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Polyaniline/poly (2-acrylamido-2-methyl-1-propanesulfonic acid) modified cellulose as promising material for sensors design

(Article begins on next page)

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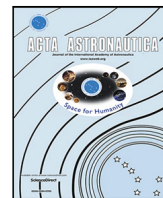
Research paper

Fast development and validation of a Sensing Suite system for CubeSats

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ABSTRACT

Educational CubeSats still remain valuable hands-on practice activity in an academic environment and provide effective and innovative solutions from technical and management points of view. Fast delivery and low cost is the paradigm driving such projects. Contemporary, the simplicity of the solutions has to match with reliability to guarantee the success of the on orbit operations. The present paper shows the entire life cycle of a Sensing Suite System suitable as scientific data collector within a CubeSat, developed in less than four months from the conceptual design to the delivery and reaching the orbit in less than six months. Technical solutions for hardware and software development and test and management good practices are provided. Circuits details to withstand and/or tolerate the space environment and data handling to maximize the number of data and measurements gathered onboard are explained. The use of breadboards and flat-sat arrangements support both the design and verification of the protoflight comprising the schedule and increasing the confidence level on the goodness of the entire solution. The reliability of the design is proven through the entire assembly, integration and verification campaign and validated by the data sent by the satellite during the operative mission.

1. Introduction

CubeSats are today systems of interest for many actors in aerospace field such as the scientific community, the industry, commercial operators, and governmental organizations and many new enterprises were born strictly focused on the small satellites business. The CubeSats can accomplish the objectives of complex missions such as Earth Observation [1,2], Telecommunications [3,4], In-Orbit Servicing [5,6], In Orbit Inspection of outposts [7] and debris [8], and, even, Space Exploration, e.g. Mars exploration [9], Moon exploration [10] and asteroids [11,12]. In many cases, the modern CubeSat technologies are quite mature or are increasing their own Technology Readiness Level (TRL): this is the case of electric [13] and chemical [14] propulsion systems, high-performance communication system [4], attitude and orbit determination and control systems, guidance, navigation and control in proximity operations missions [15,16] and in deep space [17,18], and specific elements or subsystems like docking and berthing systems [19]. Actually, CubeSats still maintain their original goal that sees the educational aspects as relevant for the preparation of a valuable, new generation of engineers, technicians, professionals and operators not only for the small satellite context but also for the entire aerospace field. In fact, CubeSats have been invented as hands-on experience in the academic context [20], and many universities around the world started educational programs based on CubeSats development [21], sometimes supported by Space Agencies. Educational CubeSat projects

have peculiar features [22]: low cost, low complexity of the design, slim but effective verifications. On the contrary, the main risks are low reliability due to Components Off The Shelf (COTS) and home-made subsystems use to avoid the cost of high TRL elements and the reduced experience of most of the people involved in these projects (mainly students) [23]. Sometimes, one of the most addressing drivers is the fast delivery of the spacecraft to exploit the launch opportunities and the need for students to earn their Degree on time while trying to gain the best from the hands-on experience participating to almost all the project phases. Moreover, the development of hardware and software for onboard subsystems is one of the most valuable activities in an educational project because it leads the students to face the threats generated by space environments and to adopt adequate solutions or mitigation actions to counteract and/or tolerate undesired events in orbit. The present paper deals with the very fast design and validation in orbit of a Sensing Suite system included on board of a 3U CubeSat with scientific objectives, entirely developed by a group of students and researchers of Politecnico di Torino. Sensing Suite is a system dedicated to collect, manage and transfer data relevant to achieve scientific purposes of the mission. The main driver for the project has been the fast delivery since the time between the beginning of the conceptual design (Phase 0+A) and the CubeSat integration on the launch (end of phase D) was less than four months and the satellite reached the final orbit

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in less than six months from the conceptual design. Fast delivery leads to compressed and overlapped design and verification phases, using model and simulation based approaches following the modern system engineering "multi-Vee" strategy [24]. At design level, the choice was to implement a slim system with low complexity but including smart solutions. The design was influenced by the necessity to select elements with quick procurement which means to adopt components already available on the market and, if possible, to re-use solution for the circuits, components and subassemblies design already developed and verified in house and/or with high heritage and know-how. The use of elements already procured in the past and/or already used in other research programmes largely helped on saving money and reduce the project cost. Assembly Integration and Verification (AIV) campaign is based on the definition of different models, from electro-functional and dummy models to the flight models, and the possibility to start the functional verification campaign as soon as possible using dedicated Ground Support Equipment (GSE) that emulates the behaviour of components not already available. That allowed to speed up the functional verification sessions and get the Sensing Suite system ready for integration within the satellite just on time for the environmental campaign. From the reliability point of view, a fault tolerant architecture has been implemented on hardware and data management, protection circuits and components are added to withstand events that could compromise the good working of the entire system or part of it.

The main contributions of this paper can be summarized as follows:

1. To provide an example of fast delivery project effective in any phase of the product life cycle. Details of the most interesting aspects of design, development, assembly, integration and verification, and validation in orbit are reported to support the reader in the development of a similar system for small satellites.
2. To propose useful good practices to face an educational, low-cost project with a really short schedule. These suggestions are supported by the strong experience and know-how gained during a real project.
3. To show the process leading to a simple and reliable avionics system for small satellites. Hardware and software for a data collection system for small satellites are described in detail. An effective way to verify the system as soon as possible, before and during the integration onboard and the full functional and environmental tests. Finally, the goodness of the adopted solutions is validated through the operations of the satellite in orbit.

The organization of the paper goes through the steps of the product life cycle: design and manufacturing in Section 2, assembly, integration and verification in Section 3, and in orbit validation in Section 4. Sections Section 5 contains the conclusions and recommendations.

1.1. CubeSat description

SPEI Satelles is a 3U CubeSat designed upon the platform developed at Politecnico di Torino in the framework of its CubeSat programme. The Sensing Suite system is the payload of the satellite and it is constituted of a dedicated embedded system equipped with an Inertial Measurement Unit (IMU) with a tri-axial magnetometer and gyroscope, while 30 temperature sensors are used to gather data useful to validate mathematical models developed by the students. The cubesat is equipped with two independent Command and Data Handling (CDH) boards (a commercial solution) and two communication systems (ComSys) (a new version of the e-star 2 communication system [25]) to improve reliability. The ensemble of one CDH and one communication system constitutes a BUS. The two buses are independent for most of the functions but BUS1 can turn off BUS2 to save power during specific mission phases. The two BUSes and other subsystems are interfaced (power and data interfaces) thanks to an interface and distribution board, called Backplane. Furthermore, the two BUSes alternate for transmission with arbitration coordinated by an arbitration circuit located on
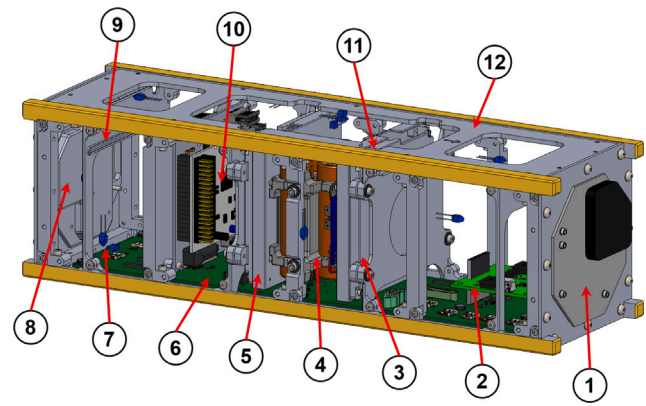

Fig. 1. Internal layout of the satellite.

