The Envisat Satellite and Its Integration

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For the early years of this new millennium, Envisat is ESA's major contribution to the study of the Earth as a system. Carrying ten sophisticated instruments – both optical and radar – it is the largest and most complex satellite ever built in Europe. It has been designed and tested over a period of more than 10 years. Much of the integration and test programme was conducted on site at ESTEC, in Noordwijk (NL). It will be the first satellite launched into a polar orbit by Ariane-5.

This article summarises the design and engineering of Envisat, and explains the model philosophy and test approaches used. The launch campaign plans are also briefly described.

Introduction

Envisat is the largest and most complex free-flying satellite ever built in Europe. It will carry a comprehensive series of instruments designed to observe a whole series of interrelated phenomena that characterise the behaviour of the Earth's environment as a system. The satellite, together with its related ground systems, will continue and extend the data services provided by the Agency's earlier ERS-1 and ERS-2 satellites. In particular, Envisat should substantially increase our knowledge of the factors determining our environment. It will make a significant contribution to environmental studies, notably in the areas of atmospheric chemistry and ocean studies, including marine biology.

The satellite makes use of the multi-mission capability of the Polar Platform originated in the Agency's Columbus Programme. This development also forms the basis for MetOp. The Polar Platform in turn has drawn heavily on the equipment and technologies developed within the framework of the French Spot programme. Almost all of the instruments on the satellite have been specifically developed for Envisat, with one or two having a strong design heritage from ERS.

The observations made by Envisat will eventually be continued and extended by a series of new, smaller satellite programmes being initiated within the Agency's Earth Observation Envelope and Earth Watch Programmes.

Background

The Columbus Programme approved at the ESA Ministerial Council Meeting in The Hague in 1987 included the development of a multi-mission Polar Platform as part of the International Space Station. Following a series of studies and iterations with potential users, an implementation re-using the equipment and architecture of the Spot-4 spacecraft bus design, although with a significantly enlarged structure, was decided upon. The main development phase (Phase-C/D) for the Polar Platform programme was awarded to British Aerospace in Bristol (UK) – later to become Matra Marconi and now Astrium Ltd. – in late 1990.

Meanwhile, in the Earth Observation area, the Agency was considering, as ERS-1 grew closer to launch, how to continue and extend the services offered. In 1988, these elements were drawn together in an ESA proposal to its Member States for an overall 'Strategy for Earth Observation'. These considerations led to the adoption of the POEM-1 programme, using the Polar Platform, at the Ministerial Council Meeting in Munich in November 1991. There continued to be an evolution in the payload complement for POEM-1. This culminated in a splitting of the payload into separate Envisat and MetOp satellites, which was finally agreed at the next Ministerial Council in Granada in November 1992. A Phase-C/D contract for the procurement and support of the Envisat payload (the so-called 'Mission Prime Contract') was awarded to Dornier Satellitensystem, now Astrium GmbH, in July 1992.

As a result of the programmatic origin of the satellite, there remain two large contracts for its implementation:
– the Polar Platform Prime Contract (Astrium Ltd.), and
– the Mission Prime Contract (Astrium GmbH).

These two large contracts, interfacing with each other at some of the technically most critical on-board locations, caused a number of problems during the development programme. The satellite integration programme has, however, largely been carried out at ESTEC following the closure of Astrium's Bristol site. As a result, many of the technical personnel have been collocated (with ESA) at ESTEC. This, and the grouping of both contractors within the Astrium company, has ensured a much smoother technical path for the programme in its final phase.

The organisation of the Agency's project teams initially also reflected the programmatic split, with separate project divisions for Polar Platform and payload. More recently the project teams have been merged within a single division. This too has simplified the technical conduct of the programme.

Major capabilities
The satellite is designed for a Sun-synchronous polar orbit (Table 1). The planned operating altitude is 800 km, although a range of altitudes can be selected allowing variations in the repeat cycle of the ground track. The local time at the equator for the descending node has been selected as 10.00 a.m., which optimises illumination conditions for part of the optical payload.

The selected orbit has a repeat cycle of 35 days and an orbital period of 100.6 min. Its inclination is 98.54 deg, which implies small, uncovered areas at the poles for instruments with limited swath widths. The on-board systems allow the ground track to be maintained within 1 km and the local hour to within 5 min. One of the on-board instruments (DORIS), when used in conjunction with a dedicated set of ground stations, provides real-time knowledge of position to within 50 cm, and a precision altitude restitution to within 5 cm.

In nominal operations, the satellite is pointed using star trackers in a 'stellar yaw-steering mode'. In this mode, the satellite is yawed to compensate for the apparent motion of the Earth across-track beneath the satellite. This compensation simplifies the processing of Doppler signals from the synthetic-aperture radar. When using the star trackers, random pointing errors will be less than 0.0085 deg, and the stability over all periods of up to 170 s better than 0.015 deg. Attitude estimation will be better than 0.04 deg. This pointing performance allows both adequate geographical location for data measured on the Earth's surface, and a vertical resolution when viewing the atmosphere at the limb of better than 3 km.

The satellite provides an average of 1900 W for instrument operations through sunlit and eclipse portions of the orbit. This enables all instruments except MERIS and ASAR to be operated continuously throughout the entire orbit. MERIS, the Medium-Resolution Imaging Spectrometer, requires sunlight to operate. ASAR, the Advanced Synthetic-Aperture Radar, produces such enormous quantities of data in its high-resolution mode that its operation is limited to 30 min per orbit.

Table 1. Major capabilities of Envisat

- Sun-synchronous orbit: 800 km, 10 a.m. descending node, 35-day repeat cycle
- Stellar yaw steering, for accurate pointing and Doppler compensation of SAR
- 1900 Watts, 2500 kg for instruments
- Data recovery at up to 100 Mbps direct via X-band or via Artemis
- On-board storage in solid-state recorders for regional and global missions
- S-band command and control; 2 kbps uplink, 4 kbps downlink

The data-handling capabilities of the spacecraft have also been sized to support the global and regional missions. All instruments except MERIS and ASAR operate continuously, together producing 4.6 Mbps. There are separate additional regional missions for MERIS (up to 25 Mbps) and ASAR (up to 100 Mbps). On-board storage in redundant solid-state recorders allows the recording and dumping of all data from the global and regional missions. Data can be downlinked directly when overflying a suitable ground station, such as Kiruna in Sweden, via a fixed X-band antenna, or when within visibility of Artemis, via a steerable Ka-band antenna.

Command and control of the satellite and payload is via an S-band transponder. The satellite will normally be operated by uplinking a 24-hour command timeline. Housekeeping data is available in real time when over a ground station, but is also included in the global mission data stream.

The satellite is designed for launch only by Ariane-5. It has a total launch mass of 8100 kg, of which 2150 kg are instruments. The physical size of the spacecraft requires the Ariane-5 long fairing, for which Envisat is the first customer.

Major components

The satellite is made up of two major subassemblies, the Service Module (SM) and the Payload Module (PLM), with a simple structural, electrical and avionics interface between them. All instruments are physically located on the PLM, to which the instruments have a largely standardised interface. The modularity of the design has made it possible to conduct by far the majority of the integration work on the SM, PLM and instruments in parallel.

Service Module

The SM provides the standard satellite support functions, and was subcontracted to Astrium SAS. It is based on the design of the Spot Mk-II service module, but with a number of important new developments, particularly in the structure and solar-array areas (Fig. 1).

Figure 1. The Envisat Service Module being readied for shock testing in the ESTEC facilities

The SM includes eight batteries, and the solar array. This is a flat-pack array designed for the Polar Platform by Fokker Space (NL), based on their standard design elements. Once deployed, the array is rotated to point continuously towards the Sun using a solar-array drive mechanism, which is attached to the base of the central cone.

The propulsion module on top of the cone contains four tanks, which hold 300 kg of hydrazine.

A single central computer containing both command and control and AOCS functions performs on-board data management. It controls the SM equipment via a standard on-board data-handling bus. The central computer also communicates with the central computer of the PLM via the same bus.

Payload Module

The PLM (Fig. 2) provides the physical accommodation for the instruments, as well as a range of instrument-related services, such as power switching, integrated payload command and control, data storage and downlinking. In addition, some equipment that would logically be part of the Service Module is physically located on the PLM for performance reasons. Such equipment includes the star sensors, the gyroscopes, and some of the thrusters.

Within the Earth-pointing and side compartments of the PLM is the Payload Equipment Bay (PEB), which contains both payload-support and instrument equipment. Astrium GmbH is the subcontractor responsible for the provision of the PEB equipment for the Polar Platform, and the design of this subassembly is based on their experience with ERS-1 and ERS-2.

Measurement data generated by the instruments is processed by a High-Speed Multiplexer (HSM) for recording or transmission direct to ground. Low-rate data (0 – 32 Mbps) are routed from the instruments to the HSM to create data streams that are passed to the recorders or to the Encoding and Switching Unit (ESU) prior to transmission. The high-rate (100 Mbps) data from the ASAR in Image Mode is introduced directly to the ESU and modulates the two 50 Mbps channels of the downlink.

Each of the two redundant Solid-State Recorders can be flexibly used to simultaneously store data from the global composite at 4.6 Mbps and from the regional missions. The communications subsystem for the transmission of instrument data or housekeeping telemetry comprises an X-band link direct to ground and a unidirectional Ka-band link via Artemis.

Figure 2. The Envisat Payload Module on the payload integration stand at ESTEC

Instruments
The roles and functions of each of the Envisat instruments are described elsewhere in this issue of the Bulletin, but some information on their accommodation is relevant to an understanding of the satellite as a whole.

There are a total of ten instruments embarked on the satellite, although one of them, the Laser Retro Reflector, is entirely passive. Two others, DORIS and the Microwave Radiometer, share a single Instrument Control Unit and are operated as though they were a single instrument. Tables 2 and 3 list the instruments, together with their primary mass and power requirements. Figure 3 shows how the instruments are accommodated on the deployed spacecraft.

