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Engineering Challenges of a CubeSat Mission Around the Moon: First Steps on the Path to SelenITA

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ABSTRACT

Flying beyond Earth's sphere of influence has been part of the main goals in space exploration. Efforts of the Artemis program now encompass different classes of missions, including CubeSats. With the challenges of deep space as mission drivers, planning, designing, launching, and operating a CubeSat for a Moon mission is proving to be a step up in difficulty. In this context, SelenITA Mission is conceived as a science mission supporting the Artemis efforts, planned to operate at Low-Lunar Orbit (LLO), flying below 200 km gathering space weather and geophysics observations, marking the first Brazilian mission to the Moon. This paper outlines the engineering challenges encountered this far in the development of SelenITA. It presents the aspects of lunar orbits and the effects of Moon's potential field on a 12U CubeSat in LLO. A Reference Scenario is established, followed by an exploration of the extreme lunar environment's effect on the satellite's thermal, radiation, and power aspects. Communication limitations in the cislunar environment are analyzed, and strategies for the Attitude and Orbit Control Subsystem are discussed. The paper also addresses the challenges associated with delivery, uncertainties, and supply chain. A conceptual overview of the system is presented, concluding with the future steps.

INTRODUCTION

SelenITA Mission is conceived as a science mission that supports the Artemis efforts marking the first Brazilian mission to the Moon. It is a dual point mission that will provide the first multi-point dust, plasma, and magnetic field measurements in lunar orbit. This mission will advance the understanding of the electromagnetic space environment at the Moon in support of the Artemis program, exploration, and the geosciences, helping to understand how future astronauts, robots, and space hardware will live and work on the lunar surface.

Upon the signing of the Artemis Accords by Brazil, the mission started being discussed on potential science to be made, the contribution to the Artemis efforts, as well as on how to design a CubeSat capable of meeting such objectives. The mission is being developed at the Instituto Tecnológico de Aeronáutica (ITA) with the interaction of different partners in Brazil and abroad. As part of the evolutionary characteristic of programs developed at ITA Space Center (CEI), the SelenITA mission is expected to share part of the development with other missions such as ITASAT-2 [1]. Such overlaps can range from direct lessons learned from previous [2] and ongoing missions for a 12U CubeSat up to the use/consideration of similar subsystems or subsystems evolved for deep space use. An artistic representation of the SelenITA mission is presented in Figure 1.

As if the challenges of deep space were not enough, planning, designing, launching, and operating a CubeSat for a Moon mission is proving to be a step up in difficulty compared to past and current missions from CEI. The following sections present major challenging aspects that the SelenITA Engineering Team faced over the last 12 months working on the mission. Wherever possible, the potential solutions identified to overcome such challenges are discussed from a general perspective. As the paper covers the engineering challenges, aspects of the scientific rationale of SelenITA Mission are not discussed herein. The science discussion is an ongoing process and the information presented in the paper are part of the interaction between the Science and Engineering teams.

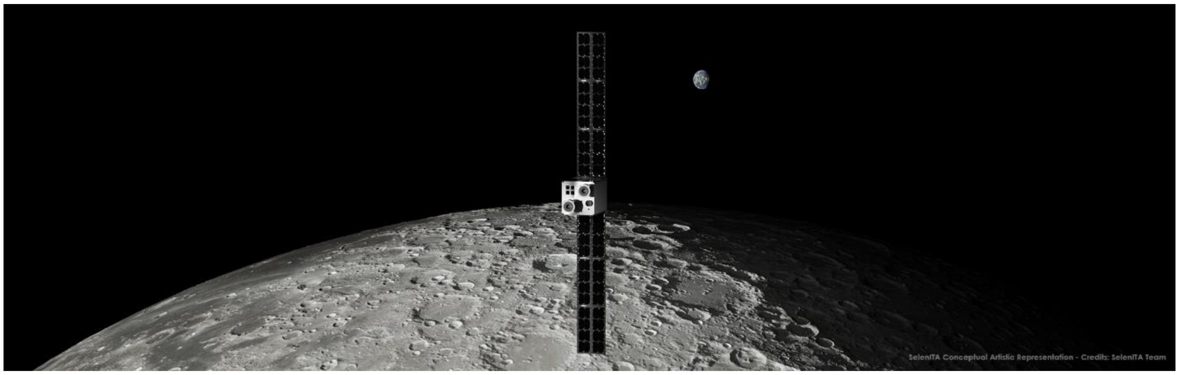

Figure 1: SelenITA – Conceptual Artistic Representation

LUNAR ORBITS AND REFERENCE SCENARIO

The effects of the irregularity of the lunar gravity field on a spacecraft orbiting the Moon are not something new. It has been discussed and explored by previous and ongoing missions such as Lunar Prospector (LP), Lunar Reconnaissance Orbiter (LRO), Korea Pathfinder (KPLO), and Kaguya (SELENE) [3 4 5 6], demonstrating the challenges of maintaining a regular orbit around the Moon while still providing a suitable condition for the science objectives of each mission.

What becomes evident from part of the literature, especially the ones covering real operational orbital data, is that missions until now are in totally different classes compared to a CubeSat. Current operational CubeSat missions to the Moon [7 8] are not specifically at a circular Low-Lunar Orbit (LLO), with either fly-by orbits or highly eccentric orbits due to their science requirements. As part of SelenITA's mission requirement of operations at a LLO, the first step was to understand the effects of the lunar irregular gravity field on a 12U CubeSat under different conditions.

The irregular lunar gravitational field is considered in the simulations using the Grail potential field (GL0660B) for the combination of 100X100 Zonals and Tesserals. Such a condition for Zonals and Tesserals was achieved after trade-off analysis on simulation time versus convergence of values. Even though other perturbations represented minor interferences in the results, the simulations also considered Solar Radiation Pressure (spherical model) and 3rd body (Sun, Earth, and Jupiter). All simulations were performed using FreeFlyer using Runge Kutta 8(9) numeric propagator and cross-checked against results from STK and GMAT equivalent propagators.

The main effect identified was the dead band associated with a given orbit around the Moon. Such a dead band represents how much the altitude of the spacecraft varies from the nominal altitude and is strongly influenced by the inclination presenting a cyclic pattern between the eccentricity and argument of periapsis that tends to increase over time until the periapsis of an orbit reaches the Moon. It is important to note that the orbits do not present a perceptible decay, as one would see on a LEO orbit, due to the lack of any dissipative forces such as atmospheric drag. Thus, the nominal altitude value for a given orbit stays the same as it is illustrated in Figure 2 for a 100 km orbit. In Figure 2 the orbit starts at the nominal altitude of 100 km with a dead band making the spacecraft vary the altitude 10 km above and below this nominal altitude. The effects of the lunar gravity field act more drastically on the eccentricity of the orbit and evolve in cycles until the dead band reaches the value of the altitude itself, thus making the spacecraft crash. In Figure 2 this is illustrated on the first days of July. It is important to point out that the orbit represented in Figure 2 does not include any correction (orbital maintenance) over time to prevent it from crashing.

