SSCAM: Micro-satellite platform for Earth observation

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Abstract: Thanks to recent technological advances in space component miniaturization, small spacecraft can assure a fast and affordable access to space, while providing a level of performance similar to larger ones. Nevertheless, only a few Earth observation micro-satellite have been demonstrated in orbit so far. This paper presents a small commercial platform concept tailored to the observation of the Tuscany Region (Italy) area. A preliminary mission analysis is performed and some potential applications are discussed. In particular, it is shown that several state-of-the-art imaging devices can be considered as the payload of the proposed platform.

1. INTRODUCTION

Recent technological advances in space component miniaturization pave the way to the successful diffusion of small platforms with reduced launch costs and short production cycles, providing a fast and affordable access to space. For this reason, spacecraft in the 10-500 kg class have become the true competitors to large spacecraft with over 1000 kg mass [1]. In fact, it has been claimed that 95% of the performance of large spacecraft can be achieved by smaller ones, incurring only 5% of the mission cost, and similarly a 70% performance can be achieved at 1% cost [2-3].

One ambitious objective of future small satellite missions is to provide low-cost remote sensing data products at unprecedented revisit rates (even constellations of small satellites can be conceived to assure a constant coverage of a given area), with a ground resolution of less than one meter. To achieve this goal, the integration of advanced Earth imaging instruments in the presence of severe mass and power constraints has been identified as a key challenge. Although the small satellite community is now pursuing a wide range of solutions, so far only a few Earth observation micro-satellites have been demonstrated in orbit, see e.g. the RapidEye and DMC constellations and the Proba family [4-6].

This paper presents the small satellite concept for Earth observation developed under the project SSCAM, funded by the Tuscany Region (Italy) POR CReO FESR 2007/2013. The reference mission and the spacecraft design are described, together with some applications for which the conceived platform may play an important role. A preliminary mission analysis, including the evaluation of the performance of the attitude and orbit control system, as well as an estimate of the power and mass budget of the spacecraft, is also discussed.

2. MISSION OVERVIEW

SSCAM is an ongoing small satellite project funded by the Tuscany Region, Italy. The project started in 2012 under the following partnership: SITAEL, Aerospazio Tecnologie, VVN, CRM Compositi, CNR-IFAC and the collaboration of the University of Siena and Polytechnic University of Milan.

Based on low mass, low power consumption and low cost architecture requirements, the SSCAM system concept consists of a single micro-satellite in Low Earth Orbit (LEO). A platform of this type is suitable for a wide range of Earth observation missions. The monitoring of glacier-melting, for instance, can reveal fundamental information about the climate change process, water availability and coastal modifications. Moreover, the observation of Earth vegetation changes can be useful for monitoring drought, deforestation and floods. Disaster monitoring, border surveillance and oil spill detection are only some of the other possible applications for which this platform could be used.

From the results of a trade-off analysis taking into account the available power, orbital lifetime and ground resolution of the spacecraft, a Sun-synchronous orbit with an altitude of approximately 470 km is considered for the mission. To enable frequent multispectral imaging of the sites of interest, a noon/midnight repeating ground-track orbit configuration with short revisit time is adopted. In particular, the selected configuration allows for one nighttime and one daytime passage over the Tuscany area every three days. The nominal ground-track of the orbit is depicted in Fig. 1. To guarantee fast and precise pointing capabilities, as required for advanced Earth observation, a three-axis attitude stabilization system is adopted. The expected mission lifetime is up to three years.

2.1 Spacecraft architecture

The spacecraft wet mass is envisaged in the 60 kg range and the bus size is assumed equal to 300x300x600 mm^3. The spacecraft is made up of the following modules:

- Payload bay: Module located in the lower part of the spacecraft, divided into four vertical compartments which contain two xenon tanks and the payload.
- Trays: Six trays superimposed on each other to form a single module located beside the payload bay, containing the tracking, telemetry and command, the onboard data handling, the attitude and orbit determination and control system (AODCS) and the power management subsystem.
- Side panels: Three solar panels installed on the sides of the spacecraft.

