ON ORBIT DEPLOYMENT OF THE EU:CROPIS SOLAR PANEL BY GFRP TAPE SPRING HINGES

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KEYWORDS

Deployment, Panel, Hinge, Tape Spring Hinge, In-Flight-Demonstration

ABSTRACT

Eu:CROPIS is a compact satellite featuring a biological payload [1-10]. The Satellite was launched on December 3rd 2018. The cylindrical Satellite of 1m diameter has four deployable panels for power generation. Those panels are connected to the main structure by glass fibre reinforced polymer (GFRP) tape spring hinges. The hinges, comparable to curved metallic measuring tapes, have elastic energy stored when flattened and folded and thus deploy the panels by simply unfolding. When unfolded the hinges snap into their original shape and support the panels with considerable stiffness. No friction or mechanical locking is involved in the deploying process.
The presented paper focuses on the practical handling of the hinges and the mechanisms during the final integration and the deployment process.
The integration of the panels requires some special consideration. The hinges are not able to support the panels under gravity. The release mechanisms only work at a correct positioning of the panels. The measures taken to ensure the integrity and functionality of the hinges and mechanisms are described and examples are given for a correct and a false outcome.
The separation is done by breaking a bolt with a heated bushing from shape memory alloy. Though reliable the separation cannot be timed down to the second and there is no direct feedback of the separation. To prevent an uneven opening of the panels several on orbit pre-tests are performed to ensure the functionality of the mechanisms for the actual deployment. At the actual separation the heating is monitored to ensure that all mechanisms are activated and the separation is working as proposed. Furthermore, a method was developed to detect the successful breaking of the bolts by use of the heating temperature data. The paper describes these checks and surveillance methods.
As not all things go as planned some decisions were to be made before and at panel deployment. Also, the unfolding of the hinges was slower than during the on ground. Tests were made to simulate and understand the on-orbit behaviour. Lessons learned for further use of the mechanisms are presented.

1. SOLAR PANEL MECHANISM DESCRIPTION

The Eu:CROPIS compact satellite has four panels of 0.8m² each that provide power to the satellite. During launch these panels are folded to the sides of the satellite (stored configuration, Figure 1). In Orbit the panels are deployed to form a plane in a windmill configuration with the satellite in the centre (deployed configuration, Figure 2).
In the stored configuration each panel is connected to the satellite by just 2 bolts, which can be released by TiNi frangibolt mechanisms [14].

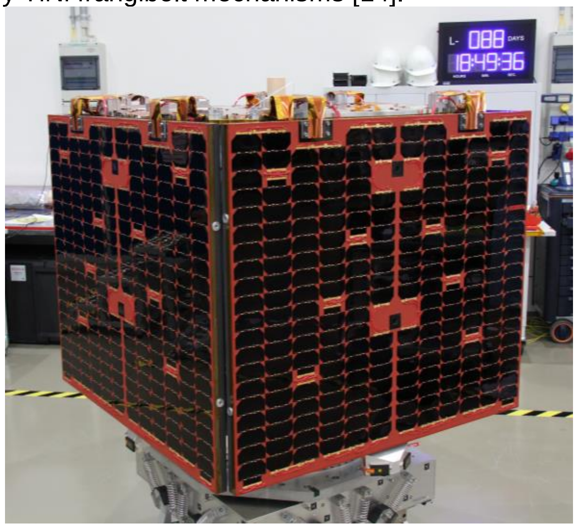

Figure 1. Satellite in stored configuration (flight model)

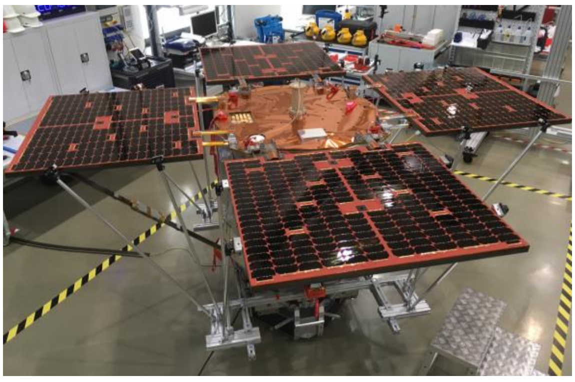

Figure 2. Satellite in deployed configuration (flight

model)

Both bolts are required to securely keep the panels in place under launch loads. The panels are further supported by standoffs, two beside each bolt. To securely push the panels on the standoff the panels are bend by the bolts by about 2mm [17].
After launch the separation can be sequentially with one bolt keeping the panel roughly in place. Once the second bolt is released the panels are free to deploy.

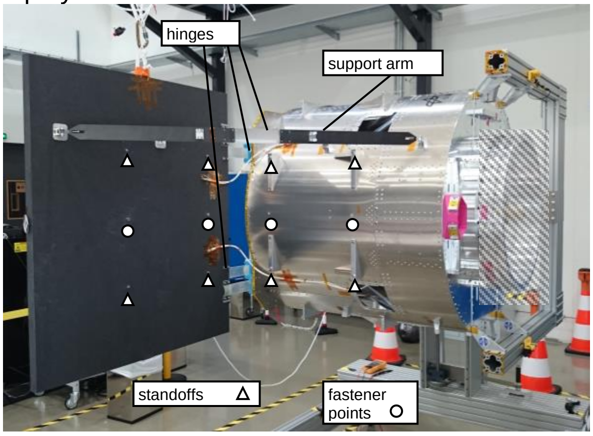

Figure 3. Satellite with one deployed panel (structural model)

The panels are supported by four tape spring hinges and one support arm. The support arm provides some additional stiffness. As the satellite is spin stabilized the rotation can trigger a vibration at the panels. Thus, the panels are required to have a deployed natural frequency above the rotational speed of the satellite which is ensured by the supporting arm.