Fig. 2. SPEI Satelles Protoflight model.

the Backplane. The Electrical Power System (EPS) is constituted of four body-mounted solar panels and a lithium-ion battery pack. Backplane distributes power among all subsystems, interfacing the EPS with all the other components. The spacecraft is equipped with a passive Attitude Control System (ACS) based on magnetic attitude stabilization with permanent magnets and hysteresis rods to stabilize the attitude and dump attitude oscillations. The inner thermal environment is regulated by a passive Thermal Control System (TCS), which relies mainly on thermal pads and specific surface finishings, and an active TCS present on the battery. All these subsystems are installed inside an Al 7075 aluminium alloy structure suitably treated with Surtec plus hard anodizing where required (e.g. for the rails).

The internal configuration is reported in Fig. 1 where the following items are highlighted and labelled: (1) Communication System 1 (ComSys 1), (2) Direct Energy Transfer (DET) circuit, (3) Command and Data Handling 1 board (CDH1), (4) Battery pack, (5) Command and Data Handling 2 board (CDH2), (6) Backplane board, (7) Example of temperature sensor, (8) Communication System 2 (ComSys 2), (9) Hysteresis rods, (10) Sensing Suite system, (11) permanent magnets and (12) structure. Fig. 2 instead shows the external view of the satellite with the access ports, the solar panels and the antennas visible.

Table 1
Objectives for Sensing Suite system.

| Objective	Label	Rationale
OBJ1	Thermal analysis data	Acquire temperature measurements on various points of the spacecraft to characterize the spacecraft thermal environment and its interaction with the space environment.
	Attitude analysis data	Acquire attitude measurements to characterize the spacecraft passive attitude control system for communication purposes.
	Earth Magnetic Field Mapping	Acquire Earth Magnetic Field measurements to characterize the environment in which the satellite operates.
OBJ2	System reliability	Help checking the spacecraft health status by collecting housekeeping data. Verify the application of system-level techniques for the mitigation of radiation effects in LEO and manage onboard misbehaviours.
OBJ3	COTS qualification in space	Verify the behaviour of a low-cost electronic system built from COTS orbiting in LEO. Verify the behaviour of a commercial MRAM in LEO.
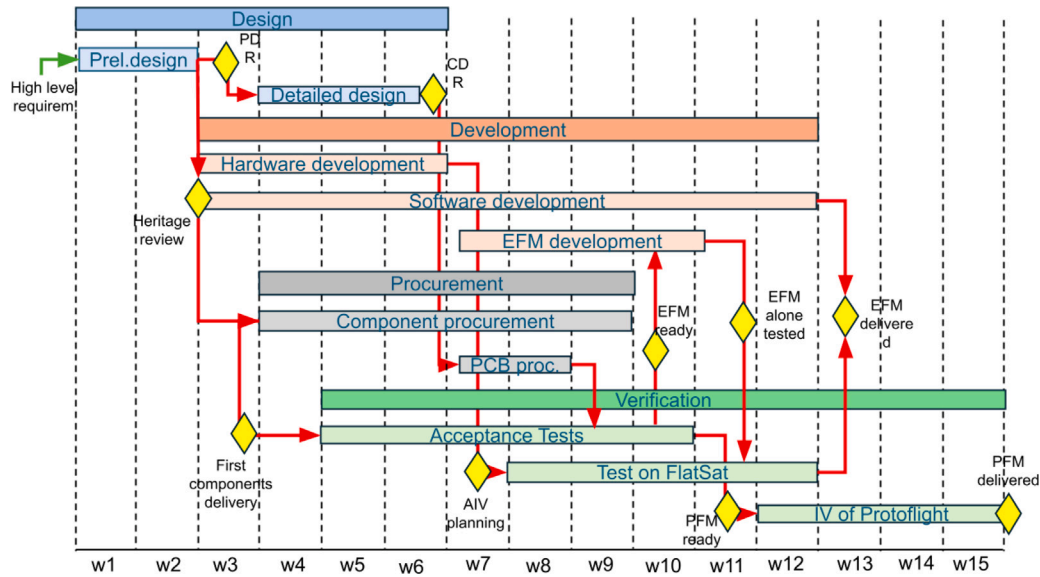
Fig. 3. Plan for Sensing Suite System development, test and integration.

2. Sensing suite design and manufacturing

The design process of Sensing Suite follows three main steps: (1) objectives definition and high level requirements and constraints analysis and definition, (2) Sensing Suite architecture definition and hardware selection, and (3) Sensing Suite software description.

2.1. Objectives and high level requirements

Sensing Suite is designed to accomplish three main mission objectives, reported in Table 1. OBJ1 refers to the scientific studies to characterize the space environment around and inside the satellite with focus on the thermal environment and the Earth Magnetic Field Mapping that allow validating thermal models and passive attitude control models developed by students. OBJ2 aims at assessing the reliability of the entire satellite by acquiring onboard housekeeping data. OBJ3 deals with qualification in orbit of low cost electronic circuits based on COTS and components with a specific technology.

The main driver of the Sensing Suite project is the short schedule reported in Fig. 3. Preliminary design took in inputs the high level (i.e. mission and system) requirements and ended with the Preliminary Design Review (PDR), hardware and software architectures were sketched and the already available and developed software modules and components/equipment were checked to concentrate on the detailed design of the new parts and the overall integration of the Sensing Suite System. PDR enabled the development of the new modules and circuits, with focus on the realization of the new firmware and testing of new protection and sensors acquisition circuits. Review of the heritage

led to the start of procurement, which was addressed by an heavy and precise research of providers that could quickly deliver the products while being limited by procurement procedures and rules of the public administration. The detailed design allowed completing the development of the Sensing Suite board being supported by the results coming from the progressive testing of all the new elements in order to confirm their correctness or to bring the suitable corrections. That increased the confidence level on the design and enabled the procurement of the Printed Circuit Board (PCB) at week 6, at the end of the Critical Design Review (CDR). The verification immediately started after the hardware acquisition and the release of the first software modules, in order to proceed with the acceptance tests on single elements. For Sensing Suite System, two Electro-Functional Models (EFM) and two Proto-Flight Models (PFM) were manufactured. EFM was adopted both for the development of the system and for its integration on the full satellite model which was represented up to week 12 by a FlatSat, i.e. all satellite elements were available for functional verification but not in the final configuration, facilitating the verification. Meanwhile, an Assembly, Integration and Verification (AIV) step-by-step procedure was defined to obtain the PFM taking advantage of the outputs from the work on the EFM, reducing the required time and effort. In particular, PFM was singularly tested on its hardware elements and subsequently the final software release was loaded after the completion of the functional test sessions on the EFM. This way the functional tests on Sensing Suite were quickly completed in less than 3 weeks. After 15 weeks, the system was integrated in SPEI-satelles, following with the full-functional tests, day-in-life test and the environmental tests campaign.
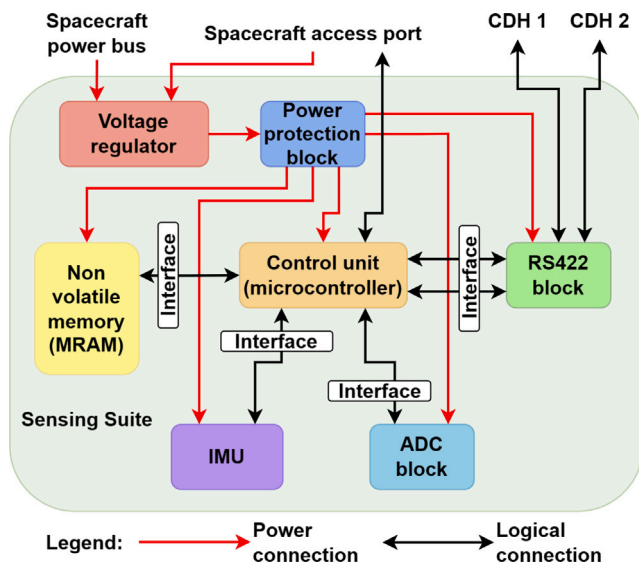
Fig. 4. System architecture.

