16th International Conference on Space Operations, Cape Town, South Africa – 3 - 5 May 2021.

SpaceOps-2021,3,x1308

Launch and Early Operations of Eu:CROPIS

Miguel Lino[a]*, Daniel Schulze[b], Gary Morfill[c], Claudia Philpot [d], Olaf Eßmann [e]

[a] Flight Director, *German Space Operations Center, Münchener Str. 20, 82234 Weßling* / miguel.lino@dlr.de
[b] Flight Director, *German Space Operations Center, Münchener Str. 20, 82234 Weßling* / d.schulze@dlr.de
[c] Operations Manager, *German Space Operations Center, Münchener Str. 20, 82234 Weßling* / gary.morfill@dlr.de
[d] Systems Engineer, *Institute of Space Systems, Robert-Hooke-Straße 7, 28359 Bremen* / claudia.philpot@dlr.de
[e] Mission Manager, *Institute of Space Systems, Robert-Hooke-Straße 7, 28359 Bremen* / olaf.essmann@dlr.de

* Corresponding Author

Keywords: LEOP, LEO, contingency, panel deployment, tomatoes, biology

Abstract

Eu:CROPIS (Euglena and Combined Regenerative Organic-Food Production in Space) is a German Aerospace Center (DLR) small satellite launched on December 3rd 2018 to a low Earth orbit in a ride-share launch on a Falcon 9 rocket. The primary payload developed by DLR-ME (Aerospace Medicine) and the University of Erlangen comprises a pair of greenhouses in which tomatoes were to grow under simulated Mars and Moon gravity. Being a spin-stabilized spacecraft, the most distinctive feature of the satellite is that its spin rate is set so that the simulated gravity level on the biological payloads can correspond to Moon or Mars gravity, as well as that of orbital space stations in near zero-g.

The sequence of events planned for the three days of the Launch and Early Operations Phase (LEOP) is explained. The main driver of the planning was the solar panel deployment. The release of the four panels is done in a non-explosive manner via shape memory alloy actuators (SMA), which are connectors that are fractured if heated to a sufficient extent. The deployment of the panels can only be verified indirectly with this system and a risk of having a partial deployment of panels had to be considered. In addition, the attitude control of the spacecraft is solely done by magnetic torquers. Therefore, the actions after the initial satellite health check at the first acquisition of signal in space targeted a verification of all the critical capabilities and system redundancies on-board required to perform the safe release of the solar panels.

Secondly, this paper reports the highlights of the LEOP. The first planned contact with Eu:CROPIS failed mainly as a result of a misconfiguration on the ground systems. Furthermore, later in the LEOP after the checkout of the platform, the panel deployment was initially unsuccessful as only two of the four panels deployed. Finally, towards the end of the planned LEOP, a severe anomaly on the flight software prevented transmission of telemetry and reception of telecommands by the spacecraft for three days, after which an automated failure detection and isolation reaction recovered the system.

Finally, the anomaly analysis performed and decisions taken by the operations team are critically assessed in retrospective. A handful of lessons learned from the experience with Eu:CROPIS are collected for future missions and LEOPs.

1. Introduction

1.1 DLR Compact Satellite

Eu:CROPIS is the first flying satellite of the DLR Compact Satellite series developed by the Institute of Space Systems DLR-RY located in Bremen. The strategic goal is for DLR to consolidate experience in the end-to-end development of satellites of approximately 1 cubic meter and a mass of approximately 200 kg [1]. Each satellite of the series is to support scientific payloads and is to be shaped on a case-to-case basis depending on the payload requirements. Additionally, the core avionic of the bus is a crucial part of DLR's research program. Candidate missions to succeed Eu:CROPIS are to be selected.

1.2 Launch and orbit

Eu:CROPIS was launched into orbit on the 3rd of December of 2018 at 18:34:05 UTC by the Falcon 9 rocket of SpaceX from the Vandenberg Air Force Base in California. The mission SSO-A SmallSat Express from Spaceflight was a rideshare launch with 64 satellites [2]. Eu:CROPIS was injected in a sun-synchronous orbit of 574km altitude, 98° inclination and approximately 10:30 local time of the ascending node. The satellite experiences eclipses with seasonal variation between 33.8 and 34.8 minutes. Not having means of propulsion for de-orbiting, the satellite's estimated lifetime in orbit is of 9.8 years.

1.3 Main payload

The main payload named Eu:CROPIS, same as the mission, was a biological experiment intended to test a combined biological system as a technology demonstrator to be used in missions or settlements in Moon and Mars. The focus of the system tries to prove the feasibility of a biological system whose main goal is to not only obtain water from urine, but also relevant substances such as Nitrogen and Oxygen to supply a biomass, chosen to be tomatoes. The filtering system consists of Algae (Euglena) and of a trickle filter (lava rock) [1]. Taking advantage of the self-spin motion used for the stabilization of the spacecraft's attitude, the satellite can be commanded to spin to simulate gravity in the Moon and in Mars. Each compartment was to run in separate periods of 6 months [3].

1.4 Secondary payloads

The secondary payload PowerCell (NASA Ames) is also a biological experiment benefitting from the artificial gravity provided by the platform. PowerCell hosts cyanobacteria capable of photosynthesis, which are to provide Carbohydrates to Bacteria (Bacillus Subtilis) capable of producing proteins [4].

The scientific payload RAMIS (RAdiation Measurement In Space) is a radiometry system of DLR that consists of two detectors. One silicon detector is mounted on the surface of the spacecraft under low shielding while the other is placed inside the main payload. The system allows to determine protons and electron fluxes in space as well as to measure cosmic ray contributions to the radiation field as function of the orbit [1].

The payload SCORE (SCalable On-boaRd Computer) is a mini computer developed by DLR-RY to be tested operationally in the Eu:CROPIS platform and to be used in future missions. SCORE operates three cameras of which one is pointing to nadir direction whereas the other two are pointing to different areas of the satellite's body [5]. The latter are an auxiliary means to confirm the solar panels' deployment.

1.5 Platform

Eu:CROPIS is spin-stabilized spacecraft that has a cylindrical shape with 4 deployable solar panels. The wet mass is of 234kg. The dimensions are of 1.1x1.1x1.1 m^3 while the panels are stowed and of 2.9x2.9x1.1 m^3 after the deployment of the solar panels [5].

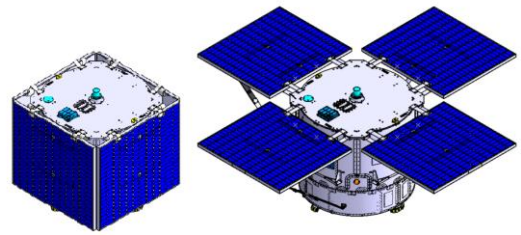

Figure 1: Depiction of Eu:CROPIS in stowed and deployed configurations.

The satellite was designed to fulfill the main requirements necessary to the success of the payload operations, while implementing a minimum cost design that also provides a maximum reliability. The biological nature of the payload was the main driver of the satellite's design, since several components require pressure and temperature stability. The satellite must ensure the possibility of commanding different attitude rates depending on the stage of the biological experiments. A high level of reliability is necessary given the fragile nature of the payloads, which is expressed in the narrow temperature ranges and required response times [6].

The satellite's default nominal attitude requires pointing the deployed panels perpendicularly to the sun vector. The induced self-spin is around its **z** body axis, which coincides approximately with the axis of the body cylinder. The **x** and **y** axes are perpendicular and form a right-handed orientation in such a way that a counter clockwise self-spin as seen from the top is a positive angular rate ωz. The sun-angle is defined as the angle between the **z** body axis and the sun vector.

The attitude control is realized with three magnetic torquers of 30 Am^2 with redundancy in its electronics while the attitude determination relies on measurements from 2 magnetometers and a set of 10 sun sensors. Four fiber optic gyroscopes measure the angular rates. The attitude determination is based on an Unscented Kalman Filter [7]. On-board navigation is provided by data from 2 single-frequency Phoenix GPS receivers with heritage from the PRISMA mission [8].

The ACS software has been adapted from that of the TET-1 mission [9]. Several attitude modes were implemented. At the nominal operation Spin and Point Mode (SPPT), the target angular rate ωz and the sun-angle can be commanded from ground. The system autonomously seeks to eliminate the angular rates in the other directions (ωx, ωy). Precession control is done following Earth's translation around the Sun during the solar year. A dedicated concept for failure detection, isolation and recovery mechanisms (FDIR) was implemented [10]. The simulation of the gravity levels for the biological

payloads is done via the dedicated placement of the respective pressurized compartments in the satellite's body.