1.1. DEPLOYMENT SYSTEM

After the release the panels are deployed by their four tape spring hinges and the support arms (Fig. 4). The support arms themselves have a tape spring hinge in the middle to support the deployment and to stabilize the arm in the deployed configuration.

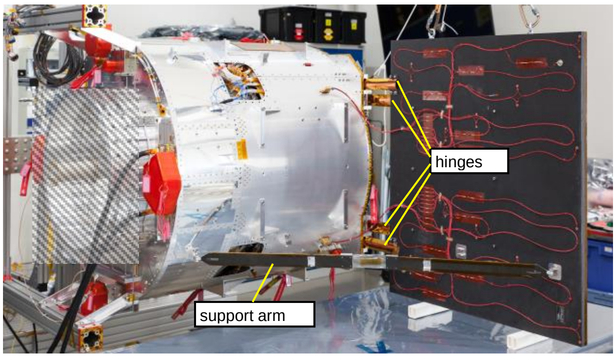

Figure 4. Satellite with one deployed panel (flight model)

The hinges are connected to the upper and lower side of the panel (The upper side being the one with cells). Therefore, the bending stiffness of the panel support is not defined by the bending stiffness of the hinges but rather by their axial stiffness. As the panels have a considerable thickness of 31mm the hinges are able to support the panels statically under earth gravity.

1.2. TAPE SPRING HINGES

Unlike conventional hinges the tape spring hinges principle is based on elastic deformation. The tape spring hinges are part circular shaped glass fibre reinforced plastic profiles. The shape gives considerable stability when straight but can be flattened and bend to a great extent. The flattening reduces the bending stiffness in the flattened area by magnitudes (factor 1800 in the given configuration). Once flattened, the hinges can easily be deformed to the stored position. The bending curvature can be oriented in the same or in the opposite direction of the undeformed profile [13] (principle depiction in Fig. 5). Further information about tape springs can be found in [1], [12] and [16].

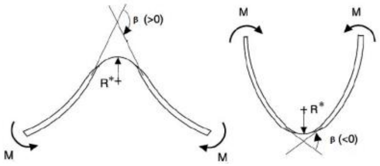

Figure 5. folding of a tape-spring with and against the curvature of the undeformed profile [13]

Curvatures in the opposite direction have a higher elastic torque. Where bended the hinges keep their flattened profile. It behaves like a thin belt of GFRP. This flattened area can be located anywhere along the hinge's length. Only at the very end of the hinge the curvature can lead to cracks, if the profile is fixed.
When released the hinges deploy the panels with moderate torque. Once the hinge straightens out it pops into the original curved profile and thus regains a higher stiffness along its length.
Therefore, the tape spring hinges perform all conventional tasks of a deploying hinge in just one piece, hinge, elastic energy for deployment and locking.
There are also differences present. The tape spring hinges do not provide a defined hinge axis. Also they do not provide an indication for the mechanical locking.
The hinges are made from glass fibre interglas 92125 and Lange&Ritter LR160/LH163 resin. The raisin is treated at 100°C. Interglas 92125 is a koeper woven fabric of 280g/m². A peel ply is not used to avoid a textured surface with possible crack initiation (note that in earlier tests peel ply was used which is visible on the hinges on some figures (e.g. Figure 6) used here).
Glass fibre is chosen because of their high ability to bear elastic energy, thus the product of allowable strain and stiffness.
The geometrical properties are:

- → free length of 140mm
- → curvature radius 30mm
- → thickness t of 0,25mm

→ width of 45mm

In contrary to the high ability to bear elastic energy in deformation GFRP has the disadvantage that the resin is quite sensitive to atomic oxygen (ATOX), temperatures above 80°C [15] and UV-radiation. Single layer insulation is used to protect the hinges against these environmental influences. The projected maximum temperature on orbit is analysed to be 40°C.

2. INTEGRATION OF THE PANELS

2.1. Hinge Integration

Four hinges are installed on each panel, two on top and two on the bottom. Top and bottom hinges are oriented in opposite direction. Due to the differing orientation and the distance of the upper and lower hinges the deformation in the folded state is significantly different (Figure 6).

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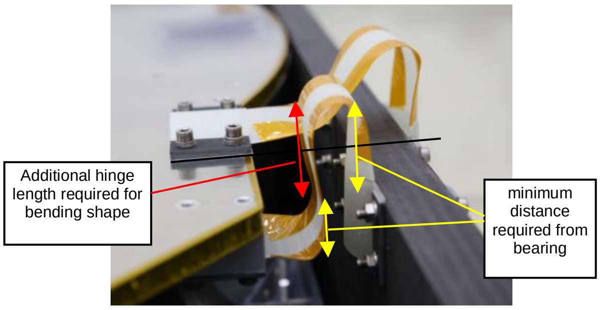

Figure 6. Tape spring hinges interface with panel in line with the upper hinge bearing.

Like conventional hinges where the axis should be aligned, the tape spring hinges need to be fixed to be aligned in the deployed state. Otherwise one or more hinges might not snap into the curved profile.