2.2. System architecture and hardware design

The Sensing Suite system architecture is summarized in the block diagram of Fig. 4.

The main building blocks are:

- The *control unit*, consisting of a microcontroller System-On-Board, connected to the spacecraft outside through the Access Port (AP) mounted on the Backplane board;
- The *ADC block* constituted by two 16-bits analog multiplexers and two Analog to Digital Converters (ADC): 32 temperature sensors are in input to this block and sampled values are sent to the control unit;
- The *RS422 block* with two independent RS422 transceivers, each one implementing an Universal Asynchronous Receiver Transmitter (UART) communication channel towards one CDH board;
- The *IMU* is constituted of COTS sensors with a tri-axial gyroscope, a tri-axial magnetometer and tri-axial accelerometer. IMU has a dedicated micro-processor that performs the measurements and processing and provides them via UART through a telemetry packet at regular intervals;
- The *non-volatile memory* consisting of a Magnetoresistive Random Access Memory (MRAM) chip where all telemetry packets sampled by Sensing Suite are stored for subsequent transmission to the CDHs;
- The *buck voltage regulator* which provides 3.3 V supply to the whole system generated from the spacecraft power bus or the AP;
- The *power protection block*, consisting of circuits to protect one-by-one each system element from latch-up and isolate a faulty element at power level from the rest of the circuit;
- The *Interface front-ends* that isolate faulty elements on the logical interconnections/General Purpose Input Output (GPIO) side (see details in Section 2.2.3).

2.2.1. Components selection

The selection of electronic components and the board design have been driven by the use of COTS components, the reuse of circuits with higher know-how and the analysis of challenges for electronic devices in space [26]. In particular, the reduced development time for our proposed design addresses the following (sketched in Fig. 5):
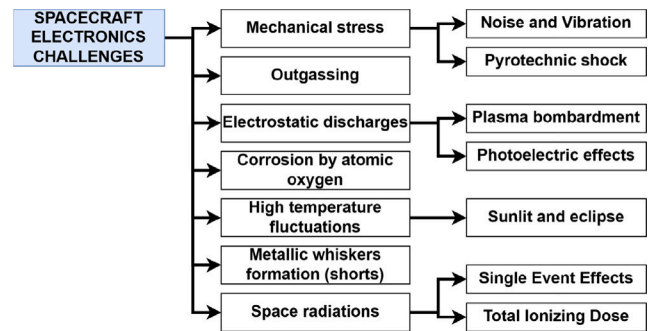

Fig. 5. Challenges for electronics devices and circuits in space [26].

- *High temperature fluctuations*: to face the expected temperature fluctuations inside the spacecraft due to the alternation of eclipse and daylight conditions along one orbit, all components were chosen as automotive grade (extended operating temperature range).
- *Space radiation*: considering that no costly rad-hard electronics can be used, techniques to mitigate the effects of the radiation environment have been included. Energetic particles affect the electronics in terms of Total Ionizing Dose (TID) [27] and Single Event Effects (SEEs) [28]: while TID is a long term failure phenomenon that can be neglected for SPEI mission due to its short duration (less than one year), redundancy on the memory and periodic resets of the CPU reduce the likelihood that a Single Event Upset (SEU) propagates, while Single Event Latch-Ups (SEL) consequences are limited by specific circuits and solutions which prevent overvoltage and overcurrent on critical lines.
- *Mechanical stress*: passive devices were chosen to have flexible terminations to better withstand the launch loads and the release from the dispenser.

Corrosion by atomic oxygen is not considered because no elements of Sensing Suite system are exposed to the "space sand" thanks to the protection of the anodized aluminium of the structure. *Outgassing* is prevented by avoiding lubricants and plastics and, in general, materials not allowed for space; moreover, residual bubbles on the system are eliminated during the thermal vacuum test campaign. The following sections pose the attention on some peculiar hardware solutions adopted for this design.

This analysis drove a careful selection of the Sensing Suite hardware components:

- *Microcontroller* of the control unit is the STM32L452RE [29,30], chosen thanks to the wide know-how of the developers with this line and relative frameworks, thus reducing the software development time. The L series embeds a code memory correction functionality (single bit-flip correction, double bit-flip detection) supporting the correction of SEU. The correction is performed during the read operations thus the software was later coded to routinely perform system resets in order to refresh the instructions that are executed more frequently. The microcontroller was mounted as daughter board on top of a Nucleo64 [31] without soldering it on the PCB, eliminating the risk of manufacturing errors and enabling fast replacement of the microcontroller if needed.
- *ADC block* is based on the AD7788 sigma-delta ADC [32] because this device has a minimal footprint and the noise shaping characteristics of sigma-delta modulators reduces noise due to upsets on the analog front-end with respect to the Successive Approximation Register (SAR) converters.
- *MTi-3 IMU* [33] has been selected because of the team experience on this series of devices [34] and the reduced dimensions and mass of this System-On-Module.

- The *Non-volatile memory* is the AS3016204 MRAM [35], a promising technology for space applications due to its intrinsic immunity to SEE [36], although still having low storage density and a higher cost. A commercial MRAM still has unhardened Complementary Metal-Oxide Semiconductor (CMOS) circuitry around the memory bank to perform read/write operations and interface with the Input/Output (I/O) BUS, so the device was in any case protected by including it in a voltage domain (see Section 2.2.2). This device also comes with the advantage of a virtually infinite write endurance (10^{14} according to the datasheet), eliminating the need for wear levelling techniques like typical for NAND flash banks.
- *RS422 interfaces*: LTC2852 [37] is the RS422 transceiver because it provides a receiver high impedance even in unpowered state (at least 92kOhm according to the datasheet), reducing the possibility of a fault to propagate to the CDH boards in case the RS422 power domain or the whole system gets disconnected by the protection circuits.
- *Power regulation circuit* is based on TMR3-2410WIR DC/DC [38]. While this System-In-Package regulator is made of CMOS technology and thus is susceptible to latch-up, it has railway certification, that proves how it is robust and withstands mechanical shocks and vibration, temperature excursions and electrical surges. It is also an insulated-type converter, specifically chosen (even if less power efficient) because the insulation avoids propagating the unregulated spacecraft voltage to the board in case the regulator stops working, with the risk of producing fumes inside the spacecraft or propagate the high unregulated voltage to the CDHs through the RS422 interfaces.
- *System Interfaces* are (1) the Communication interface to interconnect the Sensing Suite with the rest of the spacecraft (power bus and RS422 lines towards the CDHs) (2) the Access Port (AP) to get access to the system during the test campaigns. AP consists of the Serial Wire Debug (SWD) interface [39], to program and debug the microcontroller with an ST-Link programmer, and (3) the debug UART used to implement a terminal console useful to monitor the system operation.
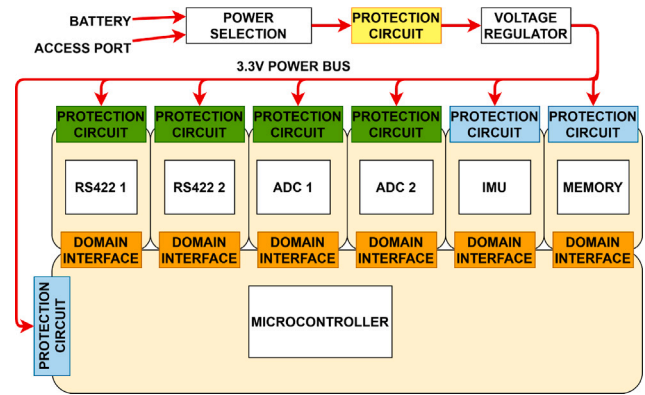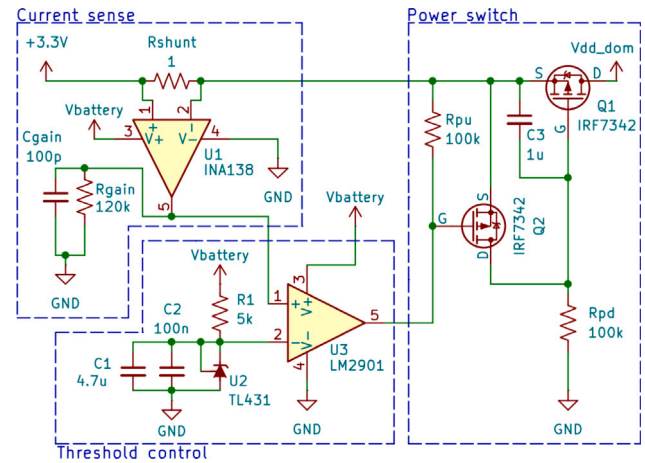
2.2.2. Latch-up protection circuits

