

CubeSIM payload for ESD detection and mitigation on small satellites

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Abstract:

The CROCUS mission will embark the CubeSIM payload on a CubeSat platform to assess the electrical interactions between the space environment and the satellites in low Earth polar orbit. A series of instruments will provide information to better understand how the satellite charges when subject to auroral arcs. The characteristics of the electrostatic discharges will be measured and a mitigation technique will be tested. In this paper, we present the status of the CubeSIM payload development.

1 Introduction

Pending on their orbit, on the space weather and on the geomagnetic activities, the satellites orbiting the Earth can face severe electrostatic charging conditions. Eventually, the deposited charges can produce high potential differences between adjacent materials, especially insulators, and ultimately generate spurious electrostatic discharges (ESD). These discharges are responsible for material degradation and electromagnetic interferences responsible for anomalies and losses. Spacecraft engineers handle this risk by applying flight heritage and the rules emanating from the international design guidelines (ECSS, NASA, ISO, etc). Nonetheless, a few tens of percents of flight anomalies are still attributed to ESDs ([1], [2], [3]).

Nanosatellites have become very common in the last decades to achieve a large variety of missions (in-orbit demonstration, science, commercial service). The guidelines related to ESDs and related electromagnetic coupling risks, initially developed for large platforms, must be adapted to the specificities of smaller platforms including CubeSats. Several factors decrease the ESD risks on nanosats surfaces. Their orbits - mostly Low Earth Orbit (LEO) - are less constraining than geosynchronous (GEO) and Middle Earth Orbits (MEO) in terms of energetic electron fluxes. The amplitude and the duration of the flash-over are reduced due to smaller solar panels. Several factors increase the surface ESD risks on nanosats: the proximity of electronics sensitive to ESD transients and EMC coupling on harnesses/components, and a relative lack of awareness of nanosats and CubeSats designers. There is thus a need to improve our knowledge on the conditions leading to ESDs and of their effects on small platforms.

Only the simultaneous measurements of hazardous surface potentials and corroborating frame potentials with the occurrence of a discharge provide reliable information that the natural space environment caused an anomaly that can be attributed to surface charging. The advent of nanosatellites and constellations provides an excellent opportunity to fly a large number of instruments on different orbits. One critical aspect for their acceptance by the space industry is to come up with a solution with limited impact on the satellite budget (volume, mass, power). There is

thus a need to develop and test new generations of miniaturized instruments. Another important aspect of the present R&D activity is to come up with efficient mitigation techniques.

The ChaRging On CubeSat (CROCUS) mission project initiated by ONERA and Centre Spatial de l'Ecole Polytechnique (CSEP) is a cornerstone for the development and the flight demonstration of new generations of dedicated instruments. The CubeSIM payload (Sensing Impulses and Mitigation on Cubesat) that will be embarked in CROCUS consists in a set of instruments under development at ONERA. Its modularity offers a variety of configurations from standalone to fully integrated instruments to fit any mission profile. The objective of this paper is to present the status of the CubeSIM development.

Section 2 shows an overview of the CROCUS mission. Section 3 introduces the CubeSIM payload. Section 4 presents the test setup. Section 5 is the summary of the planned activities.

2 CROCUS Mission Overview

2.1 Applications

The main science applications includes space weather and geomagnetic activity effects, auroral arcs and space plasma matter interaction. The in-orbit demonstrations of miniaturized instruments include the following functions: measure electrostatic charging, detect and measure ESDs waveforms, mitigate charging issues, evaluate erosion by atomic oxygen. The educational and outreach objectives include satellite development, payload development, software development, training material, internships.

