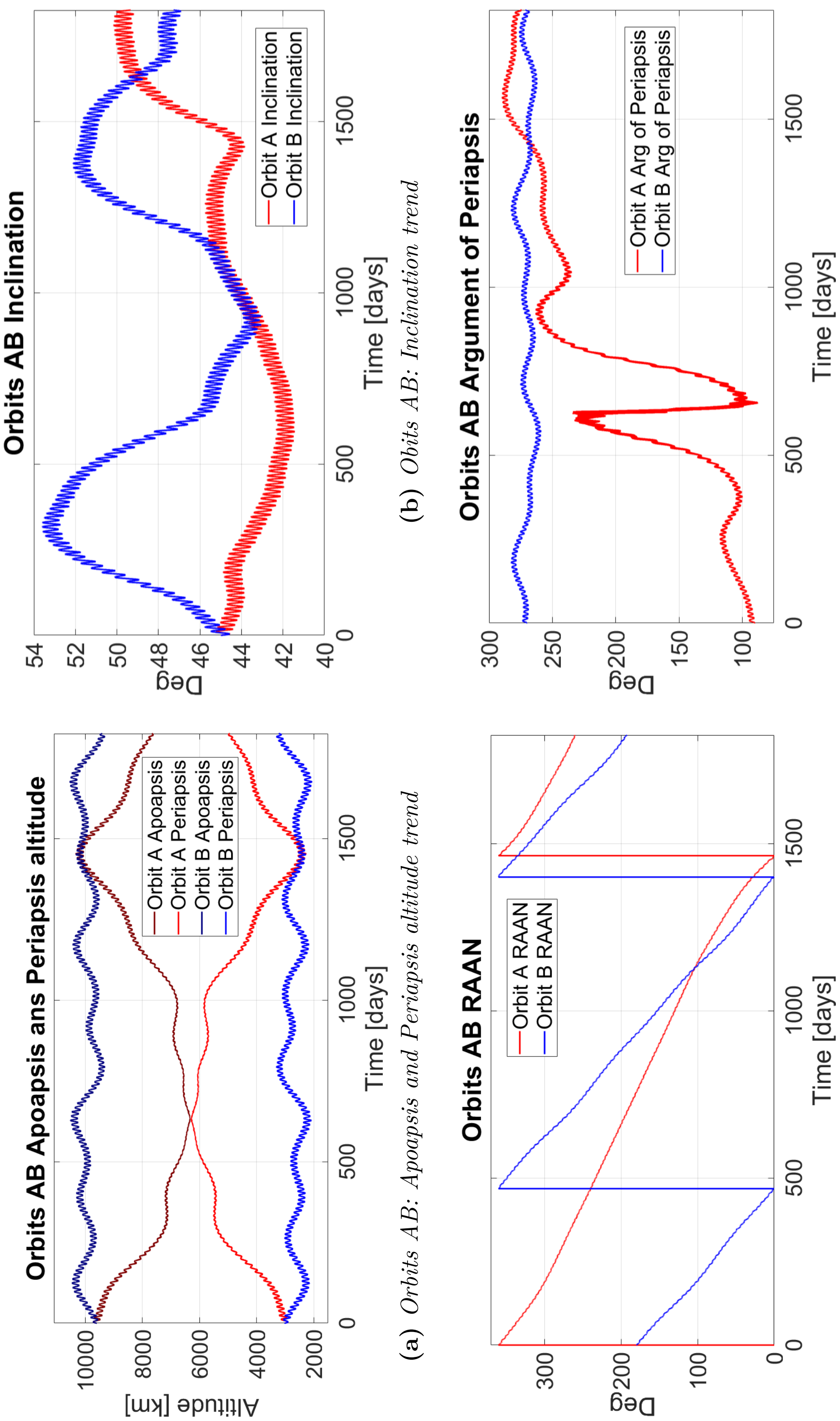
(c) *Frozen configuration 1: BD orbits*

Figure 5.2: Constellations derived by Frozen Configuration 1

Frozen configuration 1			
Orbits	A + B	A + D	B + D
Satellites	6	6	6
Mare Tranquillitatis coverage	88 %	88 %	93 %
Mare Tranquillitatis gap	12 %	12 %	7%
South Pole coverage	96 %	100%	93 %
South Pole gap	4 %	0%	7 %

Table 5.2: Coverage results for Frozen configuration 1 constellations

In these cases, due to the excellent coverage at the south pole provided by the AD constellation, despite the equatorial side has a coverage of slightly below 90%, this constellation proceeded to the next analysis step with BD to stability assessment. The stability analysis lead to the results depicted in Fig. 5.3, 5.4, 5.5 Based on the results, it has been decided to exclude orbit A as it displays a significantly unstable behavior when compared to orbits B and D. Specifically, its use is deemed inconvenient due to the rapid variation in RAAN, causing it to overlap with other orbits, and the change in argument of periapsis from 90 to 180 degrees, responsible for shifting the periapsis from the southern to the northern pole area. Furthermore, the apoapsis and periapsis altitudes vary considerably, deviating from the typical oscillatory behavior observed in frozen orbits. So among the analyzed configurations, the best compromise is the constellation composed of orbits B and D.



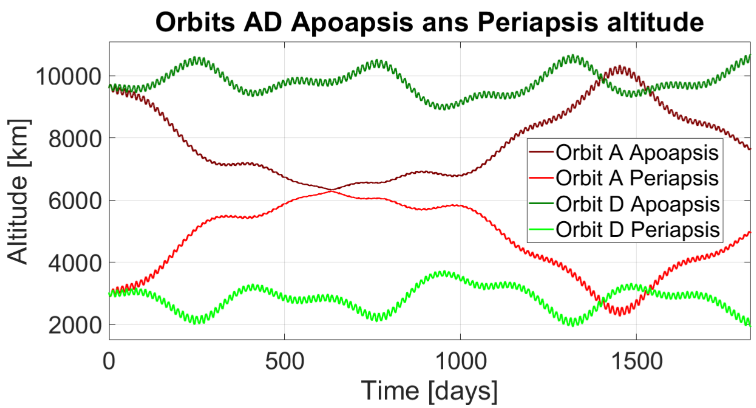
(a) *Orbits AB: Apoapsis and Periapsis altitude trend*

(b) *Obits AB: Inclination trend*

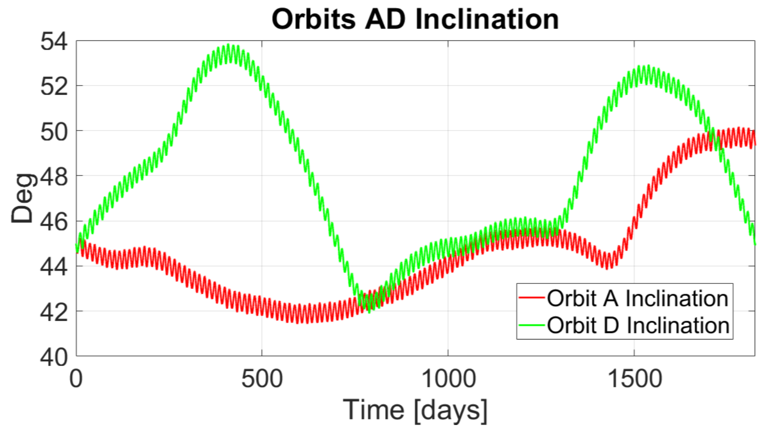
(c) *Orbits AB: RAAN trend*

(d) *Obits AB: Argument of Periapsis trend*

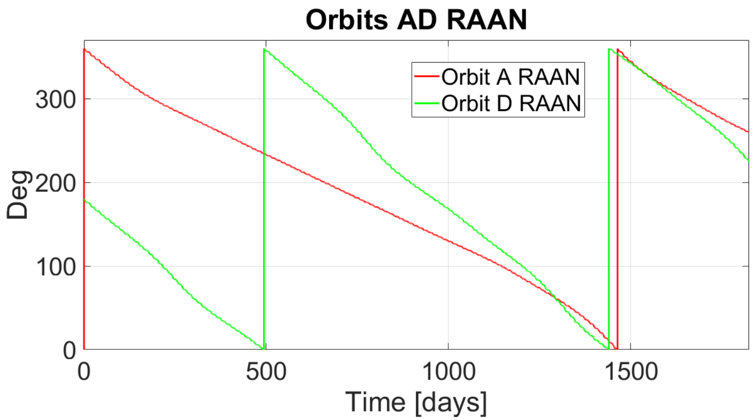
Figure 5.3: Orbital parameters evolution of constellation AB from frozen configuration 1



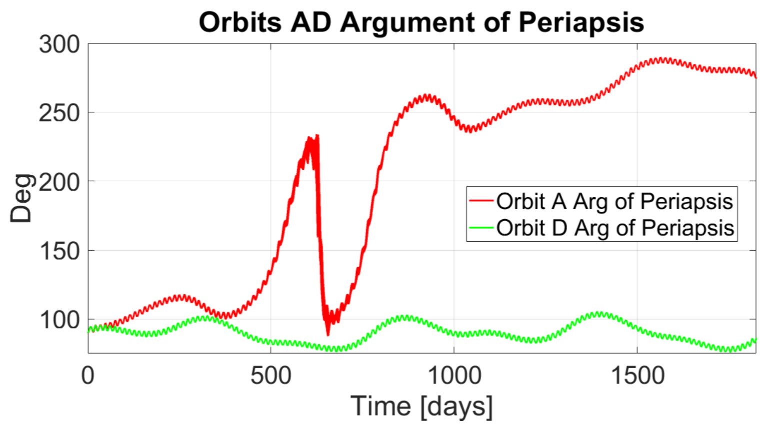
(a) *Orbits AD: Apoapsis and Periapsis altitude trend*



