

Lessons Learned and First Results of the E-Band CubeSat EIVE

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Lessons Learned and First Results of the E-Band CubeSat EIVE

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Abstract

The 6 Unit CubeSat EIVE was developed to host a novel high-speed E-band transmitter payload. Challenging aspects for reliable and demanding satellite architectures were revealed during development and operations. Reliable communication buses such as CAN, RS485 or point-to-point links should be preferred over I2C or SPI. Experiences from the implementation of the individual subsystems are presented and lessons learned derived in this paper. The on-board data handling and communication system is based on a system-on-a-chip architecture and uses the CCSDS protocol via S-band. The FPGA section proved advantageous while the use of high-priority commands is essential for in-orbit debugging. The conservative approach to the design of the power system and the high effort to test the thermal design of the EIVE CubeSat led to a stable satellite regarding the power-budget and temperature ranges. The self-developed solar panels and their deployment mechanism worked as intended with many lessons learned in the process. A major part of the time spent on integration was on the harness. The attitude control system is based on self-developed components and commercially acquired units. With most of the attitude control components being tested and calibrated on a best effort basis, more resources should have been committed to testing the GNSS receivers and the star tracker on ground. A stable safe mode was achieved after multiple bugs fixes in the first weeks after the launch. Since then, steady progress towards nominal operation has been achieved.

Keywords

6U CubeSat; Lessons Learned; Satellite Bus Design; Launch and Early Orbit Phase; Flight Results

Abbreviations

ACS	Attitude Control System
ACU	Array Conditioning Unit
ADC	Analog to Digital Converter
CAD	Computer Aided Design
CAN	Controller Area Network
CCSDS	Consultative Committee for Space Data Systems
COTS	Commercial-off-the-Shelf
DAC	Digital to Analog Converter
DLR	German Aerospace Center
DRO	Dielectric Resonator Oscillator
EIVE	Exploratory In-orbit Verification of an E-band link
EPS	Electric Power Supply
FLP	Flying Laptop
FPGA	Field Programmable Gate Array
GaAs	Gallium-Arsenide
GaN	Gallium-Nitride
GNSS	Global Navigation Satellite System
GPIO	General Purpose Input Output
HDRM	Hold Down and Release Mechanism
HPA	High Power Amplifier
HPC	High-Priority Command
I2C	Inter-Integrated Circuit

IAF	Fraunhofer Institute for Applied Solid State Physics
ILH	Institute of Robust Power Semiconductor Systems
IRS	Institute of Space Systems
ITU	International Telecommunication Union
LED	Light Emitting Diode
LEOP	Launch and Early Orbit Phase
LTAN	Local Time of the Ascending Node
MPA	Medium Power Amplifier
MEMS	Microelectromechanical systems
MOSFET	Metal-Oxide-Semiconductor Field-Effect Transistor
OBC	On-Board Computer
OBSW	On-Board Software
PCB	Printed Circuit Board
PCDU	Power Control and Distribution Unit
PDU	Power Distribution Unit
PLOC	Payload On-Board Computer
PLPCDU	Payload Power Control and Distribution Unit
RF	Radio-Frequency
RTD	Resistance Temperature Device
RPG	Radiometer Physics GmbH
SDR	Software Defined Radio
SCEX	Solar Cell Experiment

SoC	System-on-a-Chip
SPI	Serial Peripheral Interface
SUS	Sun Sensor
TCS	Thermal Control System
TESAT	Tesat Spacecom GmbH
TLE	Two-Line Element
TAS	Thales Alenia Space Germany
UART	Universal Asynchronous Receiver-Transmitter
UTC	Universal Time Coordinated
QTH	Quartz Tungsten Halogen

1. INTRODUCTION

1.1. Motivation

A new 6 Unit CubeSat bus has been developed within the scope of the Exploratory In-orbit Verification of an E-band link (EIVE) project from 2019 until 2022 at the University of Stuttgart's Institute of Space Systems (IRS) [1, 2]. The fully integrated satellite is depicted in the centre of the EIVE-Team in Figure 1. The EIVE project is conducted by a consortium of two institutes of the University of Stuttgart, the Institute of Robust Power Semiconductor Systems (ILH) and the IRS, the Fraunhofer Institute for Applied Solid State Physics (IAF) and two industrial partners, Radiometer Physics GmbH (RPG), and Tesat Spacecom GmbH (TESAT). Thales Alenia Space Germany (TAS) and AZURSPACE support the EIVE project. Following the completion of the integration, the verification campaign of the satellite bus was finished in early 2023. The EIVE CubeSat was subsequently launched from Vandenberg Space Force Base as part of the SpaceX/Transporter-8 ride share mission on the 12th of June 2023 into a sun-synchronous orbit of 529 km altitude and 1:18 Local Time of the Ascending Node (LTAN). The EIVE satellite was identified as object 23084F (COSPAR) / 56937 (NORAD) in the days after the successful deployment of the satellite from Exolaunch's Exopod Nova. The Launch and Early Orbit Phase (LEOP) was completed two weeks after launch due to stable radio contact being established, confirmation of the successful solar panel deployment and the affirmation of a stable power budget and thermal management. The commissioning phase is still ongoing at the time of publication due to improvements to the Attitude Control System (ACS) controllers and bug fixing.

During the development and testing of the EIVE CubeSat, many lessons were learned that might be of interest to the community. The following sections thus offer insights into the experiences made during the design and implementation of the EIVE satellite bus and its individual subsystems, as well as recommendations derived from the verification campaign and the first flight results.

1.2. The EIVE Mission

The EIVE CubeSat is a technology demonstration mission with the main payload being an 71–76 E-band transmitter with an analog bandwidth of up to 5 GHz and a maximum data rate of up to 10 Gbit s⁻¹ depending on the modulation format [3]. The objective of the E-band transmitter payload is to investigate the radio link quality with multiple signal modulations under different conditions like weather, slant range and elevation above the horizon [4, 5]. The E-band payload consists of two custom-designed Gallium-Nitride (GaN) solid-state power amplifiers and a Gallium-Arsenide (GaAs)-low-noise pre-amplifier, developed at IAF and integrated by RPG [6]. To achieve a positive level balance in the power plan [4], a horn antenna is employed



Fig 1. Integrated EIVE CubeSat with deployed solar panels in-between team members in the IRS clean room.

due to its high directivity. A powerful and compact Field Programmable Gate Array (FPGA)-based digital processing unit, developed by TAS [7], is employed in conjunction with a 12 GSps Digital to Analog Converter (DAC) extension board to drive the analog high-speed link. This payload computer serves as a miniature arbitrary waveform generator that stores, compresses, frames, and modulates images and pseudo-random bit sequences for transmission over the E-band link. Moreover, a 4k video camera live stream via the E-band transmitter is planned [8]. At the receiver, a 1.2 m diameter Cassegrain antenna system is placed on a pedestal together with the analog front-end and the digital signal processing [8].

An additional payload successfully characterises ten multi-junction solar cells of the company AZURSPACE by gathering IV curves, cell temperatures and total ionising doses.

2. SYSTEM ARCHITECTURE

A detailed block diagram of the EIVE satellite can be found in Figure 2. The components that were developed during the EIVE project are:

- the On-Board Computer (OBC) interface board, which connects the Xilinx Q7S OBC to the satellite bus - IRS
 - an interface board that connects multiple ACS sensors and actuators to the bus - IRS
 - a Thermal Control System (TCS) board for temperature read out and heater control - IRS
 - the ACS board which carries the magnetometers, the gyroscopes and the Global Navigation Satellite System (GNSS) receivers - IRS
 - 3-axes sun sensors - IRS [9]
 - the solar panels Hold Down and Release Mechanism (HDRM) - IRS
 - the E-band transmission chain - IAF, RPG & ILH [5]
 - a Payload On-Board Computer (PLOC) - TAS, ILH [7, 8]
 - a solar cell characterisation payload - IRS, AZURSPACE
- Most of the digital components are located in a two unit PC104 stack with the remote units connected via the harness to the individual location thereof.

The following subsections will introduce the chosen chips selection approach for the self-developed components and show the experiences that have been made with using different bus systems on the EIVE CubeSat.

2.1. Integrated Chip Selection

The EIVE satellite bus is built in the context of a CubeSat project and as such, the resources to built fully radiation

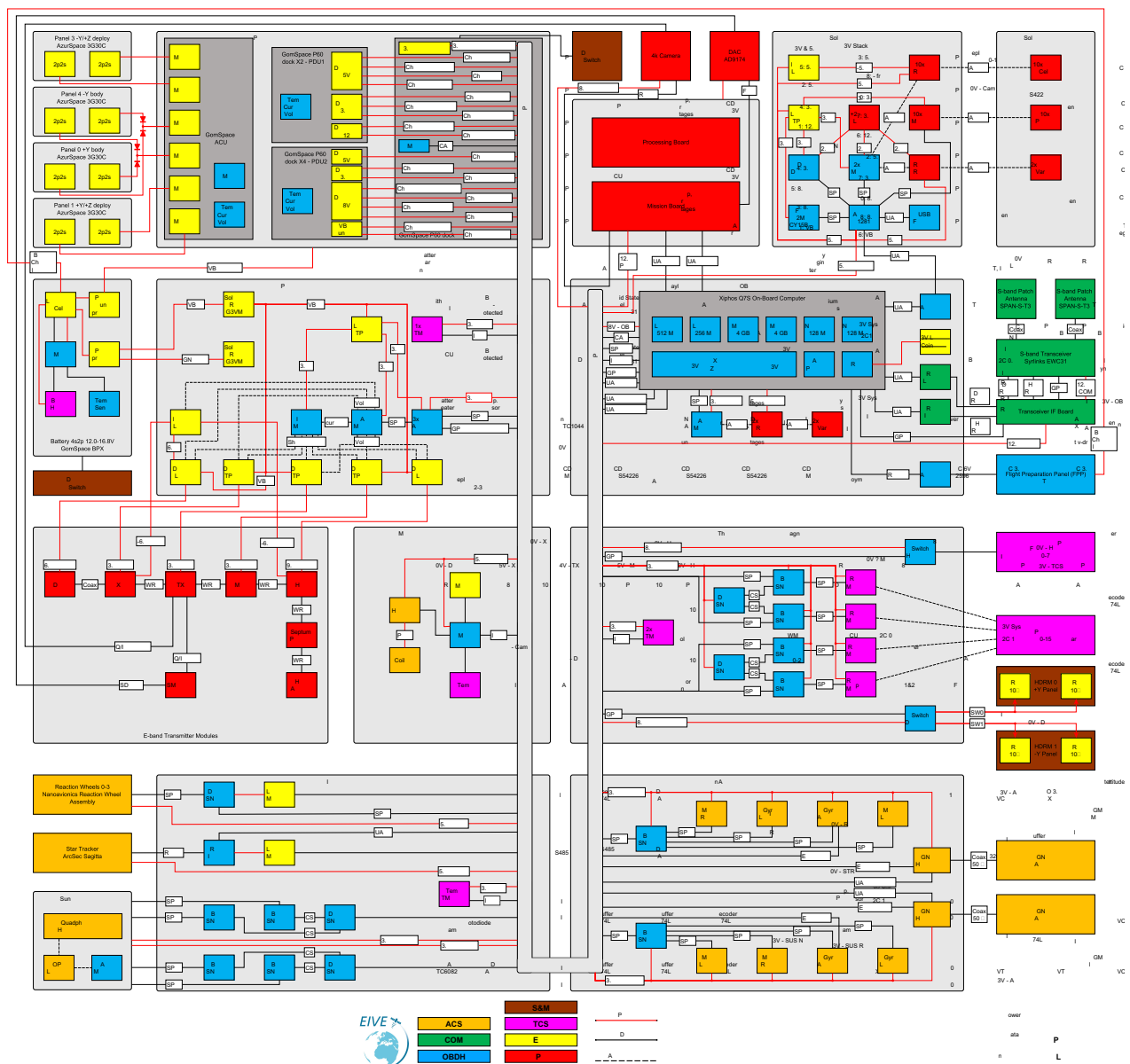


Fig 2. Block diagram of the EIVE CubeSat bus.

hardened and meticulously qualified systems are not available. Instead, a best-effort approach to the selection and testing of components was employed with most of the components having either space heritage or being related to similar parts with space heritage. As such, most components are either Commercial-off-the-Shelf (COTS) parts or self-developed.