2.2 Motivations

According to [2], a few tens of percents of flight anomalies attributed to the space environment are related to charging. According to [3], 60% of anomalies in LEO are strongly correlated with 30-100 keV electrons on NOAA-15 (Polar Earth Orbit), Kp and Dst geomagnetic perturbations, and 65 % anomalies occur in dusk-to-dawn sector. ESDs are the result of competing space environment effects. On one hand, high fluxes of medium energy electrons ($4 \text{ keV} < E < 100 \text{ keV}$) tend to produce spacecraft negative charging. These electrons are accelerated along the magnetic field lines during geomagnetic substorms. They are observed in GEO and MEO around the equatorial plane. They are also observed in LEO in the auroral zone. Typical density and temperature are $1 \text{ to } 10 \text{ cm}^{-3}$ and $5\text{-}15 \text{ keV}$, respectively. The collection of these energetic electrons can be considered as isotropic. On the other hand, in LEO, the upper ionosphere composed of high density of cold plasma reduces spacecraft charging. Typical densities range from a few hundreds up to 10^6 cm^{-3} , with the lowest values observed in the night sector during solar minimum activity. The temperature of this cold plasma is a fraction of eV. The collection of these particles is controlled by the spacecraft velocity and by the electrostatic sheath around the object. The ram and wake sides collect different fluxes of ions. Finally, vacuum ultra-violet (VUV) photons are responsible for the emission of low energy electrons (about $1 \text{ to } 10 \text{ eV}$). They very significantly reduce the satellites negative potential because the emitted fluxes are in the order of a few nA/cm^2 which is generally larger than energetic electron fluxes (about 0.1 nA/cm^2), except during very severe geomagnetic substorm in GEO.

The floating potential of a spacecraft depends as well on the electrical properties of the covering materials, especially the total electron emission yield (TEEY) and the electrical conductivity. The absolute and differential charging is governed by insulators properties and by complex 3D particle trajectories influenced by the spacecraft potential itself. Generally speaking, a fully conductive spacecraft should not charge to potential more negative than a few tens of volts negative in sunlight

due to photoemission. But a spacecraft covered with some insulators can charge down to much larger negative potentials since the insulators located in the sunlit surfaces partly prevent the photoemission to discharge the spacecraft structure. A spacecraft in Earth shadow is at risk since only energetic electrons control its potential. Figure 1 presents the fundamental collected and emitted currents responsible for spacecraft charging.

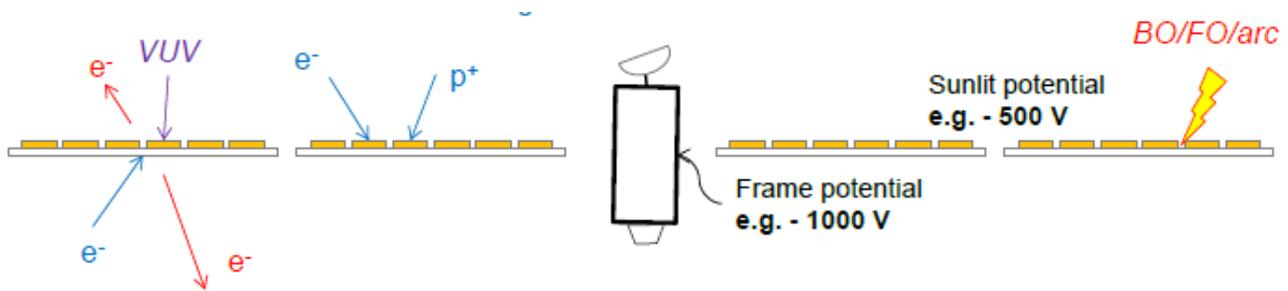


Figure 1 - Fundamental interactions between a spacecraft and its environment. The spacecraft geometry is representative of a telecom spacecraft but the same interactions apply for any type of satellite. The potential levels are illustrative of a spacecraft at GEO/MEO/LEO auroral.