Instrument procurement was via one of two routes. Instrument sub-contractors, working for the Mission Prime Contractor, designed and built the so-called ‘ESA-Designed Instruments’ (EDIs). A further three instruments were provided directly by national funding agencies

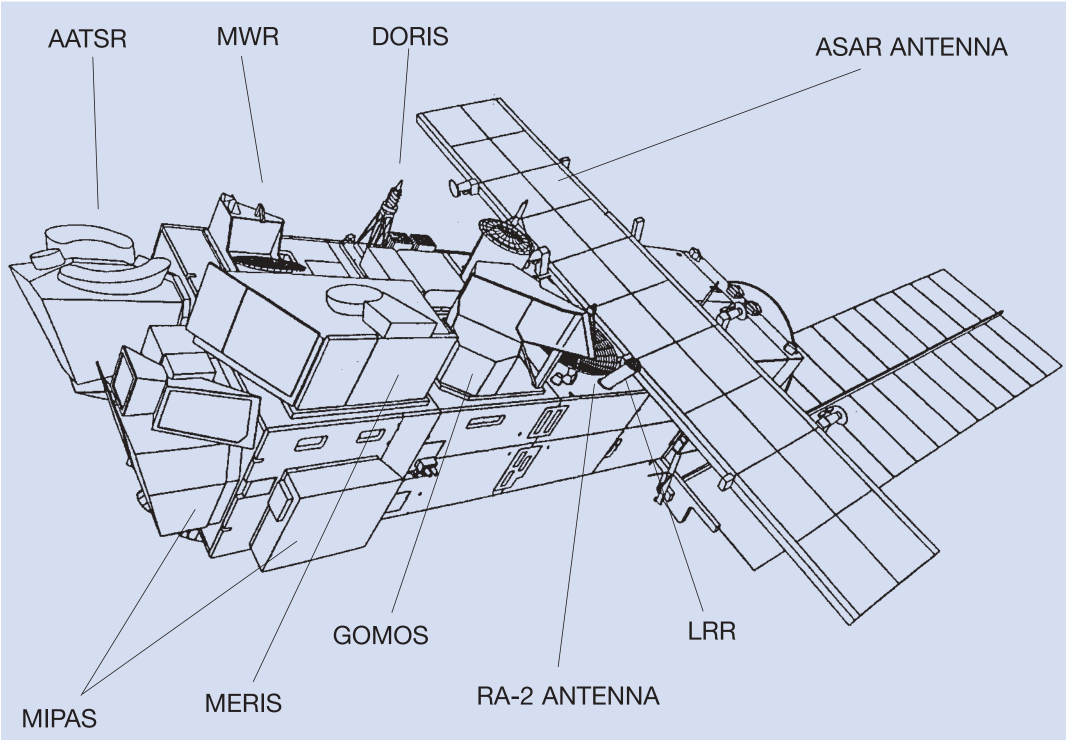
Table 2. ESA-developed instruments

ASAR	Advanced Synthetic-Aperture Radar	817 kg	1400 W (max)
RA-2	Radar Altimeter	110 kg	118 W
MWR	Microwave Radiometer	24 kg	22 W
GOMOS	Global Ozone Monitoring by Occultation of Stars	164 kg	157 W
MIPAS	Michelson Interferometer for Passive Atmospheric Sounding	327 kg	196 W
MERIS	Medium-Resolution Imaging Spectrometer	209 kg	110 W
LR	Laser Retro Reflector	2 kg	Passive

Table 3. Announcement of Opportunity instruments

SCIAMACHY	Scanning Imaging Absorption Spectrometer for Atmospheric Cartography	201 kg	121 W
DORIS	Doppler Orbitography and Radio-positioning Integrated by Satellite	85 kg	34 W
AATSR	Advanced Along-Track Scanning Radiometer	108 kg	94 W

Figure 3. The flight configuration of the Envisat instruments; the SCIAMACHY instrument is behind AATSR

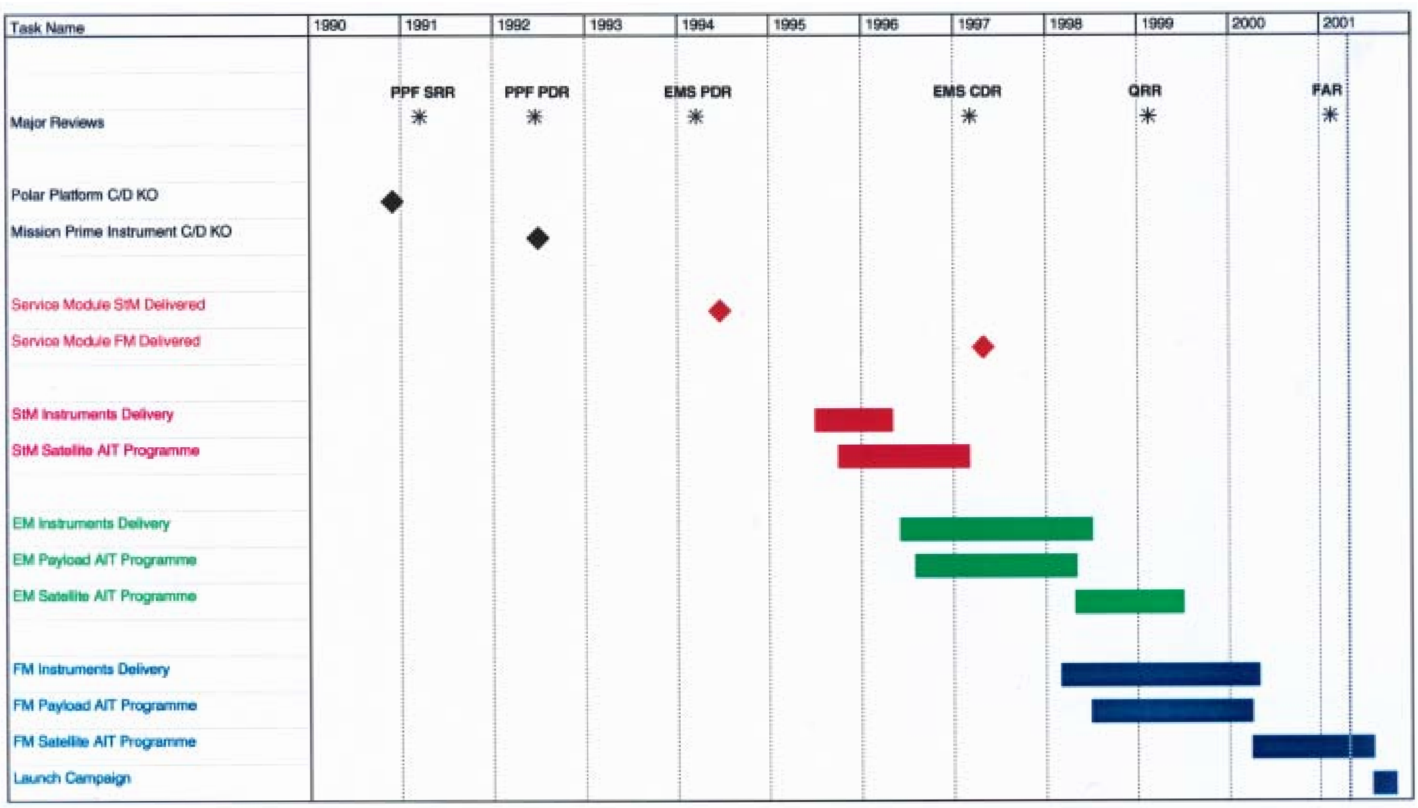


for integration with the Polar Platform. These are the so-called ‘Announcement of Opportunity Instruments’ (AOIs). The Mission Prime Contractor provided engineering and integration support for the AOIs.

With three exceptions, instruments are mounted entirely on the outside of the payload structure. This allows them maximum flexibility in terms of fields of view, both for observation and for thermal reasons. Externally mounted instruments are thermally decoupled from the Payload Module. Some units of RA-2, GOMOS and ASAR are mounted inside the PEB and share its thermal control.

Each instrument has a dedicated Instrument Control Unit (ICU) that interfaces to the Payload Module Computer (PMC) via a standardised avionics interface. The ICU is responsible for executing the macro commands sent from the PMC as their time tags become due.

Figure 4. The major spacecraft campaigns



As a general principle, instruments were integrated, qualified and acceptance tested as a unit before delivery for integration with the Polar Platform.

Development programme
Model philosophy
The general philosophy for the satellite has been to use three models:
– A structural model for mechanical qualification, including static-load, modal-survey, shock, acoustic and sine testing.
– An engineering model for electrical compatibility testing, for preliminary operability demonstration, for debugging ground-support equipment, and for a preliminary demonstration of RF compatibility.
– A flight model for thermal-balance and thermal-vacuum testing, for acceptance acoustic and sine and qualification shock testing, for RF-compatibility acceptance and for conclusive demonstration of operability.

The duration of each of the major campaigns and their relationship to the programme reviews is shown in Figure 4.

This simple model philosophy was modified in significant ways as a result of the historical development of the programme. Table 4 lists the models built of each of the major satellite items.

An RF mock-up of the entire spacecraft was built at a very early stage in the programme to allow the measurement of likely coupling factors between transmitting and receiving antennas.

Since the electrical design of the Service Module was so similar to that of Spot, only two models of the Service Module were made: a structural model and the flight model.

For the solar array (Fig. 5), there was only one complete model, the flight model. In order to avoid overstressing, it was accepted together with the flight-model spacecraft. A single panel took part in the structural-model satellite campaigns. The majority of qualification was carried out at component level.

The instrument development programmes began nearly two years after the Polar Platform development programme. Consequently, a number of instrument suppliers had difficulty meeting need dates for full engineering-model instruments. In these cases simplified

Figure 5. The solar array under test at Fokker (NL)

Table 4. Component model characteristics

Component	Heritage	StM	EM	FM	Remarks
Service Module	Spot avionics New mechanics	Yes	No	Yes	EM Satellite programme with FM service Module
Payload Carrier	New	Yes	Form and fit only	Yes	FM refurbished STM structure
Payload Equipment Bay	New (ERS)	Mass Dummies	Yes	Yes	
Solar Array	Fokker product	One panel only	No	Yes	
Ka-band Assembly	New	Partly	Yes	Yes	

Table 5. Instrument-model characteristics

Instrument	Development Category	StM	EM	FM	Remark
ASAR	New	Antenna	Reduced EM	FM	4 active tiles for EM
GOMOS	New	No	Full EM	FM	Reduced EM to serve spacecraft EM
MERIS	New	Yes	Reduced EM	FM	1 out of 5 cameras (EM)
MIPAS	New	Yes	Full EM	FM	Reduced EM to serve spacecraft EM
MWR	Rebuild	No	No EM	FM	ERS-based instrument
RA-2	Partially new	No	Full EM	FM	ERS-based instrument
LR	Rebuild	No	No EM	FM	ERS-based passive optical system
AATSR	Partially new	No	Reduced EM	FM	ERS-based instrument
DORIS	Rebuild	No	Full EM	FM	EM available for spacecraft RFC test
SCIAMACHY	New	Yes	Reduced EM	FM	Reduced EM to serve spacecraft EM

instrument models were supplied sufficient for the real needs of the engineering-model satellite programme. In a number of cases, this allowed the instrument supplier to continue development work on a full engineering-model instrument, to the eventual benefit of the flight model. Table 5 lists the development models for each of the instruments.

