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Letter to the Editor

The INTEGRAL spacecraft – in-orbit performance

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Abstract. The INTEGRAL satellite was successfully launched from Baikonur on 17 October, 2002. INTEGRAL is an observatory for gamma-ray astronomy. The goals are to provide unprecedented high resolution imaging capability for unambiguous identification of gamma ray sources and high energy resolution for line spectroscopy. This paper summarises the actual orbital evolution based on the first 8 months in orbit and provides a status of the on-board limiting life resources. The paper describes the measured in-orbit performance of the INTEGRAL satellite and summarizes the applicable operational constraints for the science user community.

Key words. gamma-ray astronomy – space observatory

1. Introduction

The INTEGRAL (INTErnational Gamma Ray Astrophysics Laboratory) satellite was successfully launched from Baikonur in Kazakstan on 17 October, 2002 at 04:41:00 (UTC) by a PROTON rocket equipped with a Block DM 4th stage and injected in to a highly eccentric transfer orbit, $700 \times 153\,000$ km inclined at 51.6° with its apogee located in the northern hemisphere. INTEGRAL reached its final operational orbit by raising the perigee height and adjusting the apogee height to achieve a geo-synchronous orbit. This was done using the onboard mono propellant propulsion system. The operational orbit: 3 days period, $9000 \times 154\,000$ km inclined at 51.6°, was chosen to maximise the time spent outside the radiation belt around the Earth to provide a stable environment for the scientific observations. INTEGRAL (Winkler et al. 2003) is an observatory for astronomy in the field of gamma-rays, the most energetic photons at the end of the electromagnetic spectrum. The aims are to provide unprecedented high resolution imaging capability for unambiguous identification of gamma ray sources and high energy resolution for line spectroscopy. INTEGRAL is a very large and complex satellite, it is in fact ESA's heaviest scientific satellite ever. It could be implemented as a low cost mission only by using a service module based on the XMM-Newton design and with international cooperation. INTEGRAL is indeed a truly international enterprise. While ESA is in charge of the overall mission the satellite development and flight operations, the launcher is provided by the Russian Space Agency and the second ground station is provided by NASA. The scientific instruments and the science data centre are provided by Principal Investigators with funding from national organisations. The Mission Operation Centre (MOC) located at ESOC/Germany is in charge of control of the satellite using ground stations at Redu/Belgium (ESA) and Goldstone/USA (NASA). ESOC has visibility to INTEGRAL via the ground stations at all altitudes above 40 000 km. The INTEGRAL Science Operations Centre at ESTEC/The Netherlands (Much et al. 2003) and the INTEGRAL Science Data Centre at Versoix/Switzerland (Courvoisier et al. 2003) are in charge of the science planning, data processing, archiving and distribution of data to the scientific community. The description of the operations concept involving the ground segment is provided elsewhere in this volume (Much et al. 2003).

INTEGRAL, in orbit since more than 8 months, is performing flawlessly and in general with much better performance than originally required. The two life limiting on-board resources: fuel for momentum management and power from the solar arrays, have ample margin. This should ensure, that INTEGRAL will continue observations for the scientific community and hopefully make many unprecedented scientific discoveries in the field of high-energy astrophysics long beyond its 5 year operational lifetime.

The INTEGRAL payload consists of two large gamma ray instruments: the gamma-ray imager IBIS, the gamma-ray spectrometer SPI, and two monitoring instruments: two identical X-ray Monitors (JEM-X), and a optical monitor

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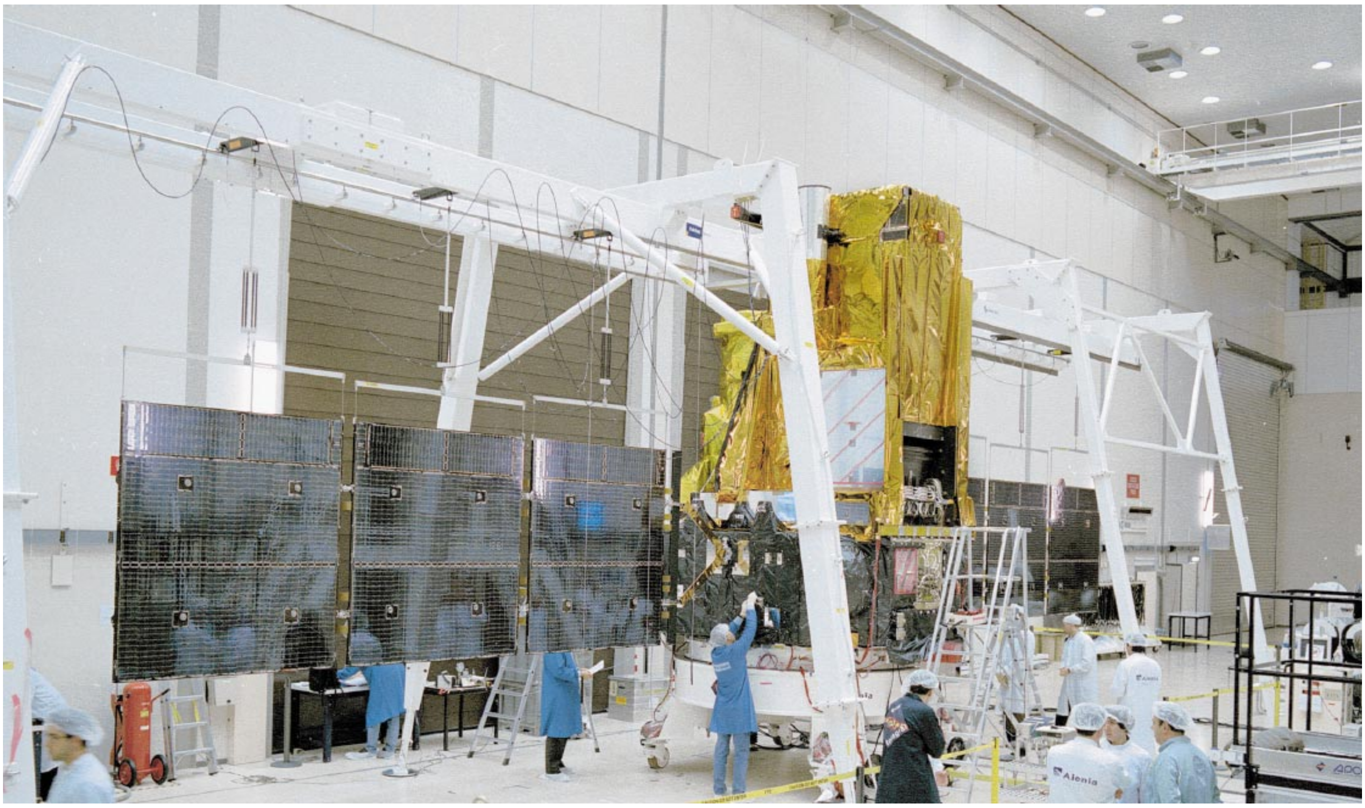

Fig. 1. The INTEGRAL flight model during solar array deployment testing at the ESTEC test facility.

camera (OMC). These instruments (Winkler et al. 2003 and references therein) are co-aligned and observe the same celestial object in the wavelength range from visible (500 nm) to MeV gamma-rays.