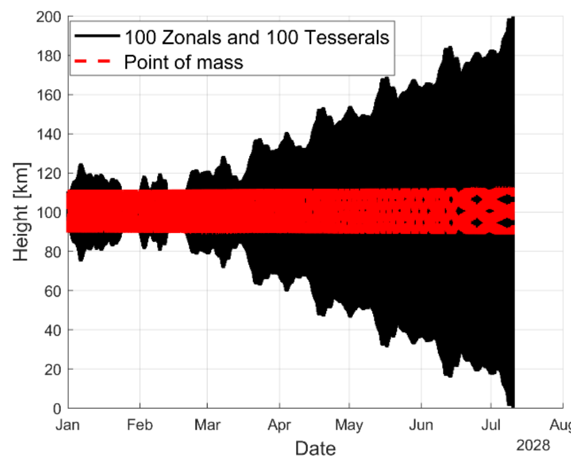

Figure 2: Altitude over time – Dead band effects

The red line in Figure 2 illustrates the same orbit only considering a point of mass for the Moon instead of the

GL0660B model. Also illustrated in Figure 2 is that for the Moon as a point of mass, the dead band does not evolve over time, as the eccentricity stays at the same value. Maneuver strategies are also discussed in other publications [3 4 5 6] and, in practical terms for a CubeSat, would need to be performed similarly, having however the limited propellant budget expected for a CubeSat. For SelenITA this is discussed later in this section.

Another characteristic of the lunar gravity field affecting the orbit is how parameters evolve in patterns, normally associated with the lunar cycle of around 27.4 days. This makes it difficult to establish a search for a preferred orbit modifying only a single parameter. An example of this inter-dependency between parameters is presented in Figure 3 with the evolution of the argument of periapsis and eccentricity over time, with the patterns being visually presented.

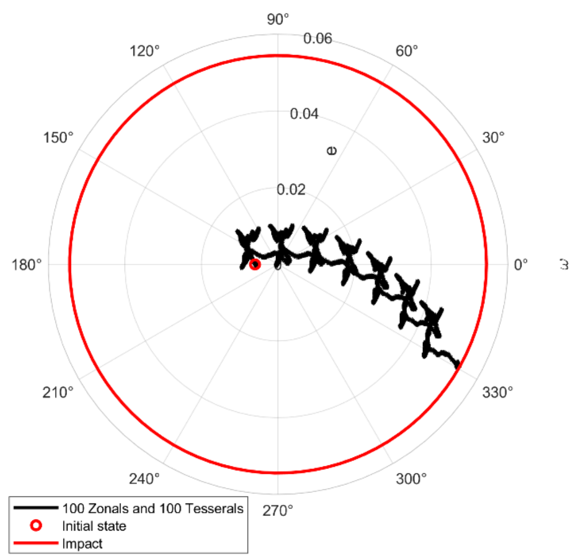

Figure 3: Evolution patterns of the argument of periapsis and eccentricity

The data from the plot in Figure 3 comes from the same orbit illustrated in Figure 2. The effect on eccentricity becomes clear as the pattern drifts outwards indicating an increase in the orbit eccentricity. Such analysis also helped to highlight the importance of the appropriate definition of the initial parameters for the orbit. The small red circle in Figure 3 indicates the starting point for the orbit at 180 deg of argument of periapsis, with the two parameters evolving with time until it crashes, being represented by the outermost red circle in Figure 3. This evolution of both parameters can be understood as a movement going from the left to the right and down in the polar chart of Figure 3. The selection of the initial parameters or eccentricity and argument of periapsis can define a "safe zone" as it will affect how long until the spacecraft will need to perform a maneuver to avoid the crash, if that is required, representing a risk mitigation factor in the mission.

The understanding of such relations and perturbations was possible with the simulation of different conditions which are still necessary as the project evolves. A variety of auxiliary simulations performed, and exploratory scenarios considered can be summarized by:

- 40 scenarios for pre-analysis and Orbit Candidates;
- 16 detailed scenarios for impulsive maneuvers;
- 57 detailed scenarios on Frozen and Quasi-frozen conditions;
- 15 different inclinations from equatorial to polar.

A Reference Scenario was then defined to explore a complete end-to-end scenario, from mission analysis and system budgets. Such a scenario also provided a way to unify all the previous analysis, standardizing for the entire engineering team. The main premises considered for the analyses supporting the Reference Scenario are presented as follows:

- Limit of a standard 12U CubeSat and related dependencies (interfaces, deployer …);
- Limit of currently available technology on suppliers i.e., no new development of subsystems;
- Limit of information currently available on services i.e., delivery and communication;
- Referenced on ongoing CubeSat Missions to the Moon and Deep-Space (solutions and lessons-learned);
- Orbital delivery provided by supplier (CLPS, lander, …) with the desired initial parameters;
- Communication based on most immediate option on the expected development of Deep-Space architectures.

Over 30 different conditions were verified to test orbit robustness, including checks on the capability to survive without propulsion/maneuvers at the deployment altitude for the entire mission time. The current approach for this scenario also encompasses a reduced number of required interventions/maneuvers and extended reaction time for operations and associated logistics (ground, antennas, system preparation). Most importantly, it explores the compatibility with the current capacities identified under the Brazilian readiness for a first mission to the Moon. The main characteristics of the Reference Scenario are presented as follow:

- Lifetime: 1 year;
- Insertion nominal orbit: 150 km;
- Operational nominal orbit: 70 km;
- Polar inclination;
- Commissioning phase: 3 months at deployment altitude;
- Operational phase: 9 months at operational altitude;
- Minimum time between maneuvers (station keeping): 45 days.

The trigger considered for the station keeping during the operational phase was the altitude reaching 15 km. Considering how the dead band varies, especially as the satellite reaches lower altitudes, this was used considering the highest mounts around the South Pole of the Moon and applying a safety margin.

Table 1 presents the main parameters used for the simulation of the Reference Scenario. It was observed that certain parameters can be considered differently between simulation suites (i.e., FreeFlyer, STK, GMAT) due to differences in the definition of the reference frame. This is pointed out in the chart specifically for the Right Ascension of the Ascending Node (RAAN).

Table 1: Reference Scenario – Parameters

Orbital Element	Value
Semimajor Axis	1888 km (Commissioning) 1808 km (Operational)
Eccentricity	0.006
Inclination	90 deg
RAAN	150 deg*
Argument of Periapsis	180 deg
True Anomaly	0 deg
Epoch	01/01/2028 00:00:00
Lifetime	1 year

* All data evaluated at the Moon Inertial (Mean Earth) Reference Frame inertial to J2000 (details on FreeFlyer documentation [9])

For the Reference Scenario, the initial eccentricity and argument of periapsis were chosen according to the pattern illustrated in Figure 3 to give more reaction time at the beginning of the mission, providing enough time for the commissioning of the platform and payload. The definition of the RAAN value depends on the launch and delivery conditions. The value used for the Reference Scenario considers the condition with the worst case for the dead band expansion. In practical terms, with such condition, in the case of the propulsion subsystem not being able to operate properly after the commissioning phase, the orbital conditions would allow the satellite to stay at the insertion orbit of 150 km without the need for orbital corrections. The dead band expansion in such case would still be within the conditions defined for the Reference Scenario for the minimum altitude of 15 km. Unfolding aspects of this potential failure scenario concerning the attitude control of the satellite have yet to be considered. The polar inclination is selected due to the science interests in the South Pole and to allow the potential coverage of all latitudes of the Moon. Such condition would also allow the measurement of a given point at the Moon equator roughly every 14 days. The inclination has a small variation throughout the period of the mission (from 89.5 deg to 90.9 deg) not requiring corrections maneuvers.