The most risky situation is the so-called inverted potential gradient (IPG) when the spacecraft frame is negative and covered with less negative insulators. An IPG level of about 300 V, meaning insulators getting charged to +300 V wrt to the spacecraft frame, was enough to trigger ESDs on solar cells edges during the Horyu-II mission on a small sat flying on SSO orbit at 680 km [4]. Such discharges occur at the junction between the conductive parts, the insulators and vacuum, also called triple point. A discharge is composed of the blow-off (BO) current which reduces the absolute spacecraft voltage (from a few hundreds negative volts to nearly zero) and of the flash-over (FO) current which reduces the differential charging of insulators located near the trigger point. Because they offer a large number of triple points in IPG situation, solar panels are subject to ESD triggering.

To our knowledge, the IPG situation is merely instrumented and if so, on a very few spacecraft. The ESD voltage threshold in IPG situation is commonly deduced from ground tests but have never been measured in flight. In addition, it is hardly feasible to get the location and waveform of ESDs. The ground tests usually provides material properties and the numerical simulations evaluate satellite charging in severe and worst-case environmental conditions. In the frame of a nanosatellite or CubeSat development, the reduced size of the spacecraft makes it possible to perform full scale test in ground facilities. That can significantly improve how the spacecraft are tested against the ESD risks. It can provide valuable comparison to measurements performed on-board a nanosatellite in its orbit, as well. Quite surprisingly, however, very few spacecraft are equipped with charging and ESD detection instrumentation. This is the main motivation of this project.

2.3 Science Objectives

The objective of the CROCUS mission is to demonstrate the ability of our science with a miniaturized payload hosted by a 2U-cubeat. The science objective is to correlate ESDs with on-board data and with other space and ground measurements. It includes the following activities: detect the occurrence of ESDs, identify the charging conditions, simulate the charging conditions, identify the space weather and geomagnetic indices and reduce charging levels.

2.4 Orbit Definition

The region of interest (ROI) for the mission is constituted by the auroral ovals located in the night sector. For a CubeSat mission in LEO, Sun-Synchronous Orbits (SSO) offer 15 to 16 auroral ovals crossing every day (geographic latitudes between 65 and 75°). The longitude of the ascending node

(LTAN) controls when (seasons) and where (geographical latitude) the satellite crosses the auroral zones in the night sector, see Figure 2. The extrema orbit cases are:

- Dawn-to-dusk orbit with LTAN 06:00 : Charging will occur mostly in the summer for the southern hemisphere. Charging in the northern hemisphere will be very rare, even during winter, due to the inclination of the orbit tilted toward the Sun in the northern hemisphere.
- Dawn-to-dusk orbit with LTAN 18:00 : Charging will occur mostly in the winter for the northern hemisphere. Charging in the southern hemisphere will be very rare, even during summer, due to the inclination of the orbit tilted toward the Sun in the southern hemisphere.
- Midnight-to-noon orbit with LTAN 00:00 : Charging will occur mostly in the winter for the northern hemisphere and in the summer for the southern hemisphere.
- Midnight-to-noon orbit with LTAN 12:00 : same as LTAN 00:00.

