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Fitting a high total impulse electric propulsion system in a student CubeSat to compensate the atmospheric drag in low-earth orbit

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Abstract

In this era where the interest in nanosatellites is growing rapidly, the next big step for them is to integrate a propulsion subsystem in order to accomplish more complex missions. With electric propulsion in particular, nanosatellites will be able to perform new maneuvers and new missions, such as missions in LEO by compensating the drag with a thruster. However, designing such a mission and the satellite for it is not easily feasible for a student project. Here we present a preliminary design for a 6U CubeSat capable of maintaining an altitude of about 300 km for more than several months. This project is a fully student project, and it is supported by the CNES and École polytechnique in Paris. It is planned to be ready for launch in the early 2020s. The phase B planning of this project allowed us to design a nanosat capable of withstanding the high demand for power and capable of performing all maneuvers necessary to reach the target altitude and maintain it. All the technical choices allowing these performances are explained: high-capacity batteries capable of providing energy for one whole thrust sequence (50Wh), large, deployable but not steerable solar panels to recharge them and a balanced ADCS strategy allowing both a high energy intake and regular thrust phases to keep a stable altitude. It is shown that a three-axis reaction wheels stabilization is necessary for such a mission, even while rotating the satellite only around a fixed thrust axis. Finally, the trajectography algorithm, for now based on periapsis raising based on GPS data, under constraints of battery charge and eccentricity, is described, as well as the structure of the on-board computer and the technical choices around them. This preliminary design shows how a satellite can handle atmospheric drag at around 300 km for several months with the constraints of a student-designed CubeSat.
Keywords: Electric Propulsion, CubeSat, Atmospheric Drag, Low Earth Orbit (LEO)

Acronyms/Abbreviations
VLEO : Very Low Earth Orbit
LEO : Low Earth Orbit
PDU : Power Distribution Unit
ACU : Array Conditioning Unit
VKI : von Karman Institute
ISS : International Space Station
VHF: Very High Frequency
UHF: Ultra High Frequency
OBC : On Board Computer
ADCS : Attitude Determination and Control System
AIT : Assembly, Integration and Test
ESD : Electro-Static Discharge
eV : electron-Volt
SE : Single Event
JEMRMS: Japanese Experimentation Module Robotic Manipulation System
DoD : Depth of Discharge

1. Introduction

Cubesats have now demonstrated to be a very good solution for conducting a large number of cheap missions, either individually or in constellations. But for a long time some missions were inaccessible to them, because their small sizes made integrating a thruster impossible. However, electric propulsion subsystems have now reached a point in miniaturization where they are small enough to be integrated in a Cubesat, while still providing significant impulse. Solid propellant thrusters in particular, offer a very compact solution, since the propellant is more dense and there is no need for it to be pressurized. However, combining the constraints of a Cubesat with the specificities and demands of an electric propulsion system is challenging, for a number of reasons. First, the power consumption of an electric thruster such as the one we integrate is very high, a lot higher than the power consumption of subsystems on a regular CubeSat. This implies a need for large, deployable solar panels to get a large power intake, but also sizable batteries and Power Distribution Units (PDU), to handle this much power. This leads to a second issue: the thruster, the propellant and the batteries are all big subsystems, so they leave little volume and mass budget for the other subsystems, making the integration of the nanosat a real challenge.

The IonSat mission aims to operate a nanosatellite in VLEO, at altitudes around 300km. The thruster is crucial as it is necessary to handle the main challenge of this mission: atmospheric drag. For that we have chosen ThrustMe's iodine thruster, NPT-30I2. At 300km, the atmospheric density ranges between $2{\cdot}10^{-12}$ kg·m−3 and $3{\cdot}10^{-11}$ kg·m−3, depending on the solar activity [18], and by our analysis the atmospheric drag causes the altitude of a 6U nanosatellite with deployed solar panels to decrease by around 500m per day.

Aside from the technological challenge, IonSat also aims to have a scientific value by integrating a scientific experiment as a payload. While no definitive payload has been chosen, multiple potential payloads are being studied, in particular one to study potential iodine contamination of the spacecraft structure.

