ERS-2 platform disposal operations

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ERS marked in many ways Earth observation at the European Space Agency. When ERS-1 was launched in 1991, it was one of the most sophisticated spacecraft ever built in Europe. It accomplished, together with its successor ERS-2, a major scientific mission covering a period of more than 20 years. During this extended period, the mission had to respond to new requirements while the actual capabilities of the ERS-2 spacecraft were declining. At the end of the mission, the mitigation of space debris had become a priority and posed a final challenge. Disposal operations were prepared to lower the orbit and to "passivate" the ageing spacecraft. The development and the execution of these operations are presented with a particular emphasis on the influence of the platform design and of its status at the time of execution. In order to give an overview of the platform status at the beginning of the disposal phase, its design is presented together with a short in-flight history of major failures and on-board software modifications. The ability of the spacecraft to comply with space debris mitigation recommendations is then analyzed before the development and the execution of the disposal operations is described in detail.

I. Introduction

THE European Remote Sensing (ERS) system was developed by the European Space Agency (ESA) and European industry during the 1980s with the objective of "measuring the Earth's atmosphere and surface properties, both with a high degree of accuracy and on a global scale".[1] The system consisted of a ground segment operated by ESA and two recurrent spacecraft -"ERS-1" and "ERS-2"- developed by an industrial consortium led by Dornier GmbH. Each spacecraft was 3-axis stabilized, weighted about 2.5 tons and featured 5 main payloads that required about 1 kW of electrical power in full operations.

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The ERS spacecraft were designed to fly in Sun-synchronous orbits at a mean altitude of approximately 780 km, an inclination of about 98.5° and a local time of the descending node of ca. 10h30; these orbits have a period of ca. 100 minutes. Repeating ground tracks were required for the mission and several operational orbits were defined with a ground tracks repetition period of 3 days, 35 days and 168 days. Whereas ERS-1 had been operated in several of these orbits; ERS-2 was maintained in the “35-days orbit” until the last year of operations.[4]

The ERS ground segment was spread over several ESA sites in Europe. Mission management, mission planning and payload data processing, archiving and dissemination were conducted from ESRIN, in Frascati, Italy. Flight Operations were conducted from ESOC in Darmstadt, Germany. Assessment of performances and development of in-flight evolutions were conducted from ESTEC in Noordwijk, Netherlands with the support of EADS-Astrium in Toulouse, France (formerly Matra-Espace).

Ariane 4 launchers inserted ERS-1 in orbit on 17 July 1991 and ERS-2 on 21 April 1995.

Figure 1. The ERS-2 spacecraft

II. Spacecraft Design

When the phase C/D was initiated in 1984, ERS-1 was far more complex than any spacecraft that ESA had flown previously.[1, 3] In order to limit cost, the spacecraft re-used the “SPOT Mark 1” platform that was being developed by CNES and Matra-Espace for the first 3 satellites of the SPOT series. This choice introduced several constraints on the orbit where the spacecraft could be operated; in particular several subsystems, such as attitude sensors, thermal control and solar array rotation required Sun-synchronism. In the meantime, a number of modifications were needed to accommodate the payloads selected for this mission. The solar array power and the battery capacity were increased, new flight software was developed to enable yaw steering and the initial complement of propellant was significantly increased.

A. Attitude and Orbit Control

Attitude measurements was provided by the following set of sensors:[5]

- 6 mechanical gyro providing each integrated angular rate measurement about one axis, these units are numbered from 1 to 6 for identification
- 2 Digital Earth Sensors (DES) that measured Space to Earth and Earth to Space transitions in a permanent conical scanning to determine the direction of the Earth
- 2 Digital Sun Sensors (DSS) providing a measurement of the Sun direction once per orbit when the Sun crossed its linear field of view
- 2 Sun Acquisition Sensors (SAS) to measure the direction of the Sun as part of the safe mode acquisition.

Attitude actuation was provided by the following actuators:

- 3 internally redundant reaction wheels (RW) to generate torque about all three axes
- 2 internally redundant magnetorquers to generate torque about the roll and pitch axes

A monopropellant propulsion system (RCS) provided orbit control capabilities and another mean of attitude actuation. The propellant, hydrazine, was stored in 2 pairs of tanks pressurized with Helium and operated in blow-down mode. Sixteen thrusters

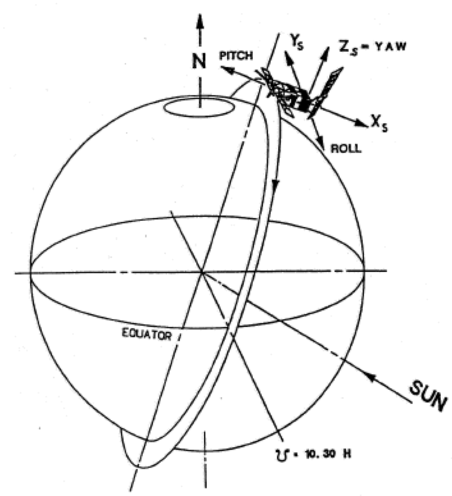

Figure 2. Spacecraft reference frames

with a catalyst bed were arranged in 2 separate branches. On each branch, 4 thrusters with a nominal force of 3.5 N could be actuated in pairs to provide torque about the roll and pitch axes and 4 thrusters with a nominal force of 15 N could be actuated in pairs to provide torque about the yaw axis or to provide thrust along or against the nominal flight direction.

In nominal modes, the central flight software (CFS) ran on the main On-Board Computer (OBC) and acted as the AOCS controller; in these modes an Earth pointing attitude was acquired or maintained and the solar array was rotated to maintain Sun pointing.

A separate control chain featuring a dedicated controller was also available to ensure spacecraft survival after a major anomaly. In this safe mode, the SAS, a single gyro and the RCS were used to acquire and maintain Sun pointing.

The safe mode implementation did not allow in-flight modifications; on the other hand, development of new versions of the CFS permitted extensive changes of the AOCS modes controlled by the OBC.

