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First Galileo Spacecraft Decommissioning: Planning and Operations execution

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Abstract

The Galileo constellation is the Europe's Global Navigation Satellite System which provides highly accurate positioning and timing information to users all around the globe. EUSPA is the Service Provider on behalf of the European Commission, while Spaceopal with its partners DLR GfR and Telespazio are responsible for day-to-day operations as the Galileo Service Operator. The constellation is composed of two spacecraft families, IOV (In-Orbit Validation) and FOC (Full Operational Capabilities) which are controlled and monitored by two control centres located in Germany and Italy. The nominal configuration of Galileo consists of 8 operational satellites plus two spares, per orbital plane. After launch "L13" which added 2 more satellites to the constellation, the Galileo constellation has been completed, although it is planned to launch 6 additional satellites to account for the satellites that need to be decommissioned. The two first IOV Galileo satellites were launched back in 2011 and with an initial lifetime of 12 year some of these spacecrafts are already overperforming, still providing accurate, reliable, and robust service. However, some other satellites are no longer providing the expected service, making it necessary to decommission them to free up slots for the new, upcoming satellites. The goal of this paper is to present the most relevant operational aspects, from the disposal into a graveyard orbit, and subsequent tank depletion, to the final passivation of the satellite. Every aspect of the campaign has been carefully assessed to ensure a safe and controlled end-of-life procedure. Some of the challenges came from the design of the spacecraft itself, while others are linked to managing the resources to maintain the nominal constellation and inserting new satellites into service by performing several LEOPs and IOT campaigns. Operational and orbital constraints have also played their part in shaping the way the activities are planned to be executed. This paper will present and discuss all these topics and will explain the reasoning behind each decision. Additionally, it will present improvements to certain operational approaches implemented to address various issues encountered, which could be applicable to a range of activities and potentially extended to other constellations with similar characteristics.

Keywords: Galileo, GNNS, Decommissioning, Operations, Satellite, Passivation

Nomenclature

r_p: Perigee radius
e: Orbital eccentricity.
i: Orbital inclination.
Ω: Right ascension of ascending node
μ: Earth's standard gravitational parameter.
θ: Orbital plane change angle.
$\Delta v_{max,inplane}$: Max allowed Δv projected in the orbital plane.
γ_{max}: Max yaw deviation w.r.t. out of plane manoeuvre.
γ_{Ctr_Lim}: Yaw deviation limit driven by satellite's control
γ_{errors}: Yaw pointing errors intrinsic to the satellite's units and configuration.
$\delta\gamma_{cal}$: Maximum gyroscope drift value.
$\vec{q}_{Gyro,Sat}$: Rotation vector from the gyroscope axis into satellite body frame yaw axis.
t_{prep_boost}: Time while the gyroscope is in charge of attitude control until the end of the boost phase.
t_{post}: Time between end of boost until sun sensor back on attitude control.
σ: Rotation of solar array drive mechanism
$\vec{u}_{sun}$: Sun vector in body frame
$\vec{u}_{sadm}$: Vector perp. to the solar array plane in body frame
E_{disch}: Discharge energy of the battery
P_{SC}: Power generated by the spacecraft
P_{loss}: Power losses of the satellite
P_{SA}: Power generated by the solar arrays
P_{SA0}: Maximum power generated by solar array with perpendicular sun incidence.
$P_{ch,max}$: Maximum charge power.
k: Optimizer coefficient for the sigmoid function
α: Sun elevation over the solar arrays

Acronyms/Abbreviations

Argument of Latitude (AoL)
Argument of Perigee (AoP)
Attitude and Orbit Control Subsystem (AOCS)
Avionic Software (ASW)
Catalyst Bed (Catbed)
Depth of Discharge (DoD)
End of Life (EOL)
Field of View (FoV)
Galileo Control Centre (GCC)
Galileo Control Segment (GCS)
Galileo Sensor Stations (GSS)
Galileo Uplink Stations (ULS)
Global Navigation Satellite System (GNSS)
Incidence Angle Modifier (IAM)
In-Orbit Validation (IOV)
Inter-Agency Space Debris Coordination committee (IADC)
Latching Current Limiter (LCL)
Low Earth Orbit (LEO)
Medium Earth Orbit (MEO)
Packet Utilization Standard (PUS)
Reaction Wheel (RW)
Right Ascension of Ascending Node (RAAN)
Solar Array (SA)
Solar Array Drive Mechanism (SADM)
Telecommand (TC)
Telemetry (TM)
Thermal Control Subsystem (TCS)
Tracking and Telemetry Control Facility (TTCF)
United Nations Office for Outer Space Affairs (UNOOSA)

1. Introduction

Galileo is the European GNNS constellation. It is a MEO constellation defined as a 24/3/1 Walker. The current 30 in-orbit satellites are distributed in 3 planes separated by 120 degrees. Within each plane, two out of ten satellites are placed in auxiliary slots, which are not part of the baseline constellation used to deliver the service. In this configuration, Galileo offers eight high-performance services all around the globe [1].

The in-orbit spacecrafts are supported by a worldwide ground segment. comprised of several GSS and ULS, and with 7 TTCFs specifically dedicated to the operations. All the operations of Galileo are carried out from the two GCCs, one located in Oberpfaffenhofen (Germany) and the other one in Fucino (Italy), working in a redundant configuration, ensuring a fully continuous monitoring and control.

