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TABLE OF CONTENTS

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EXECUTIVE ABSTRACT

The satellite´s mission is to photograph the moon´s permanent shadowed craters in search of water ice. Frequent meetings occurred in order to maintain track of everyone's work so that the weekly deadlines could be met. These meetings were based on a concurrent engineering style and each member presented status updates while the other members would ask questions if further explanation was needed.

A solution based on an 18U CubeSat that will perform a flyby and take photos near the surface of the moon was designed and will be presented through this report. This satellite will start its mission on a GTO orbit increasing the eccentricity of the orbit each time it passes the perigee by applying an impulse to the satellite. This shall be done 40 times until the CubeSat reach the moon, after years. Data will be collected during the flyby and then be transmitted to the ground station on the Earth.

INTRODUCTION

Moonllite is a nanosatellite designed by five students from Instituto Superior Técnico. The satellite dry weight is 6kg (payload included) and its size is 18U standard. The satellite´s mission is to photograph the moon´s permanent shadowed craters in search of water ice.

In this report, it explained the mission analysis carried out by the team and a description of every subsystem included on the satellite. This includes the propulsion, attitude control, electrical, communications, structure and thermal subsystems. A brief description of the launcher selection and satellite operation modes is also included in this report.

The satellite's weight will roughly be around 6kg excluding fuel (1kg for payload, 2kg for solar panels, 2kg for the structure, 0.5kg for general electronics and 0,5kg for the thrusters). Its space distribution is approximately 1U for general electronics, 1U for payload and 16U for fuel.

Luis Ferreira is the Systems Engineer and Team Leader for Moonllite. Luis supervised all other subsystems and provided the main goals for each week.

Francisco Enguita is the Communications Engineer for the team. Francisco Enguita is in charge of all telecommunications between ground station and satellite as well as command and data handling.

Beatriz Alves is the Attitude and Orbit Engineer for the team. Beatriz Alves managed the orbit, mission analysis, propulsion and attitude determination and control of the Moonllite satellite.

João Caldeira is the Electrical Engineer for the team. João Caldeira is in charge of all of the electrical and power subsystems of the satellite.

Ricardo Santana is the Structure Engineer for the team. Ricardo Santana analyzed and studied the best structural material properties for satellite usage.

APPLICABLE AND REFERENCE DOCUMENTS

List of all references used or mentioned in the main text, and may include a list of the acronyms used in the report. A change record table, like the one reported below, should be added in this section.

TABLE 1: Example of a change log record table

Reference	Document
1	http://nanostarproject.eu/wp-content/uploads/2019/02/NANOST-REQ-042-Space-mission-requirements_V2.pdf?fbclid=IwAR0S9HVgKWNqUxuCQKG5NfUToyNmEnXIfiXd-4jGu9_GNAlyNhvWmaT0BX8
2	Gil, Paulo J. S., Elementos de Mecânica Orbital, December 2015
3	https://nanostar-project.gitlab.io/main/source/preliminary-design/propulsion.html
4	https://sst-soa.arc.nasa.gov/04-propulsion?fbclid=IwAR39oHwXUGIsPccMDyfDb5uo8zOAEzkzvw_2ubYRT2Pt4Du6xBywlK2aehw
5	Challenges and Economic Benefits of Green Propellants for Satellite Propulsion, Gotzig, Ulrich
6	EUROSPACE Position Paper: Exemption of propellant related use of hydrazine from REACH authorization requirement.

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MISSION OVERVIEW AND REQUIREMENTS FLOWDOWN

As referred before, the main mission of the satellite is to photograph the moon´s permanent shadowed craters in search of water ice. An 18U CubeSat, named Moonllite, was there for idealized to fulfill the mission requirements. The satellite will perform several revolutions around the earth increasing the apogee distance from the earth each time the satellite passes on the perigee by increasing the velocity of the satellite in this point. Considering the top-level requirements, the mission is design to last for 1,4 years and perform a flyby at an altitude of 95km from the moon. Also, correct attitude control is used to ensure a good pointing of the payload towards the moon surface and to make possible data transmission the Ground segment based on ESA network, using specified bands.

Part of our team study was based on ISTSAT-1, a 1U CubeSat developed by Instituto Superior Técnico students.

Our mission can be divided into three different stages:

1- Launch and orbit insertion - the satellite is placed on a GTO orbit by a rocket;
2- Orbit expansion - the satellite orbit is expanded by applying a series of ΔV each time it passes the perigee;
3- Data acquisition and transmission - all data is collected using the payload and transmitted to the ground station.

Moonllite is composed of the following subsystems:

- Payload;
- Electrical Subsystem - containing all the electrical parts of the satellite, like OBC, COM, Antennas, EPS, sensors and Solar Panels;
- Attitude Control - containing the necessary elements to ensure the attitude control like reaction wheels;
- Propulsion - containing the fuel and the thrusters;
- Structure - aluminium based.

Further subsystem and mission detailed explanation will now be provided through the report.

SUBSYSTEMS ANALYSIS AND DESIGN

In this section is a description of all the subsystems of Moonllite.

MISSION ANALYSIS

Constraints, Requirements and Fundaments

The mission analysis and the definition of the orbits performed by the satellite had 3 main requirements[1]:

1-The altitude of the periselenium pass shall not be higher than 100 km (MR002.);

2-At least one Moon's flyby shall be performed (MR003.);

3-The mission duration from launch to end-of-life shall not exceed 5 years(MR008.).

Appart from the constraints specifically set by the mission requiremnt document, the main obstacles to this mission analysis where:

1-The proximity between the Earth and the Moon, that results in the overlap of the influence region of both planets and, therefore the patched conic approximation results in a lower precision approximation;

2-Nanostar propulsion systems have a very limited impulse capacity, which, not only sets restrictive boundaries to the Delta-V of each impulse (resulting in a higher number of necessary impulses), but also forces an increase in the burn-time, resulting in a lack of precision that will only be accentuated with the number of impulses.

In order to achieve all the stated requirements, as well as to tackle all the difficulties observed, the problem has been approached with regards to the Oberth Effect.

The Oberth Effect states that retrograde burns are most efficient close to the central/main celestial body. Hence, in order to transfer from an orbit with less energy to a more energetic one, the best moment to effectuate the impulsive burn is at the periapsis. This happens because the kinetic energy of the satellite is proportional not to the velocity of a body, but to it's square and, therefore, the same delta v, performed in 2 different points of the orbit, will always result in a bigger variation of kinetic energy, in the one with a bigger initial velocity.[2]

Taking advantage of this effect, most burns will occur in the perigee, resulting in a raise of apogee strategy.

