The INTEGRAL Spacecraft

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Abstract

The INTErnational Gamma Ray Astrophysics Laboratory (INTEGRAL) is being developed by ESA as the second medium-size satellite of the long-term scientific plan Horizon 2000. It is an observatory providing an excellent opportunity to the scientific community for detailed imaging and high-resolution spectroscopy of Gamma-ray sources.

After completion of the design and development phase the Structural Thermal Model of the Integral Spacecraft and its four scientific instruments is presently being tested. First tests took place in spring 1998 at the ESTEC test facilities. Late summer 1998 the Engineering Model test campaign will follow.

This paper summarises the key programmatic milestones of the project, highlight the final design characteristics of the satellite, report on the first test results, and describe the orbital scenario of the mission and the herewith-related requirements to the satellite.

1. Introduction

The INTEGRAL spacecraft is presently being built under ESA contract by a European industrial consortium led by Alenia Spazio S.p.A. of Turin, acting as prime contractor. The main industrial development phase (phase C/D) was kicked off in October 1996 after a successful Preliminary Design Review (PDR), which confirmed that the basic requirement of commonality with XMM, i.e. the re-use of the XMM Service Module, would be suitable to fulfil the INTEGRAL scientific mission objectives. Modifications required to accommodate INTEGRAL specific needs could be kept to a minimum, thereby reducing development risk and cost.

The Phase C/D schedule (see fig. 1) is dictated by the Launch date 1st April 2001, and is structured in three phases: STM, EM and FM programme, which will be addressed in chapter 3 of this paper.

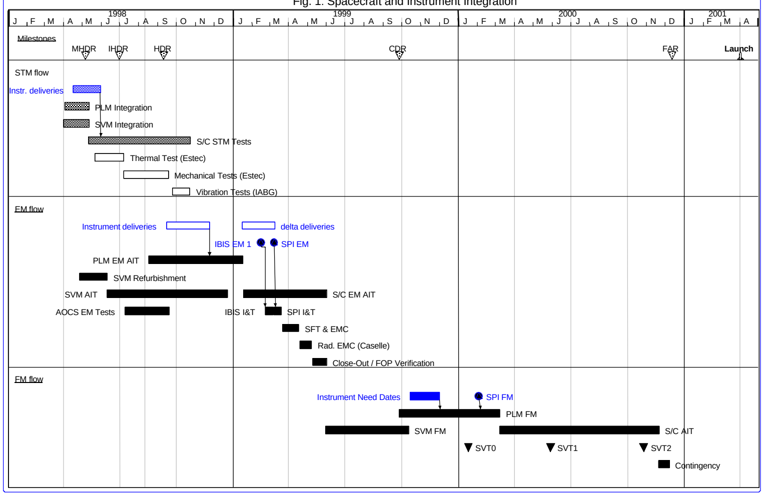
Within the INTEGRAL programme ESA is responsible for the overall spacecraft and mission design, spacecraft procurement, system integration and testing, mission planning (by ISOC) and control (by ESOC). The four scientific instruments are provided by individual scientific consortia led by a PI each and are nationally funded, with ESA contributions (Data Processing Electronics, Parts Procurement and Spectrometer cryo-cooler). Pre-processing and distribution of scientific data to the science community will be secured by the Integral Science Data Centre (ISDC), which is also nationally funded via a PI consortium. All consortia involved in the development of the instruments as well as the ISDC PI and the mission scientists will receive a guaranteed share of observing time.

Further contributions from Russia (procurement of PROTON launcher) and United States (use of NASA's Deep Space Ground Station Network) will be compensated with a guaranteed observation time for scientists of both countries. The remaining observation time (> 65 %) will be available to the general scientific community.

The Ground Segment follows a distributed architecture sharing tasks and responsibilities between three bodies:

- The Operations Ground Segment (OGS), consisting of the ESA and NASA ground stations and the Mission Operations Centre (MOC) at ESOC (see M.Schmidt, these Proceedings).
- The INTEGRAL Science Operations Centre (ISOC) (see P.Barr et al., these Proc.).
- The INTEGRAL Science Data Centre (ISDC) (see T.J.-L. Courvoisier, these Proc.).

