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# IONSAT: A STUDENT NANOSAT WITH AN IODINE THRUSTER IN VERY LOW EARTH ORBIT

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With better resolution, lesser latency and lower launch cost, Very Low Earth Orbits (VLEO, under 400km) reveal unexploited potentials. Nanosatellites and electrical propulsion (EP) provide the opportunity to pave the way for space applications at these altitudes. The main challenge at VLEO is the strong atmospheric drag, and thus the very short life expectancy of a free falling satellite. We propose to use the new miniaturized EP devices that have recently become available, and the associated high total impulse to weight ratio, in a CubeSat. Designing such a mission and the corresponding platform is not easily feasible for students, however, and no student projects with such an ambitious propulsion plan exist, at any orbit. We present the preliminary design of IonSat, a 6U CubeSat capable of maintaining a fixed altitude under 300 km for several months. This student-driven project is supported by the French space agency CNES, Ecole polytechnique (Paris), and ThrustMe, which provides a thruster working with iodine. The launch is scheduled around 2023. The team of fifteen undergraduate students is involved in all parts of the project (design, management, research of funding, etc.), and changes every year. The design conducted during the last three years led to a nanosatellite capable of withstanding the high demand for power, and achieving a successful station-keeping at the targeted altitude. The operations plan include a step-down descent from 350km to 250km, with 2-month intervals and a 10 km decrement. The goal is to achieve a lasting orbit control at the lowest possible altitude. We emphasize our presentation on three challenges: orbit control, attitude control, and thermal design. It is possible to maintain the orbit at 300km within a 10km margin and a maximal eccentricity of 0.002 for more than 6 months using a discrete strategy consisting of precisely calculated thrusts around the apoapsis, while controlling eccentricity. A specific strategy privileging minimization of drag over maximization of input power is presented. More demanding strategies can bring more precision but increase the operation complexity. The minimal attitude control system for orbit keeping is achieved without star trackers, and only with reaction wheels and magnetorquers. We also present the thermal design of the nanosatellite, which has to withstand more than 50W in 6U. We believe that IonSat will help inspire new trends of space development, regarding the use of electric propulsion for nanosatellites in the exploration and exploitation of VLEO.
**keywords:** Electric Propulsion, CubeSat, Atmospheric Drag, Very Low Earth Orbit (VLEO), ,

## 1. Introduction

Lowering the altitude of satellites provide numerous competitive benefits: better resolution, lower latency, lower launch costs... For this reason, Very Low Earth Orbit (or VLEO, defined as the orbits with a mean altitude below 400 km) have gained interest during recent years. The previously mentioned advantages allow the performance of platforms in higher orbits to be matched with smaller and simpler platforms in VLEO, thus further reducing costs.

These performances come at a cost, since the interaction with the atmosphere (mainly aerodynamic drag) has an important effect on the flight dynamics, representing a major challenge for stationkeeping. The effect of atmospheric drag starts to be significant at an altitude around 500 km and it dominates above the other perturbations below that altitude (excluding the Earth's gravity field). At an altitude of 250 km, the expected lifespan of a 6U, 10 kg satellite without propulsion is in the order of 1 day. This can be seen as an advantage for end-of-life, since satellites will de-orbit naturally, thus limiting space debris ; but it mainly acts as a central constraint and design driver : stationkeeping at these altitudes requires the use of embarked propulsion.

CubeSats have now demonstrated to be a very good solution for conducting a large number of missions, either individually or in constellations. But for a long time VLEO 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 denser 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, mainly because of the power consumption and size of the subsystems.

The IonSat mission aims to operate a 6U nanosatellite at altitudes below 300km for a minimum duration of 6 months, thus demonstrating the feasibility of complex CubeSat missions in VLEO. For this purpose, we have chosen ThrustMe's iodine thruster NPT-30I2. At 300 km, the atmospheric density ranges between $2.10^{-12}$ and $3.10^{-11}$ kg.m$^{-3}$, depending on the solar activity [1], 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 seeks to add a scientific value to the mission by integrating an experiment as a payload. While no definitive choice has been made, multiple opportunities are being studied, in particular one to study potential iodine contamination of the spacecraft structure.

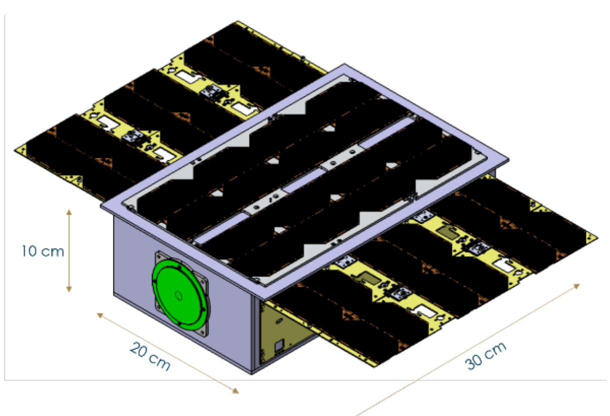



Fig. 1: Overview of IonSat: a 6U CubeSat with deployable solar panels

## 2. Mission definition

### 2.1 *Design drivers*

IonSat is a project essentially carried by students, with limited time, knowledge and financial resources. Moreover, since the mission is intended to be a demonstrator, the development needs to be quick. For these reasons, the choice has been made to rely significantly on one advantage of the CubeSat format, namely the possibility to buy on-the-shelf, standardized components. This allowed us to focus on our main objective: performing stationkeeping in VLEO.