Power is provided by four composite sandwich solar array panels in the deployed configuration [11]. The regulation is done by two Array Power Regulators (APR), each connected to two opposite panels [6]. Each panel can provide 189W to 304W, whereas each APR can provide up to 292W. The panels are mounted in alignment with the **x** and **y** body fixed axes and can be identified accordingly: +**x**,-**x**,+**y**,-**y**. The panel deployment is done via a non-explosive manner via SMA actuators, which are connectors that are fractured if heated to a sufficient extent [13]. Li-Ion batteries have been installed to cope not only with the eclipses of 33.8-34.8 minutes duration but with worst case conditions of injection in orbit as well. The power bus is unregulated, having a voltage of 22-34V. DC/DC converters were used for units with low input voltage [6].

The radio frequency equipment assembly (RFEA) comprises two cold redundant S-band transmitters, with configurable net data rate of 400kbps, 800kbps or 1600 kbps, and two hot redundant receivers. Two circularly polarized hemispherical antennae are placed on the top and bottom plates of the structure. An analysis on the influence of the satellite's spin rates on the RF link quality measured in orbit has been done [14].

Two on-board computers (OBC), cold redundant but with sub-units operating in parallel, uses the RTEMs operating system, which runs a command and data handling (CDH) system developed by DLR-RY [15]. A simulation environment with the purpose of testing and validating the on-board software using automatic procedures was set up [16].

1.6 Ground Segment

The ground operations of Eu:CROPIS are executed from the German Space Operations Center (GSOC) part of the DLR-RB institute at Oberpfaffenhofen near Munich. The main ground station providing two S-band antennae is Weilheim, located in Germany. The rest of the Ground Station Network for the LEOP comprised the stations of O'Higgins (Antarctica), Spitsbergen (in Norway from KSAT) and Saskatoon and Saint-Hubert in Canada (Canadian Space Agency). All the antennae of the network are capable of downlink and uplink to the satellite. While the extensive amount of stations allowed for a frequent monitoring of the satellite in the first two weeks in orbit, only the primary (Weilheim) WHM station is used for routine operations [3].

At the Launch and Early Operations Phase (LEOP), the operations were supported by the Eu:CROPIS team in the control room for 3 days. Each day had 2 shifts of approximately 10h duration, with 0.5h planned for the handover. Gaps between shifts had 3h-5h duration when no activities were planned and monitoring-only passages were scheduled. During routine operations, 24/7 support is ensured by GSOC, including *in situ* support in the control center and on-call support by a reduced amount of personnel.

The dissemination of housekeeping and scientific data is done at GSOC: the scientific data of Eu:CROPIS payload and RAMIS are routed via a SFTP transfer to the Microgravity User Support Center (MUSC) located in Cologne; the PowerCell data is distributed to NASA in similar fashion. The mission partners also receive auxiliary data such Two-Line Elements (TLEs) or housekeeping data related to the platform where necessary [3].

The tools used by the Operations team make maximum use of the multi-mission infrastructure built at GSOC over the last years [17]. The commanding system, GECCOS (GSOC Enhanced SCOS), benefits from the experience gained with other missions (Grace-FO, TerraSAR-X and TanDEM-X, BIROS) [18]. Nevertheless, minor specific tailoring is normally necessary to cope with requirements from different missions. The Procedure Tool Suite (ProToS) developed at GSOC [18] was used for the development of the Flight (FOP) and Ground Operation Procedures (GOP). Eu:CROPIS was the first mission to use the tool operationally. Currently, ProToS is the standard procedure tool at GSOC for any new mission.

2. Preparation

2.1 Feedback on the on-board software

The development of the satellite's features most closely related to operations were done in close cooperation between the Eu:CROPIS teams of DLR-RB and DLR-RY. The common affiliation of both institutes facilitated the communication between the ground and the space segment. Decisions could often be made not to benefit solely one of the segments but to benefit the overall the mission instead. Both parts were receptive to suggestions and to criticism.

Further, as GSOC holds a LEO (low Earth orbit) multi-mission control center where the synergies are sought as far as possible, similarities between missions, knowledge of operations personnel and tools are of importance. This approach makes it possible to reduce the financial costs of operations by making the control center more efficient.

Therefore, recommendations were made to make the operability of the satellite in terms of functionality and of naming as similar as possible to the ones already operated in GSOC. Settings related

to telecommands and telemetry packets, as well as on-board software configurations and CDH properties were modified as result of recommendations done by GSOC. One example was the implementation of so-called system event log as an extension of the Packet Utilization Standard (PUS) event service 5 [19] that constitute a non-volatile record of important events on-board that can be read out upon request. The on-board storage and retrieval PUS service 15 was also implemented [19].

A few features were implemented early on the CDH's development that turned out not to be used because there was no necessity or because they had a costly implementation on ground. An example of this is the feature of interlock in the PUS scheduling service 11 that allows for scheduled telecommands to be executed depending on the success status of other telecommands or of the generation of specified on-board events. Such feature was present in the PUS published by the European Cooperation for Space Standardization (ECSS) in 2003 [20] but no longer in the currently active release of 2016 [19].

FDIR-related features such as the ones in the scope of PUS service monitoring 12 and event-action 19 [19] could not be implemented in the on-board software as they would have required too large an effort. Instead, the FDIR implementation done on the on-board software was based on state machines, where each FDIR mechanism has a different map of states with specific conditions and timings for transitions to occur. The transition of satellite modes depending on the battery state of charge described on [6] is an example of such a case. Such an implementation limits the operators' possibilities for configuring the settings of each machine. Reconfigurations are the more necessary the more often unexpected situations on the spacecraft occur.

2.2 Validation of procedures

A Mission Operations requirement was to validate all FOPs and all GOPs. Since both the Engineering Model (EM) and the Flight Model (FM) were necessary for the assembly, integration and testing of the satellite in Bremen, their availability for validation of operations procedures and systems required coordination to schedule validation sessions. Also, OBSW updates done before the launch required revalidation of specific FOPs that had already been validated. All the GOPs were validated. The vast majority of the Flight Procedures including the payload software upload sequences were validated either using the EM or the FM. The FOPs that were not validated were either of concern to payloads only or not part of the sequence of events (SoE) of the LEOP.

2.3 Training for the LEOP

The simulations campaign was essential to the adequate training of the personnel and the respective familiarization with the operations environment. The proximity between the operations team from GSOC and the satellite development team from DLR-RY was strengthened during the simulation sessions. The trading of information and knowledge between partners was encouraged and resulted in a close cooperation on console between the team members of the ground and the space segment.

Despite the LEOP having duration of 3 days, i.e., 6 shifts, each training simulation consisted of 2 days simulating approximately one shift each. The most critical parts of the sequence of events were chosen for training, namely the first acquisition and the panel deployment. The remainder of the planned activities for the LEOP were executed in a single separate session where the contact times were not represented. This ensured the validation of the whole SoE. 8 simulations in total were done.

Apart from the personal training benefits, these sessions also served as debugging sessions, where certain aspects of the OBSW and of the spacecraft in general became evident to the team. Suggestions to improve the CDH from an operational point of view arose from the simulations. For instance, it was noticed that the request of all stored events had to be done with a specific telecommand per sub-system on-board, totalling 16 commands. As the activity was time-tagged for each passage, the corresponding commanding sequences were lengthy and thus requiring more commanding time and more slots in the mission timeline. In order to circumvent this, a few so-called meta telecommands were added to mission database, which can address all subsystems at once.

The systems used at the training were to represent the real systems as far as possible. The performance of all the mission operations tools used was validated. On the side of the space segment, significant effort was invested by the Eu:CROPIS team of DLR-RY to implement a reliable simulation setup. While this was mostly achieved, a few aspects of the training still did not reproduce the real behaviour of the systems or of the spacecraft in orbit. For example: some unit redundancies were not available on the EM; temperature telemetry or payload data was not produced. Nevertheless, the quality of the training was excellent and proved essential when anomalies had to be recovered during LEOP.

3. Planned Sequence of Events

The SoE is the plan of activities to be performed on the spacecraft during the LEOP. The goal of the LEOP was to deliver the spacecraft platform and the payloads in operating condition within 3 days, where 67 passages were planned. The multi-mission tool used by GSOC to produce the SoE is the SoE Editor [21].

