

Article

Design of CubeSat-Based Multi-Regional Positioning Navigation and Timing System in Low Earth Orbit

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Abstract: The Global Navigation Satellite System (GNSS) provides critical positioning, navigation, and timing (PNT) services worldwide, enabling a wide range of applications from everyday use to advanced scientific and military operations. The importance of Low Earth Orbit (LEO) PNT systems lies in their ability to enhance the GNSS by implementing signals in additional frequency bands, offering increased signal strength, reduced latency, and improved accuracy and coverage, particularly in challenging environments such as urban canyons or polar regions, thereby addressing the limitations of the traditional Medium Earth Orbit (MEO) GNSS. This paper details the system engineering of a novel CubeSat-based multi-regional PNT system tailored for deployment in LEO. The proposed system leverages on a miniaturized CubeSat-compatible PNT payload that includes a chip-scale atomic clock (CSAC) and relies on MEO GNSS technologies to deliver positioning and timing information across multiple regions. The findings indicate that the proposed CubeSat-based PNT system offers a viable solution for enhancing global navigation and timing services, with potential commercial and scientific applications. This work contributes to the growing body of knowledge on LEO-based PNT systems and lays the groundwork for future research and development in this rapidly evolving field.



Academic Editor: Hyun-Ung Oh

Received: 23 November 2024

Revised: 20 December 2024

Accepted: 26 December 2024

Published: 31 December 2024

Citation: Tzanoulinos, G.; Ait-Mohammed, N.; Lappas, V. Design of CubeSat-Based Multi-Regional Positioning Navigation and Timing System in Low Earth Orbit. *Aerospace* **2025**, *12*, 19. <https://doi.org/10.3390/aerospace12010019>

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Keywords: LEO; GNSS; PNT; CubeSat; atomic clocks; spacecraft system engineering; STK; MATLAB; SPENVIS; DRAMA

1. Introduction

The GNSS, standing for Global Navigation Satellite System, is a network of satellites that provides global positioning, navigation, and timing information to users on Earth. The current GNSS, including the United States' GPS, Russia's GLONASS, Europe's Galileo, and China's BeiDou Navigation Satellite System, operate in MEO or GEO (Geostationary Earth Orbit). Additionally, Regional Navigation Satellite Systems (RNSSs), such as Japan's QZSS and India's NavIC, are also in operation. Nowadays, the GNSS has become an essential element in everyday life and is the sole technology that offers free, precise, and omnipresent PNT services to users around the world. As highlighted in the EUSPA EO and GNSS Market Report, global revenues from the GNSS reached around EUR 260 billion in 2023. By 2033, these revenues are projected to climb to EUR 580 billion, with over 80% of this total coming from services enabled by GNSS devices. Additionally, global shipments of GNSS units are expected to reach 2 billion per year by 2027, with nearly 9 billion GNSS devices anticipated to be in use by 2033. Predominantly, revenues from GNSS components,

receivers, system integrators, and software/added-value services are generated by US and European companies. The US accounts for the largest market share at more than 30%, with Europe trailing at close to 25% [1]. Traditionally, navigation satellites have been placed in MEO. In these orbits, the satellites traverse the sky at a slow pace and provide global coverage via low power navigation signals. However, implementing navigation satellites in LEO offers several advantages. A significant benefit of LEO compared to MEO is the reduced spreading loss, which results in stronger signals on the ground. Additionally, LEO provides faster movement across the sky (mean motion), enhancing geometric diversity and leading to multipath whitening. More specifically, the path loss of signals from LEO satellites is so lower than those in MEO that they could have approximately 1000 times (30 dB) greater signal strength. Thus, they are resistant to interference and are advantageous for navigation in urban and indoor settings. Also, the increased signal strength along with the introduction of new signal frequencies make them more resilient against jamming or spoofing [2,3].

Nevertheless, it must be mentioned that a key drawback of LEO is its smaller satellite footprint, necessitating a significantly larger number of satellites to achieve the same coverage as MEO. This was one of the fundamental considerations in the design of the current MEO GNSS constellations. In this era, the “New Space” movement is revolutionizing the space industry with innovative approaches that significantly reduce costs and enhance accessibility. Unlike traditional, government-led space activities, New Space focuses on commercial viability, innovation, rapid development cycles, and cost reduction, making space more accessible and affordable. This sector is marked by the entry of numerous private companies like SpaceX and Blue Origin, fostering competition and technological advancements through faster development cycles and cost-effective manufacturing processes. Public–private partnerships are common, leveraging government support while harnessing private sector efficiency. In this environment, small satellites and nanosatellites are gaining popularity. There is high interest in CubeSats; small, standardized satellites that are much cheaper and quicker to develop than traditional satellites. CubeSats have democratized space exploration by allowing universities, small businesses, and even amateur groups to conduct space missions. They are usually built using COTS (commercial off-the-shelf) components, which further drive down costs and development times. This approach contrasts sharply with the bespoke components typically used in traditional space missions, leading to a more agile and flexible space industry. The combination of New Space initiatives, CubeSats, and COTS components is enabling a broader range of entities to participate in space exploration, fostering innovation, and opening new commercial opportunities in the space sector. Having said that, this paper describes the system engineering of a CubeSat-based LEO PNT system able to provide services to country-sized areas of interest.

2. Materials and Methods

As the potential of LEO navigation has begun to be explored, the concept of LEO-PNT has emerged. There are three methods for utilizing LEO constellations for positioning: leveraging LEO signals of opportunity, hosting payloads on satellites designed for PNT applications, and utilizing fused payloads embedded within telecommunication payloads [4,5]. This paper is focused on the hosted payload approach.

2.1. PNT Payloads

Space-based navigation systems utilize stable atomic clocks aboard satellites in well-defined orbits, emitting signals to Earth. These satellites establish a space–time reference frame, offering time and positioning information. A minimum of four satellite signals is necessary to accurately determine both position and time. Additional satellite signals

enhance precision and aid in detecting discrepancies and potential issues within the data. For such systems, a time epoch, defined with a specified reference frame that includes a location (e.g., center of Earth) and starting time (e.g., UTC), is necessary.

The philosophy behind this kind of systems is based upon integrating atomic frequency references (AFRs) aboard satellites that maintain a consistent frequency output, which is utilized by other electronic components to generate signals for transmission to Earth. Code data and timestamps are implemented into the signals before their final transmission to the ground. However, noise, timing variability, and environmental influences that are beyond the AFRs impact the overall performance of the system.

The clocks onboard must consistently deliver a precise and stable frequency along with a clearly defined time epoch. Any inaccuracies or instability in the clock time, or unpredictable frequency drifts, will result in accumulating errors in the pseudorange solution. To illustrate, one could consider that the GNSS demands such stable clocks that, for instance, to achieve a 1 m precision in measurements, a timing uncertainty of approximately 3 nanoseconds is required. Maintaining such timing uncertainty for a day necessitates a fractional frequency instability of approximately $3.5 \times 10^{-14} = \left(\frac{3 \text{ ns}}{86,400 \text{ s}} \right)$. This level of stability is attainable with high-quality atomic clocks, but not with other existing clock technologies or oscillators [2].

CubeSat-Based PNT Payloads

To comply with the CubeSat specifications, a PNT payload must adhere to strict constraints regarding mass, power consumption, and volume. Over the past decades, research and development efforts have focused on miniaturizing atomic clocks for commercial use, led by DARPA, which recognized the potential for small, battery-powered versions. Today's CSACs take less volume than 17 cm³ while weighting about 35 g. Moreover, they tend to have extremely low power consumption (<120 mW), and they can function across a broad temperature spectrum ranging from −40 to 85 degrees Celsius. Thus, CSACs are the ideal AFR for a CubeSat platform.

Since PNT payloads are not popular in the CubeSat industry yet, one could develop a PNT payload by utilizing COTS components as building blocks, which is also the approach of the ATOMIC payload of the AIOTY-CUBE [4]. Finally, in line with the New Space approach, the objective of a CubeSat-based PNT system is to function independently of a ground segment dedicated to timekeeping, like the ones required by MEO GNSSs. This can be achieved by utilizing GNSS observables for precise orbit determination and synchronizing the CubeSat's time reference with GNSS time, thereby eliminating the need for master clocks [4]. Such a payload would consist of a GNSS antenna and a GNSS receiver so that it obtains inputs about time and position from MEO GNSS, an On-Board Computer (OBC), and a CSAC to generate the system's time and ephemeris and finally a transmitter and an antenna to transmit the PNT signals. The architecture of the proposed payload in terms of function and hardware is outlined in Figure 1.

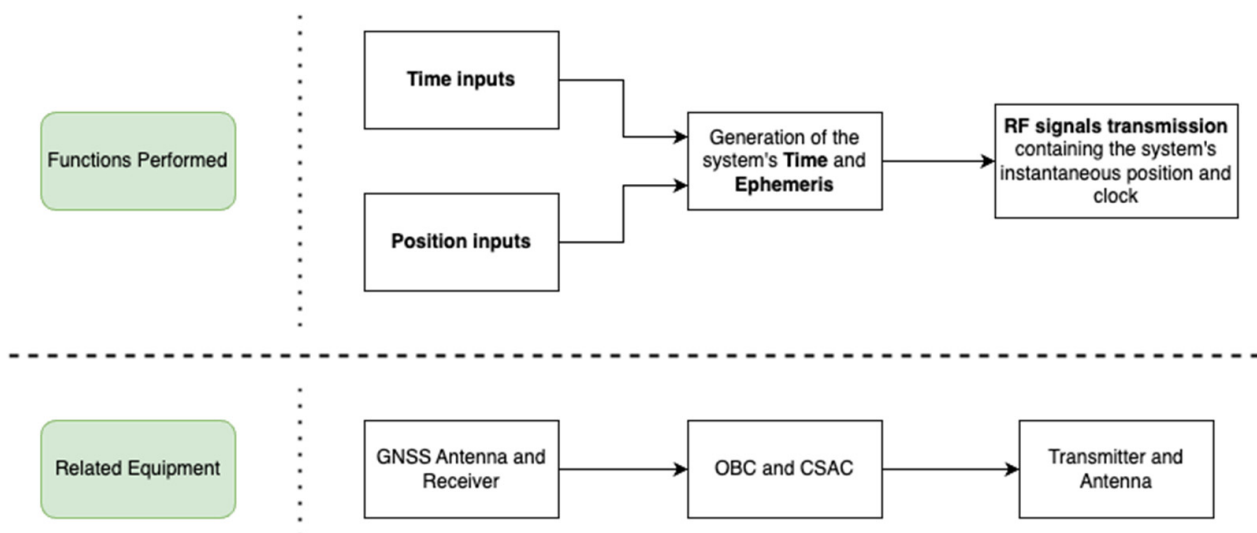


Figure 1. Architecture of a CubeSat-based PNT payload in line with New Space.