The lifetime of the Galileo satellites is estimated in 12 years, which for some of the spacecraft has already been surpassed. After more than 12 years of operations some satellites have entered the not-in-service list [2], from which GSAT0104 has been selected as the first satellite to be decommissioned from the constellation.

A proper decommissioning of satellites has become one hot topic nowadays in the space industry due to the growing risk that space debris poses for space operations. It is because of this that some guidelines have been put in place to ensure that we work together for a safe and clean environment, so no risk affects the future of space activities.

Although the focus is now more into LEO orbits, the decommissioning of satellites in MEO orbits is equally important. This orbital region houses several very important constellations around the world, the GNSS. For the safety of Galileo and every other GNSS constellation, Europe has taken very seriously the work on the decommissioning activities.

In this paper, the decommissioning activities for the first Galileo satellite that undergoes such phase are discussed. The key aim of the paper is to outline these activities from a technical and operational point of view. However, to understand the context behind some of the decisions made, it is important to first provide a brief planning overview, taking into account the available resources while maintaining the nominal constellation to ensure uninterrupted service.

Then, in the next section, these same topics, along some new ones, are discussed and further expanded providing a technical view into the activities. From orbital constraints to specific characteristics of the spacecraft design, most of the topics that have played a significant role in the design of the activities are explained.

Finally, some in-house resources are outlined. These applications, have played an important role in easing the planning of the activities, highly mitigating the impact of several constraints that could have made the activity last much longer.

The whole passivation activity is very complex, especially for a large project like Galileo. The intention of this paper is not to cover the whole activity throughout the whole process. Instead, it aims to serve as references point for future operators on how to deal with some of the challenges encountered during similar activities.

2. Strategy

2.1 Requirements and Constraints

As of now, no specific regulation regarding the debris mitigation policy for MEO orbits has been implemented. However, in accordance with the guidelines established by IADC [3] and by UNOOSA [4], once a Galileo satellite reaches the end of life it must be decommissioned in a safe way, minimizing the collision risk with other active satellites.

In particular, the following considerations were applied to Galileo IOV satellites decommissioning strategy:

- The satellite graveyard orbit should be located above the nominal altitude, being the acceptable range +300 km to +1000 km in semimajor axis.
- The decommissioned satellite should not cross any GNSS constellation orbit for at least 100 years.
- The propellant should be depleted to avoid explosions in a safe and controlled way, as much as the architecture of the satellite allows.
- Satellite's internal energy should be minimized as much as possible to avoid undesired incidents once the satellite has been passivated and is not controlled anymore.
- The satellite should not be commandable once passivation has been completed.

Moreover, operational constraints must be considered, being one of the most impacting one the definition of the decommissioning windows. Due to safety reasons, to perform the decommissioning activities, these ones should be carried out outside Earth eclipse seasons. These eclipse seasons depend on the altitude of the orbit, having a slight difference between the nominal orbit and the disposal orbit as can be appreciated in Fig. 1. Furthermore, the eclipse seasons are plane dependent, consequently, decommissioning windows differ for each orbital plane.

Last but not least, Fig. 1 take into account an additional constraint due to the gyroscope calibration activity, which is needed during orbit raising and tank depletion campaigns. This constraint will be further analyzed in section 4.1. This limitation has been mitigated and can be considered not applicable anymore, nevertheless, it is also represented in Fig. 1 for a better overview about the possible constraints applicable to these satellites.

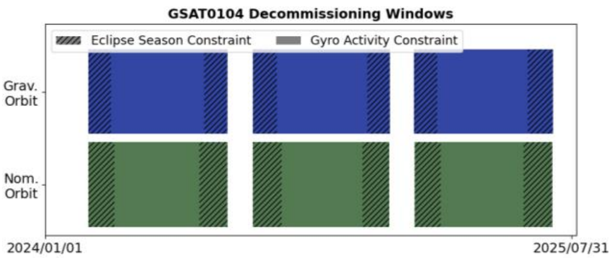

Fig. 1. GSAT0104 Decommissioning Windows

2.2 Strategy Description

Firstly, the satellite needs to be taken away from the nominal orbit, having two possibilities, to first reposition the satellite into a spare slot, or directly to start with the orbit raising phase and take the satellite to the graveyard orbit as soon as possible.

The orbit raising phase consists of a sequence of manoeuvres executed close in time, with the main objective of achieving the target semimajor axis, which considering the nominal orbit altitude, is contained in the range [29900, 30600] km. Furthermore, the eccentricity is also controlled to keep a quasi-circular orbit, just as per nominal orbit. In addition to these, the rest of the orbital parameters must be also controlled, specially the AoP/AoL, in order to ensure the correct orbit stability.

Once the previous phase is completed, the tank needs to be depleted as much as possible, undergoing a tank depletion campaign. This one consists of several out of plane manoeuvres to try to consume the maximum amount of propellant from the tank. A venting strategy is not considered as the satellite design did not account for it. It is important to mention that due to the uncertainty of the propellant measurements, it is impossible to know in which manoeuvre the tank will be depleted.

After the tank has been confirmed depleted, the next step is to passivate the remaining subsystem of the spacecraft. This is a complex activity that requires of several team working in coordination. Whereas for the planning, several requirements and constraints are considered to ensure the safety of the operations.