Latch-up conditions can be generated whenever a sufficiently energetic particle hits CMOS circuitry and an injected current or an overvoltage condition activates the parasitic Silicon Controlled Rectifier (SCR) existing in CMOS cells, creating a low impedance path between the supply rails. Once activated, the latch-up condition is self-sustaining and the only way to stop it is to perform a power cycle of the device. An uncontrolled Latch-up can have severe effects and lead to the partial or complete destruction of the IC due to overheating [40]. For this reason, each device (or group of devices) on Sensing Suite system got its power rail protected by an appropriate circuitry (Fig. 6), which separates the power bus into isolated power domains.

There are mainly two types of domain protection circuits, depending on the expected peak current consumption of the devices in nominal conditions. The simplest type of protection circuit consists of resistances placed in series to the supply rail, which drops the voltage in case of latch-up overcurrent until the self-sustaining threshold is reached. This solution is applied to low peak current devices such as ADC and RS422 transceiver (provided that no termination resistor is present on the transmission line). For devices with higher peak power consumption, like the MRAM, IMU and microcontroller, a more complex latch-up protection circuit was developed (Fig. 7).

It is inspired from [41] (specifically by using the same INA138/168 [42] current sense amplifier). The circuit consists on a current sense block (converting the device current consumption to a voltage output through a shunt resistor and an amplifier) whose output is compared to a threshold by a comparator. Whenever the threshold is reached the supply line is opened by a power switch, this power switch (an high side p-type Metal-Oxide Semiconductor Field Effect Transistor

Fig. 6. Sensing Suite power delivery architecture.

Fig. 7. Latch-up protection circuit.

or MOSFET) circuit is implemented in such a way to have fast turn-off (tens of microseconds) and slow turn-on (tens of milliseconds). In [41], a discharge MOSFET (crowbar) is also included to discharge the bypass capacitors connected to the power pins, reducing the probability of permanent damage; the solution proposed in Sensing Suite system eliminates this crowbar to reduce the complexity because the power switch and the crowbar must be well synchronized to avoid short circuit currents on the power rail during transitions, when both MOSFETs are conducting simultaneously. All devices selected for the latch-up protection circuits were chosen to be linear bipolar technology devices (INA138 current sense amplifier, TL331 comparator [43] and TL431 reference voltage generator [44] for the threshold), because it is less prone to SEL, as suggested in [45]. The actual SEL overcurrent value is difficult to determine a-priori, since it depends on the specific device but also on the specific part of the latter which is affected. In literature [46–48] there are examples of currents ranging from tens of mA up to 1 A and databases exist with experimental results on a wide variety of devices tested with particle accelerators or lasers (like the IEEE Workshop on Radiation Effects Data [49], ESA's [50] and NASA's Radiation Test Databases [51]) but still the number of tested ICs is small compared with the commercial offer and rarely the selected devices were found on such databases. For Sensing Suite circuits, the threshold current was selected to be around 100 mA for the complex latch-up protection circuits, compared to a nominal operating current of the devices in the range of around 10 mA to 40 mA, a value which was considered safe for our experimental design (close enough to the expected latch-up overcurrents and not too close as to risk causing false detections). Finally, a Positive Temperature Coefficient (PTC) fuse was

Fig. 8. The four boards produced.

placed to interrupt power to the whole board in case of irreversible failure, to isolate the faulty element on Sensing Suite from the rest of the system or the entire spacecraft.

2.2.3. Domain interfaces

The chosen configuration allows to disconnect a single power domain from the supply rail in case of Latch-up/short circuit fault on the supply pins. This represents a problem since the presence of ESD protection diodes allows current to flow from the I/O pins of a not powered device to its supply pins, with the resulting propagation of the fault to other devices/power domains. For this reason, a group of domain interfacing blocks (see Fig. 6) isolates a not powered domain from the rest of the system. 1-bit SN74AXC1T45 [52] and 4-bits SN74AVC4T245 [53] dual supply bus transceivers were chosen because they implement a partial-power-down capability which puts I/Os in high impedance state whenever the power supply is removed from one of their sides. That leads to the complete isolation of domains while the rest of the system can continue to operate at reduced functionality (if the microcontroller domain is still working).

2.2.4. Board production

The PCBs were manufactured and assembled by a local SME by providing a set of file commonly generated by CAD for electronics designs: Gerber files, drill files, Bill Of Material (BOM) and 3D models/Renders of the board were delivered. All files were easily generated by KiCad with the exception of the BOM, generated by Python script but included in KiCad as it is completely open-source and allows deep customization including the possibility of adding custom BOM scripts. The entire fabrication process took less than 3 weeks to complete a total of four boards (Fig. 8): two boards without a soldered IMU (EFM), and two boards with the IMU soldered (the main PFM and spare).

2.3. Software design

The flight software was built resorting to Hardware Abstraction Layer (HAL) libraries and runs on FreeRTOS Real Time Operating System (RTOS), allowing for fast software development, portability and testing. The firmware is organized in hierarchical layers (Fig. 9), starting from the bottom of the figure:

- *Board hardware*: i.e. the hardware elements of the system.
- *uC hardware*: the microcontroller peripherals interfacing with board hardware or implementing other vital functions. The figure highlights only the used peripherals while others (like the reset and clock controller and the interrupt controller) are not shown for simplicity.

- *uC drivers*: the drivers for microcontroller peripherals, both HAL delivered by ST and those developed ad hoc for the use on this project (like the second level UART driver);
- *Driver utilities and board drivers*: i.e. drivers developed for the board hardware and utility libraries they rely on.
- *Tasks*: the upper layer of the firmware, consisting of the actual tasks that implement the various system functionalities as state machines.

An important constraint for the firmware development has been avoiding the use dynamic memory allocation because it is unpredictable and its use is discouraged, as NASA's list of programming best-practices states in [54].

2.3.1. Drivers overview

UARTs are managed in interrupt mode, making use of FreeRTOS queue primitives to perform data transfers from and to the Interrupt Service Routines (ISR) and providing the upper layers with an easy to use UNIX-like interface (read() and write() functions). Interrupt mode is processor time-consuming, especially with high baud rate (115 200 baud in our case) and numerous UART interfaces. However, its behaviour is predictable compared with Direct Memory Access (DMA) and the processor resources are not so constrained for this mission. A series of software layers have been developed on top of the UART driver to buffer, search and decode incoming packets (buffering utilities and packet search utilities). This stack is used for communication links with the CDHs (with an additional protocol layer), the IMU driver and the debug console on the Access Port. The ADCs implement a Serial Peripheral Interface (SPI), so a library was developed to perform temperature sampling in polling mode, including the management of the analog multiplexers. Another library was developed for the configuration, measurement reception and packets decoding of the IMU. Communication with the CDHs utilizes a custom protocol on top of the UART frame, which formats source and destination addresses, command codes, payloads and Cyclic Redundancy Check (CRC) codes. The non-volatile memory driver is a crude implementation of the Common Flash Memory Interface (CFI) that executes memory transactions in single SPI mode (polling). The memory is organized as a circular log system since the data storage and retrieval requirements were minimal: telemetry packets are read in Last In First Out (LIFO) mode, with data retrieval consisting on reading the requested number of packets always starting from the last one written.