2.2. Mission Statement

The purpose of the mission is to design a LEO CubeSat Constellation that has the capability of providing persistent PNT signals in specific areas of interest, with the size of a country, around the globe. This shall be achieved by the CubeSats relying on the MEO GNSS for orbit determination and time synchronization (ODTS), facilitating a multi-tier architecture [6]. Time and frequency synchronization to existing GNSS constellations ensures compatibility and interoperability. No dedicated ground segment is needed for timekeeping, while the requirements for space infrastructure are kept at a minimum. Based on the heritage from the ERMIS CubeSat mission, the spacecraft (S/C) shall mostly consist of GOMspace components, while the ADCS shall be from CubeSpace. The ERMIS mission, led by the UOA (University of Athens), is Greece's first small satellite constellation and is part of the Greek CubeSat in-orbit validation projects. It consists of one 8U and two 6U CubeSats and aims to demonstrate technologies like 5G IoT connectivity, inter-satellite links, Earth observation applications, on-board data compression, and optical laser communications [7].

Following the philosophy described above, the PNT payload housed onboard shall consist of a multi-frequency GNSS receiver along with an antenna, a CSAC, a navigation computer, a transmitter, and an additional antenna to transmit the PNT signals. Like the ATOMIC payload, this architecture allows for the GNSS data to be processed through a real-time navigation filter to achieve accurate ODTS, while the S/C's predicted trajectory is used to create its ephemeris. The CSAC plays a critical role as the onboard time and frequency reference. Its phase and frequency are consistently adjusted to align with the GNSS broadcast time, using the onboard system. A calibration signal ensures that the signal generator's phase is fully synchronized with GNSS. This enables the generation of GNSS-synchronized navigation and ranging signals, which can be used alongside GNSS signals for single point positioning [4]. This kind of system can provide services to an unlimited number of users. Additionally, Satellite Laser Ranging (SLR) shall be used to confirm the satellite's orbit when it passes over an SLR-capable ground control station (GCS).

Furthermore, COTS components with publicly available datasheets that contain the necessary information for the system design shall be preferred.

The mission's ground segment shall consist of UOA's GCS, along with other GCSs in Greece that are available as a service. The constellation shall be deployed through multiple launches, requiring consideration of a range of different launch providers. Small launch

vehicles should be an option to consider in case the desired orbit is rarely available through Rideshare launches. Also, separate launches are considered necessary for S/C scheduled for different orbital planes. The number of S/C in the constellation shall be kept as low as possible.

Each S/C shall undergo LEOP (Launch and Early Orbit Phase), a Commissioning phase, and a Payload validation phase before starting its Nominal operations phase. Each satellite shall have an operational lifetime of at least 3 years. After the end of the mission, the S/C shall be deorbited according to the relevant regulations and laws in place.

As stated above, the system is aimed to provide persistent PNT services to specific areas/countries. This design allows for service in multiple areas in similar latitudes while minimizing the payload duty factor. In this paper, the focus is the European region; thus, the area is selected to be Germany since it is a Central European country of substantial area ($\sim 357,600 \text{ km}^2$) where many major metropolitan districts are located. Overall, such a satellite system would enhance current PNT services over Germany by implementing strong signals from LEO and new geometries that are not achievable by the MEO GNSS. This would enhance applications like drone mobility, autonomous driving, precision agriculture, and emergency response.

3. Results

3.1. Mission Architecture

To address the mission, an 8U CubeSat platform shall be utilized that shall transmit S-band PNT signals. In order to provide the coverage needed, a Walker Delta $61^\circ: 100/10/1$ constellation at 550 km shall be utilized. The mission architecture is illustrated in Figure 2, while Mission Objectives were stated, as seen in Table 1, to evaluate the mission's success.

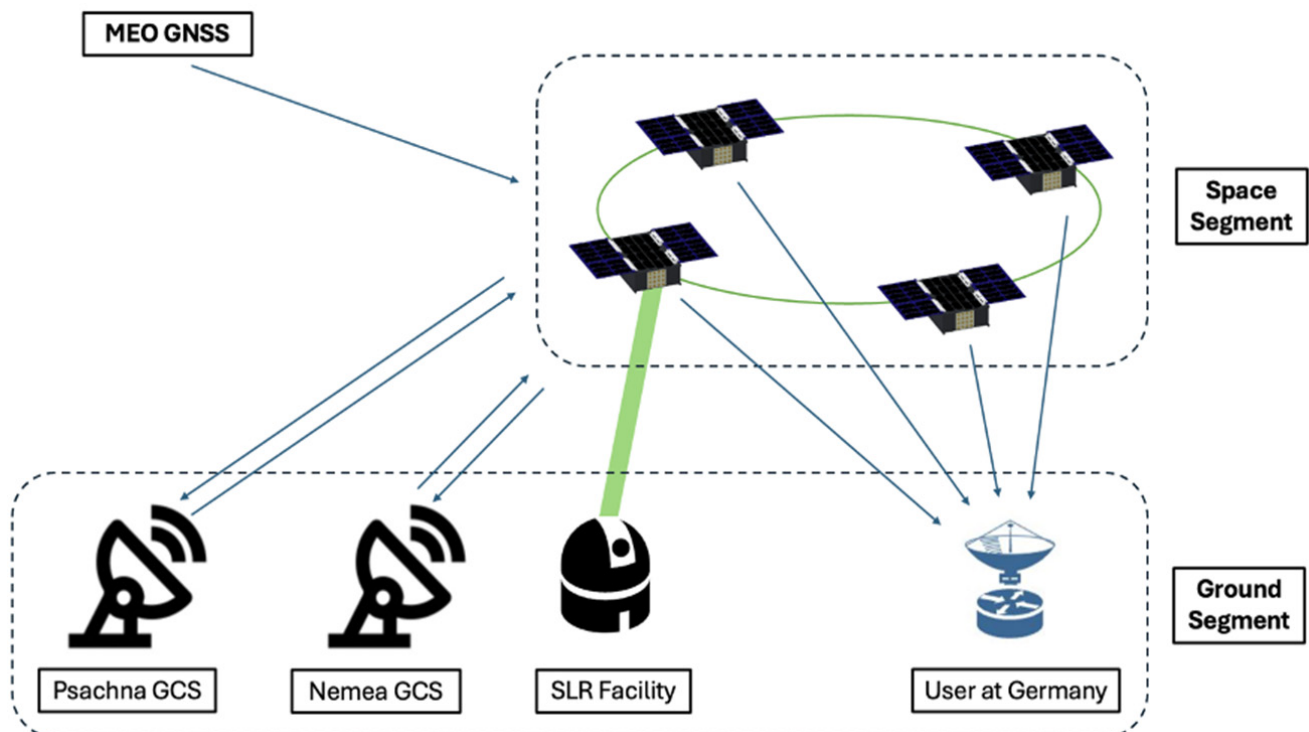


Figure 2. Mission architecture.

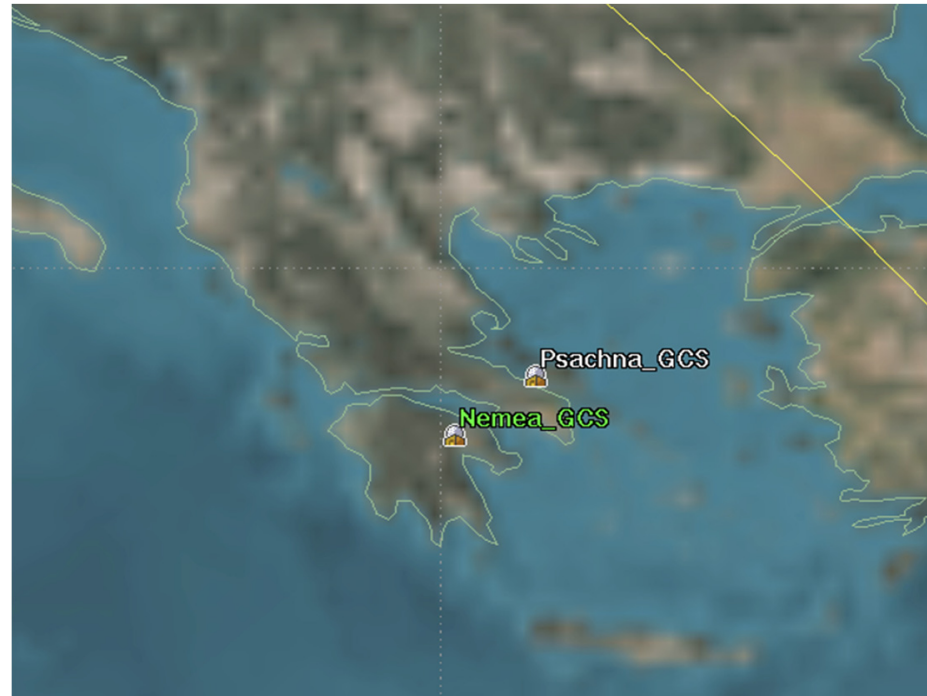
Table 1. Mission Objectives—success criteria.