The optics modules of SCIAMACHY and MIPAS were replaced during the flight-model thermal test with thermally representative models.

Satellite software development and validation approach

Envisat carries some 40 different real-time processors and a wide range of the satellite's functions are software-implemented: commanding and monitoring of SM, PEB and payload-instrument equipment, payload science data-handling functions, onboard time distribution and datation, and scientific data processing (in the case of the AOCS and of several payload instruments).

In general, these computers have used Ada as the programming language (e.g. the for Payload Module Computer), or Assembler when tight memory resources forced the designer to optimise the size of the code for a given function (e.g. for the Service Module Central Communication Unit). Most processors belong to the 1750 family (standard instruction set). The software is designed to schedule tasks in a cyclical manner where different tasks might be executed with different frequencies.

The development and validation of this software has been performed using development and validation platforms implementing a fully representative processor breadboard connected to a simulated environment modelling accurately all of the processor's interactions with its environment.

Most of the Envisat onboard software is either located in PROM and downloaded to RAM (e.g. most instrument software), or already loaded in RAM (in the case of the PMC). In most cases, software components are loadable and patchable by ground telecommands. To support this, the Flight Operation Segment at ESOC is equipped with a Software Maintenance Facility where software is modified, tests of the modified software are run, differences between the memory images extracted, and patch commands generated for uplinking. After uplinking, the updated memory is dumped for a final comparison within the software maintenance facilities.

There were difficulties associated with the initial design of the PMC software. Requirements stemming from the original Columbus multi-mission capability were never descoped to the specifics of the Envisat mission. At the same time, the separation between Mission Prime and Polar Platform Prime Contractors meant that proper requirements on mission operability were not levied on the PMC. These difficulties co-existed with a decision to use Ada on the late 1980s Mil Std 1750 processor the MAS 281.This situation led to late software modifications and additional validation effort. An independent company was contracted to perform additional intensive testing using four additional software-validation facilities. System-level schedule impacts were limited by tuning the scope of system tests to match successive and incremental deliveries of the PMC software for satellite Assembly, Integration and Testing (AIT). A mission time-line test simulating five consecutive orbits of intense satellite operation was successfully run in December 2000. The final version of the PMC software is currently completing non-regression testing.

System databases supporting in-orbit satellite operations

Two main database products have been delivered to support Envisat in-orbit operations:

- The Satellite Reference Database (SRDB) has been delivered by the Prime Contractor to ESOC to conduct in-orbit operations. It gathers together lower-level databases initially used within the various EGSEs, and ensures uniqueness of each parameter through a well-controlled nomenclature. The database as delivered to ESOC contains all macro commands to allow commanding of the satellite. It also contains all of the housekeeping parameters, their defined monitoring limits and the specific calibration curves that are required to monitor the satellite during the mission.
- The Satellite Characterisation Database was a deliverable from the satellite Mission Prime Contractor. It documents precisely, in a pre-agreed format, the results of all performance testing executed throughout the satellite development and validation phase. The contents of the database, such as alignment data or CCD spectral calibrations, have been used to build ground software for the generation of so-called ‘Level-1b products’ from each instrument.

Ground Support Equipment

The size and complexity of the spacecraft result in demanding requirements on ground-support equipment. Mechanical Ground Support Equipment (MGSE) includes conventional items such as a satellite trolley, vertical integration stands for the Payload and Service Modules, lifting beams for spacecraft and panels, and turnover trolleys. The satellite trolleys, of which there are two, to support two satellite models simultaneously, each weigh about thirty tonnes. They must be disassembled for transport and to move from one clean room to another. Even when disassembled, they are out-of-gauge items. Despite this, they are supported for movement on compressed-air pads hovering a few millimetres off the floor. This system works extremely well and the trolley is routinely manoeuvred by just two or three people.

The Electrical Ground Support Equipment (EGSE) is also large and complex, covering about 400 m^2 of floor area and consuming 0.1 MW. There are three principal elements:

- Service Module EGSE, including processors to stimulate AOCS sensors and special checkout equipment for Service Module systems
- Payload Module EGSE, including special checkout equipment for payload subsystems
- Instrument EGSE organised around a single Integrated Test Assembly, but with a number of dedicated data and processing blocks for each of the instruments.

Each of these elements was based on standard systems using the Elisa checkout language procured from Astrium SAS (formerly Matra, Toulouse). Unfortunately, evolution of the standard between the procurement initiation for each of the elements meant that neither the software, nor the databases nor the hardware were interchangeable.

A significant and late addition to the EGSE was a pair of the Front-End Processors designed for the ground segment to ingest measurement data. These provide possibilities to realistically process every virtual channel data unit and every source packet, even at the highest data rate (100 Mbps). As a result, confidence in data integrity from every test is very high.

The EGSE used to test instruments was built around a standardised element, procured by the Mission Prime Contractor, which precisely simulated Polar Platform interfaces. The Integrated Test Assembly employed at satellite level used the same processor and software. This had two important benefits: time spent debugging the instrument EGSE was minimised by sharing resources, and the instrument-level test software could be used directly at satellite level.

A consequence of the number of processors involved in the EGSE has been lost test time due to hardware defects, design deficiencies and software bugs. The duration of the engineering-model campaign was significantly affected by resolution of these issues. It seems that when procuring EGSE, paying much more attention to system integrity and maintainability would have saved significant cost in the overall programme.

Launch campaign

Envisat will be the first user of the new S5 Payload Processing Facility in Kourou (Fr. Guiana). The launch team will occupy all of the available office space, all of the checkout areas, all of the integration room, and the smaller of the two fuelling halls.

About 340 tonnes of material will be transported to the launch site using a dedicated Antonov AN124 aircraft for the flight hardware, two Boeing 747s for the EGSE, and a ship for the MGSE items. At the peak, there will be about 120 project personnel supporting the launch campaign, which will last about three months. The MGSE will be set up by an advance party about a month before the start of the launch campaign itself.

Envisat's size made it impractical to design a container to fit the integrated satellite into the AN124, which is largest commercially available cargo aircraft. Consequently, the Service Module, Payload Module, ASAR antenna and solar array will be transported separately and the satellite reassembled in Kourou.

After assembly of the solar array, a test will be conducted to check that initiation of deployment functions correctly. A full electrical test of the spacecraft will take place, followed by fuelling with 300 kg of hydrazine. All of these activities will happen without leaving the class-100 000 facilities within the S5 building. The spacecraft will then be loaded into a special transport container for the short trip to the Final Assembly Building (BAF), where it will be lifted onto the top of the launcher and encapsulated in the fairing. Prior to encapsulation, the last of the remove-before-flight and install-before-flight items will be dealt with. There are more than 300 of these in total, so this operation will take place overnight. The assembled rocket with solid boosters attached and satellite mounted will then be transported to the launch pad, along a parallel pair of railway tracks, on the enormous launch table. Hopefully, the launch will take place at the very first launch opportunity, of which there is one per day.

Design and testing

Mechanical

Mechanically, the satellite consists of the cone of the Service Module, which at its lower end interfaces with the launcher and the central cylinder of the Payload Module. Shear panels attach sidewalls to form a stiff rectangular box structure. Figures 6 and 7 show the overall structure.

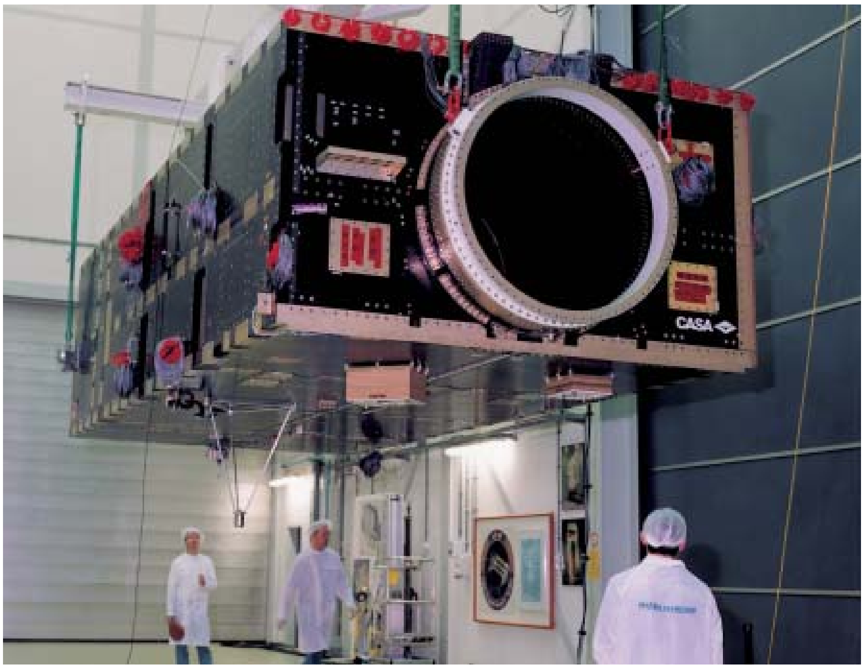


Figure 6. The Payload Module box structure around the central tube

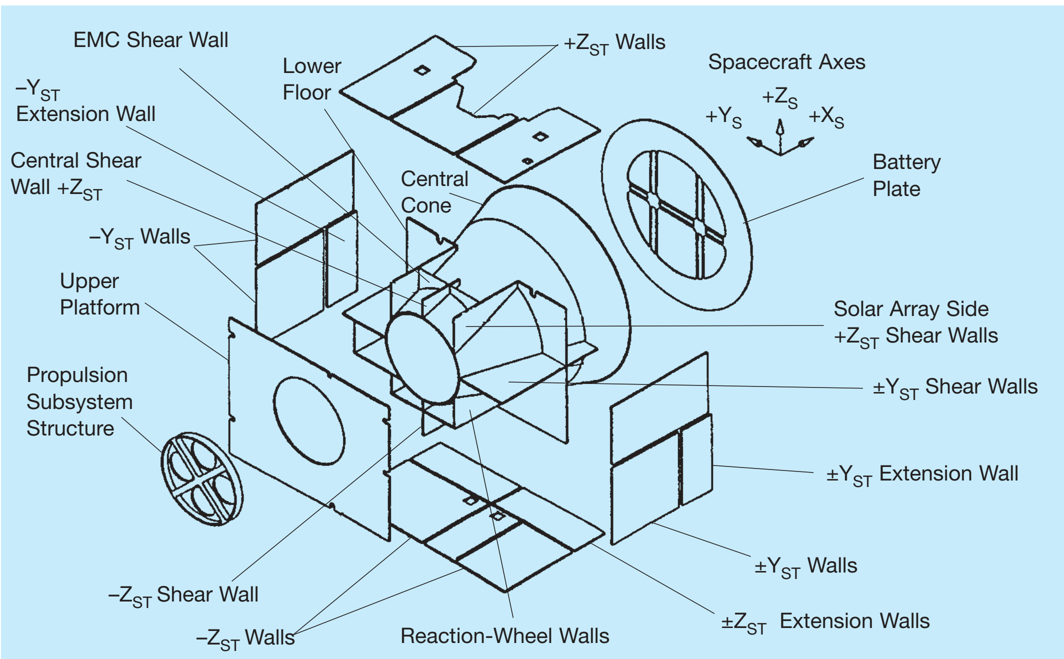


Figure 7. The Service Module's mechanical configuration

The SM is built around a CFRP cone as the primary structure, with a launcher interface at one end and the propulsion module at the other. Both interfaces are made of aluminium. A box-shaped metallic structure, with aluminium honeycomb panels, supports the electronic equipment and surrounds the central cone.