2.3.2. Memory organization

Sensing Suite packets have constant size and are sequentially written on the MRAM main array, while a pointer to the log head is stored inside a table on the MRAM Augmented Storage (AS) array, a secondary memory array of reduced size available for metadata storage (as shown in Fig. 10).

This head pointer comes under the form of a "packet counter" which is a 32 bit number that uniquely identifies each packet sampled from the start of the mission and is incremented after each sampling (with period of one minute). The packet counter is stored on the table as redundant copies and a bit-wise majority voting algorithm is applied during each access to the table in order to retrieve and eventually correct the values. The packets on main array are padded to align them to the memory size, allowing the software to retrieve the memory address from the packet counter with a simple multiplication and a modulus operation, as in Eq. (1).

$$Address = pckt\_cnt * pckt\_size \% mem\_size \qquad (1)$$

This type of memory organization is simple but reliable and supports the retrieval requirements of the mission, i.e. the transmission always starts from the most recent packet.
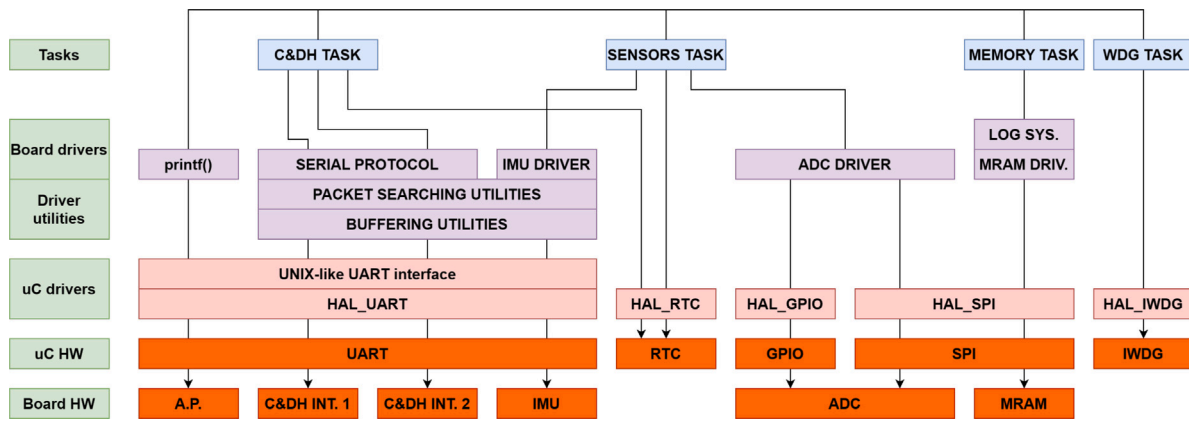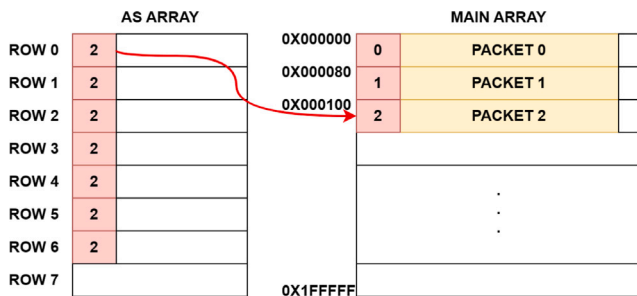
Fig. 9. Software architecture.
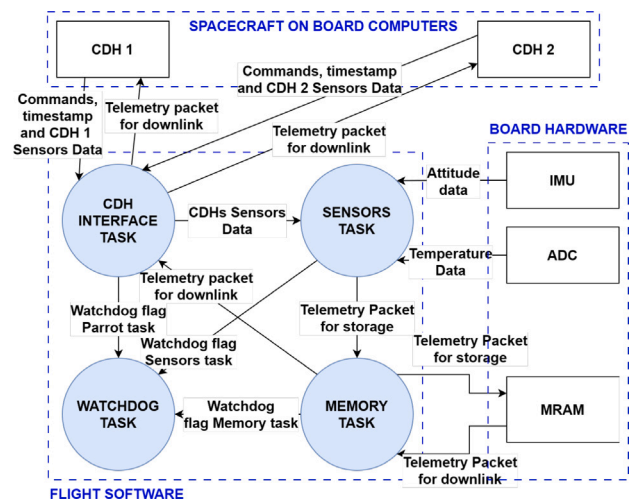

Fig. 10. Non volatile memory organization.

2.3.3. Tasks and Inter Task Communication (ITC)

At the application level, the firmware consists of four tasks:

- *Sensors task* samples and formats Sensing Suite packets including data from the temperature sensors, the IMU measurements, the housekeeping data merged with a selection of critical telemetries (e.g. battery voltage, charge/discharge currents and temperature, active operative mode) coming from the CDHs via RS422 link.
- *Memory task* performs read/write operations on the non-volatile memory.
- *CDH Interface task* manages the communication with the two CDHs, implementing the communication protocol and dispatching commands and data to/from them.
- *Watchdog task* manages the hardware watchdog and performs periodic, scheduled resets of the system.

Tasks are implemented as state machines, and the exchange of data between them is shown in the data flow diagram in Fig. 11.

The main type of exchanged data is the Sensing Suite packets which come with a timestamp for time synchronization. When a new packet gets sampled, Sensors Task requests the Memory task to write the new packet on memory. The memory executes the request while caching a copy of it in volatile Random Access Memory (RAM) (to ensure that in case the MRAM chip fails the system is at least able to downlink the last sampled telemetry packet from the RAM cache).

When a telemetry downlink command from one of the CDHs is received by the CDH Interface task, it requests one or more packets to the Memory task to then send them back to the CDH. The exchange of telemetry packets happens unidirectionally and always between two communicating tasks, allowing the implementation of a simple mutual exclusion mechanic: each packet exchange structure is paired with a corresponding atomic handshake flag which has the dual purpose of protecting it from race conditions and serve as request flag. Fig. 12 shows this handshake principle: each task which participates to the
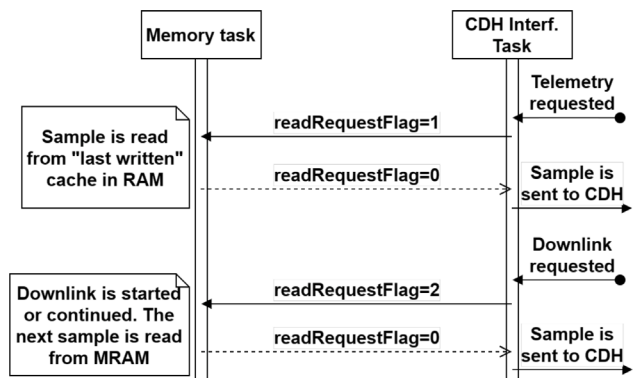

Fig. 11. Application data flow graph.

Fig. 12. Example application of handshake between CDH interface task and memory task.

transaction can enter the critical section (access the structure) only if the flag has been set to a determined value by the other one, that must stay outside the critical section until the flag is reset again by the former.