Figure 1 shows the INTEGRAL flight model during solar array deployment testing at the ESTEC test facility.

2. Summary of system design and operations

The spacecraft configuration was in many ways predetermined by (i) the decision to re-use the XMM-Newton Service Module design, (ii) the mass and field-of-view requirements of the instruments and (iii) the constraining dimensions of the Proton fairing. The basic requirement of the INTEGRAL satellite being compatible with the Ariane 5 and Proton launchers implied interesting design challenges. It required all mechanical and electrical interfaces, the environmental requirements and the spacecraft envelope to be established for design integrity with both launchers. The large mass of the spectrometer and the minimum focal length of the imager presented serious centre-of-gravity and fairing envelope problems, besides the need to distribute the loads into the Service Module structure. With local cut-outs in the fairing insulation and by rounding off the corners of the payload module upper part an almost perfect balancing of the satellite was achieved. The tight envelope also influenced the selected accommodation of the telecommunication antennae. By placing the antennae on short booms and on two diagonally opposite corners of the service module, the desire to avoid deployable booms and the fairing envelope constraints were both satisfied. A summary of the main facts and figures about INTEGRAL are given in Table 1. An exploded view of INTEGRAL is shown in Fig. 2.

2.1. Main operation modes

The INTEGRAL spacecraft provides stable pointings with pointing characteristics as described in Table 1. For the SPI instrument, the background in each of the 19 independent detectors varies in time in a different way. This variation can limit the sensitivity that is obtainable. Several types of background variations are present: (i) short-term variations due to solar activity and solar system "weather", (ii) variations over the orbital period (related to the position of INTEGRAL in the orbit), and (iii) long-term variations over the mission duration.

In order to reconstruct the image on the detectors for all sky pixels (~250) in the field-of-view with 2° resolution for a single pointing, a set of 19 equations with 156 unknowns would need to be solved. This is impossible, and the only way to increase the number of equations and make the system solvable is to observe more pointings. Thus, in order to solve this problem of background determination an appropriate dithering strategy has to be adopted for every observation.

This strategy consists of several off-pointings of the spacecraft pointing axis from the target in steps of 2°. The integration time for each pointing (all instruments) on the raster is flexible in the range between 0.5 hour to 1 hour. The integration time is adjusted in a way so that always multiples of a complete dither pattern are executed for each observation. The spacecraft will continuously follow one dithering pattern

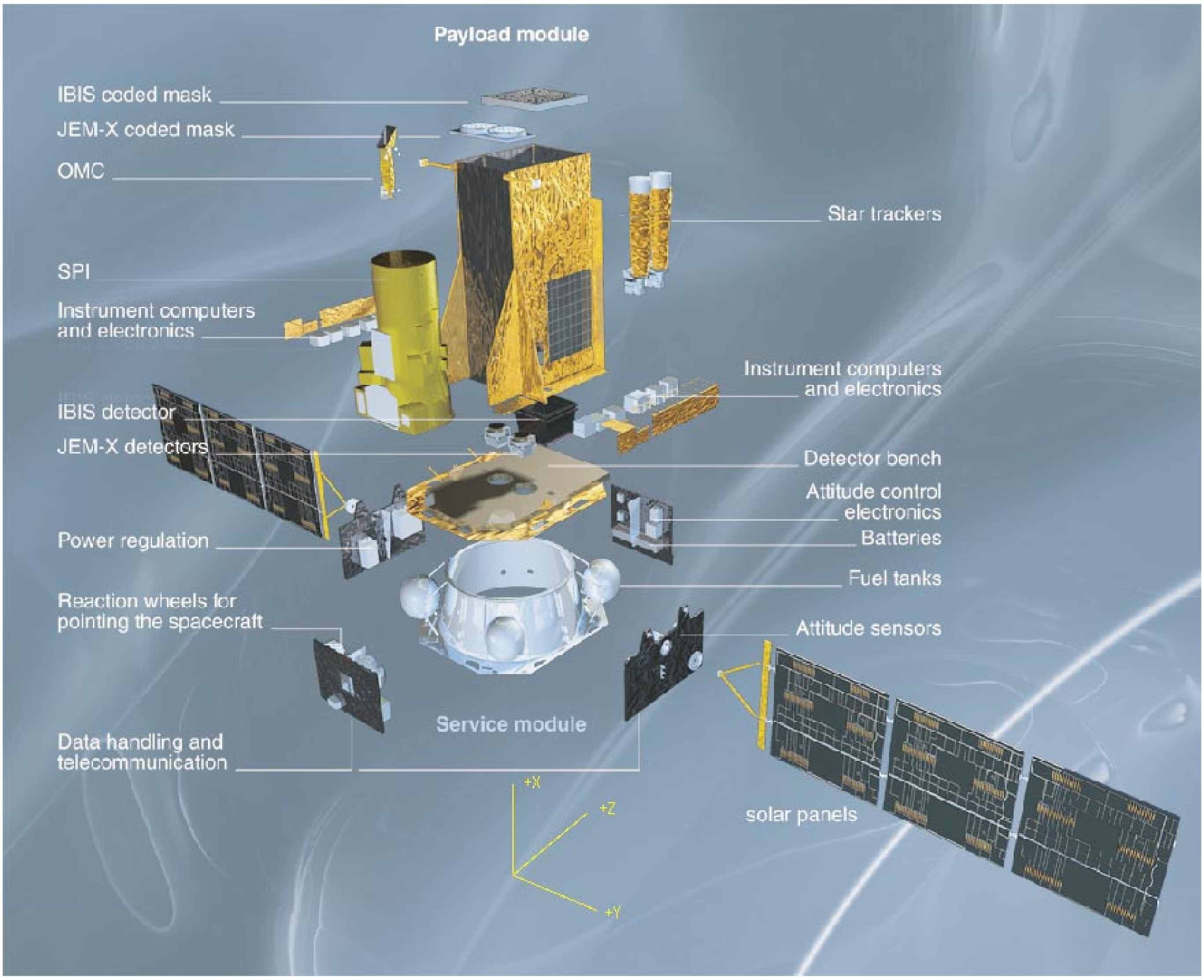

Fig. 2. Exploded view of the INTEGRAL spacecraft. Dimensions are $(5 \times 2.8 \times 3.2)$ m. The deployed solar panels are 16 metres across. The mass is 4 t (at launch), including 2 t of payload.

throughout one observation. Two different dither patterns and a staring mode (no dithering) are used as operational baseline:

(1) Rectangular dithering (baseline):
This mode consists of a square pattern (Fig. 3) centred on the nominal target location (1 source on-axis pointing, 24 off-source pointings, each 2° apart, in a rectangular pattern). This mode is used for multiple point sources in the FOV, sources with unknown locations, and extended diffuse emission which can also be observed through combination ("mosaic") of this pattern.