The approach to selecting chips is an adaptation of Sinclair et al.'s "careful COTS" proposal [10] where commercially available chips are selected taking into account the radiation tolerance of the chips as determined by total ionising dose and the single event effect tests. The accumulated total ionising dose of the EIVE CubeSat is 10 krad/year behind a 1 mm aluminium solid sphere shield as estimated with the OMERE tool. Due to the lack of funds to conduct own radiation testing, test results from the proceedings of the IEEE Radiation Effects Data Workshop were utilised. A great resource are the annual reviews of radiation tested parts from the previous workshop by D.M. Hiemstra [11]. These compiled lists are a good starting point to find integrated chips which are suitable for the specific mission under design.

Due to the Covid-19 pandemic and the ensuing chip shortage, the stock of many specialised chips decreased dramatically in the years 2020–2023. This unforeseeable issue affected the development process of the EIVE CubeSat bus, which was transitioning in 2020/2021 from the prototype to the engineering/flight model production phase. Many of the chips, that were selected during the prototyping phase, became rare or completely unavailable, which led to design changes after the original design freeze. In hindsight it would have been best to purchase a conservative amount for all selected chips during the prototyping phase. As a rule of thumb, at least four times the amount of chips that are necessary to equip all initially planned Printed Circuit Boards (PCBs) (prototype boards, qualification/engineering/flight models, backup models, ground testing hardware, etc.) proved to be adequate for the EIVE project.

2.2. Communication Buses

2.2.1. CAN bus

The core components of the EIVE satellite bus are the OBC, the Power Control and Distribution Unit (PCDU) and the S-band transceiver. The communication between the OBC and the Gomspace P60 PCDU works via a Controller Area Network (CAN) bus with an SN65HVD232 interface chip. The SN55HVD, a radiation hardened variant of the SN65HVD232 chip, was not selected due to procurement issues, yet might be of interest for other designs. Overall, the CAN bus has proven to be easy to implement and has worked flawless in both on the flatsat as well as on the integrated flight model. While no networks with larger amounts of participants were investigated, CAN buses can be recommended for new CubeSat bus designs.

2.2.2. UART with or without RS422/RS485 interfaces

The communication between the OBC and the Syrlinks EWC31 S-band transceiver is established via three full-duplex RS485 lines. The housekeeping data is routed via an ISL32600E transceiver which proved adequate by radiation testing of Irom et. al [12]. The two full-duplex clock and data lanes of the S-band transceiver are routed via an LTC2872 chip, which has a higher speed rating, and was radiation tested by Irom et al. [12] and Allen et al. [13]. The Universal Asynchronous Receiver-Transmitter (UART) lines necessary for this RS485 network in the OBC are established by FPGA cores that directly feed the data into the Consultative Committee for Space Data Systems (CCSDS) IP cores. Once these RS485 links had been established during the flatsat phase, they ran up to the time of writing without any issues. Thus, it can be concluded that RS422/RS485 point-to-point links or networks with two participants are a stable solution that can be recommended for future designs.

However, back-powering or also called parasitic powering can be an issue of UART buses. This can occur if no transceiver chips or digital isolators are used and one of the connected devices is powered off. In this condition, current may flow via an active-high TX line to the powered-off component. Internal pull-up resistors or protection diodes of chips connected to the UART's RX line can feed small amounts of current into the supply voltage line which might be sufficient to power low-consumption chips as visualised in Figure 3a and Figure 3b. Back-powering can cause components to not fully shut down, which might hamper with resetting parts, or affect the start-up of systems. This was the case with the UART connection between the OBC and the multiMIND PLOC [7]. In contrast to other areas, no digital isolators or transceiver chips were used here. The consequently occurring start-up issues of the PLOC were solved by switching the OBC's TX lines to high-impedance (tri-state) when the PLOC was powered off.

Another lesson learned for UART buses concerns baud rate errors. Many simple microcontrollers such as the ATmega128, used in the Solar Cell Experiment (SCEX), and even high-end devices such as the Zynq7020 soft cores have integer numbers as clock dividers. This leads to baud rate errors since many specific baud rates cannot be replicated with accuracy. A mismatch between the clock rates in UART transmissions can be often tolerated for short transmissions, however, long transmissions can contain bit errors produced by the clock rate mismatch. The baud rates of microcontrollers and FPGA UART interfaces

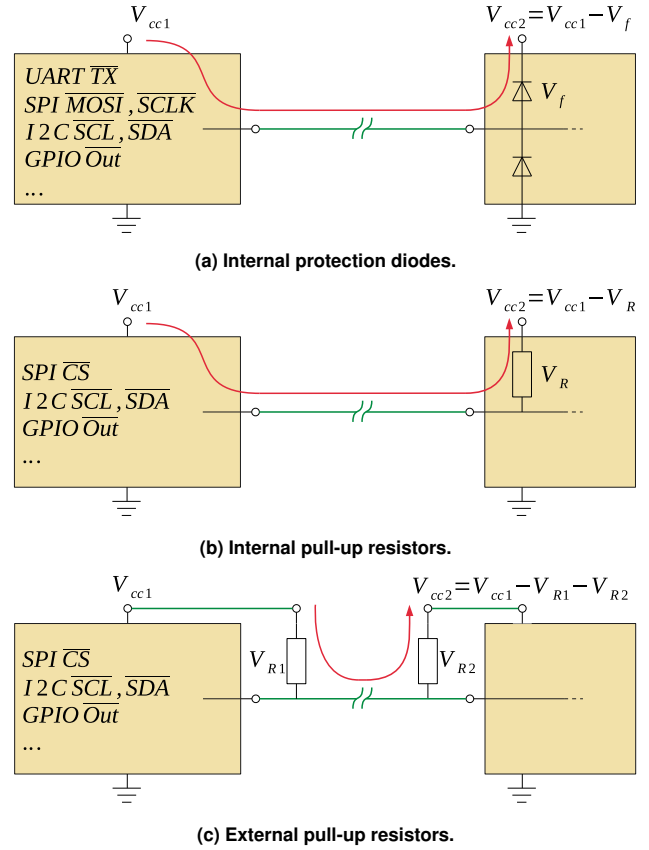


Fig 3. Examples of back-powering for chips / PCBs with split power supplies.

should thus be carefully selected and matched especially if no floating point clock dividers are available on the selected hardware.

2.2.3. I2C

Inter-Integrated Circuit (I2C) buses are simple multi-master buses that can be found as interfaces for many chips. While I2C buses are easy to implement, it must be stressed that this bus can get stuck if any of the participants blocks the bus by an incomplete transmissions or if noise is interpreted as flanks by the listeners. If the I2C state machine of a device is stuck, chips must be either power cycled or brought into a defined state by dedicated monitoring mechanisms. Another drawback is the bus capacity. I2C buses are meant to operate with a small number of chips on a single printed circuit board. An I2C bus with many devices and/or long lines, e.g. by routing the bus through the satellite, might have a high bus capacity and therefore need countermeasures such as slower transmission rates, lower pull-up resistor values or buffer/repeater chips. Similar to the UART buses, parasitic powering via external or internal pull-up resistors might be an issue if remote chips are powered-off but still connected to the SDA/SCL lines as shown in Figure 3c. It has been previously reported that I2C buses led to the failure of CubeSats and thus should be avoided for space applications or at least accompanied by fail safe mechanisms [14–16]. The authors of this paper share this assessment and strongly advice the CubeSat community to avoid I2C buses. Unfortunately, both the selected Gomspace BPX battery and the ISIS iMGTO magnetorquer board only provide I2C interfaces which had to be subsequently used for the communication with the OBC. The second I2C bus on

EIVE connects multiple TMP1075 temperature sensors that are placed throughout the satellite stack to the OBC. A failure of this bus can be tolerated since these temperature sensors are non-critical for the operation of the satellite. A critical malfunction of the EIVE satellite's I2C bus, that had not been observed on ground before, occurred shortly after deployment. The I2C bus connecting the OBC, the magnetorquer board and the battery ceased to work, thus preventing telemetry to be read from both devices and control torques to be sent to the magnetorquers. The I2C bus became functional again for a certain time in the range of minutes to multiple days after rebooting the On-Board Software (OBSW) on the OBC. This issue could be circumvented by switching the I2C buses from the FPGA I2C core with the hardware core of the Zynq 7020. While it is not clear what the failure mechanism of the I2C FPGA soft core is, it is advised to avoid I2C FPGA soft cores.

2.2.4. SPI

A Serial Peripheral Interface (SPI) bus is a simple bus designed to operate between chips on a PCB at short distances. It has separated lines for the input (MISO) and output (MOSI) of the bus master controller, provides a clock line (SCLK) and activates the connected devices by pulling the chip select pins (CS) on the connected devices low. A drawback of the SPI bus is the large amount of chip select lines that need to be connected to pins of the master-OBC. While the Zynq7020 offers many pins, the maximum amount would have been exceeded by the satellite bus. Thus, SN74LVC multiplexer chips were used to reduce the amount of chip select pins. As already mentioned in the UART and I2C paragraphs, back-powering is an issue for buses with no electrical isolation if the connected chips can be powered off. Moreover, it must be avoided that a single faulty chip can corrupt the entire bus and thus all attached devices. To tackle these issues, every four SPI devices are connected to a single SN74LVT buffer chip with the chip selects being tied to the buffer enable lines and the chip selects having a buffered line pulled to ground as shown in Figure 4. This solution fulfils multiple functions. Firstly, it acts as an isolator that only connects remote chips to the OBC, if the OBC is pulling the chip select line low. Thus, only a single chip is connected to the bus at a time which prevents potential interferences with other chips. Moreover, the buffer chips prevent parasitic powering since the buffer lines are high impedance if powered off. The signal integrity also improves since the buffer chips reduce the capacity on

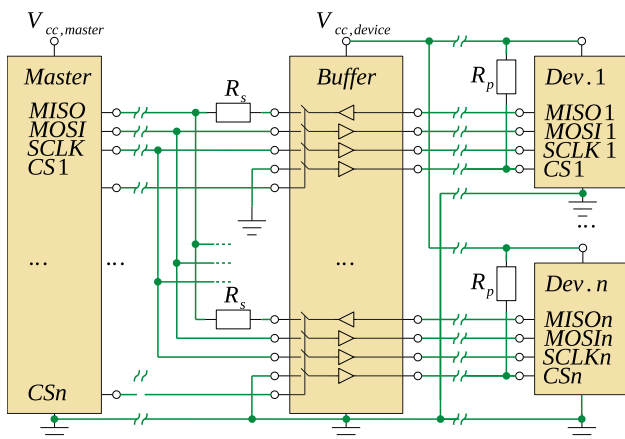


Fig 4. SPI bus buffering and branching.

