MINIATURIZED PROTOTYPE FOR GRAVITY FIELD ASSESSMENT USING DISTRIBUTED EARTH-ORBITING ASSETS

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ABSTRACT

In the past 20 years, and especially in the wake of ground-breaking space missions such as Challenging Minisatellite Payload (CHAMP), Gravity Recovery and Climate Experiment (GRACE), and Gravity Field and Steady-State Ocean Circulation Explorer (GOCE), satellite gravimetry has been demonstrated to be one of the best techniques to monitor the evolution of the Earth's surface mass transport processes and how that related the various geophysical processes.

This article provides an overview of a CubeSat platform (*miniaturized Prototype for Gravity Recovery using Distributed Earth-orbiting satellites* - uPGRADE) and its payloads designed to achieve similar scientific goals as its predecessors. Gaining from the recently emerged NewSpace era, the uPGRADE mission proposes: i) the development of a reusable CubeSat framework fitted to carry a payload to low Earth orbit; ii) extend the use of micro-electromechanical systems (MEMS) technologies to the development of a high-resolution, low-cost, low-weight accelerometer unit able to sense the minute non-gravitational accelerations at orbital altitude; iii) develop a miniaturized Star Tracker suitable for CubeSat, employing an integrated design between optics and processing units, resorting to field programmable gate arrays (FPGAs). Some of the lessons learnt during the design process are discussed and challenges ahead are presented.

1 INTRODUCTION

The uPGRADE (*miniaturized Prototype for Gravity Recovery using Distributed Earth-orbiting satellites*) mission aims at developing a nano-satellite capable of monitoring the temporal variations of Earth's gravity field and measuring the neutral thermospheric densities. To that end, we developed a CubeSat platform from the ground up to address the design challenges presented by the scientific goals. Though developed within the scope of this scientific mission, this CubeSat platform, on the long run, aims to provide the capabilities required to perform other types of missions, such as the qualification of space systems in Low Earth Orbit.

Previous missions such as Gravity Recovery and Climate Experiment (GRACE), Challenging Minisatellite Payload (CHAMP) and Gravity Field and Steady-State Ocean Circulation Explorer (GOCE) required substantial investments and development costs to provide the necessary data to reconstruct Earth's gravity field and allow the study of large mass transport processes. The uPGRADE mission arises as a first attempt to develop a small, light and low-cost satellite measurement system that will make financially feasible the continuous monitoring of Earth's gravity field by multiple systems, while addressing the issue of observing the fast mass transport processes associated with weather systems. Similarly, multiple spacecrafts make it possible to build a network of assets that are able to measure Earth's thermospheric densities within different environmental conditions (e.g., Sun illumination) and from different positions around the Earth, at the same time, providing a potentially improvement on the characterization of Earth's atmosphere, which is a crucial step towards space weather monitoring and space debris mitigation.

Following the footsteps of previous missions, uPGRADE's primary means of gravity field recovery and thermospheric density measurement is a combination of highly precise accelerometers and data based on the global navigation satellite system (GNSS). The accelerometer is sensitive to non-gravitational forces (aerodynamic forces, solar radiation pressure, and internally-generated forces such as thermally induced deformations, structural vibrations and propulsion forces). While non-gravitational forces can be directly measured by the accelerometer, the GNSS measurements enable the precise orbit determination that provides the total accelerations acting on the spacecraft and, in turn, their difference gives an estimation of the gravitational accelerations.

Common commercial of the shelf (COTS) accelerometers are usually unable to provide accurate enough measurements to allow for useful estimations of thermospheric densities. On the other hand, highly precise accelerometers are often large, heavy and expensive, making them unfit for a small and light spacecraft as the uPGRADE. Hence, the need to have a tailored high-resolution accelerometer arises and, to meet the low mass and volume needs, MEMS are most suitable. Alongside with the need to have precise accelerometers comes the need to have precise knowledge over the accelerometer attitude. To that end, Star Trackers are added on-board the uPGRADE spacecraft. These two payloads along with the uPGRADE spacecraft structure, on-board computer and software (i.e., attitude determination and control system–, housekeeping, communications, payload management), cover the novel sub-systems developed under the uPGRADE mission and further described in the upcoming sections.

The uPGRADE mission serves the dual purpose of validating the proof of concept of small low-cost satellites for gravity and thermospheric studies, and qualifying all the novel technologies developed within the mission. To accomplish these goals, the uPGRADE mission is designed to last about 3 years in a sun-synchronous orbit at an altitude of around 500 km. Besides providing the conditions needed for the mission, this type of orbit is commonly used by other types of missions (e.g., Earth observation) and, for this reason, rideshare launch services usually provide releases into this family of orbits. This comes as an added benefit to reduce the overall costs of the uPGRADE mission.

The uPGRADE is a conjoint effort of several partners involving an industrial team based in Portugal and led by Spin.Works, and a scientific team based in Austin (USA) and led by the Center for Space Research, of the University of Texas at Austin. In addition, a major contribution to the scientific work – the development of a MEMS-based highly sensitive and accurate accelerometer – is led by the Iberian Nanotechnology Laboratory and the University of Minho in Portugal. The *Instituto da Soldadura e Qualidade* (ISQ) is responsible for the testing and qualification of uPGRADE's payload and spacecraft, as well as the development of the data interpretation platform responsible for translating and organizing the scientific data generated within uPGRADE.

In the following sections, the uPGRADE spacecraft/platform is described. Starting with the Attitude Determination and Control System (ADCS), followed by the Communications system, On-Board Computer architecture, the Power Supply system, and the Main Frame design. Afterwards, the High-Resolution Accelerometer, the Star Tracker, and the Payload Harness subsections provide an overview of the main elements of the payload.