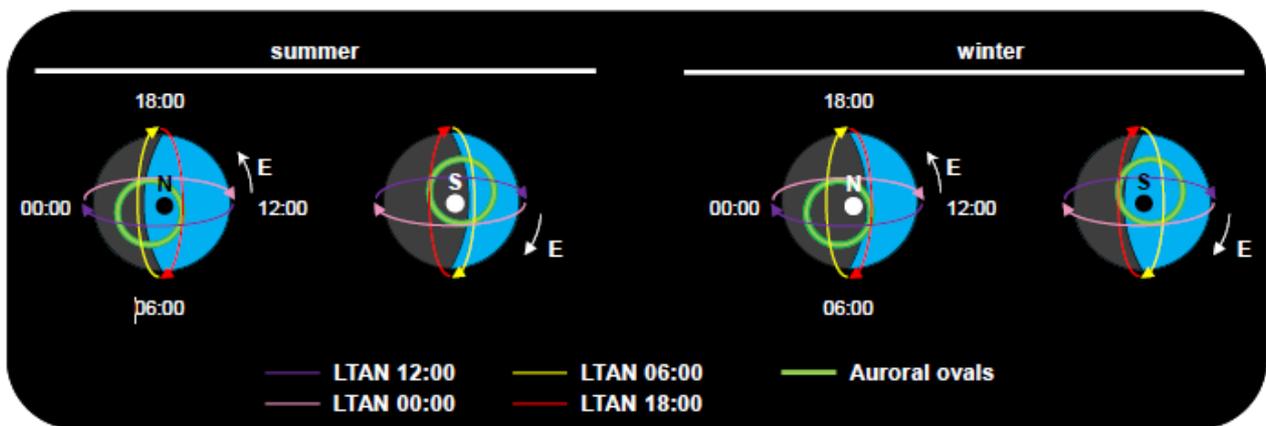


Figure 2 - Schematic representation of four typical SSO orbits and seasonal effects on ROI crossings (auroral ovals in the night sector). The dayside is shown in blue. The night side is shown in grey

The auroral arcs thickness in the ionosphere is typically several or even tens of kilometers [5]. Charging events lasting more than 10 seconds are induced by auroral arcs locally oriented in the same direction as the spacecraft velocity. Because of the tilt between the Earth rotation and magnetic axes and because of their incidence ($\sim 97-99^\circ$), SSO orbits are very good candidates. Therefore, any 90-minutes period SSO orbit crosses at least 3000 times the ROI every year. It increases up to 6000 for LTAN 00:00 and LTAN 12:00.

2.5 Organization

The development of the CubeSIM payload, currently in phase C, is internally funded by ONERA. The CROCUS mission is led by ONERA in partnership with Centre Spatial de l'Ecole Polytechnique (CSEP). ONERA and CSEP studies have jointly conducted phase 0/A/B- studies. ONERA is now conducting CROCUS Phase B/C/D activities. The activities planned until the end of 2022 include the development and the test of an engineering model of CubeSIM mounted on a structural model of a 2U platform.

3 CubeSIM Payload

3.1 Functions

The occurrence of ESDs will increase when the spacecraft gets negatively charged with respect to its environment. Negative charging will be achieved by flying the spacecraft in the ROI. The CubeSIM payload will include several functions listed in this paragraph. The core functions are the ESD

detection with ANT, PEA and TWIST; and the spacecraft negative charging mitigation with SCAPEE.

3.1.1 BEC

Negative charging will be increased inside the ROI by the Big Electron Collector (BEC). BEC is a conducting plate covered by a conductive material with a low TEEY. In standby (STBY) mode it will be electrically connected to the structure. BEC is designed to increase the negative current collection on the spacecraft frame. In its active mode, BEC will be connected to the positive end of a high voltage supply. The negative end will be connected to the spacecraft frame. The potential of the spacecraft frame will get negative under the collection of thermal electrons by the positive end of the voltage supply. Ideally, BEC will be located on a deployable panel to optimize electron collection.

3.1.2 MISTEEC

The MicroStructured Electrostatic Event Creator (MISTEEC) will be composed of a layout made of conducting and insulating materials forming triple points that will increase the probability to trigger ESDs. This layout will generate large IPG levels. The conductor will be connected to the frame. The insulators will have a large TEEY and a large electrical resistivity. Under the impact of keV auroral electrons, the grounded conductor will get negative and the insulator will get much less negative than the conductor. This configuration will be optimal for triggering ESDs. Ideally, MISTEEC will be located on a deployable panel to optimize its differential charging.

3.1.3 SPARK

The Simulated Primary ARc Kicker Emitter (SPARK-E) will mimic the waveform of ESDs. It will calibrate ESD detectors in-flight. The Simulated Primary ARc Kicker Receiver (SPARK-R) will capture the transient currents generated on a printed circuit board (PCB) by SPARK-E.

3.1.4 ESD Sensors

Pearson current probes (PEA) will detect transient electrical currents on harnesses connecting equipment to each other. Antenna monopoles (ANT) will be deployed. They will detect the transient electric fields generated by ESDs.