Objective Number	Mission Objective	Success Criteria
Objective 1	Spacecraft shall perform ODTs by relying on MEO GNSS.	At least one spacecraft shall perform ODTs once by relying on MEO GNSS throughout the mission duration.
Objective 2	The mission shall demonstrate SLR.	The altitude of a single spacecraft in orbit shall be measured via SLR technique at least once.
Objective 3	The system shall provide PNT services to Germany.	A user at ground level in Germany shall be able to determine its 3D position via the system at least once throughout the mission duration.
Objective 4	Spacecraft shall remain operational for the full 3-year mission duration.	At least one spacecraft shall be operational for 3 years in orbit.

3.2. Ground Segment

The ground segment is composed of two bands: UHF and S-Band, with S-Band being the main TTC link and UHF being the backup and the main one during LEOP. The main GCS shall be the one that UOA has been developing at Psachna, Greece. However, a second GCS is considered to ensure a successful handling of the constellation. Since UOA does not own another GCS, GCS as a service is the go-to option. For this, KSAT's GCS at Nemea, Greece, is baselined.

UOA's GCS is located at $38^{\circ}34'10.39''$ N, $23^{\circ}38'54.55''$ E (WGS'84) with an elevation of 42 m (above sea level), and KSAT's GCS is located at 37.8° N and 22.6° E, as illustrated in Figure 3 [8].

**Figure 3.** GCS locations.

3.3. Mission Analysis

3.3.1. Constellation Design

As mentioned in the Mission Statement, the system shall provide persistent PNT services over the area of Germany. That means that a satellite constellation shall be designed in a way that at least four S/Cs are always covering the service area.

It is essential that the S/Cs are placed into circular orbits with the same altitude to ensure that the distance between the S/C and ground users and GCSs is the same for all S/Cs throughout the mission. Additionally, the constellation's configuration should always remain constant. Thus, Walker constellations are evaluated. Since the area of service is not located in a polar area, Walker Delta geometries are the ones under inspection, with single-shell constellations being the ones considered.

Moreover, the design of the constellation must conform with other parameters, like the Payload link budget, the TTC link budget, the expected radiation dose, the power budget, deorbit plan, etc. Certain criteria deemed critical are analyzed in depth as there is no assurance they will be met. In contrast, criteria for which the preliminary analysis indicates a sufficient margin surplus, making it highly likely they will be met, are not considered further at this stage.

STK simulations of the constellation configuration were performed between the altitudes of 550 km and 750 km. A minimum elevation angle of 5° is assumed, meaning that, for lower elevation angles, no contacts are considered. This is chosen to mitigate the effects of the signal traveling through a longer atmospheric path, such as increased attenuation, multipath interference, and fading. Additionally, low elevation angles are more prone to encountering obstacles like buildings, trees, or terrain features, which can block or degrade the signal. The philosophy behind the coverage is that the S/C shall have their Payload antennas mounted on the nadir side and track the area of service as they pass near it. That way, the area covered by each S/C is constrained by the FOV (Field of View) of the Payload antenna. The Payload antenna is chosen to be GOMspace ANT2150-DUP. The Payload antenna's FOV is limited to a cone with a half-angle of 30° , ensuring that the antenna achieves a gain of 5.5 dB at the cone's edge [9]. Whether or not a constellation configuration meets the coverage requirements is concluded via a 4-Asset Coverage Figure of Merit object in STK that assesses if the area of Germany is always covered by a minimum of four satellites for the duration of one year.

In order to design a Walker Delta constellation, the parameters that need to be defined are the altitude, the inclination, the total number of satellites, the number of orbital planes, and the relative spacing between satellites in adjacent planes. Among those, the critical parameters are the altitude and the total number of satellites, and the main aim is to minimize them, while the variable describing the relative spacing between satellites in adjacent planes is always set to 1. The constellation design is performed in an iterative approach that implements the trial-and-error method and is described in Figure 4.

The first step for such an analysis shall be to do a preliminary tradeoff analysis between altitude, which directly translates to Payload and TTC Link Budgets and TID (Total Ionizing Dose).

For the first steps of the analysis, the inclination is selected to be 56° to match the one of Galileo because Galileo is a system optimized for coverage over Europe; and, even though this is a comparison between a LEO and a MEO system, at this point, it is considered an acceptable educated guess. This process led to finding an optimal inclination value of 61° and subsequently to Table 2, which includes information on the total number of satellites, the number of orbital planes, the minimum received power of PNT signals for a user at ground level, the TID of the S/C, and the average GDOP (Geometric Dilution of Precision) that were considered in the final tradeoff.

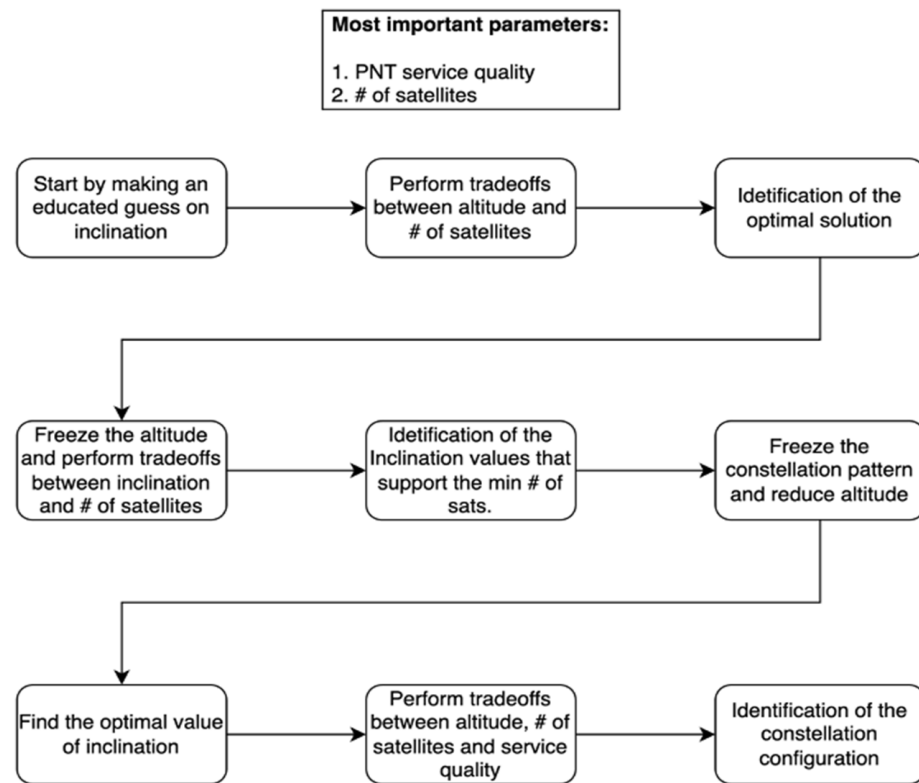


Figure 4. Constellation configuration tradeoff process.

Table 2. Tradeoffs between altitude, number of satellites, and Payload link budget for the inclination of 61° .

Altitude (km)	Total # of S/C	Orbital Planes	Min PNT Signal Strength at Ground Level (dBW)	TID in 3 Years * (krad)	Avg GDOP
750	80	10	−158.9	22.6	7
	81	9			4.5
700	90	10	−158.5	19.5	4.1
650	90	10	−158.1	17	5.5
600	90	10	−157.7	14.5	7.7
550	100	10	−157.2	12.2	5.3

* It is assumed that the S/C components are shielded by aluminum with a thickness of 1.5 mm and that the Sun is at solar maximum. The calculation was conducted via SPENVIS [10].

Given that the system's requirements, whose definition is beyond the scope of this paper, dictate specific floor values for the PNT signal strength, TID, and the average GDOP, the Min PNT signal strength at ground level should be above -157.3 dBW, ensuring that the PNT signals of the proposed system achieve a higher minimum signal strength than the Galileo E1 signal [11]. The TID should be less than 15 krad, based on heritage from ERMIS CubeSat mission regarding the radiation tolerance of COTS, and the GDOP should be less than 6 so that the system's performance can be considered sufficient [12]. The optimal solution that conforms with those is a Walker Delta 61° : 100/10/1 constellation at 550 km and is visualized in Figure 5.