### 2.2 *Major constraint: atmospheric drag*

As stated in the introduction, challenging atmospheric drag is the primary goal of the mission. It is thus critical to determine a relevant model of this force to evaluate the impact on our satellite. In a free molecular flow, the drag force is still :

$$F = \frac{1}{2}\rho S C_{drag} V^2 \quad (1)$$

but the main difficulty consists in finding $C_{drag}$

We used CelestLab (software developed by CNES for orbit and trajectory analysis) to compute the drag coefficients for each part of the satellite. The assumptions made to describe the shocks on the satellite surfaces follow a Maxwellian model, which includes 2 types of reflection [2] [3] : specular reflection (elastic shocks) ; and diffuse reflection (absorption of particles and reemission at the surface temperature).

Drag coefficients greatly vary with the orientation of the satellite. Near the equilibrium position (satellite aligned with its velocity vector), $C_{drag} \approx 2,8$. The results of the analysis are presented fig. 2.

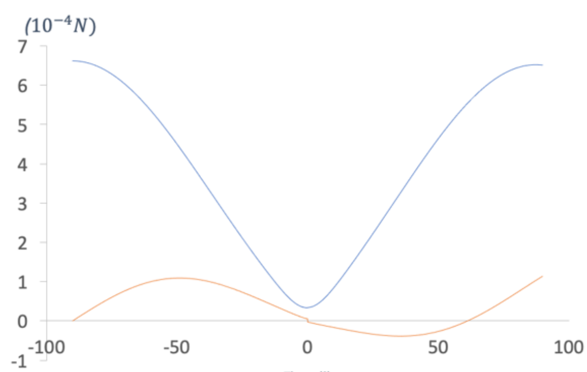



Fig. 2: Lift (orange) and drag (blue) as a function of angle $\theta$ between the relative wind and the surface (altitude 250 km, air density $6,5.10^{-11} kg.m^{-3}$)

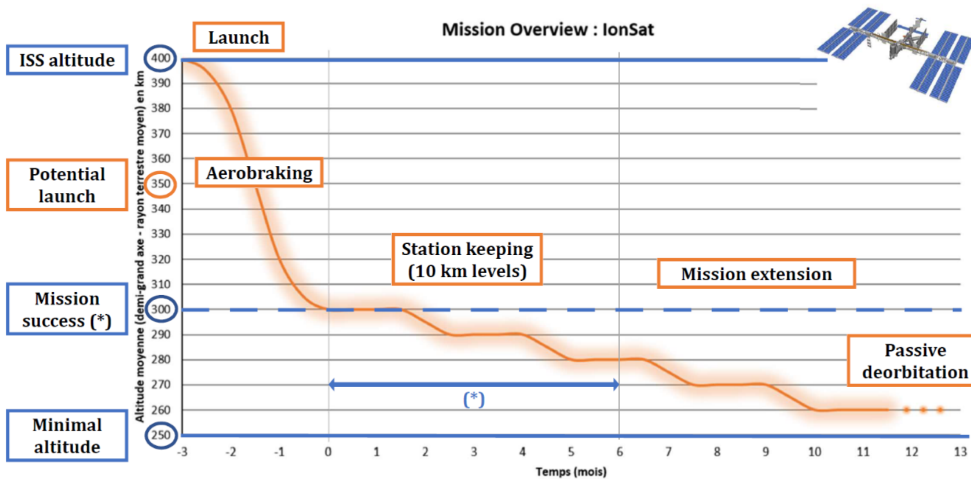



Fig. 3: IonSat mission overview: mean altitude of an orbit in function of the time (in months)

These considerations have to be put in relation with a clear understanding of atmospheric density variations, for which we have used MSIS-E-90 [1], an atmosphere model developed by NASA and implemented in CelestLab. This model accounts for the influence of various parameters on density: altitude, latitude, longitude, time of the day, solar activity.

### 2.3 *Mission Plan*

As previously established, the mission's main purpose is to provide a demonstration of the feasibility of complex missions in VLEO. The lower the stationkeeping is performed, the more striking this demonstration will be. In addition, the importance of atmospheric drag increases dramatically as the orbit altitude decreases ; meaning that even a 10 km altitude change represents a significant difference. Subsequently, IonSat's mission plan has been designed to minimize the final stationkeeping altitude while ensuring at the same time that the mission will not end prematurely. A preliminary study has shown that the critical altitude for our satellite, where even a thruster operating at the maximal duty cycle would not be able to compensate atmospheric drag, is around 250 km. This value was retained as the minimal considered altitude for further studies.

As illustrated in Figure 3, the mission plan which was decided upon is the following:

- Perform an aerobraking from the launch altitude (400km if from the ISS, see §2.4) until 300 km where the mission starts
- Perform a progressive descent (10 km levels), with stationkeeping of at least 2 months for each altitude
- The mission is considered a success after a successful hold under 300 km for a duration of 6 months. If the mission plan unfolds as intended, this should occur at the end of the successful 280 km stationkeeping
- A mission extension is then likely to take place, continuing the 10-km, 2-month levelled descent until the passive deorbitation of the satellite.

The 2 months value was chosen based on the orbital analysis, which showed a periodical fall of perigee altitude with a period of about 6 weeks ; thus making the risk of loss of control due to atmospheric drag higher every 6 weeks. Stationkeeping is therefore considered successful if its duration comprises this "risky" period.

### 2.4 *Launch Opportunities*

VLEO is an unusual altitude, thus there are very few launches that correspond to it. The first option is then to be launched from the ISS and do a phase of descent after the detumbling phase. This descent will be initiated 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 $dm^2$ (i.e. 18U), the descent from 400 km to 300 km takes about 90 days, which is considered an acceptable amount of time.

Other options involve the use of rideshare on a classic LEO launch, or a micro-launcher, to be placed at a custom altitude between 300 and 400 km ; and then the same aerobraking technique to reach 300 km.