This strategy consists in a series impulsive burns, all occurring in the a previously determined point that will coincide with the perigee of the orbits. This way, although the perigee will remain the same though all the intermediate trajectories, allowing that the most benefit of the Oberth Effect to be taken, the semi-major axis (and, hence, the eccentricity) will suffer sequential increases that will result in the raise of the apogee.

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General description of the orbit

The total trajectory of the satellite can be divided in 3 different orbits:

-Parking orbit: It's the initial orbit where the satellite is inserted and stabilized after launching.

-Transfer Orbit(s): It's the orbit(s) that the satellite describes in order to transfer from its initial parking orbit, to its operational orbit.

-Operational Orbit: This orbit provides the necessary conditions so that the satellite is able to achieve the main purpose of the mission.

Parking Orbit

The parking orbit consists of a circular orbit with the following characteristics:

Semi-major axis (SMA): 6670.9927 Km

Eccentricity (ECC): 0

Inclination (INC): 4.7º

Argument of the Perigee (AOP): 160º

Rise of the ascending apogee (RAAN): 360º

This orbit has a period of 5422.4759218s and at least one full orbit path shall be performed in order to stabilize the satellite after launching and performing detumbling.

Transfer Orbit

In order to achieve the desired operation trajectory, with smaller impulses, the team opted for a set of 39 transfer orbits that coincide in radius of periapsis, inclination and rise of the ascending apogee of the parking orbit. The theoretical values of the characteristics of the transfer orbits are briefly summed up both in table A and in the General Mission Analysis Tool (GMAT) simulation that will be annexed to this document.

Operation Orbit

The operation orbit consists of an elliptical orbit with the same radius of periapsis, inclination and rise of the ascending apogee of both, the parking orbit and the transfer orbits. The satellite shall fly-by the moon with a peri-selenium of 1832.5237km, corresponding to an altitude of 94.524km (as calculated by GMAT), achieving this event at the apogee of the earth focused orbit. The theoretical calculations of the remaining classical elements of the orbit are summarized in the last line of the following table:

V (Km/s)	SMA (km)	rp (km)	ra (km)	E (J/kg)	Impulse (Km/s)	ECC	T (s)
7.729887564	6671	6671	6671	-29.8755809	0.043	0	5422.4759218
7.772887564	6746.26533	6671	6821.53066	-29.5422712	0.044	0.011156592	5514.5026294

7.816887564	6825.50699	6671	6980.01398	-29.1992962	0.045	0.022636705	5611.9474398
7.861887564	6908.97808	6671	7146.95616	-28.8465237	0.046	0.034444758	5715.2067183
7.907887564	6996.95504	6671	7322.91007	-28.4838189	0.0469	0.04658527	5824.7172183
7.954787564	7089.53971	6671	7508.07942	-28.1118392	0.047	0.059036232	5940.7090136
8.001787564	7185.38445	6671	7699.76891	-27.7368596	0.048	0.071587603	6061.5855106
8.049787564	7286.58802	6671	7902.17604	-27.3516218	0.05	0.084482342	6190.0982809
8.099787564	7395.75736	6671	8120.51472	-26.9478825	0.051	0.097996368	6329.7302560
8.150787564	7511.26139	6671	8351.52279	-26.5334928	0.052	0.111866882	6478.5905740
8.202787564	7633.58783	6671	8596.17567	-26.1082998	0.0535	0.126099006	6637.4961444
8.256287564	7764.52579	6671	8858.05158	-25.6680196	0.054	0.140836134	6809.0045830
8.310287564	7902.23213	6671	9133.46425	-25.220722	0.056	0.155808145	6990.9451610
8.366287564	8051.29627	6671	9431.59254	-24.7537779	0.057	0.171437769	7189.6861584
8.423287564	8209.99966	6671	9748.99933	-24.2752751	0.059	0.187454291	7403.3099162
8.482287564	8382.20479	6671	10093.4096	-23.7765606	0.06	0.204147337	7637.4542498
8.542287564	8566.21307	6671	10461.4261	-23.2658233	0.062	0.221242813	7890.3184275
8.604287564	8766.49731	6671	10861.9946	-22.7342795	0.064	0.239034729	8168.6516606
8.668287564	8984.94182	6671	11298.8836	-22.1815571	0.066	0.257535537	8475.8668389
8.734287564	9223.74647	6671	11776.4929	-21.6072721	0.068	0.276758092	8816.0137141
8.802287564	9485.4947	6671	12299.9894	-21.0110286	0.07	0.296715647	9193.9301397
8.872287564	9773.24002	6671	12875.48	-20.3924184	0.072	0.31742186	9615.4379214
8.944287564	10090.6172	6671	13510.2344	-19.7510217	0.074	0.338890788	10087.5992101
9.018287564	10441.9866	6671	14212.9732	-19.0864065	0.077	0.361136892	10619.0561491
9.095287564	10837.9811	6671	15004.9621	-18.3890338	0.08	0.384479455	11228.8117846
9.175287564	11286.5342	6671	15902.0684	-17.6582108	0.083	0.408941674	11933.0682227
9.258287564	11797.6342	6671	16924.2684	-16.8932174	0.086	0.434547649	12752.7431057
9.344287564	12384.0304	6671	18097.0609	-16.0933067	0.089	0.461322383	13715.2660667
9.433287564	13062.2531	6671	19453.5063	-15.2577046	0.093	0.489291784	14857.2477391
9.526287564	13863.3021	6671	21055.6041	-14.3760844	0.097	0.518801511	16244.6833383
9.623287564	14820.786	6671	22970.572	-13.44733	0.101	0.54988892	17956.3505980
9.724287564	15982.0021	6671	25293.0042	-12.4702774	0.106	0.582592972	20107.4962309
9.830287564	17430.6459	6671	28190.2918	-11.4338849	0.111	0.617283259	22902.4235625
9.941287564	19281.0714	6671	31891.1428	-10.3365625	0.117	0.654013002	26644.5239321
10.05828756	21742.0062	6671	36813.0125	-9.16658738	0.123	0.693174589	31905.1392807
10.18128756	25158.2536	6671	43645.5073	-7.92185351	0.131	0.734838511	39712.9325383
10.31228756	30290.9435	6671	53910.887	-6.57952434	0.137	0.779769158	52466.2980512
10.44928756	38643.8289	6671	70616.6578	-5.15735645	0.1469	0.827372178	75601.6800372
10.59618756	55183.8132	6671	103696.626	-3.6115663	0.1569	0.879113102	129011.6059816
10.75308756	102906.174	6671	199141.349	-1.93671567	0.085	0.935173958	328528.7423236
10.83808756	195566.494	6671	384461.989	-1.01909072		0.96588884	860702.1690136

Table A: Theoretical calculations of the impulsive burns necessary and of the classical Keplerian orbit parameters of all the orbits.