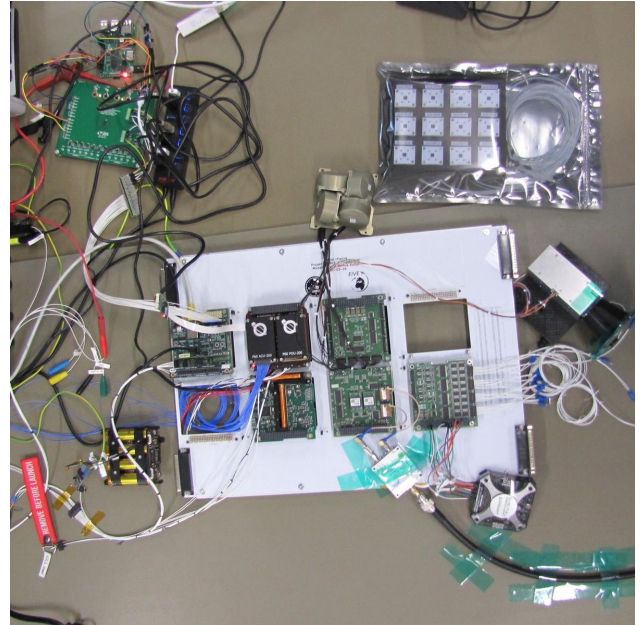


Fig 5. EIVE flatsat set-up with the white flatsat PCB adapter.

the OBC side of the SPI bus and produce sharp flanks at the chip side due to their low output impedance. The master-in (MISO) line is the critical line since all chips on the bus can potentially write on it. While the output enable signals on the buffer should not allow multiple devices to write on this line, there were issues during the development where multiple chip selects were pulled low at the same time by error. To prevent short circuits on the MISO line, 100 Ω serial resistors were introduced to limit the current flow in these fault cases. It must be noted, that these resistors must be small enough to keep the signal integrity at the desired bus speed intact.

The flatsat set-up had several SPI communication issues. Figure 5 shows that all PCBs were arranged side-by-side on an adapter PCB with PC104 connectors. The total length of the signal lines was with approximately 1 m close to the recommended maximum length for serial buses such as SPI or I2C. All inconsistent communication issues vanished in the stack set-up with a maximum length of the SPI traces in the range of 0.2 m and up to 0.4 m for the sun sensors. The lesson learned derived from this is to reduce cable/trace lengths not only on the flight model but also in test setups to prevent signal integrity issues.

2.3. General Recommendations

Even though most OBCs can handle many low-level hardware devices, it is not recommended to build a centralised system by routing all low-level sensors directly to the OBC. This assessment stems from the many issues encountered and the relevant amount of time needed to implement the software to communicate with all devices within the self-developed software framework. One notable struggle was to find a polling sequence that allows all sensors to be read out within the timing constraints of a real time operating system. Especially the 400 ms polling interval of the sensors of the ACS lead to strict time constraints that made multi-threading necessary in the OBSW. Having a centralised system means that the satellite bus is reconfigurable down to the physical communication layer which enables low-level bug fixes. This advantage, however, is almost negligible if

the system has been well-tested on ground rendering low-level programming changes in orbit unnecessary.

A better system design approach is to put a single or redundant set of low-power, high-reliability microcontrollers on every printed circuit board that are connected to the local chips and sensors. Thus, information via potentially unstable bus systems such as I2C and SPI can be gathered locally and then transferred through the satellite with a bus with higher reliability such as CAN, RS485, spacewire or similar.

3. ON BOARD DATA HANDLING AND COMMUNICATIONS

3.1. On-Board Computer and On-Board Software

The EIVE CubeSat utilises a Q7S OBC built by Xiphos Technologies. This OBC is comprised of a ProASIC watchdog that monitors a Zynq 7020 System-on-a-Chip (SoC). The Zynq 7020 has a powerful ARM dual-core processor as well as an FPGA section. The OBSW reuses the Flying Laptop (FLP)'s flight software framework [17, 18] with major adaptations to comply with the EIVE platform's requirements and hardware constraints. Moreover, the operating system on the OBC is an embedded Linux built with Yocto instead of the RTEMS used on FLP.

3.1.1. System-on-a-Chip OBC

The experience with using a high-performance (for space application) processing chip has been overall positive. Neither the processing power, the non-volatile memories or the random access memories have been a limiting factor during the software development. Moreover, the FPGA part is useful for plenty of purposes. One of them was the implementation of additional soft core peripherals so that single buses could be separated from each other to limit the amount of devices and thus avoid bus capacity issues and limit fault propagation. Most importantly, it enabled the option to include the CCSDS protocol's telemetry and telecommand FPGA IP cores directly in the programmable logic of the OBC instead of having dedicated hardware such as in the FLP. Nevertheless, the FPGA section increases the complexity of the software development by far and requires additional skill sets for the software developer.

While some of the Linux operating system features such as the file system, the time management or the GPSSD service are attractive to the software developer, other issues opened up along the way. One example was that certain hardware interfaces such as the I2C bus of the platform are directly controlled by the Linux operating system. In case the Linux based implementation is insufficient for the tasks at hand, adaptations in the Linux kernel might become necessary.

3.1.2. High Priority Commands

Part of the heritage of the IRS's FLP small satellite was the use of High-Priority Commands (HPCs). A HPC bypasses the OBSW and can trigger basic actions on the satellite without the need for a running or functional software image. The FLP routes HPCs from the transceiver to a CCSDS board and could then trigger basic actions like rebooting the OBC or the PCDU [19]. Unlike the FLP, EIVE has only a single HPC that is able to reboot the OBC which then starts the OBSW's golden image. The golden image is the last image flashed on ground and should always work. The

HPC is a 144 bit sequence that is sent non-inverted and inverted via the S-band uplink. The output of the S-band transceiver's uplink line is routed to both the Zynq 7020 SoC chip as well as its ProASIC watchdog with a Y-trace. The watchdog monitors the telecommand line for the unique bit sequence and triggers a reboot to the golden image if the bit sequence is identified. During the first weeks in orbit, multiple updates of the OBSW, the FPGA firmware and the Linux device tree were necessary. Unfortunately, mismatches in the versions of these three update components resulted in stuck OBSW images. This was solved by sending the HPC bit sequence to triggered a reboot back to the golden image. A HPC is also superior to a software watchdog timer since either the time to kick in is longer than reasonable during flights over the ground station or the software could get stuck in states where it resets the watchdog timer but ceases to communicate via the radio equipment. The latter occurred while implementing a fix of the high-data rate mode while in orbit. The authors thus encourage developers of new CubeSat buses or on-board computing hardware to implement similar fall-back features that allow immediate reboots despite possibly corrupted OBSW images.

3.2. PLOC

While this section focuses on the lessons learned from the viewpoint of the platform, more information on the multiMIND PLOC can be found in Pawlitz et al. [7] and Manoliu et al. [8, 20]. The Xilinx Ultrascale+ of the PLOC is of interest to the EIVE payload since it offers the high-speed JESD204b interface needed to operate the high-speed AD9174 DAC. Using a System-on-Chip instead of a bare FPGA gives the advantages of both worlds. One ARM core is used to implement a bare metal software. This approach is easier than implementing the control logic in an FPGA. The FPGA size allows to meet the timing of the high speed interface and to fulfil other functions in parallel. Nevertheless, these resources come at the price of a high power consumption of 15W and thus high thermal dissipation. This poses a significant challenge to the satellite bus. Hence, it is recommended to select high-performance components also based on the power consumption and to consider the implementation of power saving methods such as powering off unused processor cores and FPGA sections to save energy.

4. COMMUNICATION SYSTEM

As mentioned before, EIVE has a Syrlinks EWC 31 S-band transceiver (2083.6 MHz uplink, 2263.9 MHz downlink) together with two Anywaves SPAN T-3 patch antennas. The coverage of approx. 78% availability both in up- and downlink with the IRS ground station, that was reported by Koller et al. [2], could be confirmed in orbit. Temporary losses of the uplink and more rarely the downlink occur when the satellite's antennas are pointing perpendicular to the IRS's ground station antenna. During the LEOP, the DLR (GSOC/DFD) ground stations Weilheim, Inuvik and O'Higgins were used. Due to their larger antenna diameter and higher transmitter power, no signal integrity issues were observed in both the uplink and downlink. If the link budget cannot be closed over the entire full-sphere of the satellite, it is thus highly recommended to include additional ground stations with a higher effective radiated power during the satellite's LEOP. Nonetheless, additional ground stations increase the network complexity as well as the frequency coordination. It is highly recommended to

perform end-to-end tests for both the uplink and downlink with every ground station involved as was the case for project EIVE.

The tests of the S-band system for project EIVE started two years before the launch and the final patch of the high-data rate downlink was tested two weeks after launch. It is not uncommon to need a lot of time to test and implement the communication system which is critical to the operation of the satellite. Thus, a lesson learned is to start as early as possible with it. During the flatsat test phase, Software Defined Radios (SDRs) were used to emulate the ground station side of the communication link. While it generally is a good idea to employ SDRs for their versatility when debugging or optimising Radio-Frequency (RF) links, they offer additional challenges of their own. One of those was that it was hard to integrate the SDR into the tool chain since it affected the real time environment by introducing delays and having message queues that grew faster than they could be processed.

Using the CCSDS protocols increased the complexity of the overall communication system and needed the procurement of specialised CCSDS FPGA cores. Nevertheless, it was time well spent since the CubeSat is now compatible with the industry standard which facilitates and enables the use of external ground stations and leads to a standardised packaging system that is comprehensible and reusable.

Concerning the frequency filing with the International Telecommunication Union (ITU), it is advised to start the licensing process as early as possible to avoid time troubles due to the lengthy process. First talks with the ITU and the national regulatory body for radio frequency allocation should occur before or during the time when the transceiver hardware is bought or built since the frequencies selected at this stage might be hard to change at a later stage.

5. ELECTRICAL POWER SYSTEM

5.1. PCDU and Battery

The Electric Power Supply (EPS) of the EIVE CubeSat consists of a GomSpace P60 PCDU with one Array Conditioning Unit (ACU) and two Power Distribution Units (PDUs). Compatible to this, a GomSpace BPX 77 Wh lithium ion battery system is flown. The overall experience with these products was positive. Apart from an connector issue as detailed in Section 8.2, the unit has been functional at all times and successfully passed thermal vacuum and vibration tests without any issues. Having many controllable power switches such as with the PDUs of the P60 system enables the creation of a satellite bus where any system can be power-cycled without restarting the entire satellite. Power switches are crucial since CubeSat components can come into faulty states which usually can be recovered by power cycles. The safety features such as the ground watchdog, the latch-up protections and the power harvesting mode for critical battery voltages offer an adequate level of security for a satellite of its size. The ACU of the satellite showed a good ability to track the maximum power point of the solar panels both on ground as well as in orbit. It is recommended to acquire a solar array simulator such as the Keysight E4360A to stimulate and test the operation of the solar power inputs of the PCDU system. A critical bug of the ACU was encountered this way. For solar panel currents close to the input current limits, the ACU would measure a wrong battery voltage of up to 18 V and cease to load the battery any longer. While this might be a test set-up related

issue, it can be recommended not to operate PCDU systems close to the maximum values given in the datasheets. Since the PCDU and battery are the backbone for the functionality of all systems on-board of a satellite, it is recommended to invest time into the selection of a suitable product. Besides the electrical compatibility with the satellite bus, the overall security concept of the satellite and thus the survivability of the satellite itself is defined by the safety features of the PCDU.