The whole sequence was planned with a sensible level of activities in passages. While an approach where all activities are planned without spare time would be ideal to achieve the targets of the LEOP as soon as possible, any unexpected operational issues from the ground or space segments during the LEOP would make frequent reworks of the plan necessary. Also, it is generally recommendable that the personnel has time to get familiar with the behaviour of the spacecraft in flight as well as to cope with anomalies both in the ground systems as wells as in the spacecraft. Thus, a middle ground approach was done where nearly half of the scheduled contacts were idle.

3.1 Separation

The on-board computer is powered immediately after separation. Approximately one minute later, the OBC boot sequence is executed and the nominal transmitter sends regular housekeeping telemetry in S-band. The ACS system is also initialized automatically. The tip-off rates induced at separation from the launcher were designed to be at most 2°/s. The ACS control mode after separation is based on a B-dot controller whose target is zeroing the angular rates [7]. After reaching such conditions, the ACS switches mode to re-orient the spacecraft to achieve a 90° sun-angle, since the solar panels are perpendicular to the main body axis **z**.

In this so-called barbecue mode, a maximum of two panels can be illuminated. This stage of the mission, where a solar panel area available is reduced while having relatively strict temperature limits on the payloads due to the biology, was the main design driver for the solar panel area. A reasonable confidence level of power balance after the separation was estimated in simulations under different conditions [6]. The battery is fully charged before launch to provide power to the satellite during 5h worst-case scenario without any solar power available. The dimensioning of the solar panels in the stowed configuration resulted in that the system is significantly over-dimensioned for the deployed configuration. In nominal operating conditions of the whole spacecraft, the solar panels supply more power (~584W) than the satellite requires even with the payloads in the respective operative science modes (~165W). The first set of contacts with the spacecraft was scheduled roughly 41 minutes after separation.

3.2 Preparing the panel deployment

Assuming a nominal separation of the launcher with nominal power availability for the satellite as well as nominal attitude control to ensure it, there was no pressing deadline to perform the deployment.

Given the criticality of the solar panel deployment, which is detailed in sub-chapter 3.3., the approach chosen was to test all the fundamental units and functionalities of the platform beforehand. The main advantage is that the whole set of on-board resources necessary for a safe panel deployment or necessary to cope with anomalies is validated in flight in advance. It also gives sufficient time for the operations team to get familiar with the spacecraft. The tests comprised an active redundancy switching of critical units which otherwise would not be necessary: the redundant OBC, redundant transmitter and redundant sun-sensors were tested operationally in flight. The SMA actuator heaters were to be activated as a trial before the panel deployment too. Also, tests of the ACS were to be carried out: the sun-angle control was tested by commanding from the barbecue mode to a sun-angle of 80° and then back to 90°. The spin control was evaluated in a similar fashion by commanding from 1 rpm to 2 rpm and then back to 1 rpm. Furthermore, updates of mounting matrices for the gyroscopes and the magnetometers were planned, as well as updates of the ACS moments of inertia based on the calibrated data downlinked in flight.

The only platform checks done after the panel deployment, considered as having secondary priority, were those of the redundant command module of the power and control distribution unit and of the different telemetry transmission rates.

During the SoE, the payloads except SCORE are monitored but not commanded. The Eu:CROPIS main payload and PowerCell are powered after separation and remain in stand-by modes where no interaction from ground is needed in nominal conditions. The activation of RAMIS was planned for the 3rd LEOP day. On the other hand, SCORE is to be powered by ground command and tested in advance of the panel deployment, as pictures from its cameras are helpful to confirm the release of the panels.

3.3 Panel deployment

The solar panels are installed on the spacecraft's body by tape spring hinges, which can store elastic energy and unfold if unconstrained [12]. The release of the panels is done by breaking the respective connectors, called SMA actuators. This can be done only via telecommands that switch on resistive

heaters that cause the SMA actuators to warm until its breaking point. Each of the 4 panels has 2 SMA actuators, one placed close to the top plate and another near the bottom plate and can be identified as **-z** or **+z**. For instance, the top SMA actuator of the **+y** panel is named **+z+y**. The two SMA actuators of the panel need to break for a deployment. Each SMA actuator is supplied by two hot redundant thermistors and two cold redundant heaters. In total: 16 heaters and 16 thermistors, being a single-failure tolerant system. As there is no dedicated feedback from the chosen SMA actuators mechanism, the verification of the breaking of the SMA actuators and the subsequent release of the panels can only be done via indirect inferences from telemetry or from the pictures taken by the two selfie cameras of SCORE.

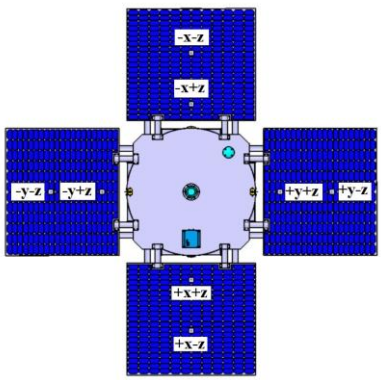

Figure 2: schematical illustration of the location of the SMA actuators.

The risk of having a temporary partial deployment of panels was considered. An imbalance of mass distribution of the satellite resulting from a mixed or partial deployment of panels would lead to deviation between the main body axis **z** and the principal axis of inertia. Should such a permanent incomplete solar panel deployment occur, a severe degradation of the mission would follow because the biological payloads could not run its experiments with the designed induced Moon and Mars gravity.

Furthermore, the ACS is only capable of limited responses to perturbations in the attitude, since the control is solely done by magnetic torquers. Inconsistent moment of inertia settings at the ACS controller with respect to the true inertia tensor of the spacecraft can further lead to a faulty attitude control. Such a scenario would not violate the power requirements: the power supplied by only two panels would have sufficed to supply the platform and the payloads in nominal mode operations.

The SMA actuator heaters and respective thermistors were to be tested in advance by activating the heaters in a limited manner. Dedicated test commands were created for this purpose, which turn the heaters for 1s only.

The breaking point of the SMA actuators is approximately at 80°C. Tests done on ground with the flight model resulted in estimation that the SMA actuators would break 30s-60s after activation of the corresponding heater. Assuming a simultaneous break of the 4 panels, the time difference between the full deployment of the first and the last panel would be 10s at most. The breaching of the SMA actuators can be confirmed indirectly by the temperature curve while heating it. As there is an expected drop in the temperature increase as the SMA actuator breaks, dedicated derived telemetry parameters for the first derivative of each of the 16 temperature parameters delivered by the 16 thermistors were inserted in the mission database. The approximation used considered the last 10 produced telemetry samples.

Apart from simple manual on/off telecommands for the resistive heaters, dedicated constraint-driven automatic telecommands were created to ensure safe operations. Using the latter, the heaters turn off automatically after 120s, if the temperature difference between two redundant thermistors is larger than 20°C or if both of the respective thermistors measure 150°C or more. In case of loss of commanding capability from the ground station during the deployment passage, this ensures that heaters are securely turned off. Also, if the temperature is not at least 5°C higher 5s later than at the moment the respective heater is switched on, the operation is interrupted. None of the aforementioned characteristics of the automatic telecommands for the SMA actuators were configurable.

All the SMA actuator-related actions were planned in the SoE with the robust automatic telecommands in mind. The manual on/off telecommands could be used as a backup solution when necessary.

Since each panel had two hinges to break, it was opted to break one of the SMA actuators of each panel before its planned deployment, avoiding that two heaters have to be on simultaneously for one panel.

A few options were considered regarding the attitude and angular rate at the deployment of the panels. It is in stowed configuration that the available solar panel area is smaller, varying between 1 and ~1.4 solar panel areas as the satellite rotates around its main body axis. Therefore, having as much power as possible before the deployment was privileged in comparison with the status after the deployment. The selected sun-angle was of 60°, which ensures a limited decrease of solar power (~14%) while in stowed configuration. Each panel that deploys reduces its effective area by approximately 42% but

by being permanently illuminated during the rotation, the averaged effective area over one rotation is nearly doubled.

The rotation rate for the panel deployment was chosen to be 1 rpm, providing sufficient robustness for the attitude disturbances expected at the panel deployment.