Note: The grey cells correspond to values of that add to suffer a slight adaptation when simulated in Gmat. This deviations can be caused by the moon's gravitational field, approximation and finite precision errors that accumulate due to the high number of iterations and solar pression radiation.

GMAT Simulation

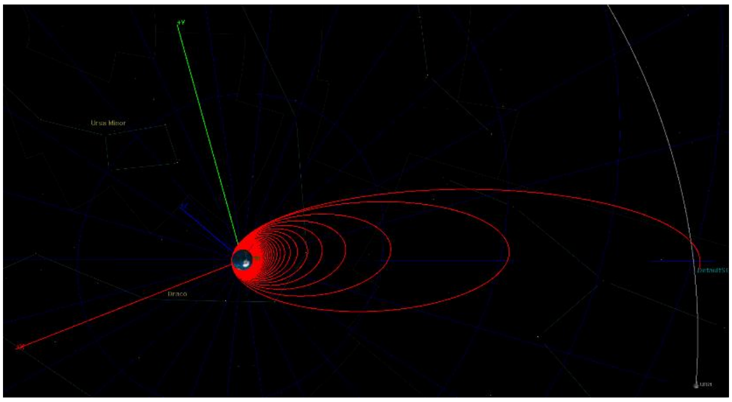

Figure 1: Orbit View

A simulation with the General Mission Analysis Tool (annexed to this document), allowed the validation of the theoretical calculations, while accounting for perturbations such as solar pressure and moon's gravitational field. This analysis also allowed some results to be produced such as an eclipse window report, an orbit view (figure 1) and its projection on Earth's surface, etc. Some relevant results will be discussed in the present section.

Although it can be verified by analysis of the annexed code, apart from the already stated Keplerian classical orbit parameters, this orbit was simulated for an initial impulse at 180º true anomaly point, at 01 Jan 2020 10:17:20.00 UTCGreg. Depending on the number of parking orbits described and on the point of insertion in orbit, the total duration of the mission may vary, however, assuming a single parking orbit is described, this corresponds to a launching at 1 Jan 2020 08:46:58.00.

The following trajectories can also be analyzed with a diagram of the altitude of the satellite (relative to Earth) during the performance of the transfer orbits, with time. The graphic C shows that even though the smallest attitude stays constant with time (since periapsis remains the same), whereas the maximum altitude keeps rising.

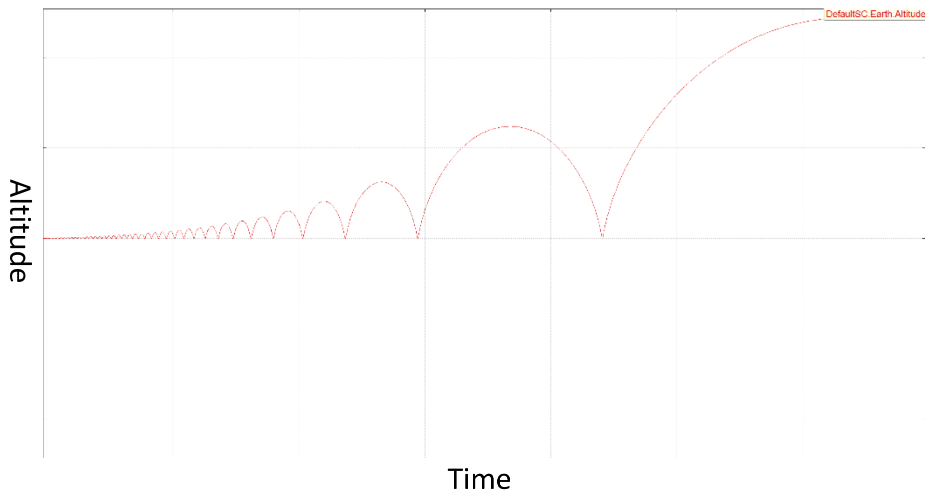

Figure 2: Altitude of the orbit in function of time

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Total duration of mission and Windows of Opportunity

The total duration of the mission depends on the number of parking orbits described. The following table gives examples of mission durations, launching times and true anomaly at insertion, for different number of parking orbits described, that are coherent with the simulated orbit:

Number of parking orbits described	Mission Duration (s)	Epoch of insertion (TDB Gregorian)	True anomaly of insertion (º)
1	1430544.431	1 Jan 2020 08:46:58.00	180
2	1435966.907	1 Jan 2020 07:16:36.00	180
3	1441389.383	1 Jan 2020 05:46:14.00	180
4	1446811.859	1 Jan 2020 04:15:52.00	180
4.5	1449523.097	1 Jan 2020 03:30:41.00	0

Table B: Examples of variations of the proposed orbit

In the event of rescheduling of the launching date (due to climacteric conditions, delays in development etc.), the same orbits shall be applicable for a launching date 29.530 days later (since this coincides with the moon's synodic period with respect to Earth). Nevertheless, through manipulation of the insertion point in the parking orbit, number of parking orbits described and true anomaly of the first impulse, this orbit can be easily adapted to a new launch date (despite possible repercussions in duration of mission, necessary thrust per impulse and delta-V budget).

LAUNCHER

Various launchers could be used to perform the initial orbit insertion needed for our mission. However, we can select a few examples listed on the table below.

Launcher	Availability (higher the better)	Injection Accuracy (perigee, km)	Max acceleration (g)		Fundamental Frequency (Hz)		Cost (FY00$K/kg)
			Lateral	Axial	Lateral	Axial	
Delta	0.68	0.25	3	6	15	35	9.8 – 10.8
Atlas Centaur	-0.11	1.7	1.2	5.5	10	15	11.6 – 12.7
Ariane 4	0.32	0.91	1.4	5.7	10	18	9.9 – 12.5

Table C: Examples of Launchers.

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All the launchers listed have large payload accommodations capable of transporting our satellite. The decision will then rely on cost and the availability of the launcher at the launch time.