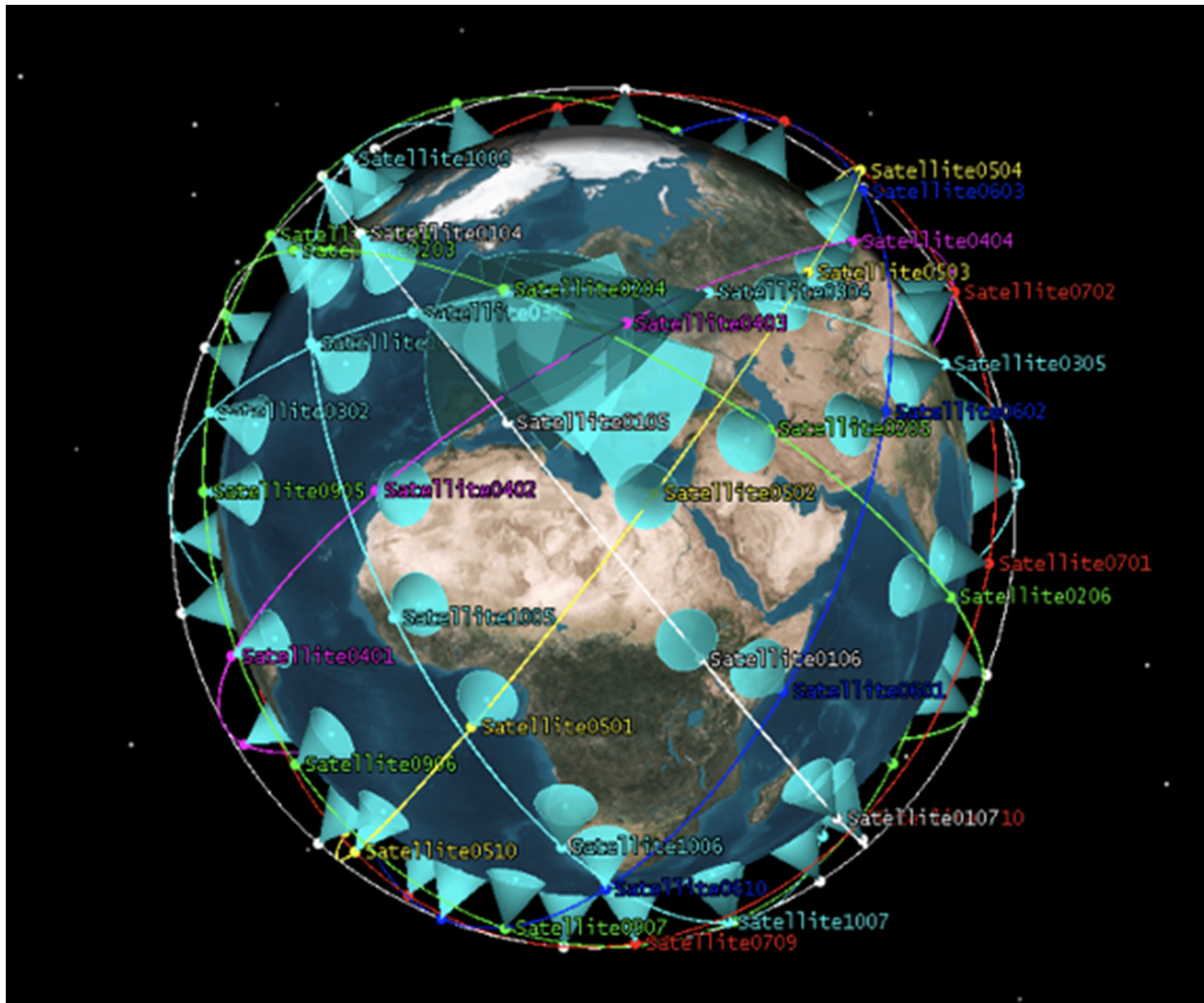


Figure 5. Walker Delta 61°: 100/10/1 constellation at 550 km that tracks Germany.

3.3.2. Contact Analysis

The contacts between a single S/C and the service area have been simulated and resulted in an average contact duration of 10.7 min and to an average of 7.2 contacts per day. The average revisit time (ART) of each S/C is about 3 h.

Additionally, the contacts between a single S/C and GCSs have been simulated. In the STK analysis performed, a minimum elevation of 20° was assumed and any contact with a duration less than a minute was considered insufficient for establishing a link and was discarded. This resulted that each S/C has an average contact duration of 267 s with both GCS and an average contact number per GCS of 2.6 per day.

3.3.3. Solar Eclipse Analysis

The solar eclipse analysis performed in STK resulted in an average eclipse duration of 31.5 min, a maximum eclipse duration of 35.6 min, and a maximum of 14 eclipses per day. The S/Cs having an altitude of 550 km have an orbital period of 95.4 min.

3.4. Space Debris Mitigation

The mission plan does not involve any intentional break-up in orbit. The components susceptible to accidental breaking-up and releasing debris are the two GOMspace NanoPower BP8 battery packs, the CubeSpace reaction wheels, and the Enpulsion NANO

propulsion system. Thus, their health shall be monitored through an array of sensors that capture their functional and environmental state throughout the mission lifetime. Apart from that, the space debris mitigation analysis is conducted via the DRAMA software [13].

Each S/C's cross section is calculated for different orientations via the DRAMA–CROC as seen in Figure 6 and Table 3.

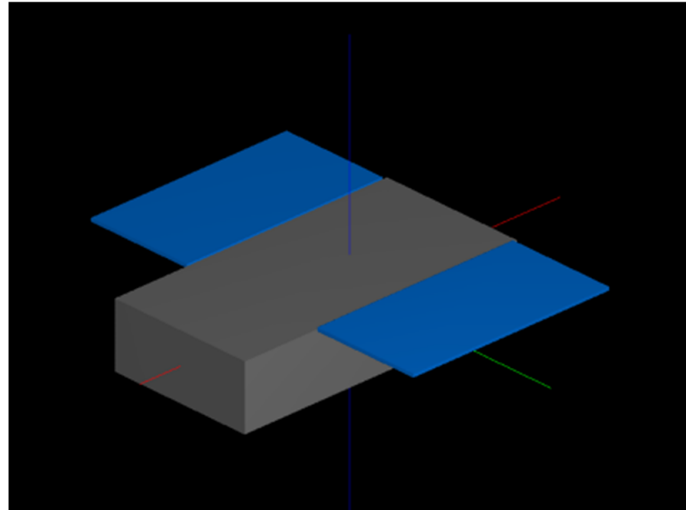


Figure 6. CROC 3D model.

Table 3. S/C dimensions and cross sections.

Main Structure Dimensions	454 mm × 226.3 mm × 100 mm
Deployed solar panel dimensions (×2)	326.55 mm × 167.17 mm × 5.2 mm
Minimum cross section—2U side	243 cm ²
Average cross section (randomly tumbling case)	1294 cm ²
Maximum cross section	2133 cm ²

Additionally, the probability of space debris or meteoroid impact was calculated via DRAMA–MIDAS as depicted in Figure 7.

Since the total number of catastrophic impacts N is known, the probability of catastrophic impacts can be calculated via $P_{cat} = 1 - e^{-N}$ [14]. The results are shown in Table 4. In all cases, it is lower than the critical value of 10^{-4} , as stated in ESA Space Debris Mitigation Requirements document, and is, therefore, considered acceptable [15].

Table 4. Probability of catastrophic impact before the end of the mission.

Object Diameter	1 mm to 1 cm	Above 1 cm
Probability of catastrophic impact during 2029	7.63×10^{-9}	5.09×10^{-6}
Probability of catastrophic impact during 2030	1.06×10^{-8}	7.11×10^{-6}

Furthermore, an orbital decay analysis was performed using DRAMA–OSCAR to verify compliance with the 5-year deorbit requirement.

The S/Cs shall have station-keeping capabilities; thus, their original altitude of 550 km shall be maintained for the full 3-year nominal operations phase. Based on the ΔV budget, in the worst case, 70 m/s will be allocated to lower the S/Cs's altitude before allowing them to passively decay. It is assumed that a single retrograde burn of 70 m/s is performed in a random point of the initial orbit.

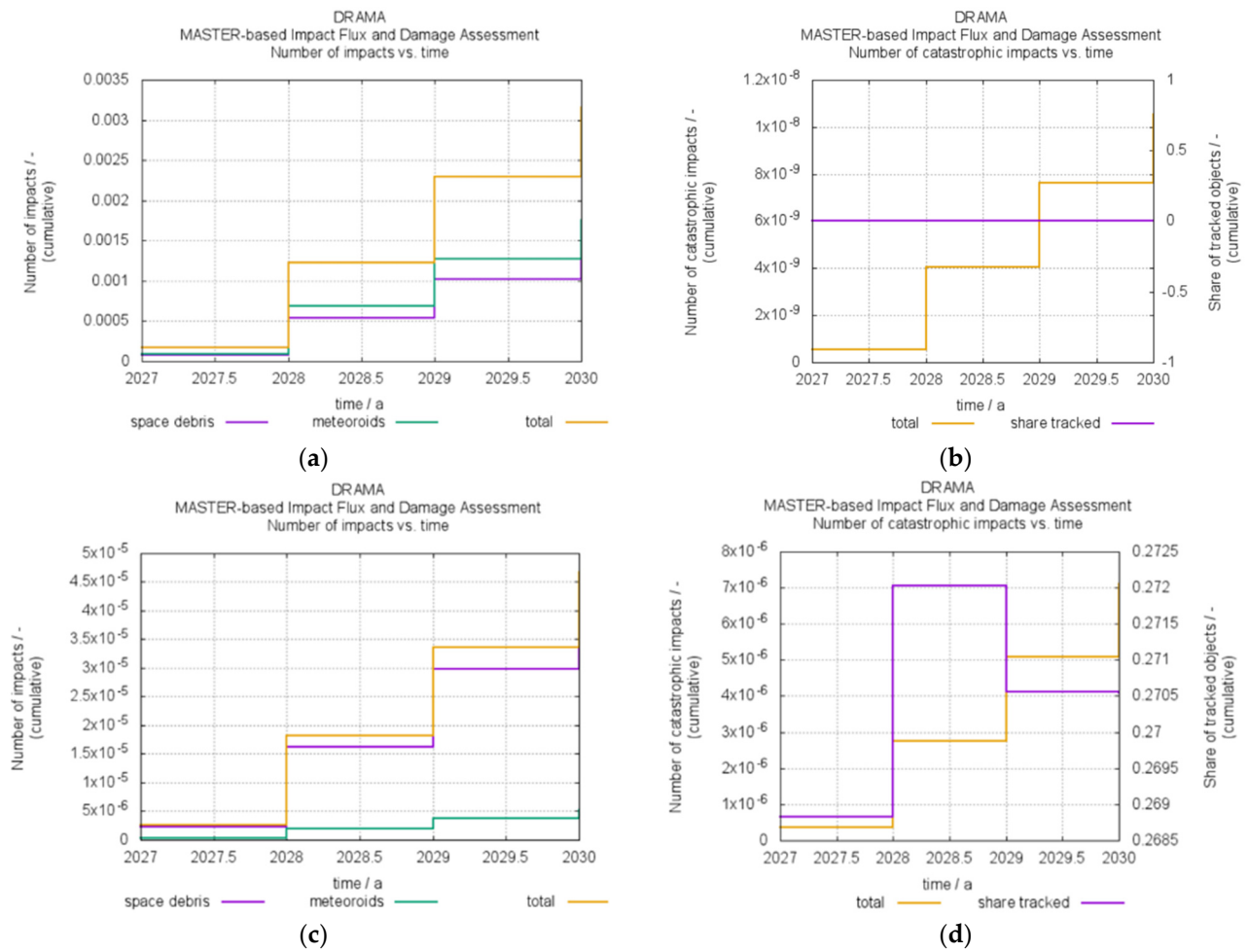


Figure 7. Number of impacts (a) and catastrophic impacts (b) vs. time for objects between 1 mm and 1 cm and number of impacts (c) and catastrophic impacts (d) vs. time for objects above 1 cm.