The original CFS version of both ERS satellites featured 8 AOCS modes for attitude acquisition, attitude control during payloads operations and orbit control maneuvers. In these modes, DES normally provided pointing reference in pitch and roll at 1 Hz; DSS provided a pointing reference in yaw once per orbit and a gyro triplet provided body rates measurements that were used to propagate attitude estimation at 8 Hz in all 3 axes.

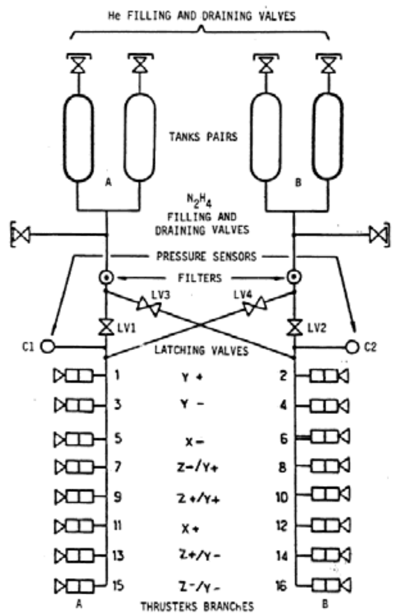

Figure 3. Diagram of ERS 2 propulsion

In the fine modes, attitude was controlled by reaction wheels and magnetorquers were used for continuous wheel de-saturation. In these modes, attitude control ensured that the instruments were pointed toward the Earth, with a yaw steering (Yaw Steering Mode) or without it (Fine Pointing Mode).

For orbit maintenance, the Fine Control Maneuver mode (FCM) was used to impart small in-plane ΔV to control the East-West drift of the satellite close to the Equator whereas the Orbit Control Maneuvers mode (OCM) was used to correct the orbital inclination. This later type of maneuver required a 90 deg slew about the yaw axis, so that the 15N thrusters could be used to impart a ΔV perpendicular to the orbit plane. OCM could also be used without a slew to impart a large in-plane ΔV

B. Data handling and transmission

TT&C operations were performed via a redundant S-band transponder within the platform. In the payload module, the Instrument Data Handling and Transmission system (IDHT) featured a redundant tape recorder with a data capacity of 6.5 gigabits and a redundant dual link X-band transponder. The IDHT was in charge of transmitting in real-time the images from the Active Microwave Instrument and to record and transmit data from other payloads and from the platform. Thus, flight operators had access to the complete set of housekeeping telemetry.

C. Electrical power supply and management

The solar array consisted of two 5.8 m x 2.4 m wings, on which were mounted 22 260 solar cells providing an average power of about 2.5 kW at beginning of life. The solar array was continuously rotated so that its surface remained perpendicular to the Sun direction. Eclipses accounted for about one third of each revolution; four nickel-cadmium batteries provided satellite power during these periods allowing continual payload operations.

III. Operation history and In-Flight Modification

When it failed on 10 March 2000, ERS-1 had been operated for more than 9 years, more than 3 times its planned lifetime.[2] ESA had to rely on ERS-2 for Earth Observation until the launch of Envisat on 1 March 2002. Envisat, together with Metop-A that was launched on 19 October 2006 eventually ensured data continuity after ERS. At that point, the payloads were still in good condition, and several applications benefited from ERS-2 observations.[6] The mission was thus further extended to mid-2011 which amounted to a total of 16 years of continuous operations. This far exceeded the design lifetime of the platform and, as could be expected, some subsystems experienced degradation of performances and failures. To overcome these problems significant changes were implemented over the years both on ground and on the spacecraft.

A. Mono-gyro and gyro-less piloting software

ERS-2 was fitted with a complement of 6 gyros providing each rotation rate measurement about one axis; in the original version of the CFS, attitude control required measurement from 3 gyros for all nominal modes. This original set of AOCS modes, referred as "3-gyros piloting" was used throughout the first part of the routine operations and offered optimal pointing performances.

From 1997 onward, ERS-2 was affected by degradation of performances and failures of several gyros. This evolution was following the events of November 1996 which saw the loss of SPOT-3 precisely because of gyros failures.[12]

In order to safeguard the mission, several new AOCS modes were developed. The first set of modes, referred collectively as "mono-gyro piloting" or "1-GP" allowed to ensure attitude control and to continue science operations with a single gyro. 1-GP was activated in February 2000; it introduced additional flight constraints and pointing performances were degraded.

Development of a second set of modes, referred as "gyro-less piloting" or "0-GP" was then started but an additional gyro failure occurred before it was ready. Mono gyro piloting had been designed to use data from either gyro#5 or gyro#6; unfortunately, gyro#5 failed on 7 October 2000. It was decided to shorten the period upon which attitude control would solely rely on gyro#6 by introducing an additional mode referred as the "Extra Backup Mode". EBM was quickly developed to allow attitude control without any gyro. This mode was intended to keep the spacecraft alive and its pointing performances were not sufficient for science operations; EBM was activated in January 2001.

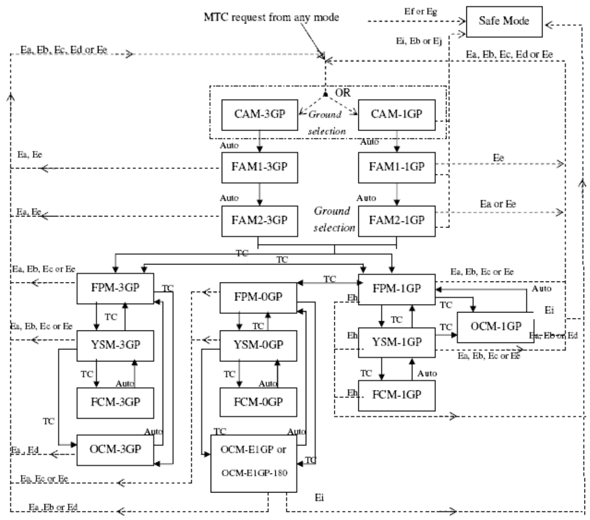

Figure 4. AOCS modes transition diagram after the introduction of 1-GP and 0-GP modes

Gyro-less piloting was finally available and could be activated in June 2001. 0-GP did not require any gyro inputs and significantly reduced exposure to gyro failures. The pointing performances in 0-GP were inferior to those in 1-GP; however, they were sufficient to resume science operations. 0-GP remained operational until the end of the mission.