The structure of the Payload Module consists of a 1.2 m-diameter CFRP tube and a set of CFRP sandwich panels forming the webs and external faces of the PLM box. The structure is conceived as a stack of four bays, each 1.6 m high. The bay furthest from the SM (bay 4) is designed to be removable from the other three.

Static load tests were conducted using hydraulic jacks on separated structural models of the Service Module and the Payload Carrier. These tests involved loading the structure to the equivalent of 2.8g at its centre of gravity.

The structural-model satellite programme began with a modal-survey test that allowed the identification of modes, and their correlation with the satellite finite-element model. Following the test, the interface to the solar array was stiffened.

Sine vibration (Fig. 8) of this very large spacecraft brought a number of problems. The structural model was the first object to be tested on the new hydraulic shaker, HYDRA, at ESTEC. HYDRA allows simultaneous excitation of the test object with three translations and three rotations, although for Envisat only one axis was excited at a time. Even so, the satellite can be tested without a physical reconfiguration between axes, saving many days on the critical path. It also allows a complete set of very-low-level runs to be performed at the start of the test campaign, to identify early any unexpected behaviour of the new test object and to confirm the safety of the test. At the time of the structural-model test, the shaker's development was not complete, and so only the longitudinal axis was attempted, to avoid controllability problems in lateral. Even so, it was considered unsafe to test between 58 and 81 Hz. And qualification in this domain was achieved by a sine response analysis using the finite-element model qualified by the modal-survey test. Lateral axes on the structural model were tested on the electro-dynamic twin shaker also at ESTEC. The inertia of the test object meant, however, that instead of starting the sine scan at 5 Hz, it was started at 15 Hz. Qualification below 15 Hz was achieved using the results of the static load tests. Despite these difficulties, the structural-model test results were useful in refining the qualification and acceptance levels for both equipment and instruments

Figure 8. Envisat being installed on the HYDRA hydraulic shaker at ESTEC

For the flight-model acceptance test, the HYDRA had been much improved. Even so there were serious concerns about its controllability. There were also concerns about the effect of cross-talk from one axis to another on the validity of the test. An extensive series of pre-tests were performed using a dynamically representative dummy of Envisat. These showed quite serious cross-talk between axes and excitation of spurious frequencies. A strategy was adopted whereby cross-talk would be accepted, and deep notches to the base excitation predicted for launch accepted, provided all critical points were exercised above their flight limit loads at least once. In the event, the damping and linear behaviour present on the flight model (but not on the

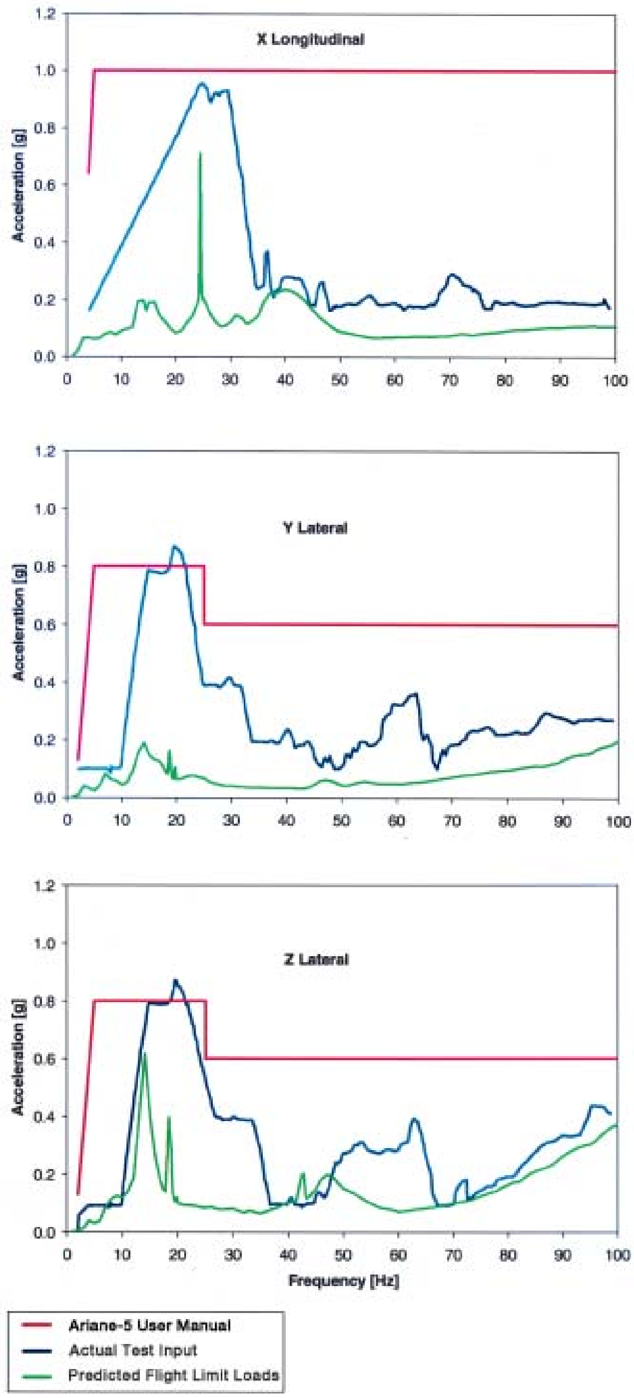


Figure 9. The sine-vibration spectra

dummy) allowed the predicted base excitation to be achieved in all three axes with only one very minor deviation. A bonus was that the entire flight-model sine-vibration test was completed in just 7 days, compared to the three weeks or more predicted for the multi-shaker (Fig. 9).

To ameliorate the difficulties expected in identifying the mode shapes of the finite-element model in the results from HYDRA, a novel 'mini-modal-survey' test was performed. This test placed a single exciter at one of the lifting points of the satellite, and was executed in less than a day. The results provided a good identification of modes in the frequency range below 50 Hz. The required first lateral frequency for the satellite is above 5.5 Hz. The measured first frequency is about 7.8 Hz.

The design requirement from Ariane-5 is for a longitudinal quasi-static load of 4.55g and survival of 1g from 5 to 100 Hz. Lateral accelerations are below 0.8g up to 100 Hz. In common with most large space structures, extensive notching was necessary to protect sensitive items, such as the solar array, the ASAR antenna, and several of the optical instruments.

The vibration spectra (Fig. 9), which were accepted by Arianespace on the basis of coupled dynamic analysis, were less than 0.3g in longitudinal and less than 0.35g in one lateral axis from 35 to 100 Hz. The other lateral axis (in the plane of the Earth-pointing face) was less than 0.6g in the same region.

Acoustic tests on both the structural and the flight models (Fig. 10) of the satellite were conducted in the Large European Acoustic Facility (LEAF) at ESTEC. The structural-model test had allowed the derivation of random qualification levels for equipment and instruments. The acoustic spectrum required by Arianespace has an overall level of 142 dB. The level for the Envisat acceptance test was 138 dB. These reductions were possible following flight experience with a number of Ariane flights, and changes to the launch pad and fairing.

One issue that caused some difficulty during the development programme was shocks propagating into the spacecraft from the separations of the fairing and of the spacecraft from the launcher. In the event, the launcher authority designed a shock-attenuation device for Envisat and a number of other missions. Also, a piece of pyrotechnic test hardware was specifically designed to produce the specified shocks at the spacecraft interface, rising from 60g at 300 Hz to 2000g at 2500 Hz. Shocks propagating to the lower levels were significantly above design levels for a number of equipment items, but the test was survived without the slightest damage to any spacecraft hardware, and without any change in the status of the relays.

A second issue of concern during the development programme was micro-vibration, with a number of the optical instruments being sensitive to small imported vibrations during the mission. Analysis and some measurements showed that most noise was produced by the AOCS reaction wheels, and by the Stirling-cycle coolers in two of the instruments. Subsequent analysis has shown that the worst-case micro-vibration at the instrument interfaces would cause about a 0.1-micron displacement and a 0.2 microrad rotation. These levels are acceptable by all instruments.

Thermal

The satellite is designed to fly along the direction of the ASAR antenna, with the solar-array end of the spacecraft pointing towards the Sun, which appears to rotate around the spacecraft once per orbit. One orbit takes about 100 min, with about 30 min in eclipse. The other end of the spacecraft points towards deep space and is used to accommodate the infrared instruments, which require a cold environment for their detectors.