5.2. Payload PCDU

The RF chain of the E-band payload consists of five modules with different requirements concerning their respective power supplies [5]. Many of the needed voltages, listed in Table 1, are uncommon which led to the development of a dedicated power supply, the Payload Power Control and Distribution Unit (PLPCDU).

Furthermore, the sequence of activation of the RF components requires all power channel to be switched independently. LM2596S DC/DC converters were chosen for the High Power Amplifier (HPA) and Dielectric Resonator Oscillator (DRO) supply. This was motivated by the output current and voltage range as well as the high heat sinking capability necessary to cope with the up to 3 W/10% of loss of the HPA DC/DC converter. The LM2596S was tested by Kessarinskiy et al. to a total ionising dose of more than 70 krad [21]. The high-efficiency, high-current TPS54226 converters are used for the 3.3 V lines of the Medium Power Amplifier (MPA), transmitter and frequency multiplier. Relevant radiation tests on the TPS54226 chips are performed by Allen et al. [22]. Currents are measured with precision shunts via INA195 current amplifiers, voltages are sampled via voltage dividers and both are digitised with a MAX1229 Analog to Digital Converter (ADC). This ADC was used on multiple self-developed PCBs such as the sun sensors and the solar cell experiment and has shown a convincing performance as a general purpose SPI ADC. The negative voltages that are necessary for the transistor gates of the X8 frequency multiplier and the HPA are created by inverting the 6 V DRO voltage with a LTC1044 charge pump. It must be noted that these charge pumps have a very limited current capacity and need longer during startup than most DC/DC converters which is relevant since especially the HPA needs to have a negative gate voltage before power is applied to it.

Gappad fillers are attached to the top of the DC/DC converters and close the 1 mm gap to the aluminium heat spreader as shown in Figure 6. This has proven effective in reducing temperature peaks if the PLPCDU is supplying the E-band RF components for up to 10 min.

A critical decision of such a high-power system is to design the shut-off and electrical insulation of the system. Since the PLPCDU is directly attached to the battery, it needs to

	$V_{CC,1}$ [V]	$I_{CC,1}$ [mA]	$V_{CC,2}$ [V]	$I_{CC,2}$ [mA]
Dielec. Res. Oscillator	6.0	20-25	-	-
X8 Frequency Multiplier	3.5	20-80	<-5	<10
Transmitter / Mixer	3.4	200	-	-
Medium Power Amplifier	3.5	540	-	-
High Power Amplifier	9.8	2500- 3000	<-5	<10

Table 1. Supply voltages and currents of the E-band parts.

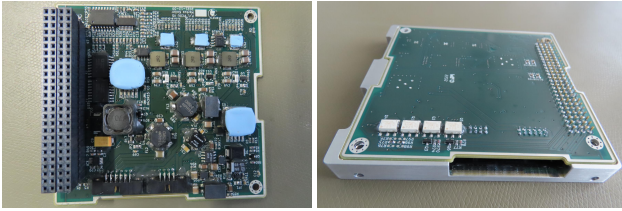


Fig 6. The PLPCDU without and with its heat spreader.

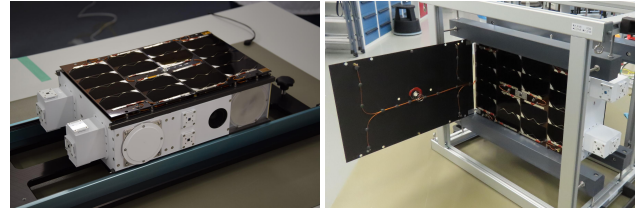


Fig 7. EIVE solar panels with harness.

be reliably disconnected from the battery if it is not active. Therefore, G3VM-31HR1 optocoupler solid state relays in a redundant configuration were inserted into the positive power rail. Solid state relays have no moving parts and only consume a small amount of power to keep the power line on. Since the optocoupler's Light Emitting Diodes (LEDs) will slightly deteriorate over time, it is recommended to be conservative with the LED current in the design process. The pins of the ADC's SPI bus as well as all General Purpose Input Output (GPIO) pins are insulated from the rest of the bus with ADUM3401 digital isolators. This is mandatory since the high power draw from the PLPCDU can lead to shifts of the reference ground which in turn can lead to unwanted signal interactions with the rest of the satellite bus connected to the P60 PCDU.

Generally, it is recommended to make a grounding scheme of the entire satellite system and identify where different ground potentials can occur. For CubeSats, the star ground topology [23] with the battery at the centre is a typical scheme and was thus chosen for EIVE. Typical points of concerns of the star grounding scheme are the connections to the chassis, which were made for EIVE via 1 M Ω /1 nF R/C elements, deployment switches, that cut off power rails, and the RF components, that typically tie their signal ground directly to the chassis.

During the development and testing it was noticed that upon start-up of the OBC, a large consumer was draining the battery for a fraction of a second. It was determined that some GPIO pins were pulled to the high state during the boot configuration of the SoC's FPGA. This led to the PLPCDU being temporarily powered and some RF components to switch on. This could be solved by disabling the power supply line for the LEDs of the optocoupler. The lesson learned in this case was that it would have been better not to route the SPI signals and the GPIO pins directly to the OBC and instead implement a local microcontroller with e.g. a CAN bus interface. This could have facilitated the control of the unit and prevented some of the activation issues encountered.

5.3. Solar Panels

The solar panels of the EIVE CubeSat were developed in a cooperation between the company SpaceTech Immenstaad GmbH and the IRS. The AZURSPACE 3G30 cell assemblies were connected in a 4s2p configuration with two such arrays per panel. The substrate of the solar panels is a 1 mm [90/90/0/0] carbon fibre reinforced polymer with a Kapton cover layer onto which the solar panels and their interconnects are glued. The attachment of the solar cells is difficult due to the fragility of the cell arrays themselves. Also, know-how is needed to manufacture bonds that can tolerate the mechanic stresses due to the different coefficient of thermal expansions with cycle amplitudes of up to 200 K. It is thus recommended to have solar panels assembled professionally. During the design, one of the main concerns was the gap between the body mounted and deploy-

able panels that had to be maintained under vibration loads to prevent damage to the underlying solar cells. Moreover, the Kapton insulated cables are routed on the backside of the solar panels as shown in Figure 7. Unforeseen issues arose at the points where the cables were routed through holes from the front to the backside. The cable bending radii and the epoxy glue attachment points led to a build-up of approximately 5 mm which had not been accounted for. This increased the total height of the panel assembly and led to the maximum CubeSat envelope of the vibration test adapter and deployment pod to be exceeded initially. Additional lessons learned on the mechanism of the solar panels can be found in Section 8.3.

5.4. Power Budget Lesson Learned

Arguably the most important lesson learned of this paper is that it paid off to be conservative with the power budget and plan for the tumbling case as good as possible. The average power consumption in safe mode as measured in orbit is approximately 5 W if the S-band transceiver is in receive mode and the magnetorquers and battery heaters are active. This is about half of the worst case assumption of the power budget of 10.17 W without margins [1]. If the S-band transmitter is activated, the power consumption rises to approximately 15 W. This power budget overestimated the OBC that draws 2.2 W instead of 3.6 W as assumed during the design. A reason for the big deviation is that the power consumption for SoC based OBC systems is dependent of the FPGA usage. This quantity is usually unknown at the beginning of the project and can even change while in orbit. Moreover, the power draw of the magnetorquers was overestimated since the average activity of this device is lower than estimated. It must be noted, that an active star tracker and reaction wheels would add approximately 3–5 W, however, these components are not active in EIVE's safe mode. Moreover, the sun-synchronous orbit parameters increased slightly with a start delay, that moved the LTAN 16 min towards a noon-midnight orbit which slightly increased the power yield.

Figure 8 shows data from the EPS system over half an orbit. The selected data contains one of the aforementioned issues, the I2C bug of the battery/magnetorquers/OBC, which occurs at 2:52 Universal Time Coordinated (UTC). This is why the BPX's battery voltage ($\leftarrow$) flatlines at this point in Figure 8 a. Before that, the safe mode controller pointed the deployable solar panels in the rough direction of the sun. The battery was charged until the cut-off limit of 16.3 V is reached as shown in Figure 8 a before the I2C bug occurred. The voltage curve reached its lowest points due to the elevated power draw when the S-band transceiver is used for transmissions and the satellite is in the eclipse. Figure 8 c shows a downward slope in the total generated power over the sun phase which indicates the battery being charged to the maximum capacity at which the PCDU limits the generated power to the consumed bus power. The input current on the strings as shown in Figure 8 d mainly de-

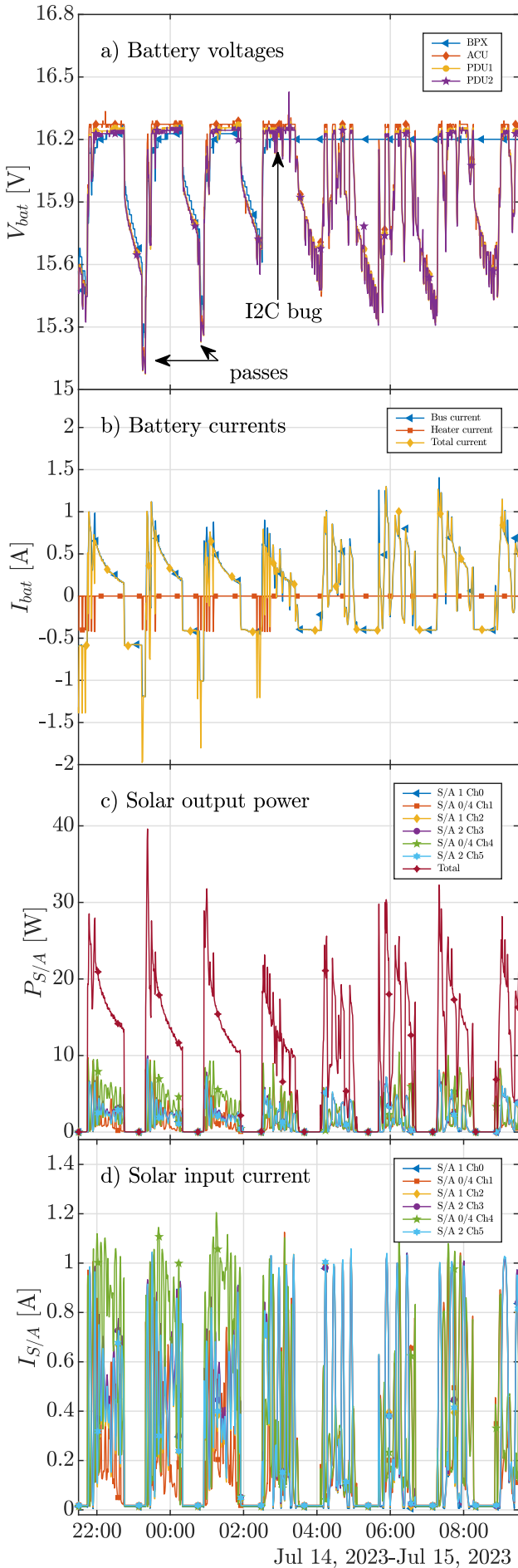


Fig 8. Battery and solar panel timelines over one day.

depends on the sun vector relative to the panels. The initially implemented ACS controller brought the deployable panels into view of the sun after which overshooting increases the error angle again with no stable pointing being reached during a single sun phase. If the I2C bug occurred, the distribution of the sun vector over the panels and thus the input currents are more or less random. The battery telemetry is lost after the I2C bug occurs, thus Figure 8 b no longer shows the battery heater current spikes (■). Nevertheless, the battery heaters are automatically controlled as visible in the battery voltage ripple during the eclipse phase in Figure 8 a.