B. Failure of transponders and TM coverage

The prime tape recorder failed in January 2003. The backup unit failed on June 2003. From this time onward, low-rate data could not be stored on-board anymore. Ground controllers could no longer access a global data set of housekeeping telemetry. Fortunately, coverage was not limited to the passes from the prime TT&C ground station at Kiruna. A ground station in Svalbard provided coverage during 2 out of the 3 daily revolutions without visibility from Kiruna and backup S-band ground stations were available to provide support during special operations. Moreover, a large network of ground stations scattered around the world was available to acquire real-time satellite data transmitted in X-band. Housekeeping telemetry was then extracted from the data set and forwarded to the control center at ESOC.

Thus, ground controllers had access to the TM generated in significant portions of the orbit corresponding to the passes supported by the S-band stations and some X-band stations. [6]

IV. Assessing capabilities against the latest space debris mitigation standards

During the 20 years of operations of ERS, the awareness of the threat posed by space debris considerably grew. Throughout this period several organizations initiated mitigation activities and high level discussions were held to determine how this issue was to be addressed. In 2006, ESA ratified a code of conduct with other European space agencies; this code was then tailored to the needs of ESA and translated into an internal instruction (ESA/ADMIN/IPOL(2008)2 "Requirements on Space Debris Mitigation for Agency Projects"). This instruction came into force on 1 April 2008 and was applicable to all space systems procured after this date.

Back to the 1980s, the ERS satellites had been developed without any requirements aimed at limiting the proliferation of space debris and ensuring a safe re-entry to Earth. In particular, the platforms had been designed to function solely within a range encompassing all the operational orbits. In 2003, ESA defined a set of priorities to minimize the impact of ERS-2 on the space environment after decommissioning and Astrium-France was tasked to perform the associated feasibility studies. This activity highlighted the robustness and the versatility of the ERS-2 design as it appeared that, despite the extensive time already spent in orbit and the loss of several critical units, the platform could support a significant orbit lowering to hasten re-entry. Moreover, the studies showed that sufficient fuel remained on-board to support several more years of operations before lowering had to be initiated. It was even possible to accomplish these objectives in case more gyros units were to be lost.

In 2010, as the end of the mission extension was approaching, the actual development of the disposal operations was undertaken. ERS-2 greatly benefited from the experience gained during the de-orbiting of previous SPOT satellites. CNES had commissioned Matra Marconi Space to study the feasibility of de-orbiting SPOT-1 as early as 1994.[7] After extensive studies, successful disposal of SPOT-1 and SPOT-2 were performed respectively in November 2003 and July 2009. As for previous SPOT spacecraft, ERS-2 could not support controlled re-entry and "de-orbiting" had to be performed in 2 steps. During the first one, orbit maneuvers would be executed to actively lower the orbit before commanding a "passive" configuration; the end of this first step marked the end of flight operations. During the second step, the orbit of the uncontrolled spacecraft would be left to decay through natural perturbations until atmospheric re-entry finally takes place.

The disposal operations encompassed all the activities performed during this first step; that is to say from the end of scientific operations (or "routine") until the final commanding of the spacecraft. In line with ESA/ADMIN/IPOL(2008)2, the primes objectives of these operations were:

- to lower the orbit in order to ensure re-entry within 25 years after the end of the mission
- to reduce the risks of fragmentation by minimizing the amount of energy stored in the spacecraft after the end of operations

In addition, to comply with ITU regulations, all radio emissions from the spacecraft were to cease definitely before expiration of the ERS-2 license.

The two prime objectives were by nature inseparable, propellant being both the mean to lower the orbit and the prime form of energy stored on board. However, it appeared that disposal operations could be organized in two consecutive stages. First, a "descent" stage would mainly consist of maneuvers to lower the orbits and deplete most of the remaining fuel; then a "passivation" stage would encompass all the operations to deprive the spacecraft from its remaining energy and to ensure the end of radio emissions.

V. Development and execution of descent operations

A. Initial drivers for a maneuvers strategy

ERS-2 was launched with a propellant complement of 314 kg for a dry mass of 2194 kg. When the mission was terminated on 4 July 2011, it was estimated that 160 kg remained on board. In other words, more fuel had to be consumed during disposal than throughout the 16 years of routine. The remaining propellant amounted to a change of velocity in the order of 140 m/s and to a combined burn duration of more than 8 hours. Most of these maneuvers would be performed against velocity in an environment that would become, as the altitude lowers, less and less favorable. For comparison, the largest maneuvers in routine were performed to control the inclination of the orbit; the associated velocity changes were perpendicular to the orbit plane and had a magnitude in the order of 2 m/s.

The first objective of the descent phase was to maneuver toward a deposit orbit with an acceptable lifetime. This lifetime could be minimized by lowering as much as possible the perigee, that is to say aiming at an elliptical orbit. However, considering the large amount of remaining fuel and the operational constraints from both the spacecraft and the ground segment this strategy could not be pursued. Below a certain altitude, it was dubious, that the AOCS (and especially mono-gyro and gyro-less modes) could operate and that safe mode could be recovered. In addition, this strategy required all maneuvers to be executed close to apogee, this requirement added up to existing constraints such as preventing illumination of AOCS sensors. On the other hand, circular orbits allowed optimal performances of the AOCS and the remaining fuel was sufficient to reach a deposit orbit with a lifetime well below the objective of 25 years.

For these reasons, maneuvers would be executed in pairs and a near circular orbit would be maintained throughout the descent stage.

B. Pre-selection of AOCS modes for the maneuvers

In the original CFS, orbit corrections along or against velocity could be performed in two different AOCS modes. FCM allowed corrections smaller than ca. 0.1 m/s whereas larger maneuvers had to be performed in OCM. The introduction of the mono-gyro and the gyro-less software provided additional options as each of these software featured its own version of FCM and OCM. With FCM, too many burns would have been required to lower the orbit. OCM was therefore preferred and one of its several implementations had to be selected.