Two important ones are the timing of the operations and the elapsed time until passivation is achieved. From the moment the tank is depleted, there is a time limit linked to the RWs' saturation levels. Adding this to the uncertainty of the tank depletion, makes clear that a dynamic planning approach is recommendable. Following this idea, all the manoeuvre contacts are planned in a way that they can be easily repurposed to hold the passivation activities. On top of this a specialised on-call team is ready to support the activities at any time.

Between the last manoeuvre and the passivation activities is necessary to fit some additional operations to achieve the desired starting point for the final passivation. These are the transition to sun-pointing, agreed to be performed as soon as possible after tank is depleted, and starting the corruption of the software, as this is a lengthy activity, then the duration of the final contact can be reduced.

The remaining activities can be fit in a single shift were every operations team, is involved. The main requirement for this phase is to provide redundant TTCF coverage. This allows for swift transition and reconnection to the spacecraft if any ground issue causes shortages of the TM/TC link. In this way, the shortage of monitoring/commandability, if any, with the spacecraft will be minimum considering the criticality of the passivation activities.

For further details, from a technical point of view, on the passivation activities refer to the section 3.2.

After the Galileo satellite has been confirmed decommissioned, then it is time to manage the presence of the satellite on the different ground facilities. Every facility will have to follow different operational procedures according to the nature of the element (type of data and how it is stored).

The teams in charge of passivating the different elements of GCS will take care of removing every reference to the satellite from the automated processes. This will only affect the future generation of products, while all the historical data generated for this spacecraft, such as TM/TC data and ground data, will be stored for future references and analysis.

3. Preparation and execution

3.1 Manoeuvre Phases

For the manoeuvre phases, the general operational, mission and subsystems constraints during manoeuvres were considered with extreme caution, since these were the longest manoeuvres in the history of the constellation.

3.1.2 Orbit Raising

In order to maximize the increase of the semimajor axis, in-plane manoeuvres are considered, while Hohmann pairs manoeuvre-type are considered to maximize the efficiency of them. The necessary boost, translated to Δv assumes a perfect in plane, tangent to the initial orbit impulse. Even if the satellite capability could perform such a manoeuvre in one single pair, multiple pairs approach was implemented to distribute in a better way the total thrust, and increase the flexibility due to unpredicted phenomena, such as: thrusters over/miss performance, accuracy of the units, disturbances, attitude control deviations…

At this moment, it is plausible to think that out of plane components could already be introduced during the orbit raising phase, to reach the graveyard altitude with an almost depleted tank. However, this would increase the operational complexity, and second but most important, it is limited by a minimum propellant that needs to be left while staying on the disposal orbit. This propellant should be enough to cope with on board contingencies for a prolonged period. This period depends on how much time would the satellite stay in the raised orbit, something that is subjected to program decisions and deconflicting with other major activities of the constellation.

With all this in mind, it is determined to execute 3 pairs of Hohmann manoeuvres. Certainly, in space operations, there is always a need to account for contingencies or unexpected behaviours, particularly during critical operations, therefore, additional backup manoeuvres are initially defined after the nominal ones.

On top of the previously defined manoeuvres, an initial smaller manoeuvre is planned, due to two very particular reasons:

1. To be used as a dress rehearsal for the thruster's performance after a long period of time of thruster inactivity, and even longer time without thrusting for long times.
2. To decrease the collision risk with the rest of the Galileo satellites placed on the same orbital plane.

To fully grasp this second reason, it is essential to understand that a satellite is part of a wider constellation, where other satellites are located in the same orbital plane but with a phase difference of 45 degrees. As a consequence, when executing a manoeuvre to increase the semimajor axis, hence increasing the period of the transfer orbit, the satellite describing this last orbit would lag behind in phase due to the drift between them. This means that there is a time when the satellites could collide, which is dependent on the final altitude, thus, on the first Hohmann manoeuvre impulse.

Fig. 2 represents the time before collision risk with other satellites of the same orbital plane, being the grey line, the impulse selected to ensure that enough time is available to execute the subsequent Hohmann manoeuvre and avoid the collision risk. Each of the data series represent the nominal slots on the same orbital plane. Notwithstanding the big times appreciated in the graph, the selection of an appropriate delta V considers potential failures and recovery times between the first and the subsequent manoeuvre. Still, the rehearsal manoeuvre could have lasted longer, however, there was no benefit on that, as the total number of manoeuvres and risks would not have been reduced.

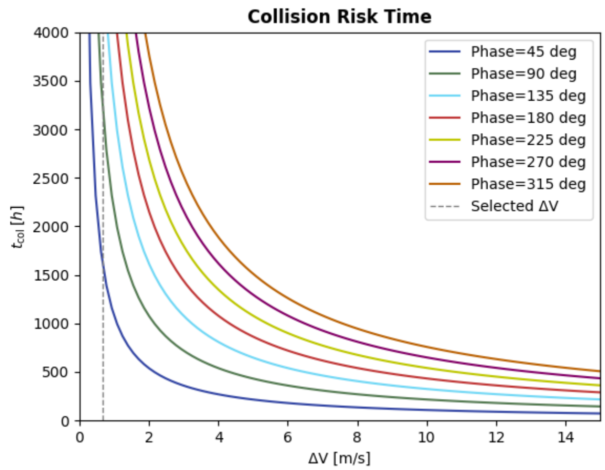

Fig. 2. Collision risk time after first manoeuvre

In summary, for GSAT0104, 7 in plane manoeuvres are executed while having planned additional backup manoeuvres which are not considered necessary to be used. The total impulse needed to acquire the target orbit is distributed among the manoeuvres, where deviations are corrected in the subsequent ones. Finally, a significant amount of propellant is left in the propulsion tank, to guarantee the safety of the satellite, as it is decided to be kept in the graveyard orbit for around 1 year.