This very basic form of mutual exclusion does not ensure bounded waiting and progress conditions but allows a reduction in complexity and an increase in performance with respect to a solution using mutexes or queue primitives of FreeRTOS. A similar handshake mechanism (without the need of mutual exclusion, only to issue requests) is used
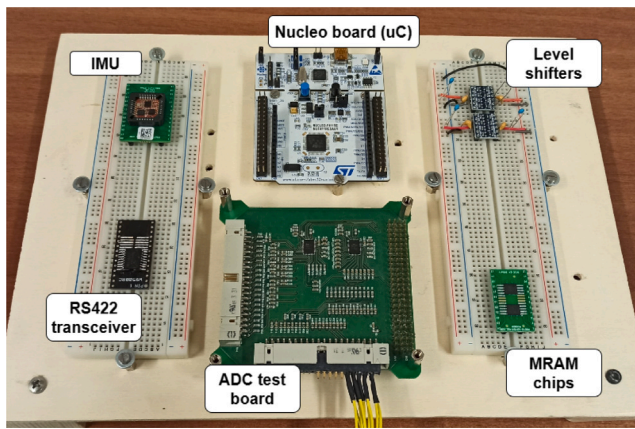
Fig. 13. Breadboard model.

Fig. 14. Electro-functional model.

between the Watchdog task and the others: the Watchdog task periodically receives handshakes from the other tasks, if one task does not send the handshake then the hardware watchdog is not reloaded and the system will reset. The Watchdog task also performs periodical and scheduled resets of the system in order to exit from eventual fault conditions and correct eventual memory bit-flips thanks to the embedded memory correction capabilities of the microcontroller.

3. Assembly, integration and verification

3.1. Model philosophy

Design and verification process of Sensing Suite was strongly supported by different kinds of digital and physical models.

- *CAD Models*: used for configuration and mechanical assessment, and delivered to mechanical engineers for satellite layout definition.
- *BreadBoard (BB) models*, Fig. 13: adopted for the design of circuits and development of the software. They are representative of the electrical and data interfaces and are subjected to functional tests to assess the features and/or the performance.
- *Electro-functional Models (EFM)*, Fig. 14: flight representative model in terms of size, configuration and mass. They also represent the first assembly and integration of all circuits, components and avionics elements.
- *Proto-Flight Models (PFM)*: flight model (i.e. the final version of the system) used for qualification&acceptance campaign performed before flight.

Fig. 15 shows how the models are used at different levels for development and verification purposes.

The final utilization of breadboards is educational, which sees their use in scholastic courses and student teams learning activities; EFMs allow training ground operators and testing operations procedures before moving on the flight module; the final utilization of PFM is flight in orbit to accomplish the mission objectives. The adopted philosophy meets the needs of the project so the strategy was to produce in-house many breadboards to quickly verify the features required by design. Moreover, the breadboards support the interfaces definition with the other onboard subsystems. The quick verification via breadboards of critical parts of the design guaranteed the production of the board with a higher confidence level. When the four boards were delivered, a precise path of Integration and Test was conducted that allowed fast testing of the EFM on the FlatSat while the same path was repeated in a slower manner and with the worthwhile corrections on the Proto-Flight Model (and its spare). In substance, the work on the EFM allowed to fix bugs on the software, assembly mistakes on the boards and validate the final AIV procedure.
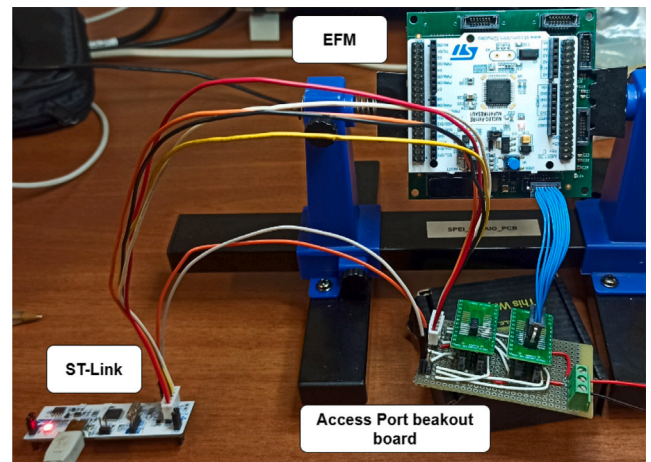

Fig. 15. Model philosophy for Sensing Suite system.

3.2. Proto-Flight Model integration and verification

The PFM integration and test of Sensing Suite system started after the completion of the activity on the EFM model integrated on the FlatSat, meaning that acceptance test of all board elements have been concluded and a complete and stable software release have been delivered. The Integration and Verification (IV) campaign followed a specific sequence (Fig. 16). First of all, the power connections and the right regulation and distribution of the voltages to all the system components were checked. Then, the integration and test of the software module-by-module was done, starting from data interfaces internal and towards other subsystems (e.g. the RS422 communication), then memory storage, temperatures values and IMU data management and onboard routines (e.g. time keeping) were verified. At the end, the hardware and software integration was completed and Sensing Suite system was delivered for mechanical integration on the CubeSat.

The next section details the most relevant steps in the IV procedure.

3.2.1. Integration and test of flight model

This test campaign aims at verifying the interfaces and the basic functions of the Sensing Suite components. The Sensing Suite is tested as a stand-alone component, first the PCB only and then integrated with the Nucleo board. The Sensing Suite is powered by a bench power supply. Communication with the board is managed by a dedicated software.

The test is executed according to the following sequence (Refer to indexes in Fig. 17):

- *Power supply verification* (not reported in Fig. 17): the Sensing Suite PCB (without the Nucleo mounted on it) is powered by a
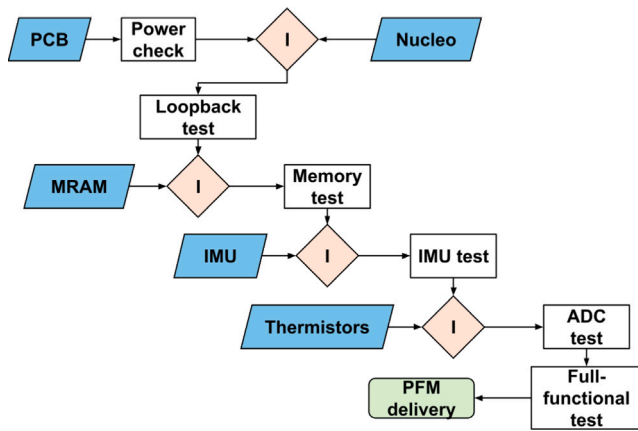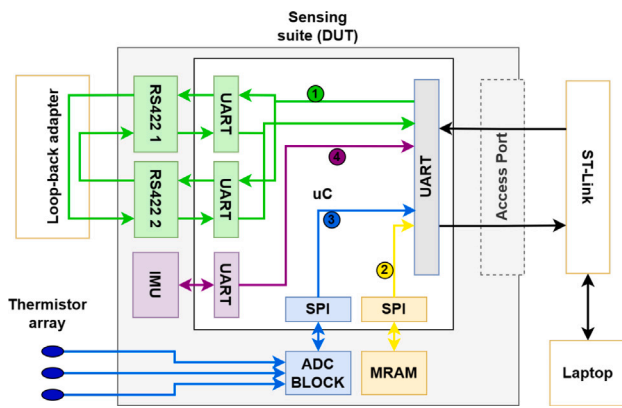
Fig. 16. Sensing Suite AIV plan.

Fig. 17. Test subdivision. 1: Loop-back test, 2: MRAM test, 3: ADC test, 4: IMU test.

power supply at the extremes and centre of the specified voltage range (9V to 13 V), the voltage at the power regulator output and on each power domain is measured by a voltmeter to verify that the power is correctly provided to each component. Main results show that the board is correctly powered and the measured voltages and current consumption are within the expected ranges. The power is then correctly being distributed to the components.