The first option was to use the original 3-GP software. But as reported previously, 3 out of the 6 gyros had failed during routine. In 3-GP, all surviving units (#3, #4 and #6) would be required to run both the nominal (FPM-3GP and OCM-3GP) and the contingency (CAM-3GP, FAM1-3GP and FAM2-3GP) AOCS modes. One of these units (#3 or #4) also had to be selected to control the safe mode. In short, with the 3-GP option, a gyro malfunction could result at best to a safe mode and in some cases to a loss of spacecraft. The risks associated with this option were considered so high that it was discarded.

With the mono-gyro software, gyro#6 had to be selected to run both nominal (FPM-1GP, OCM-1GP) and contingency (CAM-1GP, FAM1-1GP, FAM2-1GP) AOCS modes. A separate gyro would be selected in safe mode.

The last option was to use the gyro-less software. As its name does not suggest, gyros would be required in all AOCS modes apart from the FPM-0GP. The gyro-less software featured the OCM-E1GP that used gyro#4 for control and the same contingency modes as the mono-gyro software. Overall, this option was the most robust in case of failure of gyros; on the other hand, OCM-1GP had better performances and was associated with less severe flight constraints than OCM-E1GP.

Due to the alignment of gyro#4, attitude estimation based solely on the output of this unit was stable when the spacecraft was slewed. Therefore OCM-E1GP had been extensively used in routine for out-of-plane maneuvers. For in-plane maneuvers however, correct performances could only be achieved under certain conditions. First of all, the duration of a burn was limited. Astrium initially recommended a maximum of 200 s; this threshold was extended to 300 s after two successful test burns performed in February 2011.[4] Furthermore, due to its limited stability, this mode was not usable when the spacecraft was submitted to excessive disturbance torques. As a consequence, Astrium recommended the use of this mode to be restricted to altitude higher than 700 km.

OCM-1GP would therefore have to be used at lower altitude. For the initial part of the descent stage however, the respective merits of OCM-E1GP and OCM-1GP had to be further examined. This issue was closely associated to the size of individual maneuvers and the plan to manage potential failure of gyro.

C. Ground constraints

The question of maneuver sizing was strongly related to ground segment constraints as the plan under preparation was radically different from what had been experienced in routine.

Over the years, ESOC had come to operate an increasing number of missions that were in turn each increasingly complex. To find the radical gains in efficiency that were required to meet this challenge, ESOC evolved into a system of systems, that is to say, teams and systems were increasingly coordinated and pooled to offer more functionalities and performances than the simple sum of the constituent flight operations segments. This development was accompanied by growing standardization and automation of common processes (see for example Ref. 8 and Ref. 9). Throughout its extended lifetime, ERS played its full part in this evolution[6] and this environment had to be considered to design disposal operations.

For instance, the time when a ground station (let alone a network of ground stations) could be devoted to the needs of single spacecraft has long since passed. At ESOC, ground station booking is performed on a weekly basis. This process consists in allocating ground station time slots to individual missions; it concerns both ground stations operated directly by ESA and external support granted according to availabilities and contractual agreements.[9]

For the disposal operations, adequate ground station support was required to gather sufficient tracking data for orbit determination and also to collect TM for system monitoring. As the spacecraft was submitted to a new type of operations and environment, close monitoring was essential. With the loss of the tape recorders, flight controller visibility was restricted to ground station passes and after the end of routine, it was no longer possible to count on the support from the X-band ground stations. The maneuvers plan was therefore devised taking into account these requirements together with realistic expectations concerning ground station availability at the time of execution.

In addition, once an agreement over ground station usage had been reached with other missions for a given week, it was essential to ensure that ERS would not have to request late revision. Station booking was based on the expected trajectory of the spacecraft for more than a week in the future. This prediction was built upon the assumption that planned maneuvers would be executed at a given performance level. Over that period, the number and size of the maneuvers had to be such that the plan would remain valid at least when maneuvers performances remained within the nominal range. Moreover, the plan had to leave some margin for contingencies such as misperformances during burns or unit failure.

The orbit determination process also had to be considered. It was critical that the trajectory prediction remained sufficiently accurate to ensure acquisition by the ground stations. Each update required an adequate set of tracking data over a period without maneuvers; it also required the availability of flight dynamics experts. Those experts being also involved in the support of other missions, the work-plan had to be devised to be compatible with their other commitments. In particular, nominal flight dynamics support had to be restricted to daytime and orbit changes had to be within the capabilities of the existing automatic processes.

After careful considerations of these constraints, it was found that existing processes could be used to support a pair of maneuvers for a total velocity change of ca. 4 m/s each day; provided that periods of 2 or 3 days free of maneuvers were put aside between series of 5 days of maneuvers. Each of these periods was named a "maneuvers block" and its two sub-phases "maneuvers execution" and "fine orbit determination". The time separation between the 2 burns of a pair was to remain minimal so that sufficient tracking data could be gathered between pairs. This approach enabled the support of all descent operations with available personal and without interfering with the other missions operated at ESOC.

Nevertheless, it required strict management of deviation from the plan during execution. In particular, a prompt response was required in case a burn could not be executed entirely. Contingency actions included, not only the classical trouble-shooting of the spacecraft, but it was also necessary first to enable a quick orbit assessment and thus ensure ground station acquisition and second to revise the ground stations allocation plan without excessive losses for the affected missions. In order to guarantee an adequate recovery, it was found that ground stations coverage should be provided during each burn.

D. Flight constraints

The plan also had to be compatible with all the flight constraints associated with the mono-gyro and gyro-less control modes. These modes were not as robust as the original 3-gyro piloting modes. In particular, 1-GP and 0-GP modes were sensitive to the interruption of DES data, especially during the burn.

Outages of valid DES measurements would most likely be caused by the presence of the Sun in its field of view. Due to the properties of the orbit, blinding by the Sun could only occur shortly before entry to eclipse or shortly after exit. From this point of view, the most favorable section of the orbit for the maneuvers was in the region of the orbital nodes. This approach was also favorable in term of ground station support, the ESA station at Kourou, French Guyana, was already part of the ERS-2 TT&C network and support from the ASI station at Malindi, Kenya could be envisaged. With these 2 stations, it was possible to perform burns in visibility, at opposite arcs of the orbit and within a delay in the order of 6 hours.