INTEGRAL
Fig. 1: Spacecraft and Instrument Integration



2. Spacecraft

2.1 Key Design Requirements

- **Commonality with XMM:** The main programmatic goal of commonality with XMM is achieved by the re-use of the XMM Service Module including the use of recurrent XMM FM units. This approach and the agreement with Russia on the provision of a Proton launcher allowed to design a mission of the size of an ESA Cornerstone at the cost of a Medium Size Mission (in terms of the ESA Horizon 2000 Programme).

Modifications necessitated by the specific needs of the INTEGRAL mission had to be introduced mainly in the Attitude and Orbital Control System (AOCS) and Reaction Control System (RCS) areas, which are specifically affected by the different orbital and launcher scenarios. A new unit, the Rate Measurement Unit (RMU) was added to improve the AOCS capabilities for attitude failure and anomaly recovery.

Diaphragm tanks had to replace the original XMM tanks using an internal propellant management device, to account for the Proton Launch preparation scenario, where satellite and launcher are integrated and transported in horizontal orientation before launcher erection on the launch pad. The position and orientation of the thrusters has been changed to avoid disturbance torques due to unsymmetrical plume impingement on the PLM.

Further changes relate to different TM/TC structures, mission specific software and thermal control to assure a best possible adaptation to the mission needs.

- **Compatibility with Launchers:** Different static and dynamic fairing envelopes and the different mechanical environment of Proton (baseline) and Ariane 5 (backup) were taken into account in the definitions of the satellite configuration and of mechanical launch loads, both for the XMM recurring units and the newly developed instruments. This is presently under verification in the frame of the Structural Thermal Model test campaign.

- **Operational Orbits:** With respect to the system design the choice of the operational orbits (details see chapter 4) affected mainly the radiation protection. In the worst case, the Ariane case, the electronic units will be subjected to high radiation doses (see chapter 5) requiring radiation hardened parts and components.

- **Resources:** The mass situation is well under control and allowed recently to deplete part of the available margin by increasing the fuel mass by 50 kg and improving the passive shielding of IBIS in the collimator tube.

In contrast to this, the power budget remains highly critical and a reason for major concern. The actual instruments power demand exceeds the allocated power resources by 120 Watts, when the S/C reaches a Solar Aspect Angle (SAA) of 40°. ESA and Alenia are presently studying this problem, while the instrument teams are trying to optimise their electrical design. Another solution to this problem would be to reduce the SAA from 40° to 30° already from the beginning of the mission and not only after 2 years, resulting in a degraded sky coverage.

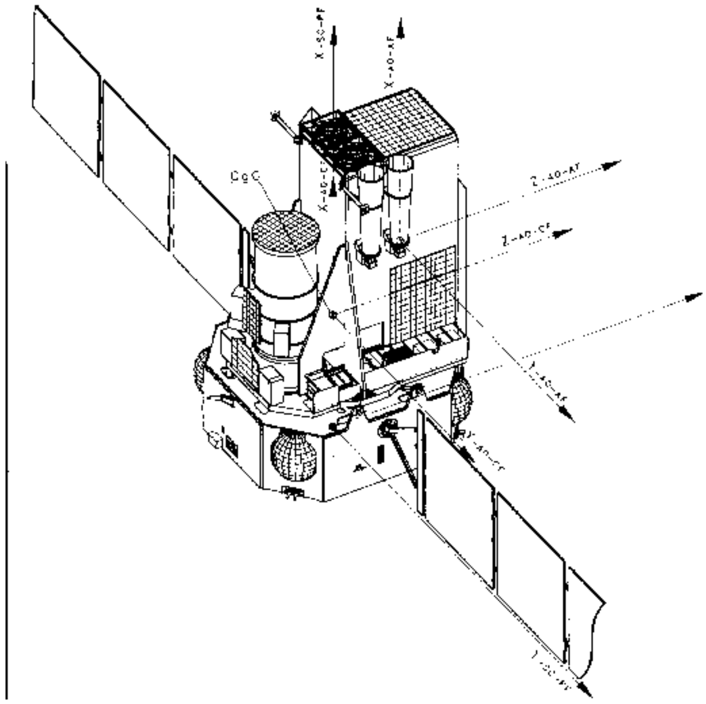


Fig. 2: The INTEGRAL S/C (schematics)

- **Pointing and Alignment:** The INTEGRAL S/C pointing and attitude requirements are less stringent than those on XMM, and preliminary analyses show that they can be fulfilled up to the end of the extended mission without relying on gyroscope information.