- *Loopback test* (index 1 in Fig. 17): the RS422 interfaces and the Access Port UART interface are tested in a loopback configuration. An adapter cross-connects the respective RS422 transceivers. Packets are sent from the Access Port terminal by an ST-Link through the UART. A test software, loaded on the microcontroller, routes those bytes on both RS422 TX lines which are alternatively exchanged between the two interfaces and then back to the AP. The test is passed if all bytes sent by the ST-Link are received back twice. As results, the RS422 correctly performs a loopback communication on both lines; therefore, the data interface towards CDH1 and CDH2 is properly functioning.
- *MRAM test* (index 2 in Fig. 17): the Sensing Suite non-volatile data memory is completely tested. The memory is erased, the erase verified, then it is written in both locked and unlocked states and the correct behaviour is verified, i.e. not written while locked, written while unlocked. Finally, the power is removed to confirm that the memory content is preserved without power by a dedicated reading. The results show that reading and writing are successfully completed and MRAM is correctly behaving and communicating with the microcontroller.
- *ADC verification* (index 3 in Fig. 17): the thermistors arrays are connected one at a time and the ADC readings are printed on

the Access Port terminal to verify the correct operation of the ADCs and the correct software binding of channels, the latter is verified by heating each single thermistor. This also verifies the integrity of the thermistors' cables. ADCs receive and convert data in realtime according to the expected sampling period.
- *IMU test* (index 4 in Fig. 17). Command sending and packet reception and decoding are both checked, the measured data is printed on the Access Port terminal and gyroscopes data is verified in its sign by rotating the Sensing Suite board. The IMU gets correctly configured by software and the measured values are correctly extracted from their packets.

No significant anomaly was detected during the test; the Sensing Suite requirements were verified and it was delivered for the integration with the spacecraft.

Sensing Suite was the last subsystem integrated onboard the Cube-Sat. which became ready for the functional tests campaign and the environmental tests campaign. The functional tests on the flight model of the satellite included the "first day-in-the-life test", the "full functional test" and the "day-in-the-life test" that, respectively, simulated the first day of the satellite after the release (when critical functions should be completed to assess that no major anomaly occurred on the satellite), the communication with ground (the capability to react to ground commands and send data) and a generic day in the operative life of the satellite. The environmental test campaign consisted of Vibration Test (sine and random sweeps testing) and Thermal Tests (short cycling test). Sensing suite did not present anomalies during both campaigns, however, a technical issue occurred during this phase: due to a weakly inserted grounding plug, a possible electrostatic discharge damaged the microcontroller by permanently shorting some of its pins while the Sensing Suite was integrated with the rest of the spacecraft, this unexpected fault allowed verifying that the latch-up protection circuits were correctly operating, since they successfully isolated the short circuit which did not propagate to the spacecraft supply bus or other avionics, the Sensing Suite was subsequently replaced with the spare PFM and the AIV operations resumed.

4. Validation using mission data

In-flight requirements validation was made possible by the continuous effort of mission operation team to gather data at every available communication window, multiple down-links were performed and the system always responded with the requested number of packets. The system correctly turned on after deployment and communication with the spacecraft was established in the following days, after the commissioning phase the first downlinks were performed from Sensing suite, confirming that it communicates correctly and with proper timing with the CDH in orbit and that the memory storage was correctly erased by CDH command after deployment. The analysis on received packets coming from the spacecraft verified that the sampling period was correctly respected; the values measured by sensors were nominal, suggesting that every sensing block was correctly functioning. Months of scientific data were gathered and compared against the predictive models developed by the ACS and thermal teams.

Fig. 18 shows an example of thermal data from Sensing Suite gathered during one downlink window and covering around 5 h (around 3 orbits), the temperature variations during the alternating sunlit/eclipse phases are clearly visible, while the absolute values are in line with the predictive models developed [55,56], with the outermost parts of the spacecraft (solar panels and structure) experiencing high fluctuations while the innermost components (electronic boards and battery) being well isolated and remaining in their safe operating range.

Figs. 19 and 20 show the IMU data over the same period, from a preliminary analysis it was possible to notice that the passive attitude stabilization system is correctly aligning the spacecraft Z axis to earth's magnetic field lines, it is also possible to appreciate the difference in
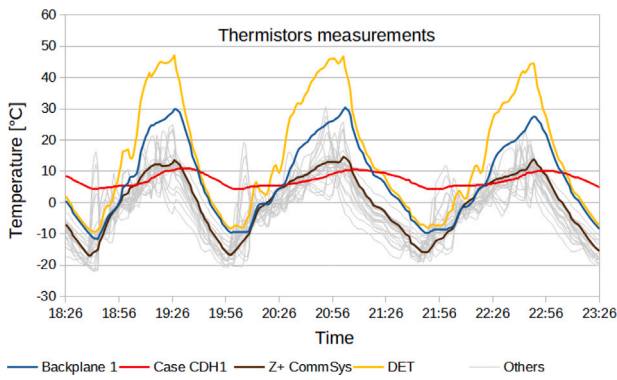
Fig. 18. Data covering around 5 h from all thermistors of Sensing Suite placed on various points of the spacecraft. Some relevant series are coloured and labelled while all the other series are greyed. X axis is local time (UTC+1) while Y axis is in °C. (For interpretation of the references to colour in this figure legend, the reader is referred to the web version of this article.)
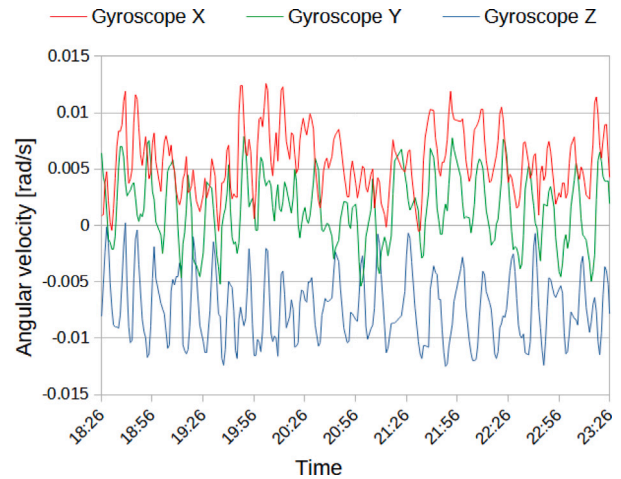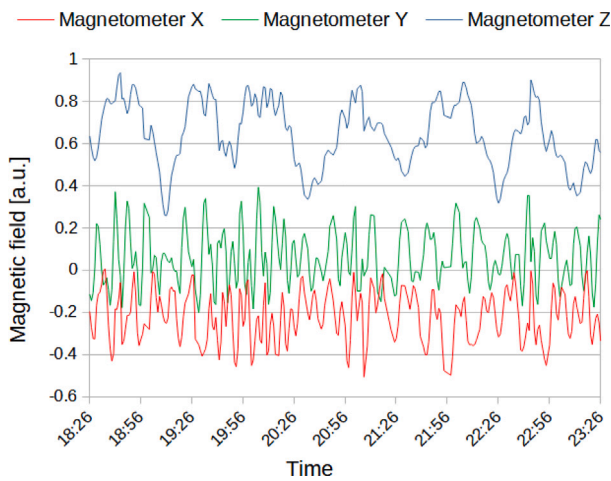

Fig. 19. Data covering around 5 h from IMU. X axis is local time (UTC+1) while Y axis is magnetometer data in Arbitrary Unit (normalized to calibration value).
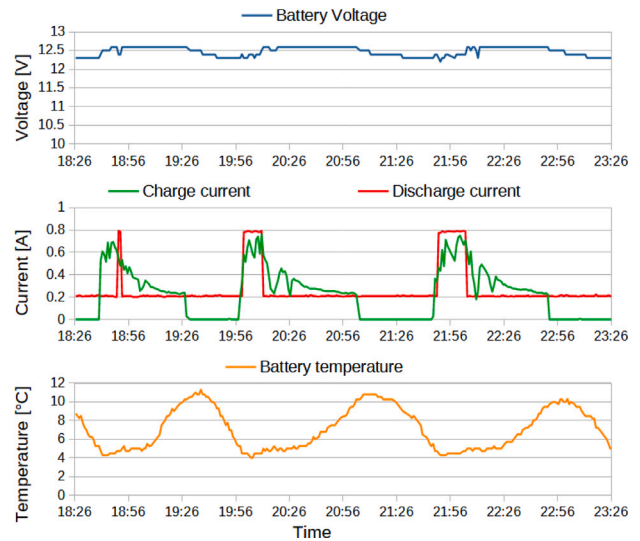

Fig. 20. Data covering around 5 h from IMU. X axis is local time (UTC+1) while Y axis is gyroscope data in rad/s.