Then, via the Vis-viva equation, it is possible to calculate the new orbit after the maneuver.

$$u^2 = GM \left(\frac{2}{r} - \frac{1}{a} \right)$$

where

u is the orbital velocity of the satellite;

r is the distance between the satellite and the Earth's center;

a is the length of the semi-major axis;

G is the gravitational constant ($6.67 \times 10^{-11} \text{ N} \cdot \text{m}^2 \cdot \text{kg}^{-2}$);

M is the Earth's mass ($5.972 \times 10^{24} \text{ kg}$).

The new semi major axis is calculated to be $a' = 6811 \text{ km}$, while the eccentricity is $e = 0.017$. By having these elements to be used as inputs in the DRAMA–OSCAR tool, the orbital decay analysis can be performed. As illustrated in Figure 8, it results that the S/Cs shall deorbit in 1.7 years.

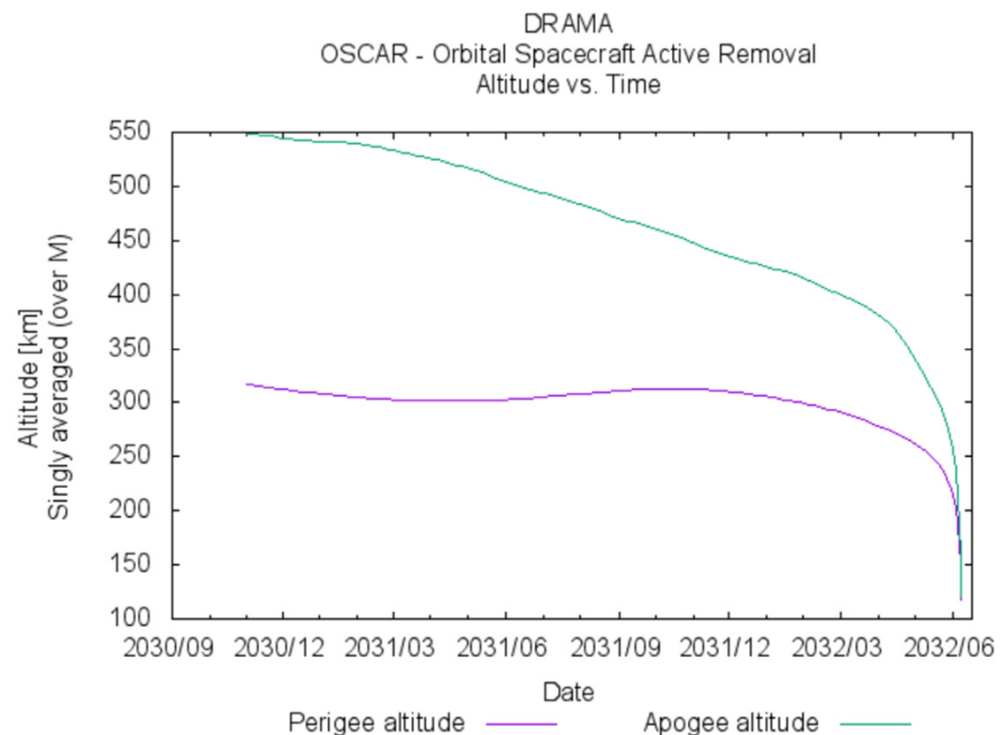


Figure 8. Orbital decay analysis—passive disposal after deorbit maneuver.

Finally, the S/C's re-entry is modeled through DRAMA-SARA. The S/C was modeled as a solid 8U cuboid ($0.8 \times 0.2 \times 0.1$ m) made entirely of aluminum (DRAMA-AA7075 material from DRAMA's library). As depicted in Figure 9, the analysis concluded that the S/C is completely disintegrated in less than 1 h after atmospheric re-entry and at the altitude of 70 km.

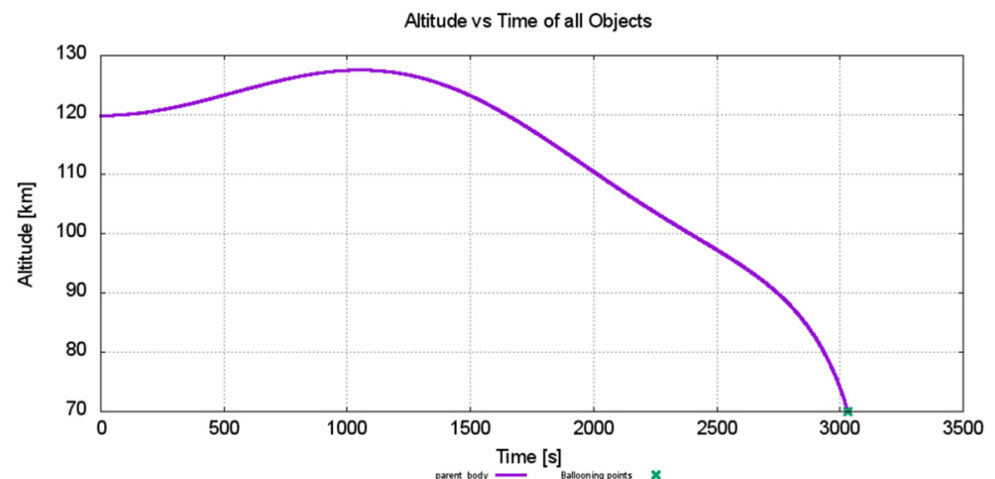


Figure 9. SARA re-entry simulation—altitude vs. time.

3.5. Space Segment

The S/C BUS consists of the following subsystems: Command and Data Handling Subsystem (CDHS), Electrical Power Subsystem (EPS), Attitude Determination and Control Subsystem (ADCS), Telemetry and Telecommand (TTC) Subsystem, Propulsion Subsystem (PRO), and Structure Subsystem (STR). The S/C's baselined configuration can be seen in Table 5, while its CAD is shown in Figure 10.

Table 5. S/C baseline configuration.

Subsystem	Component
CDHS	GOMspace NanoMind A3200 GOMspace DMC-3 GOMspace GPS Kit
EPS	GOMspace NanoPower P80 System GOMspace NanoPower BP8 ×2 Body Mount Solar Array GOMspace NanoPower MSP-B-4 ×4 Body Mount Solar Array GOMspace NanoPower MSP-B-4 ×1 Deployable Solar Array GOMspace NanoPower DSP-90deg ×2
ADCS	CubeSpace CubeADCS Core CubeSpace CubeTorquer CR0008 ×3 CubeSpace Cubewheel CW0057 ×4 CubeSpace CubeSense Fine Sun Sensor ×1 CubeSpace CubeSense Coarse Sun Sensors ×6 CubeSpace CubeSense Earth CubeSpace CubeMag Compact CubeSpace CubeMag Deployable
TTC	GOMspace NanoCom AX100 GOMspace NanoCom ANT-6F (UHF) GOMspace NanoCom Link-S GOMspace NanoCom ANT2150-DUP
PRO	Enpulsion NANO
STR	GOMspace 8U Structure GOMspace 6U Side Panels GOMspace NanoUtil MSP-FPP Flight Preparation Panel Custom mounting components
PNT Payload	PNT Solution Generator NANOLink-boost Gen2 Transceiver GOMspace NanoCom ANT2150-DUP LRA

As was already mentioned, the PNT Payload is based on AIOTY-CUBE's ATOMIC payload and consists of the following components:

1. PNT Solution Generator:
 - a. Talysman GNSS Antenna;
 - b. NovAtel OEM719 GNSS receiver;
 - c. Microchip SA.45s CSAC;
 - d. Payload OBC.
2. NANOLink-boost Gen2 Transceiver.
3. GOMspace NanoCom ANT2150-DUP.
4. Laser Retroreflector Array (LRA).

Overall, the PNT Payload is considered to have a mass of about 1.5 kg and take up a volume of 1U. Its maximum power consumption is 31.25 W.

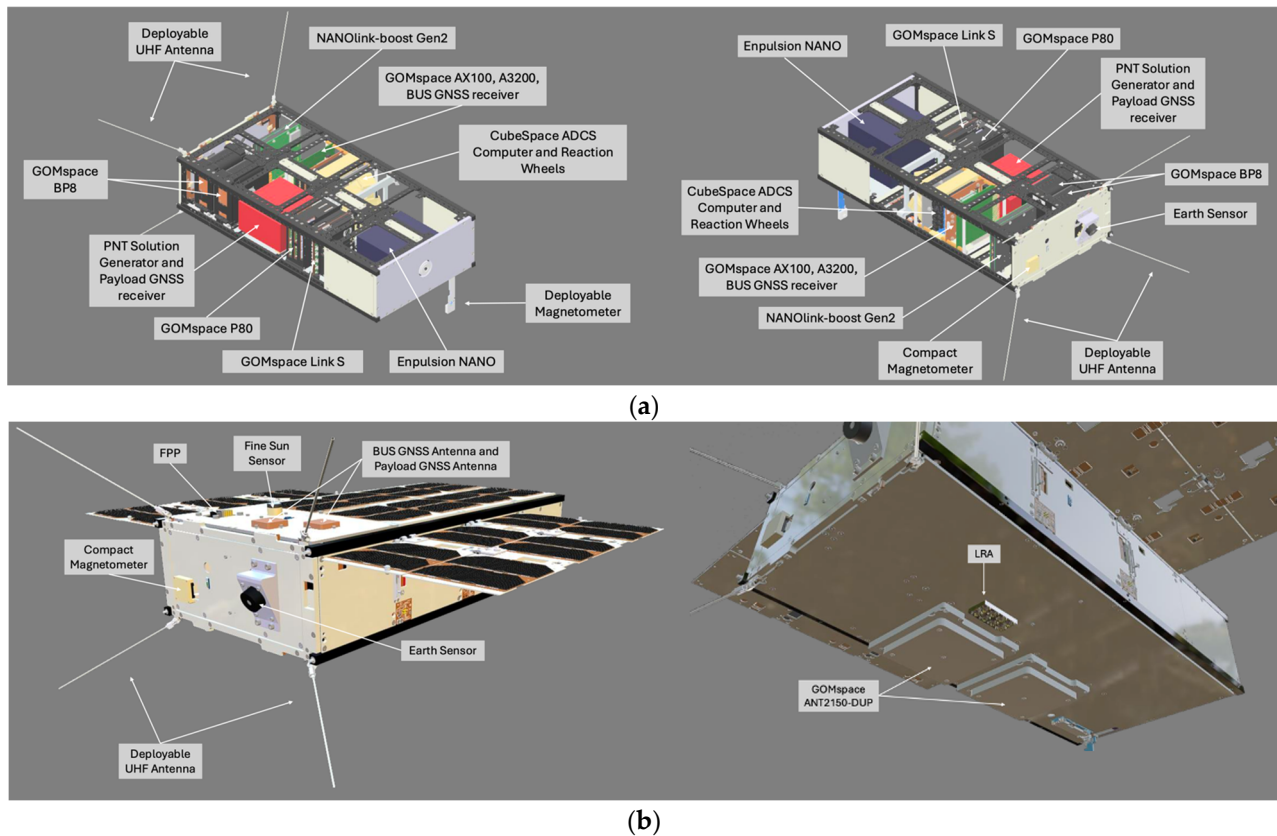


Figure 10. S/C internal (a) and external (b) view.