Figure 4 illustrates the satellite altitude over time and the total consumption of ΔV for each maneuver. Constraints defined for the Reference Scenario can be observed in Figure 4 with the minimum altitude (trigger) of 15 km, the time between maneuvers as well as the effects of the altitude dead band as the satellite is maneuvered to a lower orbit. It is also visible that, even though the satellite flies at the operational altitude of 70 km, it will still reach lower altitudes. This is of particular interest for trade-offs considering the science objectives.

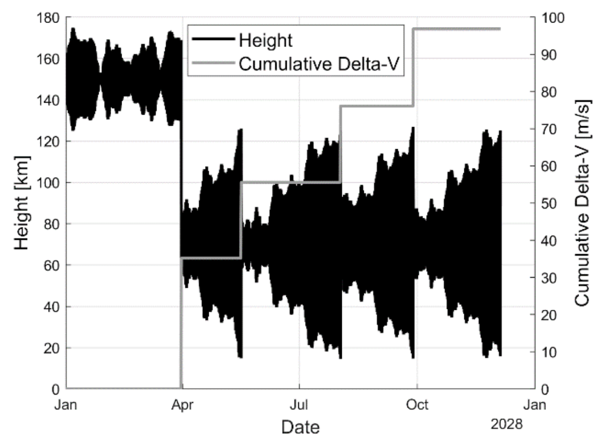

Figure 4: Reference Scenario – Altitude and ΔV versus Time

The strategy considered for the orbit control of the Reference Scenario consisted of a ΔV applied at apoapsis correcting periapsis followed immediately by a ΔV applied at periapsis correcting eccentricity (two tangent burn). Other strategies have been identified but have not yet been explored, being part of the future steps on the iteration of the Reference Scenario. A total of five orbital maneuvers are required (including a de-orbiting burn), with a mean time between maneuvers of 1.5 months, within the defined constraints for the Reference Scenario. The average burn-time of 150 seconds is identified for a thrust of 2N. Auxiliary simulations for thrust of 1N, 2N and 4N were performed to evaluate the total burn-time and the total propellant consumption with the 2N option being selected as a conservative option for the current stage of the analysis. The relationship between burn-time and total thrust was identified to be inversely proportional with further analysis still to be performed. The estimated total ΔV for the reference scenario including a 25% margin is around 130 m/s. Such ΔV is considered only for orbital maneuvers, i.e., altitude change and station keeping. Further margins are still to be considered to account for the attitude control of the satellite, to be discussed later in the paper.

Additionally, it must be noted that the maneuver epochs are still under analysis and will vary depending on several conditions such as the true launch date and conditions, and other details still to be analyzed. Simulation details can also affect the epochs for the maneuvers' triggers, and therefore it can induce minor differences in epochs for different simulations.

On a preliminary analysis, the potential candidates for propulsion subsystem were initially selected based only on ΔV capacity and power consumption. This led to electric propulsion being the immediate option. However, upon analysis and simulations it was identified that the total time to achieve the maneuvers required using low-thrust propulsion would take up to 70% of the mission time. Considering that this would conflict with the science objectives and platform demands of the power subsystem, the criteria of propulsion subsystem candidates had to be re-evaluated. New candidates for the propulsion subsystem were analyzed using a new set of criteria to provide a preliminary rank for a list of 134 propulsion options (CubeSat oriented) from 23 different companies. Table 2 presents the overview of such criteria for candidate selection.

Table 2: Propulsion Subsystem Candidates Criteria

Weight	Criteria	Score		
		1	3	5
5	Subsystem Modularity	No	-	Yes
5	Company Heritage	Based on previous successful missions		
5	Delta-V (25kg wet mass)	<170 m/s	-	>170 m/s
5	ADCS capacity	No	Yes (not 6 DoF)	Yes (6 DoF)
4	Active power	>50 W	-	<50 W
3	Subsystem Heritage	No	-	Yes
3	Dimensions	>4 U	-	<4 U
2	Thrust level (>1000 mN)	-	400 mN to 1000 mN	>=1000 mN
1	Wet mass (<5000 g)	>5000 g	-	<5000 g
1	# thrusters for Delta-V	1	2	>= 3

The top 5 selection of propulsion subsystem candidates based on the criteria of Table 2 is presented as follows:

- Aerojet Rocketdyne: MPS-135-4U [10]
- Aerojet Rocketdyne: MPS-125-4U [10]
- Dawn Aerospace: CubeDrive 4U [11]
- VACCO: Green MiPS [13]
- MOOG: Monoprop. Module [12]

The orbital analysis including the orbital maneuvers simulated for the Reference Scenario initially explored the use of the MPS-135-4U, with a backup option of CubeDrive 4U. The propellant consumption based on propulsion subsystem candidates ranges from 55% to 100% of the total capacity depending on the option selected. It must be pointed that the propellant consumption depends on characteristics considered for the propulsion subsystem (ISP, propellant mass, pressure, thrust, etc.) and on the margins considered. The analysis and selection of potential candidates will continue over the development of the mission also assisting the iteration of the Reference Scenario.

With a major part of the Reference Scenario explored from the perspective of orbital mechanics and the challenges of staying at Moon’s orbit, the baseline orbit could then be passed to other subsystems specific analysis, as it will be presented in the next sections. In addition to points mentioned over this section of the paper, further refinements of the Reference Scenario include:

- Adjustment of orbital parameters to explore a single given region on the Moon;
- Accommodate operational limitations of propulsion subsystems;
- Accommodate operational limitations of communication services;
- Risk analysis on delays or lack of correction maneuver;
- Analysis of end-of-life (EOL) procedures to de-orbit the satellite in accordance with the guidelines of the Artemis Accords on protection of heritage sites on the Moon [14].

Additional points related to the iteration of the Reference Scenario covering not only orbital analysis but the full analysis of the mission’s subsystems are presented at the final section of the paper.

EXTREME ENVIRONMENT

In addition to the irregular gravitational field of the Moon that presents extreme conditions for orbital control, operating a CubeSat in a cis-lunar environment is also accompanied by extreme conditions on thermal, radiation and power aspects.

Radiation

Satellites at the Moon encounter a unique radiation environment compared to satellites at LEO. Understanding this environment is crucial for the design and operation of lunar missions. Moving away from the shielding offered by the Earth's magnetic field means that the satellites are intensely exposed to a variety of radiation that will cause damage to the electronic components onboard the satellites [15]. One of the most concerning forms of radiation is high-energy Galactic Cosmic Ray, composed mainly of charged particles such as protons and nuclei of atoms, which can penetrate spacecraft structures and pose a radiation hazard to onboard electronics [16].