2. Mission Definition

2.1 The mission objective: VLEO

Very low-Earth orbits are very interesting for a number of uses: they provide a better resolution for imaging satellites and a smaller latency on data exchanges, which is useful for example for real-time telecommunication. However, VLEO missions are traditionally very hard to achieve for a Cubesat, since they require a thruster, which is hard to integrate in a Cubesat

The main technological objective of IonSat is to demonstrate the feasibility of a student nanosatellite mission in VLEO for a significant amount of time, with a minimum of six months. The rough target altitude is around 300 km, an altitude at which it is very hard to have a stable orbit because of atmospheric drag, and it is planned to try station-keeping even lower with the remaining fuel. We only use a rough altitude because the altitude we will reach is actually determined by the air density, since it is what creates the drag, and the altitude at which a specific air density is found actually fluctuates with the solar activity [18]. The low altitude of the mission is the reason we will need a thruster to keep IonSat in orbit.

2.2 Atmospheric Drag

Challenging atmospheric drag is the primary goal of the mission. This force is the main specificity of the VLEO, because its effects are negligible compared to

to all other terms as soon as you leave these limits (it is 10^{-9} times the main acceleration term at an altitude of 500 km [1]). Our objective is to fight against this force which has the main effect of reducing the radius of the satellite's orbit.

The drag force is described in the literature [2] by the following equation (1):

$$F_T = -\frac{1}{2}\rho S C_d V_r^2 \left(\frac{V_r}{\|V_r\|}\right) \quad (1)$$

where ρ is the density of the air, S is the surface projected according to the plane normal to the speed, Cd the drag coefficient, Vr the relative speed (the atmosphere is in motion with the Earth's surface). It is difficult to find the value of Cd because it depends on the type of the flow and shape of the structure. The literature gives Cd = 2.5 [2] in the case where S is the projected surface perpendicular to the speed. This force implies that the satellite loses energy so it will lose energy in altitude and gain speed.

In an orbit at 400 km, we lose a few meters per hour. The effect increases greatly when we approach the Earth, since density is approximately decreasing exponentially with altitude and speed increases in inverse proportion to the root of the orbit radius (2) .

$$v = \sqrt{\frac{GM}{R}} \quad (2)$$

v = orbital speed, M = mass of Earth, R = radius of the orbit. G = gravitational constant

We tried to understand why we found in the literature a force of expression similar to the force of friction usually exerted by air on the Earth's surface (Cd is constant at high Reynolds number). Indeed, the phenomena at stake are not the same: the mean free path at 300 km altitude is approximately 3 meters and increases with altitude.. So we can't consider the flow as a continuous flow just as we have on earth.

We therefore considered a simple model of flow on the satellite: the satellite is immersed in a set of animated particles with a uniform velocity V. IonSat is inclined by an angle of incidence α with the speed. Fig 1. below presents the notations used in our model.

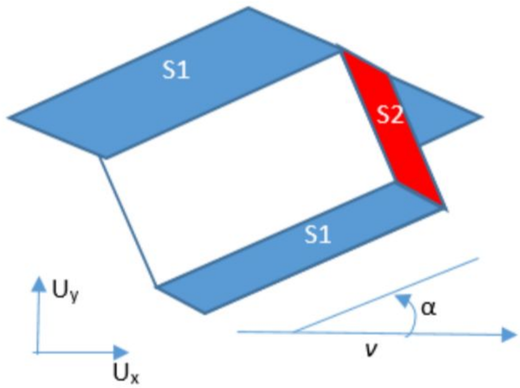

Fig. 1. notations of our simple drag model.

The shocks on the satellite surfaces are elastic and follow the laws of Descartes' reflection (accommodation coefficient is null). With such a model, we establish a balance of quantity of movement in stationary mode on the satellite and thus we obtain the following expression for the aerodynamic result of the air on the satellite (3) :

$$F = \rho v^2 [(S_1\ sin\alpha\ sin2\alpha - S_2\ cos\alpha\ sin2\alpha)U_y + (S_1\ sin\alpha\ (cos2\alpha - 1) - S_2\ cos\alpha\ (cos2\alpha + 1))U_x] \quad (3)$$

This model allows us to find the same dependence as Eq.(1) with the density of the air, the speed of the air flow and the projection of the surface. Its similarity with the equation announced in the literature [2] leads us to believe that it is a coherent model. With this simple model, we know how the lift and drag forces are evolving according to the inclination of our satellite. We were therefore able to verify that at low incidence ($\alpha < 10^\circ$) the lift was negligible in front of the drag. Moreover, at very low incidence, an interesting phenomenon occurs. The lift term is negative because the front face (in red on Fig. 1.) of IonSat deflects the particles upwards a lot.

With the effect of drag now identified, we are able to use it to define our altitude of station keeping. For a given altitude we calculate the drag and deduce the duty cycle of the thruster (proportion of the time the thruster is active) that is necessary to balance the drag, using for the thruster a working value of 1 mN (see subsection 3.2). Fig 2. below shows this logic. The satellite needs to be above the blue line to compensate for the drag.