SYSTEMS OPERATIONS MODES

For our mission, we can consider three different operation modes:

- Orbit mode - This mode is used from the initial GTO orbit until the satellite reaches the moon. The payload is off but all the other subsystems are used in this mode so it is possible to have a correct pointing of the antenna towards the ground station, a correct pointing when using the thrusters and a status update of the satellite.
- Flyby mode - This mode is used while the flyby is being performed. Correct attitude control is needed and the satellite payload must be active so the mission goal can be achieved.
- Safe mode - Not yet completely defined, the safe mode is used when something not expected occurs during the mission and while the satellite tries to return to its normal operation mode. On this preliminary design, the safe mode is similar to the orbit mode and will be better defined later on the mission project.

SPACE PROPULSION SUBSYSTEM

Propulsion of choice

Monopropellant liquid propulsion has been chosen for this mission, namely, hydrazine.

Hydrazine has a specific impulse of up to 240[3] and a thrust of up to 30.7N[4] (This system is still TRL-Technology readiness level- 9, this is, it is flight proven through successful mission operations with documented mission operation results on small spacecrafts and smallsats such as microsatellites). These characteristics allow for relatively small periods of burn, maximum 12 seconds, which correspond to a maximum deviation of less than 1º from the impulsive burn selected point.

Advantages and Disadvantages of the propulsion system

This propulsion system has been chosen, due to its ability to perform a higher thrust (and therefore, reducing the burn time and the error associated with the fact that the impulsive burns aren't perfectly instantaneous), a high specific impulse and to its ability to perform orbit insertion in both periapsis and apoapsis, orbit maintenance and maneuvering and attitude control. Hydrazine is also a common choice for both large and satellite which results in a "large and successful heritage and great variety of space qualified, off-the-shelf components"[5].

Nevertheless, the utilization of hydrazine entails some disadvantages and dangers that shall not be overlooked. The main problem with this propellent is its toxicity. Hydrazine is a carcinogenic and mutagenic chemical and, therefore special precautions must be taken "during all ground operational phases when Hydrazine or its derivatives are used"[5]. Since this propellent has been identified as a

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substance of high concern by REACh regulations (2011)[6], these systems may be forbidden in the future and thorough research as been conducted in order to develop a possible green replacement for hydrazine. Despite all efforts to create such substitute, hydrazine is still one of the propellants with the highest performance available in the market.

Propellent budget

It has been calculated that approximately 16.46kg of hydrazine are required in order to perform all the impulsive burns in table A (which corresponds to 16.43L). Due to the choice of reaction wheels as actuators chosen for the attitude determination and control system (ADCS), and to the fact that no external applicators of torque are required by these actuators, no extra propellant is necessary to perform the correct control of attitude. The orbits utilized in these mission (elliptical orbits) are highly stable, and most corrections can be effectuated by correcting individual impulses, without increasing the total delta-V budget. Nevertheless, a margin shall be applied to possible orbit corrections, therefore, it will be necessary around 16.6kg of hydrazine.

ATTITUDE, DETERMINATION AND CONTROL SUBSYSTEM

Perturbations and Necessary Torque

Due to the nature of this mission, the main perturbations that will affect the satellite correspond to the gravity gradient torque, the solar radiation pressure torque and the magnetic torque. Each of these perturbations will be further explored in the subsequent subsections.

Gravity-Gradient Torque

Due to the gravitational field

of other celestial bodies (mainly the Earth, Moon and Sun), satellites tend to align their longitudinal axis with the nadir-zenith direction which results in a torque that alters the satellite orientation. Assuming that the axis with the minimum moment of inertia is the longitudinal axis, this phenomenon can be mathematically modeled by the following equation:

$$T_g = \frac{3\mu}{r_p^3}|I_{\max} - I_{\min}|\sin(2\theta)$$

To estimate the maximum gravity-gradient torque that the satellite would be submitted to, a set of approximations were taken in to consideration:

-The satellite has been approximated to a cylinder with 340.5mm of height and 300mm of radius;

-The maximum angular deviation (θ) between the longitudinal axis and the nadir-zenith direction has been considered 45º in order to major da synodic perturbation.

-It has only been considered the Earth magnetic field, since, during most mission, the satellite will stay inside Earth’s influence region (and outside the Moon’s influence region), and since the standard gravitational parameter of the Earth is much bigger than the Moon’s.

These approximations result in a 1.14e-6Nm maximum torque due to the gravitational field of the Earth.

Solar Radiation Pressure Torque

In asymmetric satellites, the incident electromagnetic radiation from the sun is responsible for a pressure and shear stress over the intercepted surface. This phenomenon can be described by the following equation:

$$T_{sp} = \frac{\Phi}{c} A_{sp}(1+q)\cos(i)||\mathbf{c}_{sp} - \mathbf{c}_g||$$

However, since the satellite is roughly symmetric (with two symmetry planes that intersect along the longitudinal axis), this perturbation is almost insignificant and has been approximated by a security factor of 1e-5Nm.

Magnetic Torque

This perturbation is the result of the interaction between the central's body magnetic field and the residual magnetic dipole of the vehicle (that can be minimized during the production, but can't be completely annulled). This torque depends mainly on the orbit attitude and inclination (since orbits with an inclination closer to 90º and, hence, closer to the poles, will experience a higher magnetic field) and the residual magnetic dipole. Since the moon's magnetic field is insignificant, the magnetic field experienced by the satellite (magnetic field of the Earth), can be modelled as a dipole:

$$T_m \approx D_m \frac{\lambda M_m}{r_p^3}$$

Approximating the unitless function of the magnetic latitude to 1 (due to the small inclination of the orbits) and assuming a value of the residual magnetic dipole on the higher spectrum of the normal scale, it is estimated that the magnetic torque is 5.363e-5Nm.

In conclusion, the total torque expected due to perturbations is 5.474e-4Nm.

Chosen Sensors

In order to ensure attitude determination, the team has opted for the utilization of sun sensors that can detect if the sun is in its field of view or not, as well as gyroscopes (that not only enhance the precision of the determination of attitude, but also ensure this measurement, while the satellite is in an eclipse window, since gyroscopes are independent of the environment, even though, they tend to drift and therefore are not very precise on their one), that measure the angular rate, that can be integrated to obtain the evolution of the attitude.

Sun Sensors

The chosen sun sensor is the NewSpace Systems NCSS-SA05.