2 THE UPGRADE SPACERAFT

After separation from the launcher, the spacecraft assumes a safe state, where the minimum set of systems required to acquire a stable, power and thermally safe attitude are used. Furthermore, in the event of a failure during operation, the spacecraft shall be able to transition to this state for an undefined amount of time. These specifications define the Standby and a Safe states. The former is employed during launch, while the system is idle, and the latter during safe operations.

Once the Launch and Early Operations Phase (LEOP) ends, the mission enters the commissioning phase. During this phase, all the on-board systems are checked and calibrated. As previously described, the uPGRADE spacecraft is comprised of several novel systems, never flown before; hence, naturally, during the commissioning phase the first technology demonstration occurs. This demonstration will assert the operation and performance of the payload: Star Trackers and the high-resolution accelerometer. This phase lasts for a year at most to give enough time for system reconfigurations, if needed, and to subject the systems to a large set of operational conditions.

The first scientific data collection campaign begins after the successful commissioning of the spacecraft. This phase lasts for 6 months and aims to: i) validate the operations concept during acquisition and download of the payload data; ii) assess the impact of the on-board perturbations (e.g., reaction wheels vibrations) on the accelerometer measurements; iii) test the data interpretation platform responsible for converting, storing and publishing the payload data acquired on board. During this phase, the spacecraft shall be in Data Collection state.

Having verified the nominal operation of all on-board systems and the scientific data collection and download capabilities, the mission enters the second technology demonstration phase. Lasting up to 6 months, this phase is reserved to perform additional demonstrations/tests and calibrations. As before, during this phase the spacecraft shall be switched to Commissioning state.

Prior to the decommissioning of uPGRADE's spacecraft, the missions enters the detailed data collection phase. This should last for a year and provide the necessary scientific data to fulfil the mission goals. This phase is left to the very end of the mission to allow for the prior verification of all systems, assess the impact of external perturbations and, above all, enable the acquisition of data at lower orbits where the scientific return is greater.

Aside from the aforementioned states, three other are available to the spacecraft: nominal, high data link and decommissioning. In future iterations of this spacecraft, this latter state shall enable the self-destruction of the spacecraft through the use of propulsion systems, which are not necessary for uPGRADE, given its low orbital altitude. The high data link state is designed to allow for the download of large volumes of data to the Ground Station (GS). Although this function is also available within the data collection state, this state is meant to be used when other large data is required to be sent to the ground (e.g., images, time series of monitoring data). Finally, in the Nominal state the spacecraft will ensure safe power supply and communications with the ground. This state acts as a fallback state between mission phases or standby periods.

The spacecraft state architecture reflects the refinement of previous designs performed at Spin.Works. It operates within two layers: the nominal semi-autonomous and the safety layer. The first allows for the transition between states through telecommands and by scheduled or triggered events. In case of failure, the safety layer transitions the spacecraft to the Safe state restoring its control and safe power generation. This transition shall be issued by the Fault Detection, Isolation and Recovery (FDIR) system upon the detection of a failure or malfunction. Exceptionally, the decommissioning state can only be entered or left by direct telecommand (TC) from the Mission Operation Centre (MOC). Similarly, the spacecraft can only switch from Safe to Nominal state through TC. This approach is adopted to ensure the MOC has full control over the transition from a Safe state to other possibly unsafe sates, including the Decommissioning state which is meant to be destructive by design.

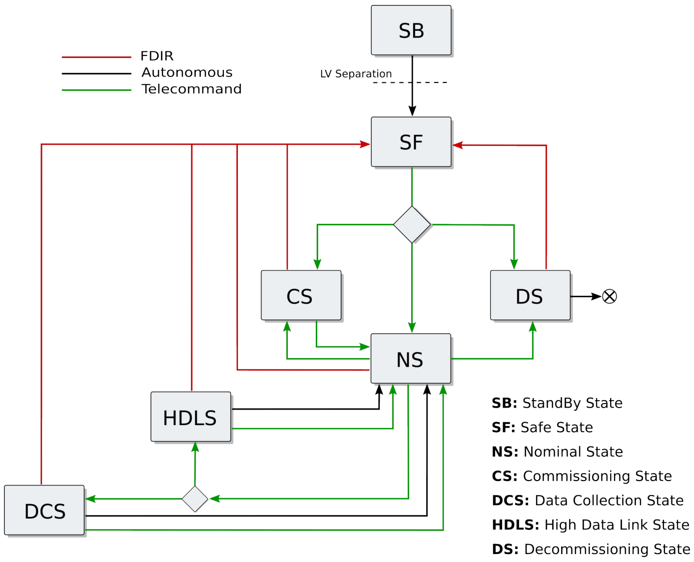


Figure 1. Spacecraft states diagram

2.1 Attitude Determination and Control System (ADCS)

The ADCS has 5 operational modes: Detumbling (DET), Attitude Acquisition (AAM), Sun Pointing (SPM), Enhanced Sun Pointing (ESPM) and Ground Tracking (GTM). Each mode is enabled according to the spacecraft state and operational goal.