3.1.2 Tank Depletion

Once the satellite is stable in the disposal orbit, the remaining propellant left on the tank is needed to be expelled. Due to the amount of propellant left in the tank and the decrease in efficiency in the propulsion subsystem when the tank pressure decreases because of fuel consumption, just as for all the monopropellant propulsion subsystem, a significant number of manoeuvres are needed. After these ones the propellant is expected to be expelled, only keeping some residuals that cannot be removed in the satellite's tank and propulsion subsystem.

As already introduced earlier, pure out of plane manoeuvres are executed, where the inclination and the RAAN are modified, simplified in Eq. 1 & Eq. 2, which are orbital parameters not considered as critical for this phase. One could think to center the applied Δv to the optimal point of the orbit so that the orbital plane would not change but only the inclination would be impacted. However, in practice, this is impossible due to the unpredictability of the thruster pulses during long manoeuvres used for attitude control, and besides, as already said the change of these parameters are not considered a problem. For these reasons, it is decided to give priority to the rest of operational and orbital constraints. Nevertheless, the RAAN can always be corrected if required by the subsequent manoeuvre, although there is no need to keep it as the beginning of the campaign.

$$\Delta v = 2v \sin\frac{\theta}{2} \tag{1}$$

$$\cos\theta = \cos i_1 \cos i_2 + \sin i_1 \sin i_2 \cos\Delta\Omega \tag{2}$$

On the other hand, the impact of the deviations from the pure out of plane manoeuvres cannot be disregarded, as always in plane components are introduced in these ones. The in-plane components play an important role in terms of orbit stability, which need to be corrected from one manoeuvre to another to ensure the fulfillment of the 100 years without crossing the Galileo nominal orbit.

Another point to consider is that as there is no limitation in the final inclination, manoeuvres could always be executed in the same direction, maximizing the sun presence on the sun sensor and minimizing the gyroscope usage. This cannot always be guaranteed due to the need to apply in-plane corrections and to the limitation on the FoV of the Sun sensors, which is smaller than the angle from the sun vector at the beginning and end of the decommissioning windows.

Last but not least, the performance of the propulsion subsystem for low pressures in the tank, is not well known in the industry, which implies big uncertainties specially in the planned last manoeuvres. Consequently, operational teams are prepared to accommodate additional flexibility at the conclusion of this phase. The maneuver strategy has been designed to incorporate a substantial degree of flexibility, ensuring that, in the event of a subsystem malfunction, the orbit will not be significantly affected.

The tank will be considered depleted when the spacecraft operations experts and space segment team in collaboration with the satellite manufacturer, observe anomalous behaviors in the telemetry received from the spacecraft in terms of propulsion subsystem, TCS and AOCS.

3.2 Final passivation

With the confirmation of a depleted tank, as already mentioned, a countdown for the saturation of the RWs saturation starts. While the spacecraft is in an earth-pointing attitude, the magneto-torquers can cope with the dumping of the RW momentum. However, when the spacecraft would transition to a sun pointing attitude, there is no dumping mechanism anymore because the thrusters are gone. The reasons to transition to sun pointing are two. To continue the passivation activities, as they are performed in sun pointing, or due to some autonomous anomaly reaction.

For this reason, a safety threshold of one week is put in place, in accordance with the manufacturer, to ensure the RWs will not reach values close to saturation at any point for the rest of the activities.

Besides the RW saturation consideration. The final passivation activities bring their own risks and requirements. The analysis and evaluation of the activity, considers the following topics to be the main drivers for the planification of the remaining activities.

- Loss of thruster capabilities
- Battery passivation
- ASW corruption

Another factor influences the design of the activities is the possible reboot of the spacecraft after the passivation activities are concluded. Due to the obvious loss of attitude of the spacecraft and the fact that solar panels cannot be disconnected from the bus, this will cause random episodes where the satellite will be powered and could reboot itself again. The activities to safely account for these factors are explained here below.

During these last steps of the activity the satellite is monitored continuously, for this reason it is decided to deactivate every on-board monitoring and reaction, PUS 12 and 19 services, to avoid any unexpected transition. It is worth to mention that only a couple of monitoring would be applicable for the active configuration and attitude at this phase, however making sure that the satellite cannot transition or reconfigure itself just adds an additional layer of safety to the activities. The trained teams on-console are the ones in charge of monitor and reacting if any anomalous behaviour appears.

Then, after the satellite is placed in a stable sun pointing attitude the battery is disconnected from the main bus. An important point to consider here was the fact that the battery was not designed to be operated in this way. However, after several collaboration between operational teams and satellite manufacturer, the contingency procedure to bypass one cell in case of failure can be repurposed to completely cut out the battery from the bus, avoiding future issues of battery overcharge. This activity has been simulated multiple times, in nominal and contingency scenarios, with the support of the manufacturer, until it has been clear that it can be carried out in a safe way.