With the hinges installed in the folded state measures have to be taken to ensure the correct position of the hinges. With thin glass fiber hinges it is not possible to ensure the correct position of the hinges if upper and lower hinge are connected separately.

To ensure the accuracy of the installation the hinges are preinstalled on the panel. On the satellite side of the hinge a bracket is used that connects upper and lower hinge. This bracket is installed with the tape spring hinges unfolded. It has to be checked that the distance and parallelism to the panel and the bracket on the other side is accurate.

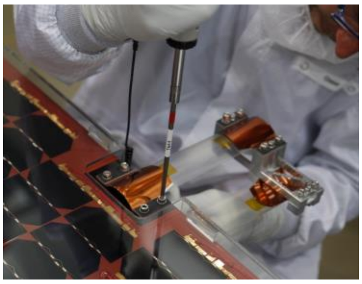

Figure 7. Pre-installation of hinges with a bracket on the satellite side.

Once the panel is fixed to the satellite the hinges can be folded and the bracket with both hinges is connected to the satellite.

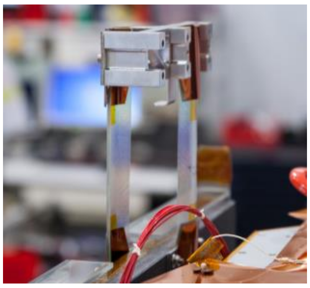

Figure 8. Panel installed at the satellite without the hinges folded.

When folding the hinges, it has to be ensured, that the curvature in the hinge is not pulled towards the ends of the hinge where the profile is stiffened by the interface. This is especially critical on the lower hinge where an additional sheet metal is used to provide a save deformation of the hinge (Figure 9).

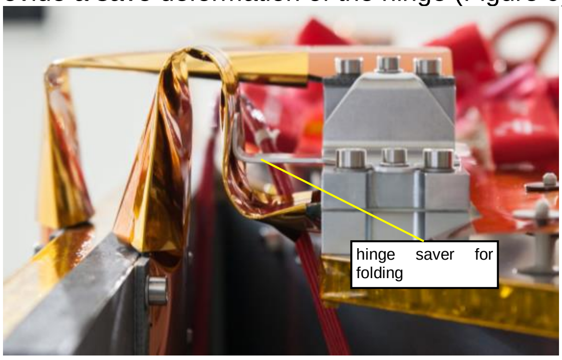

Figure 9. Hinges folded and brackets connected to the satellite. (hinges covered by SLI)

After folding the brackets have to be checked to be at the same distance and parallel as before.

3. MECHANISM INSTALLATION

The panels are connected via TiNis Frangibolt mechanism. The principle of separation is by breaking a notched bolt due to an expanding bushing around the bolt. The bushing elongates due to a shape memory alloy effect and thermal expansion. This puts the bolt under tension and breaks the bolt. It has to be ensured though, that all the parts which are compressed by the bolt – including the elongating bushing - do have sufficient stiffness. If this were not the case the elongation of the bushing might rather deform the parts of insufficient stiffness and not the bolt. This would prevent the rupture of the bolt. Insufficient stiffness in this case does not refer to materials but to faulty installation causing gaps in the load path. This can be particles in between the connected parts, insufficient alignment of the parts, or even a deformed part (e.g. a bended washer) in the load path compressed by the bolt.

In Eu:CROPIS the expanding washer is inside the satellite and can not be reached during the installation process of the panels. The same is true for the nut tightening the bolt on the satellite side. Parts are secured in place before closing the

satellite.
One possible failure case before launch would be an undetected activation of the mechanism. The mechanisms being a bushing of shape memory alloy expand once the temperature for the shape memory effect is reached. The mechanisms have to be reset in a compression tool before the next activation otherwise the heating would not break the notched bolt.
To ensure the detection of unplanned mechanism activation a full set of witness bolts is used. After the mechanism bushings are installed inside the satellite, notched bolts are torqued in the mechanisms even if no panel is installed. In case of unintentional activation, the bolt would break. Thus, in an inspection an activated mechanism can easily be detected.