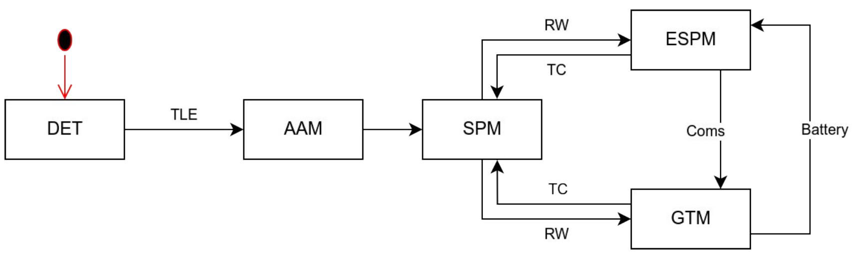


Figure 2. Attitude Determination and Control System mode diagram

During the initial phase of the mission, the ADCS calls the DET mode to stabilize the spacecraft, minimizing the tumbling effects from the launcher separation. Furthermore, this mode is designed to work as a safe fallback in case of a failure is detected, thus minimizing the dependencies on other systems (e.g., gyroscopes, reaction wheels). Also aiming at a simple and reliable design, in this mode, the ADCS makes use of the B-Dot algorithm [1] and the set of magnetorquers rods (MTQ) to keep the spacecraft spin rate below 0.2deg/s. Once Two Line Element set (TLE) messages start to be received, the ADCS transitions to the AAM making available a full attitude and position estimate for the spacecraft. The TLE messages will be updated at every communication with the GS, with the SPG4 algorithm [2] being used to propagate the spacecraft position between communications with the GS. Based on the spacecraft position estimate, the spacecraft is able to compute Earth's reference magnetic field and Sun direction which are provided to the navigation. The on-board navigation filter is based on [3], using the magnetometer and the fine Sun sensor as observations to correct errors in the state propagation which is accomplished through the integration of gyroscope measurements.

During the scientific data acquisition phases, the spacecraft will engage the on-board payload increasing the overall power consumption. To maintain a power positive balance the ADCS calls the SPM to point the solar panels towards the Sun. To perform this manoeuvre with MTQ the controller uses a sliding mode control law based on [4] to reduce the effect of external disturbances. Due to the low control authority and low torque, solely using MTQ for attitude control would result in pointing errors that would lead to large downtimes between payload data transfers to the GS and the recharging of the spacecraft battery. Thus, reaction wheels (RW) were added to the system with its use restricted to the phases without payload data collection.

Table 1 - Power difference between RW and MTQ control modes

GNC Mode	Average Consumption	Average Received	Power Balance	Downtime Ratio
SPM (w/ magnetorquers)	1.1W	5W	3.9W	66%
ESPM (w/ reaction wheels)	2.8W	10W	7.2W	42%

The use of RW allows for improved pointing towards the Sun enabling the Enhanced Sun Pointing mode. While in this mode, the ADCS controls the spacecraft attitude with RW using a double cascaded PI control law. Nonetheless, the MTQ are still used in this mode to manage the angular momentum stored in the RW. The RW's higher torque output and availability enable precise, highly dynamic manoeuvres which fit the characteristics of the High Data Link state use scenario. In this scenario, the S-band antenna's tight radiation pattern requires precise pointing to the GS during the communication window, therefore a Ground Tracking guidance module is used to update the controller references and compensate for the state's high dynamics.

2.2 Communications

The volume of scientific data to be downlinked is orders of magnitude greater than the uplinked telecommands required to operate the spacecraft. For this reason, two communication channels are used in this mission: a high-throughput simplex channel for data downlink and a low-throughput half-duplex channel for control messages, telemetry, and telecommands.

The spacecraft is equipped with a half-duplex ultra-high frequency (UHF) transceiver connected to a deployable cross dipole antenna, and an S-band transmitter connected to a patch antenna. The deployable antenna is comprised of four rigid-body spring-loaded rods with a triple-redundant burn wire release mechanism. Their rigidity raises the natural resonance frequencies of structural vibrations induced on deployment, relevant to the scientific mission's acceleration measurements. This type of antenna grants attitude-independent communication for a robust UHF control channel. S-band downlink throughput is maximised to an estimated 15 Mbps if the patch antenna points at the Ground Station during the communication window.

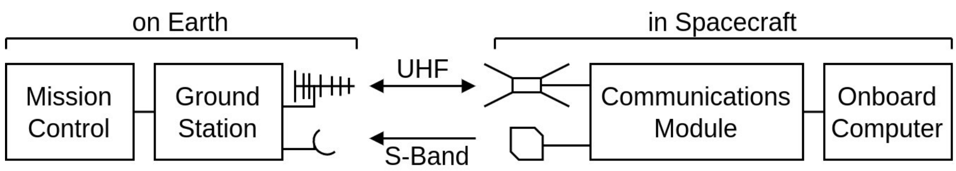


Figure 3. High-level architecture of the Communications System

In addition to hardware modules, the ground station and the communications module are also conceptually comprised of software modules that manage the messages exchanged between them. Messages are exchanged as a result of the operations and commands issued by mission control. The interaction between mission control with the on-board computer are coordinated via the communications system.

Due to the fact that the UHF channel is half-duplex, frame collisions can occur if communication is asynchronous. Though there are algorithms for collision avoidance and recovery, the simpler solution of a synchronous, master-slave model forms the basis of the protocol design. All downlinked UHF frames are responses to uplinked frames. To ensure reliable transmission, the protocol stipulates a mechanism for ground-based retransmissions, which can handle frames lost on uplink and downlink. This is implemented by retrying frame transmissions until a response is received. Duplicate frames are discarded at the spacecraft, as each message is prefixed with a unique identifier.

For large data downlink such as scientific payload or diagnostic data, the S-band channel is used for the reliable downlink of up to 500 megabytes per pass. The protocol is inspired by Saratoga data transfer protocol [9], and tailored to the transmitter's constraints. These types of protocol constitute the network layers analogous to the Data Link Protocol sublayer and Transport Layer, when compared to the network stack as defined by the consultative committee for space data systems (CCSDS). The Network Layer is omitted since data flow is point-to-point.