In addition to cosmic radiation, there is also solar radiation, which is emitted by the Sun in the form of charged particles and electromagnetic radiation. Such radiation can increase significantly during solar storms or with solar flares. Even though both Moon and LEO environments can experience Solar Particle Events (SPE) caused by solar flares or coronal mass ejections and Galactic Cosmic Ray (GCR), the lack of a substantial atmosphere and global magnetic field on the Moon results in greater radiation exposure during SPEs compared to LEO. Additionally, being outside the Van Allen belts the radiation exposure from trapped particles is less significant on a spacecraft orbiting the Moon [17].

The preliminary analysis for the radiation tolerance levels required for the SelenITA mission used the SPENVIS [18] tool, considering a satellite on a deep space mission, without the protection of the Earth's magnetic field. The analysis was performed considering the information from the Reference Scenario, considering a Near Earth Interplanetary orbit. Once the satellite orbit type was defined, the radiation model was chosen.

Table 3 shows the input parameters used in SPENVIS.

Table 3: Input parameters used in SPENVIS

Spacecraft Trajectories	
Mission Duration	1 year
Orbit Type	Near Earth Interplanetary
Orbit Start	1 Jan 2028 00:00:00
Distance from the Sun [AU]	1
Solar Particle Mission Fluences	
Solar Particle Model	King
Number of Ordinary Events	Burrel Statistics
Number of Anomalously Large Events	5
Prediction Period	Automatic
Offset in Solar Cycle	Automatic
Magnetic Shield	No

King's solar proton model was built using data obtained exclusively during the active years of solar cycle 20 (1966-1972). During solar cycle 20 there were 25 recorded events, among these 25 events, a single event was responsible for about 70% of the total fluence above 10 MeV. This event was called the anomalously large event. To make a conservative analysis of the deposited dose, 5 anomalously large events were considered. Figure 5 is given the plot of the solar flux of the protons for one year of mission.

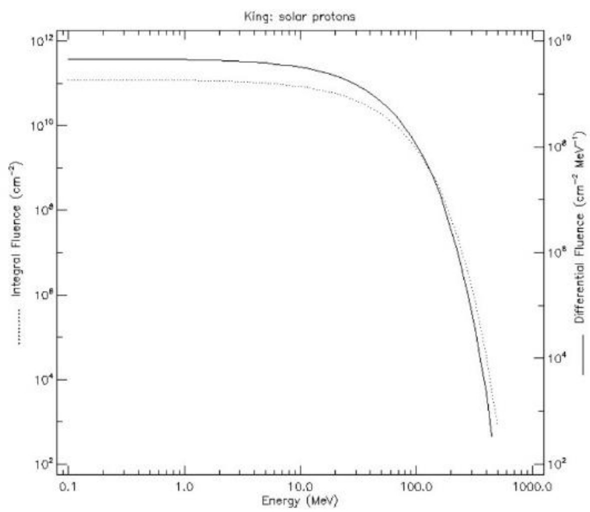

Figure 5: SelenITA Solar protons fluences

Once the model for the solar flux was defined, the Total Ionizing Dose was evaluated using the SHIELDOSE-2Q [19] code for a silicon target according to different aluminum thicknesses. Figure 6 presents the plot of the dose in krad (Si) to the thickness of aluminum.

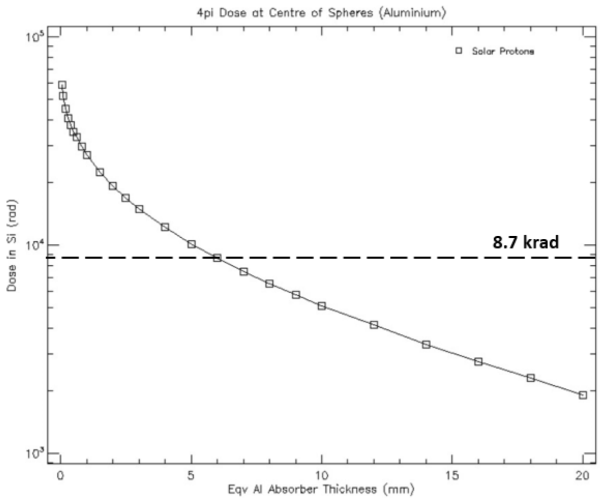

Figure 6: TID according with the aluminum thickness

In the current mechanical architecture of SelenITA, it is considered the use of 6 mm of aluminum as shielding, and the dose for this thickness of aluminum is around 8.7 krad. It is important to note that that this dose represents an overestimated analysis, as it considers 5 anomalously large events during one year of mission which has not been observed so far.

To understand which components are being used by deep space missions on the CubeSat class, an assessment was made on different types of deep space missions and the tolerances of the components that such missions are using. Figure 7 illustrates the dose supported by different subsystems of different missions based on publicly available datasheet of subsystems.

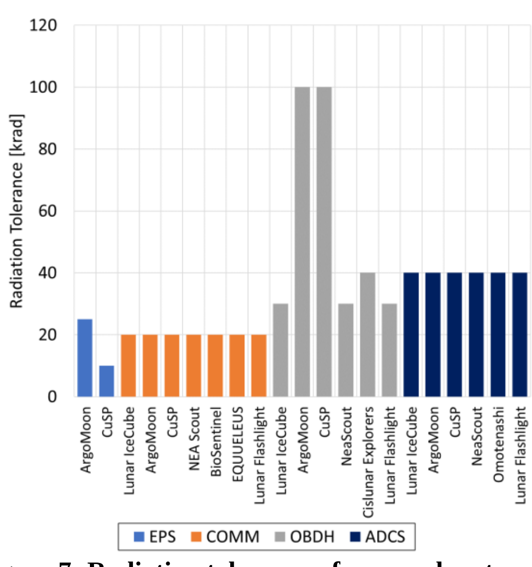

Figure 7: Radiation tolerance of some subsystem of deep space missions

Figure 7 shows that most deep space missions are using components that can be considered as COTS (Commercial of the shelf). This is in line with the results observed in the analysis which, even with overestimated conditions such as a large number of anomalously large events, the dose for the thickness considering in this study using is low. The use of 6 mm thick aluminum shielding adds a considerable mass to the satellite, but until the moment of the project this has not proved to be a limitation. If this happens during the progress of the project, new analyses should be done to resize the thickness of the shielding. It could even be considered other materials as shielding, or the specific shielding of some subsystems more sensitive to radiation.

The analysis presented in this paper considers the radiation effects for TID (Total Ionizing Dose). For SEE (Single Events Effects) other analyses should be done and other mitigation techniques should be used, such as redundancy, software identification and correction, or space weather prediction, in order to shut down the satellite during large solar events.