From this point onwards, the satellite functions exclusively on solar array power. Battery passivation and ASW corruption are both critical topics of the activity. The order of these two activities is an important decision due to their criticality. The final plan is to alternate both activities. First the redundant banks of the software are corrupted in advance of the last contact. Then during the activities, the battery passivation is done first and then the corruption of the remaining banks. The rationale behind it is that the risk of losing power due to attitude issues was deemed lesser than an undesired reboot into a corrupted SW. Simultaneously, this strategy reduces the remaining operations with the bypass battery.

The activity to corrupt the ASW has the goal to prevent that the satellite can reboot itself during the solar episodes mentioned previously. However, though the satellite will not be able to turn on from a software point of view, the bus will still be powered, and the hardware of some units will turn on as an effect of this.

As a response, to avoid the reactivation of some critical units, all the LCL connections will be disconnected. Some units that deserve a mention are the RWs, because if not properly disconnected, it could cause an uncontrolled increase of the speed in the future that could end up in a satellite break-up. To avoid this, the RWs are commanded to torque 0 and subsequently the LCL is disconnected. Other important unit to be properly switched off are the propulsion subsystem, thrusters and catbed heaters, and the TCS control units to avoid an overheating of units in the future.

This will conclude the passivation activities; the last time tag sequence will take care of switching off the transmitter and forcing a reboot into the corrupted ASW so the last activities will be performed out of visibility. To ensure a proper SW passivation several in-flight test will be performed to try to recover the satellite. Then in failing to do so, it will confirm a successful passivation of first Galileo satellite.

4. Optimization of IOV Satellite Decommissioning

4.1 Gyroscope Calibration

Due to the geometry of the manoeuvres needed to be executed and the architecture of the satellite, the sun sensors are not always available during the manoeuvre execution. To keep an accurate attitude control, when the sun sensors are not able to provide such measurements, the integrative unit onboard, the gyroscope, is needed to be on the satellite's attitude control loop.

The type of Galileo satellite examined in this paper has a 1-axis rate integrating gyroscope mounted inside one of the panels of the satellite's body. An additional gyroscope is available with the same characteristics to ensure the cold redundancy of them. Due to the characteristics of this integrative unit, in order to maintain an acceptable accuracy, prior to the usage of this unit for an extended period of time, a calibration using the data coming from sun sensors as reference is needed.

The limitations of the 1-axis rate gyroscopes are well known, particularly the loss of accuracy outside the principal rotation axis as well as the inability to detect the perpendicular rotation's components. However, due to the satellite's dynamics during its calibration as well as during the manoeuvre execution, the accuracy loss is considered to be of minimal significance. Furthermore, on the one hand, this gyro was selected as the performance is adequate for this mission, while on the other hand, safety margins are always introduced when conducting space operations.

The objective of the calibration activity is to obtain a maximum gyroscope drift value small enough to ensure that the yaw angle while having the gyroscope controlling the yaw will be kept inside some acceptable margins. To perform such a calibration the gyroscope on board algorithm requires to keep the sun inside the FoV of the sun sensor, but not only this, for a better accuracy the sun must be always contained in a narrowed cone that can be seen as a narrowed FoV of the sensor. Furthermore, the convergence of the algorithm requires some time to reach the value, having to perform several iterations in some cases as can be seen in Table 1. The definition of the target maximum drift value will be analysed in further subchapter.

Table 1. Gyroscope Drift Estimation Convergence Times

Iter n°	Drift estimation [min]	Convergence	Target Value acquired	Time needed [min]
1	28	No	No	70
2	38	Yes	Yes	
1	36	Yes	No	90
2	37	Yes	Yes	
1	31	No	No	85
2	48	Yes	Yes	
1	30	Yes	Yes	N/A

Estimating the time needed for the activity of the calibration is something that cannot be predicted, therefore, an empirical approach was decided to be followed. As presented in Table 1 the time needed to acquire a convergence with the correct target value differs significantly. For this reason, as it is a common practice in space operations, the most conservative one is selected as a constraint, which is the worst-case scenario of 90 minutes between the start of the 1st iteration and the end of the 2nd. This means that the sun is needed in the sun sensor's calibration cone for at least 90 minutes, something that depends on the satellite attitude and the sun elevation over the orbital plane.

4.1.1 Target Gyroscope's Max Drift

As introduced earlier, a drift target value needs to be defined. Initially, the target value had been defined as a very conservative value, which although being a safe approach, it also introduces an additional degree of difficulty for a fast convergence during the calibration activity. Indeed, two different values were in place, one more conservative for longer periods of attitude based on gyroscope control, another one less conservative for shorter periods.

For this reason, a new model was implemented for the tank depletion phase, on the one hand reducing the target value and hence reducing the convergence computational effort, on the other hand guaranteeing the safe operations of the satellites.