	NCSS-SA05
FUNCTIONAL CHARACTERISTICS	
Field of view	114°
Update rate	>10 Hz (limited by customer ADC)
Accuracy	<0.5° RMS error over FOV
PHYSICAL CHARACTERISTICS	
Dimensions	33 mm x 11 mm x 6 mm
Mass	<5 g
Power	<10 mA @ 5 V
ENVIRONMENTAL CHARACTERISTICS	
Thermal (operational)	-25 °C to +70 °C
Vibration (qualification)	20 g_{RMS} (random)
Radiation (TID)	n.a.
INTERFACES	
Power supply	5 V_{DC}
Data	5 analogue channels
Connector	9-way female Nano-D
Mechanical	3 x M2 threaded holes

Figure 3: NewSpace Systems NCSS-SA05 and its performance parameters.

Gyroscopes

The chosen gyroscope is NewSpace's Stellar Gyro NSGY-001:

	NSGY-001
FUNCTIONAL CHARACTERISTICS	
Rate estimation accuracy [3σ]	≤0.20 degrees/s (boresight) ≤0.05 degrees/s (cross-boresight)
Maximum slew rate	≥1.00 degrees/s
Detection capability	Mv ≥5.0
Maximum number of features tracked	15
Standard update rate	>1 Hz
Sky coverage	>99%
PHYSICAL CHARACTERISTICS	
Dimensions	37.0 mm x 35.5 mm x 49.0 mm
Mass	<100 g
ENVIRONMENTAL CHARACTERISTICS	
Thermal (operational)	-25 °C to +50 °C
Vibration (qualification)	14 g_{rms} (random)
INTERFACES	
Power supply	5 V_{DC}
Power Consumption	<200 mW (average)
Communication	SPI
Connector	nano-D (P15)
Mechanical	Front: 3 x M3 (w/ alignment slots) Top: 2 x M3 (w/ alignment slots)

Figure 4: NewSpace's Stellar Gyro NSGY-001 and its performance parameters.

Chosen Actuators

As actuators, we've opted for a reaction wheel system along the 3 orthogonal axis, since this an high accuracy approach, with almost no constraints when it comes to pointing options and attitude maneuverability. Reaction wheels typically don't require any external actuator (therefore no propellent is necessary for their correct function), however this system is inhibited by a limited lifetime (that corresponds to the lifetime of the wheels and the sensors). For a short period, mission like this one, the lifetime of the components shall not be a problem.

In order to ensure the control along the three axis, 3 MAI-400 Reaction Wheels will be necessary, placed along each axis.

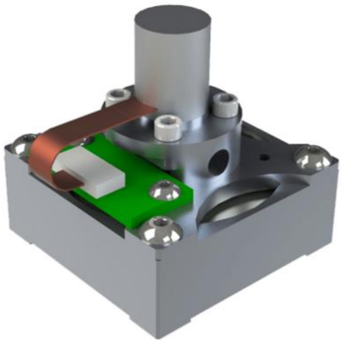

Figure 5: MAI-400 Reaction Wheel

This actuator is characterized by:

-Momentum storage: 9.35 mNms @ 10,000 RPM

-Maximum torque: 0.635 mNm

-Rotor balance: <40 mg-mm

-Power

-5 VDC required

-Peak current: 440 mA (maximum acceleration)

-Steady state current: 170 mA (500 RPM)

-Idle: 90 mA

-Weight: 90 g

-Dimensions: 3.3 cm x 3.3 cm x 3.84 cm

-Serial interface: UART, I2C

-Operating temperature: -40 to +85 degC

Functioning of the ADCS

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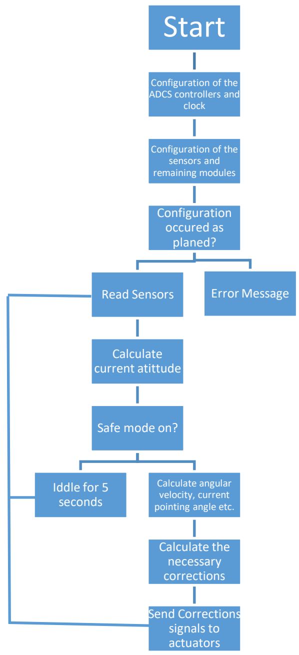
The behavior of the proposed ADCS can be summarized by the following flux chart:



COMMUNICATIONS SUBSYSTEM AND GROUND SEGMENT

Ground Segment

We have decided to use a combination of three ground for the satellite's communication, because of the significant cost and added complexness associated with an additional satellite acting as the middleman for the telecommunications. These ground stations are located in Madrid (Spain), Canberra (Australia) and Goldstone (U.S.A.).

These locations where chosen because of their geographical position, which allows us to have a constant data transmission flow. In addition to what was previously mentioned, all of the ground stations are part of the DSN (Deep Space Network) which means we have access to the 34-meter antennae, whose size and power will allow the mission to be successfully fulfilled.

The 34-meter antennae have 53.6dB of gain, will be enough to transmit the uplink data as well as receiving data such as telemetry, health status and payload, coming from the satellite. We felt this set of antennae struck a good balance between being too small, leading us to compensate with a bigger satellite antenna, which would result in a satellite size increment or being too "overkill" in power and size (ex. The 70-meter DSS14) for our mission.

As it will be mentioned again in this report, our main model for the satellite was the LUNAR RECONAISSENCE ORBITER. For communication the LRO uses a 34-meter antenna for active telecommunication.

Data Rate Selection

Due to the simplicity of our satellite command data and health status telemetry, values fall under the usual ones, totaling at 12000bps not including payload data transmission. The S-band allows have medium data rate at 100000bps/downlink which is more than enough to transmit the 10Mb data from payload. We are experiencing an intentional bottleneck from the satellite's antenna in order to keep the size as small as possible. The Uplink will be significantly higher due to higher power output from the ground station's transmitter.

Frequency Band Selection

The S-Band was used because it strikes a good balance between data rate and added power consumption due to the increased Space Loss. UHF-band has the lowest Space Loss out of the available frequencies and the transmitter had excellent power efficiency. The main issue with UHF was the bad data rate which served insufficient for mission completion. Ku band has excellent data rate transmission but due to the added Space Loss of using higher frequencies, both the downlink transmitter and antennae would have had to be more powerful, which would have increased satellite size for both the equipment and the allocation of bigger solar panels.