6. ATTITUDE CONTROL SYSTEM

This section outlines some general lessons learned of the ACS system and presents experiences with the used equipment.

Early ACS system-engineering is important. The control-cycle should be designed before implementing the software for the ACS components. The late implementation during the EIVE project caused the need to rewrite ACS software components and reschedule their polling.

Moreover, it is recommended for future satellite bus developments to test the ACS algorithms within the OBSW. While this was possible with the simulator test bed of project FLP [24], cost and shortage of manpower prevented project EIVE to implement or adapt such a simulator. This increased the risk of bugs within the ACS algorithms and also the time needed for verification, compared to the simulator testing approach. Single scripts that compare the results computed in the OBSW to the results from the ACS developing environment are a good way to start, however, cannot replace full-scale simulators.

The GAFE tool [25] was used as such a development environment for the implementation and testing of the ACS algorithms. While GAFE itself is a great simulation tool, it cannot be recommended for ACS algorithm development since there are too many parameters that can influence the simulation results. Optimisations of the controller gains often were only applicable for certain start conditions and performed poorly for others. Moreover, the framework is difficult to work with and complicates clean coding and version control.

Furthermore, it must be recommended to have a dedicated person for the development and testing of the ACS controller. Having a dedicated ACS engineer from the start of the bus development could have certainly increased the chances to have stable ACS control from launch onward. Due to project constraints this was unfortunately not possible, however, would have helped to avoid last minute re-designs and lengthy in-orbit debugging.

The verification campaign was cramped due to the launch deadline and therefore errors which could have been detected slipped through. Testing sensors and actuators within the fully integrated satellite is irreplaceable while unit tests are only the first step. Additional information on the ACS testing of the EIVE CubeSat can be found in Eggert et al. [26].

6.1. Gyroscopes

The rotational rates of the EIVE CubeSat are measured by four gyroscopes in redundant pairs. The first kind are L3GDH Microelectromechanical systems (MEMS) gyroscopes. These COTS chips were chosen due to the availability, small size, and power consumption as well as the adaptable range of down to $\pm 245^\circ\text{s}^{-1}$ and thus

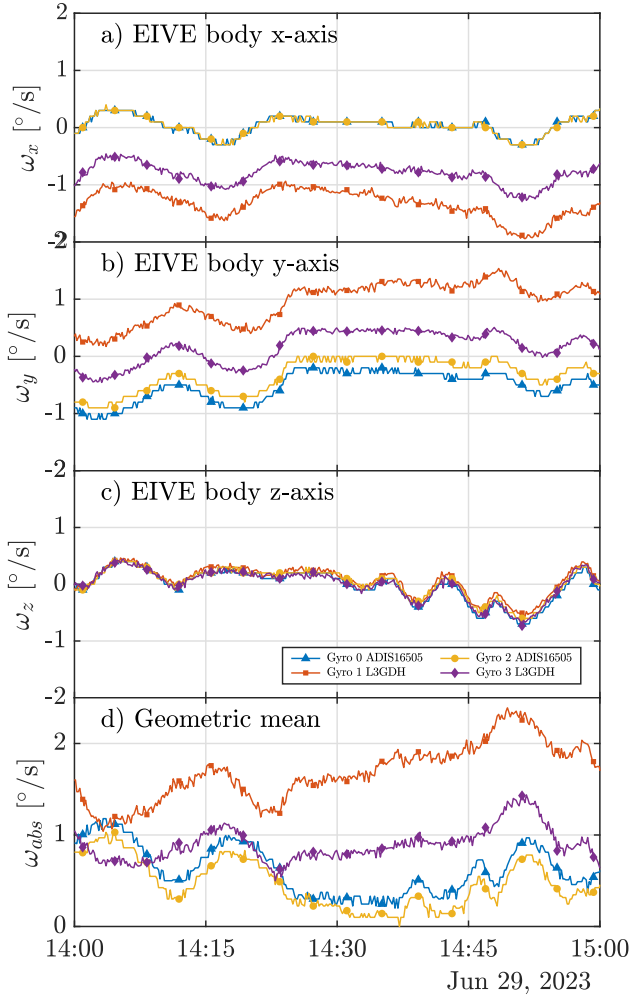


Fig 9. Gyroscope rotation rates with visible bias offsets.

adequate resolution. However, there was noticeable noise and bias of up to 1.5°s^{-1} when lying motionless in the clean room. The second type of gyroscopes flown on EIVE are ADIS16505-3 tactical grade inertial measurement units. While the accelerometer is not of interest to a CubeSat mission, the gyroscopes have a great precision and a device specific factory calibration. A drawback of this sensor was, that the model with a range of $\pm 125^\circ\text{s}^{-1}$ (ADIS16505-1) was not available at the time of manufacturing, thus the $\pm 2000^\circ\text{s}^{-1}$ version (ADIS16505-3) was selected. Although the resolution is limited in the relevant range of single degrees per seconds, both the offset and the noise of the gyroscope are small compared to the L3GDH. If COTS MEMS gyroscopes are selected as a technology for future missions, the authors recommend to use such high-precision, tactical grade gyroscopes with small rotational rate ranges, a small bias, reasonable bias stability and a good factory calibration.

The calibration of the gyroscopes included determining the individual bias of the devices by measuring the rotational rates of the motionless satellite in the clean room for extended periods. However, first flight results indicate that the bias values especially of the L3GDH gyroscopes but also for the ADIS16505 changed from the values on ground. This can be observed in Figure 9 where the rotation rates as measured by the individual chips are plotted for an hour. Whilst the ADIS16505 gyroscopes ($\triangle$, $\circ$) are in good agreement with each other, the L3GDH gyroscopes ($\square$, $\diamond$) have significant offsets which is a sign of a bad bias calibration. Interestingly, the values on the Z-axis

(local X/Y-axis for the gyroscope chips) are in good agreement for all gyroscopes. Temperature influences, but also mechanical degradation due to the vibration tests and the launch itself might play a factor in this change.

It is recommended to test the bias values of the individual gyroscope in a thermal vacuum chamber to determine the temperature impact since these values might not be provided by the manufacturer's datasheet. Additionally, it is a good idea to prepare for an in-orbit calibration campaign for MEMS gyroscopes.

6.2. Magnetometers

There are five magnetometers of three different kinds on the EIVE CubeSat. Two magnetoresistive LIS3MDL as well as two magnetoinductive RM3100 MEMS magnetometers were selected. An additional XEN1210 magnetometer is mounted on the ISIS iMGTO magnetorquer board. An adequate calibration of the magnetometers is of utmost importance since the magnetometers are influenced by the residual magnetic field of the satellite which must be compensated. For that purpose, a custom built Helmholtz cage was utilised for the calibration of the EIVE magnetometers. Information on the magnetometer calibration method can be found in Koller et al. [2]. After the calibration, the LIS3MDL and RM3100 were more or less in agreement with each other in homogeneous fields. Nevertheless, the RM3100 showed to have lower jitter and was easier to implement on the PCB since it has a bigger chip package.

While the calibration campaign on ground took longer than expected, the measurement values of the magnetometers in orbit are overall consistent and good enough to be used for actuation.

6.3. Sun Sensors

EIVE carries twelve self developed vector Sun Sensors (SUSs) which are arranged in pairs of two in all directions, as depicted in Figure 13 and outlined in Pentke et al. [9]. Testing the sensors has proven to be challenging as the only available high-intensity light sources at the IRS are a Quartz Tungsten Halogen (QTH) lamp and a construction floodlight. If the QTH lamp was put close to the sensor to increase the light intensity, the divergence of the light source became unreasonably high. If placed far away and shielded by baffles, the light could be reasonably parallelised, however, the overall intensity decreased. During the sensor calibration, a compromise was made at about a tenth of the intensity of the sun with a distance between the light source and the sensor of about one meter. The flight model tests with the SUS attached to the satellite's body showed that it was easy to determine if the SUS vector normal was implemented correctly. However, due to the immobility of the QTH light source, a floodlight was used to determine the correct rotation of the sensor around its normal axis. Due to the poor parallelism of the floodlight, the results were often inconclusive. It is thus recommended to obtain access to a better high-intensity light source and choose a set-up which offers decent parallel lighting. Mobility is advantageous since sun sensors are typically distributed on all faces of satellites. It is often easier to move the light source than to rotate the satellite in front of the light source.

The initial sun vector results are shown in Figure 10. Unfortunately, Figure 10 a-c) shows that SUS mounted on opposing sides provide conflicting sun vector components S_X , S_Y , S_Z . The agreement of sun sensor pairs among them-

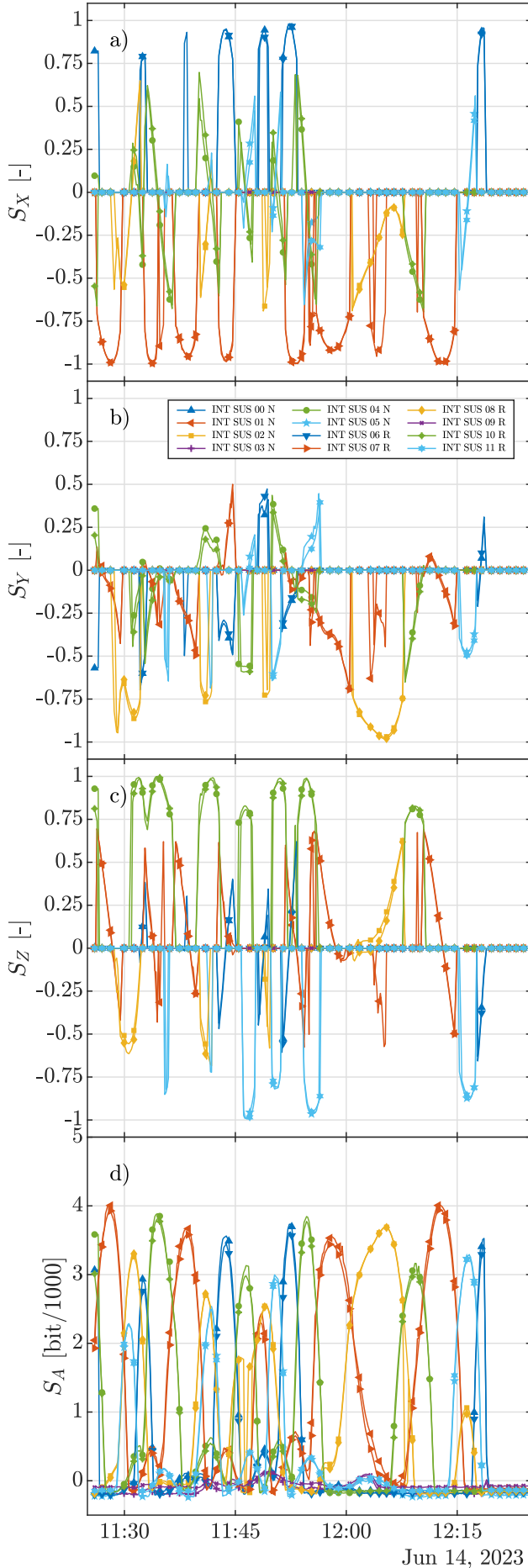


Fig 10. Sun vector measurements during one sun phase.

selves is generally fine. The mistake was identified as an inadequate albedo suppression, stemming from a misjudgement of the Earth's albedo strength. As depicted in Figure 10 d, the sum of the current of the four photo diode segments for valid sun vectors are up to 4000 bit. The signals induced by the Earth's albedo are much smaller in the range 0 to 750 bit, or about 20% of the maximum signal strength. The initial implementation was adapted to discarding sun vectors with less than 70% of the sun vector with the highest measurement intensity if multiple sun vector results are available at once. This reliably discards the sun vectors generated by the Earth's albedo while keeping possibly redundant sun vector measurements with smaller intensity valid, thus, leading to a successful determination of the sun vector.