## 3. Design Elements

### 3.1 *Propulsion*

One of the key elements of the design is the integration of a thruster in order to compensate atmospheric drag. We chose the NPT30-I2 thruster from the French start-up ThrustMe ; a gridded ion thruster using solid iodine as fuel. The 1,5U format weighs about 1.8 kg ; and for a 60-65 W consumption, it can provide a thrust of 1 mN (with a total impulse up to 9500 Ns) [4].

ThrustMe is a major partner of the IonSat mission, providing a fully qualified thruster. The interfaces between the thruster and the satellite are studied and defined through technical dialogue with the start-up.

### 3.2 *Station-keeping*

The success of the mission is defined by the ability of the satellite to hold a precise VLEO orbit for more than six months. For that, the satellite must compensate the drag and maintain its orbital parameters in the mission margin. The satellite in under the influence of several forces, the main ones being the thrust ($\sim$ 1 mN) and the drag ($\sim$ 0.1 mN).

The first approach to this problem is to consider the power involved. A satellite on a stable orbit has a given constant energy E. By applying the law of conservation of energy, being given $P_D$ the power loss due to drag force and $P_T$ the power gain thanks to thrust applied by the engine, $dE/dt = 0 \Leftrightarrow P_D = P_T$. As we consider mean variation on long duration (a month long hold versus an hour long orbit), we have $\langle P_D \rangle = \langle P_T \rangle$. Therefore $\langle F_D \rangle = \langle F_T \rangle$ with $\langle X \rangle$ the mean value of $X$, $F_D$ and $F_T$ the drag and thrust.

Since $F_T$ is fixed before the launch and $F_D$ depends on the semi major axis $a$, let $\eta(a) = {}^{F_T}/_{F_D}$ be the orbital hold duty-time. Let also $\eta_T = {}^{\tau}/_{T}$ be the thrust duty-time, i.e. the proportion of time when the engine fires, with $\tau$ the mean thrust time on an orbit and $T$ the orbital period. Having $\eta_T = \eta(a)$ assures that the orbit remains stable energy-wise. As we established previously, the expression of $F_D$ at atmospheric density $\rho$ is $F_D = \frac{1}{2}\rho S C_{drag} V^2$

Thus :

$$\eta(a) = \frac{1}{2} \frac{S C_D \rho(a) G M_\oplus}{F_T a} \qquad (2)$$

This formula enables us to compute the required time proportion for propulsion in order to hold the altitude. Knowing the inclination of the orbit, we can also compute the maximum duty-time. If the solar panel produce a mean power of $P_{el}$ and the engine consumes $P_E$, $\eta_{\max} = P_{el}/P_E$. For a given inclination, we can calculate the maximum sun exposure and deduce the maximum duty time, and thus the minimum mean altitude. In the case of IonSat, following this approach the minimum altitude at which stationkeeping is feasible is found to be about 250 km . One can also compute the maximum mission time at a given altitude by dividing the total impulsion I of the engine by the drag force :

$$T = \frac{I}{\langle F_T \rangle} = \frac{I}{\eta(a) F_T} = \frac{2aI}{S C_D \rho(a) G M_\oplus} \qquad (3)$$

If the duty-time allows to know the ability of the satellite to hold a given orbit or “how long to push”, the study of the orbital elements enables one to plan precisely the maneuvers, or “when to push”.

Out of the six orbital parameters, whose variation can be computed using the Gauss Planetary equations system (equations (4) to (9)), two of them define the mission margin and success: the semi-major-axis $a$ and the eccentricity $e$. The inclination i is defined at launch and does not change during the mission, and the right ascension of the ascending node $\Omega$ evolves linearly through time (nodal precession phenomenon). The last two parameters, the argument of the periapsis $\omega$ and the true anomaly ($\nu$ or $\theta$) are not known a priori and must be found on spot. Considering the mission plan, it is reasonable to assume that the orbit is quasi-circular ($e \ll 1$ ). In facts, for a semi major axis margin $h = 10$ km, $e = h/2a \simeq 10^{-3}$. Therefore, we can assume that $\theta$ equal to the eccentric anomaly $E$ and the mean anomaly $\mathcal{M}_0$.

$$\frac{\dot{a}}{a} = \frac{2\,h}{\mu\,(1-e^2)}\left[e\,\sin\theta\,F_r + (1 + e\,\cos\theta)\,F_\theta\right] \qquad (4)$$

$$\dot{\mathcal{M}}_0 = \frac{h}{\mu}\,\frac{(1-e^2)^{1/2}}{e}\left(\left[\cos\theta - \frac{2e}{1-e^2}\frac{r}{a}\right]F_r + \left[1 + \frac{1}{1-e^2}\,\frac{r}{a}\right]\sin\theta\,F_\theta\right) \qquad (5)$$

$$\dot{e} = \frac{h}{\mu}\left[\sin\theta\,F_r + (\cos\theta + \cos E)\,F_\theta\right] \qquad (6)$$

$$\dot{\omega} = -\frac{h}{\mu e}\left[\cos\theta\,F_r - \left(\frac{2 + e\cos\theta}{1 + e\cos\theta}\right)\sin\theta\,F_\theta\right] - \frac{\cos i\,\sin(\omega+\theta)\,r\,F_z}{h\,\sin i} \qquad (7)$$

$$\frac{di}{dt} = \frac{\cos(\omega+\theta)\,r\,F_z}{h} \qquad (8)$$

$$\dot{\Omega} = \frac{\sin(\omega+\theta)\,r\,F_z}{h\,\sin i} \qquad (9)$$

As mentioned previously, for a given mission plan, $i$ is given and does not evolve. Therefore, the variation of $\Omega$ is known ($R_\oplus, J_2$ are constants):

$$\dot{\Omega} = \frac{-3}{2} \frac{R_\oplus^2}{\left(a(1-e^2)\right)^2} J_2 \omega \cos i \tag{10}$$