(b) *Obits AD: Inclination trend*

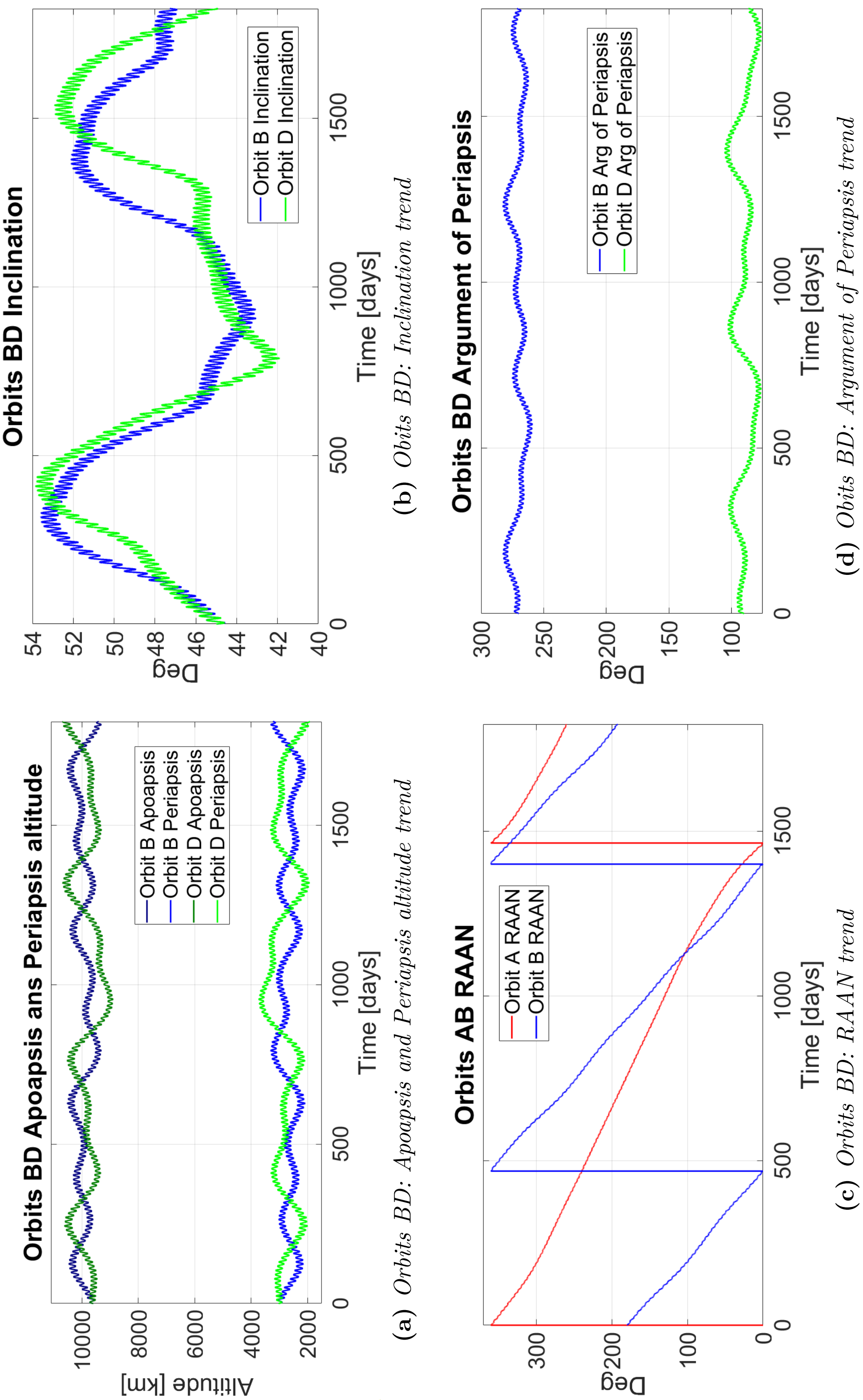


(c) *Orbits AD: RAAN trend*



(d) *Obits AD: Argument of Periapsis trend*

Figure 5.4: Orbital parameters evolution of constellation AD from frozen configuration 1



(a) *Orbits BD: Apoapsis and Periapsis altitude trend*

(b) *Obits BD: Inclination trend*

(c) *Orbits BD: RAAN trend*

(d) *Obits BD: Argument of Periapsis trend*

Figure 5.5: Orbital parameters evolution of constellation BD from frozen configuration 1

5.1.3 Frozen configuration 2

The Fig. 5.6 shown the constellation derived from [67], with parameters in Tab. 5.3

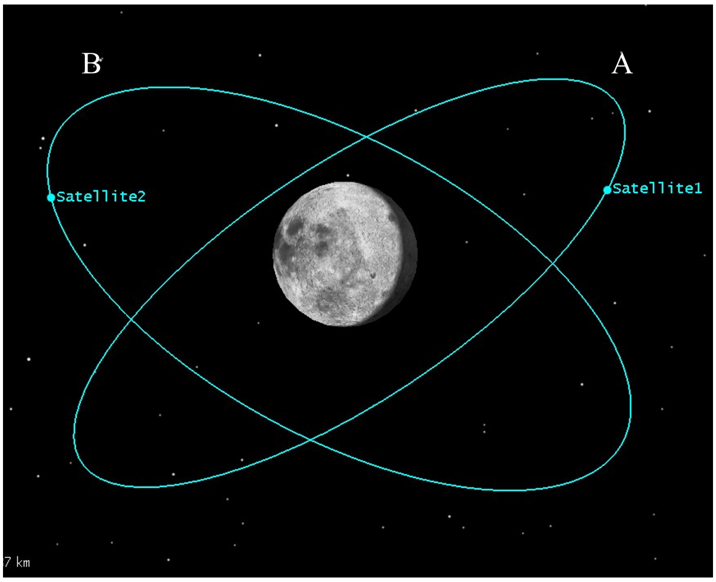
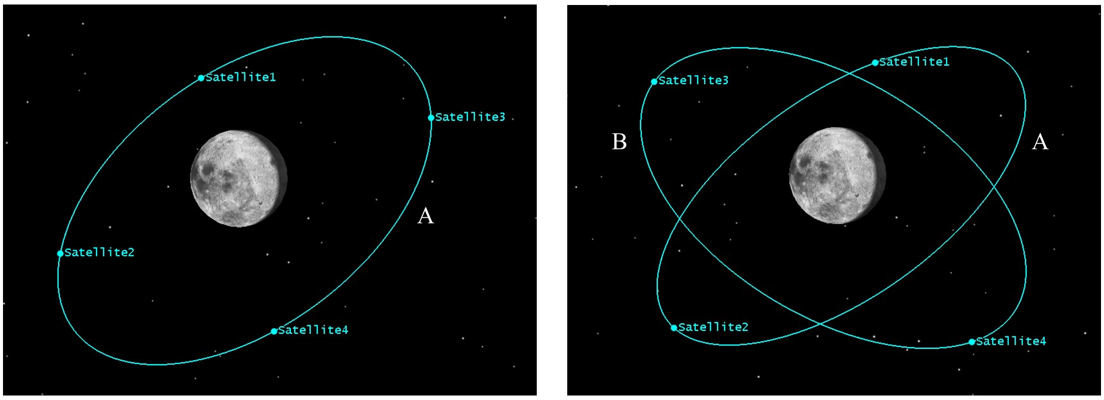


Figure 5.6: Frozen configuration 2

Frozen Configuration 2								
S/C	Plane	Orbit	Period [hr]	a [km]	e	i [deg]	RAAN [deg]	AgP [deg]
1	1	A	16.2	7500	0.05	40	0	90
2	2	B	16.2	7500	0.05	40	90	90

Table 5.3: Frozen configuration 2 orbital elements

The constellation initially comprised 6 satellites, evenly divided between 2 orbits. This configuration was reorganized into two setups, as illustrated in Fig. 5.7. In the first scenario, only the **A** orbit with 4 spacecraft is sufficient to achieve the required communication coverage. The second configuration involves both **A** and **B** orbits of the original, with the number of elements reduced to 4, equally distributed across the trajectories.



(a) *Frozen configuration 2: orbit A* **(b)** *Frozen configuration 2: AB orbits*

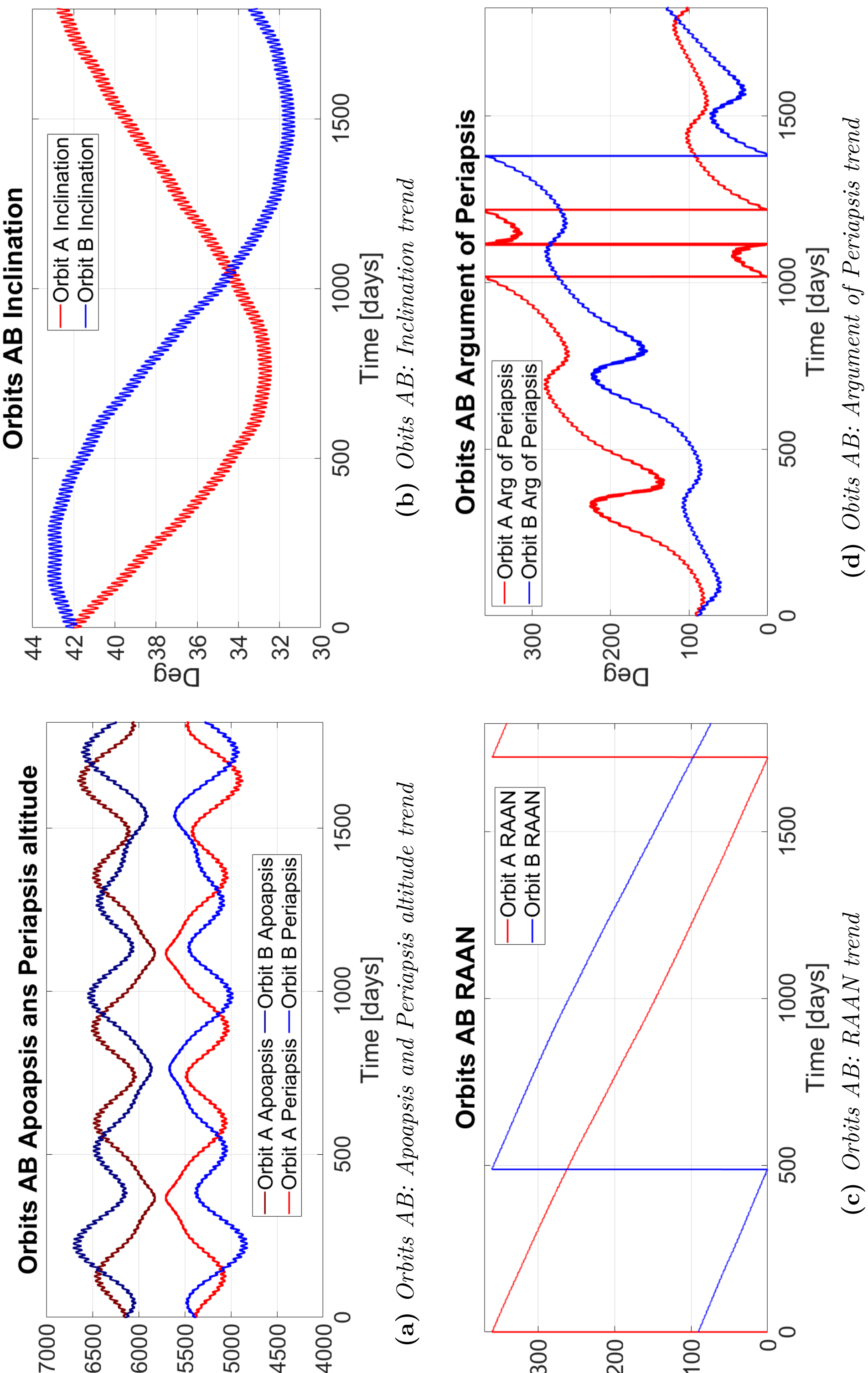
Figure 5.7: Constellations derived by Frozen Configuration 2

The coverage results for the 5 year propagation of the orbits are summarized in Tab. 5.4.

Frozen configuration 2		
Orbits	A	A + B
Satellites	4	4
Mare Tranquillitatis coverage	90 %	91 %
Mare Tranquillitatis gap	10 %	9 %
South Pole coverage	91 %	92%
South Pole gap	9 %	8%

Table 5.4: Coverage results for Frozen configuration 2 constellations

Perturbation of orbits for the 5 year mission duration, allowed to obtain the results of Fig. 5.8. In this case both constellations exhibit a good stability. Furthermore, the configuration composed of orbits A and B shows a good evolution of relative RAAN over time that doesn't require any control to avoid superimposition. The Argument of Periapsis complete rotation in this case is not a problem, because both orbits have a very little eccentricity, so the orientation of Periapsis is not a problem for communication. For what concern inclination, some control may be implemented in order to avoid that an excessive reduction influences coverage performances.



(a) *Orbits AB: Apoapsis and Periapsis altitude trend*

(b) *Obits AB: Inclination trend*

(c) *Orbits AB: RAAN trend*

(d) *Obits AB: Argument of Periapsis trend*

Figure 5.8: Orbital parameters evolution of constellation AB from frozen configuration 2

5.1.4 Frozen configuration 3

In the following Fig. 5.9, there are the second constellation proposed by [67] and orbital elements are in Tab. 5.5 below.