6.4. GNSS Receivers

EIVE uses two cold redundant Hyperion HT-GNSS200 receivers connected to an active AP.35A antenna each. While the size and power consumption of the GNSS receivers are suitable for the EIVE ACS system, the connection between the receivers and the antenna was troublesome. The GNSS receiver and antenna both have u.fl connectors which are mechanically fragile and only have 30 mating cycles. In order to avoid a bound harness situation, a screw-lock coaxial connector was mounted in-between the receivers and antennas. Here, the small diameter of the u.fl cables made it hard to achieve a proper connection to the adapter connectors. It is therefore recommended to take the connector and the coax cable type into consideration when selecting a GNSS device.

Testing the GNSS receivers has been a challenge as well. The first approach included routing GNSS signals from an antenna with low noise amplifier on the IRS's roof via a RF-to-glass fibre converter into the clean room. There, another RF-to-glass fibre converter, a GNSS amplifier and a passive GNSS antenna created an RF link that was directed at the satellite's GNSS antennas. This method needs no expensive equipment such as a GNSS simulator and allows the satellite to find the current time stamp. However, the GNSS solution is stationary and the signal strengths at the receiver antenna are unknown. To test the dynamic performance of the GNSS device, GNSS signals of a full orbit were recorded at the GNSS simulator of the Institute of Navigation of the University of Stuttgart. This recording was replayed to the satellite with a GNSS recorder and a passive GNSS antenna. It was noticed that the received signals were badly conditioned which is why a 48 dB GNSS amplifier was inserted into the chain to compensate for approximately 33 dB loss of the radio link between the recorder's passive antenna and the satellite's antenna. A drawback of both methods is, that it is hard to stimulate the GNSS receivers at the correct signal strength due to the uncertainties of the additional radio link. Nevertheless, having access to a GNSS simulator/recorder is of advantage for ground testing since at least the principal functionality of the GNSS receivers at high velocities and altitudes can be confirmed. Time line plots of all gathered GNSS from the first 1.5 months after the launch are presented in Figure 11. A valid solution was only computed for a fraction of the mission time by using some GNSS satellites (◆) while Figure 11 c depicts that the receiver usually recognised at least 15 to 20 GNSS satellites (◀). The GNSS time can be provided even when no 3D fix is calculated. However, Figure 11 a displays intervals with a significant time offset of multiple days while no 3D fixes could have been computed (◆).

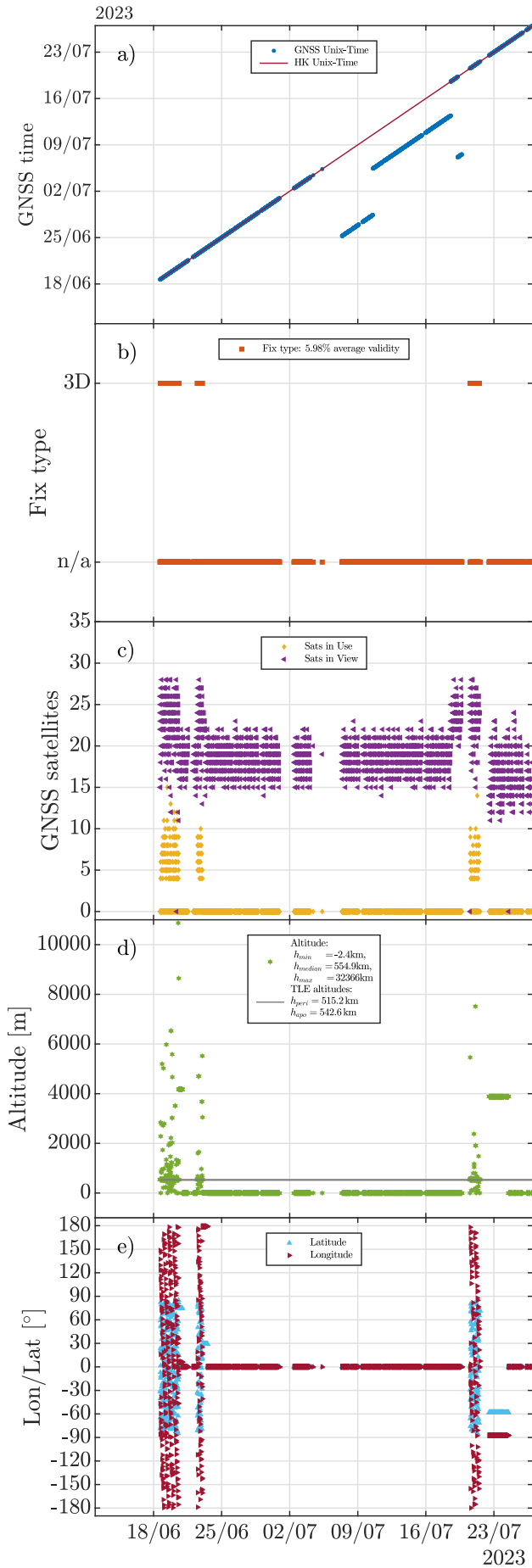


Fig 11. GNSS data points for the first month in orbit.

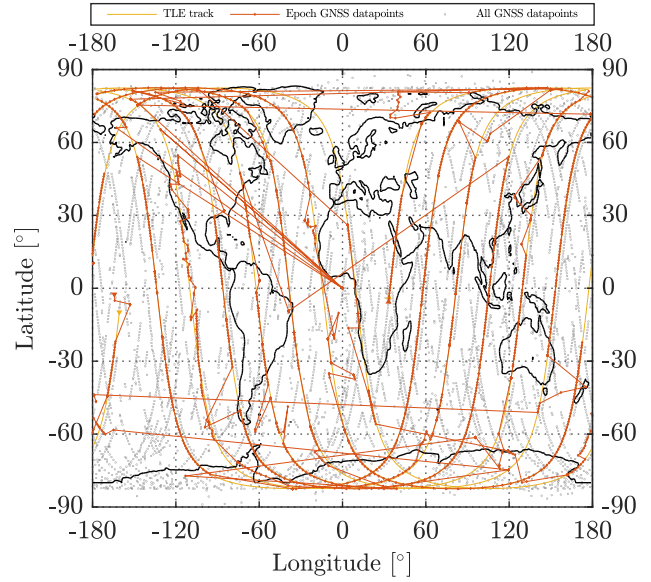


Fig 12. GNSS data points and TLE track for the epoch 2023-06-19 00:00:00–11:05:00 and all GNSS data points until 2023-07-27.

The altitude plot Figure 11 d shows that the altitude stays close to the expected flight altitude during periods with a valid fix. However, there are episodes where the altitude is calculated as sub-zero or just below a geostationary orbit. The calculated latitudes and longitudes show that there are gaps in-between valid data points. This can be clearly seen in Figure 12 where the longitudes and latitudes are plotted on a map. All data points gathered within first 1.5 months (●) highlight the aforementioned gaps. As an example, data points of six orbits in the epoch starting on the 19th of June (—♦—) are superimposed on the ground track as calculated from this epoch's Two-Line Element (TLE) (—). While most of the data points in this interval follow the TLE track, there are frequent outliers of single or multiple data points visible in addition to aforementioned gaps. Another interesting finding is that there are almost no data points over eastern Europe which might be caused by interference from ground based GNSS jamming equipment. While the cause of the poor behaviour of the GNSS system remains unclear, low signal-to-noise ratio of the GNSS are the prime suspect. This is supported by the fact that the signal-to-noise ratio of the satellites in view as taken from the receivers skyview data set is mostly around 0 dB. Equipment malfunction or signal deterioration due to the satellite's rotational rate are possible. The Beesat-9's operations team, which operate the same HT-GNSS200 receiver, reported a solid performance of the device in personal correspondences.

6.5. Star Trackers

The EIVE CubeSat uses an ArcSec Sagitta 1.0 star tracker. It is integrated tilted by 49° relative to the +X/Y-plane of the satellite in order to avoid sun blinding during the pointing modes. A baffle was added to avoid reflections into the star tracker's optics as shown in Figure 13. Unfortunately, ground testing the star tracker proved fruitless. During procurement, the option to secure the lens system with the focus set to infinity by applying epoxy glue was chosen. This made it impossible to test it in the star tracker test bed in the clean room of the IRS. Tests under the open sky were only pursued until the functionality had been proven in order

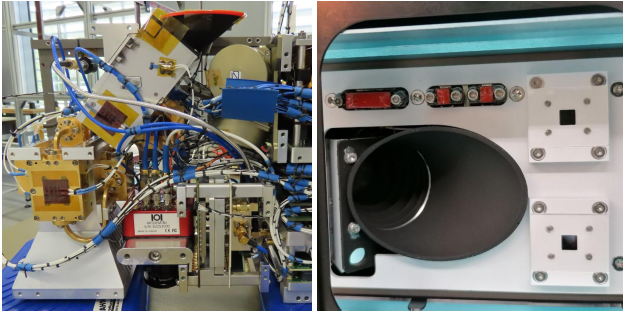


Fig 13. Star tracker with baffle and cap during integration, star tracker baffle, sun sensors and umbilical in flight configuration.

to not expose the flight hardware to uncontrollable climatic conditions longer than necessary. It is thus recommended to choose the manufacturing options with the star tracker tests set-up in mind. Another lesson learned was to have access to engineering models of components even when the flight model is already in orbit.

First flight results indicated regular fixes during the eclipse phase and some fixes if the satellite is in the sun. The fix quality is expected to improve once the parameters for the blinding suppression are fine-tuned. While this might be a minor inconvenience, in-orbit optimisation must not be underestimated as a risk factor to both the equipment as well as the time schedule.

7. THERMAL CONTROL SYSTEM

The temperature control system of EIVE is designed to be cold-biased [1]. This means, that the satellite is meant to be in a thermal equilibrium at the lower operational range of the components leaving enough head space for temperature increases if additional consumers such as the payloads are turned on. Heaters can increase the temperature of critical components when they drop below their operational limit. EIVE's thermal control system consists of 16 PT-1000 Resistance Temperature Devices (RTDs), temperature readings from many chips and components, six foil heaters and the internal battery heaters. The RTDs are distributed throughout the entire satellite at points of interest to the system. These devices were chosen due to the small size and high accuracy. Each RTD is read out by a MAX31865 chip with an SPI interface as described in Section 2.2.4. When working with this chip, it was noticed that after indeterministic periods of time, the individual chips stopped providing reasonable values, which could be temporarily fixed by a OBSW reboot. While the cause of this malfunction remains unknown, a better workaround turned out to be resending the MAX31865 set-up register before each readout command and not just during the initial configuration of the system. This simple trick comes at the minute expense of sending some additional bytes, however, can fix broken configurations on the chips. Thus, it is highly recommended to the embedded programmer to send the configuration registers of simple devices in frequent intervals.