3.1.5 TWIST

The Track Weak Inrush Signal and Transient (TWIST) function will digitalize the analog signals with a bandwidth of 10 MHz or more. The numerical code will be deployed on FPGA.

3.1.6 CPA

A charging plate assembly (CPA) is based on flight-proven techniques ([6]-[7]). It will monitor the differential charging of an insulator on the surface of the satellite. CPA will detect both normal and inverted potential gradient but the priority will be put on IPG level measurements. CPA will provide a continuous information on spacecraft charging during both passive and active instrument operations to address spacecraft charging interactions with the auroral arcs (passive mode) and with the dense plasma (BEC active mode with an artificially charged spacecraft).

3.1.7 SCAPEE

The Spacecraft Charging Alleviation by Passive Electron Emission (SCAPEE) reduces the negative spacecraft charging when activated.

3.1.8 RESISTACK

In option, the RESISTACK atomic oxygen detector is also embedded on the CubeSIM payload.

3.2 Design

The computer-aided design (CAD) of the CubeSIM payload is presented in Figure 3. Its size is 0.7 U. In that assembly, CubeSIM is mounted on a 2U CubeSat platform (PF). The BEC and MISTEEC functions are mounted on additional 200 x 80 x 2 mm deployable panels. The angle between the panels and the PF can be adapted from 90° to 135°. BEC is covered with conductive black polyimide. The MISTEEC panel is made of aluminum with 2 μm gold deposit. Eight alumina sheets are glued with EPO-TEK® conductive H20E on each side of the MISTEEC panel. Two ANT antennas are deployed perpendicularly to CubeSIM. CPA and SCAPEE are located on two 1U surfaces located at the front and at the rear (only the rear side is presented in Figure 3). The PEA, SPARK and TWIST functions are located inside the 0.7 U CubeSIM payload case.

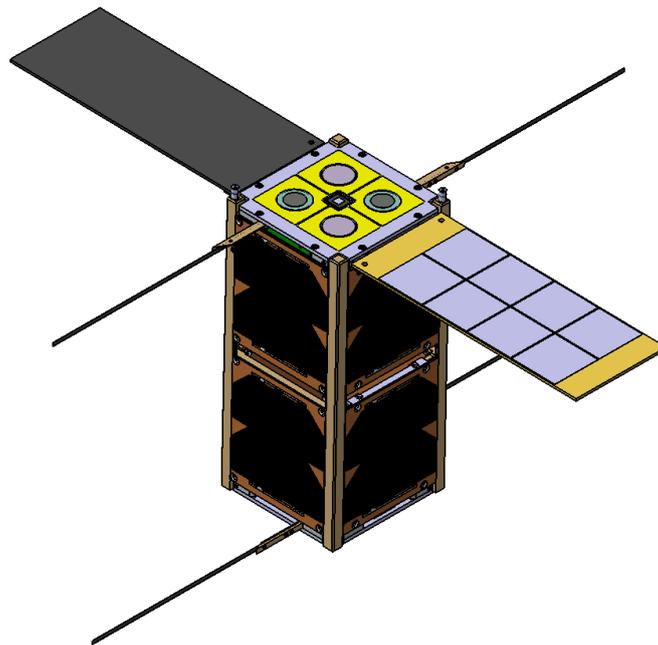


Figure 3 – CAD of the CubeSIM payload mounted on a 2U CubeSat platform

Figure 4 presents the CubeSIM electronic architecture. A FPGA controls and monitors the instruments and communicates with the on-board computer. Figure 5 is a CAD of the electronic boards stacked to one another.