(2) Hexagonal dithering:
This mode consists of a hexagonal pattern centred on the nominal target location (1 source on-axis pointing, 6 off-source pointings, each 2° apart, in a hexagonal pattern). This mode will only be used for a single, strong, known point source, where no significant contribution from out-of-view sources is expected (Fig. 3). Experience from earlier observations has shown that this is not very often fulfilled (e.g. because of transient sources).

2.2. Autonomy

Integral has a large number of on-board autonomous functionalities despite it is operated in near-real time. This is mainly to control the spacecraft during unforeseen and planned periods with no ground contact and to recover from on-board anomalies where a quick reaction is required. INTEGRAL is designed to survive any 36 h outage period from ground assuming any single point failure. During the perigee passage there is a planned ground outage period of 4 to 6 h. During this period all the eclipses take place. Before and after an eclipse passage the spacecraft attitude and orbit control system and power system reconfigure itself as the Sun is not available for navigation and power source. A specific INTEGRAL feature is the "Broad Cast Package" (BCP) dsitributed on-board to all the instruments every 8 s. This BCP contains on-board generated

Table 1. INTEGRAL – main facts and figures (see also Fig. 2).

Objective	Fine imaging and spectroscopy of celestial gamma-ray sources in the energy range 15 keV to 10 MeV
Instruments	
Imager IBIS	15 keV–10 MeV, coded aperture mask, 16483 CdTe dets, each $(4\times4\times2)$ mm, 4096 CsI dets, each $(8.4\times8.4\times30)$ mm
Spectrometer SPI	18 keV–8 MeV, coded aperture mask, 19 Ge dets, each (6×7) cm, cooled @ 85K
X-ray Monitor JEM-X	4 keV–35 keV, coded aperture mask, micro-strip detector $(2\times)$, (each Ø250 mm) Xe/CH_4 gas
Optical Monitor OMC	*V*-band (500–600 nm), CCD detector, refractive optics
Launch Vehicle	Proton with Block DM upper stage. Launch from Baikonur/Kazakhstan
Operational orbit	Apogee height: 153 657 km
(as achieved on	Perigee height: 9050 km
01 Nov 2002)	Inclination: 52.2°
	Argument of perigee: 302°
	RA ascending node: 103°
	Period 3 days
	Max eclipse duration: 1.2 h
Ground stations	Redu (Belgium) and Goldstone (California)
	Coverage 100% above 40 000 km
Lifetime	2.2 years nominal 5.2 years extended
Dimensions	Satellite body dimension : $(2.8\times3.2\times5)$ m
	Solar array span 16 m
Mass	Total mass: 3954 kg
	Dry mass: 3414 kg
	Fuel (hydrazine, at launch): 540 kg
	Instrument mass 2013 kg
Power/Energy Storage	28 V regulated power bus
	Advanced Rigid Solar Arrays, Silicon cells.
	@ launch (@ 2.2 years): 2377 W (1960 W), Sun aspect angle: 0°
	@ launch (@ 2.2 years): 1834 W (1630 W), Sun aspect angle: 40°
	Two 24 Ah NiCd rechargeable batteries
Communication	*S*-band up and down link, 2 fixed antennae
	Telemetry rate: 113 kbps (payload: 108 kbps)
	Telecommand rate: 2 kbps
Mechanical Properties	First axial mode: $\geq$38 Hz
	First lateral mode: $\geq$12 Hz
	Load case 1: ±9 g longitudinal, ±1.5 g lateral
	Load case 2: ±0 g longitudinal, ±4.5 g lateral
Pointing and alignment	3-axes stabilised spacecraft (instrument pointing axis: +X, Sun pointing: +Z)
(as specified)	Absolute pointing error: $5'$ (Y, Z), $15'$ (X)
	Instrument alignment: $1'$ (Y, Z), $3'$ (X)

information and on-ground loaded information and includes information about orbital events to which the instruments should react: (i) time/altitude when an observation can start or shall finish due to the radiation belts, (ii) time of eclipse entry to warn instruments before being switched off during eclipse, (iii) status of "On Target Flag", which relates to the attitude stability. When the flag is set it indicates that instruments can start observations, e.g. after a slew, and, (iv) radiation monitor readings according to which the instruments shall power down the high voltage and go to safe mode, as soon as an instrument specific threshold is met. All these features are designed to optimise operational efficiency and protect the instruments in case of anomalies.

INTEGRAL has several on-board failure detection and recovery functionalities, which are mainly focussed on the attitude control and the on-board power system, which are the only two sub-systems which eventually could endanger the survival of INTEGRAL in case of failure. The attitude and orbit control system operates with three hardware groups: nominal operation, failure detection and failure recovery. This ensures recovery for any single point failure. The failure recovery mode is a hardwired control mode using the thrusters for actuation. The spacecraft has in addition a software based on-board monitoring and action routine. Any on-board generated telemetry can be monitored, and in case a pre-set threshold is passed three consecutive times the defined action will be invoked. This function is already used to monitor the compressor power demand from the SPI cooling system. The function will be important later in the life cycle of INTEGRAL, when more on-board anomalies can be expected.

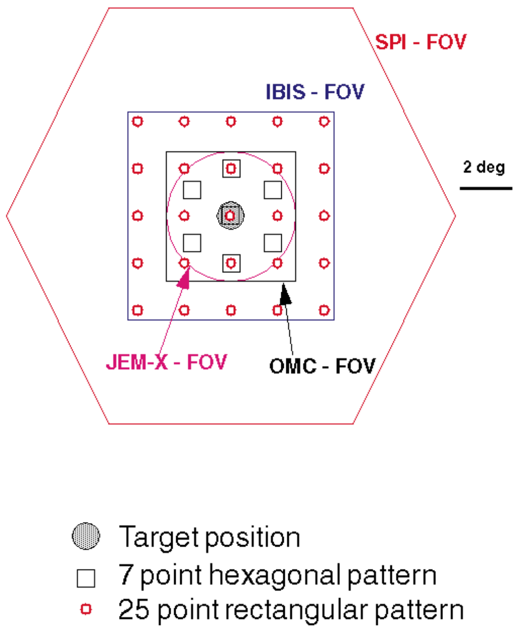

Fig. 3. Dither patterns for INTEGRAL (see text).