3.6. System Budgets

The margin philosophy applied is as follows. Regarding the mass and power budget, a margin of 5% is applied for existing units, 10% for existing units requiring minor modifications, and 20% for new developments or existing units requiring major modifications. On top of that, a 20% margin is applied on system level. A 50% margin is used for the ΔV , while a margin of 5 dB is applied for communication link budgets.

3.6.1. Mass Budget

In the mass budget shown in Table 6, the harness is assumed to have a mass equal to 5% of the total mass of all components. Finally, given that a deployer for 8U CubeSats could have a maximum mass capacity of 15 kg [16], this leaves a mass margin of the deployer pod of over 3.5 kg.

Table 6. Mass budget—subsystem level.

Subsystem	Mass (g)	Mass with Margin (g)
CDHS	132	138.6
EPS	2564	2692.2
TTC	540	567
ADCS	819	860
PRO	900	945
STR	1873	2024
PNT Payload	1558	1821.4
Harness	374.3	393
Total	8760.3	9441.1
Total mass with system margin (g)		11,329.3

3.6.2. Power Budget

Regarding the power budget, the S/C is equipped with 40 solar cells in total; 16 body mounted and 24 deployable solar cells. According to GOMspace, each solar cell can generate up to 1.2 W in LEO [17,18]. However, for the following analysis, it is assumed that a solar cell can generate 1 W when it is exposed vertically to sunlight. That way, when the S/C is in Sun-Tracking Mode and the solar arrays are fully deployed, it is assumed that 40 W can be generated.

As seen in Figure 11, a MATLAB Day in the life simulation has been conducted to calculate the batteries' state of charge (SoC) based on the following assumptions. The EOL (End-of-Life) capacity of a single battery is 50 Wh, and since there are two batteries on board, the S/C has a total EOL battery capacity of 100 Wh. The S/C operates in the Sun-Tracking Mode, which can generate 40 W of power while exposed to sunlight. Over a 24 h period, the S/C will enter the PNT Service Mode an average of seven times, corresponding to the typical number of daily contacts with the service area. Those are separated by approximately 3 h, which is the ART to the service area. Each contact, based on STK analysis, lasts an average of 10.7 min. Additionally, the S/C will engage in the S-Band Link Mode twice within a 24 h period for 5 min each time. After completing either an S-Band Link Mode or PNT Service Mode, the S/C will return to the Sun-Tracking Mode. It is assumed that no power is generated (0 W) when the S/C is in a different mode than the Sun-Tracking Mode. Moreover, the PNT Solution Generator shall maintain a duty factor of 100% during nominal operations. The CSAC needs to be monitored and steered a couple of hours prior to operating with the required performance, and since the ART of 3 h is a shorter timeframe, it needs to remain active at all times. The power consumption breakdown for the modes mentioned above is shown in Table 7.

Table 7. Nominal operation modes power breakdown.

Component/Function	Power w/Margin (w)	Sun-Tracking Mode	S-Band Link Mode	PNT Service Mode
Battery Heaters	1.1	100%	100%	100%
EPS	0.6	100%	100%	100%
GNSS kit	1.5	100%	100%	100%
CDHS	0.3	100%	100%	100%
ADCS-3-axis control	2.8	100%	100%	100%
UHF Rx	0.5	90%	0%	0%
UHF Tx	3.2	10%	0%	0%
S-Band Tx/Rx	17.9	0%	100%	0%
PNT Solution Generator	3.0	100%	100%	100%
PNT Signal Transmission	37.5	0%	0%	100%
Mode Consumption w/Margin (W)		10.1	27.1	46.8
Mode Consumption w/system margin (W)		12.1	32.6	56.1

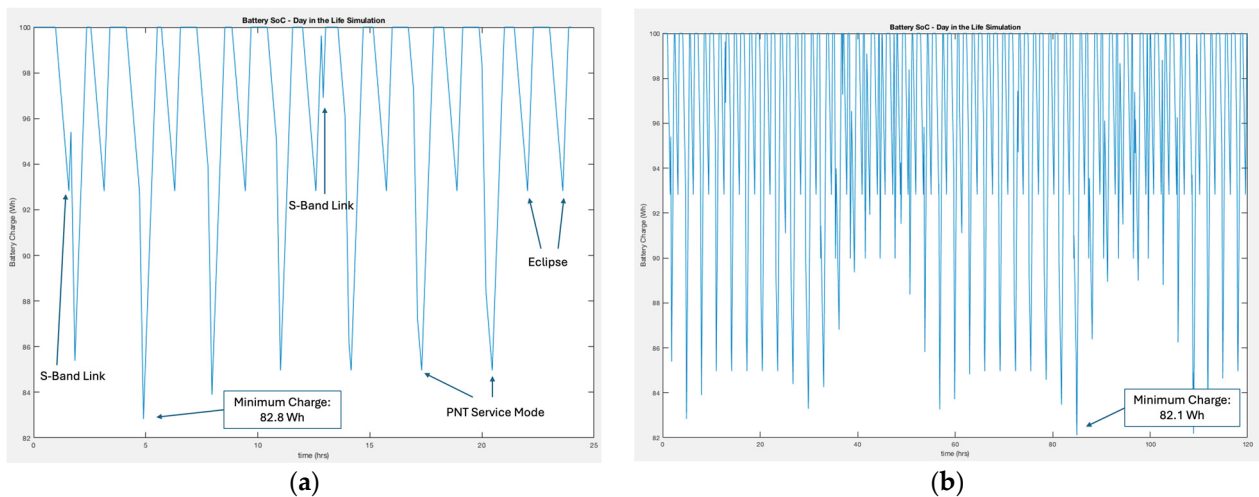


Figure 11. Battery SoC—day in the life simulation—nominal scenario for a single day (a) and for 5 days (b).

3.6.3. Link Budget

A link budget analysis was performed, taking into account UOA's GCS and the TTC components of the S/C. This resulted in the UHF link at 403 MHz to close when the elevation is 5° with a surplus of 20 dB for uplink and 16 dB for downlink. Regarding the S-Band TTC link, it is able to close with the 5 dB required margin at an elevation 25° in the case of uplink at 2030 MHz and at an elevation 30° in the case of downlink at 2210 MHz.

Moreover, the PNT signal power at ground level was calculated, which is also assumed to be transmitted at 2210 MHz, and it resulted in -157.19 dBW at 5° of elevation and -143.89 dBW at the elevation of 90° . As there is not currently an existing standardized user segment for LEO PNT applications, no analysis was conducted on compatibility with the user segment, and it is not considered a part of this paper.

3.6.4. ΔV Budget

Regarding the ΔV budget, it consists of two sources of fuel consumption; the station-keeping of the S/C and the phasing maneuver required for the S/C to be equally spaced within each orbital plane. An STK analysis was performed in both cases.

The primary causes of a satellite's loss of altitude are atmospheric drag and solar radiation pressure. Their combined vector was modeled in STK, and by analyzing its values over a year, the average deceleration exerted on the satellite can be determined. Therefore, the ΔV required for 3 years of station-keeping can be calculated. In the STK simulation, a worst-case scenario was considered, where the S/C's cross-sectional area was assumed to be 2133 cm^2 (S/C's larger cross section). This resulted in 41.7 m/s or 62.6 m/s with a 50% margin needed for station-keeping.

The mission requires ten S/Cs to be phased within a single orbital plane at an altitude of 550 km so that they are evenly spaced in it. To account for the worst-case scenario, the ΔV required to phase a single satellite by 180 degrees is to be calculated.

Additionally, it is assumed that after this maneuver is completed, N days or approximately $N \cdot 15$ orbital periods of the satellite at its original altitude of 550 km are going to pass before readjusting its orbit to the original one with a second burn.

The philosophy behind these maneuvers is that by performing a burn to change the altitude of the satellite, its period is going to change by ΔT . More specifically, by lowering the altitude of a satellite that has a period of T , it will change to T' , and the periods will be related by the formula $T' = T - \Delta T$. Over the course of N orbits, this difference in orbital period will accumulate into a 180-degree phase shift, equivalent to half of the original

orbital period, and then by performing a second burn in the opposite direction, the satellite shall return to its original orbit. This can be expressed mathematically as $N \cdot \Delta T = T/2$. By having knowledge of the S/C's initial state and its desired orbital period after the maneuver, it is possible to calculate the ΔV required via STK. This is achieved using the Astrogator propagator by inserting a "Target Sequence" item with a "maneuver" nested within it. The results are shown in Table 8.

Table 8. ΔV with respect to phasing duration.