Interference from the Moon was also considered; this type of problem could only happen when the angle between the line of nodes and the Earth-to-Moon vector was within certain ranges. Fortunately, it could be totally excluded by phasing the "orbit determination" sub-phase of each maneuvers block with the revolutions of the Moon.[4]

E. Maneuvers plan

As reported earlier, above 700 km, maneuvers could potentially be performed using either OCM-1GP or OCM-E1GP. However, in OCM-E1GP, burn duration was limited to 300 s. At the initial tank pressures, a burn of 300 s was expected to impart a ΔV slightly inferior to 2 m/s. This value corresponded nicely to the maximum per maneuver that had been derived from the ground segment constraints. Hence using the '300 s' instead of '2 m/s' threshold to size the initial maneuvers was only slightly detrimental to the pace of the descent but it allowed ground operators to execute the maneuvers plan in the event of the malfunction of either gyro#4 or gyro#6. Below 700 km however, this approach did not make sense any longer; operations had to rely on gyro#6 and to minimize the risk of a failure of this unit before the completion of the descent larger maneuvers were desirable.

It was therefore decided to split the descent stage into 2 phases. In the first phase, individual burn duration would be limited to 300 s; these maneuvers would be executable in either OCM-1GP or OCM-E1GP. The second phase would be entered when the altitude had passed below the 700 km threshold and it would consist of maneuvers imparting a ΔV in the order of 2 m/s that could only be executed in OCM-1GP.

This approach was refined considering other constraints and a maneuvers plan comprising 64 burns arranged in 7 blocks was devised. The first 4 blocks belonged to the first phase and individual burn duration was limited to 300 s. During the second phase, the duration of individual burns was gradually increased to maintain the desired impulsion despite the declining force of the thrusters.[4]

F. Satellite configuration for descent

To initiate the descent, several modifications of the on-board configuration were necessary together with new tuning of a few on-board parameters.

Once payloads operations were terminated, yaw steering was no longer necessary, and the spacecraft remained in FPM outside of maneuvers.

Throughout routine, the two pairs of tanks had been maintained isolated from each other and each was connected to a branch of thrusters. The tanks selection was regularly modified to prevent excessive difference of pressures within the RCS. For disposal, it was decided to interconnect all the feeding lines. Hence frequent valves actuation could be avoided and a single pressure could be measured from 2 separate sensors.

Simulations and the test burns performed in February 2011 had revealed a risk of a transient saturation of a reaction wheel. In such a case, two separate surveillances designed to identify a failure in the commanding chain of the wheels and of the magnetorquers would have triggered a reconfiguration. However, for the case considered here, it was found that such a reaction was not adapted because the affected wheel normally returned to a safe range through the nominal off-loading algorithms. Hence it was decided to disable both surveillances at the beginning of disposal. In addition, the value of the control gain about X used in OCM-E1GP was modified to prevent excessive momentum transfer to the wheels at the end of the thrusters phase.

The selection of the safe mode gyro was also subject to a risk analysis. Either gyro#4 or gyro#3 could be selected. Unfortunately, gyro#4 shared a converter with gyro#6 and as gyro#6 was used for nominal control, a failure of this converter could have directly resulted in a loss of spacecraft. The alternative was gyro#3, however this unit had shown strong signs of degradation and there was doubt about its ability to keep operating throughout disposal. Gyro#4 was finally selected and indeed gyro#3 failed one week before the end of disposal operations

Although they were no longer required, the payloads were left switched on and commanded to the so-called “heater mode”. In this configuration, the thermal control of the payload module -where some platform units were also located- was performed by software which ensured less power demand than the hardware controlled heating configuration associated with payloads off.

As the altitude lowered, the apparent size of the Earth increased; analysis of this issue resulted in the update of a control parameter used during acquisition and the inhibition of the so-called “Earth diameter” surveillance throughout disposal. Altitude lowering was also associated with a decrease of the orbital period and therefore affected the algorithm driving the rotation of the solar array; this issue was addressed by ensuring frequent updates of the relevant control parameters and by relaxing the associated surveillance.

Finally, degraded pointing performances were expected at a lower altitude. In order to prevent unnecessary reconfigurations, the “yaw de-pointing in FAM2” and the “Sun presence” surveillances were also inhibited.

G. AOCS reference configurations during descent

Gyro failure was the sword of Damocles hanging over the disposal operations of ERS-2. Without gyro#6, the execution of maneuvers and the recovery from a safe mode would have become precarious. As detailed earlier, this unit was required to execute certain steps of the maneuvers plan. This plan, however, also accommodated some margins in case gyro#6 was temporarily unavailable. Analysis of previous gyro failures had highlighted the relation between failure and time of usage; in other words ground was expected to use gyros -gyro#6 in particular- with maximum parsimony.

This was not straightforward, because the mono-gyro and the gyro-less software retained some vestiges of their 3-GP precursor. In particular, transitions in and out of OCM were associated with an autonomous switch on of the selected gyros triplet. Hence, all remaining gyros had to be on for each of the maneuvers. In order to avoid repeated power cycles, it was decided to let all 3 units powered throughout the “maneuvers execution” sub-phase. This approach also had the advantage of enabling the estimation of angular rates about the 3 body axes. During the “orbit determination” sub-phase, all unnecessary gyros would be powered off.

For the first phase, OCM-1GP was preferred over OCM-E1GP for its superior performances. Ground operators could however resort to OCM-E1GP if the situation required. All 3 remaining gyros would be powered on throughout the “maneuvers execution” sub-phase and powered down during the “orbit determination” sub-phase when FPM-0GP would be used.

For the second phase, maneuvers could only be performed in OCM-1GP. For the “maneuvers execution”, the same configuration as the first phase would apply. However, FPM-0GP would no longer be suitable at low altitude and FPM-1GP would have to be used during the “orbit determination”. Consequently, gyro#6 would have to remain permanently on for several weeks.