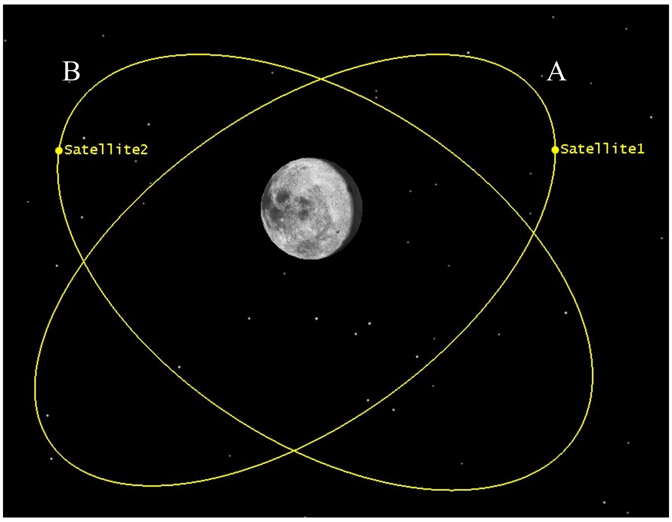
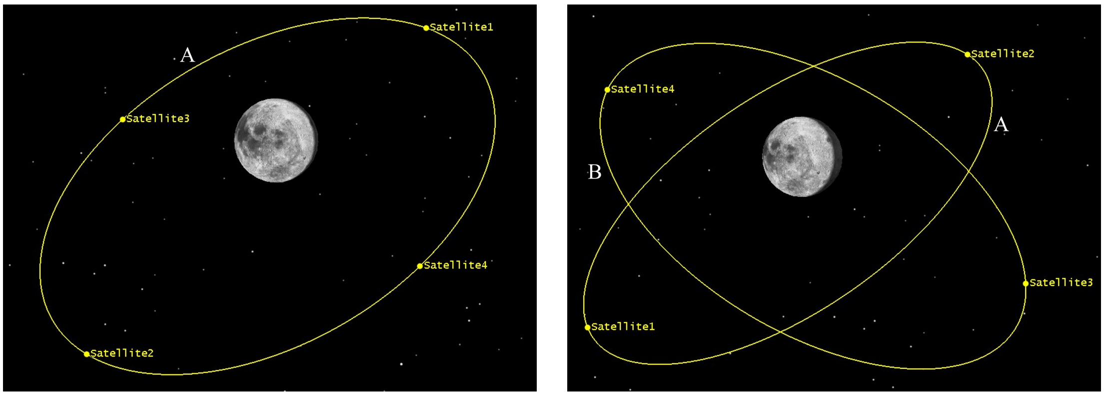


Figure 5.9: Frozen configuration 3

Frozen Configuration 3								
S/C	Plane	Orbit	Period [hr]	a [km]	e	i [deg]	RAAN [deg]	AgP [deg]
1	1	A	24.5	9870	0.185	40	0	90
2	2	B	24.5	9870	0.185	40	90	90

Table 5.5: Frozen configuration 3 orbital elements



(a) *Frozen configuration 3: orbit A* (b) *Frozen configuration 3: AB orbits*

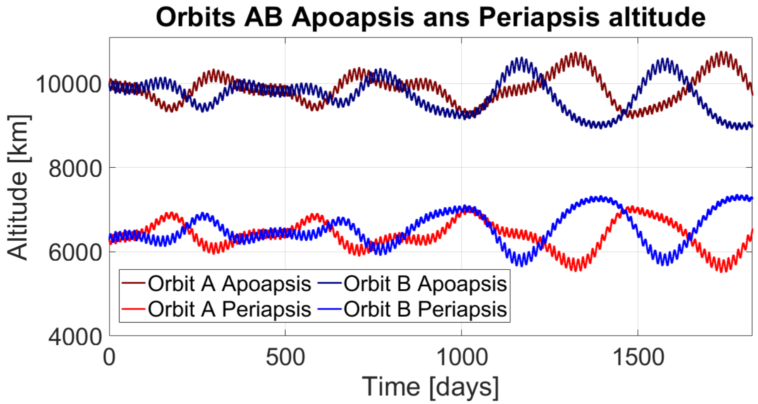
Figure 5.10: Constellations derived by Frozen Configuration 3

This case, very similar to the Frozen configuration 2, consisted of 6 satellites. The results from simulations conducted for the 5-year mission period allowed the identification of 2 constellations that met the requirements, as depicted in Fig. Specifically, both constellations contain 4 spacecrafts, which were positioned in a single orbit in case *a* and across two orbital planes in case **b**.
Tab. 5.6 gather the results of conducted analysis.

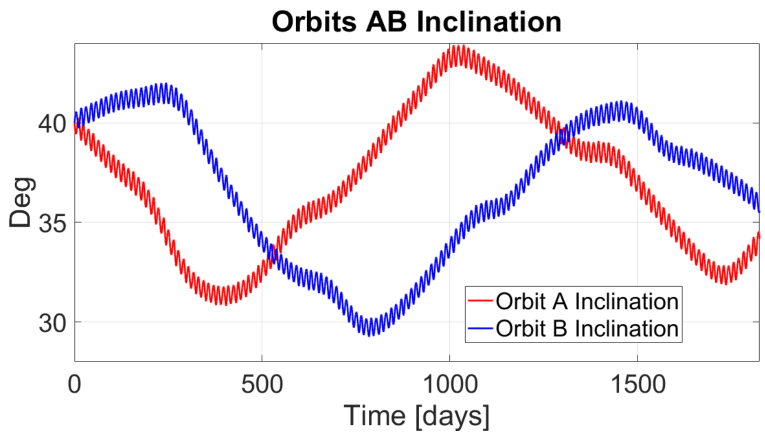
Frozen configuration 3		
Orbits	A	A + B
Satellites	4	4
Mare Tranquillitatis coverage	90 %	93 %
Mare Tranquillitatis gap	10 %	7 %
South Pole coverage	92 %	98%
South Pole gap	8 %	2%

Table 5.6: Coverage results for Frozen configuration 3 constellations

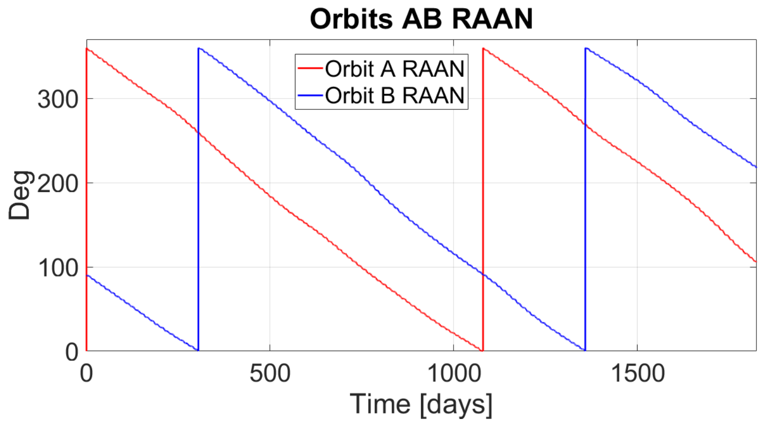
The results of both orbits propagation is represented in the graphs of Fig. 5.11. It can be seen a general good evolution of parameters. For the Argument of Periapsis the same reasoning can be applied as in the previous case, considering the orbits almost circular shape. In the final part of the mission, the orbits exhibit an increase in altitude oscillations, but it has been decided to keep them nonetheless to carry out analyses of the station-keeping maneuvers. Like the Frozen Configuration 2, these orbits manifest a reduction in inclination that may need some control.



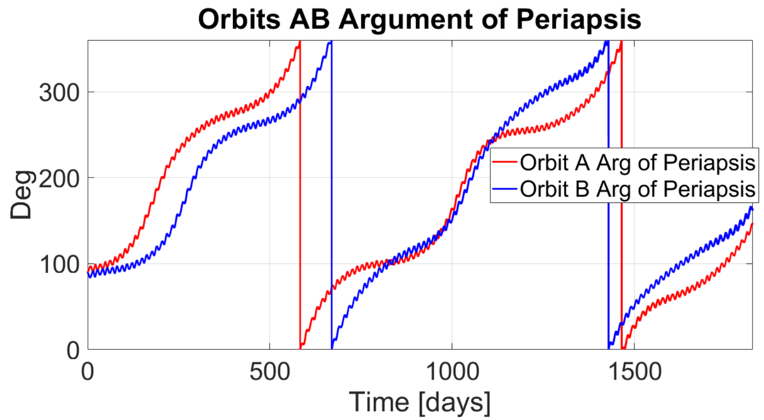
(a) *Orbits AB: Apoapsis and Periapsis altitude trend*



(b) *Obits AB: Inclination trend*



(c) *Orbits AB: RAAN trend*



(d) *Obits AB: Argument of Periapsis trend*

Figure 5.11: Orbital parameters evolution of constellation AB from frozen configuration 3

5.1.5 Hybrid configuration

The last constellation simulated is in in Fig. 5.12. It is composed of the frozen orbit defined by [66] for polar coverage, and a circular orbit to provide equatorial communication. The Tab. 5.7 comprises frozen and circular orbits parameters.

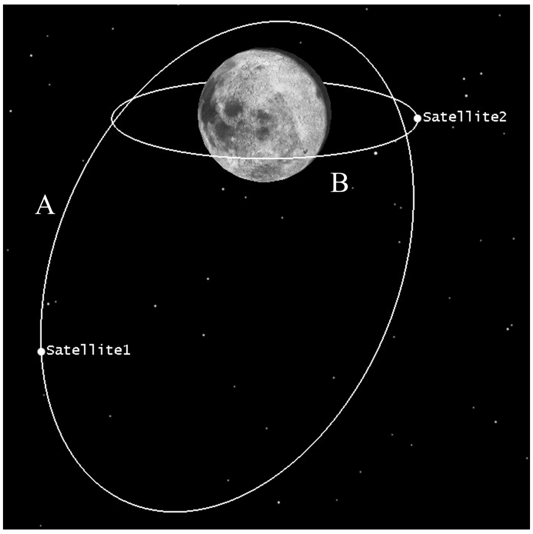


Figure 5.12: Hybrid configuration

Hybrid configuration								
S/C	Plane	Orbit	Period [hr]	a [km]	e	i [deg]	RAAN [deg]	AgP [deg]
1	1	A	15.3	7210	0.6	56.2	0	90
2	2	B	6.2	3937	0	0	0	0

Table 5.7: Hybrid configuration orbital elements

The choice to analyze such a specific configuration was suggested by the analyses found in [51], which proposed, among various constellations, the use of two circular orbits, one polar and one equatorial, for communication within their respective areas as can be seen in Fig. 5.13

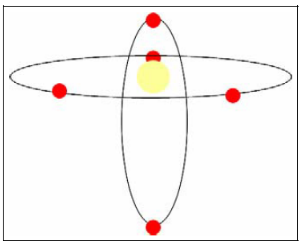


Figure 5.13: Space communication architecture working group constellation proposal. Image Credits: [51]

The concept has been applied to this mission by revising the used trajectories. Specifically, the polar circular orbit was replaced by an elliptical Frozen orbit, maximizing coverage over the pole of interest while avoiding spacecraft spending too much time in the northern hemisphere where coverage is not required. The equatorial circular orbit was retained but carefully redesigned in height considering perturbation effects and achieving desired coverage with the minimum number of satellites. This lead to the final configuration of Fig. 5.14 that with 5 satellites satisfy the requirements for coverage.