The instrument alignment requirements are listed in the summary table below. Actual values could not be included in this paper since the system distortion analysis is still ongoing. However, preliminary assessments give confidence that the actual performance will be well within the requirements.

POINTING and ATTITUDE DOMAIN			
	Y,Z	X	Over
Absolute Pointing Error (APE)	5'	15'	
Relative Pointing Error (RPE)	0.3'	1'	10e3s
Absolute Pointing Drift (APD)	0.6'	2'	10e5s
Absolute Meas. Accuracy (AMA)	1'	3'	
INSTRUMENT ALIGNMENT			
Overall Misalignment Error	1'	3'	
Misalignment Variation	0.1'	0.3'	10e3s
Misalignment Variation	0.3'	1'	10e5s
Error of a-posteriori determination of misalignment	1'	3'	

- **Autonomy and Ground Outage:** The mission is designed to be a real-time mission under continuous ground control. The spacecraft design takes into consideration that short term reactions (in less than 3 min.) have to be excluded and medium term reactions (in less than 30 min.) are to be minimised. The satellite is designed to cope with 36 hours of ground outage. On payload side this capability needs to be confirmed for the two major instruments.

- **Instrument accommodation:** The spacecraft design is driven by the gamma ray and x-ray instruments (SPI, IBIS, Jem-X) which are all based on the coded mask principle. Here a coded mask is positioned at a large distance (~ 3.5 m) from a position sensitive detector. To achieve and maintain the required relative position accuracy between each detector and its mask as well as the Instrument Line of Sight (ILS) w.r.t. the satellite Star tracker reference axis is a challenging engineering task. It requires a high degree of alignment and dimensional stability of the supporting structure under all environmental conditions encountered from integration to the end of the mission.

Field-of-View and mass distribution constraints had to be taken into account in the accommodation of all instruments, especially of SPI, weighing alone 1321 kg. Interferences of the S/C with PROTON fairing envelope were solved among other by smoothing of IBIS mask corners allowing marginal accommodation.

The layout of the payload module (PLM) has been conceived in terms of a separate science-payload module containing the instruments. The PLM is integrated and tested independently from the SVM and later mated with the SVM for satellite level activities. The interface to the SVM has been designed to be as simple as possible to reduce complexity, time and cost.

2.2 Payload complement

The following table summarises the key P/L parameters flown on INTEGRAL. The information given here is limited to a brief overview, while detailed information can be obtained from other papers provided by the individual instrument teams and contained in these proceedings.

Payload:

- Main Instruments:
  - SPI: Germanium spectrometer
  - IBIS: Cadmium Telluride / Caesium Iodide imager
- Monitors:
  - JEM-X: X-ray monitor
  - OMC: Optical monitoring camera
- Characteristics:
  - Energy Range
    IBIS: 15 keV-10MeV; SPI: 20 keV-8MeV; JEM-X: 3 keV-35keV; OMC: 500-850 nm;
  - Field of view (fully coded)
    IBIS: 9°, SPI: 16°, JEM-X: 4.8°, OMC: 5°
  - Angular resolution (FWHM)
    IBIS: 12', SPI: 2°, JEM-X: 3', OMC: 17.6"
  - Spectral resolution (E/E)
    IBIS: < 7% @ 100 keV; 6% @ 1 MeV
    SPI: 0.2% @ 1 MeV
    JEM-X: < 47% (E/1 keV)$^{-0.5}$
  - Source location
    IBIS: 1', SPI: <30', JEM-X:<30", OMC: ~8"
  - Continuum sensitivity
    IBIS: $2x10^{-7}$ ph/(s cm² keV) @ 1MeV in $3x10^{6}$ s
    JEM-X:
    $9x10^{-6}$ ph/(s cm² keV) (@ 6 keV
  - Narrow line sensitivity
    SPI: $3x10^{-6}$ ph/(s cm²) in$3x10^{6}$s @ 1MeV
    $2x10^{-5}$ ph/(s cm²) in$3x10^{6}$s for 511 keV-line

3. Test Philosophy and first test result

The actual industrial development schedule is set up to meet the 1st April 2001 as a nominal Launch date and is structured according to the model philosophy required by the INTEGRAL programme and its relationship with the XMM programme. In particular the verification philosophy will benefit from the test campaign completed on the SVM of the XMM spacecraft by re-using its STM and EM.