The satellite power subsystem is sized by considering solar panels that can generate up to 50 W each and Lithium-ion batteries able to manage the power requirements during eclipse periods. A schematic view of the spacecraft is reported in Fig. 2.

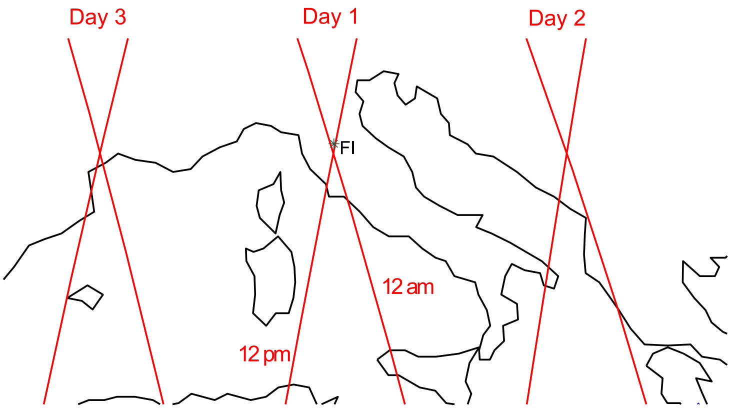

Figure 1. Nominal ground-track.

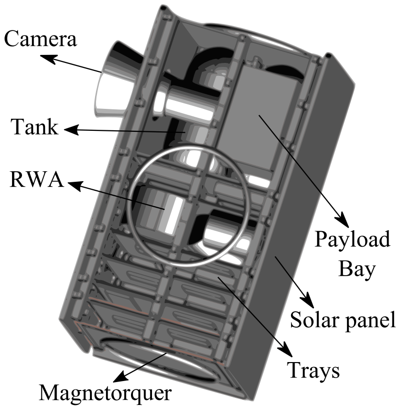

Figure 2. Spacecraft bus.

2.2 Payload

With the above figures, there are several possible state-of-the-art, space-qualified instruments that could be installed onboard the spacecraft. The Lightning and Sprite Imager (LSI), for instance, is a multi-mission consolidated equipment already flown on the Sprite-Sat spacecraft, which monitors luminous emissions in the upper atmosphere induced by lightning [7]. It consists of a wide-band and a narrow-band CMOS cameras operating at different wavelengths, weighs 640 g and requires less than 10 W of power when imaging. The CHRIS Imager is a hyperspectral system which has already been used onboard the Proba-1 spacecraft to collect bidirectional reflectance distribution function data [8]. CHRIS provides up to 63 spectral bands in the visible and near-infrared range at a ground sampling distance below 34 m, weighs approximately 14 kg and has a power consumption of about 10 W. Another possible option is the CIRC camera, an infrared camera installed on the ALOS-2 and JEM-CALET spacecraft and used for wildfire detection [9]. CIRC is equipped with an uncooled infrared array detector and usually mounted at an off-nadir angle of 30º. It weighs about 3 kg and has a peak power consumption of less than 20 W. The three instruments are depicted in Fig. 3.

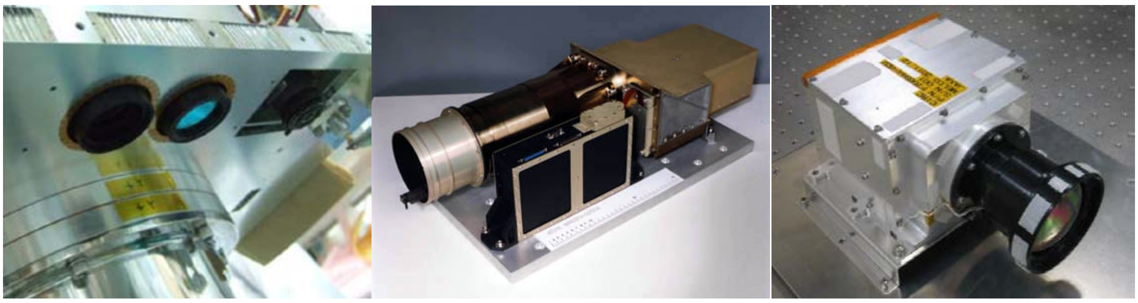

Figure 3. LSI (left), CHRIS (center) and CIRC (right) instruments.