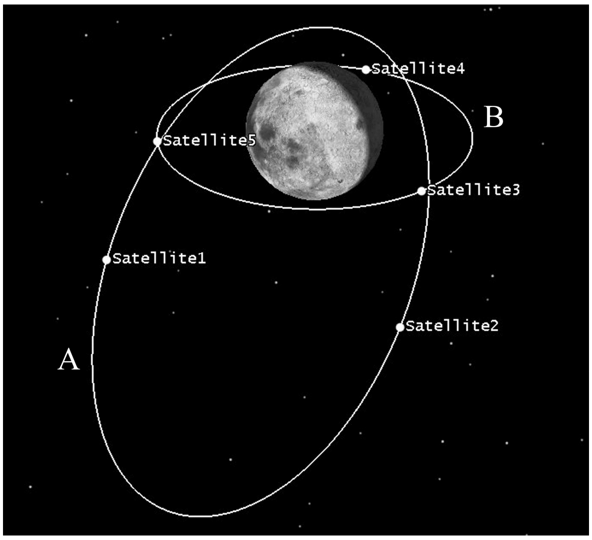
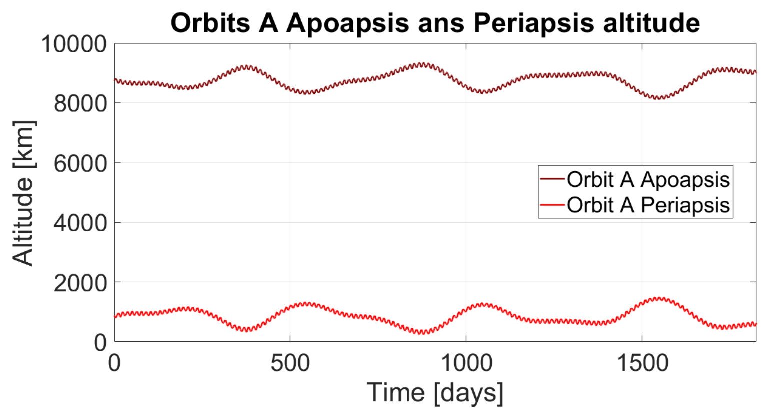
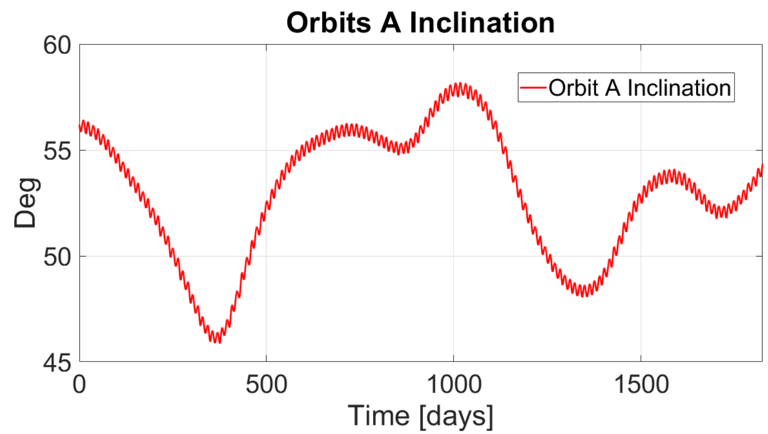


Figure 5.14: Hybrid configuration: AB orbits

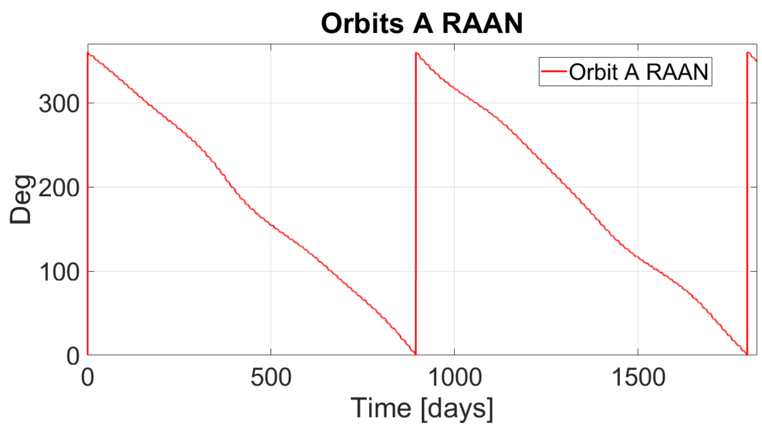
As one can see in the Graphs of Fig. 5.15, 5.16, both orbits shows a good evolution over time, and despite some variations in inclination, they remain generally stable. This behaviour suggest that the configuration could accomplish the goals even without any orbital correction.



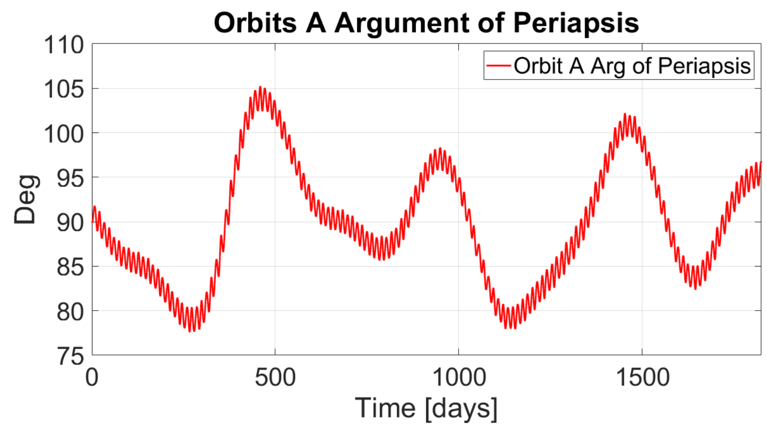
(a) *Orbits A: Apoapsis and Periapsis altitude trend*