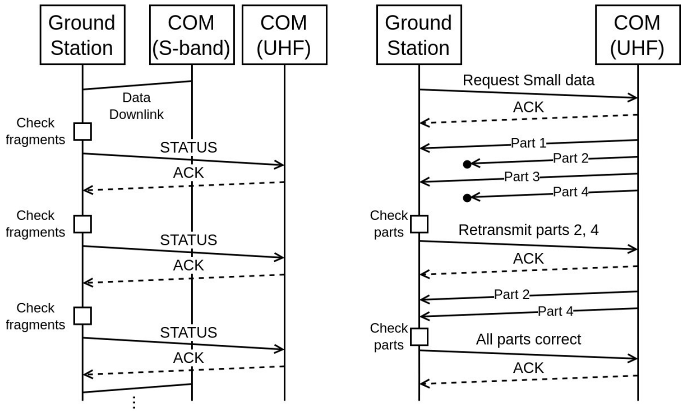


Figure 4. Sequence diagrams exemplifying Large data downlink (left) and Small data downlink (right)

The main objectives are high transmission rate and data reliability, as the communication window is relatively small. This is achieved by streaming fragmented data via S-band, and periodically acknowledging correctly received fragments, and requesting retransmission of missing or corrupted fragments, via UHF. Earth-bound files are fragmented into individually addressable parts of 1250kB.

To allow for the retrieval of telemetry and telecommand responses, the low-throughput UHF channel supports reliable downlink of small data, in the order of kilobytes. This is achieved similarly, by acknowledging which parts have arrived intact and which haven't. With this functionality, a byte stream is fragmented into parts of 126B. Aside from the reduced scale of data, this mechanism differs in its synchronous nature from its high-data rate counterpart, limited by its half-duplex channel.

Apart from the functional messages exchanged, the spacecraft will periodically downlink spacecraft system telemetry as a beacon decodable by amateur radio enthusiasts around the world. The content of these messages consists of telemetric information such as battery levels, signal strength of received frames, and Sun sensor measurements. The message format will be publicly available.

2.3 On-Board Computer (OBC)

As the brain of the spacecraft, the On-board Computer shall perform tasks critical to the successful operation of the spacecraft. Such tasks include the spacecraft housekeeping, the implementation of the required communication stack, the gathering and storage of scientific and housekeeping telemetry, and the ADCS computations in order to achieve the required spacecraft attitude. Gaining from the past experience and reliability from available real-time operating system (RTOS), the OBC software is built on top of the Azure's ThreadX and NASA's F' (F Prime) framework. While the ThreadX RTOS provides the hardware-software interaction layer and the real-time core functions, F' provides a space-proven software framework that abstracts the logic modules' implementation from the base RTOS and provides the basic building blocks for the construction of the flight software (e.g., command storage, command dispatching). Furthermore, F' provides a space-certified framework that ensures a reliable operation throughout the mission.

To attain these tasks, the OBC relies on a processing unit with dual ARM Cortex-M cores, delivering a reduced power consumption and a set of communication peripherals for the interconnection with the remaining spacecraft subsystems. Connected to this processing unit lies a memory unit comprising different types of non-volatile memory, with small sized radiation tolerant memories to be used primarily for storage of critical data and configurations to be deployed on the remaining subsystems. On the other hand, memories with higher storage capacity but more prone to radiation effects are designed for temporary payload data storage prior to transmission to the GS.

Being based on the PC/104 specification, the interconnection with the remaining spacecraft subsystems is mainly performed through the corresponding stack connector. A single inter-integrated circuit (I^2C) interface is shared for communication with the Electric Power System (EPS) module, for the subsystem comprising both magnetometer and magnetorquers, and for the UHF antenna deployment mechanism. The interaction with the Communications subsystems is performed with two distinct universal asynchronous receiver-transmitter (UART) transistor-transistor logic (TTL) interfaces, one of them relying on a RS485 differential bus for faster throughput. The spacecraft gyroscope and GNSS rely on a serial peripheral interface (SPI) and UART interfaces, respectively, to provide data to the OBC. Finally, a controller area network (CAN) interface for control and processed data transfer plus an SPI interface for image transfer is used for the Star Tracker.

Three sets of additional connectors are also implemented on the OBC, for communication with the fine sun sensors and reactions wheels that share a I^2C interface, and for interconnection with the payload accelerometer that relies on an SPI interface for configuration and data transfer.