A full forward burn (in order to minimize the drag) affects, at the first order in eccentricity, the semi major axis a, the eccentricity $e$, and the argument of the periapsis $\omega$. Since all effects depends on $\omega$, we will first determine the influence of a thrust on $\omega$. Having a quasi-circular movement, at the first order:

$$\theta(t) = \frac{2\pi t}{T} - \omega(t), \text{ with } \omega(0) = \omega_0. \tag{11}$$

Thus

$$\dot{\theta} = \frac{2hF_\theta}{\mu e} \sin\left(\frac{2\pi t}{T} - \omega(t)\right) F_\theta \tag{12}$$

Resolving this equation analytically reveals two very distinct regimes and an intermediate one. First, if $e < 10^{-5}$ ($\Delta a \simeq 100$m or less),

$$\omega(t) = \left(\frac{2\pi}{T} - \frac{4T}{\pi}\left(\frac{hF_\theta}{\mu e}\right)^2\right) t \tag{13}$$

This is interesting, because the periapsis seems "pulled" behind the satellite if the satellite pushes, and "repelled" if the only perturbation is drag. The true anomaly $\theta$ is then equal to $\frac{4T}{\pi}\left(\frac{hF_\theta}{\mu e}\right)^2 t$.

If $e > 10^{-3}$ on the other hand, the periapsis does not move as long as the maneuvers last less than a period. The intermediate regime, $e \in [10^{-5}, 10^{-3}]$ is difficult to handle because the equations are too complicated to solve analytically. Therefore, it is interesting to be able to plan the maneuvers using an orbit propagator.

Maintaining the circularity of the orbit implies the control of the eccentricity value. Integrating the associated Gauss equation, we get :

$$\Delta e = \frac{+TF_T}{\pi m_{sat}} \sqrt{\frac{r_p}{\mu}} \left[\sin\frac{2\pi(t_f - t_p)}{T} + \sin\frac{2\pi(t_p - t_i)}{T}\right] \tag{14}$$

having $t_p$ the epoch of the periapsis, $t_f$ the epoch of the end of the burn, and $t_i$ the epoch of the beginning of the maneuver, and considering $\omega$ as constant. It then appears that a short prograde burn at the apoapsis (of a short retrograde burn at the periapsis) reduces the eccentricity. However, a continuous perturbation, e.g. the drag, does not change the mean eccentricity, since $\Delta e = 0$ if $t_f - t_i = T$.

Holding the semi-major axis around a given value implies to control its value. For that, the integration of the associated Gauss equation brings us to

$$\Delta a = \frac{2a^{3/2}}{\sqrt{\mu}} \frac{F_T}{m_{sat}} \left(\Delta t + \frac{T}{4\pi} e \begin{pmatrix} \sin\left(\frac{2\pi(t_T - t_p)}{T}\right) \\ -\sin\left(\frac{2\pi(t_i - t_p)}{T}\right) \end{pmatrix}\right) \tag{15}$$

and, neglecting the first order term,

$$\Delta a = \frac{2a^{3/2}}{\sqrt{\mu}} \frac{F_T}{m_{sat}} \Delta t \tag{16}$$

For a half an hour burn, centred on the periapsis, it comes that $\Delta a = 450$m. Drag has a similar effect:

$$\Delta a_{Drag} = \frac{2a^{3/2}}{\sqrt{\mu}} \frac{F_D}{m_{sat}} \Delta t \tag{17}$$

Computing with the known value of $F_D$ brings :

$$\Delta a_{Drag} = \frac{\rho(a) C_d A \sqrt{\mu}}{a m_{sat}} \Delta t \tag{18}$$

It is worth noting that, since the semi-major axis can be altered whatever the position of the satellite, it is in our interest to privilege maneuvers around the apoapsis, as these have the greatest effect on the reduction of the periapsis. It is counter-intuitive, because usually orbital maneuvers are more efficient while pushing deep inside a gravity well (Oberth Effect). Here however, pushing only around the periapsis causes a huge raise of the apoapsis and therefore, a huge increase in eccentricity. It may be interesting to compute more accurately the effect of the atmospheric drag on the satellite trajectory: as the atmospheric density is higher near the periapsis, atmospheric drag should reduce the eccentricity. Furthermore, the use of osculating orbital elements could be interesting to study: since the satellite evolves in very low and circular orbits, the argument of the periapsis moves very quickly and therefore, the previous analysis is incomplete.

Having an engine on a satellite also unlocks the ability to do complex orbital maneuvers such as plane inclination modification or transiting between orbits. The satellite has for example enough $\Delta v$ to raise its altitude back and forth from 300km to 600km, or to change its orbital inclination by more than 2 degrees. These abilities open up the possibilities for complex mission plan involving SSO insertions or powered altitude/inclination change.

It is finally worth noting that even the simplest strategy (pushing for 600 seconds around every apoapsis) enables the orbit to remain stable while keeping $e$ under 0,002.