Two solar panel deployment sequences were considered: all-at-once deployment and a bi-phased deployment. The second option consisted in deploying at first two opposite facing panels and secondly the other two opposite facing panels. This still maintains the symmetry in the satellite's mass distribution. This method would have required an additional upload of intermediate moments of inertia. Also, the panels to deploy first would be the ones covering the sides where the PowerCell enclosures were placed (at +**y**,-**y** axes), since the enclosures are warmer than in design as the solar panels hinder their heat dissipation into space. One advantage of this approach would have been the smaller power consumption during the heating of the SMA actuators. By guaranteeing that one of the SMA actuators of each panel had broken already, a double panel deployment can be done by turning only two heaters on.

However, the power dimensioning yielded very large margins, even in this situation. One panel can provide about 189W in worst case (i.e. 110°C) and 304W in best case scenario (-110°C) conditions. Considering that the satellite is spinning at 1 rpm with 60° sun-angle and at most two panels are illuminated, the total power the panels was estimated of being approximately 250W. Each SMA actuator heater consumes approximately 32W, while the rest of the platform requires 72W.

The battery has a capacity of 1024 Wh. Even after eclipses of 33 minutes, assuming the battery was fully charged before, the discharge is only of approximately 4%. Since the maximum discharge power of the battery is 230W, the battery can be considered in nominal situation totally available for discharges if necessary. Considering that the panel deployment was planned to be executed in sunlight in order to guarantee that the satellite's body was visible for the selfies, any insufficiency in the power provided by the panels could be compensated by the battery.

The SMA actuator actuation was estimated to take at most 60s. To ensure a deployment of all targeted panels, a maximum of 120s of heating was be accounted for in the planning of the LEOP. The result of the power balance in these conditions yielded that even activating 8 SMA actuator heaters simultaneously would have been possible.

Therefore, it was decided to plan in the SoE the release of the 4 panels at once, which is a simpler procedure than the discussed bi-phased deployment.

The SMA actuator-related preparation activities for the deployment are started with the test of all SMA actuators, done firstly on all the nominal SMA actuator heaters and then on the redundant. The result is evaluated by observing the temperatures measured by the 16 thermistors during the 1s of activation.

Secondly, having the spacecraft with a 60° sun-angle and rotating with 1rpm, one SMA actuator of each panel was planned to be broken in two separate stages: in a first passage both the **-y-z** the **+y-z**; in a second passage the **-x-z** the **+x-z** SMA actuators. In sum, the 4 lower SMA actuators. In case of a premature deployment of one of the two targeted panels at each passage, which could have happened if an upper SMA actuator had already broken e.g. during launch, the reaction would have been to force the opposite panel to deploy too, so as to ensure a symmetrical configuration and thus better equilibrated attitude.

Finally, the panel deployment was planned for an extended visibility opportunity on the 2nd day of the LEOP where the commanding time and the telemetry availability were maximized. This was a combined contact with the Weilheim and Spitsbergen ground station, where an overlap of coverage is possible. Further, the satellite is illuminated in the selected opportunity which favours the power conditions and the illumination of the satellite for the selfies. As backup opportunities, a double contact with the same ground stations one orbital revolution later and a short passage were scheduled. This optimal constellation of passages occurred once a day and therefore a delay on activities would have caused at least a day of delay in the deployment.

4 Launch and first acquisition

The launch date underwent multiple delays towards the end of 2018. The two last delays were caused by unfavourable weather conditions or by extended technical preparations required by SpaceX. This caused challenges in terms of personnel planning, as travels and accommodations for key people had to be postponed or cancelled. Rearrangement of the ground station passages had to be done several times on short notice. Since the LEOP passages had elevated priority in case of conflicts with other missions in the ground stations, no Eu:CROPIS passages had to be cancelled as a result of the delays. The final launch date of 3rd December was announced on the 2nd of December.

The request of passages as well as the re-generation of the Sequence of Events was done on the very same day of the launch. The most critical issue arising from the delay was that no official predicted Orbit Parameter Message (OPM) containing the orbit injection elements was received in advance. The OPM is an essential input for Flight Dynamics to generate TLEs, which are distributed to the ground stations and are used by the Flight Director to generate the SoE. The last OPM received from Spaceflight was for the lastly cancelled launch on the 2nd of December. On that case, the OPM was in all parameters equal to the one received for the previous launch date, on the 28th of November. Therefore, having no other input, the same was assumed for the 3rd of December. The planning had to be done by adapting the OPM received for the cancelled launch in the 2nd of December, i.e. with by shifting all scheduled activities and passages by one day. The TLEs distributed to the first acquisition ground stations were based on the adapted OPM, with an assumed launch time of 18:32:54 UTC.

The first acquisition contact was scheduled from 19:46 to 20:03 UTC in a South-North double contact over the Weilheim and Spitsbergen ground stations with a few minutes of overlap. Each station had two antennae available for tracking Eu:CROPIS. The Weilheim ground station acquisition strategy consists in having one antenna in program track following the TLEs and the other scanning the expected track in azimuth at first and then following the program tracking after 20s with time increasing azimuth deviations on both sides. Even considering worst-case scenarios (3σ) of errors in the semi-major axis (6km) and in the inclination (0.03°) at injection and taking into account the half-beam of the 4 antennae used for tracking (between 0.3° and 0.6°), the Flight Dynamics team determined that the first acquisition would be successful. The mission analysis also estimated the cumulative worst-case scenarios of time offsets and azimuth offsets in each passage of the LEOP. The maximum azimuth offset in the first acquisition double contact was of -0.26°, while the equivalent in AOS was of plus 2.5s. The offsets grow relatively fast with the mission elapsed time, underlining the importance of a successful first acquisition. 4-5h later, the time offsets were in the order of 20s-25s and the azimuth offsets of the order of 1°, making a first acquisition at that point difficult.

The SSO-A launch was observed via video stream by the operations team in GSOC. The OPM of SpaceX was not received via Spaceflight before the first acquisition. It came to knowledge of the Flight Dynamics team via unofficial communication channels as well as from the SpaceX video streaming timeline that the actual launch time was more than one minute later than the foreseen. The TLEs were adapted by Flight Dynamics assuming a launch epoch of 18:34:05 UTC, i.e. 71 seconds later than the assumed one at 18:32:54. This orbit related products were then distributed to the ground stations on short notice at approximately 19:00 UTC. Fortunately, this was well in advance before the first acquisition passages. Had the previous TLEs been used, the tracking would not have been successful in the first acquisition as such large tracking deviations for LEO satellites are too high.

Therefore, the first telemetry of Eu:CROPIS was expected in the control room at the first pass. The 4 antennae observed a signal, with a few instabilities throughout the contact. The observed bitrate was of 400kbps as expected and the frame synchronization could be done. Nevertheless, all the data was deemed faulty and therefore no telemetry was forwarded by the 2 ground stations. The correct spacecraft identification number in the telemetry frames could not be matched with the expected one. Despite having the possibility of commanding in the blind, the approach decided by the operators at the control was careful and conservative and thus no telecommands were sent.

As soon as the passages finished unsuccessfully, the Operations Team debated the situation. Firstly, it was considered that the cloud of satellites by the launcher still in proximity with Eu:CROPIS could be interfering with its emitting signal. The maximum separation between spacecraft was estimated by Spaceflight to be between the order of ~1 km to at most 15 km one orbit after Second Engine Cut-off. The frequencies used for data transmission by the other spacecraft were unknown to GSOC. It is public however that for instance the launcher transmitted telemetry in S-band as Eu:CROPIS [2]. It is likely that other spacecraft were transmitting in near S-band frequencies. In addition, it is also unknown if the impact of radio frequency transmission disturbances, including e.g. reflections or interceptions by other satellites were considered by Spaceflight.

The fact that no true OPM had been received and the TLEs distributed to the ground stations were based on the video streaming also was a topic of discussion. Nevertheless, as the spacecraft was tracked by the ground stations and the correct spacecraft ID was processed, the inaccuracy of TLEs was not the cause of the unsuccessful acquisition. The official separation confirmation and OPM was then received at 20:40 UTC. The injection elements were within specification.

Still before the next double passage at 21:21 UTC, the operations team finally uncovered the reason for the failed contact. The ground segment was expecting randomized telemetry format whereas

the satellite was not configured for this type of encoding. Randomization is a technique that ensures that the bit transition density is kept within a recommended range of values. The telemetry encoder on the satellite was configured not to use telemetry randomization although the ground segment was setup to process it that way, including the respective synchronization marker.