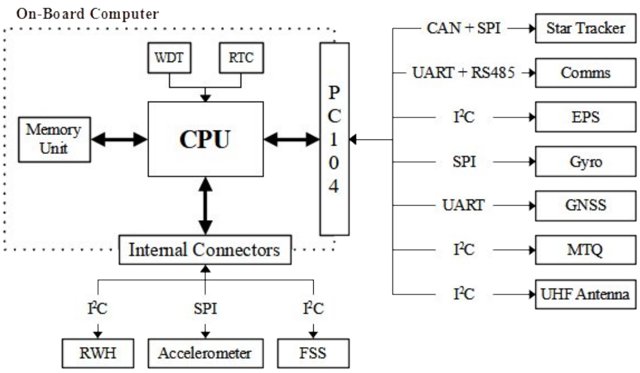


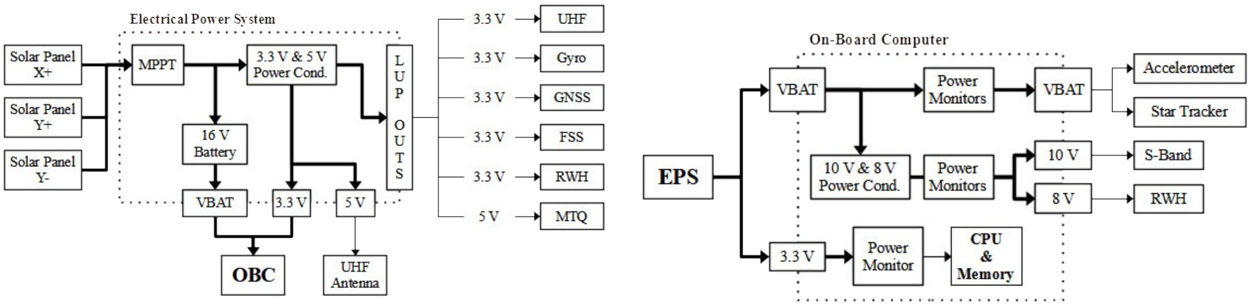
Figure 5. High-level hardware architecture of the On-board Computer module

2.4 Power Supply

The spacecraft power architecture is performed by both EPS and OBC modules to deliver the required power supplies to the remaining subsystems.

On the EPS side, three distinct solar panels shall be used on the X+, Y+ and Y- panels of the 6U satellite (cf. Figure 9), which shall deliver the acquired power to three photovoltaic power converters tuned for maximum power point tracking. The resultant power shall charge the spacecraft battery pack, which is directly routed to the PC/104 connector and used as input for two different power regulators. These regulators deliver both 3.3 V and 5 V power supplies, which are directly routed to the PC/104 connector and to six independent output switches with a configurable latch-up current limiter.

Such latch-up protected outputs are used to power most of the spacecraft subsystems to prevent damage to the electronics due to high current conditions caused by single event effects (SEE), their correspondent current limit tuned to the specified maximum current consumption of each subsystem. Figure 6 (a) shows the architecture used to deliver power from the EPS to the different devices.



(a) (b)

Figure 6. Power architecture and delivery performed by the EPS module (a) and the OBC subsystem (b)

On the OBC side, the main 3.3 V power supply is used to power its processing and memory units, with a dedicated power monitor to protect such electronics against latch-up currents as performed on the EPS. Additionally, the direct battery voltage is used for additional power delivery and regulation performed on the OBC, being delivered to the payload accelerometer and star tracker, and to two distinct regulators delivering 8 V and 10 V power supplies to the reaction wheels and S-band transmitter, respectively (cf. Figure 6 (b)). Nevertheless, all power delivery performed by the OBC is protected with power monitors so that the corresponding subsystems are shut down by command or automatically due to high current conditions.

For LEO orbits between 450 km to 600 km, eclipses can last up to 40 minutes, causing periodic charge-discharge cycles on the EPS buses. The EPS battery design/selection ensures enough capacity to power the spacecraft in all possible configurations during eclipse periods. Based on the maximum power consumption of each on-board system, in eclipse, the spacecraft may consume up to 14W. In the remainder of the orbit, the battery recharges with the surplus power provided by the solar panels, amounting to 22W. Under the expected mission conditions, this charge cycle will be able to ensure normal scientific operations for 5.5 hours (3 orbits) until the battery requires recharging.

2.5 Main Frame

The uPGRADE spacecraft main structure is compliant with the 6U CubeSat classification for nanosatellites, with 366×226.3×100 mm overall dimensions (these dimensions do not account for additional protrusions from the side panels). The bare mainframe weighs 900 g, and is made of aluminium.

The structure is composed of two panels, top and bottom (X+/X-, respectively), which are joined by two lateral ribs that bolt onto these panels. The lateral ribs support the solar panels installed on the Y+/Y- sides of the spacecraft. The ends Z+/Z- are capped by two additional panels, which are bolted onto the top, bottom, and lateral structures. These end caps accommodate the communication antennas. There are holes on both the bottom (X-) and end (Z-) panels to clear the field of view of the star trackers housed on the inside of the structure. In Figure 8, the spacecraft structure can be seen, along with its principal axis.

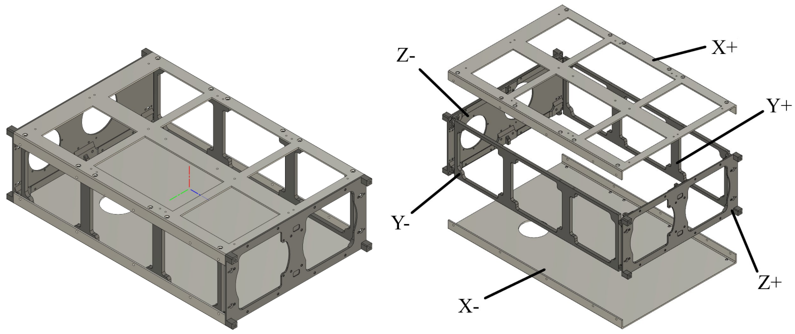


Figure 8. The main frame of the uPGRADE spacecraft, assembled and exploded view

The design of the main frame is centred around the constraints and requirements imposed by the launchers, with particular attention to the required verification testing of the structure. Compliance with such procedures drives the design, which aims to minimize weight, manage temperature cycles, optimize packaging and flexibility, and provide unobstructed access to the interior.