3. Mission analysis

3.1. Orbit design

INTEGRAL was launched on a three stage Proton rocket, which is a powerful launcher plus a Block DM upper stage that is capable of making several re-starts. This allowed a variety of different orbits from circular to highly eccentric that were all studied in detail. An inclined, highly eccentric orbit with the apogee in the northern hemisphere and with nominal parameters as shown in Table 2 was selected.

This orbit was preferred over a more circular orbit due to simplification of the injection scenario and being less demanding on the spacecraft thermal and power subsystems. The orbit also provides 84% of the time above an altitude of 60 000 km which is completely outside the Earth's radiation belts. This provides perfect conditions for real time, long duration undisturbed scientific observations which is one of the important design requirements. The two ground stations in Redu and Goldstone together provide complete telemetry coverage for satellite altitudes above 40 000 km.

3.2. Initial orbit and evolution

INTEGRAL was injected into the expected highly elliptical transfer orbit. Table 2 presents the parameters of the actual transfer orbit.

The injection into the transfer orbit was followed by perigee raise manoeuvres at the 3rd, 4th and 5th apogee passage in order to raise the perigee height to the nominal 9000 km. At the 5th perigee passage a minor apogee height adjustment was performed to achieve a perfect synchronous orbit. INTEGRAL was designed for a maximum eclipse duration of 1.8 h but by careful selection of the local launch time it was possible to reduce this maximum eclipse time to 1.2 h and still achieve an orbit where the perigee remains outside the proton belts. The short eclipse minimises the thermal drift and thermal/mechanical stress of the detectors and electronics, which is important for the instrument calibration. There are two periods every year with eclipses, each lasting approximately 21 days and containing 6 to 7 eclipses.

Table 2. INTEGRAL transfer orbit (achieved).

Parameter	Value
Epoch 2002/10/19	23:58:04 UTC
Semi-major axis	82 746.8 km
Eccentricity	0.915145
Inclination	51.71°
Ascending node	104.45° °
Arg of perigee	300.25°
True anomaly	0.007°
Perigee height	643.3 km
Apogee height	152 094 km

Table 3. INTEGRAL final orbit (achieved).

Parameter	Achieved orbit	Nominal orbit
Epoch	2002/11/01 20:15:00 UTC	
Semi-major axis	87 731.5 km	87 678.14 km
Eccentricity	0.824148	0.813202
Inclination	52.246°	51.6°
Ascending node	103.07°	104.9°
Arg of perigee	301.72°	300.0°
True anomaly	180°	0.0°
Perigee height	9049.6 km	9000.0 km
Apogee height	153 657.2 km	153 600.0 km
Orbital period	72 h	72 h

The finally achieved operational orbit parameters are described in Table 3 and are very close to the nominal (designed) operational orbit.

The INTEGRAL operational orbit is predicted to slowly increase in perigee height from 9000 km to 12 500 km and to increase in inclination from 51.6° to 87° during the first 5 years in orbit, due to the natural disturbances from Sun, Earth and Moon. This evolution provides an increasingly favourable orbit in terms of the radiation belt avoidance and ground station coverage. A very minor propellant budget is required to maintain the geo-synchronous orbit, i.e. to maintain the longitude of perigee passage in a range providing maximum ground station coverage. From a coverage point of view, it becomes less important to control the longitude of perigee passage with increasing inclination. This feature is reflected in Fig. 4, which shows the accumulated ground station coverage above 60 000 km for the nominal ground stations at Redu and Goldstone at launch, 2.2 years and 5.2 years.

Spacecraft operations also disturb the operational orbit: The thrusters used to perform the momentum off-loading of the reaction wheels are all oriented with a 12° tilt with respect to the Spacecraft bore-sight axis (X-axis). Any thruster firings

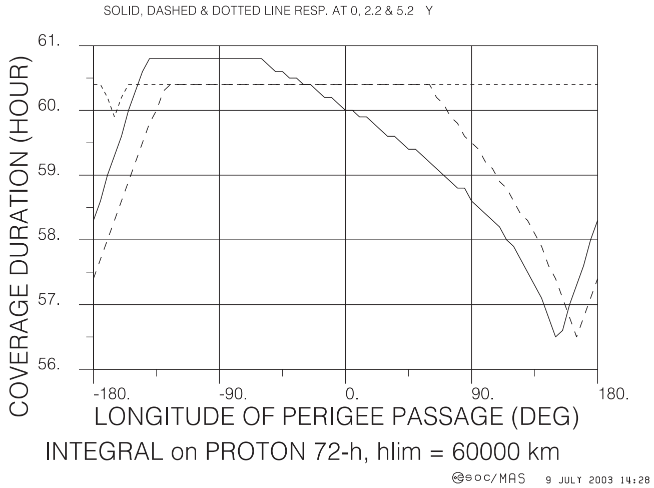

Fig. 4. Accumulated ground station coverage as function of longitude of perigee and orbital time.

will therefore result in a parasitic thrust in the *X*-axis, which eventually could disturb the operational orbit. However, by using an elaborated strategy to perform the reaction wheel off-loading, the Mission Operation Center (MOC) can actually turn this into an advantage and control the orbital evolution using this parasitic thrust. The main effect of the off-loading on the orbit is around perigee passage, and by choosing the descending or ascending trajectory – while performing the off-loading – the desired effect on the orbit will be achieved.

The predicted orbital evolution is very close to the actual evolution (up to June 2003), i.e. the actual perigee and apogee heights differ by less than 300 km from their predicted values, and the actual inclination differs by less than 0.8°. The actual evolution of the longitude of perigee is very close to the predictions and favourable from a ground station coverage point of view (Fig. 4). No dedicated orbital correction manoeuvres have yet been required.

3.3. Radiation environment

The space environment is very harsh in terms of the radiation environment. Particle radiation degrades the quality of the collected data by increasing the background, but moreover it can degrade instrument components, can cause high voltage breakdowns and it is responsible for the radiation damage in the SPI Ge detector crystals. Therefore the INTEGRAL orbit (see above) was selected to minimize the radiation effects and the orbit stays above the proton radiation belts at all the time.