Phasing Duration	ΔV Required (m/s)	ΔV with 50% Margin
1 day (15 orbits)	175.2	262.8
3 days (45 orbits)	57.8	86.7
5 days (75 orbits)	35.0	52.5
6 days (90 orbits)	29.2	43.8
7 days (105 orbits)	25.2	37.8
10 days (150 orbits)	18.0	27.0

Thus, the total ΔV required for a 10-day phasing maneuver and station-keeping is 89.6 m/s. Using the Tsiolkovsky equation and having the assumptions that the S/C's mass is 11.3 kg, out of which 200 g is propellant and an I_{sp} of 1500 s, the S/C's ΔV capabilities are calculated to be >260 m/s. This results for a residual ΔV for collision avoidance maneuvers and deorbiting of >170 m/s.

3.7. Thermal Analysis

In the space environment, the main method of heat transfer is via radiation. When a satellite is in orbit around a planetary body, there are three external heat sources that need to be taken into consideration: the direct solar radiation, the albedo, and the thermal emittance from the celestial body. Additionally, the internal heat dissipation of the satellite has to be accounted for.

3.7.1. Solar Radiation

The solar radiation is the main heating source for an S/C in orbit. Its intensity varies regarding the distance between the S/C and the Sun. For a satellite orbiting the Earth, the solar constant has an average value of 1367 W m^{-2} , with a maximum value of 1414 W m^{-2} and a minimum value of 1322 W m^{-2} , that varies while the Earth orbits the Sun. The above values are recommended by the World Radiation Center in Davos in Switzerland. Another variation of the solar constant is created due to the 11-year solar cycle, but it is in the magnitude of 0.1%, and it is considered negligible [19].

The heat gained (W) due to the direct solar flux is as follows:

$$Q_{\text{solar}} = a S_{\text{solar}} A_{\text{sun}}$$

where

a : the absorptivity coefficient of the spacecraft's surface;

S_{solar} : the solar radiation flux (W m^{-2});

A_{sun} : the area of the spacecraft vertical to the solar rays (m^2).

For this analysis, it is considered that the CubeSat maintains nadir pointing attitude, with one of its 8U sides facing to nadir as illustrated in Figure 12.

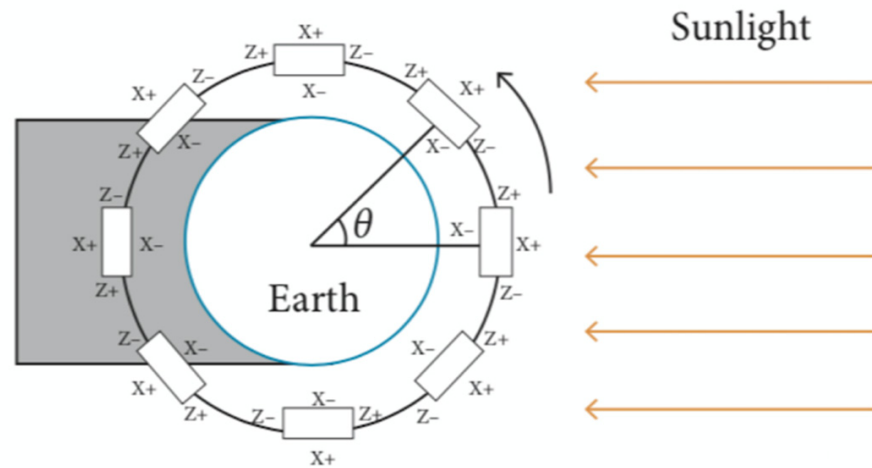


Figure 12. A CubeSat in nadir pointing mode remains oriented towards the center of Earth throughout the orbit [20].

While the CubeSat orbits a celestial body, it rotates in a way that both its 2U sides and its one 8U side (since the other is constantly facing to nadir) are exposed to direct sunlight. If a surface A is inclined at an angle γ to the solar rays, the projected area of the surface vertical to the rays (A_p) is given by the following:

$$A_p = A \cos \gamma$$

While in orbit, if the given surface performs a complete rotation, the angle γ is constantly changing. Since a surface rotates in and out of the Sunview, the average surface area over a full revolution (i.e., 0 to 2π radians) needs to be determined.

$$A_{p,average} = \frac{1}{2\pi} \int_0^{2\pi} A_p d\gamma = \frac{1}{2\pi} \int_0^{2\pi} A \cos \gamma d\gamma$$

By considering the zero radians position being with the surface normal facing away from the solar vector, the integral then goes to zero when the surface normal rotates between $\pi/2$ and $3\pi/2$ (i.e., the active surface is out of Sunview). Thus, the integral limits can be reduced, and the $A_{p,average}$ relation rewritten as [21].

$$A_{p,average} = \frac{A}{2\pi} \int_{3\pi/2}^{\pi/2} \cos \gamma d\gamma = \frac{A}{\pi}$$

The total surface of the CubeSat projected vertically to the solar rays is as follows [21]:

$$A_{p,total} = \frac{1}{\pi} (A_{8U} + A_{2U} + A_{2U}) = \frac{1}{\pi} (A_{8U} + 2 \cdot A_{2U})$$

3.7.2. Albedo

The albedo is the measure of the diffuse reflection of the solar radiation, and it is measured on a scale from 0 to 1. The proportion reflected is not only determined by properties of the surface itself but also by the spectral and angular distribution of solar radiation reaching the Earth's surface. For the Earth, the reflectivity is greater over land areas, and it generally increases as the local height of the Sun decreases and when the cloud coverage increases. Data related to albedo values for orbits with different inclination are available in NASA's data library [22].

The heat gained from a surface of the satellite (W) due to the albedo flux is as follows:

$$Q_{albedo} = aS_{solar}a_{albedo}AF_{1-2}$$

where

a_{albedo} : a dimensionless albedo constant that describes the reflectivity of the celestial body;

F_{1-2} : the view factor from the surface of the celestial body to the spacecraft;

A : the CubeSat's surface area exposed to albedo radiation.

The view factor is the proportion of the radiation that is directed from one surface to another, and it has to do with the relative orientation between the two surfaces. It is considered that the CubeSat is nadir oriented, so it has its sides either vertical or horizontal to the nadir direction (both 2U and 4U sides are horizontally oriented, and one 8U side is vertically oriented since the other 8U side constantly faces the zenith direction). The respective view factors can be considered as follows [23]:

$$F_{ver} = \left(\frac{1}{h^*}\right)^2 \text{ and } F_{hor} = -\frac{\sqrt{(h^*)^2 - 1}}{\pi(h^*)^2} + \frac{1}{\pi} \tan^{-1} \left(\frac{1}{\sqrt{(h^*)^2 - 1}} \right)$$

where $h^* = \frac{R+h}{R}$, R being the radius of the celestial body radius and h being the altitude.

3.7.3. Planetary Infrared Radiation

All objects, including stars and planets, radiate energy to their surroundings. The heat radiated is described by the Stefan–Boltzmann law and depends on the temperature of the body (T) and its surface emissivity (ϵ), which is the effectiveness of the surface material to emit energy as thermal radiation. For satellites orbiting the Earth, there are data related to the orbit's inclination from NASA's data library [22].

The heat gained from a surface of the satellite (W) due to planetary radiation is as follows:

$$Q_{planetary-IR} = aS_{planetary-IR}AF_{1-2}$$

with $S_{planetary-IR}$: the radiation flux of the planetary body ($W\ m^{-2}$).

3.7.4. Internal Heat Dissipation from the Spacecraft

While the S/C is operational, internal heat is generated by its electrical components. In this analysis, the heat dissipation is directly correlated with the S/C's CONOPS (Concept of Operations) and its power consumption, with thermal dissipation considered as a portion of the total power used. In this paper, the two cases considered are that 50% or 100% of the power consumption is converted to heat emissions.

3.7.5. Spacecraft's Thermodynamical Equilibrium Equation

By having knowledge about the thermal loads that apply on the satellite and its thermodynamical equilibrium equation, it is possible to have complete knowledge of its thermal state. The CubeSat is considered a compact object that has a uniform thermal capacity and temperature throughout its whole body. The equation is as follows:

$$Q_{in} = Q_{out} + mc \cdot dT/dt$$

where

Q_{in} : the thermal power absorbed by the satellite (W);

Q_{out} : the thermal power radiated from the satellite (W);

$m \cdot c$: the thermal capacity of the satellite (JK^{-1}), with m being the satellite's mass (kg) and c being the specific heat capacity ($\text{JK}^{-1}\text{kg}^{-1}$).

The thermal power absorbed by the satellite Q_{in} consists of the thermal power from the Sun Q_{solar} , the thermal power due to albedo Q_{albedo} , the thermal power from the celestial body CubeSat that orbits $Q_{planetary-IR}$, and the internal heat dissipation from the CubeSat $Q_{internal}$. While in eclipse, Q_{solar} and Q_{albedo} are considered to be zero.

$$Q_{in} = Q_{solar} + Q_{albedo} + Q_{planetary-IR} + Q_{internal}$$

The thermal power radiated from the satellite Q_{out} is given by the following equation:

$$Q_{out} = \varepsilon \sigma A_{total} T^4$$

where

ε : the dimensionless constant of the surface emissivity of the CubeSat;

σ : the Stefan–Boltzmann constant ($5.67 \times 10^{-8} \text{ Wm}^{-2}\text{K}^{-4}$);

A_{total} : the total surface of the satellite (m^2).

The differential is approached by the derivative's definition. While having defined beforehand the repetition K , the time step Δt , and the corresponding temperatures, the following is assumed:

$$dT/dt = \Delta T/\Delta t = (T[K] - T[K-1])/\Delta t$$

It is also assumed that the S/C has not deployed its solar arrays, that it weighs 11.3 kg, and that its body is homogenous and has the specific heat capacity of aluminum Al 6061-T6.

3.7.6. Emissivity and Absorptivity Coefficients

It is assumed that the satellite's body is made with aluminum alloy Al 6061-T6. It is also assumed that the 60% of the total surface area of the CubeSat is uniformly covered by solar panels. With the material properties of the panels defined as absorptivity coefficient $a_{panel} = 0.9$ and emissivity coefficient $\varepsilon_{panel} = 0.85$ and the aluminum surface properties as $a_{Al} = 0.16$ and $\varepsilon_{Al} = 0.03$, the CubeSat's coefficients are as follows: $a = 0.6$ and $\varepsilon = 0.52$, resulting from a weighted average [19].