(b) *Obits A: Inclination trend*

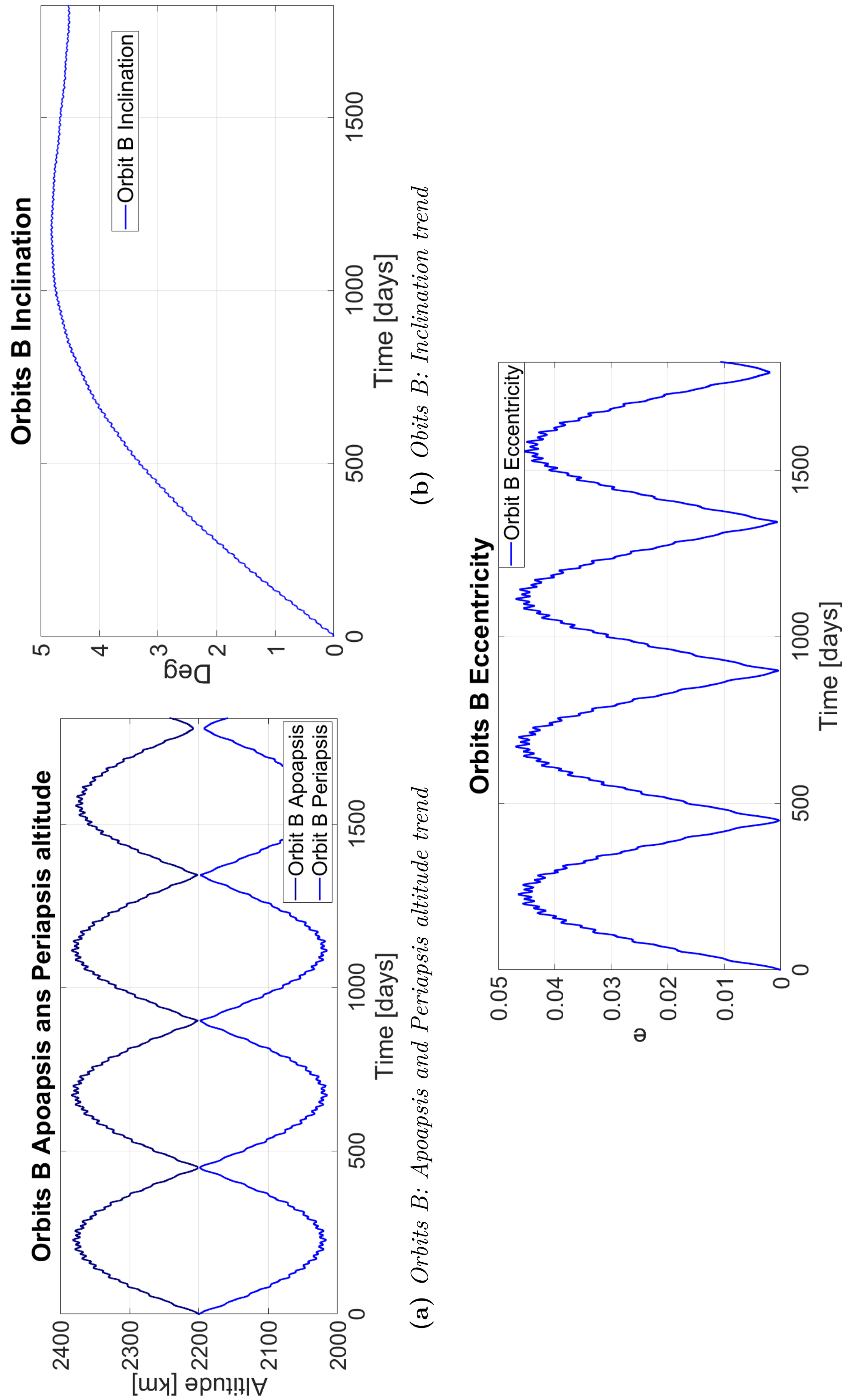


(c) *Orbits A: RAAN trend*



(d) *Obits A: Argument of Periapsis trend*

Figure 5.15: Orbital parameters evolution of orbit A from hybrid configuration



(a) *Orbits B: Apoapsis and Periapsis altitude trend*

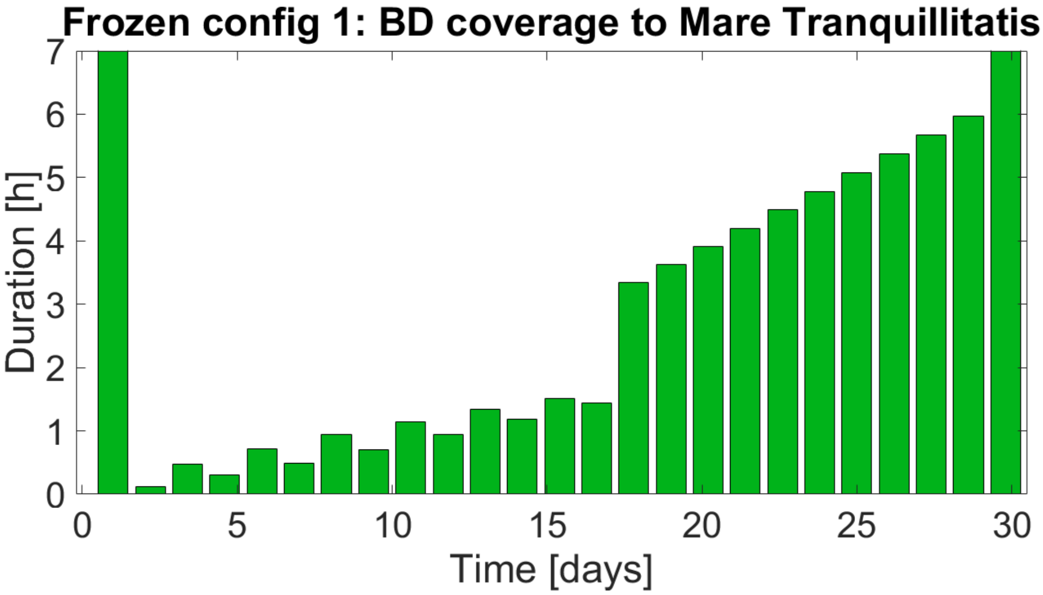
(b) *Obits B: Inclination trend*

(c) *Orbits B: Eccentricity trend*

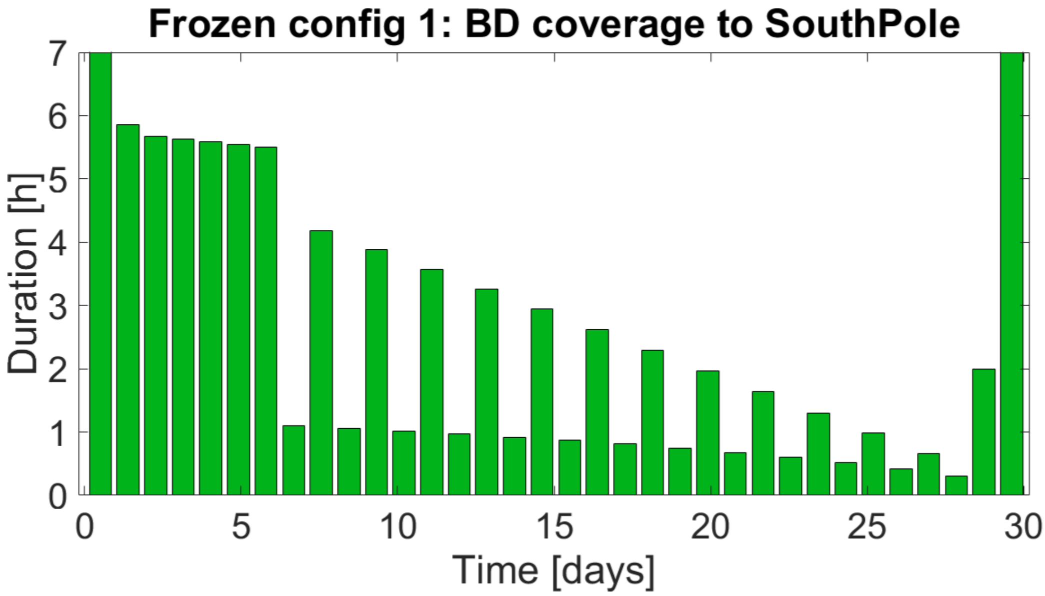
Figure 5.16: Orbital parameters evolution of orbit B from hybrid configuration

5.1.6 Detailed coverage of lunar outposts

To proceed with the evaluation of the selected constellations, as anticipated in Chapter 4, a more detailed analysis regarding coverage has been conducted. Subsequently, the results and considerations drawn from these simulations are reported. In particular, focusing on a complete lunar rotation, influence of gaps over coverage can be assessed. From the coverage results, it emerged that the analyzed orbits exhibit an interruption time accounting for less than 10% of the total duration of the analyzed scenario, equating to a maximum of 3 days within the 30-day period. In order to more thoroughly evaluate the characteristics of the coverage pertaining to the sites of interest, it was crucial to understand the distribution and average duration of these interruptions. This aspect is critical because, with a maximum interruption limit of 72 hours, solutions with a higher number of shorter interruptions are preferred over configurations with fewer but longer inactive periods. In the following Fig. 5.17 5.18 5.19 5.20 there are the representation of those periods of gaps. To enhance clarity, the y-axis has been constrained, and the longer intervals of duration have been labeled.
In all configurations, the gaps have a very low frequency, meaning they occur after many hours of communication and there are no days where gaps prevails over coverage. The only exception is found in the hybrid configuration, where gaps occur approximately every less than 2 hours of coverage for a duration of less than 10 minutes. To enhance comprehension of the data portrayed in the preceding graphs, the results have been compiled in the subsequent table. This table summarizes the total duration of coverage and gaps observed over the span of 30 days. Additionally, to underscore the quality of coverage, the number of gaps and their average duration are presented in Tab. 5.8 with other parameters, offering insights into whether these gaps can be disregarded or warrant consideration during specific periods. Furthermore, the average frequency of occurrence of gaps has been assessed; thus, it is possible to quantify the impact of periods without coverage, even if of short duration. Cases where the gap frequency is low and their duration is high will be considered unfavorable, indicating a concentration of these gaps within a restricted time frame.

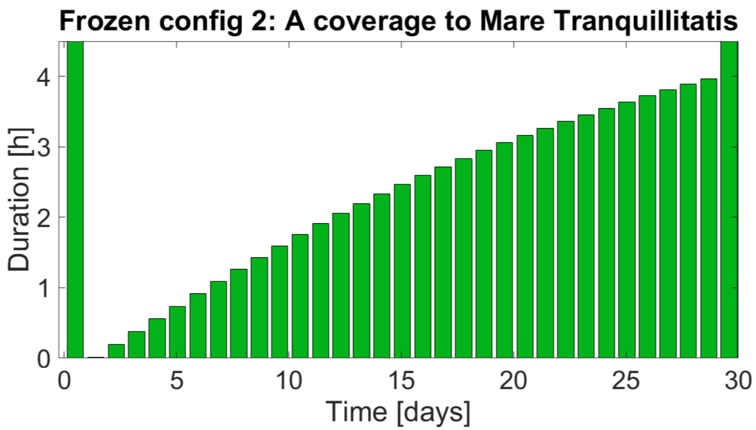


(a) *Config. 1 constellation BD coverage over 30 days to MT*

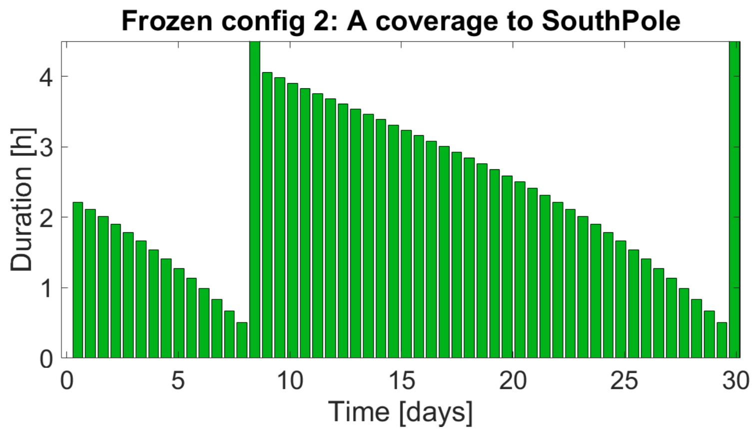


(b) *Config. 1 constellation BD coverage over 30 days to SP*

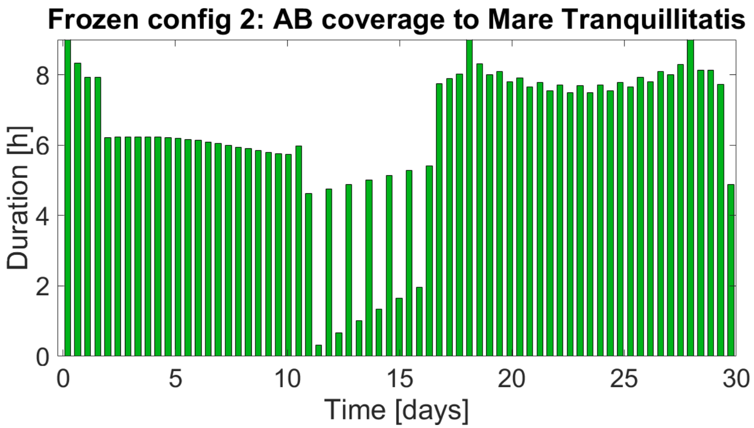
Figure 5.17: Coverage trend of frozen configuration 1 orbits over 30 days



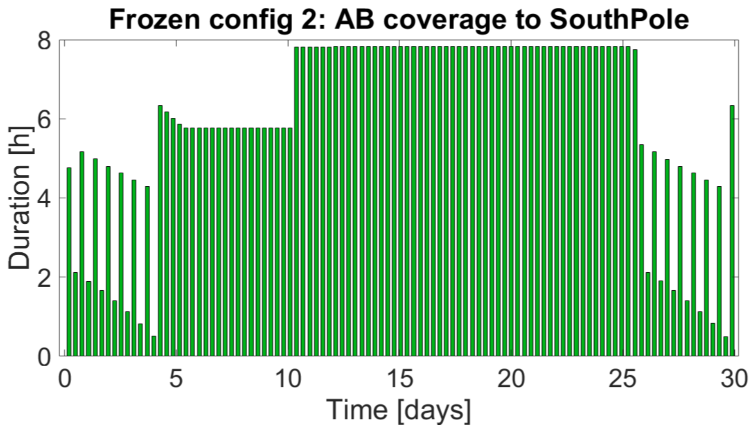
(a) *Config. 2 constellation A coverage over 30 days to MT*