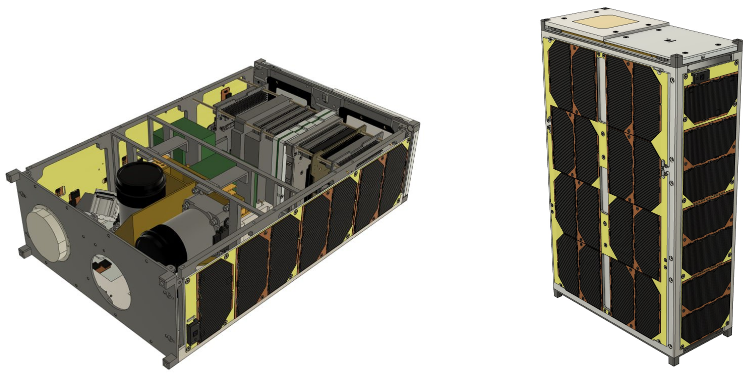


Figure 9. Full assembly of the uPGRADE spacecraft

3 THE PAYLOAD

The uPGRADE mission has a scientific goal of gathering gravimetric data to study geophysical mass transport processes and the neutral thermosphere. To that end, the proposed payload consists of a set of instruments composed of a three-axis accelerometer – yielding a measurement of the non-gravitational acceleration for a given position and attitude – a GNSS receiver, and a set of star trackers – yielding the position and attitude. The attitude is necessary to orient the accelerometer in inertial space and to locate the position of the GNSS antenna relative to the Centre of Mass (CoM) of the satellite. The GNSS positions, once double-differentiated, provide information on the total accelerations, i.e., those of gravitational and non-gravitational origin.

3.1 High-Resolution Accelerometer

Current space accelerometers are based on the electrostatic principle, where a rectangular proof mass with cm-size (4 × 4 × 1 cm and a 72 grams weight, made in Chromium coated Titanium alloy in case of GRACE's accelerometers) is encased in a cavity with electrodes on its sides. Any accelerations acting on the instrument induces the movement of the proof mass towards one side of the cavity, changing the measured capacitance and triggering an increase of the electrode voltage to create a restoring electromagnetic force that keeps the proof mass centred.

Gaining from the consortium extensive experience designing MEMS accelerometers, a novel high-resolution MEMS accelerometer is designed based on pull-in time measurement. Although the need for a front-end readout to detect the pull-in occurrence (in this case a capacitive one, although others could be used such as piezoresistive [10] or magnetic [11]), the acceleration data is not translated directly from this measurement; instead it is transduced from the pull-in time. This makes the front-end readout requirements much more relaxed and allows for an extremely high precision time measurement.

Considering that a time-based transduction mechanism is used, the readout circuit is implemented in the analogue domain, since it is not bounded by an ADC sampling frequency or any clocks limits. Furthermore, usage of active components is minimized as much as possible due to their own added 1/f noise to the system which is higher than that of passive components. Several approaches were tested for modulation and demodulation processes. In the end, a charge amplifier-based instrumentation along with synchronous demodulation has provided the best result from the performance and accuracy points of view.

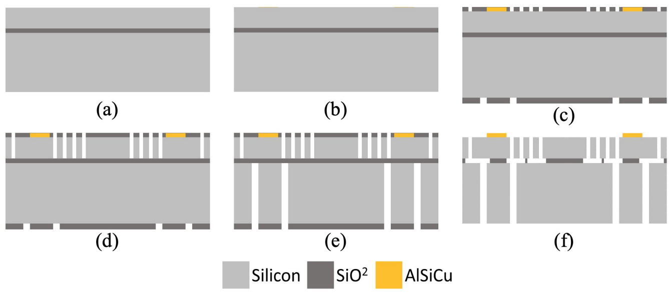


Figure 11. Accelerometer SOI-based fabrication process: (a) wafer preparation; (b) metal layer patterning; (c) FS and BS SiO2 hard mask patterning; (d) FS DRIE; (e) Tape application and BS DRIE; (f) HF structure release

Additionally, in order to maximize acceleration sensitivity, the MEMS design focused on increasing both spring compliance and proof-mass weight, while tuning the damping coefficient, critical in pull-in time-based operation principles; this results in a beneficial trade-off between mechanical-thermal noise and sensitivity [9]. The highly compliant springs have been achieved by exploring fabrication limits (at the limit between manufacturability/yield and compliance maximization), resulting in a design with six folded-springs suspending the heavy proof-mass (170mg, obtained by leaving a portion of the handle wafer attached to the movable structures – Figure 11). Dummy dampers have been implemented in the device layer to lower the quality factor and maximize pull-in time sensitivity. In Table 2, the main design parameters of the fabricated devices are presented: two different types of devices have been designed, with a different number of dampers.

Table 2. Microstructure's parameters

Parameter	Theoretical		Experimental	
	S1	S2	S1	S2
Mass, m [mg]	170		174	177
Resonance Freq., Fn [Hz]	21.15		11.8	14.9
Spring stiffness, k [N/m]	3.00		2.64	
Initial gap, d0 [µm]	3.00		2.2	

Besides the performance specifications, space applications also require that the MEMS structure sustains large accelerations during launch and detachment from the upper stage. For the sake of increasing the robustness of the structures, the designs included both the conventional in-plane mechanical stoppers at 2.9µm (full gap: 3µm) as well as cantilever-like ones at 2.8µm to gradually absorb impact (not shown). Additionally, out-of-plane displacement stoppers were implemented by fabricating encapsulation structures, as shown in the process diagram of Figure 12.