3. AODCS DESIGN

The design of the AODCS relies on the typical sensors used for small LEO spacecraft: one GPS, one magnetometer, one coarse Sun sensors, one fine Sun sensor and one Earth sensor [10]. For the considered application, an AMR type magnetometer has been selected. It operates in the ±100 μT range with a precision of ±300 nT, resulting in an attitude determination accuracy of about 1° [11]. The two Sun sensors are used during different mission phases. The coarse Sun sensor, composed by properly arranged solar cells, provides an estimate of the Sun position during the early mission phase. Once coarse pointing is established, the fine Sun sensor, composed by a position sensitive detector, is used for accurate pointing, providing an attitude determination accuracy below 0.1°. The Earth sensor operates in the infrared domain, scanning a wide field of view to detect the Earth/space transitions, and can be used either in combination with the magnetometer or as a backup system during eclipses.

Besides the above-mentioned sensors, the following set of actuators have been selected: four groups of three cold gas thrusters (CGT) for orbit control, three orthogonal reaction wheels for attitude control and three orthogonal magnetorquers for wheel desaturation (see Fig. 2). Reaction wheels are sized so as to provide a maximum torque of 20 mNm and an angular momentum capacity of 60 mNms. Cold gas thrusters are designed to provide a thrust level of 20 mN each, using xenon propellant at an estimated specific impulse of 45 s. The thruster configuration is depicted in Fig. 4. The magnetorquer system consists of magnetic coils mounted on three perpendicular faces of the spacecraft body, able to produce a magnetic dipole moment of 2.5 Am^2 each, which ensures a wheel desaturation time shorter than 1 hour. In order to meet the constraints dictated by the satellite size, a coil diameter of 22 cm has been considered.

A ground-based orbit control strategy has been selected to perform the periodic semi-axis boost maneuvers required for drag compensation, whereas an autonomous control unit has been implemented to maintain the attitude of the spacecraft aligned with the desired orientation. The attitude control unit consists of two boards in direct communication: a Mother Board (MB) and a Daughter Board (DB). The MB is a general purpose processing board containing a SpaceWire Remote Terminal Controller (SpW-RTC) device, the memory blocks and the power supply circuitry. The SpW-RTC chip is in charge of the communication with the on-board control system and the embedded microprocessor. In particular, it runs the data processing required by the attitude control algorithm, consisting of a Linear-Quadratic-Gaussian (LQG) regulator. The DB contains interfaces to sensors and actuators and the associated signal conditioning electronics. It houses a FPGA of the ProAsic3E family, which manages sensor and actuator I/O. The control unit is shown in Fig. 5.

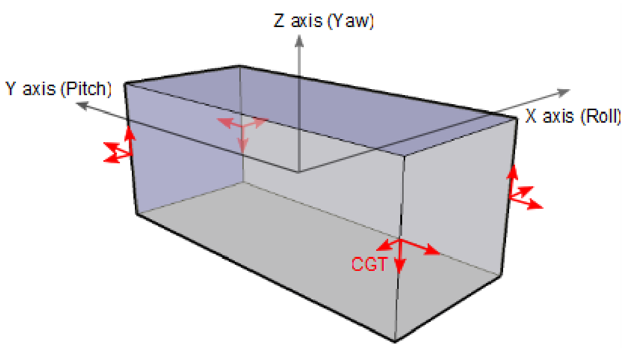

Figure 4. Thruster configuration.

Figure 5. Attitude control unit.

4. MISSION ANALYSIS

A preliminary mission analysis has been performed by simulating the evolution of the orbit and the attitude of the spacecraft with a high-fidelity numerical propagation routine [12], and evaluating the mass and power budget of the system. The objective of the ground-based orbit control strategy is to keep the satellite ground-track within 60 km from the nominal value. The attitude control accuracy requirements are specified as 0.1° for daytime observations and 1° for nighttime observations (during eclipse periods), which ensures the compatibility with the considered AODCS and Earth observation payloads.