(b) *Config. 2 constellation A coverage over 30 days to SP*

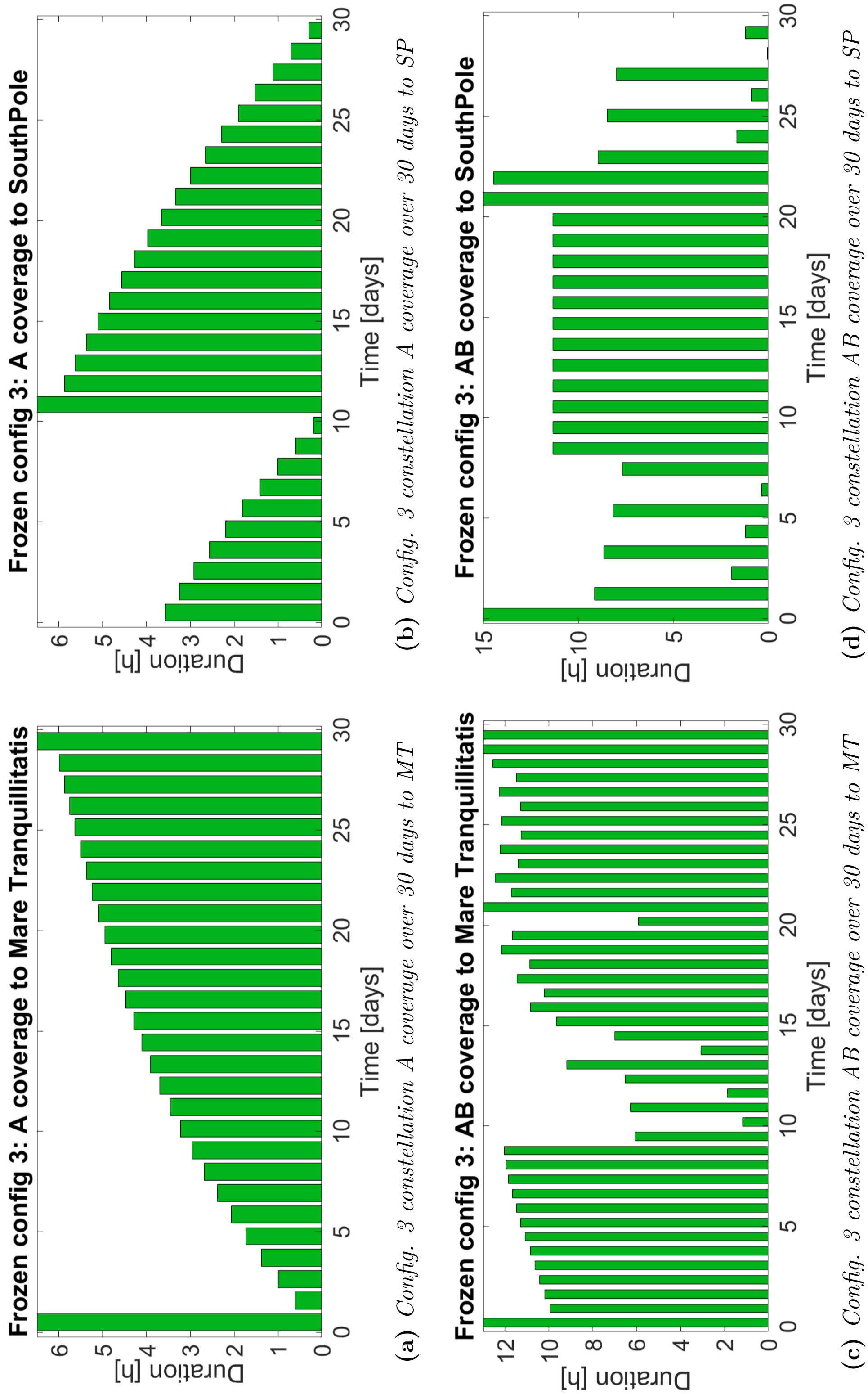


(c) *Config. 2 constellation AB coverage over 30 days to MT*



(d) *Config. 2 constellation AB coverage over 30 days to SP*

Figure 5.18: Coverage trend of frozen configuration 2 orbits over 30 days



(a) *Config. 3 constellation A coverage over 30 days to MT*

(b) *Config. 3 constellation A coverage over 30 days to SP*

(c) *Config. 3 constellation AB coverage over 30 days to MT*

(d) *Config. 3 constellation AB coverage over 30 days to SP*

Figure 5.19: Coverage trend of frozen configuration 3 orbits over 30 days

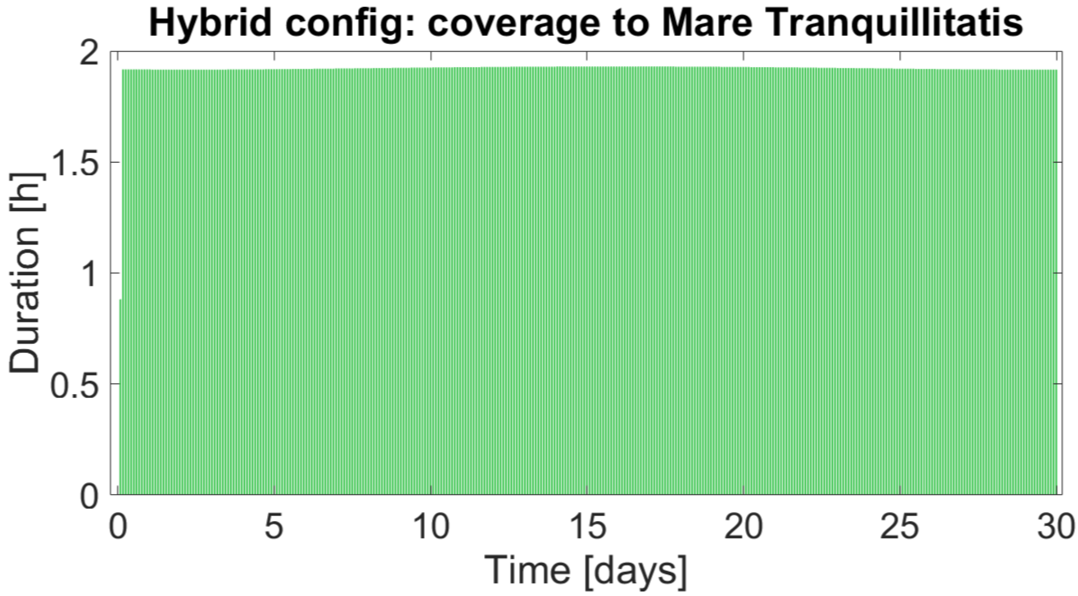


Figure 5.20: Coverage trend of hybrid configuration orbits over 30 days

Gap influence over 30 days coverage					
Constellation	**Total coverage [hr]**	**Total Gap [hr]**	**Mean gap duration [hr]**	**Gap number**	**Gap frequency [gap/day]**
Conf 1 Orbits BD to MT	676	44	1,8	24	0,8
Conf 1 Orbits BD to SP	675	45	1,3	33	1,1
Conf 2 Orbit A to MT	652	63	2	32	1,1
Conf 2 Orbit A to SP	624	95	2	52	1,7
Conf 2 Orbits AB to MT	647	73	1,1	66	2,2
Conf 2 Orbits AB to SP	630	90	0,9	102	3,4
Conf 3 Orbit A to MT	656	64	2,4	27	0,9
Conf 3 Orbit A to SP	623	90	3,2	28	0,93
Conf 3 Orbits AB to MT	663	57	1,4	41	1,4
Conf 3 Orbits AB to SP	686	34	1,2	28	0,93
Hybrid conf to MT	668	0,86	0,15	347	11,7
Hybrid conf to SP	720	0	0	0	0

Table 5.8: Constellations gap evaluation over 30 days

5.1.7 Eclipse time evaluation

The evaluation of eclipse time for satellites across the various orbits holds significant importance due to its direct impact on the power subsystem. Eclipse time refers to the duration during which a satellite remains in the shadow of a celestial body, such as the Earth or Moon in this case, diminishing or cutting off solar power availability to the satellite's solar panels. This parameter becomes pivotal as it directly influences the power requirements of the satellite. Specifically, the longer a satellite spends in eclipse during its orbital period, the greater the need for an adequate power storage system, typically fulfilled by larger capacity batteries. Consequently, the increase in battery size not only augments the power storage capability but also leads to an escalation in both the overall size and mass of the satellite. A general understanding and assessment of eclipse time across different orbits become crucial aspects in the design and operational phases of satellite missions. To assess the influence of eclipses, mean values over time and maximum experienced eclipse time have been evaluated, and consequently a ratio between the average eclipse duration and the satellite's orbital period. In the following, the results derived from the orbits under analysis are summarised.

5.1.8 Station keeping analysis

In this section of the study, the outcomes derived from the station-keeping analysis have been thoroughly explained. The methodology employed for conducting these simulations has been presented upon in Chapter 4. As previously outlined, two distinct propulsion systems were employed, and the optimal solutions for each orbit have been delineated in the subsequent discussion. The analyses conducted took into account the excursions experienced by uncontrolled orbit parameters, specifically focusing on their influence on mission performance, aiming to minimize propulsion costs.

Frozen configuration 1: Orbits BD

The results of implementing station keeping on the Frozen configuration 1 are reported in the Fig. 5.21

Results for chemical and electrical propulsion are summarized in Tab. **??**

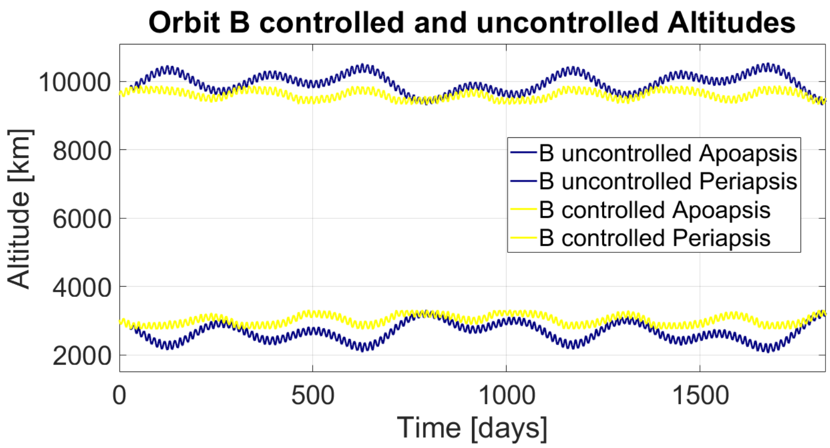
Frozen Configuration 1 Station-keeping		
Chemical Propulsion	**Orbit B**	**Orbit D**
N° maneuvers	357	325
DeltaV [m/s]	284.4	261.9
Mfuel [kg]	3.705	3.43
Time finite burn [min]	33.31	30.82
Electrical Propulsion	**Orbit B**	**Orbit D**
N° maneuvers	271	245
DeltaV [m/s]	219.74	193.84
Mfuel [kg]	0.555	0.5
Time finite burn [hr]	604	534

Table 5.9: Station keeping control for Frozen configuration 1 results

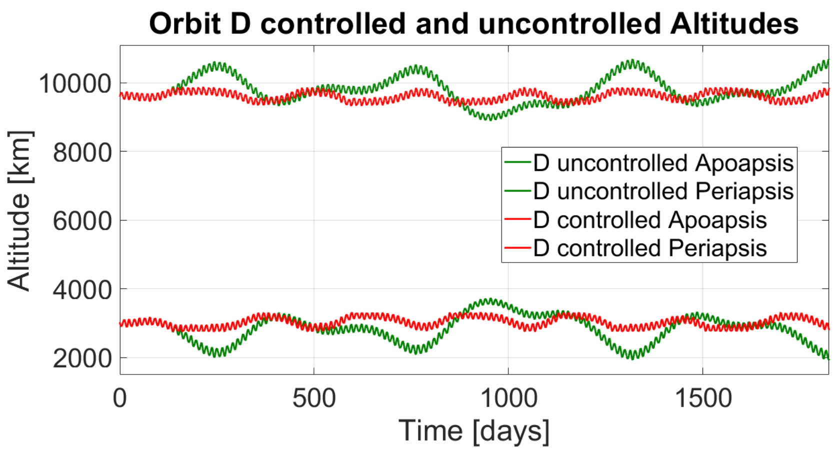
Frozen configuration 2: Orbits A and B

The results of implementing station keeping on the Frozen configuration A and B are reported in the Fig. 5.22