As the monitoring of the radiation environment is critical for the operation and safety of the instruments, a radiation monitor, the INTEGRAL Radiation Environment Monitor (IREM), was placed on the spacecraft. IREM detects and counts electrons, protons and cosmic rays with a coarse spectral resolution. It also measures the total radiation dose encountered by its sensors. IREM comprises of two detector systems, a proton sensitive coincidence detector system and a single detector that measures electrons (≥0.5 MeV) and protons (≥20 MeV). INTEGRAL has the capability of distributing onboard two IREM counter values to the instruments in so called broadcast packets. The IREM S14 counter and the TC3 counter (Zehnder & Hajdas 2001), were selected for this purposes, because these two counters are *pure* counters, i.e. S14 is effectively only sensitive to protons and TC3 only to electrons. Each instrument is automatically put to safe mode, once a radiation level reported in the broadcast packet exceeds the relevant, instrument specific threshold.

The proton threshold was always reached well below 40 000 km, when the instrument were already switched off. The critical radiation level is reached at higher altitudes at perigee exit compared to perigee entry. There is only very little evolution of the altitude when entering perigee.

The critical electron radiation level is reached around 30 000 km at perigee exit. This altitude is more or less stable within the fluctuations, though a small decrease may have occurred during revolutions 39 to 60. Soon after launch it was realized that the electron radiation is still high above 40 000 km at perigee entry. Consequently the operational altitude was adjusted to 60 000 km, i.e. by default the instruments high voltage are switched off at 60 000 km. However, the electron counter threshold for JEM-X was very often reached above 60 000 km and therefore JEM-X is now routinely switched off at 70 000 km. In the period from revolution 35 to 52 a JEM-X transition into safe mode was triggered several times by the IREM broadcast packet even above 70 000 km, reflecting the high electron radiation in this period.

Figure 5 shows a comparision between the measured IREM S14 (proton) and TC3 (electron) counting rates with the predictions from the NASA AE8/AP8 radiation belt models for the perigee passage from revolution 7 to 8 (from 01 h to 10 h UTC on November 6, 2002). The model predicts for both counters the measurements qualitiatively, but not quantitatively. This is not suprising considering the rather low threshold settings (0.3 S14 counts/s and 60 TC3 counts/s) compared to the maximum radiation level reached during perigee (3 S14 counts/s and 200 000 TC3 counts/s). The threshold settings are sensitive to the noise in the radiation belts. It is therefore required to continue monitoring the evolution of the radiation environment with the IREM and to adjust the operational altitudes accordingly.

4. Spacecraft performances

4.1. On-board resources

There are two key life limiting resources on board: propellant used to off-load the reaction wheels and electrical power produced by the solar arrays, which degrade continuously due to the radiation environment. The present INTEGRAL on-board resources as per 1 June 2003 have been evaluated and compared with the estimated figures at the Flight Acceptance Review (FAR).

Propellant versus lifetime The propellant quantity inside the tanks is regularly computed by ESOC Flight Dynamics to confirm the satellite remaining lifetime. The INTEGRAL spacecraft was loaded before launch with 543.7 kg of hydrazine propellant and 4.2 kg of pressurant (corresponding to a pressure of

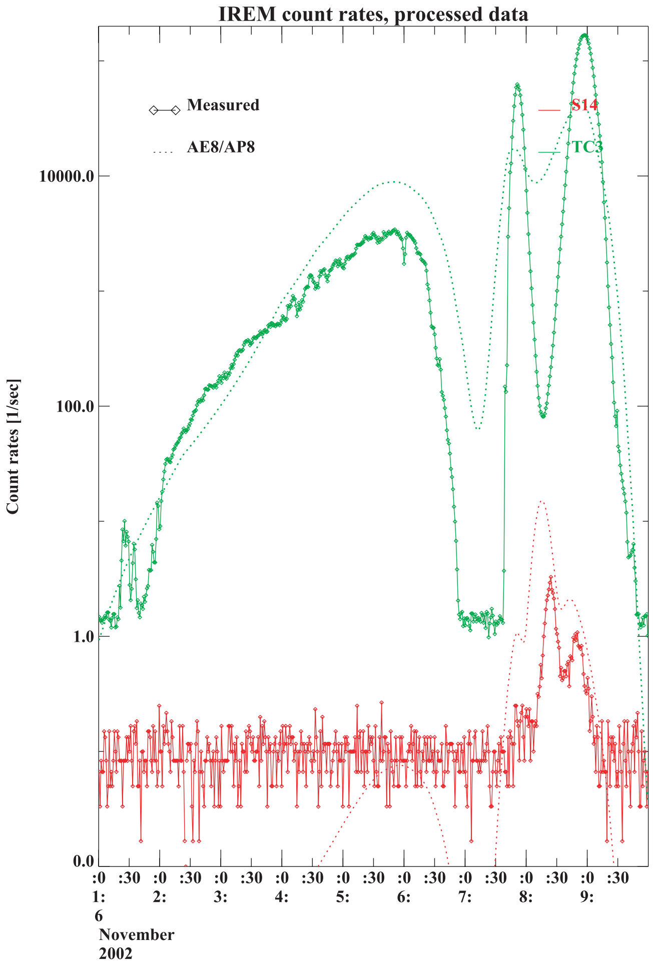

Fig. 5. Comparison of the IREM S14 (protons – red solid line) and TC3 (electrons – green solid line) counting rates with the predictions from the NASA AE8/AP8 radiation belt models (dotted curves). The perigee passage from revolution 7 to 8 is shown on November 6, 2002 from 01 h to 10 h UTC. In both cases the model predicts the measurements qualitiatively, but not quantitatively.

24 bar at a temperature of 30 °C). The fuel budget presented at the FAR was based on a loading of 540 kg of fuel and a final perigee height of 10 000 km. The estimated fuel margin at End of Life (EOL) after 5 years in orbit was 49 kg. The initial fuel used on INTEGRAL was related to the initial Sun acquisition and Sun sensor acquisition control mode of the Attitude and Orbit Control System (AOCS). The actual consumption was 0.61 kg against a budgeted value of 1.63 kg at FAR. The fuel consumption for the perigee and apogee manoeuvres was evaluated to be 360 kg based on in-flight data. This compares with a budget estimate of 384 kg, showing a total saving of 24 kg. This saving was mainly due to the very accurate injection in to the transfer orbit the very efficient perigee raise manoeuvre planning and execution. The only other sources of propellant consumption are the reaction wheel momentum off-loading, orbit maintenance and the emergency attitude control mode activated in case of an attitude anomaly.