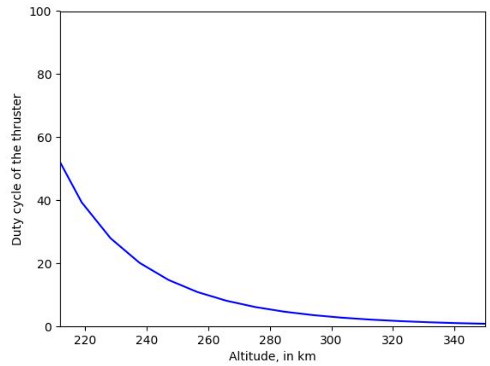

Fig. 2. Duty cycle of the thruster as a function of altitude in order to balance the drag.

2.3 High Power Consumption

Because of the thruster, our spacecraft will consume a lot of energy: during thrust phases, the thruster uses 50 W of power. Our consumptions are given in Table 1. Most components are on all the time, but the thruster and the antennas are not. Since we only have access to one ground station yet, we can make a rough estimate that our antennas have a duty cycle of 2.5%. The duty cycle of the thruster will be determined by the drag, hence will depend on the altitude.

Table 1. Consumption estimates for the subsystems

Subsystem	Consumption (W)
Thruster	50
OBC	0.4
Energy chain	0.6
ADCS	3.5
Thermal control	3
Antennas	12

We need to evaluate how much power can be harvested, because this will put a limit on the duty cycle of the thruster. Hence a limit on the lowest altitude we can hope for.

The maximum surface of solar panels on our satellite is 0.123m^2, that corresponds to 40 solar cells (see Fig.1. in section 3.1.). Then we considered 3 kinds of attitude control.

- One was a complete 3-axis control. It allows for a perfect aiming of the sun but is hard to design and therefore risky.
- Another was with the thrust axis fixed to be always aligned with the velocity axis, and a degree of freedom in rotation around that

common axis. This does not bring as much energy but it is easier to design.
- The last one is the same as the second but without the degree of freedom in rotation.

For each of those models for attitude control, we computed the energy input for each day of the year and determined the worst case scenario as the worst day. We then calculated the maximal duty cycle of the thruster to have the consumption equate the input for this worst case. The results are given in Fig.3. For the second kind of attitude control. We have to be below the blue line in order not to run our batteries dry.

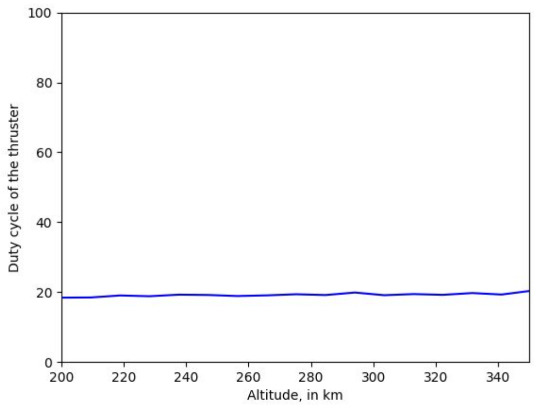

Fig. 3. Duty cycle of the thruster as a function of altitude in order not to overconsume.

We also considered the case where the attitude control is slow or imprecise, and found out that with our ADCS hardware, we loose at most 1% of energy to these issues.

2.4 Mission plan

Fig. 4. sums up the conditions to meet for the mission to be feasible. The red line represents the balance between drag and thrust, we have to be above it. The blue line represents the balance between power harvest and power consumption, we have to be below it. And the green line represents 300km line that is our altitude goal for station keeping. Our mission is to do station-keeping at an altitude under 300 km, which on this graph corresponds to being on the left side of the green line.

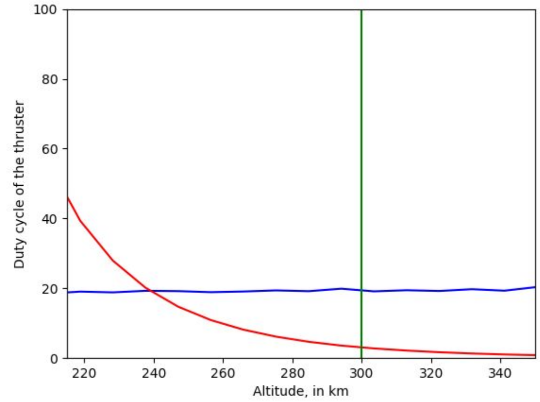

Fig. 4. Admissible altitudes and duty cycles of the thruster

This leaves a reasonably large space for our mission. In fact we can be confident that station-keeping at 260 km is possible. Although the red line might rise because of solar activity, as explained in subsection 2.1.