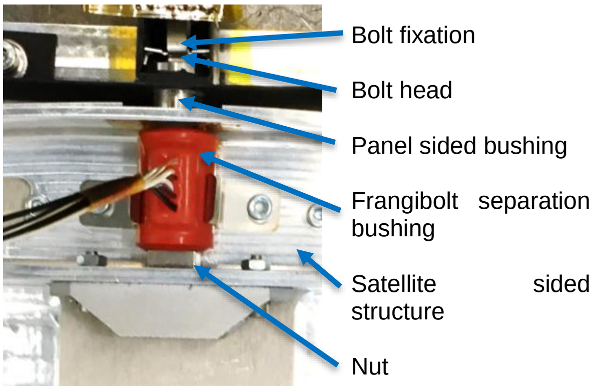

Figure 10. Separation mechanism setup

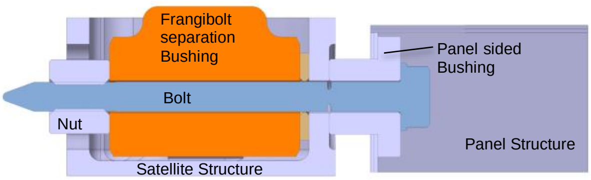

Figure 11. Separation mechanism setup cut view

It needs to be ensured that the bolt will break in orbit even though the setup cannot be visually inspected. Therefore, the installation of the bolt is not done by simply using a torque wrench, but by measuring the torque over rotation angle. The incline of the curve gives a measure of the stiffness of the compressed parts by the bolt. The procedure is to rotate the bolt by steps of 10deg rotation and measure the torque needed. At first the torque rises at much lower incline. This is the bending of the panel putting preload on the standoffs. Once the bracket at the bolt is in contact the torque is strongly increasing with each rotational step. Once the torque is reached the final curve is compared to the previous testing curves and checked for the correct shape and incline.
If the torque increase per step is not in the same range as in previous test this indicates a flaw in the system.
The following Figure 12 shows three examples, a torqueing procedure in the test campaign, the torqueing of a bolt for flight and a torqueing with the nut stuck.
The measurement starts at roughly 25Ncm. Below that value the measurement software does not react. From there it takes roughly 150deg of bolt rotation until the panel is deformed and on block. From there it should be less than 60 degrees of rotation until 200Ncm are reached.

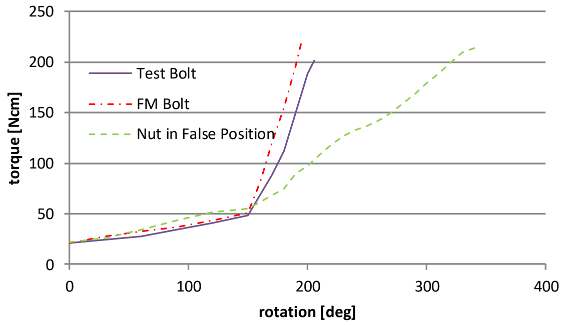

Figure 12. Torque over rotation for separation bolts

The dashed curve shows a testcase where not the complete setup was assembled thus the nut was not kept in its slid. The nut settled on the back of the satellite sided structure and deformed the structure. It is directly visible in the curve that the incline is reduced and the bolt is not likely to break.

4. MECHANISM ACTIVATION

The mechanism has two heating cycles and two temperature sensors. Each is marked with either primary or secondary

5. ON ORBIT SEPARATION SEQUENCE

Eu:CROPiS is a spin-stabilized satellite. A non-synchronized deployment leads to tumbling of the satellite which causes problem in the attitude control if the state consists for too long. The mechanisms used cannot be timed to the very second since they are heated up to break the bolt which has a spread of about 5 seconds. In the following the sequence and checks to deploy the panels are explained.

5.1. Heating Check

This test is done to check the functionality of the primary and secondary heating cycle in each actuator. The actuators are switched on for one second. A clear change in temperature needs to be visible in each of the two temperature sensors per actuator. The tests for the primary and secondary heaters are done on different passes to give the actuators time to cool down.

5.2. Separation order

The interfaces to the solar array are not separated in one pass to lower the risk of bigger time gaps between the panels. In a first run the ones further away from the hinges are separated. The folded hinges provide some force away from the satellite. That pushes the panel in place, as long as the upper Bolt is installed.

5.3. Heating sequence

The heating is done with some automation involved. This is done in case the contact to the satellite is lost. It shall either separate all 4 mechanisms or stop the heating before one panel deploys.
Once the command for the mechanism heating is send it is checked that the temperature rises on all mechanisms. If the temperature has not increased by 5k after 5 seconds the heating is stopped at all mechanisms. Also, if the two thermistors deviate by more than 20°C the heating is stopped. This is to check, if a thermistor has failed. If these starting criteria are passed the heater is kept on until the maximum temperature of 150°C is reached or 120s have passed.

5.4. Separation detection

With the given setup the separation can not be detected directly. To ensure that the mechanisms are separated the temperature over time can be used. With increasing temperature, the heat is conducted into the adjacent structure. The flow of heat increases with increasing temperature in the mechanism, thus the temperature over time increases at a lower rate. Once the bolt is broken the contact from the bushing to the adjacent structure is reduced and the temperature increases at a higher rate (Figure 13). This can be detected as an inflection point as shown with the first derivative of temperature over time in Figure 14. An average of 10 values is used to smooth the derivative and get a comprehensive curve.

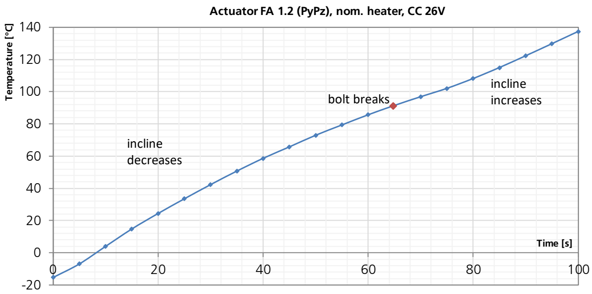

Figure 13. Temperature over time while heating the actuator

This method was tested several times up to the end to end test and a test controlled from the control room in a simulated environment.

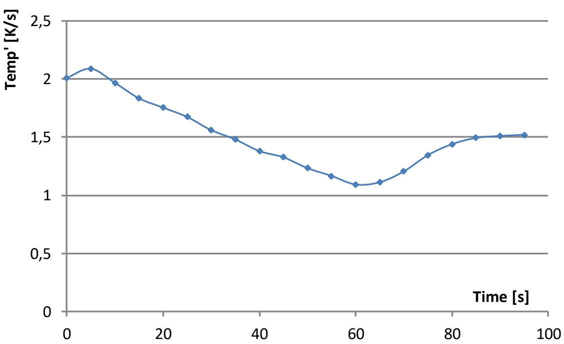

Figure 14. Incline (first derivative) of temperature over time while heating the actuator

6. THERMAL VERIFICATION

The tape spring hinge mechanisms were part of two thermal verification tests on acceptance level.