The RTD chips are mounted on a dedicated PCB together with 10 SQJ431EP Metal-Oxide-Semiconductor Field-Effect Transistor (MOSFET) switches connected to seven 24 ■ 20 mmx20 mm and one 19.2 ■ 50 mmx50 mm Kapton foil heaters as well as the two HDRM resistor circuits. Since heaters can have a significant impact on the power budget, there should be at least two instances necessary to activate them to prevent discharging the battery by mistake. In the

case of EIVE, the heaters are connected to an 8 V power line that can be switched via the PCDU. In case the battery voltage decreases below the critical threshold voltage of 13.4 V, the PCDU goes into a power saving mode turning all heaters off. In addition to the PCDU command, the OBSW needs to toggle GPIO pins that control the heater's MOSFET switch.

The heaters inside Gomspace's BPX battery are controlled via a dedicated microcontroller attached to the battery cells. The heater settings are stored in the batteries configuration memory and are set to "off" by default whereas in flight configuration they should be on "auto" to enable the automatic heating of the battery unit once the temperature of the battery cells dropped below 0 °C. By mistake, the battery heater settings were on "off" thus disabling the automated battery heating in orbit. This mistake happened since the engineering and flight model were swapped during the integration in order to fly the fresher battery. While the heater settings were set correctly on the original flight model tested in the thermal vacuum chamber [1], the setting of the configuration was forgotten on the engineering battery model.

The functionality to set the battery heater settings via telecommand was not part of the OBSW at launch and was urgently included in the first software update. In the meantime heaters in the proximity were used to heat the battery indirectly which was quite inefficient. A good design decision was to wrap the batteries in single layer isolation foil. Once the battery heaters were active, this helped to reduce the radiative losses of the battery pack.

Figure 14 shows the battery temperature and battery currents for a single orbit. The battery heaters typically activate ten to twenty times per orbit with a majority of the activations during the eclipse, however, some occur slightly delayed in an early stage of the sun phase for upto 30 s each. This amounts to 5–10% of orbital time for the chosen heater threshold temperatures of 1 °C at turn-on and 5 °C at turn-off.

Moreover, small bugs in the temperature controller such as a swapped heater mapping led to the activation of wrong heaters. Further, heaters turned on, but sometimes failed to turn off although the linked temperature sensor reached the required temperature threshold. These bugs led to the operation team needing to turn heaters on and off manually via telecommands from ground. Leaving on heaters for an

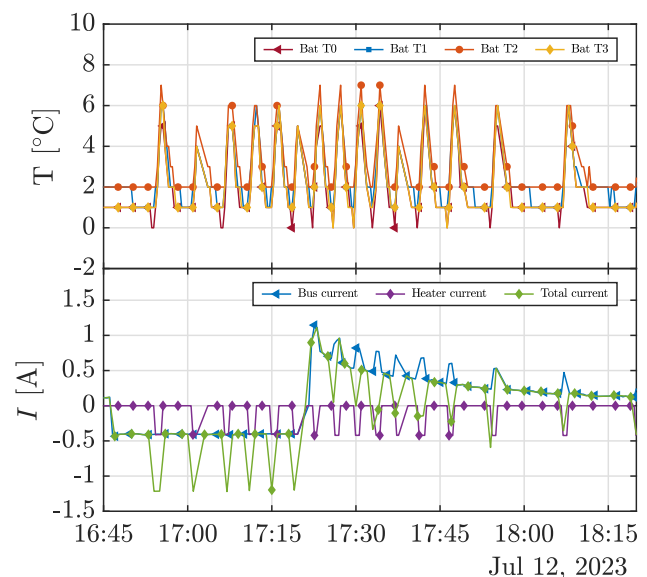


Fig 14. Battery temperature and currents over a single orbit.

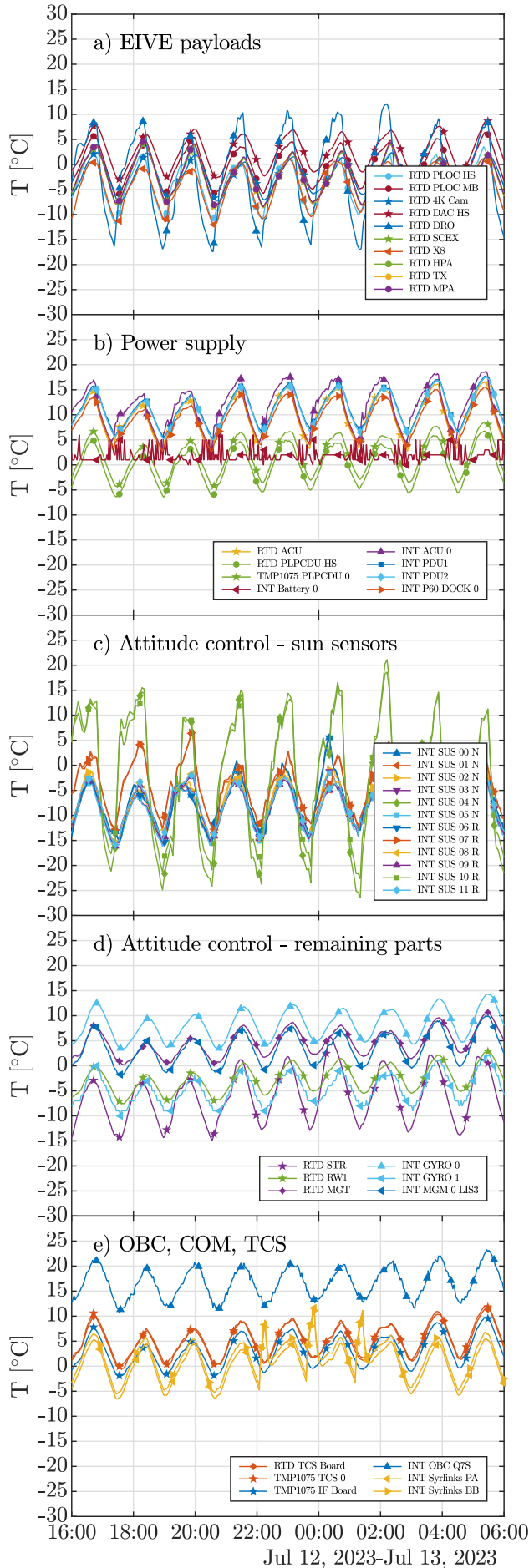


Fig 15. Temperature fluctuations over nine orbits.

extended period of time also can be dangerous since a negative power budget or an overheating of components could occur. Due to the limited verification and satellite access time, the OBSW thermal controller was only written after the final thermal vacuum tests of the EIVE CubeSat. With the satellite being at room temperature in the IRS's clean room, these bugs were never identified. A recommendation therefore is to conduct the environmental tests with an implemented and active TCS controller so that these bugs are found and fixed while the satellite is still accessible.

Figure 15 shows logged temperature data from nine typical orbits. The mostly structurally well-attached payload components of EIVE were turned off during these passes and are located in places without constant heat dissipation. The payload's temperatures as shown in Figure 15 a thus fluctuate around -2°C with $\pm 12\text{K}$ amplitude. The PLOC, however, is located at the end of the PC104 stack. It has an aluminium cover which connects to critical chips with aluminium fingers and thermal gap fillings (gappad). Since the range of the two RTD sensors on the heat spreader ($\bullet$) and the mission board ($\bullet$) is comparable to the sensors on components outside of the PCB stack, it can be concluded that the thermal connection of the PLOC to the structure is as good as planned. The power supply components as depicted in Figure 15 b of the PCDU are always active and stay within a comfortable range, while the deactivated PLPCDU ($\bullet$, $\star$) fluctuates at a lower temperature. The sun sensors have the highest temperature fluctuations of up to $\pm 50\text{K}$ as displayed in Figure 15 c. The sun sensors 4 & 10 are mounted on the back of EIVE's solar cell experiment seeing the steepest gradients. Interestingly, the sun sensors on the tuna can have the smallest fluctuation which is likely due to the thermal capacity of the tuna cans itself. The hottest internal component in the satellite's safe mode, as visible in Figure 15 e, is the OBC ($\bullet$).

The thermal simulation of EIVE showed, that thermal paints can reduce the temperature of the overall satellite into the desired cold-biased state. As a result, the outer panels, tuna cans and sun sensors were coated with white Magsil SG121 coating. To reduce reflections, the star tracker baffle as well as the inside of the sun sensor cases are coated with black Aeroglaze Z307 coating. An advantage of using those paints is that the surface's α/ϵ properties are well-defined unlike with bare metal surfaces. If no in-house experience with thermal coating exists, it is recommended to let experienced manufacturers do the paint job as the equipment and environmental conditions can be a challenge.

Overall the equilibrium temperatures of the satellite bus are within $< \pm 5\text{K}$ of the latest thermal simulation with the corrected orbit parameters. This excellent result could have only been achieved with the good correlation of thermal vacuum test results from the structural-thermal model and the flight model to the simulation [1, 2]. It must be noted, however, that the testing, post-processing and modelling proved to be rather time consuming. To sum up, the value of such an approach cannot be underestimated if the temperature simulations are meant to correlate closely with the behaviour in orbit.

At the present time, the satellite's safe mode is thermally stable. First activations of the E-band payload showed moderate temperature rises of the critical components thus validating the thermal design of the satellite.

8. STRUCTURE, HARNESS AND MECHANISMS

8.1. Structure

The structure of the EIVE CubeSat consists of two aluminium walls with cut-outs that are connected with struts and through components such as the PC104 stack or the horn antenna. The rails are integrated into the outer walls which are hard anodized to reach the required surface smoothness necessary for the glide rails. Aluminium cover plates of 3 mm thickness seal-off interior components from the space environment. The support structure of the solar panels is made of carbon fibre composite.

In preparation for the construction of the flight model, a structural-thermal model with similar structural and thermal properties was built [1]. This proved to be useful to verify the structural integrity by vibration testing. Although EIVE's Computer Aided Design (CAD) model was accurate down to the electrical component level, minor mistakes such as misaligned screw holes, tolerance issues and assembling issues were identified at this stage and prevented in the flight model. It can thus be recommended not to skip the development of a proper structural-thermal qualification model. Moreover, CubeSats may have delicate features at the outer surfaces so that it might be hard to handle the satellite safely. It is thus recommended to build an integration and handling cage such as depicted in Figure 16a, that can be used during transport and testing. The EIVE CubeSat has a two unit PC104 stack that bundles most custom and COTS PCBs with this connector standard as shown in Figure 16b. The connectors on a PC104 stack often have different heights and pin lengths and cannot be completely inserted into each other. Moreover, tolerance deviations add up which can cause a two unit stack to have an overall height tolerance of up multiple mm per height unit. To avoid issues with the attachment of the stack to the bus, it is recommended to be generous in the height of the spacers that set the PCB distances so that tolerance problems can be addressed by slightly decreasing the height of individual

spacers. Another advice is to use countersunk screws only in locations with well-defined tolerances to avoid warping panels or components. The secondary retention method for the screws on EIVE is Scotchweld DP2216 epoxy. While glue dots on screw heads may look improvised, this method has the major advantage over thread-locking fluid by being easily removable. It should always be assumed that each connection needs to be disassembled multiple times during the satellite's integration. While it never was a problem for project EIVE, the tolerances of the glide rails of CubeSats are tight and often underestimated. Therefore, the tolerances should be measured as early as possible to avoid surprises at later stages in the development. Also, the surface build-up of hard-anodising processes must be accounted for since this will increase the thickness of the rails depending on the process by up to 100 μm .