2.5 Launch Opportunities

Due to our orbit being in VLEO, which is an unusual altitude, there are very few launches that correspond to it. The idea then is to be launched in LEO, either through the ISS or through a regular launch, and then do a phase of descent after the detumbling phase. This descent will first be started using the thruster oriented towards the retrograde vector, and then using aerobraking by placing the satellite in an attitude that maximizes its cross-section, effectively using the solar panels as brakes. Our analysis showed it to be a viable option: with a solar panels surface of 18 dm² (i.e. 18U), the descent from 425 km to 325 km takes 90 days, which is a reasonable amount of time.

3. Preliminary Design

At the highest level, a spacecraft is the union of a structure, a thruster and a tank, a radiation shielding, a communication subsystem, an OBC, an ADCS, an energy subsystem, a thermal control subsystem, and a payload.

This section presents, for each of these subsystems, the design choices that are very specific to a CubeSat mission with a thruster in VLEO. Subsection 3.7 shows how to integrate all the parts in a 6U spacecraft.

3.1 The Structure

We chose to design and manufacture our own structure. This allows for more flexibility during integration, as off-the-shelf manufacturers make structures adapted to their other components.

We chose a 20cm*30cm*10cm shape, because it allows for a center of thrust that is close to the center of mass, making for a more stable system. We also add two 3U*2U deployable solar panels. They deploy along their 3U vertices, to form a solar-reactive plan of surface 18U (see Fig. 5.). We show in Section 3.7 how this fulfills the high needs in energy of the mission.

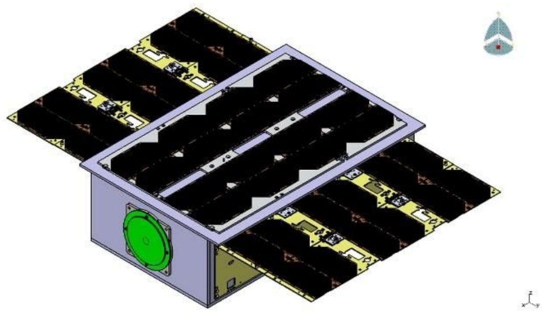

Fig. 5. Shape of the spacecraft and position of the solar panels.

It is crucial to think early in the design process about the AIT. We want the internal subsystems to be very easily accessible even in the late steps of integration [3]. The solution we choose is to make one piece for each face of the spacecraft, and to simply remove the top piece if we need to access any inside component during integration. We also designed some pieces to fix the thruster to the structure (see Fig. 6.).

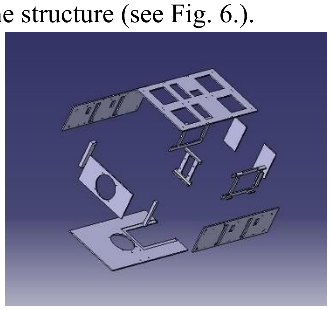

Fig. 6. CAO open model of the structure

The thruster generates heat as radiation. This heat needs to be evacuated toward space. We chose to let one of the lateral faces of the thruster in contact with space to help on this matter.

3.2 Thruster

We need a propulsion subsystem that is fully integrated, because the requirements on fuels from the JEMRMS aboard the ISS are very stringent [4]. It would be difficult for a student team to meet them.

We chose the NPT30-I2 thruster from ThrustMe. It is a gridded ion thruster that uses solid iodine as fuel. The development is in progress so we worked with approximate value that might differ from the actual values: this is considered in our margin management. We considered it is 1.5U*1U*1U in size, weights 1500g, consumes about 50W when active, has an ISP of 3000 Ns and a thrust of 1mN [17].

3.3 Radiation Shielding

There are numerous types of issues that can arise because of radiation. They can be classified by the energy of the particles that causes them, as in Table 2 [5].

Table 2. Issues caused by energetic particles

Type of particle	Energy	Damage
Plasma	Low (eV)	Perturbation of measurements
Electron	Middle (keV)	Charge build-up, ESD
Protons and electrons	High (MeV)	Internal charge build-up, ESD, ionization, displacements, SE
Ions	Very High (GeV)	SE

In most missions, the main part of the low-energy radiations happens in the Van Allen belts. Those are 2 zones where the planet's magnetic field traps most of the electrically-charged particles originating from the solar winds. The inner Van Allen radiation belt typically starts at 1000km for non-extreme latitudes [1]. Since our mission will spend very little time at 400km and then a short time (i.e. less than a year) below 300km, we do not have to shield against low and middle energies.