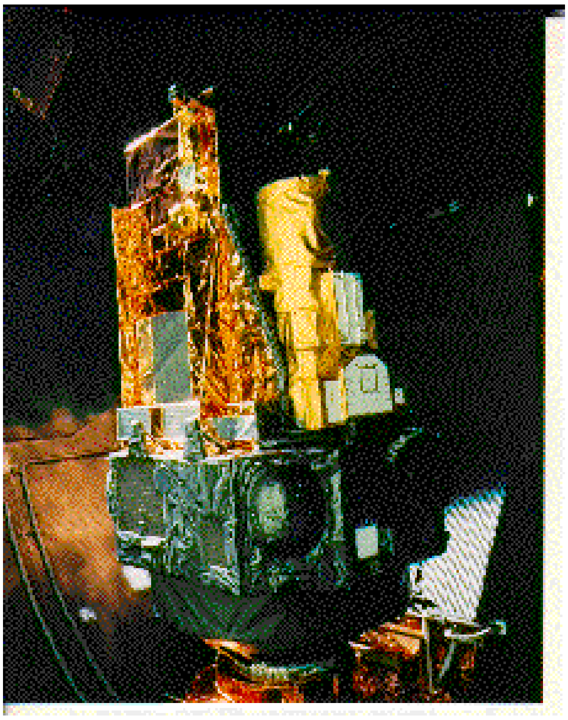


Fig.3: The INTEGRAL STM in the ESTEC Solar Simulator Chamber

3.1 Structural Thermal Model (STM) Programme:

The goal of the STM programme is to qualify the spacecraft by test against the mechanical and thermal environment with special emphasis on the newly developed Payload Module Structure (PLM), since it is not a recurring item from XMM and vital for the alignment and imaging capabilities of the instruments.

The STM test programme started in spring 1998. As of today, the Thermal Balance Test (TBT) has been successfully completed and Modal Survey Test will be completed by the end of August 1998. The other tests will follow in autumn 1998.

The preliminary analysis of data gathered during the TBT indicates a good agreement between measured and predicted temperatures, instruments included. The correctness of the overall satellite thermal control is therefore confirmed. Final results will be available in November 1998.

3.2 Engineering Model (EM) Programme:

The EM programme started in June 1998 with the hand-over from the SVM EM from XMM and will be in full swing in autumn 1998 when the instruments EMs become available. It aims to verify all electrical and software interfaces as well as electro-magnetic compatibility (EMC). The sub-system and instrument performances within the system environment and the verification of the operational procedures are key goals of this phase.

3.3 Flight Model (FM) Programme

After completion of the STM and EM programmes the integration of the Flight Model will start. During this phase final qualification and acceptance of the spacecraft and instruments will take place and therefore parts of tests already performed during the STM and EM test campaigns will be repeated at acceptance test levels. They will start spring/summer 1999.

4. INTEGRAL Orbits

4.1 Requirements

The INTEGRAL orbit scenario has to comply with requirements and constraints dictated not only by science objectives but also by ground segment configuration, satellite and launcher capabilities.

In order to allow undisturbed scientific measurements and guarantee maximum science return, it is required to optimise the time spent outside the Earth's Radiation Belts (proton and electron belts). In the equatorial plane the proton (electron) belts are typically assumed to extend out to a geocentric radius of about 4 R_E (10 R_E), with the maximum flux of protons with $E_p > 10$ MeV ($E_e >$ 1 MeV) occurring at 2 R_E (4 R_E) [Ref.: E.J.Daly, 1994]. Measurements from SIGMA, analysed by M.Vargas at ISDC [Ref.: M.Vargas, 1998], show however that the influence of the radiation belts on gamma ray instruments extends up to 60000 km. As a starting point for an adequate INTEGRAL orbit evolution analysis a minimum perigee height of 7000 km, maximum time above 60000 km and an inclination greater than 50° were selected as orbital constraints.