During the tank depletion phase, as it was already explained, pure out of plane manoeuvres are planned to be executed, however, the in-plane component is the one to be considered as it is unavoidable and would be the one impacting on the eccentricity, consequently, the one driving the crossing of a nominal Galileo orbit. For this reason, the maximum drift for each manoeuvre varies by playing.

$$\Delta v_{max,inplane} = \sqrt{2\frac{\mu}{r_p}\left(1-\frac{1-e}{2}\right)} - \sqrt{\frac{\mu}{r_p}} \qquad (3)$$

$$\gamma_{max} = \sin^{-1}\left(\frac{\Delta v_{max,inplane}}{\Delta v}\right) \qquad (4)$$

$$\delta\gamma_{cal} \cdot \left(\vec{q}_{Gyro,Sat} \cdot \vec{k}\right) \cdot t_{prep\_boost} = \gamma_{max} - \sum \gamma_{errors} \qquad (5)$$

Additionally, an already defined maximum yaw deviation constraint is considered in order to ensure a proper yaw control, not only during the boost but until the attitude control can be resumed again with the sun sensor. This time after boost, t_{post}, has been obtained from operational experience.

$$\delta\gamma_{cal} \cdot \left(\vec{q}_{Gyro,Sat} \cdot \vec{k}\right)\left(t_{prep\_boost} + t_{post}\right) \leq \gamma_{Ctr\_Lim} \qquad (6)$$

In case the resulting maximum gyroscope drift for a given manoeuvre violates the yaw deviation constraint as per Eq. 6, then the upper limit is defined by this last one instead of the obtained value (represented as dashed lines in Fig. 3).

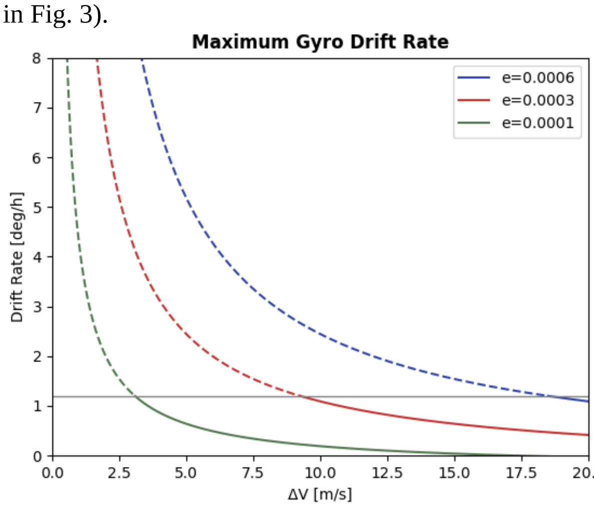

Fig. 3. Maximum Gyroscope Drift Rate

4.1.2 Gyroscope Calibration Split

During nominal orbits, Galileo satellites rotate around the z axis following a yaw steering law, a complete description of this law is described in [5]. One of the many reasons is to maximize the sun visibility in the sun sensor. As a direct consequence of this motion, together with the mentioned requirement of having the sun inside a cone in the sun sensor to calibrate the unit, the gyroscope can only be calibrated in specific windows.

Fig. 4 shows the evolution of the sun inside the FoV of the sun sensor, during one entire orbit, considering the origin at the projection of the Sun vector on the orbital plane. In addition, 2 different scenarios are presented: low and a high sun elevation angle over the orbital plane. Analysing the impact of this is extremely important, as the maximum sun elevation depends on the orbital plane, which could lead to short gyroscope calibration windows where it is not enough to ensure a convergence which would block the manoeuvres execution.

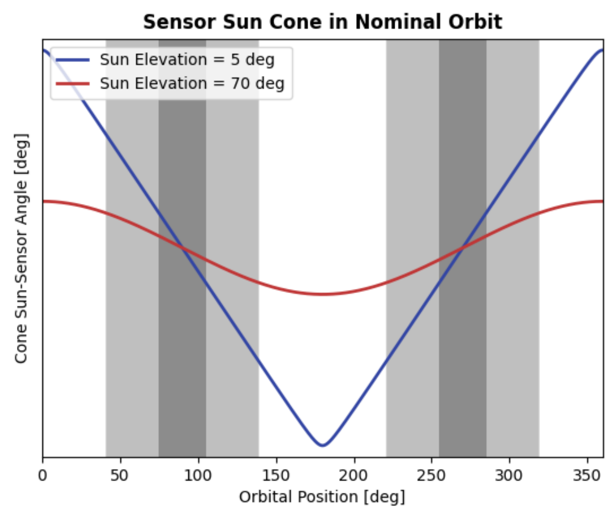

Fig. 4. Sensor's sun cone in nominal orbit

Fig. 4 offers insight into the variability of the windows according to the different sun elevation angles, highlighting in grey the orbital positions where the calibration can be performed for a nominal orbit of a Galileo satellite. It can be clearly seen that for high sun elevations the sun vector falls more time into the sun sensor cone (light grey), while for low sun elevations as a higher rotation is needed per orbit, the window is significantly decreased (dark grey). For the cases that were illustrated in Fig. 4, there are 2 windows per orbit to perform the calibration activity with a duration of 70 min and 231 min. This means that the gyroscope calibration activity, which is defined as a 90 min duration as seen in Table 1, cannot be performed in a single slot.

The duration of the windows is directly impacted by several variables, being the most important ones: the altitude of the orbit and the vertex angle needed by the sun sensor to perform the calibration of the unit. Fig. 5 demonstrates the evolution of the windows duration comparing the nominal altitude with the maximum graveyard altitude and an increase of the sun sensor cone vertex angle of 5 degrees. At this point, it is reasonable to assume that the cone vertex angle could be increased to the sun sensor FoV, however, this would imply a loss in the accuracy, reason why the angle is kept low as specified by the unit's characteristics.