As for high energies and especially SE, there are software solutions (watchdogs). Hence the usual choice for CubeSats [6], and the one we made, not to include any kind of radiation shielding.

3.4 Communications Subsystem

On-board telecommunications consist of of sending data to the ground – housekeeping, error reports, payload measures – and receiving commands and software updates from the ground.

A complete analysis of the type and length of the data collected on board to be sent resulted in the estimation of 50Mb/day, for a payload observing the environment of the satellite, considering the temporal scale of the phenomena at this altitude.

Thus, we compared 3 combinations of frequencies to communicate: VHF (uplink) / UHF (downlink), UHF/UHF and S/S band, considering that we have an operational VHF/UHF ground station at École Polytechnique and that ISAE, Toulouse, can collaborate on the project with their new S/S ground station. As only the S band can handle a downlink of 50Mb/day, the final choice was a duplex S/S band link between the ground – at least ISAE ground station – and the cubesat.

We chose the S/S EWC-31 transceiver from Syrlinks and the S-band patch antenna from Anywaves to fit the specs. Both were chosen for the EyeSat project by CNES, and our contact with the EyeSat development team, and Anywaves and Syrlinks engineers, ensures an important experience and support for this critical subsystem.

As the ISAE ground station is not operational yet, we have no access to the specs of the antenna. So, the link budget was computed using a worst case for the specs of this antenna, the ISIS S-band ground station. With those specs, we simulated the average downlink per day to choose the optimal configuration. We considered a minimal bit error rate (BER) of 10^(-3) to consider the link stable, with a convolution coding (7,1/2) as error correcting algorithm. The non-isotropic radiation of the antenna implies to make a choice between 2 configurations: 1 antenna below the satellite, or 2 antennas, 1 ahead and 1 below. As the power delivered by the transceiver is fixed, choosing 2 antennas divides the power delivered to each antenna by 2, so the problem is not trivial. The numerical simulation we did resulted in a better average link to the ground with 2 antennas instead of 1. Concerning the modulation to choose, the deal issue is about having a safe modulation, weakly sensible to the noise, or having an efficient modulation, that carries a lot of information. In this case, between the modulation proposed by the EWC-31 transceiver, the simulation resulted in a better average link with the OQPSK modulation (Offset Quadrature Phase-Shift Keying).

The final subsystem is composed of an EWC-31 OPT-C31-DAC/OPT-C31-TM2 transceiver (meaning S/S dual antenna configuration and OQPSK modulation transceiver) from Syrlinks, 2 S-band patch antennas from Anywaves, and an S-band ground station at ISAE-Supaero. This configuration leads to an estimated average of 37Mb/day of downlink. The difference with the initially estimated 50Mb/day can be managed with a diminution of the acquisition frequency of housekeeping. However, the ground segment is a point of improvement, and the case studied here is a worst case. Moreover, we are looking for several antennas to improve communication and reach at least the estimated 50Mb/day.

3.5 On Board Computer

3.5.1 Integration of the OBC in the cubesat

The specifications of the mission require the cubesat to control its position and its attitude in space. In order to keep the orbit on the altitude we chose, a precision of pointing of +/-10° is needed and we designed it a +/-5°. The ADCS module is composed of both the actuators and the board that proceeds information and control them. For modularity and clarity of design reasons the OBC and the ADCS will be split in two different distinct boards and subsystems.

The main on-board computer is the brain of the cubesat. It will take the decisions and control the general behavior of the satellite. It will communicate with several modules : PDU, transceiver, sensors, thruster and ADCS. The PDU will control the power management onboard (power between the batteries, the solar panels and all the modules) and the ADCS will control all the ACS actuators and sensors. The ADCS includes a computer which can help the main computer to process heavy calculus.

The OBC main ship is based on a Cortex ARM-4 microcontroller and runs with FreeRTOS, which is enough calculus power with the help of the ADCS computer and standard to program.

3.5.2 Flight software design

The flight software has been designed to fit the mission in two phases. At the first initialization, the OBC will start all modules and follow a pre-designed mission plan that will allow the cubesat to reach the station-keeping altitude. Then, the cubesat will work as a state machine, which will change its state according to the data sent by all the modules.