The real-time nature of the INTEGRAL mission requires full ground station coverage of the operational orbit above 40000 km with maximum use of available coverage below. In addition scientific observations shall be possible up to the end of the Extended Mission and hence the orbit shall be stable for 5.2 years following the Launch, restricting basically the perigee height evolution.

The requirement for maximum visibility from ESA's European ground stations imposes high inclination and an apogee position in the Northern Hemisphere.

For technical and operational (less station hand-overs) but also for cost reasons, the number of needed ground stations shall be minimised, while for critical operations (like orbital manoeuvres) simultaneous coverage from two stations is required. The orbital period shall be a multiple of 24 hours to keep an optimal coverage pattern for all revolutions and to allow repetitive working shifts on ground.

The satellite requirements on the orbital scenarios are dictated by power, thermal and operational considerations. In order to guarantee sufficient power throughout the mission, the Solar Aspect Angle has to be constrained to within 40° from the S/C axis during the nominal mission. The maximum duration of (umbra plus penumbra) eclipses shall not exceed 1.8 hours for thermal and energy reasons. Further on, there shall be no eclipse from separation of INTEGRAL from the launcher up to the first apogee, and the perigee shall be raised by a total delta-v of not more than 223 m/s through three individual manoeuvres.

Regarding launcher requirements, INTEGRAL is to be compliant with PROTON (baseline) and ARIANE 5 (backup) launchers. Hence two different orbit scenarios result.

4.2 PROTON Orbit Scenario

In the baseline case, INTEGRAL is launched by PROTON from Baikonour into a 51.6° inclined transfer orbit with a perigee altitude of about 700-km, and with argument of perigee (300°) and apogee altitude (153000 km) of the operational orbit. Satellite separation from the PROTON Upper Stage will occur shortly after perigee. Injection from the transfer to the operational orbit will be performed by INTEGRAL's own propulsion system to reach the initial perigee altitude of 10000 km of the operational 72 hours orbit.

The INTEGRAL Launch Window has been constructed on the basis of the requirements mentioned previously. The strongest constraint is that of avoiding eclipses longer than 1.8 hours, followed by the Solar Aspect Angle (SAA) constraint, whereas the stability constraint of perigee heights never below 7000 km is relatively weak. The constraint of no eclipses from separation to first apogee is completely masked by the SAA constraint.

Assuming a required minimum daily launch slot of 30 min, the window is open throughout the year, except for 9 days in August. An increase of the required minimum perigee height is possible, however at the expense of a reduced launch window.

A reference orbit providing optimum coverage from the ground stations at Redu and Goldstone has been defined for the beginning of the launch window in April 2001. Its initial osculating parameters in mean Earth 2000 equator are:

Epoch:	2001-04-01 @ 19:28:20 UT
Perigee height:	10000.0 km
Apogee height:	152669.0 km
Inclination:	51.6°
R.A. of asc. node:	79.4°
Argument of perigee:	300.0°
True anomaly:	0.0°

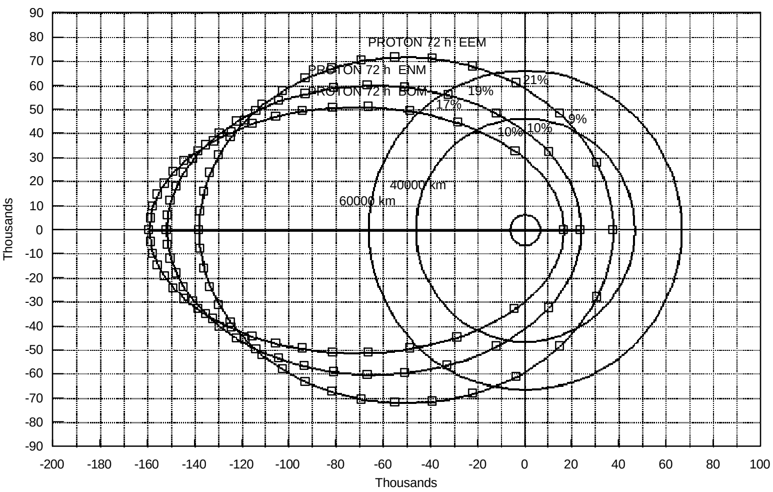


Fig.4: PROTON 72 h Orbit at BOM, ENM (2.2 years) and EEM (5.2 years) for the Reference Orbit case
[tick marks are shown every 2 hours]