3.7.7. Thermal Simulation Results

The simulations are run for a duration of 5 days for the two worst-case scenarios: 50% and 100% electrical power-to-heat conversion ratio. The results of the simulations are shown in Figure 13, and their combined temperature range is 17–36 °C.

Among all components of the CubeSat BUS and the PNT Payload, the CSAC is regarded as the most temperature sensitive. However, as illustrated in Figure 14, its frequency errors remain minimal when operating within the 17 °C to 36 °C temperature range.

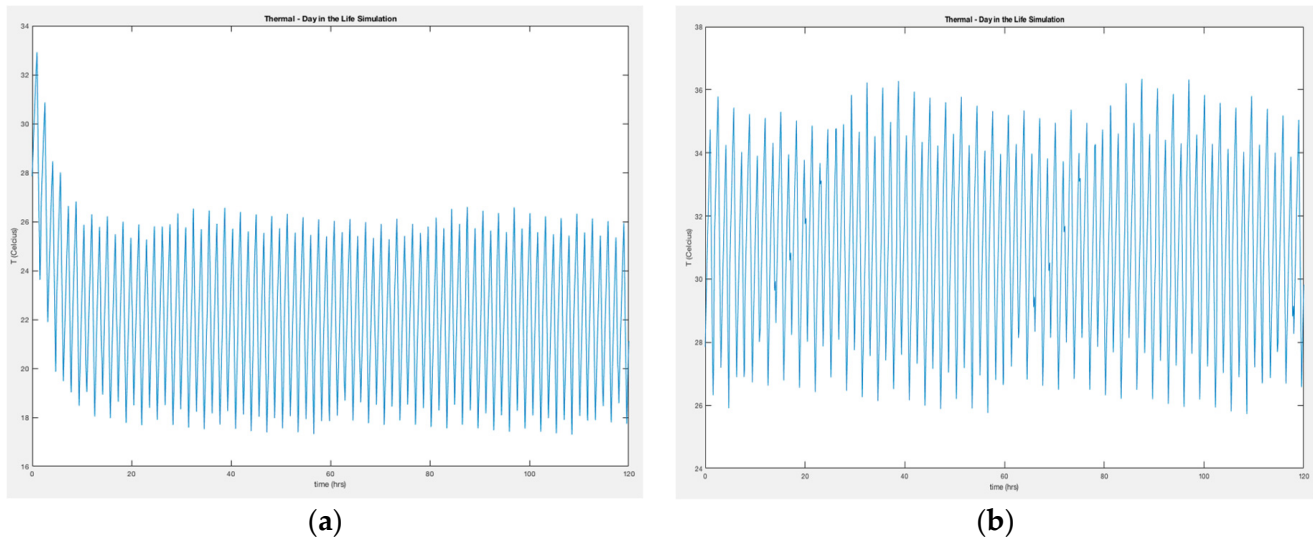


Figure 13. Thermal simulation: 5 days in the life—50% electrical power-to-heat conversion ratio (a) and 100% electrical power-to-heat conversion ratio (b).

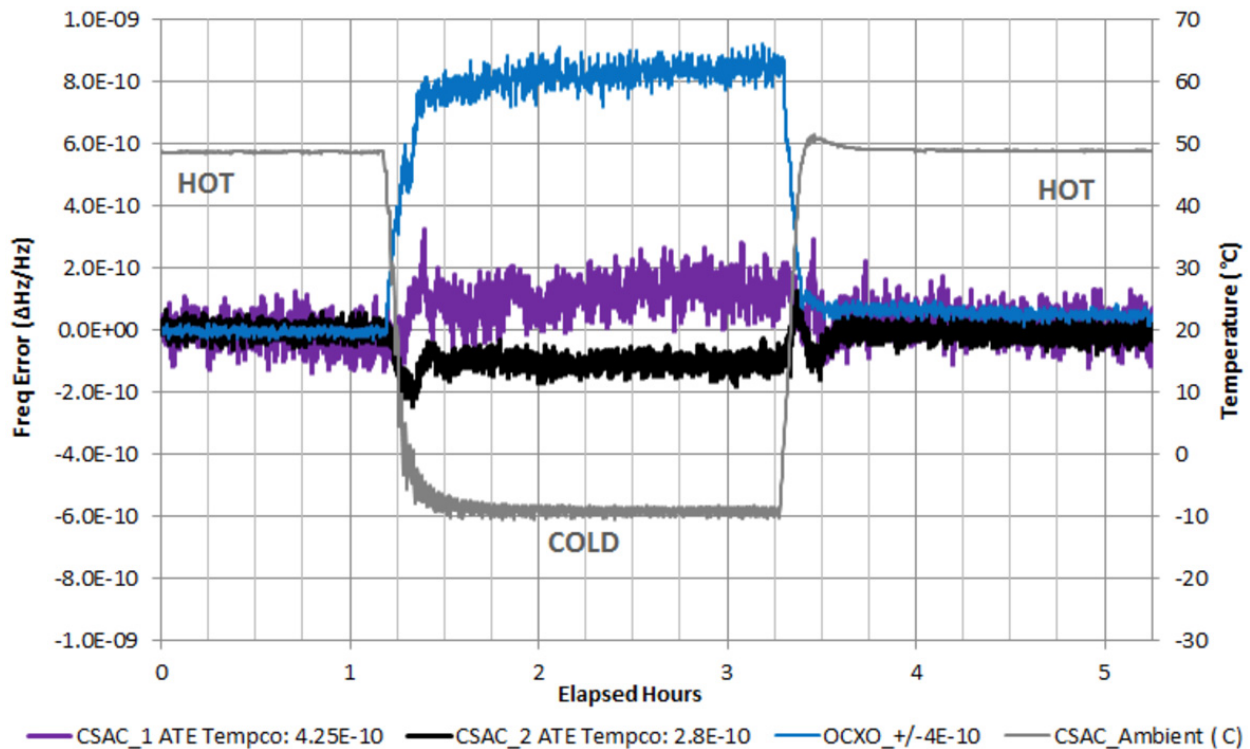


Figure 14. Frequency response of the CSAC while exposed to $-10\text{ }^{\circ}\text{C}$ to $+50\text{ }^{\circ}\text{C}$ [24].

4. Discussion

This study discusses the system engineering aspects of a CubeSat-based LEO PNT system. The constellation design and coverage analysis determined that at least 100 S/Cs, distributed across 10 orbital planes, are necessary to provide continuous coverage over Germany. Additionally, with the ground segment located in Greece and comprising two GCSs in Psachna and Nemea, each spacecraft is expected to achieve over 2.5 contacts per day, which is sufficient for operational procedures. A space debris mitigation analysis was also performed via DRAMA, confirming that the proposed system complies with all relevant regulatory requirements and, thus, maintaining the sustainability of space

activities. Moreover, the CAD model established that the spacecraft design adopts an 8U CubeSat form factor. This stands as one of the earliest examples of utilizing nanosatellites of this size for PNT services that could have a commercial impact.

The system budgets were thoroughly analyzed, confirming the system's validity. The mass budget verified that the S/C design is compatible with deployer pod specifications. The power budget, based on findings from solar eclipse analysis and the defined CONOPS, demonstrated that the S/C's battery SoC remains above 80% during nominal operations, providing a safety margin for handling anomalies and reducing battery degradation over time. The link and ΔV budgets further confirmed the feasibility of the proposed system. Additionally, the radiation analysis performed via SPENVIS confirmed the S/C's survivability in LEO for a minimum of 3 years.

Lastly, the preliminary thermal analysis indicated that the S/C would operate within a temperature range of 17 °C to 36 °C. This temperature range lies within the operational temperature limits of all S/C components and ensures their consistent and reliable performance.

5. Conclusions

In this paper, the feasibility of a CubeSat-based LEO PNT system that could offer services to multiple country-sized areas in the same latitude bounds was assessed. A preliminary design of the system that is based solely on COTS components is presented and accounts for every aspect of such a mission. The design is based on the ERMIS mission of the Greek CubeSat In-Orbit Validation program, and it concluded that it is possible to provide the PNT services mentioned above with 100 8U CubeSats without the need for a ground segment dedicated to timekeeping. Instead, the CubeSats shall operate by relying on the MEO GNSS and relaying the PNT signals on S-Band.

Since such a mission is feasible, a more detailed system analysis is recommended to lead to the CDR (Critical Design Review) of the system. However, in order to proceed, additional information is required for the subsystems and the payload. It is more probable than not that this can only be obtained after an NDA (non-disclosure agreement) with the respective provider.

Overall, the proposed system's design is fully compatible with the CubeSat standard and holds the potential to revolutionize PNT systems in LEO. Rather than replacing GNSS services, it complements and enhances them, transforming them into a multi-layered system-of-systems [3]. This approach lays the groundwork for significant advancements in navigation technologies, such as autonomous vehicle systems, asset tracking, and a wide range of other applications. In EUSPA EO and GNSS Market Report, it was concluded that GNSS cumulative revenues are forecast to reach EUR 4.6 trillion in the next decade with the service revenues accounting for almost 80% in 2033 [1]. The current and the predicted economic growth of the PNT market along with the success story of the New Space and the CubeSats suggest that a business offering commercial PNT services via a CubeSat-based PNT system in LEO is positioned for success in an emerging market.

Author Contributions: Contribution, G.T. and N.A.-M.; methodology, G.T.; software, G.T.; validation, G.T., N.A.-M. and V.L.; formal analysis, G.T.; investigation, G.T.; resources, G.T.; data curation, G.T.; writing—original draft preparation, G.T.; writing—review and editing, G.T.; visualization, G.T.; supervision, N.A.-M. and V.L.; project administration, G.T. All authors have read and agreed to the published version of the manuscript.

Funding: This research received no external funding.

Data Availability Statement: Data are contained within the article.

Conflicts of Interest: The authors declare no conflicts of interest.

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