The main states will be NOMINAL, THRUSTER and TRANSMISSION. NOMINAL will be the state when the satellite collects data on its sensors and its payload. As the solar panels are mostly oriented on the same side, the attitude in this mode will be solar panels towards the sun, in order to get as much power as possible. THRUSTER is the state when the cubesat needs to thrust and therefore uses almost all its power to thrust and maintain the right attitude. TRANSMISSION is another state when the satellite communicates with the ground segment. S-band antennas are not omni-directional, so we decided to use 2 antennas in order to wider the diagram of the antennas and allow a greater angle of attitude to communicate.

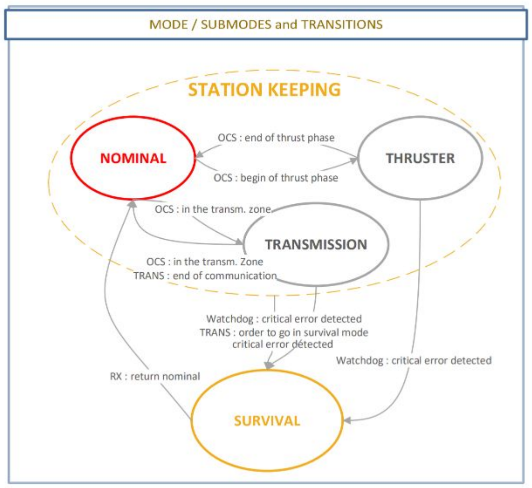

Fig. 8. States and global transitions of the flight software

TRANSMISSION is a separate mode because as we chose to use S-band, the power needed to use the 2 S-band antennas is not compatible with thrusting at the same time. The choice between thrusting and transmitting will be done according to the current state of the satellite (importance of the message to send, state of the memory, altitude and orbit…).

To enter the THRUSTER mode, we designed two possible ways of general operating. The first way is 2 weeks' mission plan sent to the cubesat. The calculus of the trajectory will be done by students on the ground and then sent to the satellite. Those mission plans will include dates of the thrusting phases, lengths, attitudes and possibly future windows to communicate with the ground station. The second way we want to enable is autonomous station-keeping, which could allow the cubesat to autonomously detect the right moment to enter a thrusting phase and manage the states accordingly to all the signals sent by the modules. This model of behavior will be based and validated on a multi-physic simulation.

3.6 ADCS Hardware

To ensure that the thrust vector is in the correct direction, and to maximize the solar energy received, we need to control all three axes with a minimal accuracy. The VLEO implies also greater aerodynamic torque to deal with.

3.6.1 ADCS Requirements

For both the solar panels and the thruster, a deviation of a short θ angle results in a loss of $1 - \cos(\theta)$ in energy. To make this loss lower than 1%, the acceptable deviation is 8°. To include margin, we fixed the accuracy requirement of the attitude control system at 5°.

The main external torques sources on the satellite are the atmospheric drag, the thruster, the gravity gradient, and the earth's magnetic field. Using the drag model from subsection 2.2, the maximum drag torque with a 10° pointing accuracy is 10^{-7}Nm. Using the 3D model of the satellite, in the worst conditions the gravity gradient results in a $2.\ 10^{-7}$ Nm torque [7]. The torques due to the thruster and the Earth magnetic fields are null if the assembly of the satellite is perfect and can otherwise be controlled: for the thruster by adjusting the alignment between the thruster and the center of mass during the final integration tests; for the magnetic field by using magnetorquers. With the inclusion of margins, we considered as the final value for external torques 10^{-6} Nm.

The flight plan makes the satellite change between thrust configurations, where it's oriented towards the prograde, and energy configuration where the solar panels are oriented towards the sun. In order to turn in less than one minute into a specific mode, the peak torque must be higher than 5.10^{-4} Nm.

To recap, the ADCS hardware must be able to control the satellite on 3-axis with a 5° accuracy, a permanent torque of 10^{-6}Nm and a peak torque of 5.10^{-4} Nm.

3.6.2 ADCS Hardware choices

We decided not to design our own ADCS but use a Out-of-the-Shelf component. We chose the CubeSpace CubeADCS [8], which include a 3-axis magnetometer, a 3-axis gyro, 3 reaction wheels, magnetorquers, and a sun sensor. This fits all the requirements, in pointing accuracy, torque, mass, and consumption, as shown in Table 3.