Power Budget

Power budget is indeed a significant challenge in CubeSat missions both in LEO and in deep space. The amount of power that can be generated is constrained by the size, which is limited in a CubeSat [20], and efficiency of the solar cells used. This limited power production capability can restrict overall mission operations and payload utilization [21]. Additionally, CubeSats need to store the energy generated to use when they are not exposed to direct sunlight. The energy storage system, often battery-based, must be carefully designed to provide sufficient power during these periods, considering the weight, volume, and longevity of the batteries. Meeting the subsystems power demands while considering the overall power budget can be a challenge. It is important to note that each mission often involves dynamic power requirements based on different mission phases, such as data transmission, payload activation, and attitude control maneuvers.

Managing these varying power demands can be complex, requiring effective power budgeting. Another challenge that power generation brings is that the power input generated by the solar panels is dissipated as heat inside of the satellite, which needs to be managed to ensure proper operation and prevent overheating. Efficient thermal design is essential to maintain acceptable operating temperatures while minimizing the need for additional power-consuming thermal control systems, such as heaters or cooling mechanisms [22].

This section presents a preliminary analysis for the power generation and consumption required for the SelenITA mission during the science mode of operation and evaluates the power limitations to the platform and payloads. The first step was to estimate the power consumption to the platform and payloads. Table 4 presents the value estimated based on a preliminary analysis of subsystems candidates.

Table 4: Power budget estimation to SelenITA in science mode

Segment	Power [W]
Available to Payload	10
Platform (+25% margin)	35
Total	45

Once the platform and payload consumption were defined, the dimensioning of batteries and solar panels was carried out. Table 5 shows what the battery capacity should be, given the input parameters such as eclipse time, battery efficiency, and depth of discharge.

Table 5: Battery dimensioning for SelenITA

Battery Sizing Estimation		
Power Required at Eclipse	45	W
Time of Eclipse	0.7805	Hours
η of Battery	1.045	-
Depth of Discharge	0.2	-
Mission Lifetime	1	Years
Battery Degradation	0.00735	Per Year
Capacity Required	35.1225	Wh
Capacity at EOL	168.0502	Wh
Capacity at BOL	169.2946	Wh

Table 6 presents the dimensioning for the solar panels, to supply energy to the platform and platform, and to charge the batteries, during periods of sunlight.

Table 6: Solar panels dimensioning for SelenITA

Solar Panels Sizing Estimation		
Power Required at Eclipse	45	W
Time of Eclipse	0.7805	Hour
Power Required at Daylight	45	W
Time of Daylight	1.183	Hour
η From Array to Loads (Xd)	0.8	-
η From Array to Batteries (Xe)	0.6	-
η Solar Cell	0.28	-
Solar Flux	1323	W/m^2
Worst Solar Incidence Array	15	Degrees
Degradation Factor	0.9548	-
Yearly Degradation Factor	0.0092	Year
Mission Lifetime	1	Year
Power Solar Array	105.7322	W
Ideal Solar Flux	370.44	W/m^2
BOL Power Flux	268.6986	W/m^2
Lifetime Degradation	0.9908	-
EOL Power Flux	266.2266	W/m^2
Area	0.3971	m^2

According to the current architecture of deployable solar panels, the profile of energy generation by these panels was simulated using STK. Figure 8 shows the energy generation graph for one year of mission according to the Reference Scenario.

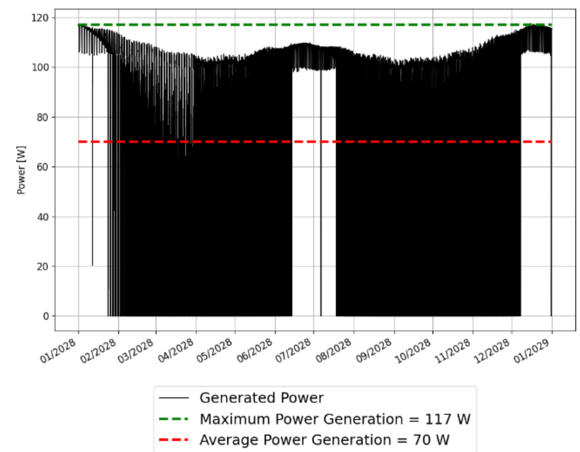

Figure 8: Power generation

According to Table 6, to fully charge the batteries the solar panels must generate above 105 W. Figure 8 illustrates that for some periods of the mission the current architecture of solar panels can generate energy above dimensioned value, but for other periods the energy generation is not enough to fully charge the batteries, making it necessary to change the operation mode in order to save energy. Therefore, for specific periods when there is not enough power, or, during orbital maintenance maneuvers, or when data transmission is being carried out to the ground stations, the satellite shall not be in science mode, due to the power restriction.

Another challenge that must be taken into consideration is that in the SelenITA mission there will be an eclipse with a longer duration in some periods of the year due to the obstruction of sunlight by the Earth. During this period, the satellite shall be in an operating mode that has the lowest possible consumption, and the thermal system will need to act to ensure that the internal temperature of the satellite is within the operating limits of the components.

Thermal

Another challenge of missions going to the Moon is on the aspect of thermal management. Such challenge becomes more demanding when dealing with a CubeSat where Size, Weight, and Power (SWAP) pose demanding requirements on the systems. Currently, the thermal aspect of SelenITA is at the initial steps, demanding more analysis to get to the same level of understanding as the other aspects of the mission. All the thermal analysis performed to this point focused solely on the understanding of the environment and main sources of heat at a low lunar orbit.

What became evident during such preliminary analysis is the effect of infra-red radiation on the satellite. Normally, the satellite would be subjected to conditions of hot and cold cases, associated with the Beta angles evolution over the mission timeline. For SelenITA, the values and dates for the Beta angle are presented in Figure 9.

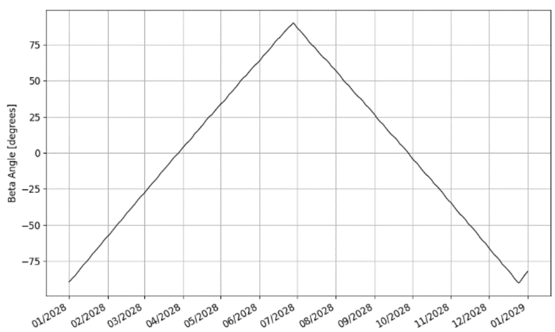

Figure 9: Beta Angle variation over mission lifetime

For the case of SelenITA's orbit, the maximum Beta angle (hot case) is marked by the moment the satellite orbits around the terminator line with the solar radiation heating the satellite continuously, having the sun-vector pointing perpendicular to the orbital plane. On the opposite condition, minimum Beta angle (cold case) for SelenITA's orbit is marked by the moment the orbital plane is in line with the Sun vector. This means that the satellite would absorb the least of solar radiation. However, the infra-red radiation emitted by the Moon poses an inverse condition of hot/cold case when compared to solar radiation. When the satellite is on the hot case for solar radiation (Beta max), the infra-red radiation on the satellite is the minimum. When the satellite is on the cold case for solar radiation (Beta min), the infra-red radiation on the satellite is at its maximum as the satellite flies over the sub-solar point on the surface of the Moon. Essentially, this subjects the satellite to a nearly always hot condition.