The LUNAR RECONNAISSANCE ORBITER served as a great starting point for the communications point of view, due to its same frequency and similar orbit. Unlike the LUNAR RECONNAISSANCE ORBITER, we opted to sacrifice slower data rate for a smaller footprint of the satellite (further explanation will be given in the antennae and OBC segment).

RF Link Budget

The Moonllite satellite will only start transmitting payload data when it has direct path to the ground station. The expected $L_{TM-ant,SAT}$ is 1.5dB.

The antenna half power beam width is 71deg and its maximum gain is 8.3 dBi and will fit the satellite without any issues, due to its PCB like format.

The space loss is measured by:

$$L_S = 20\log(C) - 20\log(4\pi) - 20\log(S) - 20\log(f) = -211.4\text{dB}.$$

To calculate downlink and uplink respectively E_b/N_0 we will need:

	$EIRP = G_{TM,SAT} - L_{TM-ant,SAT} + G_{ant,SAT} = 7.16dBW$ $L_{Tot} = L_S + L_a + L_\theta = 211.4 + 2 + 1.5 = 214.9dBW$ $T_{sys,GS} = L_T + L_{filter} + 10\log(Effective\ noise\ temperature) = 2 + 0.7 + 28.87 = 31.57K$

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DOWNLINK	$\frac{G}{T} = G_{ant,GS} - T_{sys,GS} = 53.6 - 31.57 = 22dB/K$ $\frac{C}{N_0} = EIRP + \frac{G}{T} - k - L_{Tot} = 7.16 + 22 + 228.6 - 214.9 = 43.46dBHz$

UPLINK	$EIRP = 69.3dBW$ $L_{Tot} = 211{,}5 + 2 + 1.5 = 214.9dBW$ $T_{sys,SAT} = L_T + L_{filter} + 10\log(Effective\ noise\ temperature) = 1 + 0.9 + 25.05 = 26.95K$ $\frac{G}{T} = G_{ant,SAT} - T_{sys,SAT} = 8.3 - 26.95 = -18.65dB/K$ $\frac{C}{N_0} = EIRP + \frac{G}{T} - k - L_{Tot} = 69.3 - 18.65 + 228.6 - 214.9 = 64.35dBHz$ $\frac{E_b}{N_0} = \frac{C}{N_0} - R = 9.6dB$

Due to S-Band frequency being 2.2GHz the L_a is 0.5dB, plus 0.3dB loss to account for polarization mismatch for large ground antennae.

Command and Data Handling

We chose the Stamp9G20 (OBC) used in the MCubed-2/COVE due to its low power consumption, plenty of storage (128MB) for the mission payload, considering each picture is 10MB. The MCubed-2/COVE´s mission close resembles our satellite´s mission, making it an easy fit with minimal modification, for our Data Handling segment.

All the other options have a considerable increment in power consumption and some in size, something that is not recommended/needed for the assigned mission as our main goal is to have the satellite as small as possible.

At 0.12W average power our solar panel array has more than enough feeding capacity to provide constant power to our C&DH system.

A couple of sun sensors and gyroscopes were added to provide helpful pitch and position information for better usage of the attitude control system.

Equipment

Quantity	Equipment Name
1	Endurosat Custom S-Band Antenna ISM
1	Endurosat S-Band Receiver
1	IQ Wireless HISPICO transmitter
1	Stamp9G20 (128Mb model)

Table D: Equipment components necessary.

ELECTRIC POWER SUBSYSTEM

Eletrical power system (EPS) is a fundamental system when talking about designing a satellite. This subsystem is responsible of generating, storing and distributing of power necessary in the satellite.

Typically, EPS is composed by a power source, to collect energy when the satellite is in direct sunlight. Photovoltaic solar cells are the most common power source in most missions.

One other fundamental part of EPS is energy storage, that typically occurs in a battery. Energy produced by solar panels with sunlight, is used to charge the battery, which is needed during the time when satellite is in eclipse.
Once the energy is collected it must be provided to each subsystem, according to their power needs, or used to charge the baterries.

In order to size each of these components, a detailed power budget was identified. In table E is shown the power budget necessary in our satellite, with detailed consumed energy by each subsystem.

Subsystem	Consumption (+10% margin)
Payload	5 W (5.5 W)
Propulsion	0 W
Attitude Control	6.85 W (7.535 W)
Communications	7 W (7.7 W)
Data Handling	0.12 W (0.22 W)
Thermal	0 W
Power	5.6 W (6.16 W)
Structure	0 W

Table E: Power budget.

The power consumed by the EPS was calculated by the following way:

$$Power\ consumption = \frac{Consumption\ of\ all\ other\ subsystems\ \times\ T_e}{T_d}$$

Where T_e is the length of the period in eclipse per orbit and T_d is the length of the period in sunlight per orbit.

$$Power\ consumption = \frac{18.97\ \times\ 2175}{7440} \sim 5.6\ W$$

For our mission, we consider three different operation modes:

Operation Mode	Consumption (+10% margin)
Orbit Mode	19.57 W (21.615 W)
Flyby Mode	24.57 W (27.115 W)
Safe Mode	19.57 W (21.615 W)

Table F: Power budget by mode.

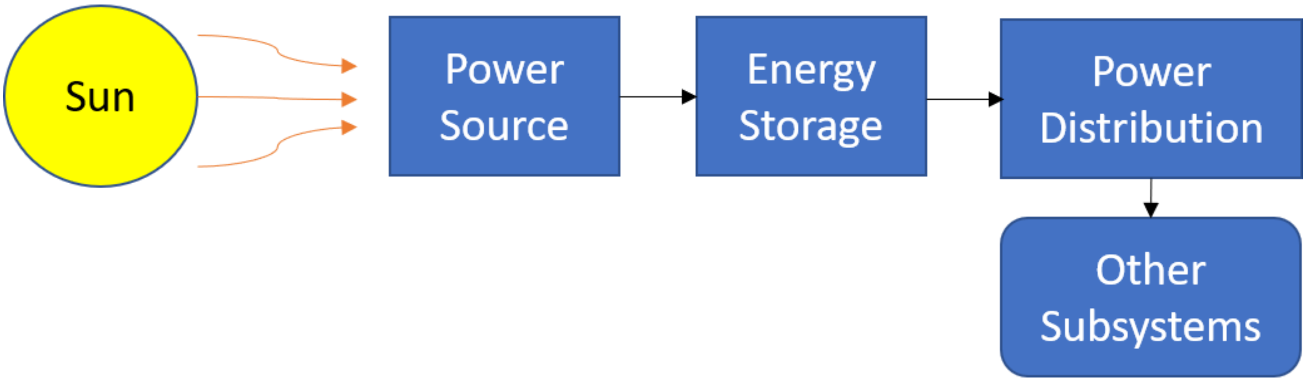

Figure 6: Architecture of Power System.