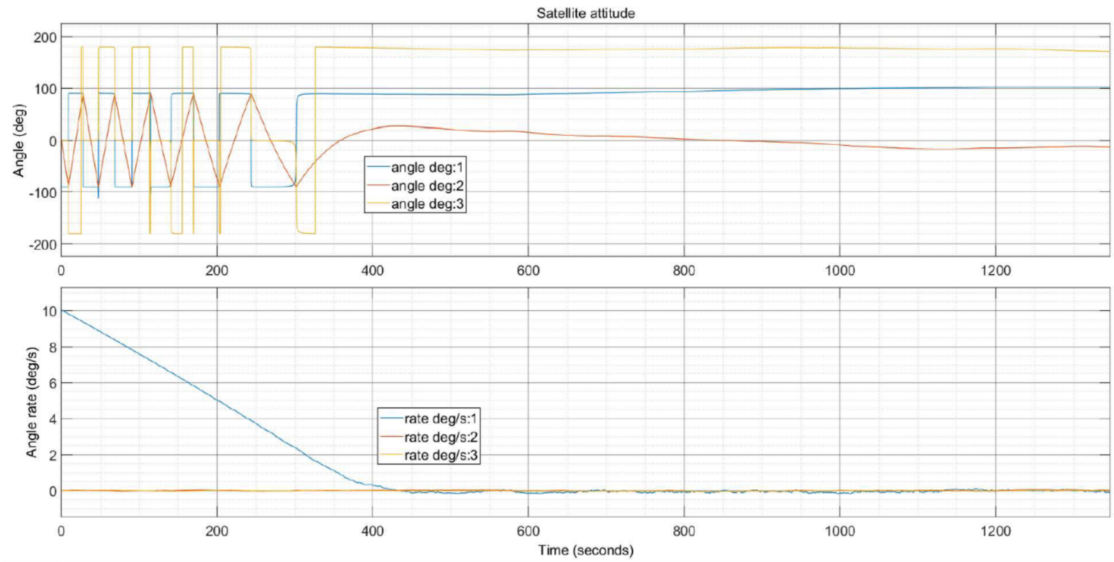



Fig. 4: B-dot implementation and test (angle and angle rate of the satellite under a strong initial rotation)

### 3.3 *Attitude determination and control system*

The specifications of the mission require the CubeSat to control its attitude on all 3 axes with accuracy: to ensure that the thrust vector is in the correct direction, limit drag and maximize the solar energy received. In order to maintain the desired orbit, a precision of pointing of $\pm 8°$ is needed. We designed it at $\pm 5°$ in order to include a margin. The ADCS must compensate perturbative torques from various sources, the most important being thruster misalignment and atmospheric torques in VLEO. Other perturbations such as solar pressure or gravity gradient are negligible. The order of magnitude for external torques is estimated at $10^{-6}$ Nm. Finally, one of the main challenges for the ADCS is to guarantee a successful initial detumbling as well as a simple and reliable safe mode. We chose a hardware configuration allowing to fit all the requirements in pointing accuracy, torque, mass, and consumption ; namely the CubeSpace CubeADCS, which includes :

- a 3-axis gyrometer,
- two 3-axis CubeMag magnetometers : 1 fixed, 1 deployable for redundancy and increased precision,
- a sun sensor,
- three Cubewheel Medium reaction wheels,
- three magnetotorquers to desaturate the reaction wheels : 2 Small CubeRods, 1 double strength CubeCoil

The ADCS module will be mounted as a stack containing the sensors, the actuators and the CubeComputer board processing the information.

The CubeComputer features an embedded software which already implements : communication routines for each component, the IGRF model for Earth magnetic field, a Kalman Filter to compute the quaternion describing the state of the satellite, multiple conversion functions and standard control loops (B-dot, PID). We will ourselves implement control loops (PD controller) and pointing routines. Even if most functions will already be embedded in the actual ADCS, we wrote them in SIMULINK to have a full simulation environment where we could verify the validity of our choices ; interfaced with VTS Timeloop, a tool developed by CNES to visualize the attitude of the satellite in 3D. In order to facilitate computation and avoid singularities (with the Euler angles for example), we used quaternions to represent the state of the satellite in our algorithms. The curves presented here, however, use Euler angles.

For example, the simulation allowed us to validate the commonly used B-Dot algorithm for the detumbling phase, relying on a direct control of the magnetic moment of the magnetotorquers [5]. Figure 4 show these results for an initial rotation $\vec{\Omega} = (0; 10; 0)°/\text{s}$. An angle rate of less than $1°/\text{s}$ is reached in 450s. This result complies with the chosen requirement (detumbling under $2°/\text{s}$ in less than an orbit).

Other simulations have shown that the magnetotorquers are indeed powerful enough to desaturate the reaction wheels under all circumstances. The saturation rate is kept under 50% with a constant desaturation, even with a perturbation.