6.1. Thermal Vacuum Cycling (TVC)

The System Thermal Vacuum Test was part of the Eu:CROPIS FM Campaign and served as thermal functionality test for the fully integrated satellite bus with applied radiator surface and solar panels.
Aim of this test is to prove the operability of the system in hot-case and cold-case conditions and to collect the orbital equilibrium temperatures for the eclipse cycle in a solar simulation run. To simulate the environmental conditions, the Eu:CROPIS flight model was set up inside the DLR-RY thermal vacuum chamber (WSA) and saw three thermal cycles while non-operative. Functionality was shown by booting to safe mode in the first and last cycle (hot switch-on, cold switch-on).
Afterwards the flight model was cycled to orbital average temperature with cold background and five orbits of 61 minutes sun- and 35 minutes eclipse phase were simulated using the vacuum chambers solar simulator to prove the feasibility of a later to be performed full Orbit Simulation Test.

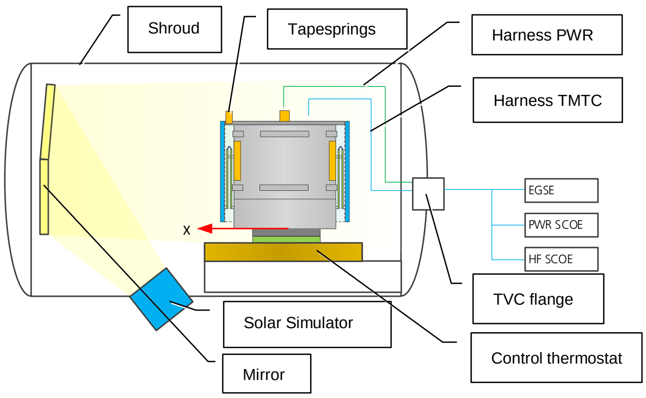

Figure 15. General Thermal Test Setup

For the TVC, two tapespring hinges on the +x side of the spacecraft were instrumented with PT100 temperature sensors to evaluate the thermal behaviour. Therefore, the temperature sensors are applied on a strip of Kapton tape with a cyanoacrylate glue in the bending area of the outer tapespring. To test the thermal impact of the single layer insulation (SLI) the +X/-Y-tapespring was covered with SLI on the outer side, while the +X/+Y tapespring remained uncovered. For operative reasons the SLI only covered the sun-facing side of the GFRP, thus deviating from the actual flight setup. The chamber parameters are given in Table 1.

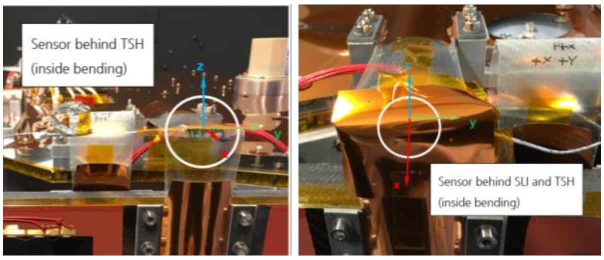

Figure 16. Tapespring instrumentation +X/-Y (left) and +X/+Y (right)

Table 1: TVC parameters

Parameter	Gradient to Cold Case	Cold Case	Gradient to Hot Case	Hot Case
Vacuum	<1e-5mbar			
Shroud temperature	-5°C	+5°C	+35°C	+27°C
Control thermostat temperature	-5°C	+5°C	+35°C	+27°C
Operation time / Dwell time	12h est.	1h	12h est.	1h

The cycling of the spacecraft was driven by both WSA shroud and a control thermostat setup serving as external heat sink. This combines heat transfer by radiation (shroud) and conduction (thermostat) to enhance the maximum temperature gradients. Figure 17 shows, how a spacecraft internal thermistor follows the conductive heat transfer (green), while the tapesprings are only affected by the radiation environment (red/blue). Temperatures deviated approximately 5 °C from the maximum shroud temperature (HC 30 °C / CC 0 °C).

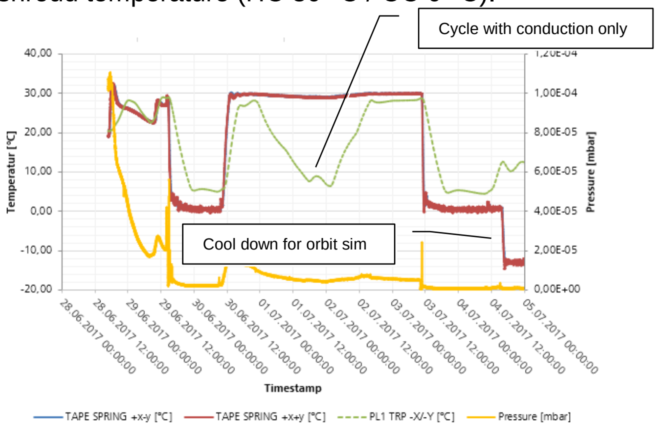

Figure 17. Thermal Vacuum Cycling

During the orbit simulation part of the test, five solar cycles with a power density of 1kW / m² full solar spectrum have been performed. During this period, the temperature of the tapesprings again followed the radiation environment, but to a far less effect compared to shroud-driven cycling, with temperatures between -11.5 °C to -6 °C approximately (Figure 18). A slight difference in the time constant and maximum temperatures can be observed for the covered and uncovered tapesprings. The solar array backplane was cycling between HC 50 °C and CC -5 °C. It was then estimated, that the peak temperature gradient between +X tapesprings and +X panel may be caused by the single layer insulation wrapped around the tapespring hinges.