In conclusion, the structural concept of the EIVE CubeSat proved adequate to fulfil all structural requirements and to cope with the challenges that arose during the development and integration.

8.2. Harness

The harness of the EIVE CubeSat was completely routed in the CAD model. While this is a time-consuming job, it facilitated the cable length estimation, routing and building of the overall flight harness.

Also, it led to a quite precise estimation of the final mass of the satellite. The CAD model estimated 8.4 kg without harness and 0.3 kg of harness mass which sums up to a total of 8.7 kg. The final measured mass was 8.756 kg, just 56 g short of the prediction. The initially assumed harness margin in the mass budget of 10% finally turned out to be 4.2%. The harness CAD also helped to prevent bound harness which is a major complication during the satellite's integration.

An important decision for the CubeSat harness is the choice of the connectors. Most stackable PCBs on EIVE use PC104 through-hole connectors. These four row connectors can be hard to solder in a reproducible fashion. The most effective way for project EIVE was to manually solder the pins using self-made Sn60Pb36 solder doughnut. Many CubeSat COTS products use Harwin M80 or Molex picoblade connectors. EIVE's star tracker uses an Omnetics nano-D connector. While nano-D connectors are excellent due to their small size and the screw locking feature, they are also quite expensive. M80 connectors are sturdy and can be secured with screws, however, they are quite bulky and take a lot of PCB floor space. Picoblade connectors are slim, as depicted in Figure 16c, and have a decent power rating for the size. However, the allowed mating cycles are limited and it is difficult to secure the clip-lock with epoxy glue as a secondary retention in a reversible fashion. Unlike bigger satellites, connectors used in CubeSats rarely are made out of metal and thus usually do not offer a way to attach a cable shield. This fact and space limitations yielded the decision to use loose cable bundles instead of multi-conductor cables with shields. The harness was bundled with Relek 20DR lacing tape using the knots detailed in NASA-STD-8739.4 [27]. A major lesson-learned is to be cautious at locations where harness is moved by the attachment or integration of other parts. This can lead to pressure being applied to PCB connectors. During project EIVE, the attachment of the tuna can panel led to the solar panel harness pushing on the ACU connector which sheared off leading to a half year delay until a replacement part was procured. To prevent the harness

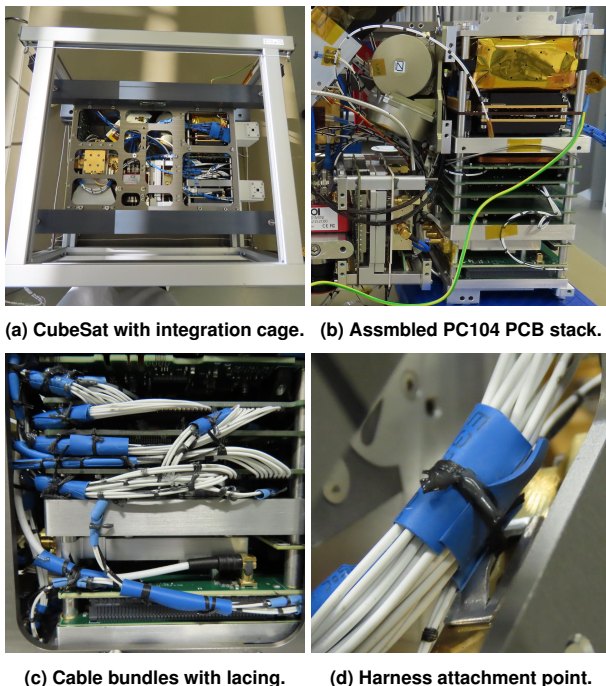


Fig 16. Features of the EIVE structure and harness.

from wobbling during vibrations, it was secured with clove hitches to structural features or to small aluminium tie bases glued to the structure with epoxy as depicted in Figure 16d. Cables can ream at the edges of components which is why the AWG22–28 Teflon cables were wrapped at critical points with polyolefin shrink sleeves. Again, DP2216 epoxy was used to secure shrink tubes, lacing tape knots and the tie bases to the chassis of the satellite. This approach has worked well and can be recommended. The manufacturing and integration of the harness took up half of the time spent during the integration of the satellite. Tying knots in spaces with difficult access requires a lot of time and practice. Nevertheless, the harness was built, integrated and tested successfully.

8.3. Mechanisms

The EIVE CubeSat has two mechanisms with movable parts. The first one are the four ITW-18-40419 deployment switches that are compressed by indexing plungers integrated into the rails. This solution, presented in Figure 17a, turned out to be mechanically and electrically solid and can be recommended.

As mentioned before, the deployment of the solar panels was imperative to ensure a positive power budget. The self-developed HDRM mechanism of EIVE is based on a thermal knife concept where two parallel 10■ high-power resistors are pushed onto a Dyneema wire loop by flat springs. When power from a 8 V line is supplied, the resistors heat the wire above its 130–140 °C melting point to cut it. The body mounted and deployable solar panels are then pushed apart by two torsion springs at the hinges and a compression spring at the HDRM itself. The HDRM mechanism is located at the center of the panels with V-shaped cups/cones of dissimilar materials defining the distance at the panel's edges. While the mechanism ultimately worked, many challenges needed to be solved during the implementation. Firstly, it must be ensured that

the thermal contact of the thermal knife resistors to the structure is as low as possible since thermal leakage paths will draw heat away from the melting wire. The Dyneema wire loops run through set screws that can be tensioned with nuts attached to the deployable panels as shown in Figure 17c and Figure 17d. The wire can be damaged at sharp edges of the drill hole in the set screw. Wrapping the affected wire section in Kapton tape prevented this issue. Moreover, the epoxy that secured the knot in the wire needs to be viscous enough as not to infiltrate the wire and increase the melting point of the latter. These issues show that it is imperative to apply good and reproducible workmanship principles. It is recommended to test thermal knife mechanisms often and under various conditions, such as vacuum and different temperatures, in order to identify critical influences on the functionality.

Lessons were also learned during the manufacturing of the solar panels itself. The hinges are made of aluminium with steel torsion springs connecting the flanges. During vibration tests, the aluminium surfaces produced abrasion dust that prevented a full deployment. Moreover, the manufacturing tolerances result in varying deployment angles. Both issues could be solved by adding steel shims into the hinges as dissimilar materials.

The solar panels were deployed automatically two minutes after the first OBC boot in orbit. Deployment could be confirmed when the satellite was in its first sun phase by measuring voltages on both deployable panels simultaneously over the O'Higgins ground station.

9. CONCLUSION AND OUTLOOK

A specialised satellite for a novel E-band transmission payload was developed [1, 7, 20], tested [2, 8] and launched into a 530 km sun synchronous orbit.

The EIVE CubeSat employs a centralised system architecture with a SoC OBC at its heart. Using communication buses such as SPI and I2C for inter-system communication in CubeSat should be avoided as far as possible and replaced by buses with a higher reliability such as CAN or RS422/RS485 to prevent low level issues and temporarily stuck buses.

Using a SoC led to a potent system due to the ability to include the CCSDS FPGA cores directly in the OBC and have an overall platform with a high flexibility in terms of additional buses and reconfigurability. The use of HPCs to recover the satellite's OBSW is explicitly encouraged. With the overall positive experience of using an S-band transceiver with the CCSDS protocol, the additional effort to implement this standard and to conduct the end-to-end communication tests with all relevant equipment and ground stations must not be underestimated.

The design of the power system was based on a conservative estimation of the power budget which ultimately led to a stable system in the safe mode even with a slowly tumbling satellite. The non-standard voltages and high currents of the E-band RF components led to the development of a custom power supply unit with the focus on thermal dissipation methods and overall system security.

The ACS MEMS gyroscopes and magnetometers yielded mixed results. Good magnetic calibration on ground led to reliable magnetic field measurements. The bias calibration of the gyroscopes did not prove reliable and needed to be repeated in space. The self-developed sun sensors initially had issues since the Earth's albedo noise signals reached approximately 20% of the sun signals and were improperly

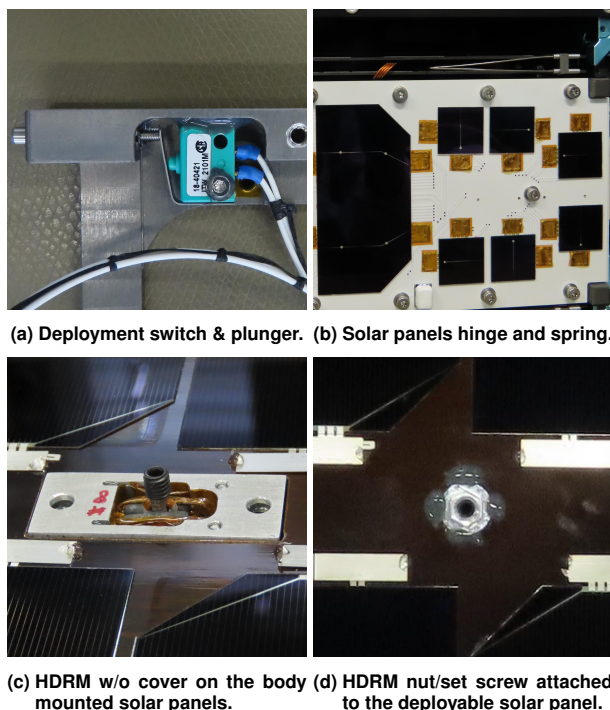


Fig 17. Features of the EIVE deployment switch and the solar panel deployment mechanism.

filtered that ultimately could be fixed in orbit. The GNSS receivers on EIVE were difficult to test on ground due to the need to establish a GNSS RF link with unknown signal strength in the clean room. The overall performance of the GNSS receivers in orbit is poor with few valid solutions being computed and sometimes days long intervals between them. While the reason for the malfunction remains unclear, signal strength issues, the rotation of the satellite or equipment malfunction are suspected. The star tracker currently performs well in the eclipse phase but has some issues with blinding during the sun phase. While this might improve once the attitude is stabilised and the star tracker cannot be blinded by the sun, future iteration of the configuration might be necessary. In order to have a well implemented attitude controller at launch, it is recommended to invest more time and use a better suited simulation environment or in the best case a simulator which includes the ACS OBSW code. The cold-bias design of the TCS system is monitored by a multitude of thermal sensors and regulated if needed by resistive heaters. The effort put into the verification of the TCS with a structure-thermal model of the satellite [1] and two major thermal vacuum test campaigns [2] led to a thermal model that is in excellent agreement with the temperature results measured in orbit. Bugs, that led to the battery heaters not turning on or other heaters being stuck on, underline the importance of thermal tests with active heater control.

The structural design of the EIVE CubeSat was verified by vibration tests and with some minor lessons learned along the way. The harness was fully routed in the CAD model and could thus be accurately predicted to 4.2% of the total mass of 8.8 kg. The manufacturing, integration and securing of the harness consumed approximately half of the total integration time. The thermal-knife solar panel deployment mechanism worked flawlessly in orbit, however, a lot of effort was spent during development and testing.

At the time of writing, the EIVE CubeSat was still in the commissioning phase with all bus components being fully operational. Once the difficulties with the ACS system are solved and all payloads are checked out successfully, the nominal operation phase can commence.

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DECLARATIONS

Conflict of interest

The authors declare that they have no conflict of interest.

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