Figure 6 shows that the ground track error can be maintained within the required accuracy during the three years of estimated mission lifetime, by performing a semi-axis boost maneuver every 60 days, whereas the equator crossing time error is kept below 3 minutes. The performance of the attitude control system is illustrated in Fig. 7. It can be seen that the attitude error, reported for the roll axis over a single orbit, satisfies the control accuracy requirements (a similar behaviour is observed for the other axes).

Figure 8 shows that approximately 2.16 kg of xenon are required by the CGT system for orbit control over the entire mission lifetime, which is consistent with the propellant mass stored onboard the spacecraft. The power generated by the three body-mounted solar panel is reported in Fig. 9 (solid line) over the orbital period, together with the power absorbed by the control system (dotted line). Notice that approximately 30 W of power are generated on average, of which about 20 W are available for payload operation.

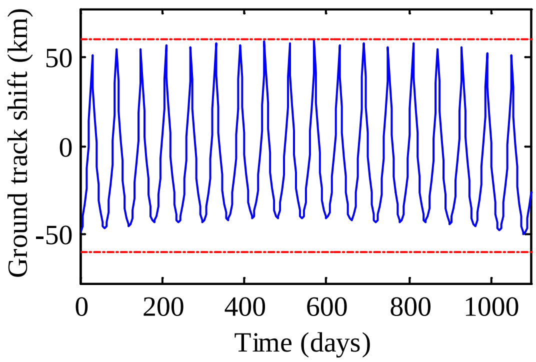

Figure 6. Orbit control.

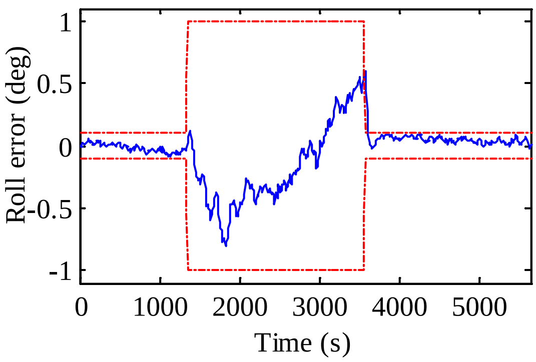

Figure 7. Attitude control.

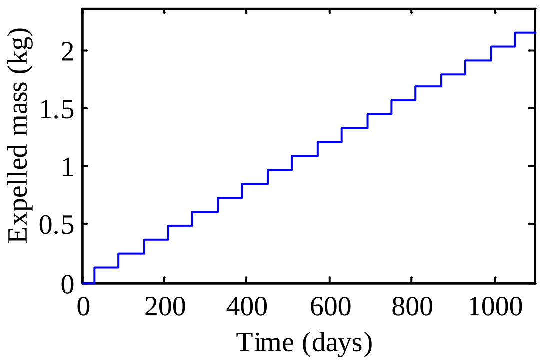

Figure 8. Mass budget.

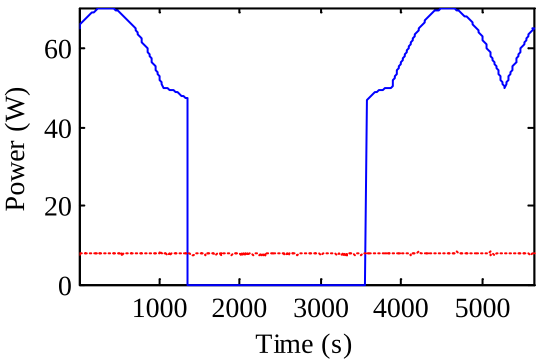

Figure 9. Power budget.

5. CONCLUSIONS

This paper has presented the SSCAM small satellite concept for the remote observation of Region Tuscany, Italy. The design of the spacecraft and a preliminary mission analysis have been discussed. Given the compatible power consumption of state-of-the-art imaging instruments, such as the ones presented in Section 2.2, and the results of a simulation case study of the performance of the AODCS, it can be concluded that the proposed platform can meet the expectations of future small satellite remote sensing missions, in terms of achievable ground resolution and revisit rates.

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