Four different reference configurations were defined. The “descent-1GP” configuration was the prime configuration during the “maneuvers execution” sub-phase for both descent phases. “Descent-0GP” could be used as a backup for the "maneuvers execution" in phase 1. "Wait-0GP" was the prime configuration for the "orbit determination" in phase-1. Whereas for this sub-phase, "Wait-1GP" was the prime configuration in phase 2 and the backup in phase 1. Flight control procedures were developed to manage the transitions between these configurations.

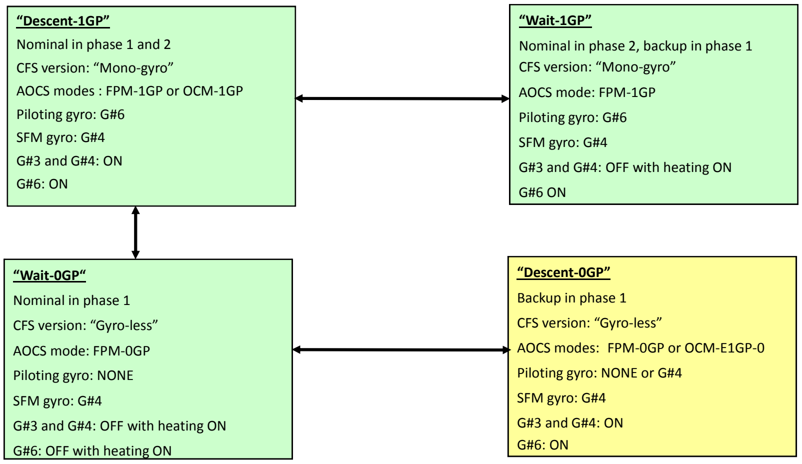

Figure 5. AOCS configurations during descent

H. Execution of descent

End of mission was declared on 4 July 2011 and the descent phase was immediately initiated. The last descent maneuver was executed on 22 August 2011.

Over these 50 days, 64 maneuvers were performed that imparted a total ΔV of 111.9 m/s and consumed an estimated 125.8 kg of hydrazine. During the descent, the semi major axis of the orbit was reduced by 210 km to reach a near circular orbit with a mean altitude of 572 km.

All maneuvers were executed in OCM-1GP with nominal performances, and interference for the other missions was minimal despite an extensive TT&C coverage. Further details are provided in Ref.4.

I. Gyro performances

In order to anticipate a gyro failure, a close monitoring process was put in place for the duration of disposal. Fast Fourrier Transform (FFT) analyses were performed daily to compute the power spectral density in the 1 Hz – 4 Hz range during individual passes ('PSD 1-4'). The measurement was compared against the given threshold of $1.67 \cdot 10^{-4}$ deg/s which was associated with a warning against an imminent failure of the unit.

For gyro#3, this threshold was violated early in the disposal; noise constantly increased until the failure on 30 August 2011.

The PSD 1-4 of gyro#4 remained below the reference threshold throughout disposal, although an increase of the noise was noticed for the last 2 weeks of operations.

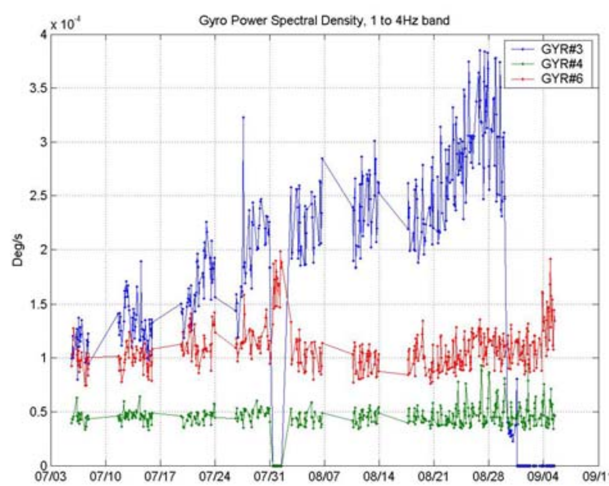

Figure 6. PSD 1-4 of gyro 3, 4, and 6 throughout disposal

For gyro#6 a momentary increase of noise was experienced in the first week of August when Gyro#6 remained on after the switch off of gyro#3 and #4. Noise came back on previous level as soon as these units were switched back on. This increase was due to a cross coupling effect with gyro#4 that shared the same converter unit as gyro#6. A genuine increase of noise was then experienced during the very last days of operations. Fortunately, gyro#6 remained functional until the end.

J. Control of batteries charge

Balancing used and available electrical power also required close attention. In routine, science operations were tuned to adapt power demand to expected power generation. For disposal, tuning of the Unregulated Power Bus (UPB) voltage limit (N_{vl}) was required to achieve an acceptable balance.

16 levels were available from 34.353 V to 36.978 V in increments of 0.175 V. On ground the need to tune the N_{vl} was assessed by monitoring the battery "K-factor" (ratio of charge to discharge). Values outside the range from 1.075 to 1.125 (at 0 ºC) triggered a need to lower or raise the N_{vl} level.

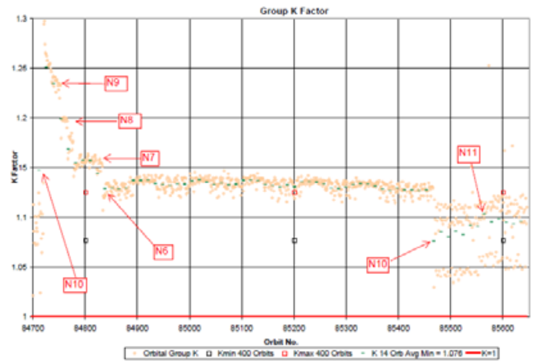

Figure 7. K-Factor and N_{vl} evolution during disposal

This strategy was required since UPB voltage was controlled higher than the maximum battery voltage. With this approach, full charge of the batteries was ensured by the end of the sunlit phase as only a hardware taper charge management was implemented. Update of N_{vl} was only required during LEOP and disposal. For all other phases, the N10 value (35.928 V) had been adequate. The adjustments performed during disposal are represented in Fig. 7.