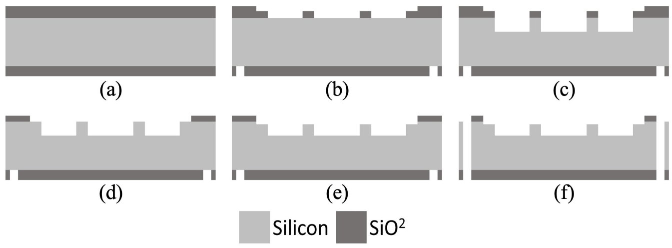


Figure 12. Out-plane stopper cap fabrication process: (a) FS and BS SiO2 deposition on SSP wafer; (b) FS and BS hard-mask patterning; (c) FS 1st DRIE (19 µm); (d) FS hard-mask thinning DRIE; (e) FS 2nd DRIE (1 µm); and (f) BS DRIE and cap release

Two different caps have been fabricated: one for the device bottom part that must accommodate the out-of-plane displacement due to Earth gravity (not present in the final application, it is necessary to consider during the device characterization). Moreover, a top side cap that limits the device displacement, preventing the collision of the proof-mass with the dummy dampers in the device layer. The encapsulation scheme and the final device are depicted in Figure 13 [13].

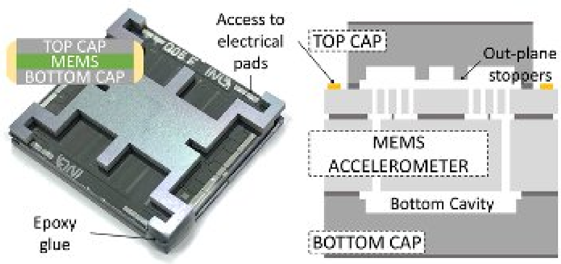


Figure 13. Device encapsulation (top and bottom caps)

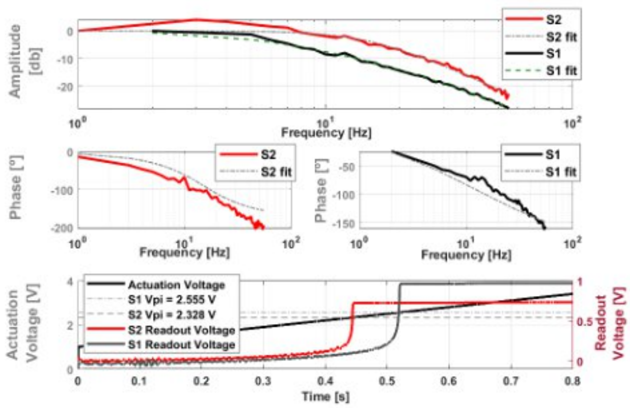


Figure 14. Experimental preliminary characterization of a fabricated S2 device

The experimental setup for the MEMS development comprises a charge amplifier circuit for capacitive displacement readout and a data acquisition board for actuation signal generation. The device is mounted on a high-resolution 6-axis stage from ThorLabs. Both devices (S1 and S2) were experimentally characterized for natural frequency (Fn), quality factor (Q) and pull-in voltage (VPI), cf. Figure 14. The fabricated devices showed a silicon over-etch (evaluated by SEM inspection) smaller than expected in the backside and electrodes regions (larger proof-mass and higher damping) and larger in the springs (lower stiffness). This resulted in lower Fn and Q (estimations extracted from measurements, cf., Table 2). Device S2 was used to validate pull-in operation and achieved a pull-in voltage sensitivity of 218 V/g [13]. Pull-in time sensitivity, noise level and long-term stability are currently under investigation.

3.2 Star Tracker

The Star Tracker unit is developed by Spin.Works from a natural progression of the in-house knowledge in the development of space qualified cameras and with the objective to produce a commercial sensor able to provide attitude estimates with arc-second accuracy.

The Star Tracker optics is based on a COTS 50 mm lens that is adapted to withstand the environment of space. In particular, venting holes were drilled and thread locking mechanisms were added. The structure was designed in aluminium 7075-T6, black anodized to reduce stray-light. A thermal analysis revealed the need to couple the sensor, through a thermal pad, with the surrounding structure to ensure a proper operating temperature.

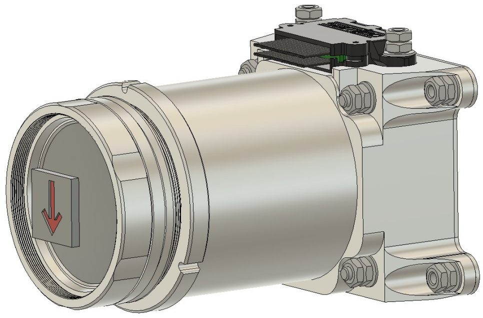


Figure 15. Star tracker unit (3D rendering)

The underlying software estimates the star tracker inertial attitude employing the QUaternion ESTimation Method (QUEST) algorithm [7] that uses a Newton-Raphson two-step iteration to easily converge towards a good estimation for the pose values. This algorithm offers an almost real-time convergence to Wahba’s optimization problem.

Overall, the Star Tracker algorithm can be decomposed into four main sub-routines:

i) image processing, where low exposed images of the dark sky are filtered by convolving the raw frame by a Gaussian kernel to enhance the visibility of the faint stars in the image. From the filtered image, a first pass of the connected components algorithm [3] builds a list of coordinates of the centre of the different blobs detected in the image;

ii) star centroid estimation: in this step the rough centroid estimate (usually in a sub-pixel scale) resulting from the connected components analysis is further refined employing a 2D Gaussian fitting algorithm, as in [5];

iii) star identification, this step builds upon the Multi-Poles Algorithm [6], a highly promising star identification method that not only provides a fast star catalogue searching but also some effective proofing layers against false objects or even highly corrupted environments; and

iv) attitude estimation which employs the aforementioned QUEST algorithm to return a fully defined inertial attitude estimate with accuracies on the order of a few arc-second.