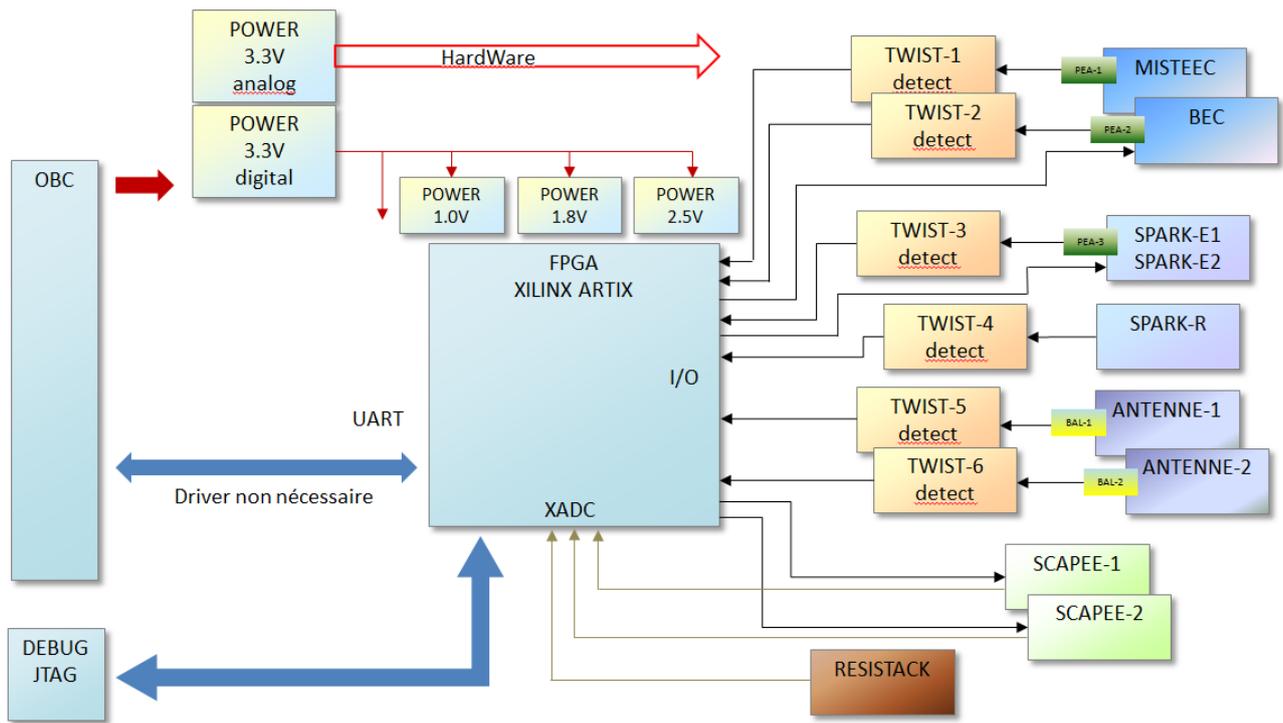


Figure 4 – CubeSIM Electronic architecture.

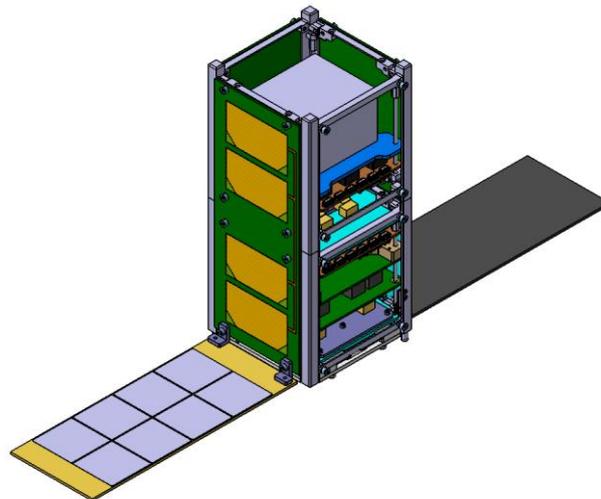


Figure 5 – CAD of the electronic boards stacked to one another on the PL EM and PF STM. From bottom to top: 1IU external face equipped with CPA and SCAPEE; board equipped with PEA; analog board; numerical board; electrical power supply; battery. ANT antennas are not represented.