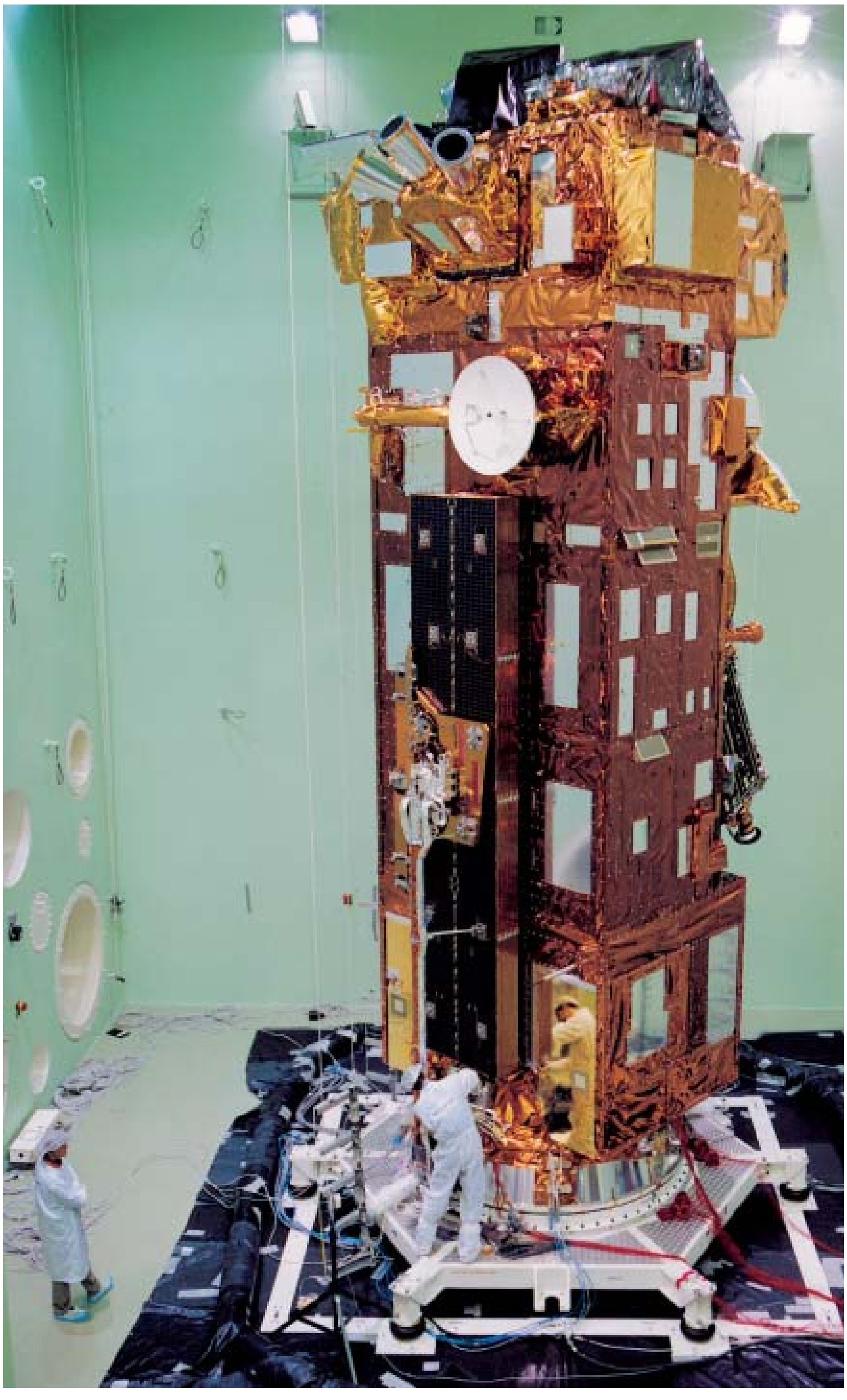
Reuse of Spot and ERS heritage technology dictated a flight direction at right angles to the longer dimension of the spacecraft. This implied that instruments on the cold tip (which points towards deep space) had to be specifically designed to fit into the limited surface area. It also implied that the ASAR antenna had to be designed to fold up for launch.

The spacecraft's thermal design is driven by the orbital conditions, and by the need to survive in the Sun-pointing safe mode. The thermal design of the box structure of the basic spacecraft is classically passive. Portions of the box's surface serve as radiators, whilst the rest is clad in multi-layer insulation. Externally mounted items, especially the instruments, are designed to have as little thermal interchange as possible with the box structure itself. Systems of heaters operated by thermostats protect equipment in safe mode. Specific thermostat-controlled heaters protect the hydrazine system and the batteries. Software-controlled heaters are operating at other times when on-board computers are available to control them. The Instrument Control Unit of each instrument controls the thermal management of that instrument when it is on. A number of optical instruments incorporate heat pipes to transfer heat away from detectors to appropriately positioned radiator plates.

In view of its size, it was impossible to perform a thermal test on the spacecraft as a whole. The flight models of the Service Module and Payload Module were tested separately. Even so, the Payload Module could not be loaded directly into the ESTEC Large Space Simulator, but was separated between levels three and four and the pieces re-mated inside the chamber (Fig 11).

The thermal-balance test on the Service Module was of the classical type, using solar simulation and liquid-nitrogen-cooled shrouds. Performed during December 1996, it validated the performance of the thermal hardware and led to satisfactory correlation of the thermal mathematical models. It also revealed that the RCS heater power was insufficient. The necessary hardware modifications, such as the wiring up of available trim heaters and new additional heater circuits, were therefore incorporated after the test.

Figure 10. Envisat being installed in the LEAF acoustic facility at ESTEC, with the solar array in the foreground



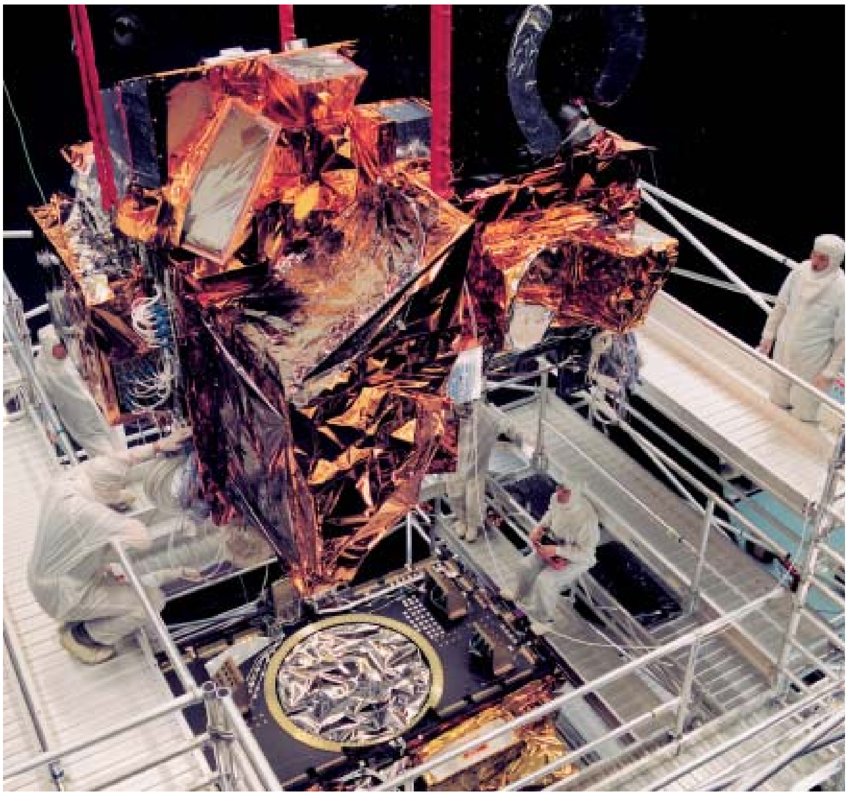


Figure 11. Upper level of the Envisat payload being loaded into the Large Space Simulator (LSS) facility at ESTEC, for mating with the three lower levels

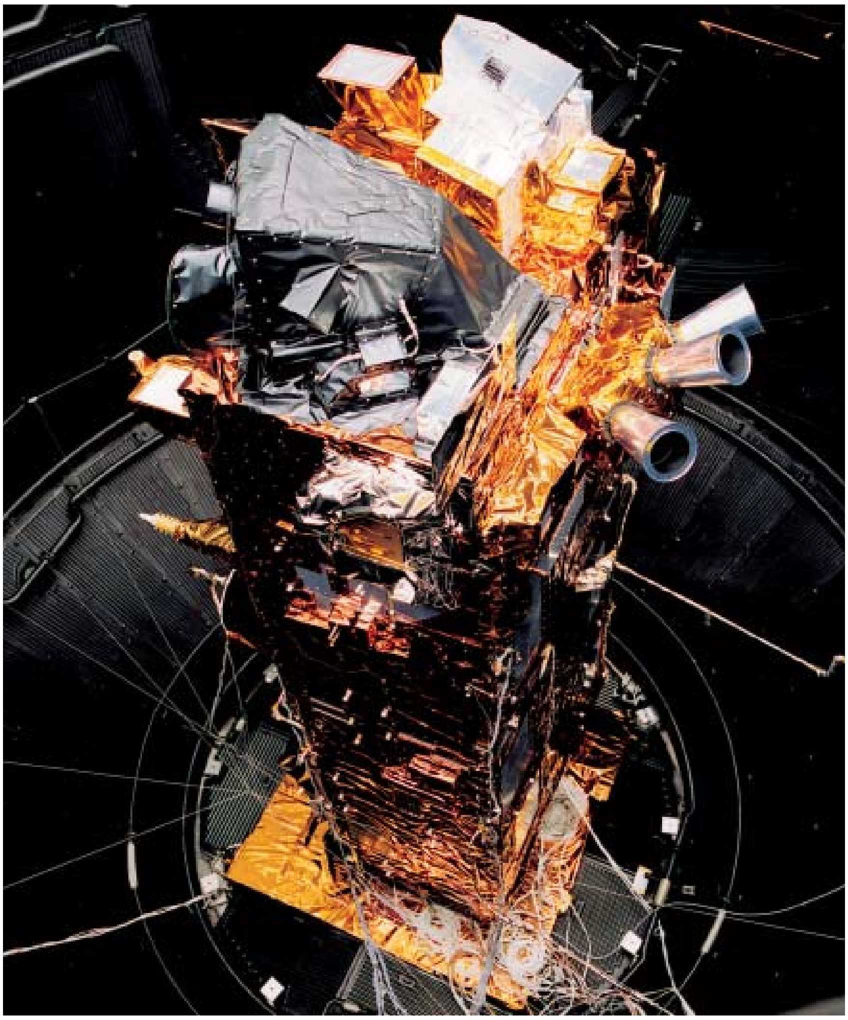


Figure 12. The complete Envisat payload in the LSS at ESTEC

The Payload Module thermal-balance test was performed during August 1999. The physical size of Envisat (Fig. 12) also made it impossible to illuminate it adequately with a simulated Sun. Consequently Sun, Earth and albedo inputs were modelled by the use of elevated shroud temperatures and test heaters. The size of the Payload Module made it necessary to utilise the full facility data-handling capacity of more than 800 thermocouple channels.

Thermal testing of the flight payload showed a poor design of the vent holes for the internal compartments. Inappropriate siting of ion gauges also hindered measurements of vacuum. As a consequence, high-voltage equipment could only be switched on safely late during the test, such that the thermal-balance test phases had to be executed without X-and Ka-band operation. Additional test phases characterising the thermal performance of these two subsystems had to be added. After the test, the vent holes were enlarged and a short handbook prepared on the use of ion gauges.