Preliminary analysis for SelenITA to understand the main sources of heat identified these "always hot" conditions. In such analysis, no specific CONOPs, operational modes or cycles of power dissipation were considered. A simplified simulation of a 12U satellite with a similar panel arrangement was analyzed for a satellite in plain aluminum for the body surface. The temperatures identified for an "operationally-off" condition were in line with what previous CubeSat lunar missions have pointed [24]. With the future analysis expected to consider different cases of internal power dissipation coming from platform and payload, this condition is expected to worsen, with the satellite reaching higher temperatures, also in line with [24].

As presented in the previous section, the power demands of the satellite are a challenge to be solved. Thermal analysis is fundamentally coupled with the understanding of power. As more energy is absorbed by the satellite, any dissipated power comes in the form of heat, meaning that simply adding more power capacity to the satellite would solve one problem and create another problem. As mentioned, the thermal analysis for the SelenITA mission is at its initial steps. An appropriate analysis plan is in development and will follow the steps described by [23]. Lessons learned as presented by [24] are also to be incorporated in the future analysis of SelenITA from the thermal perspective.

COMMUNICATION IN THE CISLUNAR ENVIRONMENT

The challenges associated with communicating with a lunar satellite are diverse and, up to the current stage of development of the SelenITA mission, they can be summarized into two main ones: uncertainties regarding access to deep space communication services from Earth ground stations, and the resulting constraint on the data budget.

The lunar communication architecture for the years 2018-2030 is outlined in "The Future Lunar Communications Architecture" report by the Interagency Operations Advisory Group (IAOG) [25]. According to this report, there is a consensus on frequencies, modulation, coding, and ranging schemes, as well as space data link and network layer protocols for missions planned until the end of this decade. One of the conceptual lunar communications architectures defined by the group involves establishing a communication network between Earth ground stations and lunar satellites via a relay satellite, such as NASA's Lunar Gateway or ESA's Lunar Pathfinder. However, due to the uncertainties surrounding the operational availability of this service, particularly for the SelenITA mission, the present analysis focuses on the Earth-Moon, Moon-Earth direct lunar communication architecture.

The purpose of this study is to comprehend the key parameters of lunar communication and explore communication alternatives that involve Brazilian ground stations. The objective is to avoid relying solely on the Deep Space Network (DSN) and other ground stations with deep space communication, because these may be overloaded in a few years [25].

The link budget analysis encompasses all gains and losses in the communication link, including noise. The data budget analysis considers the number of antennas available for the link per day, the available time for the downlink, as well as the modulation scheme and symbol rate. The lunar architecture considered in this study is direct communication (Earth-Moon, Moon-Earth).

The assumptions of the study are as follows:

- Data downlink analysis is performed for X-Band (8.4 GHz);
- A reference satellite radio with an output power of 2W is used;
- A satellite antenna with a gain of 12 dB is assumed;
- The modulation scheme used is BPSK;
- The symbol rate considered is 1.2 kSps.

Considering two 11-meter dish diameter antennas in Brazil, located in Brasilia and Sao Jose dos Campos, the average access time for both antennas was estimated to be approximately 68 minutes. However, only half of the total access time for the antennas was considered for downlink data since these antennas also serve other purposes besides communication with the SelenITA satellite.

After performing a link budget analysis, which indicates a margin greater than 3 dB with the assumed antennas, the initial results reveal that the total data budget in the Brazilian reference scenario is approximately 1 megabyte per day. Out of this, 400 kilobytes per day is allocated to the satellite platform, while 600 kilobytes per day is allocated to the payload. When compared to other lunar missions [26 27 28], it becomes evident that the data budget can be increased. However, achieving this requires considering larger dish sizes for antennas, increasing the number of antennas in the communication network with the SelenITA satellite, exploring alternative modulation techniques, and possibly incorporating a relay service into the communication architecture. It is important to note that the feasibility of this scenario is contingent upon uncertain external factors at present.

ATTITUDE AND ORBIT CONTROL SUBSYSTEM

The preliminary architecture for the SelenITA AOCS (Attitude and Orbit Control Subsystem) is defined based on current methods of attitude and orbit determination end control [29 30 31 32] and on lessons learned from SPORT mission [2]. Figure 10 shows the flow diagram of this architecture.

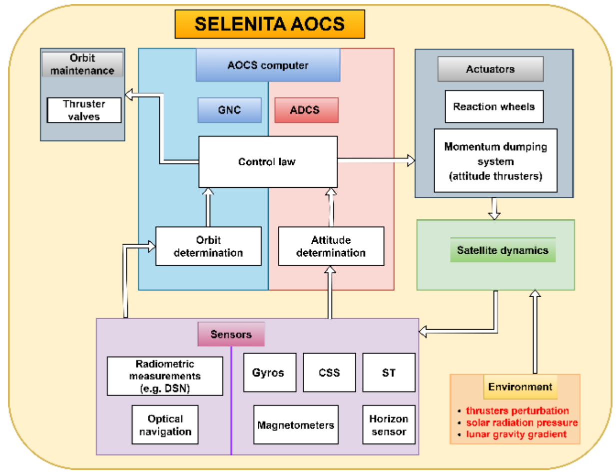

Figure 10: AOCS Preliminary Architecture

Assuming that attitude states can be determined using onboard measurements through star trackers or Sun/horizon sensors we can divide the SelenITA challenges for AOCS into two issues. The first concerns the choice of attitude actuators to perform stabilization, pointing, and momentum dumping. Usually for low-Earth orbits, the reaction wheels and magnetic torques perform these tasks. But for high Earth orbits or interplanetary missions the magnetic field presence is weak and unpredictable. Therefore, thrusters are required in place of magnetic coils for spacecraft de-tumbling and desaturation of wheels, causing increase in the design complexity of control subsystem and, therefore, the mission costs.

Another relevant challenge relies on orbit determination for navigation, and station-keeping maneuvers around the Moon. The orbital position and velocity determination approaches for LEO are based on the Global Positioning System (GPS) and Inertial Units measurements. The navigation problem around the Moon is more complex than that found in Earth orbits once the link range is greater, and the trajectories are significantly influenced by the lunar gravitational attraction. Thus, the orbit determination is directly coupled with uncertainties in the force model and in the astronautical constants. Therefore, specific means must be developed to determine the spacecraft trajectory in flight or the deviation between the actual and nominal trajectory. A number of feasible navigation techniques for lunar navigation can be found in literature, and the observations fundamental to those techniques may be made basically from the spacecraft itself, from tracking stations on the Earth, or from another spacecraft [33].

Lunar and deep space missions often adopted onboard and offboard navigation methods as the main paradigms to obtain state data. Deep space navigation solutions also can be classified according to their autonomy as: autonomous, non-autonomous, and semi-autonomous. These approaches are based chiefly on optical navigation, crosslink (inter-satellite) radiometric navigation, and ground-based radiometric methods [34]. Furthermore, the fusion of radiometric data with optical data can yield more robust and accurate trajectory determination [35].

The use of ground-based radiometric methods via DSN and optical navigation has been initially considered for the SelenITA mission. According to [36], there are three feasible radiometric high-level solutions provided by DSN. Two-way Doppler tracking gives sufficiently long tracking passes and provides a wealth of information about the orbital motion of the spacecraft [37]. However, it requires a dedicated support from a DSN antenna, limiting their availability.