The second double passage, again with Weilheim and Spitsbergen ground stations, was supported again with two antennae each. Each had one baseband equipment configured to have randomization and the other still not to have it. As the signal was tracked by the ground stations and the telemetry was processed correctly by the antennae without randomization, it was communicated to the whole ground station network to switch the telemetry randomization off. In addition, the signal gaps observed one orbit earlier and the external presences in the spectrum were no longer visible. According to [2], the maximum distance between satellites of the SSO-A rideshare had increased from 1 km to 35km, as the cloud of satellite dispersed.

The first received real-time telemetry at 21:22 UTC showed the satellite was in a healthy status. The ACS mode was the expected SPPT and the transversal attitude rates (ωx, ωy) had stabilized on values of the order of 0.01 rpm. The spin-up to 1 rpm in the main body axis had already started. However, the sun-angle was still oscillating significantly between 97° and 155°. Later on, it was confirmed that at approximately 23:05 UTC, i.e., about 4 hours after separation, the spin-up of ωz to 1 rpm had been completed and the sun-angle was stabilizing with smaller deviations (below 20°), from the target of 90°. The power conditions were healthy.

After downlinking the historical data recorded between the initialization sequence of the OBC and the first failed passage, it was possible to evaluate the behaviour of the ACS after separation. The ACS controller was activated one minute after the boot as expected, transitioning from “suspend mode” to “detumbling mode”, where the B dot controller targets to reduce the total angular velocity to 0.5°/s. The data showed an anomaly occurred in separation since the tip-off rate was of 10.01°/s, whereas the specified maximum rate was of 2°/s [22]. Fortunately, the controller was robust enough to cope with this situation and stabilize the attitude. In fact, the design of the ACS and EPS system was driven by the conditions after separation in which the available power and attitude are most unfavourable in the course of the mission. 4000 random tumble cases were simulated during the development of the power system assuming a worst-case scenario of a total rotation speed of 5 rpm. The mission success rate estimated was of 98% [6].

5 Panel Deployment

The delay in the execution of the SoE caused by the difficulties in the first acquisition was recovered by the 5th passage. The remaining activities done until the panel deployment preparations were generally successful, with a nominal behaviour of the whole spacecraft including payloads. The ACS tests with different sun-angles and different rotation rates were nominal. Test pictures with the cameras of SCORE had been taken. The 1s duration SMA actuator heater tests using automatic telecommands worked properly. In sum, the satellite was in nominal condition with a sun-angle of 60° prepared for the next steps towards the solar panel deployment in the morning of the 5th of December.

Figure 3: test pictures with the cameras of SCORE, showing the panels stowed.

After uplinking the aforementioned automatic telecommands for the SMA actuator heaters, the rupture of the two first SMA actuators, **-y-z** and **+y-z**, could be confirmed via the respective decrease in the temperature derivative. However, at the following passage, at 04:56 UTC, the execution of the equivalent automatic telecommands for the two nominal heaters for the **-x-z** and **+x-z** SMA actuators got interrupted 5s after powering up. The redundant thermistor for the SMA actuator **+x-z** only increased 3.5°C after 5s, which is below the hardcoded threshold set of 5°C. This can be explained by the different placing of the thermistors relatively to the SMA actuator. Approximately 1.5 minutes later during the same passage, a retry with the redundant heaters also failed. A second retry with the nominal heaters on the next passage at 05:28 UTC had the same result, despite the temperature at the mentioned thermistor being 1°C lower at start. It was decided then to use the manual unconstrained telecommands for activating the two heaters. Time-tagged telecommands that turn all heaters off 2 minutes after the sending of the heater on commands were inserted in advance on the on-board schedule, mimicking the automatic telecommands behaviour. Afterwards, the two nominal heaters were turned off. One minute later real-time telecommands were sent to switch off

the heaters. The temperature derivatives showed it took approximately 30s for the breaking point to be reached. The issues had with the automatic telecommands showed the disadvantage of having non-configurable parameters on such critical software features, all the more when realistic tests simulating in-orbit conditions had not been possible on ground. Had it been possible to set the mentioned threshold from 5°C to 3°C, the issue would have been more easily tackled.

With the four lower SMA actuators broken, the conditions for panel deployment were met for the double passage between 08:54 at 09:12 UTC, supported by the Spitsbergen and Weilheim ground stations. The satellite was stably spinning with 1 rpm with a 60° sun-angle in balanced power conditions. As the pass started, the telemetry housekeeping generation rates of all parameters related to the attitude, power and thermal subsystems were increased. The ACS mode was changed to "deployment mode", in which no control is executed. Since the satellite's inertia tensor was to change significantly during the deployment, any control assuming the stowed configuration would be inaccurate and could even further disturb the attitude. With the goal of ensuring the return to the nominal ACS mode after the deployment, a telecommand was sent with time-tagged execution at the end of the passage.

The automatic telecommands for the four nominal heaters of the four upper SMA actuators (**-x+z**, **+x+z**, **-y+z** and **+y+z**) were sent at 08:58 but the heating got aborted 5s later. The redundant thermistor of **+y+z** had an increase of roughly 4.5°C after the 5s, falling short by 0.5°C of the minimum of 5°C. Then, the redundant heaters were commanded again using the constrain-driven telecommand, this time successfully. The heaters stayed on for 44s and got switched off automatically as soon as all the 8 thermistors measured 150°C or more. The temperature curves of the 4 SMA actuators indicated inflexion points roughly 30s after activation of the heaters [23]. Hence, it could be assumed all the 8 SMA actuators had been broken and the panels would deploy.

4 pictures were taken roughly 1.5 minutes after the assumed rupture of the 4 SMA actuators. The images were inconclusive as the cameras were partly blinded. Two more selfies taken afterwards were of no avail too because the data transmitted to ground was corrupted. The pictures would have been the most evident means of confirming the deployment of all panels.

Further, no conclusion could be made on short notice based on the temperature on the solar panels. As the satellite was exiting the eclipse at 08:57 UTC, its effect on the temperatures at the panels is for several minutes of larger magnitude than that of the decreased sun exposure a panel would have after deploying. Concerning the power provided, both APRs showed an increase in the maximum provided power, from approximately 210W to 250W. Perturbations in the sun angle, which was varying between 40° and 80°, contributed to the variations in the temperatures and the power, making it difficult to confirm the deployment of panels during the pass.

Having had no conclusions provided from power, temperature nor pictures, it was an analysis of the attitude that provided insight into the situation. Firstly, a discontinuity on the three angular rates and sun-angle was observed at around 08:59. It is trivial to conclude that at least one panel had deployed.

Also, before the panel deployment, the transversal angular velocities ωx and ωy had negligible values while ωz was roughly 1 rpm. Therefore, the angular momentum can be approximately considered as having only contribution from the main body axis **z** before deployment. Further, the expected increase in the principal moment of inertia around the z axis was of 41%. Assuming conservation of the angular momentum, a similar decrease in ωz was expected. However, the decrease of the mean value was of half only, of roughly 20%. This was considered at the time by the ACS team already an early indicator that only two panels had deployed during the passage.

The information passed in the control room was that of uncertainty as to how many and which panels had deployed, despite these assumptions. In any case, it was certain that a full and successful panel release had not taken place.

Doubts were cast to whether all the SMA actuators had broken. In one case, the temperature variation at rupture could not induce 100% certainty. In light of this and making use of the remaining uplink time still during the mentioned Weilheim-Spitsbergen double pass 08:54-09:12 UTC, it was decided to retry the activation of the four heaters at the end of the passage. As a result of the failures experienced with the automatic telecommands, the manual commands were used to switch on the heaters. Time-tagged switch-off commands were sent as precaution from a backup commanding stack with one opportunity every minute during the double passage. After 2 minutes of activation, none of the SMA actuator temperatures showed inflexion points, further pointing to that the SMA actuators had indeed already broken.

Towards the end of the pass, at approximately 09:10 UTC, a significant conclusion could be made by the ACS team. If the four panels had deployed, ωx and ωy after the panel deployment would have

continued having a mean of about 0 rad/s, despite oscillations, due to the symmetry of the body. However, the mean of not one but of both angular velocities jumped to a positive value between 0.15 and 0.16 rpm. These observations could only be done after the oscillation period, which in this case was approximately of 8 minutes. Due to the symmetry of the system in the x and y planes and the preservation of the angular momentum, it can be concluded that either both +**x** and +**y** panels had deployed or that -**x** and -**y** panels had deployed. After analysing the pictures of SCORE taken at 09:14 UTC, it could be concluded that only the -**x** and -**y** solar panels had deployed.