The test verified the performance of the thermal hardware and allowed correlation of the various mathematical models of the Payload Module and instruments. However, it also revealed that the heater power for the RCS thrusters was insufficient for flight and corrective measures have been implemented.

Electrical

The power subsystem uses eight nickel-cadmium batteries, each of 40 Ah, and a solar array delivering more than 6.75 kW at end-of-life. Power is provided to the satellite via unregulated (22 – 37 V) and regulated (50 V) buses. The main power supply to the Payload Module is via four permanent high-power unregulated buses routed directly from the Service Module power source point. The power-bus voltage follows the battery charge/discharge characteristic. A junction and shunt regulation unit (RSJ) acts as the main interface between the solar array and batteries, taking into account varying power demands. A power distribution unit (BD) provides the necessary switching and protection of power buses supplied to the Service Module.

Within the Payload Module, power is distributed via two Payload Power Distribution Units (PPDUs), which are controlled and monitored by the Payload Module Computer. Each unit provides power switching and protection of power feeds. One of the distribution units is dedicated to the instruments, the other to Payload Module equipment. Both distribution units are located as close as possible to the SM/PLM interface in order to minimise power-harness voltage drops. During testing on the flight-model satellite, a design fault was discovered in a PPDU that allowed the unit to be switched off in a configuration in which it could not be switched on again. A design modification was introduced to cure this.

The size and complexity of Envisat has resulted in more than 700 kg of harness being used on-board. It seems that electrical design approaches such as are being applied on large aircraft might become appropriate for this class of spacecraft.

Electromagnetic and radio-frequency compatibility

EMC/RFC aspects of the spacecraft are dominated by the radiated field from the on-board radars, and from the Ka-band transmissions to Artemis. Test requirements include:

- 62 V rms from 2.0 to 2.5 GHz S-band
- 80 V rms from 3.0 to 3.4 GHz RA-2 S-band
- 160 V rms from 5.15 to 5.5 GHz ASAR
- 80 V rms from 13.2 to 13.9 GHz RA-2 Ku-band.

While testing transmit/receive modules for use in the ASAR antenna, it was discovered that the receivers were susceptible to RA-2 transmissions. Filters were fitted to the transmit modules for 12 tiles to avoid this. In all other cases, tests conducted at equipment level demonstrated compliance with these levels.

RF compatibility tests on the engineering- and flight-model satellites (Fig. 13) with deployed antennas demonstrated compatibility of the entire operating satellite with the transmitted power levels. These two tests each involved the construction of a large RFC facility covered with RF-absorbent cones within the clean room. The supporting structure was built by Brilliant Stages, a company better known for its work on outdoor rock concerts, but it was able to work quickly and effectively in parallel with normal testing. In the case of the engineering-model spacecraft, the facility entirely surrounded the spacecraft. The facility for the flight model was simplified so that only a corner and a roof were constructed to avoid reflections of the Earth-facing antenna beams. The two antennas on the rear side were not radiating during this test, having been validated on the engineering-model satellite.

Figure 13. Flight model of Envisat being prepared for RF compatibility testing, with the ASAR antenna at the bottom of the picture

These radiated-field requirements were supplemented by the usual requirements for compatibility with the launcher. One late demand from the launcher authority originated from a new military radar protecting the launch site itself, but it has also been possible to show that it will not affect Envisat.

Low-frequency conducted interference is dominated by power-bus ripple originating from the on-board Stirling-Cycle coolers, and from the pulsed power demand of the ASAR. Tests at the equipment, Service Module and satellite levels have demonstrated self-compatibility of the satellite in the presence of these signals with a margin of 6 dB.

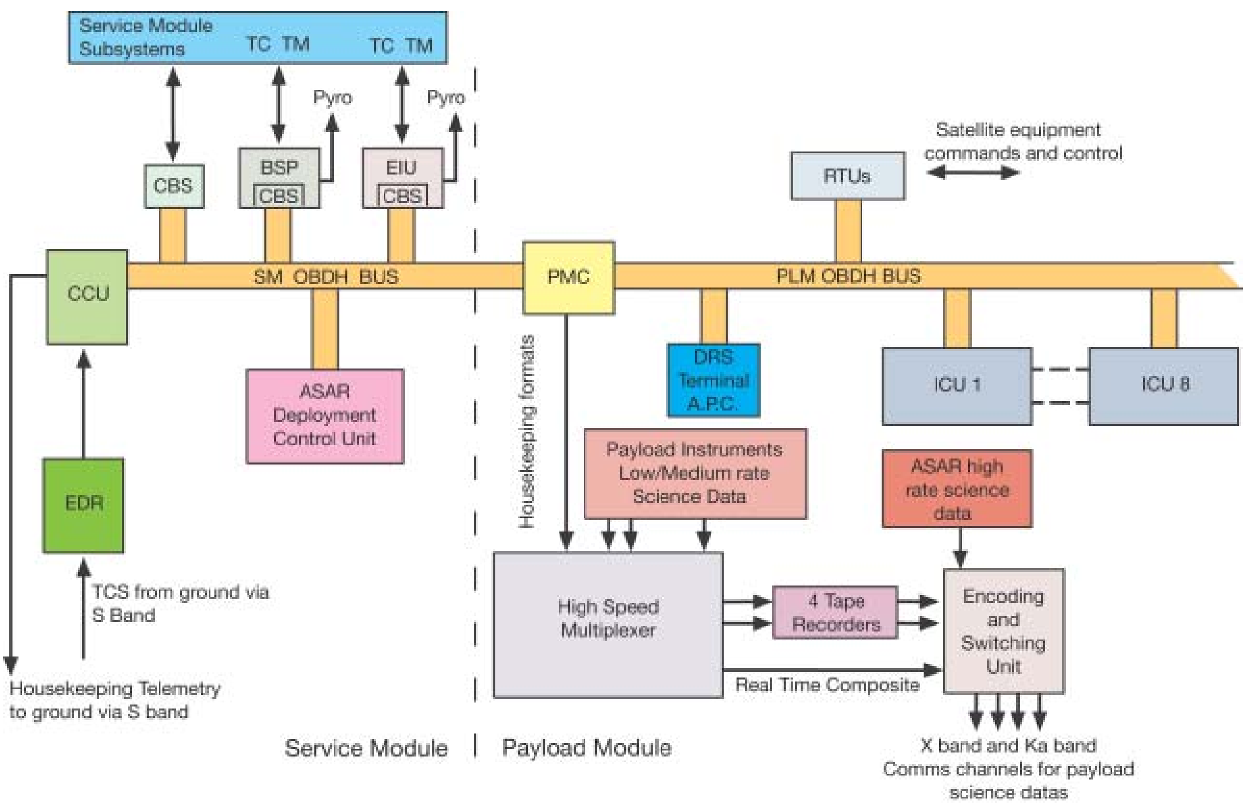
The satellite has been successfully designed and tested for avoidance of, and immunity to electrostatic discharges. All conductive elements of the structure, including the solar array, CFRP and aluminium panels, thermal blankets and thermal finishes, are electrically bonded together to form an effective equipotential conductive surface, thereby avoiding differential electrostatic charge build-up. All panel surfaces and cable harnesses have also been designed to form a Faraday cage providing electromagnetic shielding.

Measurement data handling
This function is responsible for the acquisition, formatting, recording, and communication to the Payload Data Ground Segment of measurement data generated by the Envisat payload (Fig. 14). The High-Speed Multiplexer (HSM) collects CCSDS Instrument Source Packets (ISPs) generated by the payload instruments (at bit rates up to 22 Mbps in the case of MERIS). It formats and encodes these ISP data into Virtual Channel Data Units that are multiplexed together to produce two types of data frames generating Channel Access Data Units (CADUs). One frame is for recording by one of the two solid-state recorders or by the tape recorder, the other is for real-time generation to the Payload Data Ground Segment.

CADU data streams (either from the dumping of a recorder, or generated by the HSM in real time), are collected by the Encoding and Switching Unit (ESU), which routes each of the different CADU data streams to either the X-band or the Ka-band communication subsystem. The ESU also collects the two 50 Mbps data streams from the ASAR payload instrument (this high data rate results from the operation of ASAR in Image mode), and routes them either directly to the communication systems or to one of the two solid-state recorders for later dumping.

The Kiruna X-band ground station provides only about 10 minutes of satellite visibility per orbit. The add-on capability of data relay via Artemis provides an additional 20 minutes of visibility per orbit. Payload data must be

Figure 14. Block diagram of Envisat's On-board Data-Handling (OBDH) System



delivered not later than 3 h after data generation. To meet these constraints, the Payload Module carries two Solid-State Recorders (SSRs), each providing a dynamic RAM capacity of 70 Gbit. These two SSRs allow independent recording (4.6 Mbps) and dumping (50 Mbps) of the Envisat low-bit-rate composite, of the ASAR high-bit-rate data (100 Mbps record and dump), and of the MERIS full-resolution data (22 Mbps record and 50 Mbps dump). Initially, the Payload Module embarked four tape recorders. These units proved to be so questionable that the Agency and its Prime Contractor decided to procure two SSRs. A single tape-recorder unit has, however, been retained on-board as a additional spare capability to record and to dump the low-bit-rate composite.

Envisat is equipped with two distinct communication subsystems accommodated within the Payload Equipment Bay (Table 6). The X-band communication subsystem has three communication channels operable independently and, if need be, simultaneously. Each channel is equipped with a Quadrature Phase-Shift Keying (QPSK) modulator and a Travelling-Wave-Tube Amplifier (TWTA) and, after band-pass filtering, the modulated signals are added through a wave-guide output multiplexer. The resulting composite spectrum is radiated to the Payload Data Ground Segment via a shaped-reflector antenna. The three frequency carriers used by this subsystem are 8.1, 8.2, and 8.3 GHz.

The Ka-band assembly (Fig. 15) is also composed of a three-channel communication subsystem. The assembly uses a 2 m deployable mast located on the opposite side of the satellite to the Earth. The mast is equipped with an azimuth and elevation pointing mechanism, which steers a 90 cm-diameter Cassegrain antenna pointed to the Artemis satellite. Pointing commands are generated by an Antenna Pointing Controller, which is accommodated together with the RF part of the assembly within the Payload Equipment Bay. The subsystem carrier frequencies are 26.85, 27.10 and 27.35 GHz. The system operates in open loop, which means that the Artemis Ka-band transponder will track Envisat using its telemetry, without the need for an extra tracking beacon or tracking receiver.