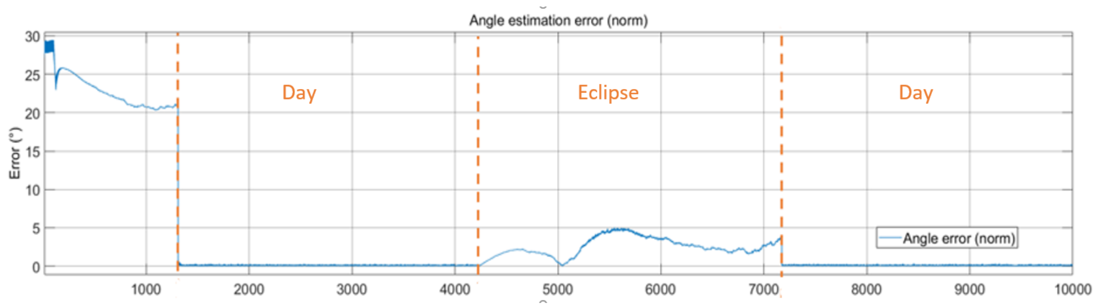



Fig. 5: Angle estimation error using the UKF

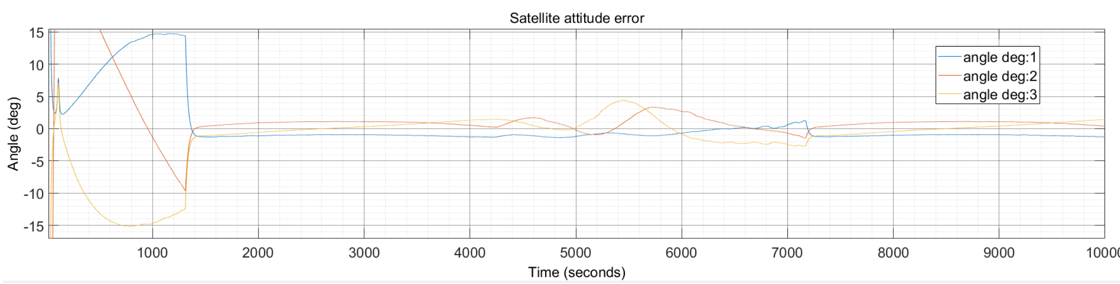



Fig. 6: Pointing error of the satellite using a PD controller and the UKF

In order to improve the measurements and eliminate noise from the sensors [6], we have implemented a Kalman filter in Simulink, allowing us to estimate both the state of the satellite (angle and rotation speed) and the error covariance. Since the equations are non-linear, we have used an Unscented Kalman Filter (UKF). The fact that sun sensors are unavailable during eclipse phases increases dramatically the estimation error given by the filter (difference between the measured quaternion and the actual one) as shown figure 5 : the angle estimation error is below 0.2° during the day but reaches up to 5° in eclipse. The angle rate estimation error (not represented here) stays lesser than $5.10^{-4}$ °/s.

We then need a control loop algorithm to deduce, from the difference between the estimated values and the desired values of angle and angle rate, the speed which needs to be applied to the reaction wheels. For reasons of simplicity and reliability [7], we chose a PD-Controller (Proportional-Derivative). This control loop, including the Kalman filter, allows to fulfil the precision requirement (< 5° on an orbit) : as shown in figure 6, the pointing error is inferior to $1,5°$ during the day and below 5° in eclipse. We expect the final performance of our ADCS to be even better, since the UKF which will be used in the actual algorithm will be coded by CubeSpace; thus, it will have access to more accurate values for the estimation error of their sensors. For example, it will include a zero-rate offset for the gyrometers, which we couldn't include due to a lack of data.

### 3.4 *Power chain*

Peak power use inside CubeSats does not usually exceed 10 W. Mainly because of the thruster, we have a much higher power budget (see table 1) : 71 W when the thruster is on [4], 8.25 W during transmission, and 5 W in the nominal mode (values given with 10% margins). The power chain has to answer several challenges :

- provide sufficient battery nominal capacity ( superior to 140Wh)
- provide the 60W needed by the thruster
- handle at least 4 different tensions needed by the different components
- ensure a solar panel input > 40W

| Subsystem | Power (W) |
|---|---|
| ADCS | 2 |
| OBS | 1 |
| Thruster | 60 |
| Transmission | 3 |
| Power chain | 1.5 |

Table 1: Power budget of the various subsystems

The Power Distribution Unit (PDU) is a card which role is to gather energy from the batteries and redistribute it to the other components. We chose the GomSpace P60-PDU [8] for its capacity to handle voltages and currents amounting to more than 50W. Since we handle 4 different tensions and this model provides 3 convertors, we have to embark 2 PDUs.

For compatibility reasons with the PDUs, the rest of the power chain mainly includes GomSpace components. Through a preliminary mission analysis we concluded that a total of 40 solar cells were needed. In order to meet this need, we chose two 3Ux2U GomSpace deployable solar panels on the side plus 16 fixed solar cells in the middle (see figure 7). The deployable panels will include a built-in thermal knife release mechanism ; once deployed, 40 solar cells will provide 48W of power. In order to monitor and control the solar cells, we will be using 2 GomSpace P60-ACU [8] with the disposition shown below figure 7.

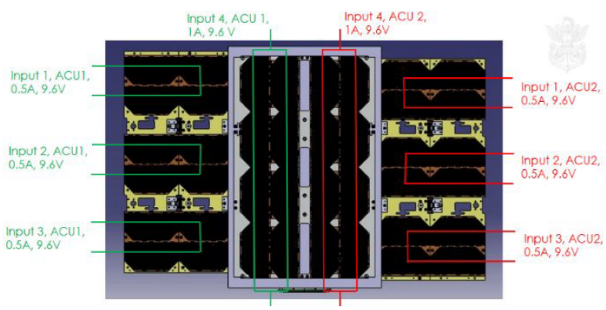



Fig. 7: Solar panels and ACU repartition

Once again for compatibility reasons with the PDUs, we chose the GomSpace NanoPower-BPX batteries [8] with a high nominal capacity (77Wh per pack) and a good life expectancy (61Wh capacity after 1000 cycles). We fixed a maximal DoD (Depth of Discharge) at 40%, because beyond this limit the batteries degrade very quickly. To evaluate the number of batteries needed, we computed the discharge of the batteries for each orbit of the year and considered the lowest point reached for configurations including 1,2 and 3 batteries. We reached the conclusion that 2 packs of batteries were needed.