For this reference orbit, Fig.4 shows the orbit ellipses at Begin of Mission (BOM), End of Nominal Mission (ENM, 2.2 years), and End of Extended Mission (EEM, 5.2 years) in relation to the radiation relevant altitudes of 40000 and 60000 km. The perigee height increases throughout the mission from the initial 10000 km to about 31000 km after 5.2 years. In addition to the orbit shape, also the inclination changes drastically over the mission from the initial 51.6° to 74.0° at ENM and 86.5° at EEM. This has an influence on the percentages of time above 60000 km and below 40000 km. Both decrease slightly over the mission, as the table below shows.

% of the orbital period (72 hrs)	Time above 60000 km	Time below 40000 km
BOM	83%	10%
ENM	81%	10%
EEM	79%	9%

The orbit analysis showed that the longest eclipse will not exceed 1.7 hours. All eclipses will occur at altitudes below 40000 km, either shortly before perigee (in summer) or shortly after (in winter).

Ground station coverage of the orbit arc above 40000 km is achievable by the combined use of Redu (ESA station) and Goldstone (NASA DSN station). The schematics in figures 4 and 5 indicate the individual visibility arcs of the orbit from these stations, at BOM and ENM respectively, together with the eclipse regions. Simultaneous visibility from Redu and Goldstone exists during a large part of the orbit.

The lack of visibility for about 4 to 5 hours around perigee is however limiting the monitoring capabilities of summer eclipses to their beginning (Goldstone). Worst case scenarios combining loss of attitude during eclipse and no ground coverage thereafter is being studied and will be taken into account in the S/C design and operational scenario.

Concerning station hand-overs it must be noted that the second hand-over (Goldstone to Redu near 45 hours) occurs at elevations close to 5° at both stations; investigations are needed to overcome related difficulties with commanding during the hand-over period.

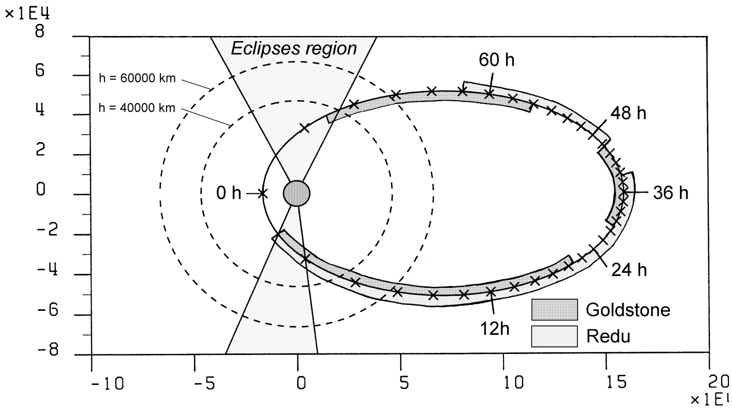


Fig.5: Coverage from Redu and Goldstone and eclipse regions at BOM, PROTON 72 h orbit, reference orbit

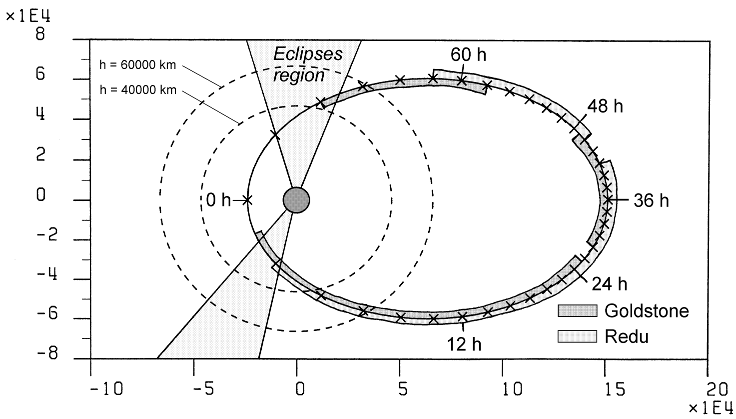


Fig.6: Coverage from Redu and Goldstone and eclipse regions at ENM, PROTON 72 h orbit, reference orbit