VI. Development and execution of passivation

A. Design driver for the passivation stage

As mentioned already, the passivation stage encompassed all the operations to deprive the spacecraft from its energy and to terminate all radio emissions. To fulfill these relatively straight-forward objectives, operations had to be designed against a key constraint: after fuel depletion once fuel had been depleted, the spacecraft would not remain functional for long.

Fuel depletion was to occur during a thrusters based mode and once it had occurred, attitude would not be controlled anymore. Ground was then expected to perform all the remaining activities within a given duration of 25 minutes. In other words, ground had to ensure that "chemical passivation" (i.e. the depletion of fuel) was immediately followed by "electrical passivation"; that is to say the disconnection of battery charges and the switch off of the transmitter.

B. Precision of fuel estimation

This requirement was challenging because the amount of fuel remaining on board was not known with sufficient precision. The "pulse counting" method had been used throughout the mission. After each period of thrusters' actuation, the amount of fuel consumed in the period was derived from the pulses count provided in the TM and the conditions of the thrusters at the time (e.g. pressure of the propellant). Fuel book-keeping was maintained by subtracting these estimates of consumptions from the initial fuel complement. With this method, errors accumulated, and the precision of the fuel estimation was particularly poor at end of life.

A second method made use of the relation of Pressure, Volume and Temperature. The mass of remaining fuel was derived from the pressure and temperature measurements provided in the TM. Although, this "PVT method" was not affected by the accumulation of errors, estimations remained imprecise because of the limited resolution of the on-board pressure sensors.

For the disposal, the relative merits of both methods were assessed and the pulse counting method was selected for all the relevant flight decisions. Assessments with the PVT method would also be performed for verification. Fuel depletion was considered possible any time once the estimate had reached 30 kg. At this point however, the actual mass of fuel on board could have been significantly higher than 30 kg and this range corresponded to a burn of more than 2 hours.

C. Ground station coverage, reference orbit and maneuvers plan

Thus, a single burn would have to last more than 2 hours to cover fuel uncertainties whereas electrical passivation was expected to be performed within 25 minutes of fuel depletion. This meant that, the commands for the electrical passivation could not be time-tagged. Instead, ground station support had to be sufficient so that engineers could identify fuel depletion as it was happening in order to immediately initiate relevant commanding. However, near continuous support over such a long duration was far beyond the capabilities of the ground stations network. It was therefore necessary to spread passivation over several individual burns. During each of these burns, fuel depletion could happen at any time.

In addition, as observed during the disposal of SPOT-1 and SPOT-2, the depletion of the hydrazine was not expected to be sudden.[10, 11] Longer burns made complete de-priming more likely; hence maneuvers were designed with the maximum duration that ground could support. The best series of overlapping (or shortly separated) visibilities in the ERS2 network was provided by the following ground stations: Katsuura (located in Japan and operated by JAXA), Kiruna (located in Sweden and operated by ESA), Svalbard (located in the Spitsbergen island and operated by KSAT), Maspalomas (located in the Canary islands and operated by ESA) and Kourou (located in French Guyana and operated by ESA). Kourou, the last of the series would be used as a back-up to ensure that electrical passivation would be performed in time.

Overlapping visibilities from these stations only occurred when the spacecraft trajectory was within specific boundaries. Once a burn had been performed, it could take days before this constraint was met again. Fortunately, by the time fuel estimation reached the upper threshold of 30 kg a circular orbit with a mean altitude of ca. 570 km was expected; and around that altitude orbits with a ground track repetition of 1 day could be acquired. An orbit that would provide adequate visibility every day could thus be selected.

The passivation stage would start when the "1-day repeat" orbit at 570 km would be reached and a series of "passivation burns" would be executed to deplete the remaining fuel; ground would monitor their execution, ready to perform the "electrical passivation" if required. Passivation burns would alternatively raise and lower the orbit to maintain the ground track repeat pattern. Individual maneuvers would be separated by 1 to 3 days to acquire sufficient tracking data and to accommodate DES illuminations constraints.

The maneuvers plan was developed accordingly, it consisted of 2 short maneuvers to acquire the passivation orbit, followed by 2 shorter passivation burns of ca. 25 minutes each and 6 longer passivation burns of ca. 40 minutes each. The duration of these burns exceeded the thrusters' qualification range but was deemed acceptable and good performances at execution later confirmed this analysis. The passivation burns were designated from P#1 to P#8.

D. Configuration for passivation burns and electrical passivation

Prior to each passivation burn, a number of surveillances that could have resulted in a transition to safe mode were inhibited. If safe mode had been entered without a sufficient fuel reserve, electrical passivation would have been delicate.

Payload was also switched completely off with external heating permanently on. This configuration consumed more power that the one used during the descent; but it allowed the passivation of the payload without requiring an additional sequence once the fuel exhaustion was confirmed. Higher consumption induced higher depth of discharge (DoD) that was acceptable between the maneuvers. For the burns however, higher energy margin was required to ensure that electrical passivation would be completed despite possible off-pointing resulting from the fuel depletion. This was obtained by disabling permanent heating for 15 minutes during the eclipse of the burn. This approach had been validated in-flight by a set of tests during which heating duration in eclipse had been gradually reduced.

Once fuel depletion had been detected, the last actions on the spacecraft would be to disconnect the batteries charge and to switch off the transmitter. The order of these operations was subject to a risk analysis. Passivation was expected to occur in sunlit and in visibility from a ground station. On one hand, the last disconnection of battery charge would result in a loss of power regulation whereas, on the other hand, switching off the transmitter would prevent ground to monitor any subsequent operations. In the end, the following plan was adopted: first, disconnection of 3 battery charges, then switch off of transmitter and finally disconnection of the last battery charge in the blind.

E. Execution of passivation

The passivation orbit was acquired on 25 August 2011 and the first passivation burn was performed on the following day. The performances of the 5 initial burns were nominal. However, on 30 August 2011, gyro#3 failed, and angular rates could no longer be computed for 3 axes.