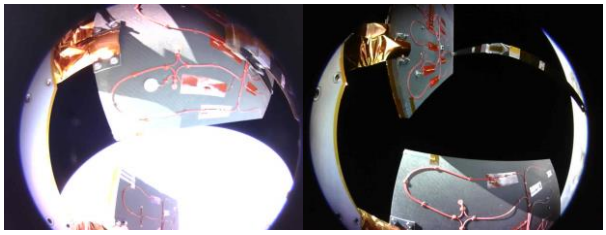

Figure 4: Pictures showing the partial deployment of the solar panels (09:14 UTC). *On the left, the upper panel (+**y**) has only barely initiated its arc towards the deployment, while the lower panel (-**x**) has almost fully deployed. On the right picture, the situation is inverted: the upper panel (-**y**) is close to its full deployment range but the lower (+**x**) hasn't. Figure 1 and Figure 6 can be used as a means of comparison.*

With this in mind and having no more accurate values to upload to the ACS, the moments of inertia for the full deployed configuration were uploaded in the last minute of the passage. The ACS control was reactivated. Finally, the last action in the double pass was to take two more selfies. The pictures showed that the panels -**x** and -**y** were deployed. It was unclear why the other two panels had yet to deploy.

At the next 5-minutes duration passage over the O'Higgins station at 09:39 UTC, it was decided to use the opportunity to activate all the nominal heaters using the manual telecommands, in order to exclude the possibility that any SMA actuator was yet to break. After doing so, no immediate impact on the attitude nor any inflection point on the temperature readings was observed. However, there were some positive news during the contact: the ACS team detected that the angular velocity over the x axis had a mean of approximately 0 rad/s. This was a consequence of the deployment of the +**x** solar panel, which had to have had occurred before the start of the pass, as no discontinuities were seen in the real-time data. Also, a selfie of that panel showed the deployment. Unfortunately, as the pass was short (8.5° was the maximum elevation) no historical data was downlinked. It could have been confirmed that the deployment of that panel had taken place at 09:38 UTC, 1 minute before the start of that passage.

The plan for the next contact with Eu:CROPIS at 10:30 UTC was to continue observing the satellite with increased telemetry rates and to upload the moments of inertia for a configuration where all panels are deployed but the +**y** panel is stowed. Speculative discussions on how to deal with the undeployed panel later on were had within the operations team. One idea that could further be analysed was that of spinning-up the spacecraft to 5rpm. However, this was not necessary. As soon as telemetry was available in the control room at the 10:30 UTC passage, it could be observed that the panel had already deployed. Not only ωx but also ωy had a mean of 0 rad/s. Both angular rates had negligible values already as the stabilization of the attitude was being done autonomously by the ACS. The housekeeping data stored on-board revealed later that the deployment of the +**y** panel had happened at 09:49 UTC, about 4 minutes after the previous O'Higgins pass had finished.

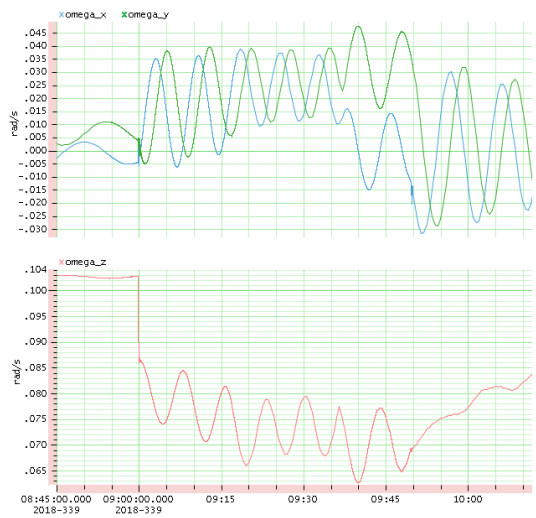

Figure 5: angular rates during the deployment of the panels. *1) until 09:00, the angular rates are stable: ωx and ωy have a mean of approximately 0 rad/s, while ωz is stable at roughly 0.103 rad/s ~1rpm; 2) shortly before 09:00, there is a sharp decrease of ωz, explained by the deployment of two (-**x** and -**y**) of the four panels. The asymmetry in the configuration and mass distribution causes tumbling. The amplitude of the oscillations of all angular rates increases; 3) between 09:00 and 09:36, ωz further decreases and both ωx and ωy increase their mean as the two already deployed solar panels were progressively reaching the full deployment; 4) At ~09:36, panel +**x** deploys; 5) At ~09:47, the panel +**y** deploys. Onwards, ωx and ωy oscillation's amplitude is larger but their means are around the 0 rad/s axis, since the mass distribution is again symmetric. ωz slowly starts increasing back to 1rpm due to the rate control by ACS.*

Pictures of the satellite body were taken using SCORE, which then validated the hypothesis – all panels were visibly completely deployed. Finally, the target sun-angle at the ACS was commanded to 0° as planned. The satellite was overall in a healthy state.

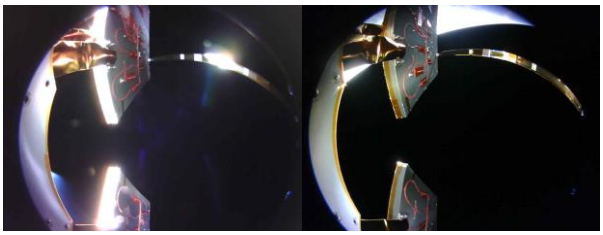

Figure 6: Selfies taken by the cameras of SCORE at 10:40 UTC confirming the full deployment of the four panels.

A dedicated analysis of the angular rates with respect to the sequential deployment of the panels will be published in June 2021 [23]. Furthermore, a structural investigation on why the panels took longer to deploy than what was tested on ground concluded that the likeliest explanation is related to temperature effects more critical than analysed and tested on the hinges made from glass fibre epoxy resin. The resin might have gotten deformed during the several in flight sun phases with temperatures that favour creep in the folded state of the panels. Because the bolts were ruptured after an eclipse phase was over, where naturally the temperatures were at nearly at its lowest over an orbit, the hinges were then stiff in the deformed state. As the sun illumination that followed in the next hour warmed up the hinges, their rigidity decreased progressively, allowing the shaded hinge from the support arm to push out the panels and therefore may have contributed to the prolonged deployment motion of the panels [23].

6 On-board computer anomaly

At the end of the sequence of events on the 6th of December before the team on site would be reduced to the minimum and to a limited on-call support, a few last activities of minor importance were planned. It was scheduled for the North-South double passage Spitsbergen-Weilheim 10:34-10:52 UTC to test the experimental high rate 3 (HR3) mode, taking advantage of the presence of the RFEA experts from RY still in the control room. Such a mode allows to double the bitrate (1600 kbps) with respect to the standard high rate mode 1 (HR1) (800 kbps).

The test of HR3 was to be combined with a recovery of historical housekeeping data of the spacecraft. In order to recover the data, a reset of the redundant memory address pointers for each memory partition had to be executed in advance. This had already been done for 2 times in the LEOP and was thus considered a riskless activity to do in parallel with the HR3 test.

The Spitsbergen passage started at 10:35 UTC with the ground station antenna configured to receive Eu:CROPIS data using HR1. The passage went nominal and the spacecraft status was green. The handover to the Weilheim antenna was performed at 10:46, still in HR1. As planned, after performing the uplink connection test and confirming a good downlink signal, the operators of the station in Weilheim were informed via voice loop of the change of the Eu:CROPIS transmission mode to HR3. Afterwards, the respective telecommand was uplinked at 10:48. The transition to HR3 was successful and the real-time telemetry continued to be processed correctly by the MCS and was available for the operations team. No indicators of any anomaly were visible whatsoever. Another uplink connection test was executed, also acknowledged successfully.

At 10:49 UTC the on-board memory address pointer reset command was uplinked as planned. This command was unacknowledged and the telemetry stream stopped. At that point a computer reboot was assumed as the most likely cause of this behavior. One minute later, which is the time that approximately the on-board computer takes to be available for telecommanding after a start-up, a blind acquisition of signal from Eu:CROPIS was attempted, but without any success.