4 AIV/AIT

The development of the CubeSIM payload (PL) is composed of an engineering model (PL EM) and of a protoflight model (PL PFM). The PL EM will be assembled, mounted and tested on a structural model of the platform (PF STM). The thermal vacuum, cycling and vibration tests will be performed at unit level and/or at model level. The PL PFM will be mounted on the PF PFM. The PL EM and the PF STM are currently under preparation.

4.1 PL EM / PF STM

The PF STM is composed of the frame of a 2U satellite covered with dummy solar panels. It is equipped with the PL EM and with an additional electrical ground support equipment (EGSE). This EGSE is connected to the same sensors as the CubeSIM PL. It is composed of three LabNation® oscilloscopes (2 channels each, 10 MHz). They are used to double check and calibrate the CubeSIM PL. The STM and all embedded electronics are electrically floating with respect to the ground. To do so, we use optic fiber for data exchange between the model and the laboratory acquisition system. The satellite mockup is powered with a battery controlled by an electrical power unit that delivers 3.3 V and 5 V lines. Figure 6 shows the CAD of the PF STM equipped with the CubeSIM EM and with EGSE. The total PF STM length is 50 cm.

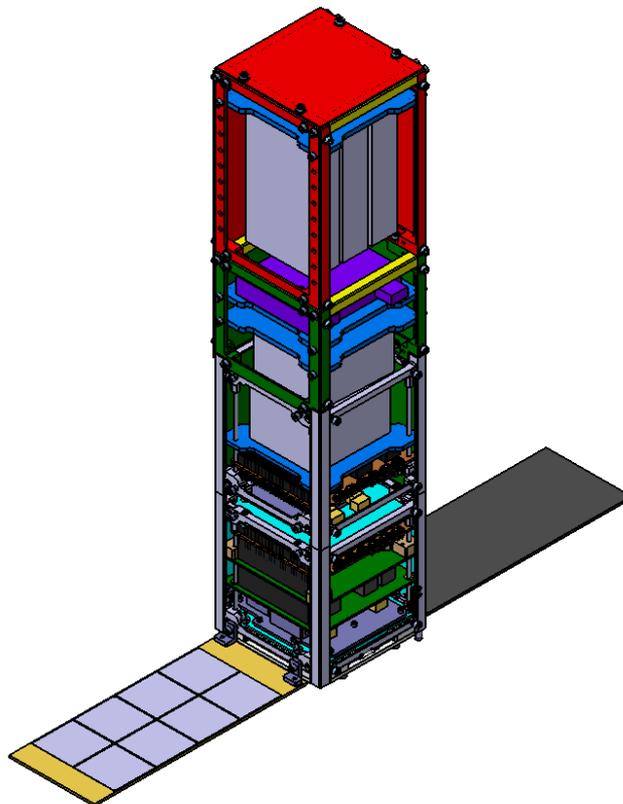


Figure 6 – CAD of the CubeSIM EM and of the additional EGSE mounted on the 2U PF STM.

4.2 Test Facilities

The JONAS chamber is located at ONERA Toulouse Center. It is a cylinder 3.4 m long and 1.85 m in diameter. It is equipped with an electron gun of energy up to 20 keV, a VUV source and a plasma source [8]. The CubeSat model is mounted at the center of the JONAS chamber with insulating nylon wires. The PF STM and the embedded electronics are thus entirely floating wrt the tank ground. This setup limits the electromagnetic disturbances of the test setup itself on the electrical signature of the ESDs [9]. Figure 7 is a schematical view of the facility. It includes a Faraday cup (FC), a VUV pach (PA) and a contactless voltage probe (KP). Figure 8 is a picture of a 2U-like mockup previously tested with this apparatus [9].