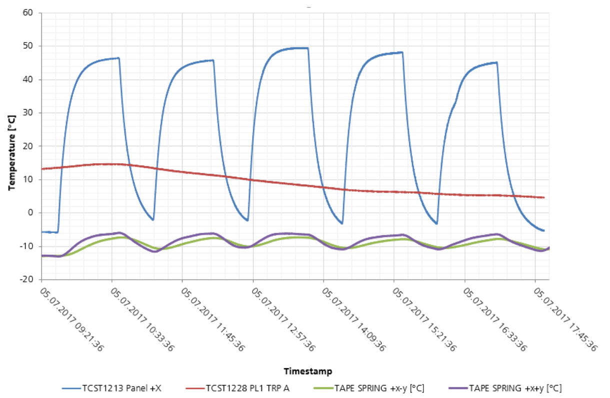

Figure 18. Orbit cycles during TVC

6.2. Orbit Simulation Test (OST)

The System Orbit Simulation Test was part of the Eu:CROPIS FM Campaign and served as thermal shakedown test for the fully integrated satellite bus with applied radiator surface and solar panels.
To simulate the environmental conditions, the Eu:CROPIS flight model was set up inside the WSA and cycled to orbital average temperature. At least 2 x 48h of 96 minutes orbit simulations were performed using the WSA solar simulator while operating the satellite as in early operations phase and in an autonomous state. Furthermore, the test served as a low temperature pre-flight bakeout for the flight hardware. The test featured a setup as in Figure 15, but lacked the conductive path provided by the control thermostat. The chamber parameters are given in Table 2.

Table 2: OST parameters

Acquisition Mode (89W)	Dwell to equilibrium	Sun Case	Eclipse Case
Vacuum	<1e-5mbar		
Shroud temperature	6°C	-20°C	-20°C
Control thermostat temperature	n/a	n/a	n/a
Operation time / Dwell time	~24 hr	61 min	35 min
Solar Simulator	off	on (1400W/m²)	off

This time the tapesprings have not been instrumented due to operational reasons. The second phase of the test, 48 h autonomous mode, had to be stopped after 15 h due to several payload non-conformances, thus the total simulation time added up to 63 h.
During the TVC post-test analysis it was discovered, that the geometry of the simulated sun beam during orbit simulation did not cover the whole solar panel plane of the spacecraft due to a mismatch of the optical axis and the panel centre point. This led to a

significant shadowing of the panel edges, causing the tapesprings to accumulate much less solar energy than anticipated (Figure 19). Even with a correction of the spacecraft position in the WSA beam axis for the OST, the total diameter of the condensed beam still causes partly shadowing of the tapesprings and therefore leads to an unintended undertesting.

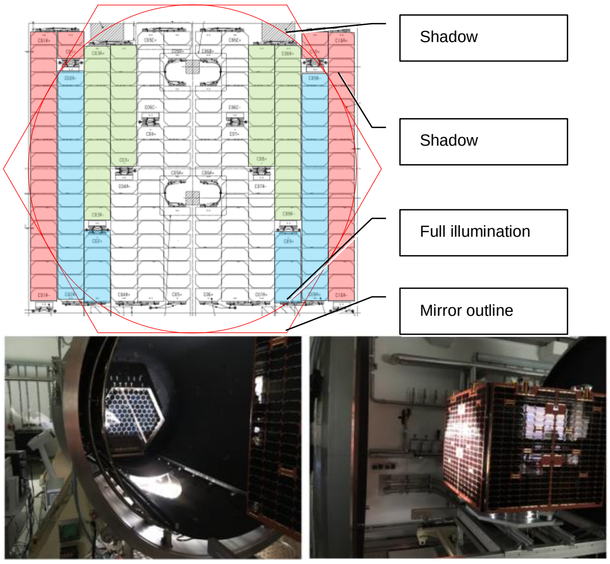

Figure 19. FM illumination test

7. ON ORBIT DEPLOYMENT

During the deployment the satellite was spinning at a rate of 1rpm. This is to keep the satellite stable and to keep the temperature in a moderate range.

7.1. Separation

For the actual on orbit deployment a pass was chosen right after the satellite left the eclipse and had a double pass to have a long satellite contact in daylight. In the control room telemetry, the temperatures and the first derivative of the temperature are included as graphs. The temperature values are read every second (Further information on operations is found in [18]).

The heating checks were performed successfully for the primary and secondary heater on all 8 mechanism.

For the actual separation of the panels the heating check stopped the heating. Even though the heating was not slower than in previous test the internal calculation of the heating suffered from a delay and the reading cycle of one second.

The problem could be solved by using the secondary heater. Previous tests showed that the secondary heater takes longer to separate the bolt, but has a higher temperature reading. It seems as if the secondary heater foil is closer to the temperature sensor and further from the bushing.

Though the use of the secondary heater solved the problem to pass the positive heating check it made the detection of the separation more difficult. While the separation of the bolt by heating with the primary heater typically happens at a temperature reading between 80°C and 95°C, the separation by heating with the secondary heater happens at a temperature reading of about 120°C. This is just about 3 to 4 seconds from the switch of at 160°C. For this reason, sensors had only 3 to 20 additional temperature readings before the heating stopped. 3 readings are not sufficient to detect the separation with an averaging over 10 readings. The following Figure 20 shows the temperature reading and the averaged incline. Only one incline shows the separation clearly.