POWER SOURCE

Once the power budget is known, the next step is size the power source that will be used.

As told above solar power generation is the predominant method of power generation on small spacecraft, and therefore it was our obvious choice.

To size our photovoltaic solar cells is necessary to meet power requirements at the BOL (beginning-of-life) and at the EOL (end-of-life). Two fundamental parameters are mission life and average power requirement. The first one is already known, and his value is 2.7119 years. The second one is taken from power budget in table E.

Now we can determine the total power that the solar array must provide during daylight (P_{sa}):

$$P_{sa} = \frac{\frac{P_e \times T_e}{X_e} + \frac{P_d \times T_d}{X_d}}{T_d}$$

Where P_e and P_d are spacecraft's power requirements (excluding regulation and battery charging losses) during eclipse and daylight, respectively, $T_e$and T_d are the lenghts of these periods per orbit and, X_e and X_d are the efficiency of the paths from the battery to the individual loads and the path directly from the arrays to the loads, respectively.

Besides the +10% margin in power consumption in each subsystem, was considered a +15% over all the satellite power.
Considering $P_e = 24.1$ W and $P_d = 31.18\ W$, $T_e = 2175\ s$ and $T_d = 7440\ s$.

For the power distribution was chosen Direct Energy Transfer, that tipically use shunt resistors to maintain bus voltage at a predetermined level. This type of power regulation is normally better instead of Peak Power Tracking in systems with less than 100 W.

Based on that, for Direct Energy Transfer, $X_e = 0.65$ e $X_d = 0.85$.

$$P_{sa} = \frac{\frac{24.1 \times 2175}{0.65} + \frac{31.18 \times 7440}{0.85}}{7440} \sim 47.53\ W$$

There are many types of cells for solar panels (Silicon, Gallium Arsenide, Indium Phosphide, Multijunction), differentiated by efficiency, degradation coefficients and prices. For our satellite we have chosen silicon cells, despite of the low effiency, but with the advantage of the low cost. For example Gallium Arsenide and Indium Phosphide cost about 3 times more. With a little more research we discover that triple junction cells have a high effiency-cost ratio compared to other cells and could be a better choice. A higher effiency would mean lower solar array area.

So, since the chosen cells were silicon ones, their theoretical efficiency is 20.8% and the achieved one is 14.8%.
Another important parameter to calculate the power production capability of the manufactured solar array is inherent degradation (I_d), that is associated to the assembled array being a group of cells and

not just one, shadowing and temperature variations. This value vary between 0.49 and 0.88, with the considered value being 0.77.

$$P_{BOL} = P_{SUN}\,\eta\,I_d \cos\theta$$

Where solar illumination intensity $(P_{SUN}) = 1367\ W/m^2$, and where $\cos\theta$, cosine loss, with angle θ being the angle between the vector normal to the surface of the array and the Sun line. The Sun incidence angle used is the worst-case, in our case we used $\theta = 45°$.

$$P_{BOL} = 1367 \times 0.148 \times 0.77 \times \cos 45° = 110.2\ W/m^2$$

From this, to calculate the power production in the EOL, we must consider factors that degrade the solar arrays's performance, L_d.

$$L_d = (1-c)^{t_f}$$

Where c is the degradation per year and t_f is the satellite lifetime.

The degradation per year, c, for a silicon solar array is about 3.75% per year, and lifetime is already known, and has the value of 2.7119 years.

$$L_d = (1-0.0375)^{2.7119} \sim 0.902$$

That means,

$$P_{EOL} = P_{BOL} \times L_d = 110.2\ \times 0.902 = 99.4004\ W$$

With all the information adquired we can now size the solar array area, A_{sa} required to support spacecraft power requirements,

$$A_{sa} = \frac{P_{sa}}{P_{EOL}} = \frac{47.53}{99.4004} \sim 0.48\ m^2$$

This solar array sizing is highly affected by angle θ, since this angle is constatly changing. Also, the distance between satellite and sun is always chaging, which affects P_{SUN}.

ENERGY STORAGE

Our spacecraft besides the power source requires a system to store energy for peak-power demands and eclipse periods.

In most missions that occurs in batteries, and our is not an exception. There are two types of batteries: primary and secondary batteries. First ones are not rechargeable, and only used in missions with short durations. Second ones are rechargeable and will be the ones we will use.

These components represent the heaviest parts of any spacecraft, but that can be minimized by using lithium ion batteries.

MECHANICAL DESIGN AND STRUCTURE

Starting with a simplified analysis, we have considered our Cubesat structure as a hollow cylinder with uniform thickness. Thus, we can easily locate the center of mass at mid-length of the cylinder and know his momentum of inertia. With this approach we started our preliminary structure design that would be useful in the choice of structure material and the kind of launcher we would choose. Our cylinder has a radius that allows it to be tangent to the vertices of our 18U.

Although, to the mass control we have used the real structure. So we have:

$$m = \frac{[0.2 \cdot 2 \cdot (30 \cdot 20 + 30 \cdot 34.05 + 20 \cdot 34.05)] \cdot 2.7}{1000} = 2.11896kg$$

considering the final thickness as 0.2cm and the density of aluminum 2.7g/cm^3.

The most critical moment through the mission at structural levels is the launching. With exception to the thermic expansion in space, this is the only moment in which the structure is under external efforts. Because of this, our structural analysis is focused on this period of the mission.

To start we have computed de lateral and axial forces applied to the satellite in the launching process.

$$P_{ax} = mgLF_{ax}; \quad P_{lat} = mgLF_{lat}$$

Knowing that the lateral acceleration will produce a momentum on the base of the structure and that this moment will increase the axial equivalent force, we can compute that as the maximum equivalent axial force which acts on the structure. Making the quotient between this value and the area of the surface we determined the maximum amount of stress that the structure material will need to support.

$$P_{eq} = P_{axial} + \frac{2M}{r} = P_{axial} + \frac{2P_{lat}L}{2r} \qquad \frac{P_{eq}f_u}{A} < \sigma_u; \quad \frac{P_{eq}f_y}{A} < \sigma_y,$$

There we obtain that our equivalent weight equals to 1716.567N. To obtain the maximum amount of stress that our structure may have to support, we do the quotient between the equivalent weight and the area of a horizontal section in the cylinder. This value equals to 0.76MPa.