Table 3. ADCS characteristics [9-11]

Attitude determination accuracy	0.2° while sun tracking 3° with magnetometers
Magnetorquers moment	0.24 Am²
Wheel speed range	6000 rpm
Wheel max torque	1.0 mNm
Wheel momentum storage	10.82 mNms
Peak power	< 7 W
Nominal power	< 850 mW
Total Mass	900 g

3.7 Energy Subsystem

Peak power usage of CubeSats are usually not more than 10W [12]. Because of our S-band antennas and mostly because of the thruster, we have a much higher power budget of around 60W when the thruster is on, 16W when the antennas are on, and 6W otherwise. This means our system needs to handle these high voltages and high currents.

3.7.1 Power Distribution Unit

The PDU is the card that gather the energy from the batteries and distribute it to the components of the spacecraft.

The only one we found that handles voltages and currents amounting to 50W is the GomSpace P60-PDU. Hence we chose to use it, which explains the compatibility choices we made for the batteries, the ACU and the solar cells (see sub-subsections 3.7.1, 3.7.2 and 3.7.1).

The characteristics of this PDU are given in Table 4 [16], and the needs of our components are given in Table 5.

Table 4. Characteristics of the GomSpace P60-PDU

Number of outputs	9
Number of voltage converters	3
Voltages possible for converter 1	3.3V, 5V, 8V, 12V, 24V
Voltages possible for converter 2 / 3	3.3V, 5V, 8V, Vbatteries
Current for each output	Max 2.5A

Table 5. Characteristics of the spacecraft's components

Component	Voltage (V)	Maximum power (W)
Communication subsystem	8	12
OBC	3.3	0.4
ADCS block	3.3 & 5	1
GPS	3.3	1
Heaters	24	50
Thruster	3.3	3

We need to handle 4 different tensions, so we chose to embark 2 PDUs. The first one will take care of the thruster, the transceiver, the OBC and the heater. The second one will handle the ADCS block and the GPS. In case we need to add active components to the spacecraft, this leaves us plenty of outputs to do so.

3.7.2 Batteries

Because we chose the GomSpace PDU, we had to choose the GomSpace NanoPower-BPX batteries for compatibility reasons. They have a nominal capacity of 77Wh per pack, and a capacity after 1000 cycles of 61Wh per pack [13].

It is important to notice that the power budget made in subsection 2.4 was made on average, we showed that over a cycle of a few orbits the spacecraft would not consume more energy than it acquires. But there are questions of storage : the batteries have a maximal capacity, so there are times where the batteries are fully charged and the spacecraft will not harvest any power. This can perturbate our averaged power budget. We must have batteries with a large enough capacity that there is no point in time where they are below the target DoD.

To evaluate how many batteries we need, we computed the energy input and the energy output of the spacecraft for every orbit of the year. We then looked at the lowest point reached by the batteries' charge for 1, 2 or 3 packs of batteries. We want this maximum discharge to be less than 40%, because over this DoD the batteries degrade very quickly [13].

We concluded that we need 2 packs of batteries (see Fig. 8.). More precisely, we need 95Wh of capacity at End-of-Life so with 2 packs we expect the batteries to last 1250 cycles.

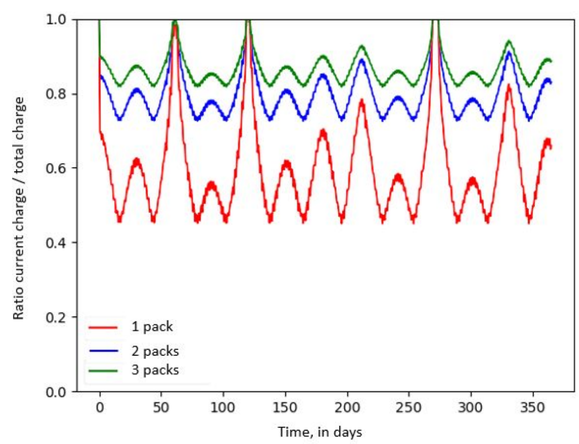

Fig. 9. Charge over time for 1, 2 or 3 packs of batteries after 1000 cycles, and 40 solar cells.

3.7.3 Solar panels

The analysis in the precedent sub-subsection was done for 40 solar cells, and it yields a maximal DoD of 30% for 2 packs of batteries. Therefore it validates the conclusion made in subsection 2.4. that 40 solar cells are enough.

We have considered several options to integrate those 40 solar cells, and concluded that the best choice for our spacecraft is two of GomSpace's double deployable solar panels on the side plus 16 fixed solar cells in the middle (see Fig. 5.).This solution offers a fully integrated deployment system, and will yield no issue of compatibility with the rest of the energy chain.