Results for chemical and electrical propulsion are summarized in Tab. **??**

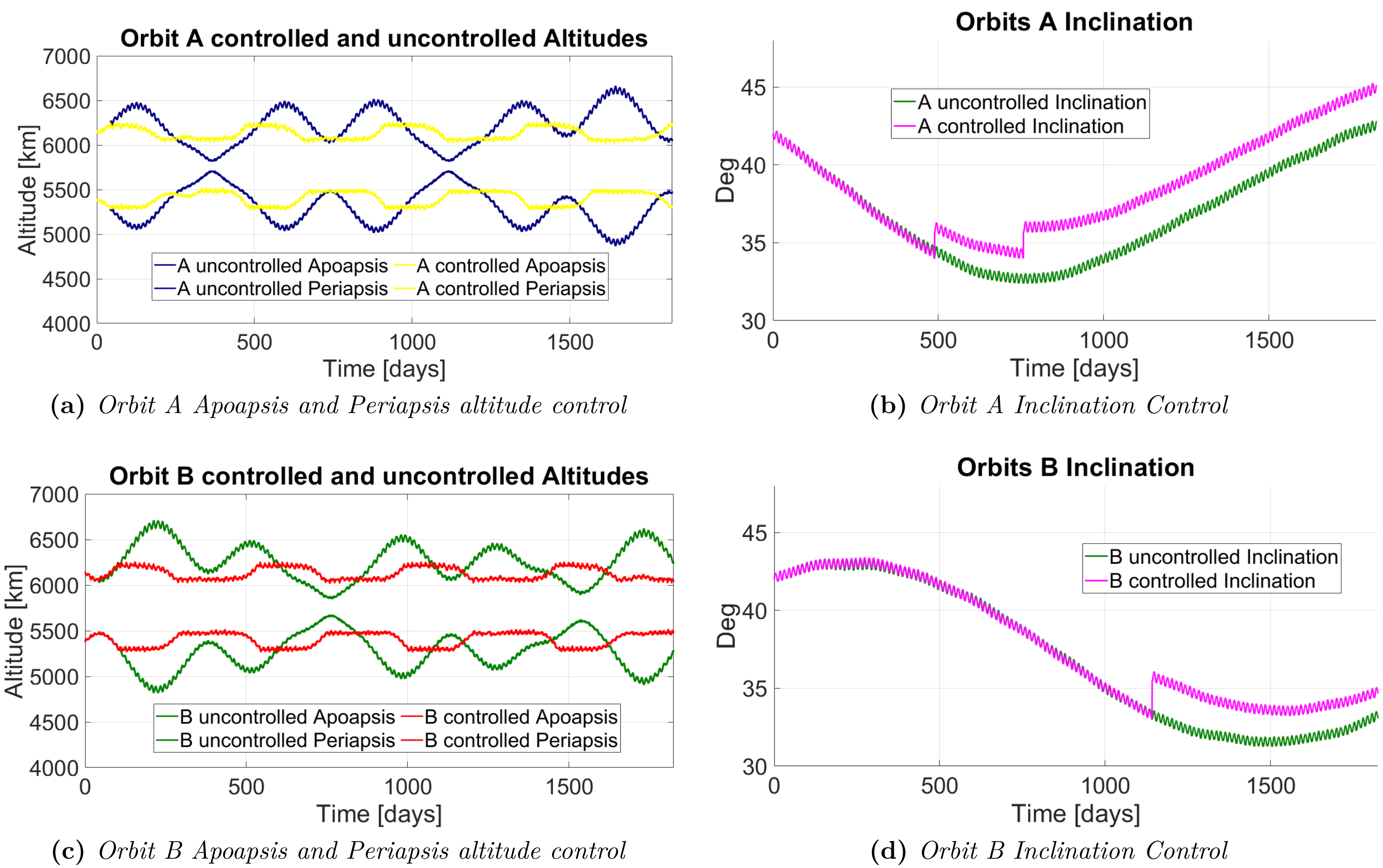


(a) *Orbit B Apoapsis and Periapsis altitude control*



(b) *Orbit D Apoapsis and Periapsis altitude control*

Figure 5.21: Altitude station keeping control for Frozen configuration 1 constellation



(a) *Orbit A Apoapsis and Periapsis altitude control*

(b) *Orbit A Inclination Control*

(c) *Orbit B Apoapsis and Periapsis altitude control*

(d) *Orbit B Inclination Control*

Figure 5.22: Altitude and Inclination station keeping control for Frozen configuration 2 constellation

Frozen Configuration 2 Station-keeping		
Chemical Propulsion	**Orbit A**	**Orbit B**
N° maneuvers	286	319
DeltaV [m/s]	197.6	214.4
Mfuel [kg]	2.62	2.84
Time finite burn [min]	23.6	25.5
Electrical Propulsion	**Orbit A**	**Orbit B**
N° maneuvers	290	373
DeltaV [m/s]	200.3	215.8
Mfuel [kg]	0.506	0.545
Time finite burn [hr]	551	594

Table 5.10: Station keeping control for Frozen configuration 2 results

Frozen configuration 3: Orbits A and B

The results of implementing station keeping on the Frozen configuration A and B are reported in the Fig. 5.23

Results for chemical and electrical propulsion are summarized in Tab. **??**

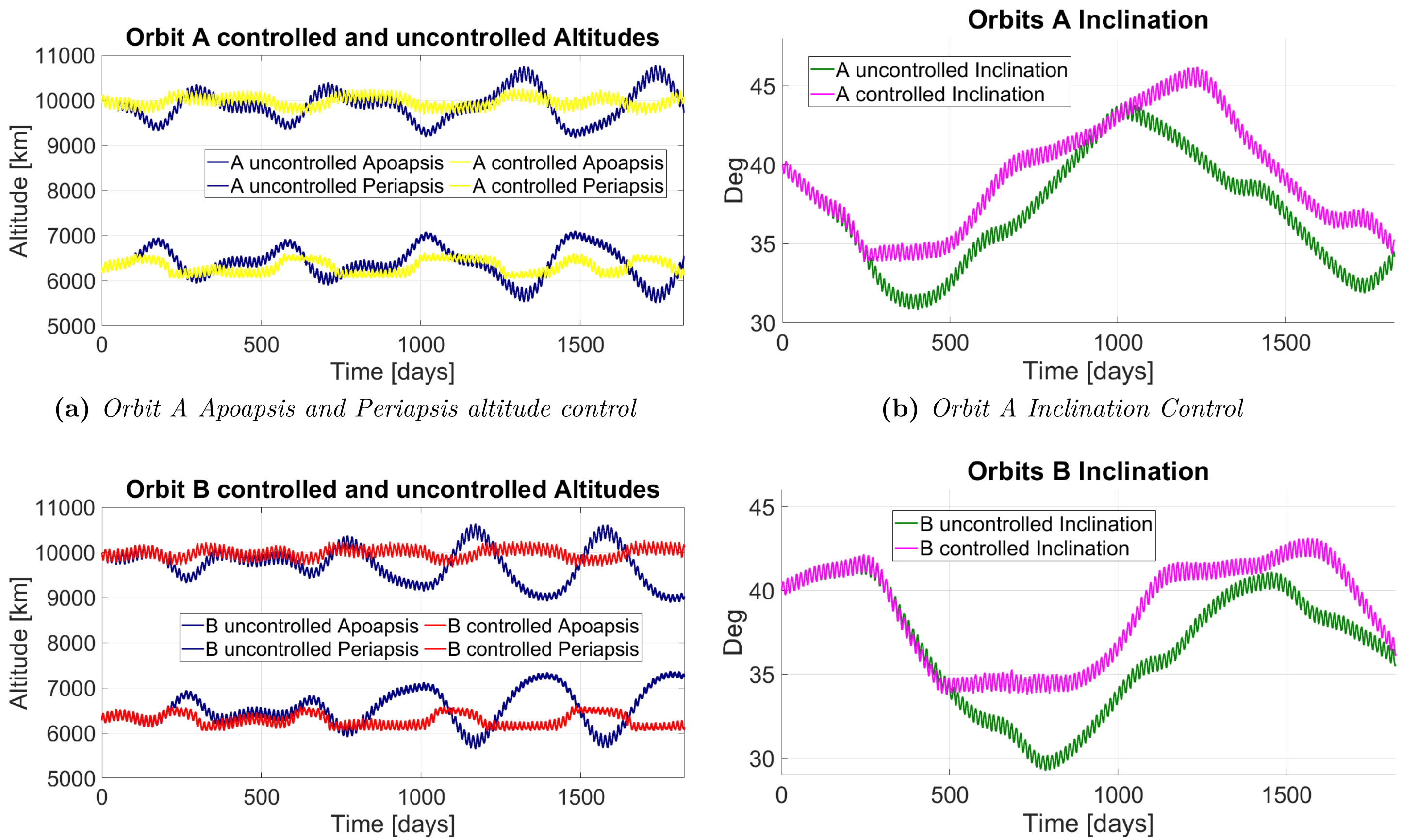
Frozen Configuration 3 Station-keeping		
Chemical Propulsion	**Orbit A**	**Orbit B**
N° maneuvers	616	503
DeltaV [m/s]	470	350
Mfuel [kg]	5.88	4.5
Time finite burn [min]	52.88	40.46
Electrical Propulsion	**Orbit A**	**Orbit B**
N° maneuvers	362	486
DeltaV [m/s]	618	502
Mfuel [kg]	0.600	0.92
Time finite burn [hr]	650	941

Table 5.11: Station keeping control for Frozen configuration 3 results

Hybrid configuration: circular and elliptical orbits

The results of implementing station keeping on the Frozen configuration A and B are reported in the Fig. 5.24

Results for chemical and electrical propulsion are summarized in Tab. 5.12



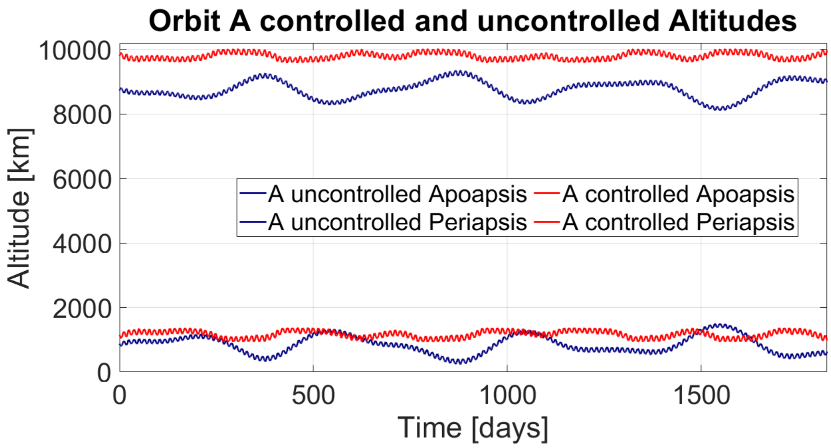
(a) *Orbit A Apoapsis and Periapsis altitude control*

(b) *Orbit A Inclination Control*

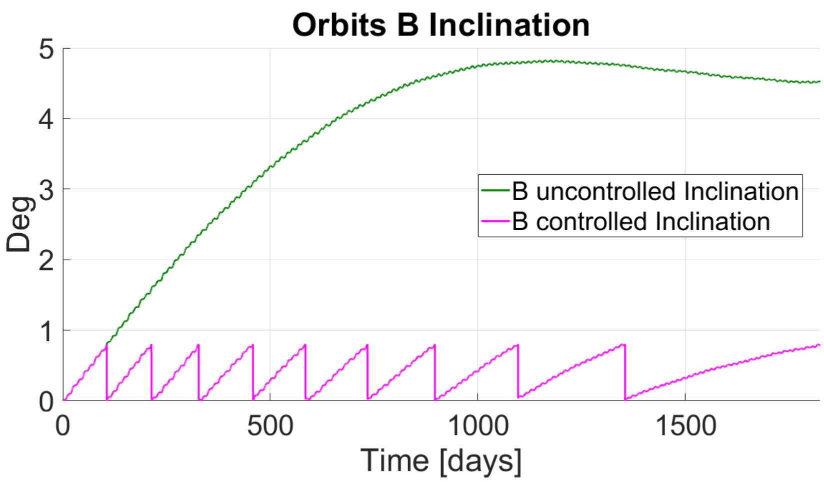
(c) *Orbit B Apoapsis and Periapsis altitude control*

(d) *Orbit B Inclination Control*

Figure 5.23: Altitude and Inclination station keeping control for Frozen configuration 3 constellation



(a) *Orbit A Apoapsis and Periapsis altitude control*



(b) *Orbit B Inclination Control*

Figure 5.24: Altitude and Inclination station keeping control for Hybrid configuration constellation

Frozen Hybrid Station-keeping		
Chemical Propulsion	**Orbit A**	**Orbit B**
N° maneuvers	245	9
DeltaV [m/s]	235.45	140.6
Mfuel [kg]	4.305	1.893
Time finite burn [min]	27.88	17.02
Electrical Propulsion	**Orbit A**	**Orbit B**
N° maneuvers	233	9
DeltaV [m/s]	223.06	140.4
Mfuel [kg]	0.563	0.356
Time finite burn [hr]	614	388

Table 5.12: Station keeping control for Hybrid configuration constellation results

5.2 Trade-off solutions

This paragraph presents the results of the Analytical Hierarchy Process and consequently, the trade-off regarding the constellation that provides the best compromise among the various identified figures of merit to best achieve mission objectives and thereby meet mission requirements. Moreover, this has allowed for a more detailed definition of the propulsion system to be adopted in the satellite, contributing to the final refinement of subsystem details.