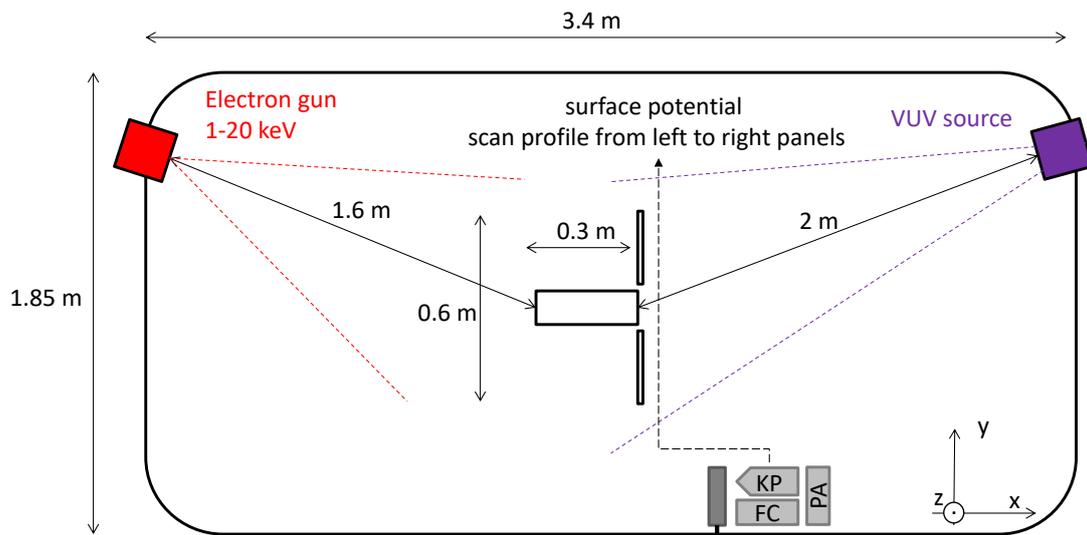


Figure 7 – Schematical overview of the PF STM installed in the JONAS chamber

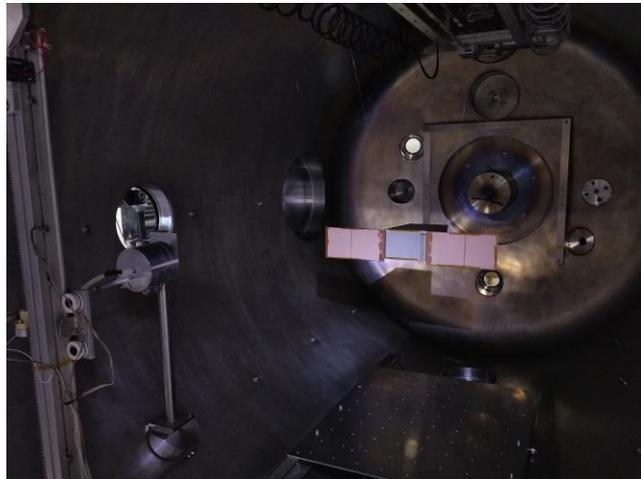


Figure 8 - Picture of 2U a cubesat mockup mounted inside the JONAS chamber

5 Summary

The CubeSIM Phase C will be reached by end of 2022 after the test of the EM in the JONAS chamber. Phase D (manufacturing and test of PL PFM) is expected to be reached by mid 2024. The platform phases C/D are expected to be reached by end 2024 in collaboration with a new partner. We plan to launch the CROCUS satellite in 2025. The satellite will pave the way to ESD sensors on small satellites. As shown in previous work ([9], [10]), the sensors will provide precise information on ESD waveforms with reasonable bandwidth (10MHz). They are easy to adapt to any nanosat or small sat with their specificities (size, materials, electrical circuit, environmental conditions).

6 Acknowledgments

The authors thank Pascal Chabert and Antoine Tavant from Centre Spatial de l'Ecole Polytechnique for supporting the CROCUS educational program through academic projects and internships funding. They also thank the student teams from Ecole Polytechnique involved in this project.

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