3.7.4 Array Conditioning Unit

The role of the ACU is to monitor and control the solar cells. It measures the temperature and the output voltage of each cell. It turns the cells on and off with the goal to provide a stable voltage to the batteries.

For compatibility reasons, we chose the GomSpace P60-ACU. Its main characteristics are summed up in Table 6 [14].

Table 6. Characteristics of the GomSpace P60-ACU

Number of inputs	6
Maximum number of cells per input	8
Voltage for each input	Min 4.5V, Max 16V
Current for each input	Max 2A

Since one solar cell is nominally close to a voltage generator at 2.5V working with a current of 0.5A [15], we can put 4 but not 8 solar cells in series on the same input. As we see on Fig. 5., we have 10 groups of 4 solar cells (3 on each deployable panel and 4 in the middle). The solution we chose is to use 2 ACU, and make the links described in Fig. 9.

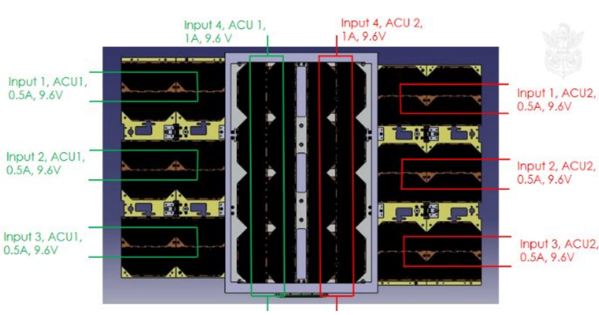

Fig. 9. Connection links between the solar cells and the ACUs

3.8 Thermal Control

At the time this article is being written, the thermal analysis is still in progress. We do not know what the thermal control subsystem will be.

3.9 Integration

We need to have the center of thrust and the center of mass aligned along the thrust axis to avoid creating a torque. Since the thruster pushes along its 1U*1U face, we have no choice for the position of the thruster : it has to be at the back, in the middle of the 2U*1U face.

The solar panels will be facing as much as possible toward the Sun. Since one of the antennas needs to communicate with the Earth, we place it on the face opposite to that of the solar panels. We chose to put the other antenna on the forward face, as it needs to communicate with other satellites.

The reaction wheels need to be as close as possible to the center of mass to maximize their efficiency. They also need to be along 3 orthogonal axes.

We have more freedom for the position of the other components. But since the thruster generates a lot of heat, we try to place all the other components as far from it as possible. We must also place them in order to have the center of mass of the satellite as close as possible to the thrust axis.

We find the solution in Fig. 10, which yields the results in Table 7.

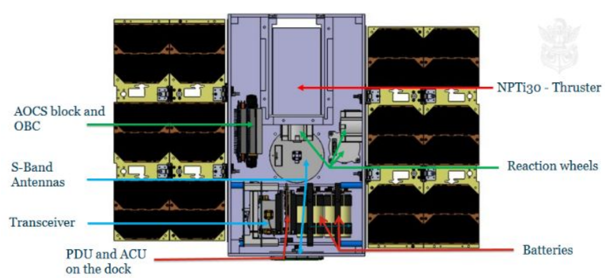

Fig. 10. Position of all the subsystems inside the structure

We did not choose where to integrate the payload yet. There are 3 possibilities : on either side of the thruster, or above the Earth-facing antenna. The choice will depend on the exact dimensions of the payload, but we would prefer the spot above the antenna because it will not move the center of mass of the spacecraft.

Table 7. Mass properties of the spacecraft (margins included)

	Solar panels deployed	Solar panels not deployed
Total mass	9.5 kg	9.5 kg
Center of mass, x	183 mm	183 mm
Center of mass, y	103 mm	103 mm
Center of mass, z	59 mm	58 mm

Maximum term of the matrix of inertia	0.148 kg.m2	0.124 kg.m2

The position of the center of mass is given from the corner of the satellite that is at the back, and at the bottom.

4. Conclusion

Through this analysis, we have presented a mission design to build, launch and operate a student nanosatellite in VLEO using an iodine electric thruster. The mission analysis shows that a mission was indeed possible when we imposed the constraint to be under 300 km. The preliminary design then shows in more precision how to achieve this, using an iodine thruster, a simple software radiation shielding, a 3-axis reaction wheel ADCS, 2 PDU and 2 packs of batteries, all integrated into a custom structure.

Once again, Cubesats show themselves capable of accomplishing missions that were thought inaccessible for them, and make these missions a lot cheaper and simpler than before.

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