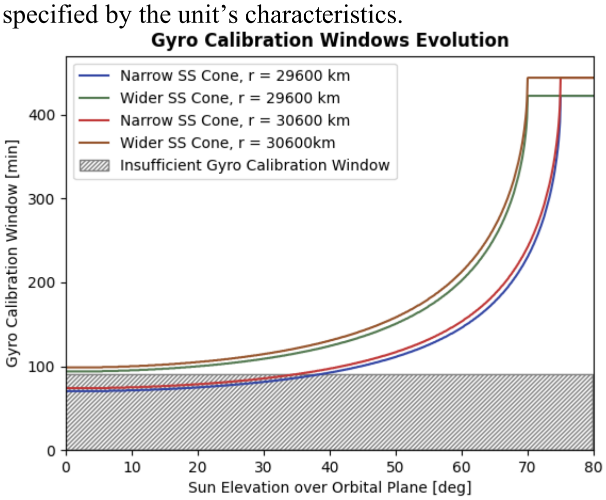

Fig. 5. Gyroscope calibration windows evolution w.r.t. sun elevation and semimajor axis

Although the increase in the altitude of the orbit improves the calibration window in terms of time, there are still minimum values of the sun elevation over the orbital plane needed to ensure the successful operation; around 38 deg in the nominal orbit, around 35 deg in the graveyard orbit.

All this means that the windows for decommissioning of the Galileo IOV satellites would be limited by this sun elevations. Even though this could be considered acceptable, this has 2 important implications:

- IOV satellites contained in orbital planes where the maximum sun elevation over the plane is low could hardly go through safe orbit raising and tank depletion campaigns.
- The decommissioning widow is narrowed down as shown in Fig. 1 while having to perform a significant number of manoeuvres, which could lead to a lack of time to complete the orbit raising, tank depletion and passivation in the same window, hence, increasing the operational risks.

Due to the previous conclusions, an improvement was implemented in collaboration with the satellite manufacturer. As presented in Table 1 in the worst-case scenario a couple of iterations are required for the convergence to the target drift rate. The solution in place is to split the activity of the gyroscope calibration into 2 shorter activities where only 1 iteration is performed, taking advantage of the fact that there are 2 windows per orbit. With this approach, the gyroscope calibration window duration, which originally was of 90 min, is reduced by a factor of 2, although there is an increase on the effort by the operators as the activity is prepared and executed twice. This is considered a suitable solution, as it mitigates all the problems previously mentioned, any IOV satellites from all the orbital planes can go through safe orbit raising and tank depletion phases, and the decommissioning windows are not limited by this activity. Lastly, although the scope of this paper is IOV decommissioning, this improvement can be applied to any type of IOV manoeuvre campaign where the gyroscope is required due to the loss of the Sun from the sun sensor's field of view.

4.2 Battery discharge prediction

4.2.1 Mathematical model

In Galileo, the manoeuvres are divided in three main phases. Two slews, forward and backwards, where the satellite rotates around the yaw axis to achieve the target boost direction or return to the yaw steering law, respectively. The earth-pointing is always maintained to measure the roll and pitch. Then the boost itself, which also includes an initial subphase to dump any residual momentum, if needed, to achieve more precise manoeuvres.

The idea behind this model is to be able to calculate the maximum discharge of the battery during a manoeuvre. The implementation of this model allows performing longer manoeuvres, maintaining the safety margins of the battery. The model uses the battery and spacecraft characteristics to simulate the internal energy exchange, and then the sun position vector in body frame to calculate the received solar power.

The following assumptions were considered:

- Battery fully charged at the beginning of the manoeuvre.
- No eclipse foreseen after the manoeuvre.
- Battery will be discharged and charged during the whole manoeuvre, the 2 slew phases and the boost phase.
- Battery EOL corrections considered.

The energy discharged, Eq. 7, from the battery is obtained by integrating the difference between the satellite consumption and the power generated by the solar array

$$E_{disch} = \int_0^t (P_{SC} + P_{loss} - P_{SA})\,dt \tag{7}$$

The power generated by the solar arrays depends on the sun incidence angle over the solar panels. To calculate this, from the sun vector only, the model calculates the optimal SADM angle, which provides maximum power, at each instant of the orbit with Eq. 8 Then, it rotates the SADM, considering the maximum SADM rotation as shown in Fig. 6. With the value of the SADM angle calculated, it is trivial to find incidence angle over the solar panels. As it is shown in Fig.

$$\frac{d}{d\sigma}\vec{u}_{sun} \cdot \vec{u}_{sadm}(\sigma) = 0 \quad (8)$$

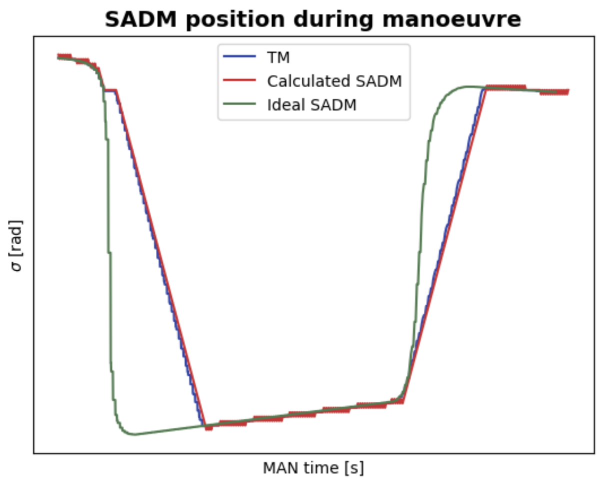

Fig. 6. SADM position during manoeuvre

On top of this, the performance of the generated solar power has an empirical term, the IAM, which must be finetuned for each solar array. There are several complex models that used advanced optics or complicated vacuum testing to obtain the IAM curve. However, because the scope of this model is to provide an approximation of the discharge of the battery, those option have been discarded. Instead, a mathematical function is implemented and tweaked to fit the TM data from previous manoeuvres from the same satellite. Then, after several iterations, it was decided that a modified sigmoid function is the better fit. As shown below in Eq. 9