3.3 Payload Harness

The payload harness is the structure designed to accommodate the instruments, both the accelerometer and the two Star Trackers. It is housed inside the main structure and needs to be attached to the frame, complying with structural and functioning requirements. It is divided into two sub-systems, each with its own design goals. These two sub-systems are 1) the rigid mounting structure that holds the star trackers and the accelerometer, and 2) the flexures that connect the aforementioned rigid structure to the spacecraft's main frame.

The rigid mounting structure needs to be as stiff as possible, allowing little deflection between the three orthogonal mounting surfaces. Since the instruments are directly attached to it, this structure also defines the position of the sensors relative to the satellite's coordinate system. This is an important part of the design since there are strict requirements regarding the spacecraft's centre of gravity position, constraints imposed by both the launcher service, and the scientific purpose of the satellite. With these factors in mind, the design of the rigid structure is centred around efficient packaging of the instruments, minimizing weight, maximizing stiffness, and providing thermal routing/managing of the payload. The rigid mounting structure can be seen in Figure 10, highlighted in orange.

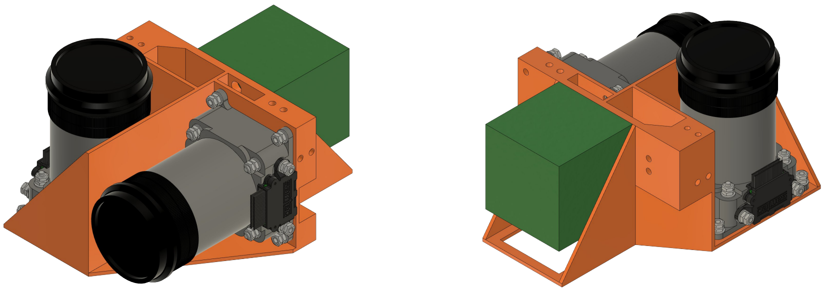


Figure 16. The rigid mounting structure, in orange, and the mounted sensors

To mitigate the vibrational and displacing influence on the payload harness, six flexures suspend the structure, decoupling it from the deflections caused by the expansion/shrinking of the main frame and other structural vibrations. These also act as radiative devices, due to the large area/volume ratio, further maximizing the thermal decoupling of these devices. In Figure 11, the six flexures can be seen highlighted in yellow.

The design goal of the flexures is to have high rigidity in one of its axes, and high flexibility in the remaining two. This characteristic allows them to resist the launch environment, isolating the payload from the main frame's resonance frequencies, and protecting the delicate instruments of the payload. The flexibility in the remaining axis permits the flexures to bend and deform when the main frame shrinks/expands, mitigating the influence on the harness structure, and allowing it to comply with the orthogonality requirements. Another aspect to take into consideration is that the deformation of these flexures causes a slight deviation of the satellite's centre of gravity, which can contaminate the readings of the sensors. To account for this, the design of the blades was iterated to find the right trade-off between, axial rigidity, weight, length, and flexibility.

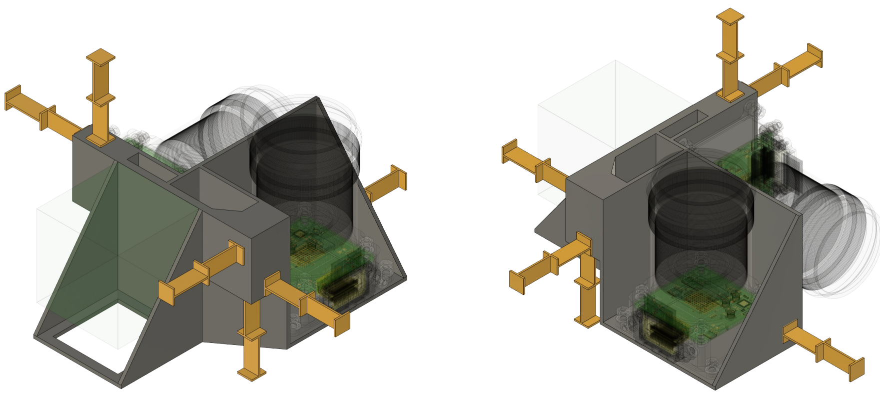


Figure 17. The flexures that connect the rigid mounting structure to the main frame

4 CONCLUSION

Previous missions designed to monitor the temporal variations of Earth's gravity field and measuring the neutral thermospheric densities are characterized by substantial investments and development costs. The uPGRADE mission develops a novel spacecraft and payload system that reduces these costs by leveraging the CubeSats' small size and weight and a novel design for a MEMS high-resolution accelerometer.

To solve the miniaturization challenges of high-resolution accelerometers, the uPGRADE mission employs MEMS technology and increases the overall resolution of the accelerometer by measuring the pull-in time instead of pull-in voltage or capacitance. A special effort is made in the reduction of measurement noise by using active components and the overall sensitivity is enhanced by increasing the MEMS spring compliance and proof-mass weight, while tuning the damping coefficient.

On the spacecraft platform, emphasis is given to the development of a reliable system able to keep perturbations to a minimum while transmitting large volumes of scientific data to the ground. Magnetorquers are used for attitude control during the vibration-sensitive phases of the mission. Additionally, reaction wheels are included to ensure fine attitude control during high data rate communications through S-band.

A payload harness is specifically developed to hold the high-resolution accelerometer and two Star Trackers ensuring these systems are rigidly coupled and that thermal deformations are minimized. A novel Star Tracker is developed as part of uPGRADE's technological goals, providing arc-second attitude knowledge crucial for the scientific objectives of the mission, and to the ADCS.

The uPGRADE mission is currently finalizing the critical design review and starting the verification and implementation phase. This phase will last for the remainder of 2022. In 2023, the qualification tests will take place leading to the final acceptance of the uPGRADE's spacecraft.

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