4.2 ARIANE 5 Orbit Scenario

In the backup case, INTEGRAL is launched by an ARIANE 5 from Kourou, applying a strategy similar to the one of the PROTON case. The transfer orbit has an inclination of 65° with a perigee altitude of about 200-km, and with argument of perigee (280°) and apogee altitude (114000 km) of the 48 hours operational orbit. The initial perigee altitude is around 7000 km.

The ARIANE 5 launch window is open all year except from May 16 to June 15 (31 days) and from November 12 to December 15 (34 days), assuming a required minimum daily launch slot of 30 min.

A reference orbit at beginning of the window in April 2001 has been defined. Its initial parameters are:

Epoch:	2001-04-1 @ 22:53:32 UT
Perigee height:	7000 km
Apogee height:	114000 km
Inclination:	65°
R.A. of asc. node:	241.1°
Argument of perigee:	280°
True anomaly:	0°

The following figure 6 shows the orbit ellipses at BOM, ENM and EEM in relation to the altitudes of 40000 and 60000 km. In this reference orbit scenario the perigee height increases during the first four years of the mission from the original 7000 km up to a maximum of 19000 km and then decreases to about 17400 km at EEM. This is the reason why the ellipses shown for ENM and EEM are very similar.

At the beginning, about 75% of the time are spent above an altitude of 60000 km, and 14% below 40000 km. The former percentage decreases during the mission and the latter increases by a few percent points. Overall, the percentages of time spent above the two key altitudes is of course inferior to that of the PROTON baseline scenario due to the different periods of orbits with roughly the same eccentricity of around 0.8. Contrary to the PROTON case above, the inclination varies only slightly during the mission between 65.3 and 64.1°.

In the ARIANE 5 case the longest eclipse duration is around 1.5 hours. All eclipses occur close to the perigee, summer eclipses within 0.5 and 4 hours after, winter eclipses about 2 to 4 hours before perigee. Almost all eclipses are confined within altitudes below 40000 km.

Ground station coverage of the complete orbit arc above 40000 km is achievable by Redu alone during most of the mission. Also in this case, no visibility is available around perigee for about 4 to 5 hours, increasing to about 6 hours towards EEM.

Throughout the mission simultaneous visibility from Redu and Goldstone exists for more than 24 hours accumulated over one revolution. None of the summer eclipses can be monitored from ground, winter eclipses occur generally under ground coverage.

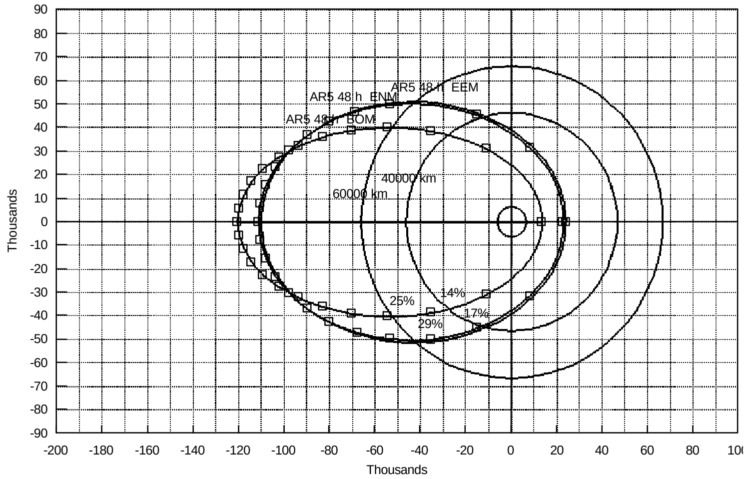


Fig.7: ARIANE 5 - 48h Orbit at BOM, ENM (2.2 years) and EEM (5.2 years) for the Reference Orbit case
[tick marks are shown every 2 hours]

5. Radiation Analysis

As the low altitude proton belt is the most harmful for electronic components and detectors, it is obvious that the Ariane orbit with the lower starting perigee and the relatively slow raise will remain the design criterion as far as parts selection and radiation protection is concerned. The expected total radiation dose for the reference orbit is about 50 kRad.