Without any telemetry nor signal available, further actions of increasing severity were done with the goal of retrieving signal from the spacecraft. After that double passage where the signal was lost, 5 passages (O'Higgins and Spitsbergen) in the following 4 hours were used. Blind acquisitions in different telemetry rates and with different telecommands for switching off and on the transmitters were attempted. Different virtual channels (VC) for telecommanding, including VC2, which bypasses the application hierarchy on the software, were also used to no avail. On-board computer restarts as well as switchovers had no effect.

The unsuccess of all the commanding excluded a simple software failure passible of an easy recovery. A double hot-redundant receiver failure which would explain the unresponsiveness of the spacecraft was considered extremely unlikely. The transmitters are cold redundant but the two would have had to have a critical failure.

The recovery attempts assuming more and more severe scenarios on the satellite showed no immediate effect, and proved even prejudicial. Later

it was determined that the first VC2 sent TC would have been processed correctly. A first OBC switchover or reboot telecommand uploaded using VC2 would have recovered the system. Unfortunately, that was unknown and that opportunity had been irrevocably lost by sending other procedures with VC2.

Further, the fact that the pass was done in the experimental high rate 3 contributed to lively in the discussions, as indeed it later got proven it had nothing to do with the loss of signal. Also, time was consumed in experimenting solutions whereas a deeper analysis requiring more time was necessary for finding the root problem. However, it is natural to assume simple anomalies and corresponding simple solutions first.

Speculation was had on other external causes for the inexistence of signal and tracking lying outside of the satellite itself. For instance, if the TLEs distributed to the ground stations were incorrect due to unknown reasons on the processes in the ground segment. This was confirmed not to be the case. A fatal collision of Eu:CROPIS was also discussed. The GSOC Flight Dynamics team reported that the tracking of the spacecraft's body by any space object surveillance network would not yet be possible as the 64 spacecraft of similar size of the SSO-A were yet to close to one to be identified.

After the last attempt of commanding, the whole team of both shifts, gathered in a state of emergency for two hours to thoroughly discuss the situation where the anomaly arose and the possible ways forward. The most likely explanation proposed was that of the software being in an unresponsive state where no telecommands are processable.

The on-board computer was designed in a way where it is not possible to telecommand Eu:CROPIS circumventing the software to perform any critical functions such as rebooting the OBC. Such implementation with so-called hardware-direct telecommands is costly and was precluded in the design phase. As a countermeasure on design level, an FDIR had been implemented where if no telecommand from ground is received for 76 hours, the on-board computer restarts.

It was unclear whether the approach of waiting 76 hours would guarantee a recovery as it was not initially certain that the on-board application responsible for the execution of this FDIR would have been affected. The common decision of the team was then to suspend the commanding of Eu:CROPIS for 3 days and 4 hours, continuing only with observation-only passages scheduled until the 76 hours elapsed. The satellite was assumed to be in a safe state in terms of power and attitude as the systems on-board including the active attitude control should be operating nominally.

Tests were made on the engineering model to replicate the anomaly. Eventually, the anomaly could be reproduced on ground using the telecommand that set the backup packet storage memory addresses to a different position. The problem was later understood to be a deadlock between different software threads generating telemetry: the first being any autonomously generated TM packet and second the acknowledge telemetry packet. Both threads waited on each other in this case, as the generation of the autonomous telemetry waits for the execution of the telecommand, which in turn waits for the autonomous telemetry to be generated. The possibility of such a deadlock state was not foreseen because commands are generally processed fast enough that such a blocked condition is not possible. However, the resetting of addresses of the packet stores of the on-board computer is the most processing demanding telecommand of the database because the memory addresses have to be calculated from stored TM packet to stored TM packet until the target is reached. In this case, 5 partitions were addressed and the data dated back to 12h in the past.

After 76 hours since the last sent telecommand had passed, a contact over the Spitsbergen station on 9th of December at 18:49 UTC was planned. Finally, after a blind acquisition was done, a telemetry stream from Eu:CROPIS could be retrieved. The satellite was therefore in the expected orbital position and was found in a healthy state. However, many configurations on-board had been changed as a result of the FDIR's execution.

The full recovery was completed in the afternoon of the following day, in which then the LEOP officially ended, 7 days after launch.

The mission database at the MCS was patched soon after to exclude the generation of acknowledgement packets after any telecommands addressing the memory service 15 [20]. This temporary workaround prevented the behavior of reoccurring altogether.

A new software version that prevented any TM deadlock state was provided by RY and then uploaded by February 2020. The aforementioned MCS patch could be then reverted.

7 Routine Operations

7.1 Eu:CROPIS payload (PL1)

Unfortunately, due to a permanent loss of communication suffered at different times by both the greenhouses, one intended to experiment in Moon gravity and the other in Mars gravity, the biological experiment could not be carried out. In one compartment, it is assumed that the cause may have lied in critical permanent unrecoverable radiation damaged suffered by essential data processing elements at the hardware. The other compartment is expected to have undergone a malfunction during a software upload where, due to a software failure, the memory management function got permanently deleted. The attempts on the satellite to recover using a procedure tested on the ground proved unsuccessful.

7.2 PowerCell (PL2)

Out of the four sub-modules contained in two enclosures, three experiments could be activated and have produced scientific data on the course of 2019. One experiment was executed simulating microgravity, another Moon and finally other Mars gravity. The spin rate of the satellite was commanded using the reference radius of PL2. For instance, the spin rate to simulate Mars gravity is of 26.552. Overall, PL2's operations can be considered successful.

7.3 RAMIS (PL3)

The RAMIS payload for the measurement of the radiation field parameters has been operated nominally since its start on-board. RAMIS collected up to mid of March 2021 over 800 days of science data which enabled to measure the galactic cosmic radiation (GCR) and the contribution from passing the Earth inner (proton) and outer (electron) radiation belts. The collected data so far has been used to study for example the variation of GCR over the North- and South Pole of the Earth in dependence on the solar cycle and also the variation of the trapped electron contributions over the time of the mission. Further data evaluation and a dedicated webpage for data retrieval is currently under construction.

7.4 SCORE (PL4)

SCORE is permanently switched on and produces telemetry that is analysed by DLR-RY on a routine basis.

7.5 Satellite's platform

The satellite's platform continues healthy and completely performant after more than two years in orbit. The single anomaly to note is that 2 out of the 4 gyroscopes have become defect due to a power supply anomaly. As long as one gyroscope is functioning, the spin rate control can be successfully done also in eclipse phases where the sun sensors can't provide measurements.

The satellite standard state is that of a target of a 0° sun-angle and a spin rate of 17.5 rpm along the **z** axis. With the current monitoring and data retrieval requirements, 1 passage per week using HR3 mode is sufficient to observe the status and downlink the data. Eu:CROPIS will be used to test new operations concepts that may be used on the on-board software of future missions of the Compact Satellite program.

8 Conclusions

DLR, as an R&D center, seeks projects that employ cutting edge technology and science. The Compact Satellite missions combine high risk with reliability and therefore a few lessons learnt from the space operations of the first days in orbit of Eu:CROPIS will be taken into consideration for following missions.

Firstly, this LEOP was a representative example of how a well-prepared team is essential for a successful mission. It provides the robustness against adversities, may they arise from the spacecraft, from ground or external disturbances. When missions have severe risks, for example because of possible critical failure points in the spacecraft or oversensitive moments in the LEOP, it is generally recommendable that the effort invested on the training of the operations personnel is larger to ensure a good response, as was the case here.

The several and extensive training sessions helped the Eu:CROPIS team to better react to the obstacles during those days. As an example, the panel deployed was trained even 8 times in a simulation environment before the LEOP. Most importantly, the team spirit between the operations team (DLR-RB) and the satellite developer team (DLR-RY) was such that one hardly could tell whose DLR institute each person was affiliated to. The handling of tense situations that often arise in LEOPs benefits greatly from a good atmosphere in the control room, for which the common simulations before the launch are a main contribute.

Secondly, the principle "stop-think-decide-do" showed its importance in several instances. This principle is the more critical in Low Earth Orbit missions the more the available time for uplink and downlink as well as the number of passages are strongly constrained. As such, there can be a natural tendency of operators to haste reactions and to try to use every minute and every pass to act, where in some cases a pause to reflect about the situation is the

best approach to find the optimal way forward out of a problem.