Table 6. Link budgets

Ground Station Antenna: 32 dB/K G/T	**X-band**
100 Mbps (QPSK)	3.6 dB
50 Mbps (BPSK)	5.7 dB
Ground Station Antenna: 37.2 dB/K G/T	**Ka-band**
100 Mbps (QPSK)	1.64 dB
50 Mbps (BPSK)	4.64 dB

The satellite command and control system

This system is organised around two On-Board Data Handling (OBDH) systems (Fig. 16), a master Service Module data-handling system, and a Payload Module data-handling system slaved to the SM OBDH bus. Each module has a dedicated computer to which equipment items are connected through OBDH digital serial data busses.

Service Module command and control

The SM data-handling subsystem performs the processing and sorage of housekeeping data, the management of telecommand and housekeeping telemetry (at the interface with the S-band ground stations), the data-bus control, the satellite alarms management, and the clock generation to all OBDH peripherals, including the Payload Module.

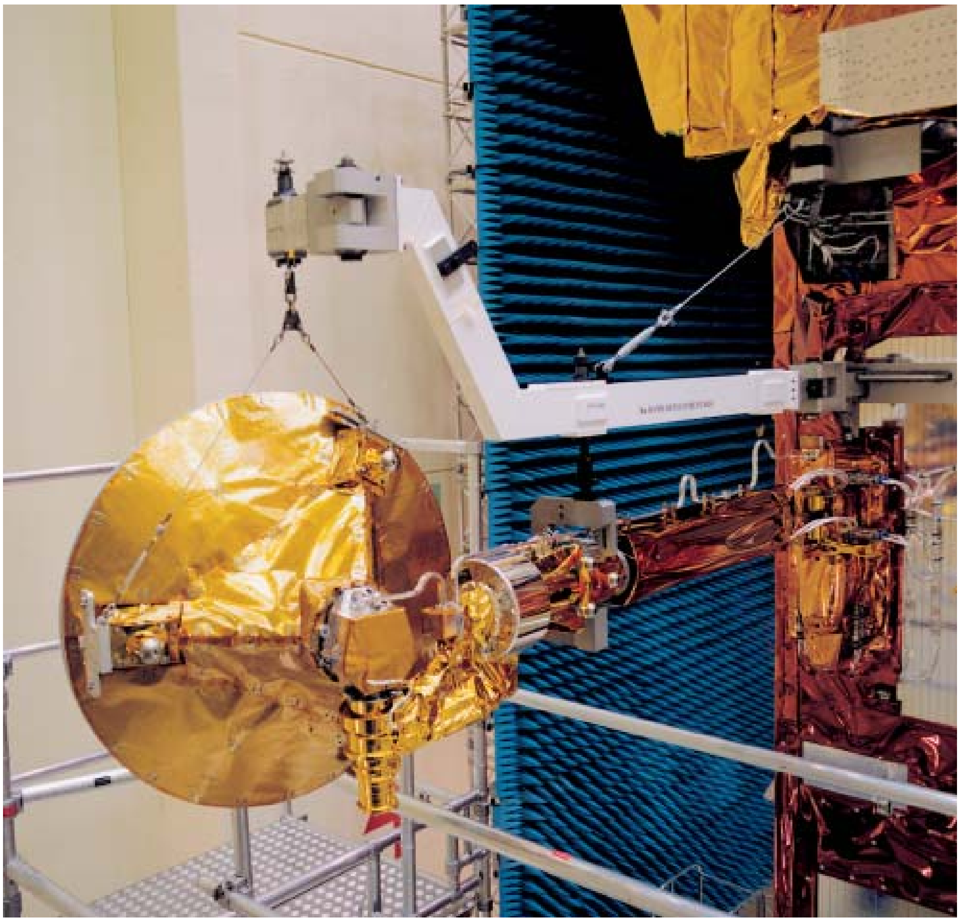


Figure 15. The Ka-band antenna undergoing deployment testing

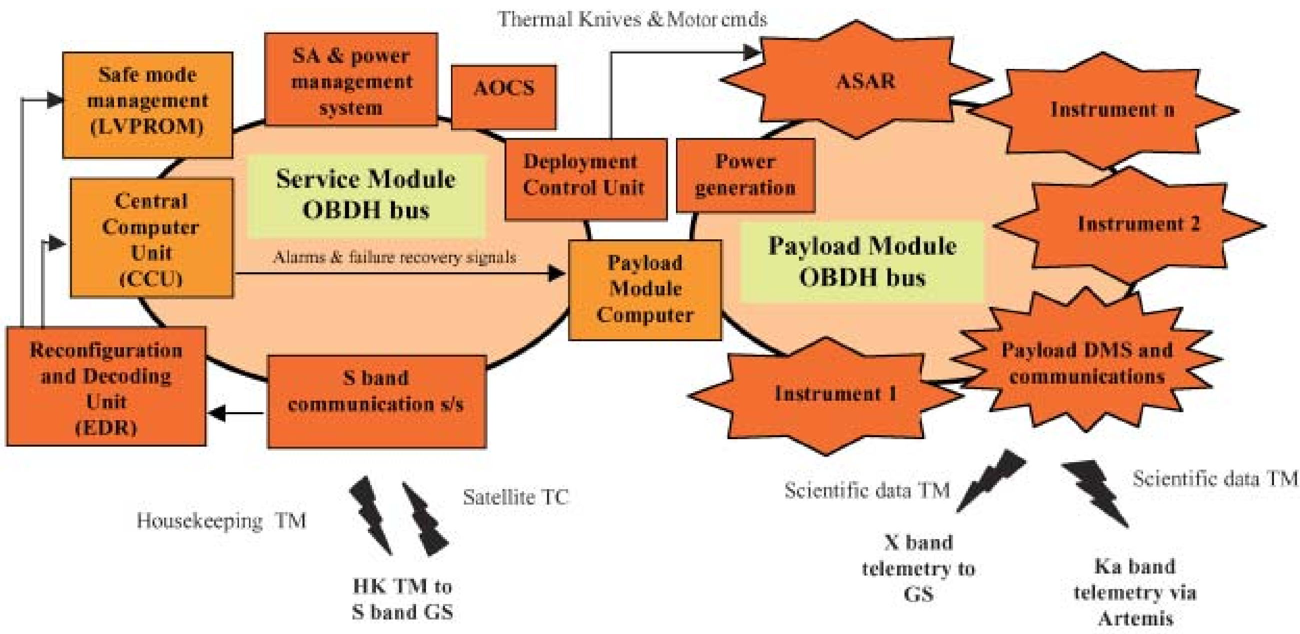


Figure 16. The Envisat satellite command and control concept

It is based on a Central Communication Unit (CCU), which handles the exchange of information between the ground and the SM equipment via the OBDH data bus. It contains an ultrastable oscillator, from which all satellite timing is generated. The CCU communicates through the SM OBDH bus with the Payload Management Computer located within the PLM. The CCU also serves as the main computer for the Attitude and Orbit Control Subsystem (AOCS). Dedicated control algorithms and monitoring functions are implemented into the CCU software.

The Decoding and Reconfiguration Equipment is responsible for receipt of the telecommands generated by the ground segment, and their decoding. This unit also acts as a watchdog with respect to the CCU and can command satellite switch-over to a safe mode in the case of a major CCU anomaly. In such a case, command and control of the satellite's main survival functions relies on software contained in a PROM

The S-band communication assembly provides a bi-directional telemetry and telecommand link with the ground and is used for overall satellite commanding, monitoring and control. It provides a 2000 bps forward and 4096 bps return S-band link to ground, via two S-band antennas operating with cardioid radiation patterns and opposite circular polarisations. Omni-directionality of the link allows communication in any non-nominal attitude. The antennas are connected via a passive hybrid to two S-band transponder systems. Unlike the Spot and ERS transponders, an internal real-time processor pilots the transponder on Envisat. The mission-critical software in this processor was subjected to independent software validation prior to its acceptance. The S-band transponder modulates housekeeping telemetry, demodulates telecommands, and supports ranging and range rating by the terrestrial command and control ground stations.

The SM housekeeping system comprises two units. The housekeeping and pyrotechnic unit (BSP) monitors temperatures and issues thermal-heater and pyrotechnic-firing commands. The Electrical Integration Unit (EIU) arms and powers the thermal knives and motors used to release and to deploy the solar array, monitors its deployment, and enables retraction if required.

The problems experienced during the development and validation of the SM command and control system have been limited because of the experience acquired with this bus for previous missions such as Spot-4 and Helios-1. One exception has been the inheritance of the dual-mode transponder design from Columbus serviceable missions. Because of its mission-criticality for Envisat, this subsystem had to be carefully analysed. Its software was simplified and modified to increase its reliability. To avoid permanent uplink loss due to simultaneous latch-up of both DMTs, the CCU will automatically reset the DMTs if it has not received a valid telecommand within the last 24 h.

Payload Module command and control

The Payload Module avionics is decoupled from the Service Module. Its provides its own power distribution, operates its own OBDH bus, and is controlled by a specific Payload Module Computer (PMC), which performs tasks including scheduling of the commanded mission, monitoring of the Payload Module systems (Payload Equipment Bay avionics, and payload instruments).

The PMC operates as the payload-instrument master controller. It communicates with the SM via the SM OBDH bus, and commands and controls all the Payload Module equipment and instruments via a separate Payload Module OBDH bus. It also routes housekeeping telemetry back to the PMC for transmission to the SM.

The PMC manages two types of macro-commands; immediate macro-commands, for example an immediate request of a special housekeeping telemetry format during visibility over a ground station, or time-tagged commands destined for PEB subsystems or payload instruments. Mission-planning commands aimed at performing autonomous round-the clock operations without ground intervention are stored as a queue of time-tagged commands.

The PMC gathers and formats PLM subsystem and payload-instrument housekeeping telemetry, which will be merged by the SM CCU with SM housekeeping telemetry prior to downlinking. The PMC transmits to the PEB subsystems and to the payload instrument ICUs, time-synchronisation signals derived from the pulses generated by the SM. This time-synchronisation function allows correlation of the onboard time-of-occurrence of specific events with the UTC time used on the ground.

The PMC also packetises the overall satellite housekeeping telemetry generated by the SM into a stream of ancillary source packets. These ancillary data contain in particular the spacecraft's orbital position parameters with respect to the satellite onboard time. They are used by some instruments to schedule specific activities (e.g. Sun or Moon occultation activities for SCIAMACHY). These ancillary data are also routed to the High-Speed Multiplexer, which allows the ground segment to receive the satellite housekeeping telemetry generated during non-visibility periods.

Instrument Control Units (ICUs) are responsible for the instrument command and control functions, ranging from the management of telemetry/telecommand flows between the PLM OBDH and the payload instrument units (MWR, DORIS), to the management of instrument internal data buses and of communication with secondary scientific processors connected to them (MERIS, RA-2, ASAR). Each payload instrument is equipped with an ICU, as is the Ka-band Antenna Pointing Controller.