The sixth burn was initiated on 5 September 2011. At the end of the Kiruna support, 91% of the burn had been executed and there was still no sign of depletion. After the TM gap, the final 48 seconds of the burn could be monitored. Depletion was positively identified from 2 independent observables:

- During the TM gap, there was a significant drop from both propulsion pressure sensors. In less than 3 minutes, the pressure had decreased from 5.6 down to 5.2 bars (see Fig. 8)
- Gyro outputs clearly showed an increase in the angular rates that was characteristic of irregular thrust level (see Fig. 9)

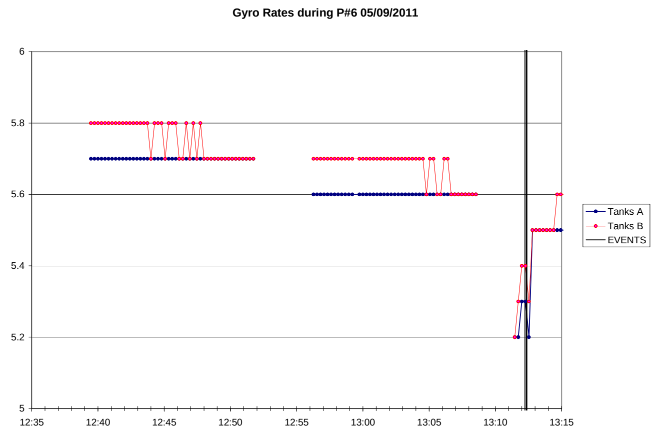

Figure 8. Pressure evolution during the final burn

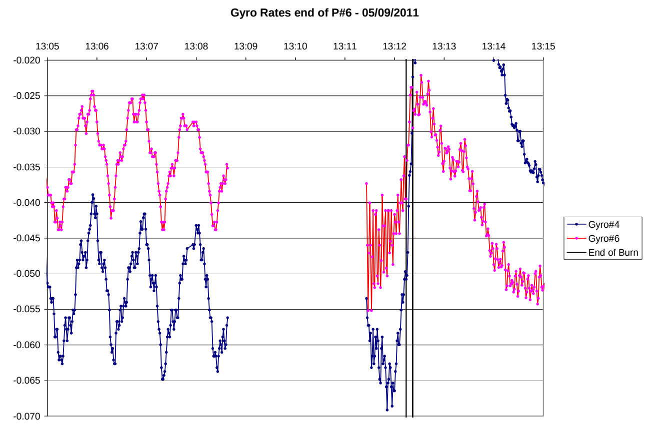

Figure 9. Gyros output at the end of the final burn

The electrical passivation sequence was uplinked shortly after the autonomous transition to FPM. The disconnection of the first set of 3 batteries was confirmed by TM. The loss of downlink was confirmed by ground station operator after the transponder switch off and the disconnection of the last battery was commanded in the blind. The last TM frame was processed at 13:14:24 UTC. Termination of radio emission was confirmed in the subsequent Kourou and Kiruna passes.

After the passivation, Fraunhofer-FHR supported several observations with the "Tracking and Imaging Radar" in Wachtberg, Germany. Tracking data were acquired and the disposal orbit was precisely determined. A series of images of the spacecraft was also generated; they confirmed that the rotation of the solar array had stopped.

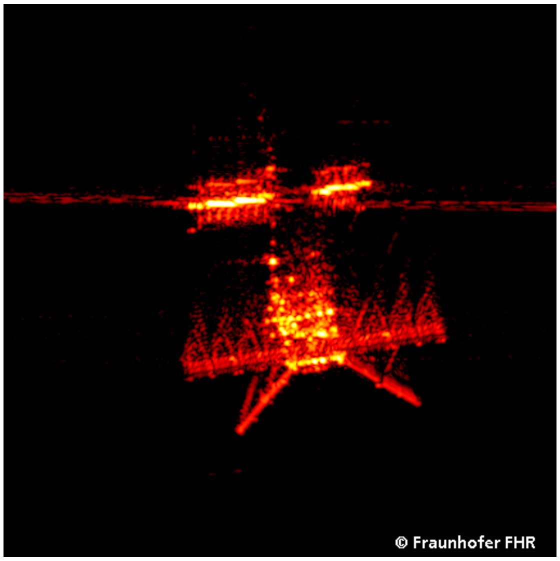

Figure 10: Radar image of ERS-2 acquired with TIRA on 07/09/2011
(Courtesy of Fraunhofer-FHR)

VII. Conclusion

This closed the final chapter of more than 20 years of continuous operations of ERS that was marked with radical evolutions of the control center, major on-board failures and new mission requirements. After several mission extensions, a successful disposal phase was implemented. This success is owed to the work of numerous professionals who developed a robust system and made it evolve over the years.

Appendix A

Acronym List

AOCS	Attitude and Orbit Control System
ASI	Agenzia Spaziale Italiana
CAM	Coarse Acquisition Mode
CFS	Central Flight Software
CNES	Centre National d'Etudes Spatiales
DES	Digital Earth Sensor
DoD	Depth of Discharge
DSS	Digital Sun Sensor
EADS	European Aeronautic Defence and Space Company
EBM	Extra Backup Mode
ERS	European Remote Sensing
ESA	European Space Agency
ESRIN	European Space Research Institute
ESOC	European Space Operations Center
ESTEC	European Space Technology Center
FAM	Fine Acquisition Mode
FCM	Fine Control Maneuver
FFT	Fast Fourrier Transform
FPM	Fine Pointing Mode
GP	Gyro-Piloting
IDHT	Instrument Data Handling and Transmission System
ITU	International Telecommunication Union
JAXA	Japan Aerospace Exploration Agency
KSAT	Kongsberg Satellite Services AS
LEOP	Launch and Early Operational Phase
OBC	On-Board Computer
OCM	Orbit Control Mode
PSD	Power Spectral Density
PVT	Pressure, Volume and Temperature
RCS	Reaction Control System
RW	Reaction Wheel
SAS	Sun Acquisition Sensor
SPOT	Système Pour l'Observation de la Terre
TIRA	Tracking and Imaging Radar
TM	Telemetry
TT&C	Telemetry, Tracking and Commanding
UPB	Unregulated Power Bus
UTC	Coordinated Universal Time
YSM	Yaw Steering Mode

Acknowledgments

This article is dedicated to the memory of Hugues Dufort, our long-time friend and colleague.

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