5.2.1 Orbital configurations ranking

As exposed in Chapter 4, the Hierachy Process have been developed with FOMs definition. In the following Tab. are summarised the results on Figures of Merit with the corresponding weight factor.

As can be seen, access duration, ΔV, fuel for station-keeping and gap distribution are associated to larger values due to their importance over mission. In the following Tab. 5.14 the final results regarding the orbital configurations are reported.

As anticipated from the previously presented results regarding coverage, orbit stability, and consequently, station-keeping, the configuration using a single orbit, specifically the Frozen Orbit 2, emerges as the winning configuration. Particularly, this outcome highlights the effectiveness of two other configurations: Frozen 2 with orbits A and B, and the hybrid configuration. These configurations have also exhibited notably optimal characteristics for the mission's objectives in earlier analyses.

Upon reviewing the values of primary importance (Coverage, ΔV for station-keeping, and required fuel) the winning configuration proves superior, except for

Figures Of Merit	Weight Factor
Access duration	0.2295
Gap distribution	0.1044
Eclipse time	0.0651
Station-keeping ΔV	0.2198
Fuel for station-keeping	0.1718
Maneuver time	0.0863
N° of maneuver	0.0588
N° of S/C per orbit	0.0313
N° of orbits	0.033

Table 5.13: Figures of Merits and relative weight factor

Figures Of Merit	Frozen conf 1	Frozen conf 2 A	Frozen conf 2 AB	Frozen conf 3 A	Frozen conf 3 AB	Hybrid conf
Access duration	0.03855	0.03751	0.03710	0.03763	0.03875	0.03996
Gap distribution	0.02109	0.00718	0.00883	0.01217	0.01986	0.03531
Eclipse time	0.01139	0.01098	0.01097	0.00979	0.01004	0.01196
Station-keeping deltaV	0.03760	0.05944	0.05215	0.01720	0.01816	0.03526
Fuel for station-keeping	0.02934	0.04450	0.04013	0.01418	0.01489	0.02772
Maneuver time	0.01480	0.02295	0.02024	0.00715	0.00751	0.01368
N° of maneuver	0.00820	0.01479	0.01521	0.00454	0.00500	0.01104
N° of S/C per orbit	0.00382	0.000573	0.00573	0.00573	0.00573	0.00421
N° of orbits	0.00412	0.00824	0.00412	0.00824	0.00412	0.00412
TOTAL	**0.16889**	**0.21232**	**0.19447**	**0.11663**	**0.12406**	**0.18364**

Table 5.14: Weight and total scores of analytical hierarchy process over configurations

the first parameter crucial for the mission. To address this issue, options could involve increasing the number of satellites in configuration A to 5 or 6, or choosing to compromise on the overall satellite weight containment by adopting the hybrid configuration. Alternatively, one might reduce the frequency of maneuvers in the second constellation, leveraging its high stability to reduce the fuel consumption. However, this could potentially be done to the advantage of the winning configuration.

Further assessment can be conducted to determine the applicability of these three best configurations in terms of coverage for potential exploration sites, as previously hinted. A simple analysis of the coverage of these locations throughout the entire mission duration revealed results summarized in the Tab. 5.15

As deduced from the outcomes, excluding the hybrid configuration due to its extremely limited ability to communicate at the North Pole, among the remaining

	Frozen conf 1 A		Frozen conf 1 AB		Hybrid conf	
Location	Aitken Basin	North Pole	Aitken Basin	North Pole	Aitken Basin	North Pole
Coverage %	62%	83%	68%	70 %	85 %	10.5%

Table 5.15: Coverage evaluation of exploration sites from best constellations

two, the merit of the single-orbit configuration 2 becomes more evident. It manages to provide a good compromise in coverage for these areas.
In conclusion, considering the results from previous analyses, coupled with the evaluation using the FOMs and this additional verification, the constellation derived from Frozen Orbit 2 with only orbit A emerges as the best compromise in terms of meeting mission requirements while minimizing system complexity, given the reduced number of satellites (4) and orbits.

5.2.2 Satellite subsystems: design refinements and budgets

After determining the orbit to place the satellites, the final step involves revisiting the preliminary analyses conducted on the spacecraft's subsystem evaluation. This step aims to further detail the analysis, enriching it with the results obtained from station-keeping simulations. These simulations allowed the determination, along with the properties of the analyzed orbits, of the capability to perform control maneuvers using specific thrusters and quantities of fuel.
In particular, these outcomes made it possible to specify in more detail the mass of the propulsion system and the required fuel, which consequently contribute to the overall satellite mass. Upon selecting the thruster, using data sheets helped derive the necessary power, thus enhancing the power budget. Briefly revisiting the satellite data as preliminary defined in Chapter 3, in the table, the decision was made regarding the propulsion system.
In the evaluated optimal solution for the constellation to be used, the use of an electric propulsion system is feasible. This is due to the average maneuver duration of approximately 2 hours, which is not prohibitive considering the propulsion type, as well as the low frequency of approximately 6 maneuvers per month. Several reasons have led to the preference for the electric propulsion system over the chemical one. These include a lower total mass, encompassing both the propulsion system and the required propellant, that lead to a reduction in ΔV. Greater reliability stemming from the absence of pressurized tanks, easier integration of the propulsion system within the satellite due to its compliance and ease of installation provided by the manufacturer. Additionally, the examined electric propulsion system occupies less space, as the dimensions reported for the chemical propulsion system do not include the pressurized fuel tanks, which add weight and volume

to the satellite. Furthermore, the absence of pressurized tanks further enhances the reliability of the electric system. Another reason influencing the choice of the electric propulsion system is the increased precision in maneuvers, allowing its potential use for attitude control, although this aspect has not been evaluated in this context.

Concluding the preliminary definition of the satellite subsystems, the following Tab. **??** recalls the data defined by the various analysis to make a comparison with Hemeria's platform.

Satellite subsystem data from simulations	
Payload Tx power [W]	22
Propulsion system power [W]	50
Propulsion system mass [kg]	1.2
Fuel mass [kg]	0.4

Table 5.16: Satellites subsystems values derived from simulations

With a clear understanding of the propulsion system type, encompassing its mass and required power characteristics, and having assessed the necessary fuel for orbital maintenance maneuvers, the satellite subsystem design process can now be finalized. This step aims to establish a definitive architecture as a reference point, enabling us to draw the final conclusions. The results derived from the station keeping analysis, comprises propulsion system with related power consumption and fuel. As can be seen from Tab. 5.17 recall Hemerias data, there is the possibility to use this terrestrial Cubesat as platform to accomplish the mission objectives, using the architecture of communication chosen.

Hemeria HP-IoT satellite		
Platform	Regular	Boosted
Size [mm]	220x230x500	
Platform Mass [kg]	20	22
Max Payload Mass [kg]	15	
Max Payload Volume	8 U	
Battery Capacity [Wh]	172	
Payload avg. Power [W]	50-80	80-120
Payload Peak Power [W]	200	
Uplink Rate [kbps]	64 (S-band)	
Downlink Rate [kbps]	1000 (S-band)	1000 (S-band) 150-300 Mbps (X-band)
DeltaV [m/s]	>150	

Table 5.17: Hemeria satellite Data sheet. Credits: [99]

As can be seen, the Hemeria satellite data can meet the requirements determined through the conducted analyses. It can therefore be concluded that the preliminary design of the small satellite has been successfully completed, demonstrating the feasibility of using a small satellite platform for the intended purpose of establishing a satellite communication system in lunar orbit.

Chapter 6

Conclusions and future work

In this study, feasibility and preliminary design of a constellation comprising small satellites in lunar orbit were completed, aiming to facilitate communication to specific sites of interest on the lunar surface for upcoming missions focusing on lunar exploration and space exploitation. The objectives and requirements were achieved in terms of communication performance using a small satellite platform, supported by lunar terminals for direct to Earth communication, capable of providing an agile, reconfigurable, and easily replaceable system, in line with the trends of the "New Space".

Extensive research in the initial part of this study assessed the proposed system's feasibility, considering both existing terrestrial and lunar architectures, as well as identifying potential trajectories suitable for deploying the constellation, accounting for challenges and issues with lunar orbits.
Utilizing tools typical of the mission design process allowed for a thorough delineation of all aspects to be studied, requirements, and objectives, enabling the formulation of two mission architecture proposals. Being a preliminary study without existing literature references, multiple strategies were considered, especially in satellite design definition, ultimately suggesting the possibility of employing an already functioning terrestrial platform.
Critical evaluations were conducted using simulations via Agi STK software. Communication requirements with ground and lunar terminals were assessed, defining the necessary communication system and power requirements in both cases. This led to the identification of the Hemeria CubeSat as the reference for defining the point design and further exploration of the architecture, which includes satellite communication solely with lunar antennas. To validate the quality of this preliminary analysis, data derived from the larger-sized satellite were compared with those from a previously studied platform, Compass, showing comparable results in mass and overall power.

Further simulations were conducted to evaluate the quality of selected orbits, excluding those failing communication and stability requirements. This led to a clearer definition of potential solutions among the selected constellations. Analyses of gap occurrence frequency, eclipse impact on individual satellite orbital periods, and simulations of orbital control maneuvers were performed.
Each simulation yielded results utilized to assess the best constellation among the proposed ones using the Analytical Hierarchy Process method and suitable Figures of Merit.
The selected architecture comprises a single Keplerian Frozen orbit, exhibiting excellent stability with coverage requirements fulfilled using only 4 satellites strategically placed along the trajectory. Although the orbit showed capability for autonomous orbital control, station-keeping analysis demonstrated excellent performance using either chemical or electric propulsion, resulting in DeltaV = 125/140 m/s, depending on the chosen propulsion method (electric or chemical, respectively).
Thus, the selected orbit allowed for the detailed definition of the propulsion system, mass requirements, and fuel, completing the preliminary satellite design, resulting in an overall definition of the primary subsystems' masses and powers

Future studies stemming from this research may focus on:

- Developing a specific methodology for preliminary satellite design, potentially based on an existing satellite dataset, including subsystem characteristics concerning mass and power allocation, categorized by satellite class and type. This would enable the development of a parametric study method to provide rapid solutions given mission requirements and a starting point value, such as results from a link budget study in terms of communication powers.

- In-depth evaluation, through simulations extending beyond existing literature studies, of stability performances and potential maneuver methodologies for orbital control in the two most discussed orbit families in this field, namely Halo and Frozen orbits.

- Further exploration of the derived satellite design, delving into a detailed definition of the required subsystems, assessing thermal and power budgets, bus systems, and specific payload elements like antennas.

- Evaluating the possibility of reconfiguring one or more satellites from the initially imposed orbit to a secondary one to address coverage requirements in areas not adequately covered by the proposed configuration, for potential surface exploration purposes. In this case, the interest would lie in studying an algorithm capable of evaluating trajectory and maneuver strategy for positioning from any point in the initial orbit to the destination orbit.

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