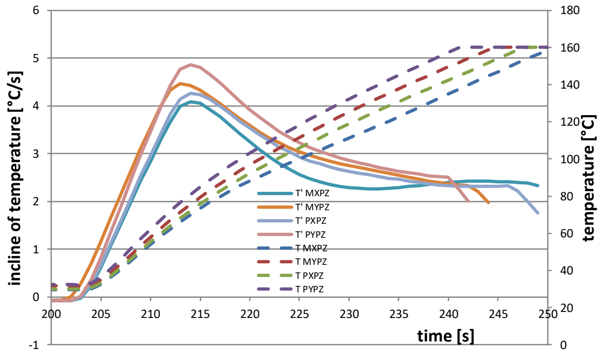

Figure 20. Temperature reading and incline of temperature from the final deployment. The incline is averaged over 10 temperature reading values.

This could be solved by manually checking the data offline after the actual separation. With an averaging of just 3 readings the separation can also be detected in the other temperature readings (Figure 21).

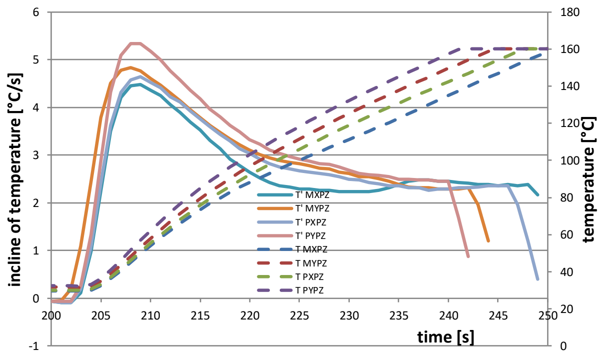

Figure 21. Temperature reading and incline of temperature from the final deployment. The incline is averaged over 3 temperature reading values.

7.2. Deployment

The deployment of the panels was expected two happen in the time frame of about 10s. In reality it seemed like the panels did not deploy at all. The first sign after the separation was a change from a rotation of only one axis to a tumbling of the rotation around the different axis. Pictures from onboard cameras confirmed that the panels had not open fully, 2 panels had barely moved. As the check for mechanism separation was not conclusive at first (see chapter 7.1) a second heating of the separation mechanisms was tried. Since this was not the cause it did not solve the problem. The images of the panels showed a slow movement towards the deployed state at closer inspection (Figure 22 and Figure 23).

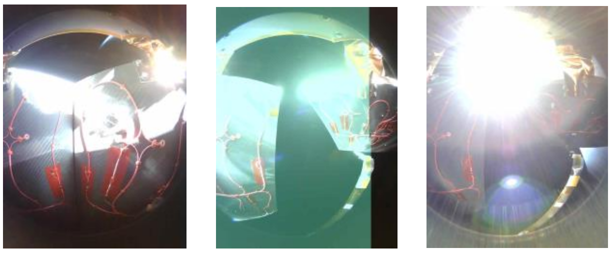

Figure 22. Panel +X-Y positions before, 3 and 6 minutes after deployment.

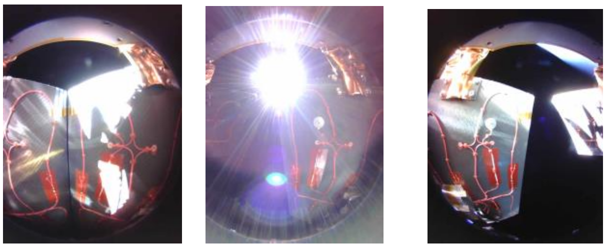

Figure 23. Panel -X+Y positions before, 3 and 6 minutes after deployment.

The tumbling was still present at the next contact half an hour later and had disappeared at the following contact. Additional images confirmed the deployment.

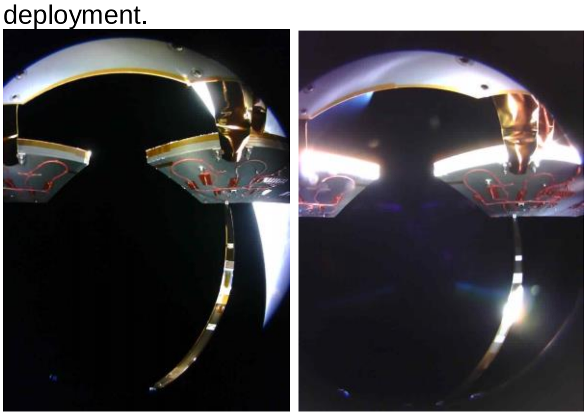

Figure 24. Deployed panels: X+Y (right), +X -Y (left)

The tumbling could be spotted on the rotation telemetry of the satellite. Figure 25 shows the angle between sun and satellite and the angular rates around the three axes.