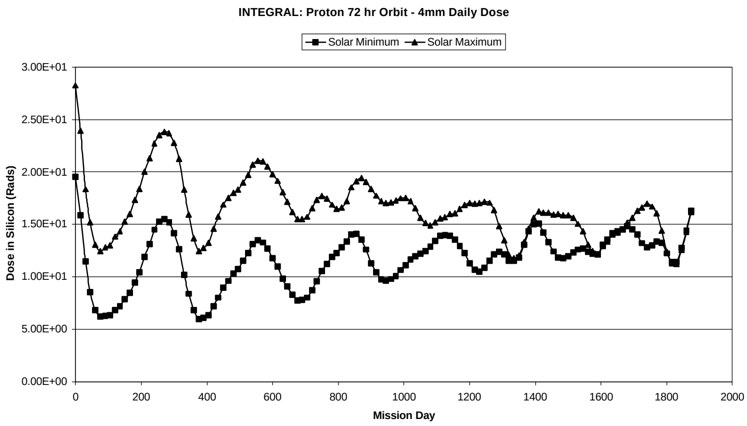
For the baseline (PROTON) orbit, a systematic analysis was started: The orbits with the highest perigee heights were calculated as function of launch date and hour. The orbit with the lowest of these maximum height perigees, occurring within the allowed launch window, was used as input for the radiation calculation. Its initial parameters are:

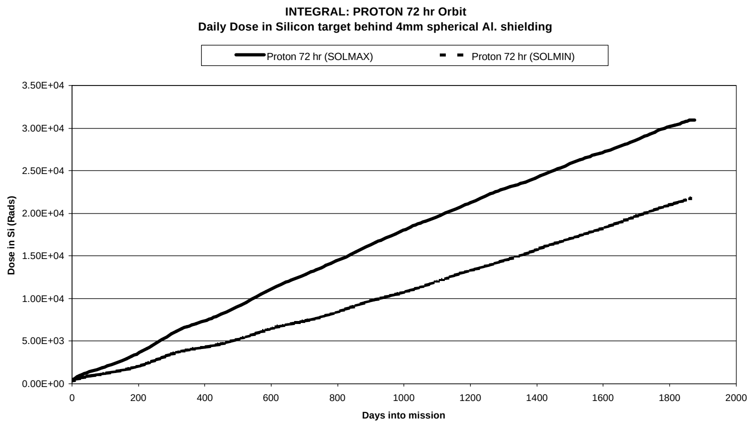
Epoch:	2001-10-04 @ 05:24:00 UT
Perigee height:	10000 km
Apogee height:	153000 km
Inclination:	51.6°
R.A. of asc. node:	103.4°
Argument of perigee:	300°
True anomaly:	0°

The result is reported in the following diagrams showing the daily radiation dose and the accumulated total dose as function of day in orbit. The solar activity has been kept constant for both curves defining the best and worst case situation.

The calculated radiation doses are dominated by the electron contribution, which is more significant at higher altitudes (perigee) and less affected by the atmospheric absorption. It is however highly depended from the fluctuation in the Earth magnetic field and therefore higher during solar maximum.

Fig.8/9: Daily and Accumulated Radiation Dose for Proton Orbit





6. Conclusion

Commonality between satellites, i.e. the re-use of a common "bus', is standard practice in commercial telecommunication satellites, where typically a large number of satellites of the same make is required to fulfil adequate coverage. With XMM and INTEGRAL, this approach was successfully applied also for scientific satellites reducing development risk and costs. The development of completely new instruments and their qualification remains the real challenge and critical issue to be carefully monitored in the upcoming months.

Acknowledgements

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