### 3.5 *Configuration*

The first phase of the project (feasibility study) concluded that a CubeSat mission in VLEO could be achieved with a minimal size of 6U. The retained shape was 20x30x10 cm shape, since it made the center of thrust closer to the center of mass, allowing a more stable system. The thruster pushes along its 1U*1U face ; since we want to minimize the perturbative torques, we need to align the centers of mass and thrust along the thrust axis. Subsequently, the thruster needs to be placed at the back, in the middle of the 2Ux1U face. The other components were placed so as to answer the following constraints :

- the reaction wheels need to be along 3 orthogonal axes, as close as possible to the center of mass
- components should be as far away from the thruster as possible to minimize thermal constraints
- some components (e.g. batteries and PDU) need to be close to one another
- total cable length should be minimal in order to limit perturbative magnetic fields
- the center of mass should be as close as possible to the thrust axis
- maximize the remaining space for a potential additional payload
- take into account the accessibility of components for AIT-related issues

The design which resulted from these constraints is presented figure 8. It includes a compact "brain stack" comprising the ADCS module, the OBC, the power chain and the telecommunications subsystem. The structure will be the only component that we will manufacture ourselves, using aluminum plates. This choice was made in order to provide more flexibility, allowing evolutions with the design ; and after noticing that on-shelf structures were expensive and over-resistant compared to our needs. Since the mission is relatively short and in low orbits, we do not need to include any form of radiation shielding.

### 3.6 *Thermal analysis*

In order to avoid malfunctions, lifespan shortening and/or irreversible damage, it is critical that on-board electronics and components remain within given ranges of temperature. Table 2 gives a summary of these constraints :

| Subsystem | Min. ($°C$) | Max. ($°C$) |
|---:|---|---|
| Deployable panels | -60 | 100 |
| Mounted panels | -40 | 80 |
| Batteries: charge | 0 | 45 |
| Batteries: discharge | -20 | 60 |

Table 2: Thermal range of the critical components (other components have less restrictive constraints than the batteries but are in the same environment).

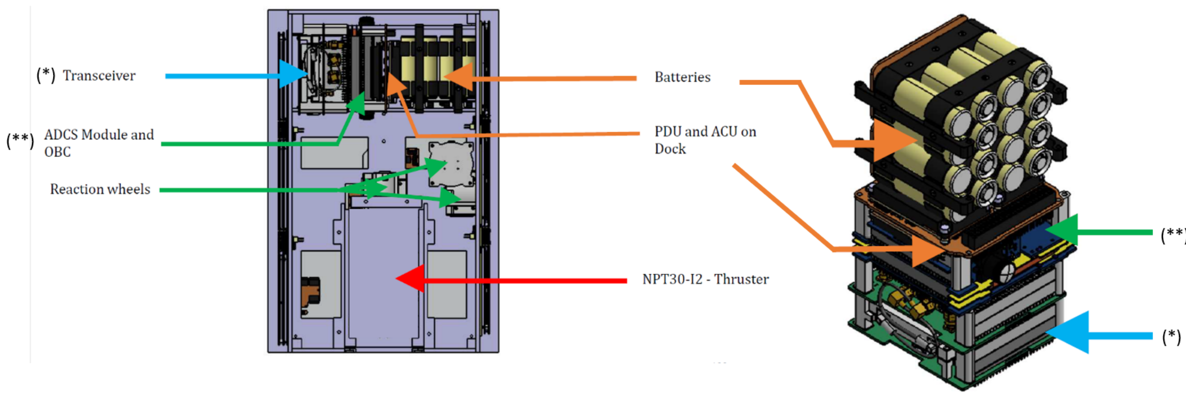
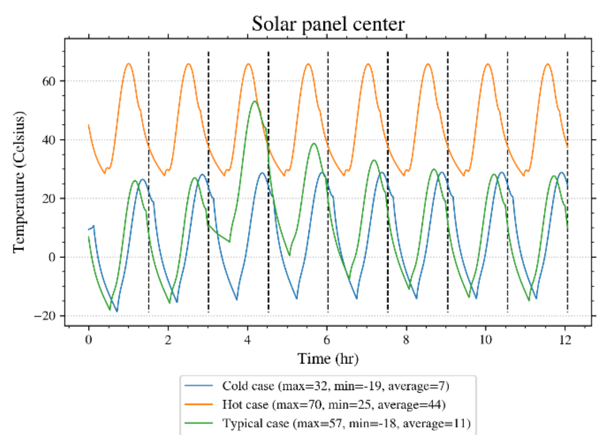



Fig. 8: CAO model of the design for IonSat (left) and detailed view of the "brain stack" (right)

Thus, we need to model and compute thermal transfers and temperatures of the satellite along its trajectory. For this purpose, we used THERMICA, a software developed by Airbus relying on the Finite Elements Model, and allowing us to simulate the environment, orbit and attitude of the satellite. We took into account the radiative flux coming from the Sun and the Earth, the heat-emitting sources aboard the satellite and the radiative loss. Three cases were under study : Extreme cold (minimal sun exposure and internal dissipation, thruster off) ; Extreme heat (maximal sun exposure and internal dissipation, thruster constantly on); and a typical case (average sun exposure and expected functioning scenario). In this last case, the thruster is switched on during the third orbit and the telecommunications during the fourth.

The results, presented figure 9, show that the solar panels always stay within the acceptable range. Batteries also never exceed their maximum operating temperature; but in the cold and typical cases, they can reach negative temperatures, which would normally be an obstacle for their charge. However, this situation occurs only in eclipse, already excluding the possibility to charge the batteries. The amplitude of the oscillations on an orbit are far more important for the solar panels than for the batteries, the latter being representative of the internal components in general. This shows that the latter are relatively protected from exterior radiative flux. Finally, the temperature shows a significant increase upon activation of the thruster and telecommunications, demonstrating the good quality of the thermal coupling within the structure, which is important in order to avoid high gradients.