The One-way Doppler tracking baseline measurements, obtained from the spacecraft to ground stations via the DSN's Multiple Spacecraft Per Aperture (MSPA), enable the DSN to receive signals from up to four spacecraft simultaneously [38, 39], providing track even when the primary target is another spacecraft. Additionally, the signal power required for tracking is significantly reduced compared to a two-way approach, allowing the spacecraft to utilize lower gain antennas. However, this tracking architecture requires precise timing from an accurate onboard clock to minimize environmental sensitivity of the frequency and timing reference [39].

Finally, the third architecture considered from DSN is the Delta Differential One-way Ranging (DDOR) that performs precise state tracking using the differential one-way range technique. This technique provides information about the angular location of a target spacecraft relative to a reference direction through the measurement of range difference between two ground station antennas separated by a large geographic distance [40]. The main challenge of this architecture is the requirement of two antennas, and DDOR passes can only occur in areas where there are overlaps in coverage [36].

SERVICES AND DELIVERY UNCERTAINTIES AND SUPPLY CHAIN CHALLENGES

Other challenges identified over the initial discussions and analysis for the SelenITA mission are the uncertainties for service assumptions and the process of acquiring such information. Even though this can be considered outside the engineering realm, it has direct effects on how the analysis of the mission unfolds.

With the ecosystem of missions going to the cis-lunar environment expected to grow over the next decade, specific services are also expected to be offered.

Services of communication (space as a service) for deep space can offer an alternative to the DSN [40] as this one will have to deal with multiple missions that can be considered of higher priority for its main administrator, NASA. Even though it is possible today to include DSN in the logistic and budget planning for SelenITA, the realistic approach is that other specific services will have to be selected. Understanding the technical aspects of the network of each provider is a key factor in understanding the limitations the mission might have. As presented previously in the paper, the current assumptions for link and data budget assumed a simple network of antennas in Brazil, which poses challenges on the amount of data that can be downlinked. Thus, the number and the size of antennas that can be offered in a service, the available bands, and the main concepts of use for such service are points that the next iteration of the Reference Scenario will include. The availability of such services for telemetry gathering and sending of telecommands impacts the logistical aspect of the mission. This concern overlaps with discussions on how the mission will have to arrange ground operations to send commands and instructions in advance as well as how certain operational aspects, i.e., maneuvers, might require the supervision of the satellite control team.

Another type of service commonly offered for CubeSats is the delivery service at a given orbit, either on rideshares or specific placements with more recent services of orbital logistics. When dealing with deep space, the most immediate option discussed is the Commercial Lunar Payload Service (CLPS) [41]. The providers of CLPS missions are dealing specifically with NASA requests to assist the Artemis Program and concurrently offering, wherever possible, the commercialization of space on their landers/rovers. Presently, specific options and categories of services are offered more vaguely by each provider demanding a more tailored discussion compared to how delivery services are offered for LEO. From an engineering perspective, once again the CubeSat paradigm is not clearly outlined as part of these delivery services even though they can be expected. What can be expected is that, with clear objectives and locations defined for future CLPS missions, the providers will then offer any of the extra available space in a case-by-case option. As consequence, the design of missions such as SelenITA will have to consider for a large flexibility on the CONOPs and on the resources to reach the planned orbit in case of minor divergences between the specific objectives of a CLPS delivery and the planned objectives for an orbital delivery such as SelenITA.

The current Reference Scenario did not include major flexibility explorations and assumed that a given delivery service would be able to meet the initial orbital parameters and schedule. Variations of the initial RAAN value have been explored and will continue to be analyzed. Still, the challenges on uncertainties of delivery services can affect the CONOPs, schedule and potentially the cost of a mission. As part of the risk mitigation for the mission, the engineering team established a plan to try to interact with such providers to then have the sensitivity analysis for the Reference Scenario based on potential candidates of delivery services for SelenITA.

Lastly, supply chain challenges are part of the mission planning that can directly affect the engineering analysis and definitions. Among the factors observed over the work on SelenITA as well as from lessons learned from SPORT [2] and ITASAT-2 [1] the following are presented:

- **Suppliers' heritage in CubeSat subsystems to deep space:** current options (normally COTS) are focused on LEO missions. Missions flown on SLS-1 provided the heritage to a substantial group of providers and subsystems. Current engineering analysis and selection of potential candidates prioritizes this point.
- **Access to such suppliers/providers:** Challenges on previous missions from Brazil stayed around on reaching specific suppliers. For CubeSats it is important, and a common practice, of offering subsystems outside ITAR restrictions and with accessible export control. Still, some technologies, for instance AOCS and propulsion, can still present a less straightforward process. Current strategies to deal with this challenge are on the line of trying to interact with such suppliers as early as possible, even though specifications or definitions are not yet complete.

CONCEPTUAL VIEW

With the analysis covered for the vital subsystems it was possible to explore the concept of the spacecraft. Here challenges of the CubeSat paradigm become evident especially in the space available for platform and payload.

The current conceptual view of the system presented in Figure 11 shows an exploratory view of potential candidates of subsystems for the SelenITA satellite to fulfill the Reference Scenario. The objective of presenting the conceptual architecture is to integrate the knowledge acquired over the previous analysis and to highlight the current limitations of the system, for both platform and payload. It is important to note that the conceptual view of the satellite does not represent a final definition of the system as this will evolve to respond to the challenges discussed herein and to encompass the discussion on the science objectives.

The main characteristics of the current system's conceptual state are listed as follows:

- 1 year of lifetime;
- 70 W average of power;
- 1 MB/day of data;
- Total Wet Mass < 30 kg;
- from 3 to 4 kg of mass available to payload;
- 2.2 U of space available to payload;
- ~ 170 m/s of ΔV capacity (maneuvers + AOCS + station keeping);
- The payload must be prepared for high temperatures.

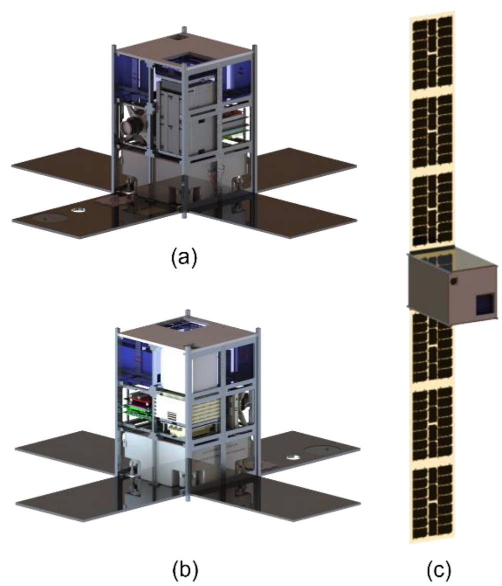

Figure 11: SelenITA – Platform current Conceptual View – (a) and (b) internal components, (c) unfolded panels

Still on the integration of the knowledge acquired, Figure 12 presents a potential operational timeline of the Reference Scenario with the different disciplines analyzed and discussed previously in the paper.