The amount of fuel allocated for reaction wheel momentum off-loading was estimated pre-launch to be 45.2 kg for the 5.2 years extended life and based on certain assumptions on

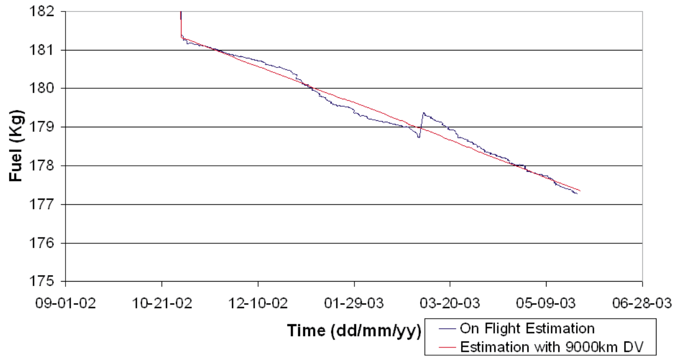

Fig. 6. Measured (blue) and predicted (red) fuel depletion.

slew manoeuvre amplitude and frequency, in orbit disturbance torque due to the solar pressure, and reaction wheel set under usage. Two momentum off loading manoeuvres were baselined for each orbit giving an average consumption per orbit of 0.0572 kg.

Figure 6 shows the comparison between the fuel depletion trend, as recorded from on board data, and as estimated. The two curves are consistent, confirming the assumptions made for the estimate. Extrapolating the actual measured propellant consumption until now throughout the mission lifetime of 5.2 years gives an estimated fuel consumption of about 37 kg for momentum off-loading. This results in an expected margin of 82 kg after 5.2 years.

Each momentum off-loading causes a small parasitic delta velocity to the orbit. However, by performing the momentum off-loading at an appropriate spacecraft attitude, the parasitic delta velocities averaged out such that no orbit maintenance manoeuvre is foreseen up to now. Also up to now no anomaly causing an emergency attitude control mode has occurred.

The need to do momentum off-loadings depends on the operational use of INTEGRAL and in particular on the amount of large slewing. With the present fuel situation and assuming a similar operational scenario without any on board failures, there is enough fuel for approximately 15 years in orbit.

Power The INTEGRAL spacecraft employs fixed solar arrays. Two elements need to be considered related to the on board power resource: (i) the power supplied by the solar arrays as a function of the Sun aspect angle, and (ii) the power degradation as a function of the satellite life (2.2 years nominal mission). The first point is shown in the Fig. 7, where the measured power is corrected at Winter solstice conditions, to allow comparison with supplier's predictions (continuous line). The agreement is good and minor differences are due to the fact that the attitude is not sufficiently stable over a long period, as INTEGRAL is almost continuously slewing. In addition, the resulting non-steady temperatures make it difficult to refer the measurements to the predictions at 25 °C.

The lifetime degradation of the solar arrays was evaluated from the "Beginning of Life" performance by applying degradation factors derived from supplier's information based on their experience. This exercise was primarily needed to evaluate the need for reducing the maximum allowed Sun aspect

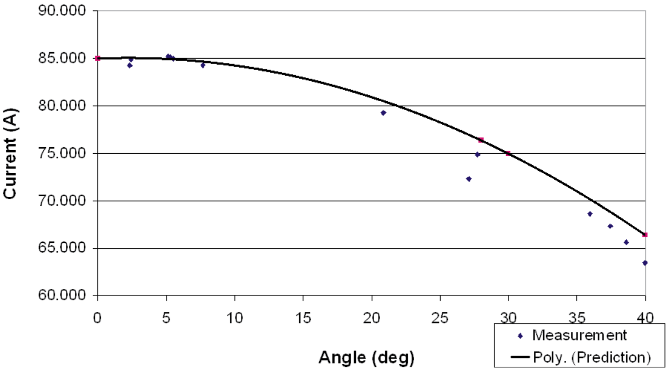

Fig. 7. Current generated by solar arrays (see text).

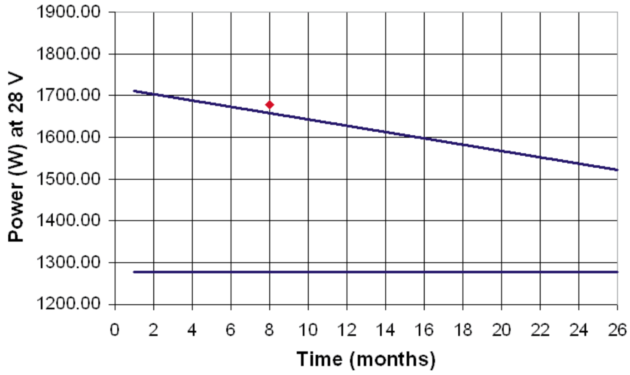

Fig. 8. Power margin of solar arrays at Sun aspect angle of 40°. Actual measurement (red diamond), predicted degradation (line), and power demand (horizontal line at 1280 W).

angle during eclipse season and during the extended mission after the nominal 2.2 years in orbit. INTEGRAL is designed to operate with a Sun aspect angle of 40° during the first 2.2 years in orbit and with 30° during eclipse seasons and during the extended mission after the initial 2.2 years in orbit.

Figure 8 shows the expected degradation of the solar array power, and the satellite power consumption including an allocation for on-board failures (horizontal line at approximately 1280 W). The power supplied after 8 months in orbit is also indicated. From this we can conclude that it will not be required to reduce the Sun aspect angle during eclipse season of the initial 2.2 years nominal life of INTEGRAL and that the extended mission can be carried out with a Sun aspect angle of 40° outside the eclipse season. The overall power situation must be reviewed on a 6 months basis in order to benefit from the largest possible Sun aspect angle and thereby achieve the maximum sky visibility.

4.2. Telemetry

Soon after launch it became evident that the radiation background in the gamma-ray instruments was higher than expected. As a consequence the instrument event rate was exceeding the telemetry downlink capability. Enhanced on-board processing for background rejection could solve the problem only partially. It was therefore decided to investigate a potential increase of the telemetry rate. The telemetry system consists of three major elements

- the on-board date handling which acquires source packets from instruments and subsystems in a sequence determined by the programmable polling sequence table and then delivers the packets to the telemetry manager;
- the telemetry manager, which assembles the packets in frames, performs the encoding and finally provides a continuous bit stream to be modulated on the radio frequency carrier;
- the *s*-band radio frequency downlink from the on-board transmitter to the ground station.

Ground station off-point tests confirmed that the downlink had good margin providing a 1dB head room for telemetry rate increase. Also the on-board data handling had margin for acquiring an increased rate of packets and was furthermore easily programmable.

The problem was the telemetry manager, designed for a fixed bit rate. However, it turned out that a divider which derived the telemetry clock from an oscillator, could be set differently by a software patch. After careful validation on the INTEGRAL engineering model (after all, the telemetry is the operational life line of the satellite) the patch was uplinked changing the crucial divider stage from 1/10 to 1/8. The overall telemetry rate (payload and spacecraft) thus was increased from about 105 to 131 kbits/s at frame level. This allowed to change the polling sequence table from 204 to 256 packets/8 s making full use of the data handling capability margin. The number of packets available to instruments went up from 195 to 246 packets/8 s, a gain of 26%.