At times, such as the failed first acquisition, this principle was indeed applied, as no actions were taken as a result of the missing telemetry data. However, in other cases, as in the early partial panel deployment, some reactions were actually of no use, even if of no harm either. For instance, all the attempts to force the deployment with reactivation of the heaters did not help, as the SMA actuators had already been broken. The example of the panel deployment shows that sometimes waiting is a better remedy or even the only one, rather than that of troubleshooting by trial and error. Moreover, there is a risk that reactions turn out to be prejudicial to the mission. This was visible in the reaction to the unresponsive OBC, where the commanding with VC2 during the troubleshooting precluded what would have otherwise been a recovery of less complexity from the deadlock state the OBC was in at the time. Nevertheless, especially in such scenarios where no status of the spacecraft is known, it is urgent to by all means reacquire telemetry to then urgently recover from a potentially life-critical anomaly. Any analysis is even the more difficult, the less is known of the current spacecraft state.

All in all, it must be stated that the distinction between what a severe anomaly for which a stop in operations is necessary and what a minor anomaly for which a simple reaction is sufficient is difficult to define. The Flight Director is the responsible, based mostly on experience and partly on intuition, to decide whether an approach or the other is the most adequate and then conduct the operations team accordingly. Also, the fact that major anomalies are more notable should not understate that the majority of operational problems are in fact minor incidents, which can be and should be dealt with in a “fix-as-you-go” fashion.

Another interesting point to note from the LEOP is that sometimes anomalies were not analysed correctly at first because of distracting circumstances at the appearance of anomalies. Often unnecessary effort and time is consumed to analyse occurrences which are absolutely unrelated to operational issues, delaying the closure of the problem. The fact that it was during the HR3 test that the link with the spacecraft was lost was a source of multiple discussions and analysis into the possible effects of the test, while this was not related in any way.

Therefore, while it is clear that problem analysis benefits from a wide search for root causes, it is pertinent to recommend that whenever possible sensitive satellite activities should not be planned or executed in parallel, since the risk of an incident always exists and the troubleshooting will be easier and faster the closer the conditions are to a nominal status.

Acknowledgements
Alice Bonfanti
Ansgar Heidecker
Carlos Garcia Acero
Erika Barkasz
Lukas Grillmayer
Manfred Schneller
Olaf Mierheim
Thomas Berger

References

[1] J. Hauslage et.al, Eu:CROPIS – *Euglena gracilis: Combined Regenerative Organic-food Production in Space - A Space Experiment Testing Biological Life Support Systems Under Lunar And Martian Gravity*, Microgravity Science and Technology, 2018.

[2] J. Roberts, A. Hadaller, *Behind the US’s largest Rideshare launch: Spaceflight’s SSO-A*, SSC19-X-03, 33rd Annual AIAA/USU Conference on Small Satellites, Logan, USA, 2019.

[3] D. Schulze, C. Philpot, G. Morfill, B. Klein. T. Beck, *Food Production in Space – Operating a Greenhouse in Low Earth Orbit*, SpaceOps - 14th International Conference on Space Operations, Daejon, South Korea, 2016

[4] G. McCutcheon et al., *PowerCell Payload on Eu:CROPIS – Measuring Synthetic Biology in Space*, SSC16-XI-04, 29th Annual AIAA/USU Conference on Small Satellites, Logan, USA, 2015.

[5] T. Delovski, C. Düvel, F. Greif, A. Heidecker, S. Kottmeier, O. Mierheim, F. Nohka, F. Orlowski-Feldhausen, *Eu:CROPIS AIV Program: Challenges and Solutions for a Spin-Stabilized Satellite Containing Biology*, International Journal of Aerospace Engineering, 2019.

[6] J. Fromm, *Power System for the Eu:CROPIS Satellite - Results from Design, Trade-offs, Analysis, Simulation and Testing*, European Space Power Conference, Thessaloniki, Greece, 2016.

[7] A. Heidecker, T. Kato, O. Maibaum, M. Hölzel, *Attitude Control System of the Eu:CROPIS mission*, International Astronautical Congress, IAC-14.C1.4.11, Toronto, Canada, 2014.

[8] J. S. Ardaens, S. D’Amico, O. Montenbruck, *Flight results from the PRISMA GPS-based navigation*, 5th ESA Workshop on Satellite Navigation Technologies, Noordwijk, The Netherlands, 2010.

[9] O. Maibaum, A. Heidecker, *Software Evolution from TET-1 to Eu:CROPIS*, 10th IAA International Symposium on Small Satellites for Earth

Observation, ISBN 978-3-89685-575-6, Berlin, Germany, 2015.

[10] O. Maibaum, A. Heidecker, F. Greif, M. Schlotterer, A. Gerndt, *FDIR Handling in Eu:CROPIS*, 12th IAA Symposium on Small Satellite for Earth Observation, IAA-B12-0704, Berlin, Germany, 2019.

[11] T. Glaser, O. Mierheim, C. Hühne, *Design and Analysis of a Composite Sandwich Solar Array Structure for the Eu:CROPIS Compact Satellite with as few Release Mechanisms as possible*, International Astronautical Congress, Jerusalem, Israel, 2015

[12] O. Mierheim, T. Glaser, C. F. Hobbie, S. Kottmeier, C. Hühne, *The Tape Spring Hinge Deployment System of the EU:CROPIS Solar Panels*, European Conference on Spacecraft Structures, Materials and Environmental Testing, Noordwijk, Netherlands, 2018.

[13] J. Budroweit, M. Drobczyc, *Dynamic link analysis and its in-orbit verification on a spin-stabilized satellite mission Eu:CROPIS*, International Journal of Satellite Communications and Networking, 2020

[14] F. Dannemann, F. Greif, *Software Platform of the DLR Compact Satellite Series*, Proceedings of 4S Symposium, Mallorca, Spain, 2014.

[15] A. Ofenloch, F. Greif, *Design and Implementation of an Event-oriented Simulation in a Distributed System for Automated Testing of On-board Software*, SESP 2017 - Proceedings of ESA Workshop on Simulation and EGSE for Space Programmes, Amsterdam, Netherlands, 2017.

[16] G. Morfill, E. Maurer, S. Zimmermann, R. Nibler, *Multi-Mission Synergies in Routine Operations of Low Earth Orbiting Satellites*, SpaceOps - 14th International Conference on Space Operations, Daejeon, Korea, 2016.

[17] C. Stangl, B. Lotko, M. P. Geyer, M. Oswald, A. Braun, *GECCOS – the new Monitoring and Control System at DLR-GSOC for Space Operations, based on SCOS-2000*, SpaceOps - 13th International Conference on Space Operations, Pasadena, USA, 2014.

[18] T. Beck, L. Schlag, J. P. Hamacher, *ProToS: Next Generation Procedure Tool Suite for Creation, Execution and Automation of Flight Control Procedures*, SpaceOps - 14th International Conference on Space Operations, Daejeon, Korea, 2016.

[19] ECSS-E-70-41C: Telemetry and Telecommand Packet Utilization, 15th April 2016.

[20] ECSS-E-70-41A: Telemetry and Telecommand Packet Utilization, 30th January 2003.

[21] J. Hartung, R. Nibler, C. Peat, A. Spörl, M. T. Wörle, C. Lenzen, *GSOC SoE-Editor 2.0 - A Generic Sequence of Events Tool*, SpaceOps - 14th International Conference on Space Operations, Daejeon, Korea, 2016.

[22] A. Heidecker, M. Schlotterer, O. Maibaum, E. Panzenböck, S. Löw, M. Markgraf, *Magnetic Attitude Control of a Spinning Spacecraft: Flight Results and Lessons Learned from DLR's Compact Satellite Eu:CROPIS*, 12th IAA Symposium on Small Satellites for Earth Observation, Berlin, Germany, 2019.

[23] O. Mierheim, T. Glaser, F. Orlowski, S. Kottmeier, C. Hühne, *On Orbit Deployment of the Eu:CROPIS Solar Panel by GFRP Tape Spring Hinges*, European Conference on Spacecraft Structures, Materials and Environmental Testing, 2021.

[24] A. Heidecker, M. Schlotterer, *In-Orbit Experiences with the Spin Stabilized Attitude Control System of Eu:CROPIS*, 11th International ESA Conference on Guidance, Navigation & Control Systems, to be published in June 2021.