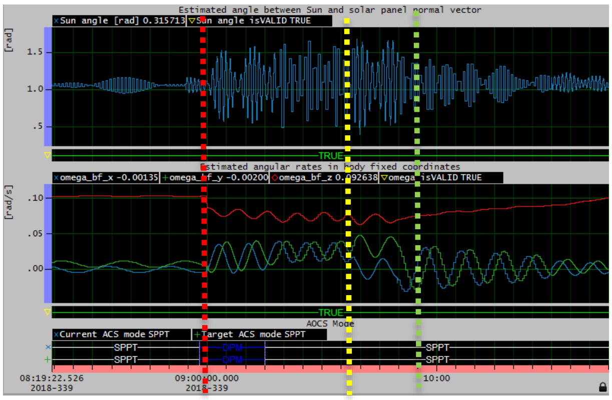

Figure 25. Angle between satellite Z-Axis and the sun and rotation rates during the deployment

The panel release is marked with the red dotted line. The increased oscillation in sun pointing angle and the changes in angular rates around all axes is clearly visible. It is believed that at the yellow dotted line an additional change is visible and from the green line on all solar panels were fully deployed, such that the attitude control system was able to stabilize the satellite again.

8. HYPOTHESIS ON THE REASON FOR THE DELAYED PANEL DEPLOYMENT

To find the reason for the delayed deployment several hypotheses are checked though some can be ruled out right away:

The panels might have gotten stuck. This is very unlikely though. The design has been severely checked for collisions and it would not explain why all panels got stuck and it would also not explain the slow movement. Thus, this is not assumed to be the cause.

The hinges made from glassfibre reinforced epoxy resin could have several causes of permanent deformation.

Loads above the failure loads can lead to fibre and interfibre failure. Especially interfibre failure does not necessarily lead to complete structural failure but can reduce the stiffness significantly. A structural failure would not explain the final deployment though.

Radiation or atomic oxygen could have degraded the resin. But this would also not explain why the panels where finally able to deploy.

Creep is a slow deformation over time a deformation that epoxy resin is known to be prone to do. But the hinges were stored for a long in tests and the deformation was not critical.

Higher temperatures can have two effects on the epoxy raisin. It can accelerate the creep rate and it can weaken the epoxy raisin once the glass transition temperature is reached. This could explain the behaviour. The satellite left the eclipse right before the separation. Therefore, if the hinges had been deformed before during the sunlight passes, the hinges would have kept their folded state. Once the hinges are warmed by the sun to higher temperatures they lose the stiffness to stay in the folded state. Each panel has one hinge on the support arm. These hinges were covered from the sun by the panels before deployment. Therefor it can be expected that these hinges were still able to push the panels outward once the hinges at the edge of the panel lost their stiffness. This possible explanation has the contradiction that the presumed temperatures are far below the glass transition temperature of the resin used.

Due to renovations in the test facilities and reduced presence at the laboratories the checks on this hypothesis are not all finished yet.
It was found though that a hinge that was kept in folded state in the oven showed permanent deformation down to 65°C. The following Figure 26 shows a hinge that was kept folded at 80°C for 1 hour and was released afterwards at room temperature. The shape after release fits quite well to what was observed at the on-orbit deployment.

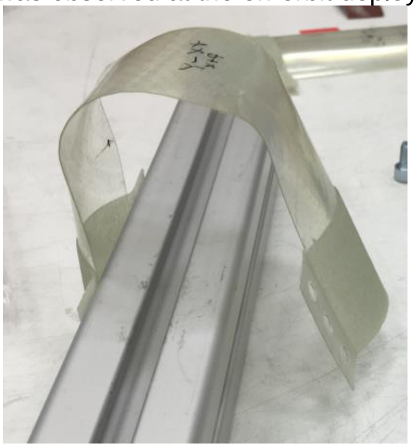

Figure 26. Deformed hinge after being subjected to 80°C for 1 hour.

A preliminary study showed that under several assumptions like the way the insulating foil touches the hinges, and the way the sun hits the hinges at the curved position can lead to temperatures up to about 68°C. For that the foil directed towards the sun needs to touch the hinge and the foil directed away from the sun needs to not touch it. The sun must also heat the foil for a longer duration of time. As can be seen in Figure 9 this situation is actually present as the curvature of the hinge puts tension on the one side of the foil while it puts the foil on the inner curvature into wrinkles. Until further tested this is the most probable explanation for the observations at the panel deployment.

9. CONCLUSION, LESSONS LEARNED AND OUTLOOK

Several measures were taken to ensure the deployment of the panels. Most of them proved very effective. But some problems were not detected in advance.
The installation of the panels ensured the correct positioning.
The tracking of torque over rotation proved very effective to detect any installation flaws and gave much confidence for the release. The same is true for the witness bolts used.
The detection of the separation worked generally well. It was missed though to test the detection with the secondary heaters as an end to end test including the telemetry provided by the satellite and the control room interface. This was done only with the primary heater. This cost some time and added to some confusion when the deployment did not go as planned.
The temperature sensibility of the hinges was judged to not be an issue in advance as the prognosed highest temperature of 40° was well below the expected glass transition temperature of 80°C. Thus, no special focus was put on the hinge temperature in the TVC. Test, that could have uncovered this issue could - and in retrospective should - have been:

→ Test of the folded hinge for the highest temperature and time after which it straightens sufficiently.
→ Test for the temperature seen by the hinge under orbital conditions. This test could have been part of the orbital simulation campaign, but with dedicated focus the hinge temperature, the radiation on the hinges and the functionality of the hinges after the test.

For further use the following steps are foreseen:

→ An orbital simulation test with the Eu:CROPIS configuration to reproduce the on orbit behaviour.
→ Change of the resin system to one with higher temperature stability.
→ Dedicated test on the behaviour of the folded hinges stored under higher temperature conditions.
→ Orbit simulation tests of hinges with insulating foils including a deployment at the end of the test.

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