4.3. Pointing stability

The spacecraft orthogonal co-ordinate system is described by *X*-, *Y*-, *Z*-axes (Fig. 2) with origin at the centre of the separation plane between spacecraft and launch adaptor. The *X*-axis is perpendicular to this spacecraft/launcher separation plane, pointing positively from the separation plane towards the spacecraft (i.e. the *X*-axis is the pointing direction). The *Z*-axis is orthogonal to the solar array surface, pointing positively to the Sun. The *Y*-axis completes the coordinate system.

Scientific observations with INTEGRAL require that the spacecraft pointing is stable to within 18″ around *Y*- and *Z*-axis and 60″ around *X*-axis (line-of-sight). This stability is achieved in the inertial pointing and slew (IPS) mode of the Attitude Orbit and Control System (AOCS), which uses the fine measurements of the star tracker and fine Sun sensor as inputs to the three control loops. The output from these controllers are momentum demands for the reaction wheels that are operated in speed feed-back loops. The performance of the steady pointing stability is measured by the controller errors around the three X, Y, Z-axes which are provided as Φ, Θ and Ψ angles in the telemetry.

Figure 9 is the Φ error evolution over a 4 hrs time period, showing a complete fulfilment of the requirement (7.5″ maximum error vs. 60″ requirement). Two samples at different flight

Table 4. Pointing stability (operational vs. worst case simulation). Controller errors Φ, Θ, Ψ are given in arcseconds.

	Simulation			Operational 18 Dec 2002			Operational 25 May 2003		
	Φ	Θ	Ψ	Φ	Θ	Ψ	Φ	Θ	Ψ
Max	7.5	8.1	9	6.958	0.8906	2.59	6.62	0.885	1.85
Min	−7.5	−8	−7.5	−5.57	−1.0738	−4.86	− 7.69	−0.834	−3.06

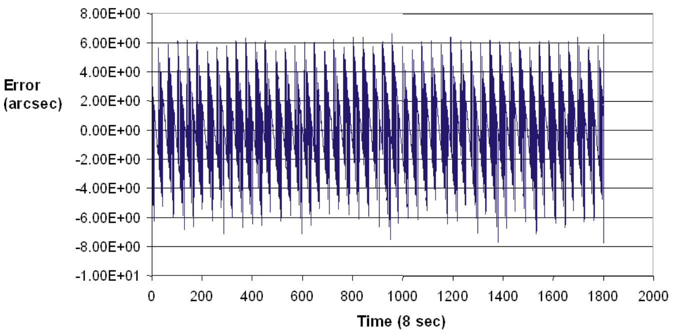

Fig. 9. Φ control error stability over 4 h. Requirement is 60′′.

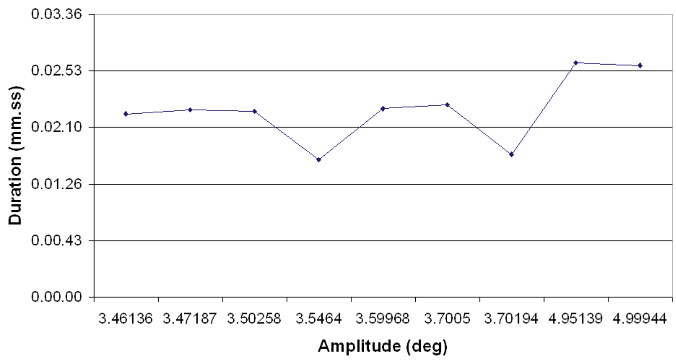

Fig. 10. Slew duration for open loop slews.

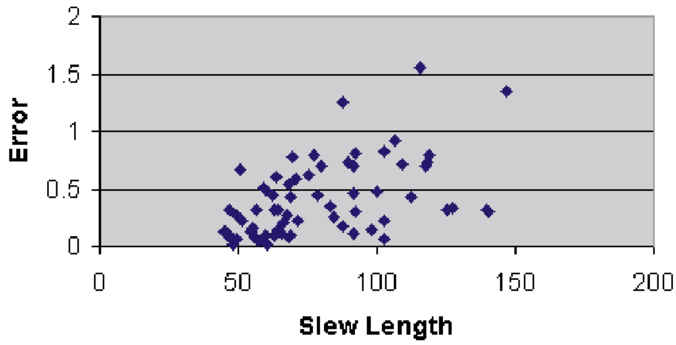

Fig. 11. Slew error for large open loop slews. Units for slew length and error are degrees.

times are shown in Table 4 where a comparison with worst case simulation results is provided.

4.4. Operational efficiency

The operational efficiency is given by the percentage of the total mission time that is available to science observations above an altitude of 40 000 km. The time available to science is reduced by the time to perform slews and the number of reaction wheel off-loadings. Two different type of slews are performed by INTEGRAL:

- Closed loop slews (manoeuvre with control on all three axes, Fig. 2), i.e. typically 2° slews during dither observations.
- Open loop slews (manoeuvre with the Z-axis uncontrolled, and X- and Y- axes controlled), i.e. typically 7° slews during Galactic plane scans and very large slews to re-point to another target.

The performance of the slews can be rated as a function of slew duration and pointing error at the end of the slew and the need to perform a corrective slew to achieve the desired pointing.

From flight data, the duration of open loop slews between 2° and 7° was always found within the required value of 5 min; a snapshot of open loop slews duration is shown in Fig. 10. The duration of closed loop slews (≤2°) is also always within the requirement of 3 min.

The present statistics from large open loop slews with lengths between 45° and 147° showed a maximum error (length of the corrective slew) of 1.55°, compared to a requirement of less than 5% as depicted in Fig. 11. Open loop slews up to 7° are well within the requirement as shown by Fig. 12 taken from a set of open loop slews performed during the commissioning period.

Reaction wheel off-loadings are needed for mainly three independent reasons:

- to compensate the momentum accumulation caused by external disturbance torques;
- to compensate the momentum accumulation due to slew manoeuvres;
- to meet the momentum slew constraints (basically for large angle slews).

Data from 121 reaction wheel off-loading maneuvers performed since the end of the commissioning phase (November 2002–April 2003) show that the time spent for off-loading is in line with the expected one and does not have significant impact on science time.

5. Operational constraints

This section summarizes constraints for operating the spacecraft.