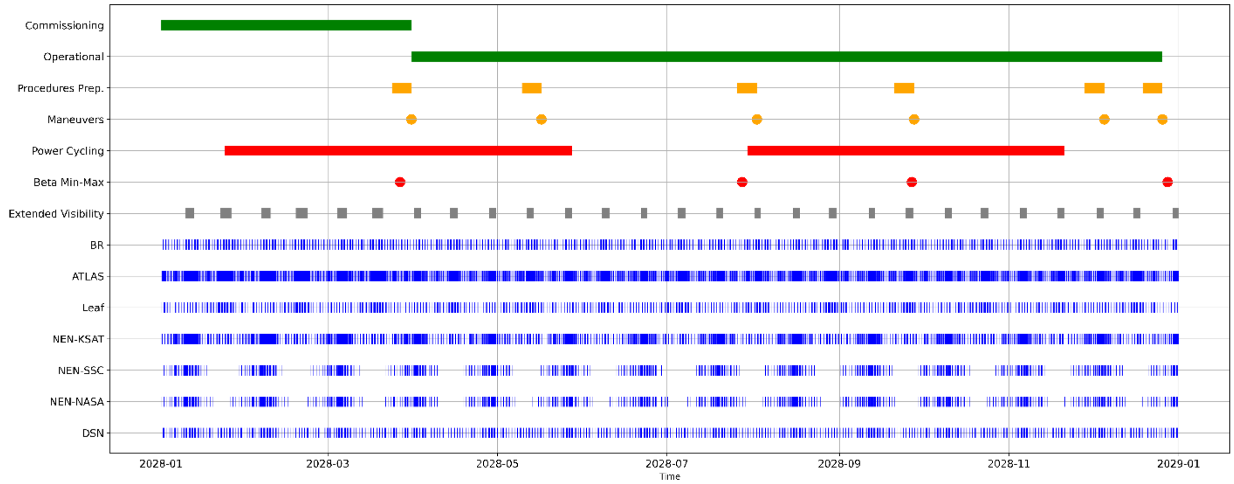

Figure 12: Operational Timeline Conceptual View

The timeline presents different events of the mission that can be defined by a set of dates. It provides a view of potential conflicts and highlights areas where more analysis is expected.

Conflicting or concurrence of specific events can be identified with such a timeline. Examples of where such preliminary timeline can be used are, 1) discussions if maneuvers might require access from ground stations, not for commanding but just for assisting and gathering of telemetry, 2) discussions on science measurements expected to occur under a specific set of required conditions (visibility, power, communication, etc.). It can also assist in resources planning. On the same example of maneuvers, understanding the logistic of the operation, resource available, and preparation time will also assist in the understanding of reaction time, among other factors of the mission.

Additionally, the consideration of specific solutions, services, and providers can be explored using the timeline. An example using the communication aspect of the SelenITA mission can be demonstrated based on Figure 12. As discussed in the paper, the current stage of the work considered a network of two antennas in Brazil, showing the main challenges related to logistics, link, and data budget. Still, in Figure 12 are presented the accesses for the antennas of a variety of networks (in blue) from commercial providers. This can be later used as the baseline for the analysis of the suitability of each of those services, providing a context for the interaction with the providers.

As the Reference Scenario is iterated over the next steps, such a timeline will provide a snapshot of the operational aspects of the mission and the effects of the engineering and science decisions.

CONCLUSION AND FUTURE WORK

Planning and developing a CubeSat mission to the Moon comes with challenges on the same level as the novelty of the scientific discoveries expected for the near future. Identifying such challenges at early stages is critical to any team, especially in the case of the first Brazilian mission to deep space.

The points covered in this paper outline the top-level concerns, indicating wherever possible the directions to be taken on the next steps. It is important to note that the challenges discussed herein are not the only ones and as the project progresses, unfolding aspects of each area will be identified and will have to be analyzed. Also, the directions pointed out here are not the only possible solutions to deal with each challenge and the conceptual view of the spacecraft under such limitations is not the final form of the system, which will still evolve until the Critical Design.

It is also important to note that the points covered in the paper deal solely with the engineering aspect of the mission, as the title suggests. The interaction of the Engineering and Science teams is a process kept under constant update to assure that the science objectives of the mission are in pace with what the system can do. Managerial and inter-organizational aspects of the mission are beyond the scope of this paper.

Different CubeSat missions to cis-lunar environment have discussed part of the challenges faced over the development and operation of their systems [24 42 43 44]. The analysis and incorporation of the lessons learned, and the solutions adopted by each of them are part of the next steps in the development of the SelenITA mission. Even though the ongoing CubeSat missions to the Moon and deep space vary in their objectives and characteristics, the CubeSat paradigm is present in all of them, making a valuable source of information for the next round of missions going to the Moon.

For the capabilities and limitations identified for the system, main directions are presented as potential ways to improve such limitations. The understanding of the current stage and next steps are presented as follows:

- **Orbit maintenance challenges:** Moon's gravitational field poses challenges to maintaining a stable orbit without or with a minimum number of interventions/corrections while still presenting a relevant case for Science Objectives. The in-depth knowledge available now and further analysis will enable a broader exploration of the orbital conditions and effects of lunar gravity on a 12U CubeSat.

- **Lifetime limitations challenges:** Necessity of constant interventions to account for the irregular gravitational field, posing hard requirements and limits on the operational lifetime. The in-depth knowledge available now with potential solutions depending on interaction with Science Objectives. Further steps will include consideration of different subsystems, failures, and the potential of an extended life.

- **Mission logistics and risks:** The necessity of interactions with the spacecraft is highly limited to access times, and available resources (ground stations). Logistics can be lengthy and reaction time must not be overlooked. A consolidated view of major operational markers of the mission assists in the understanding of potential issues. Close interaction with Science Objectives is required.

- **Thermal/Radiation environment challenges:** Heat sources poses hot conditions to the system. The necessity of evaluating critically the effects of Infra-red radiation, Solar Radiation, Albedo as well as critical areas of the satellite. Analysis in early stages using Systems Engineering tools to explore upper/lower-bound cases. Need for further developments.

- **Extremely limited budget (SWAP):** Tight margins to accommodate multiple instruments. Current CubeSat missions to the Moon (mostly 6U) take no more than 1.5U of payload. Required interaction and definition of Science Objectives.

- **Services and Supplier challenges:** Analysis based on assumptions of what service providers can achieve. Estimates based on public information of suppliers due to lack of direct response to inquiries. Margins are used to accommodate potential discrepancies requiring interactions with providers to start as soon as possible.

- **Reference Scenario:** The definition of a Reference Scenario proved to be a useful way to identify and address the challenges related to the SelenITA Mission. Further iterations are expected to incorporate responses to issues and concerns identified during the first iteration.

- **Payloads:** The main limitations identified can now be used to iterate with the science team on how to accommodate the desired payload instruments, serving also to define possible modifications of such instruments. The inclusion of payloads in the Reference Scenario will be a step in the definition of a formalized Concept of Operation for the SelenITA Mission.

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