The conclusion of the thermal analysis shows no limitations : all components should survive during the mission. However, solutions (such as radiators) could be investigated to reduce thermal stress in case of extreme heat, since components could be operating close to their limits.



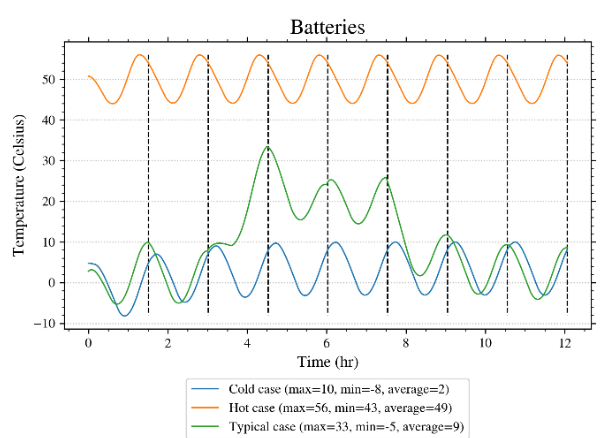




Fig. 9: Thermal analysis for the mounted solar panels (up) and the batteries (down)

### 3.7 *On Board Computer*

For modularity and clarity of design reasons the On-Board Computer and the ADCS will be split in two different distinct boards and subsystems. The main OBC is the brain of the CubeSat. It will take the decisions and control the general behavior of the satellite, while communicating with several modules : PDU, transceiver, sensors, thruster and ADCS. The specifications of the mission have led to the conclusion that we need an orbit propagator able to predict the trajectory over a day (e.g. the SGP4 algorithm [9]). Subsequently, the hardware has to include a FPU (Floating Point Unit) with double precision in order to avoid error propagation ; as well as a minimal RAM of 8 MB. Among other requirements, this has led us to choose EnduroSat's OBC Type I (ARM Cortex M7 processor) running with FreeRTOS.

The flight software has been designed in the form of a state machine, each state corresponding to a primary goal while maintaining the capacity to handle secondary systems. The state changes according to the data sent by all the modules. The modes we have chosen for IonSat are :

- Nominal : normal functioning state of the satellite: data acquisition from the sensors and payload, orbit propagator
- Thruster : activated during the thrust phases, limiting the power for other components
- Transmission : activated when the satellite communicates with the ground segment, answering the pointing needs
- Survival : in case of error ; with division into substates according to the severity of this error
- Brake : destined to be used only during the detumbling and aerobraking phase at the beginning of the mission (ADCS emphasis).

In order to maximize the reliability of our flight software, we need to operate our components as independently as possible in order to avoid error propagation. Thus, the OBC and the ADCS will be in a master-slave relation only. For the same reason, the OBC will operate using threads with minimal communication channels : MAIN (data centralization, choice of the state, management of the slave threads); THRU (manages the engine - warming/start/stop); ACQ (collection and formatting of the acquisition data); TRAN (transmission with the ground station); OCS (orbit propagator). Finally, to avoid memory issues, we will mostly use static variable allocation on boot ; keeping redundancy on each critical system, as well as unallocated memory.

### 3.8 *Telecommunications*

IonSat's telecommunication subsystem will have to send data (housekeeping, error reports, payload measures) to the ground and receive commands and software updates - including emergency patches - from the ground. The analysis of the type and length of the data collected on board to be sent through the downlink resulted in the estimation of 23 MB/day. It is important to state that this value was largely overestimated during the early phases of design (250 MB/day). This led to the choice of a duplex S/S band link with the ISAE Supaero ground station.

This choice was reevaluated in the light of new specifications (e.g. data collected from the thruster) and the decision was made to switch to the UHF (downlink)/VHF (uplink) band. Indeed, this technology is more commonly used, has a higher flight heritage for CubeSats, as well as a widespread network of ground stations, allowing to increase the daily transmission time; and is easier to handle and cheaper than S-band technology. Ecole Polytechnique also has a fully functioning UHF/VHF ground station. Finally, UHF/VHF antennas are omnidirectional, meeting the need for a “safe mode”, allowing transmission no matter the angle of the satellite The only constraint resulting in this change is a limitation in the uplink capacity, which could make software uploads more difficult.

The link budget shows a need for an average of 27 minutes of visibility per day, which is achievable using 5 ground stations. Several options are considered to achieve this coverage, including paid services, partnerships with foreign universities and most importantly with radio-amateur networks (AMSAT, SatNOGS). The French branch of the AmSat has notably become involved in the project this year.

## 4. Conclusion

Through this analysis, we have presented a complete preliminary design for a nanosatellite capable of achieving stationkeeping in VLEO with the help an iodine electric thruster. We have demonstrated the viability of our solution from the point of view of orbital hold, attitude control and thermal analysis. Once again, CubeSats prove themselves to be capable of accomplishing complex missions that were thought to be inaccessible for them, while making these missions a lot cheaper and simpler.

### Acknowledgments

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### Acronyms

**ACU** Array Conditionning Unit.

**ADCS** Attitude Determination & Control System.

**AIT** Assembly, Integration and Testing.

**CNES** Centre National d'Etudes Spatiales.

**DoD** Depth of Discharge.

**EP** Electrical Propulsion.

**FPU** Floating Point Unit.

**IGRF** International Geomagnetic Reference Field.

**ISS** International Space Station.

**LEO** Low Earth Orbit.

**OBC** On Board Computer.

**PDU** Power Distribution Unit.

**UHF** Ultra High Frequency.

**UKF** Unscented Kalman Filter.

**VHF** Very High Frequency.

**VKI** Von Karman Institute for Fluid Dynamics.

**VLEO** Very Low Earth Orbit.

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