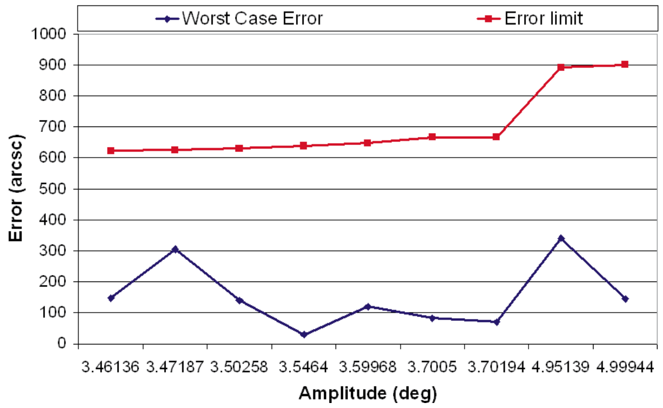

Fig. 12. Slew error for small open loop slews.

Sun aspect angle The power design requires that the angle (Sun aspect angle) between the spacecraft $+Z$ axis (Fig. 2) and the Sun vector must not exceed 40°. The original constraint to limit the Sun aspect angle to 30° after the first 2 years was replaced by a more rational approach based on the supplied current and the main bus (consumed) current: as long as the difference between them is greater than 5 A there is no need to reduce the Sun aspect angle. If the difference becomes less than 5 A, a reduction to 30° would be needed to safeguard the spacecraft operations. Further to this limitation and because of thermal protection of the instruments the roll angle around the Z-axis must not exceed ±5° with respect to the Sun vector.

Minimum approach angle To avoid excess stray-light from Earth and Moon into the star tracker field of view, the angle between the X-axis and the Earth and Moon limbs must be larger than 15° and 10° respectively.

Slew constraints The open loop slew for angles larger than 7° is constrained by a limitation on the initial angular momentum on the spacecraft axes to avoid instability of the slew controller. Moreover, to account for the slew error, a margin on the final angular momentum needs to be left on each wheel to avoid triggering of the so-called autonomous momentum dumping (see below).

Wheel speed range Reaction wheel operation must avoid that each of the wheels crosses the region +120 rpm to −120 rpm at a low speed, and, exceeds its maximum (4000 rpm) speed. The constraint must be guaranteed by a ground controlled reaction wheel off-loading function. To cope with the outage requirement or on-board malfunctioning a autonomous momentum dumping function is implemented on-board.

6. Spacecraft status

The general status of the spacecraft is good, the baseline configuration selected before the launch is still maintained and no need to use redundancy or back-up functions occurred.

Radio frequency subsystem The radio frequency subsystem is working nominally and link budgets show sufficient margin so that it is even possible to keep the ranging permanently on. A check of those margins showed that the link budget could accommodate an increase of the telemetry data rate.

On-board data handling The On-Board Data Handling (OBDH) subsystem is working properly. No problems or degradation have been identified so far. Modifications have been implemented in the software to allow the increase of the telemetry rate; with this modification a 26% increment of the data collection from payload instruments was achieved.

Power subsystem The power subsystem is working properly. The solar arrays provided power of 2380 W at "Beginning Of Life" at 0° Sun aspect angle is as expected. No degradation of the solar arrays has been identified so far. The performance of the two 24 Ah NiCd batteries was measured during the first eclipse season. The end of discharge voltage at the end of the eclipse is a good indicator of the battery degradation: the measured value is above 39 V for both batteries at the end of the longest eclipse, significantly above the minimum operating voltage of 31 V. This result shows that the batteries are in good health. The maximum depth of discharge measured during the eclipse season was less than 15%. Therefore it was decided to postpone the battery reconditioning (a maintenance operation aimed to restore the energy storing capability requiring a complete battery discharge followed by a complete recharging) to after the second eclipse season.

Attitude and orbit control subsystem (AOCS) The function and performance of the AOCS meets all the subsystem specifications. All the units are periodically checked and for some of them a regular maintenance campaign is performed, in particular: (i) noise and drift of the inertial measurement unit are measured every six months, and the drift value is compensated in the attitude control computer, (ii) the drift of the rate measurement unit is measured daily on the main unit and twice a year for the redundant one; the measured value is compensated on-board. Both these units show values well within the expected range. The reaction wheels are indirectly checked comparing the expected speed profile with the one provided in the telemetry; the good correlation of the two and the correspondence between the measured and the theoretical power demand indicate that the units are operating as expected. The star trackers are correctly operating showing full performance; however, the number of guide star losses is higher than expected. These events mainly occur during perigee passages and the only consequence is a more frequent intervention of the inertial measurement unit in the control, as foreseen by the autonomy design. Therefore this behaviour is no matter of any concern.

Reaction control subsystem All the components of the subsystem are nominal and perform as expected. The fuel consumption is better than expected as shown above. The stability of the reaction wheel off-loading manoeuvres indicates that no thruster degradation has occurred so far.

Thermal control subsystem After an initial threshold adjustment during commissioning phase, the thermal control subsystem performs nominally. Payload module temperatures are basically maintained by the lateral radiators. The substitution heaters, controlled by thermostats, do not intervene except when the units are switched-off (i.e. during eclipse passage). All units of the service module have been operational since few hours after spacecraft/upper-stage separation and the thermal control performed as expected keeping them within the predicted range. To verify the thermal mathematical model, two quasi-steady state cases have been identified and analysed. The results show in general good agreement and we conclude that the thermal control is working correctly without degradation of radiators and multi-layer insulation.

7. Conclusions

The INTEGRAL satellite was launched precisely on the second as planned at 4:41:00 UTC from Baikonur. The injection into operational orbit and the commissioning of the satellite were carried out as planned. The INTEGRAL satellite performs in general better than specified and no failure of any on-board units has happened. The performance continues to be optimised. The operational constraints are continuously under optimisation to increase the operational efficiency.

INTEGRAL has adequate on-board resources to ensure, that INTEGRAL will continue observations for the scientific community and hopefully make many unprecedented scientific discoveries in the field of high-energy astrophysics long beyond its 5 year operational lifetime.

INTEGRAL is the result of good design practices, solid engineering and thorough on-ground testing. Despite this, it has been a very cost efficient mission mainly due to the re-use of the service module from XMM-Newton, the participation from the international partners and last but not least an efficient project management.

Acknowledgements. The successful INTEGRAL project is the result of the great efforts and dedication of all the teams of the principal investigators, the industrial teams spread all over in Europe, and the ESA teams and individuals both at ESTEC and ESOC, who have contributed to making the satellite an outstanding observatory. In particular the team of the prime contractor at Alenia Spazio, Turin is acknowledged for their dedication, professionalism and excellent cooperation and the operation teams at ESOC, ISOC and ISDC for their dedication and efficiency in operating the INTEGRAL satellite for the benefit of the scientific community. It has truly been a privilege to work with so many competent people. The authors are also grateful to P. Bühler for supplying the comparison between IREM measurements and the NASA AE8/AP8 radiation belt models and to P. Favre for supplying the IREM data.

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