From this calculation we can multiply the last value to 1.1 to compare it to the yield tensile strength of different materials and to 1.25 factor to the ultimate tensile strength.

By comparing our values with the Aluminum yield and ultimate strength conditions, we understood that this material would easily support the amount of effort we need to the launching process.

Furthermore, aluminum has high strength related to his weight which is positive because the more weight our satellite has, the most expensive will be the launching. The ductility and lower density of this material, when compared with the other materials between we must choose, are also positive points because this way it could endure more effort until the fracture and reducing the total mass of de Cubesat. Finally, aluminum is easy to machine, what would facilitate the satellite construction process.

To finish this preliminary analysis, we calculated the structure natural axial and lateral frequencies. The respective launcher frequencies must be under the computed frequencies.

$$f_{ax} = \frac{1}{2\pi}\sqrt{\frac{EA}{mL}} = \frac{1}{2\pi}\sqrt{\frac{2E\pi rt}{mL}},\ f_{lat} = \frac{1}{2\pi}\sqrt{\frac{3EI}{mL}} = \frac{1}{2\pi}\sqrt{\frac{3E\pi r^3 t}{mL}}.$$

So we have that the axial frequency equals to 2340 Hz and the lateral frequency equals to 516 Hz.

THERMAL CONTROL SUBSYSTEM

	Typical Temperature Ranges (°C)	
Component	**Operational**	**Survival**
Batteries	0 to 15	-10 to 25
Power Box Baseplates	-10 to 50	-20 to 60
Reaction Wheels	-10 to 40	-20 to 50
Gyros/IMUs	0 to 40	-10 to 50
Star Trackers	0 to 30	-10 to 40
C&DH Box Baseplates	-20 to 60	-40 to 75
Hydrazine Tanks and Lines	15 to 40	5 to 50
Antenna Gimbals	-40 to 80	-50 to 90
Antennas	-100 to 100	-120 to 120
Solar Panels	-150 to 110	-200 to 130

Table G: Typical temperature ranges for all the components

With the table as a reference we started a thermal design that allows us to accomplish the necessary requirements. There are some theorical concepts we had to have in consideration for this control. Our Cubesat will be irradiated externally both from the sun, the earth atmosphere albedo (the proportion of incident radiation that is reflected by the surface) and earth infrared emissions similarly to a black body that variates with the distance to earth – to the maximum temperature, we have considered the time of the launching at earth surface; to the minimum, the time of the moon fly-by.

To balance this external heat inputs, we need to consider the emissions of our satellite and the heat dissipation by the payload and all the internal subsystems.

So, we can right the following equation and compute temperatures:

$$Q_{env} = \alpha S(A_p + RA_R) + \varepsilon IRA_{IR} \quad , \quad Q_{env} + Q_{in} = \sigma T^4 \sum_{1}^{n} \varepsilon_n A_n$$

The albedo has been considered of 30% and the IR emission by earth has been calculated as the potency of earth radiation as a black body, multiplied by the area of earth and then we have made the quotient between this and the area of the spherical surface with radius equals to the distance between the satellite and earth center. To the maximum temperature we have considered the earth radius because it corresponds to the launching time and to the minimum, we considered that plus the distance between the earth and the moon.

At this stage we computed the maximum and minimum temperatures. To the maximum temperature we have considered the maximum area of incident radiation both from the sun and earth. This area is the section produced by a diagonal cut on the parallelepiped.

To the minimum temperature we have considered the minimum area of incident radiation. This is the area of the base of de parallelepiped again both for the earth and sun radiations.

We have also considered the internal, payload and subsystems heat dissipation equals to 12W. This value comes from a real CubeSat like the one we are designing.

Using polished aluminum, we obtained about 32 degrees Celsius to the minimum temperature and 75 degrees Celsius to the maximum temperature. So our satellite was extremely hot comparing these values with the typical temperature ranges in the table. Varying between the different treatments of aluminum that we were able to use we have not found any one with the relation between the absorptivity and emissivity to satisfy the requirements. At this stage we have decided to use an aluminum paint bright to coat our satellite. This way we have obtained a maximum temperature of 3.05 degrees Celsius and a minimum of -42.9 degrees Celsius.

Because the most sensible components will be in the interior of the CubeSat and more exposed to the heat dissipation of all the components, they will be stable at a medium temperature that would allow them to survive operational.

RISK ANALYSIS AND MITIGATION

In this section, a risk analysis related to all mission phases is presented on the table below. Here is specified the most important mission risks and some mitigation strategies to deal with them.

Description	Likelihood	Criticality	Actions
Initial orbit insertion failure	Medium	Low	A preliminary impulse should be applied to preform the expected orbit insertion
Orbit characteristics different from expected	Medium	Low	An orbit correction shall be made by applying an impulse
Communications failure	Low	High	Safe mode
Attitude control failure	Low	High	Safe mode
Battery charging failure	Low	High	Safe mode
General electronics failure	Low	High	Safe mode
Propulsion failure	Low	Medium	Orbit correction shall be made
Satellite flyby is performed too far from the moon surface	Medium	High	Orbit correction shall be made
Satellite crash on the moon surface	Low	High	No actions needed
Data transmission failure	Low	Medium	A second attempt shall be made
Incorrect pointing of the payload	Medium	High	Orbit correction shall be made
Payload failure during flyby	Low	High	Orbit correction shall be made

Table H: Risk analysis and Mitigation

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CONCLUSIONS

As presented in this report a solution based on an 18U CubeSat with 6kg of dry weight that will perform a single flyby to the moon was considered the best option by the team to meet the top-level requirements of the mission.

This satellite trajectory is based on a series of impulses each time the satellite passes the perigee of the orbit until it reaches the moon. This is probably the most critical aspect of this mission. As the fuel available as little margin and the mission characteristics may require some orbit corrections, because of the large number of times an impulse is applied to the satellite, an analysis should be made to overcome this problem. The possibility of performing more than one flyby to the moon can be studied as well, even though it will require more fuel.

APPENDICES

Appendices can be used to attach additional materials that your Team may find relevant to understand the report. In particular, the list of all subsystem requirements (that permit to meet the mission-level requirements), other tables created during the project (e.g. change logs of the subsystem requirements, tables of conformity), or the description of the tools developed by the team, can be included here.