Fig. 21. Data covering 5 h from battery sensors onboard the CDH1. X axis is local time (UTC+1) while Y axis is, from top: battery voltage in V, charge and discharge currents in A, battery temperature in °C.

field intensity at the equator (valleys) vs. at the poles (peaks), with a notable difference between the intensity at the equator in sunlit (higher) vs. in eclipse (lower) due to the deformation of the field due to the solar activity. From the gyroscope data is possible to extrapolate a maximum rotation period of around 7 min around the spacecraft axis, highlighting the unfeasibility (as expected) of high precision pointing with this type of attitude stabilization. It can be concluded that the scientific objectives are achieved since information for thermal assessment, attitude angular velocity profiling and mapping the Earth Magnetic Field are obtained thanks to the data gathered by the Sensing Suite System.

Although the fast development precluded to implement specific solutions to directly gather diagnostics about the system behaviour in orbit, an indirect extrapolation from the available data was done: the power requirements were validated from the measurements about the overall space-craft consumption (battery voltage, charge and discharge currents) and by observing that the battery never discharged completely during mission operation.

Fig. 21 shows the battery data sampled by CDH1 and stored by Sensing Suite over the same period, the measurements demonstrate that power is correctly managed and the solar panels correctly recharge the battery pack.

From this dataset, an average spacecraft consumption of 3 W and an average charge power of 3.2 W can be computed, with peaks of 800 mA on the charge current and 750 mA on discharge current (corresponding

on the activation of the spacecraft self-heating elements). Data about the correct work of the protection solutions is extrapolated from the Sensing suite reboot counter (which also counts the normal periodic self-resets), as shown in Fig. 22.

The gathered data showed no abnormal resets of the microcontroller. That proves that the housekeeping acquisition for reliability purposes is done by Sensing Suite. The correct activity of Sensing Suite in any phase of the mission in which it is involved also confirms that the selected COTS and their arrangement in the designed circuits as well as the overall system redundant architecture work properly in space environment, achieving OBJ3.

5. Final discussion and conclusion

Sensing Suite System is an example of affordable avionics system for data collection on an educational small satellite constrained by a very short schedule and a limited budget. In this regard, the estimated man effort was 1650 hours: the board development involved (1) an hardware/software engineer which designed the board and developed the major part of the firmware, (2) two additional software engineers
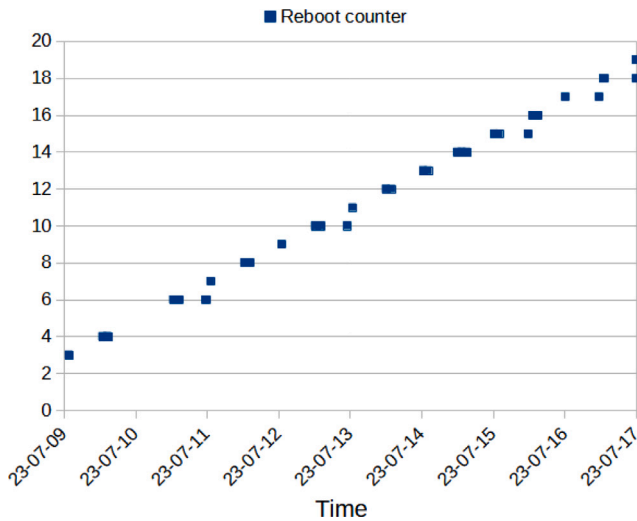
Fig. 22. Data covering 8 days showing the Sensing Suite reboot counter.

(one working on the IMU driver, the other working on the interface with the CDHs), and (3) a system engineer responsible of requirements formalization and a test engineer responsible of interfacing with the AIV engineers to define the verification path and produce test procedures.

The total cost of Sensing Suite System, including manufacturing and procurement costs and taxes, was 530€ per unit for EFM and 930€ per unit for PFM, the cost was mainly dictated by the fast production request, with the total value of electronic devices (IMU excluded) being less than 100€ per unit. The increased cost of the PFM was due to the soldering and mounting of IMU (with unit value of 300€). Sensing Suite System successfully met the requirements as defined in the design phases. The test campaign at any level demonstrated the capability to achieve the objectives providing on a regular base the data about temperature, satellite attitude and magnetic fields. Moreover, fault tolerance and fault recovery was observed during the test campaign and the operativity of the system also during the environmental test campaign. Even, Sensing suite features have been validated in orbit because it continuously operated for the scheduled duration of the mission without exception: it means that the system works properly and, in case of failure, it is able to recovery the misbehaviour. The Sensing suite project inside SPEI-satelles programme allowed to trace a set of good practices and lessons learned. Depending on the type of these outputs, each of them can classified in three main groups: technical, management and educational aspects.

Technical lessons learned derive from the analysis of Sensing Suite strength and weakness points:

- System-level techniques for radiation effects mitigation are the best allies for university programs to strengthen avionics by maintaining low cost figures. Another approach would be to couple this solutions with a comprehensive study on the expected radiation profiles and components SEE cross-sections but this is not always a possibility and requires specialized instrumentation.
- The existence of databases which gather radiation tests on different devices can be a precious resource during components selection but the number of tested devices is somehow limited and this goes in contrast with the experimentation of new avionics and electronics solutions which is typical of academic missions. A good compromise for a permanent university program could be to reserve some space on each mission for experimental devices or systems, run on a protected environment and sided by higher TRL avionics to run the main mission.

- Failure of avionics is an option in low-budget spacecraft. A modular and distributed architecture with decentralization of duties can be beneficial for academic missions with poorly hardened electronics. Redundant solutions resorting to multi-master electrical BUS types like the CAN-bus could allow multiple subsystems to interface with the radio transmitter(s) with an increased resilience to single subsystems failures.

The Management activities conducted during the project highlight some good practices along product life cycle:

- Establishing a permanent space project in universities increases the space missions and systems design know-how of students and professors. The heritage from previous programs constitutes a formidable starting point for new projects because it increases the confidence level and limits the waste of time and resources.
- The importance to establish a precise and overlapped schedule, with well-defined milestones and tasks to be completed in order to guarantee no/low delays. Interlacing design, development and testing in a sort of multi-vee approach allows early test to support the choices of the detailed design.
- The importance of the model philosophy and its effective application have high relevance to gain time. The use of breadboards proves the validity of new parts of the design never developed before and prevents future issues during the implementation and verification phases.

Finally, some educational aspects emerged:

- A multidisciplinary approach requires cross-skill and different knowledge. Hardware and software are prevalent and students with skills in electronic, communication and computer science engineering cover a relevant role. However, it is fundamental the contribution of students in aerospace engineering and system engineering: the former add specific know-how on avionics aspects and the latter give a relevant contribution on the requirements definition, AIV planning and execution and interface with all parts of the system.
- Students skills, attitude and motivation need to converge to create a team, not a group. Team means defining roles and responsibilities: everyone can contribute but each task has a responsible in charge that takes care the right and on time execution and reports the status of the activities to the management. In a short schedule, the work efforts and the level of know-how requested are very high. Knowledgeable students (and experienced researchers) with strong motivation and focus on the project are essential. If they are also friends it brings to valuable results but they must have complementary skills and be organized as a team, with specific roles deriving from technical competencies and soft-skills.

CRediT authorship contribution statement

Simone Bollattino: Writing – original draft, Validation, Software, Formal analysis. **Fabrizio Stesina:** Writing – review & editing, Validation, Supervision, Methodology, Conceptualization.

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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