$$P_{SA} = P_{SA0}\cos\alpha(t)\left[2\left(\frac{1}{1+e^{-k\cos(\alpha(t))}}\right) - 1\right] \quad (9)$$

Where: $\alpha(t) = 0 \;\forall\; \alpha(t) < 0$

Alternatively, the phases of the manoeuvre when the spacecraft receives enough solar power to charge the battery is defined as per Eq. 10 below. The charge power is limited by a maximum value that depends on the specific architecture of the PCDU on board.

$$P_{ch,max} > P_{SA0} - (P_{SC} + P_{loss}) > 0 \quad (10)$$

4.2.2 Results

In this subsection the results of the model are presented against the real TM.
The spacecraft parameter affected directly by the incidence angle is the SA power as already mentioned. Fig. 7 shows the SA current profile during a nominal manoeuvre.

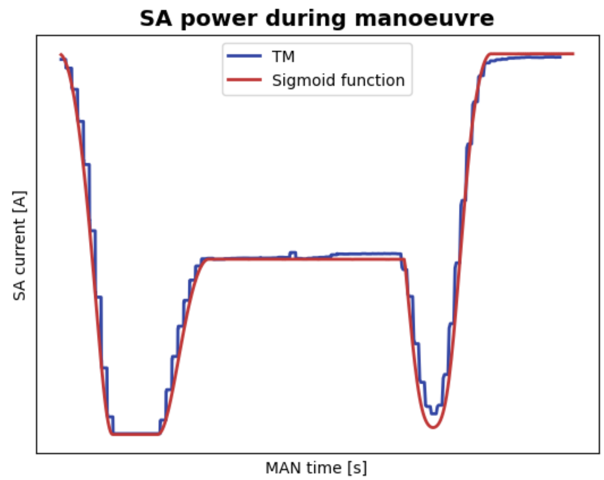

Fig. 7. Solar Array power during manoeuvre

Aside from getting a good fit to the TM, another more relevant criterion has been to always have a conservative fit. What this means is that the calculated SA current must always be below the real TM to always have a higher battery discharge. In Fig. 7 above, the data corresponds to the manoeuvre with the best fit. Some other manoeuvres have worse fits, but they are still in the acceptable range. The idea is to keep validating, and updating if needed, the parameters of the sigmoid function to fit the upcoming manoeuvres from the depletion campaign.

While the SA power is a better way to finetune the model, it is the DOD the one being monitored on board. Below in Fig. 8 the DOD during the same manoeuvre is presented.

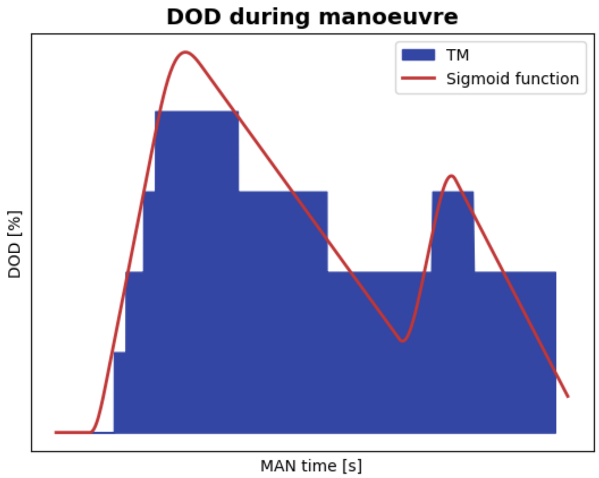

Fig. 8. Depth of Discharge during manoeuvre

This model has been designed in the framework of the manoeuvres analysis, however with some modifications this tool could be used in several scenarios where the battery performance is relevant for the activity. If the sun vector and the specific satellite consumption are known, the model can calculate the battery discharge for the whole duration of the activity.

5. Conclusions

The decommissioning planning for the first Galileo satellite has been carried out with safety as the highest priority, making sure that the IADC guidelines are met so the integrity of the MEO environment is maintained. The process has involved lengthy technical assessments and discussions to identify and address key challenges, including operational constraints and space debris mitigation strategies.

Although the activity has not yet been fully executed, at the time of this publication, the planning approach outlined aims to provide operational references that can be adapted to similar decommissioning efforts in the future. By documenting the technical considerations and decisions, it is desired that other operators could benefits from these practices during their operations. The insight into more than 12 years of experience operating a large constellation, will prove beneficial for the whole space community.

The execution phase, once initiated, will offer an opportunity to validate the proposed strategy. Any deviations from the expected outcomes will provide important lessons learned, contributing to the continuous improvement of decommissioning procedures for the remaining satellites of the constellation.

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