Satellite Attitude and Orbit Control System
The AOCS provides three-axis stabilisation of the satellite body, which remains fixed in the local orbital reference frame. It ensures acceptable pointing and pointing-stability performance for the payload instruments and sensors (Table 7).

In nominal operational modes, the CCU software processes all AOCS signals received from the sensors and controls the AOCS modes. Figure 17 shows the satellite AOCS mode transition diagram. The sensors are:
- A set of four independent two-axis gyroscopes: two are used for nominal operations, with the other two remaining in cold backup.
- Two Digital Earth Sensors (DES): one is used for nominal operations, while the other is kept in cold redundancy.
- Two Digital Sun Sensors (DSS): one is used for nominal operations, while the other is kept in cold redundancy. DSS is not used for attitude control in SYSM.
- Three Stellar Star Trackers (SST): any pair used for SYSM, with the third kept in cold redundancy.

In nominal operations, the CCU software computes the attitude and pointing estimation using correction algorithms that are AOCS-mode dependent. It also generates commands to the following AOCS actuators:
- 16 thrusters, 8 allocated to each nominal and redundant actuator chains. They are used for initial attitude control, in- and out-of-plane manoeuvres, and safe mode. The backup thrusters are kept in cold redundancy.
- 5 reaction wheels, each with a 40 Nm/s capacity (two wheels on Y- and Z-axis, one wheel on X-axis)

Table 7. Pointing stability performance for the most demanding payload instruments

Time Period (sec)	Budget for Worst Axis (deg)	Requirement for All Axes (deg)
1.6	0.0017	0.0040 (SCIAMACHY)
4.0	0.0036	0.0040 (MIPAS)
75.0	0.0135	0.0150 (MIPAS)
110.0	0.0138	0.0200 (SCIAMACHY)
120.0	0.0143	0.0300 (AATSR)
170.0	0.0139	0.0300 (MERIS)

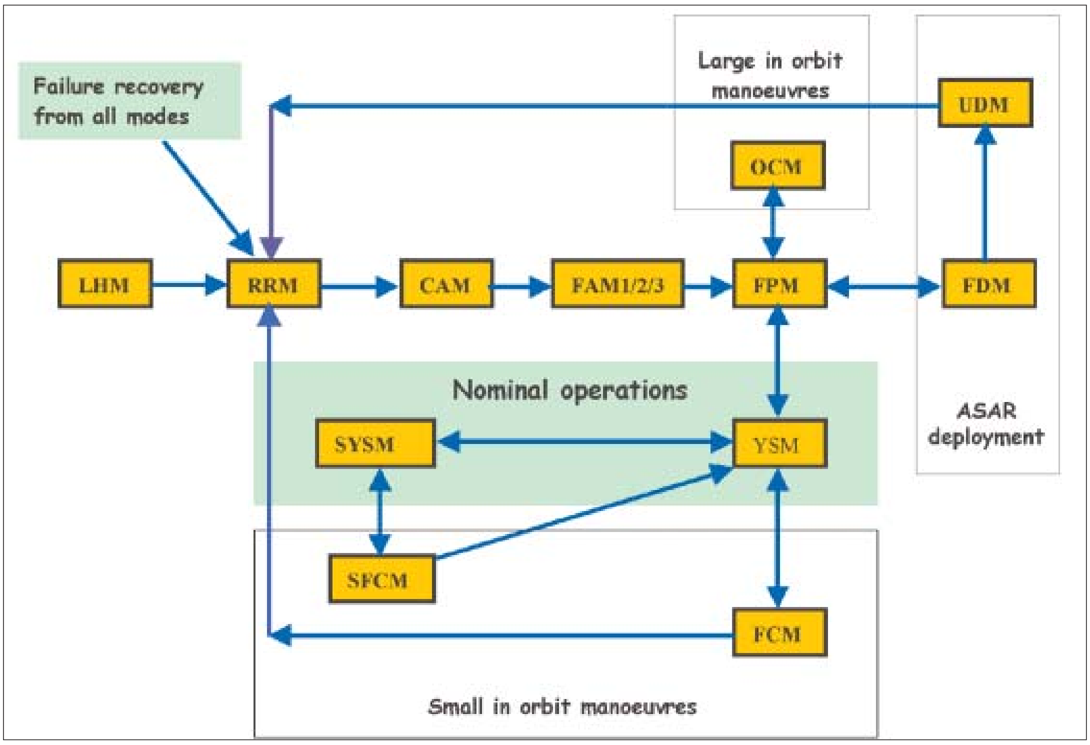


Figure 17. Mode transition diagram for nominal AOCS modes

– 4 magneto-torquers, forming two pairs. They are actuated at the correct satellite orbital phase (depending on their alignment wrt the Earth's gravity field) to control the maximum spin rate of the reaction wheels on the relevant axis.
– 4 tanks containing 300 kg of hydrazine for initial orbit acquisition and orbit-maintenance activities.
– A solar array drive mechanism (MEGS) to control the solar array's rotation (1 rotation per orbit). At a second level, the MEGS is also controlled to compensate for the solar array perturbations due to flexible modes, which are detected by the gyroscopes.

Table 8 provides an overview of the sensors and actuators used for each AOCS mode.

Table 8. Satellite AOCS modes and associated main characteristics

Mode of Operation		Comments	Sensors and Actuators
Operational Mode	YSM or SYSM Geocentric pointing, correction for the local normal pointing, and yaw steering bias	Operational mode with fine satellite pointing	Reaction wheels with magnetic torquers YSM: 1 Digital Earth Sensor, 1 Digital Sun sensor and 2 gyros SYSM: 2 SSTs and 2 gyros
Orbit Control Mode	OCM	Out-of-plane orbit maintenance manoeuvres. ΔV > 0.05 m/s along Y-axis after 90 deg rotation around Z -axis. Payload mission interrupted.	1 Digital Earth sensor 2 gyros 8 thrusters
	FCM or SFCM	In-plane orbit-maintenance manœuvres Mission not interrupted	8 thrusters FCM: 1 Digital Earth sensor, 1 Digital Sun sensor and 2 gyros SFCM: 2 SSTs and 2 gyros
Satellite Safe Mode	SFM	+Z Sun pointing 0.4 deg/s < Satellite rate < 0.65deg/s TM/TC and AOCS control through T4S PROM software.	T4S PROM software-controlled 8 thrusters 2 SAS (+Z and –Z) 2 gyros
Attitude Re-acquisition (exit from safe mode)	RRM->CAM>FAM	Earth re-acquisition by Earth sensors Payload Module put into safe mode	8 thrusters 1 digital Earth sensor 1 digital Sun Sensor 2 gyros

If major anomalies prevent the CCU from performing its nominal tasks (control of the modes above), a specialised Service Module PROM-driven processor (the T4S) can take over AOCS control and keep the satellite in a healthy power and thermal configuration, in so-called 'Satellite Safe Mode' (SFM). The SFM uses attitude information from two specialised Sun-acquisition sensors and rate information from two two-axis gyros. In SFM, the satellite's attitude is controlled by thrusters, the nominal satellite zenith axis is pointed towards the Sun, and a slow angular rate (0.4 – 0.6 deg) is maintained around Z to keep all of the satellite equipment in a safe thermal equilibrium.

The prime satellite AOCS operational mode is stellar yaw-steering, deriving the satellite attitude from the SSTs and the satellite rates from the gyroscopes. Commands are sent to the SST sensors using a ground-based star catalogue tuned to the spectral characteristics of the SST optics. The time of detection of the commanded star, and its position within the SST field of view allow the onboard software to derive an attitude measurement for the satellite. The SYSM software uses SST attitude-measurement data from two of the three SSTs (all accommodated on the Payload Module) along with rate data from two-axis gyroscopes. Within the CCU, these data are combined to derive an estimate of the satellite pointing with respect to the orbital reference frame (close to the most pointing-demanding limb-sounder instruments such as MIPAS), along with the necessary pointing corrections to be applied through commands to the reaction wheels. The generated commands create control torques using the five reaction wheels accommodated along the three satellite axes to keep the satellite pointing as close as possible to the required orbital reference frame, thereby ensuring the required satellite pointing. In addition to attitude-control torques, the magneto-torquers are activated in phase with the satellite position on the orbit to maintain the reaction wheels within operational limits.

In addition to SYSM, there are two other modes (FPM and YSM) that use wheels for fine attitude control. Both of these modes use gyroscopes, Earth sensors and Sun sensors for attitude determination. In the event of complete SST failure, the YSM mode can form a backup operational mode (used nominally on ERS-1 and -2) with degraded pointing and rate performances arising from sensor errors and measurement frequency. Currently, these two modes are only used as transition modes to enter SYSM.

Attitude control using thrusters results in coarse pointing performance. The other satellite modes are used for orbit maintenance or operational transition towards fine-pointing modes. These modes use monopropellant thrusters for attitude control. In these coarse pointing modes, attitude measurement is provided by a Digital Sun Sensor, a Digital Earth Sensor, and two two-axis gyroscopes.

Orbit manoeuvres are required to achieve the correct orbit following separation and to correct the spacecraft altitude (compensation of the air-drag effect) and inclination over the mission's lifetime. Inclination manoeuvres require delta-V firings normal to the orbital plane, which is achieved by rotating the spacecraft by 90 deg, before and after the firing. For protection against mis-pointing, vulnerable instruments are switched to either standby or heater mode during the manoeuvre. Large in-plane manoeuvres (via Orbit Control Mode, OCM) for initial orbit acquisition or repeat-cycle changes are also undertaken by using an OCM sub-mode. Small firings in the plane of the orbit, through the FCM, are used to compensate for air-drag effects. During these FCM manoeuvres, the satellite maintains its nominal attitude and the overall mission is unaffected, although pointing degradation occurs (Table 9).

Table 9. Performance of AOCS non-operational modes

AOCS Non-operational Mode	Axis	Budget (deg)	Requirements (deg)
Fine Acquisition Mode	all	4.90	5.0
Fine Control Mode	X and Y	0.27	0.2
	Z	0.33	0.7
Orbit Control mode	All for in-plane manoeuvres	1.10	2.0
	All for out-of-plane manoeuvres	1.70	2.0
Safe Mode	Z in sunlight	10.50	8.0
	Z in eclipse	17.50	20.0

Acknowledgement
Some of the engineering challenges involved in the long development of Envisat have been described in this article. The development effort has involved literally thousands of people. They should consider themselves thanked here. Obviously they are far too numerous to name. For the authors, it has been an immensely rewarding experience and we look forward to a successful mission.