The Venus Express Spacecraft System Design

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Although the Venus Express and Mars Express spacecraft are very similar, key modifications were made to meet the requirements of a Venus mission. This paper provides an overview of the main mission drivers that led to the design changes, and describes the main spacecraft functions.

1. Introduction

The Venus Express mission was proposed in 2001 in response to a call for ideas to reuse the Mars Express platform. The first studies demonstrated that a Venus orbiter mission could indeed be done by adapting the Mars Express spacecraft. Finally approved on 5 November 2002 with a launch date of October/November 2005, the development, integration and test of the spacecraft was limited to 3 years. With such a tight schedule and restricted budget, it was imperative to limit the modifications and adaptations of the Mars Express design to those essential for coping with the constraints of a Venus orbiter. An overview of these changes are provided in the first part of this paper, followed by a description of the spacecraft functions. On 11 April 2006, Venus Express was inserted into orbit around Venus, with the full range of resources available for an extended operational lifetime.

2. Mission Requirements

2.1 Reuse of Mars Express

The Venus Express spacecraft is designed to cope with the specific mission requirements of a scientific spacecraft orbiting Venus, with extensive reuse of the Mars Express design. As expected, the Venus Express spacecraft is very similar to Mars Express:

— the same system concept: a core box structure with body-mounted instruments, fixed radio antennas, and two solar arrays mounted on one-degree-of-freedom mechanisms (Fig. 1);
— the same avionics units and propulsion subsystem;
— the same store-and-forward operational concept: Venus observations during specific portions of the orbit, with onboard storage of the science data, alternating with Earth-pointing for ground communications and forwarding of the data.

However, four main Venus mission-specific features had to be taken into account, as described below.

2.2 New payloads

New payloads had to be accommodated (VIRTIS, VMC, VERA and MAG) and two payloads that were Mars Express design-drivers were removed (Beagle-2 and MARSIS). This was accomplished with no major changes to the structure. The

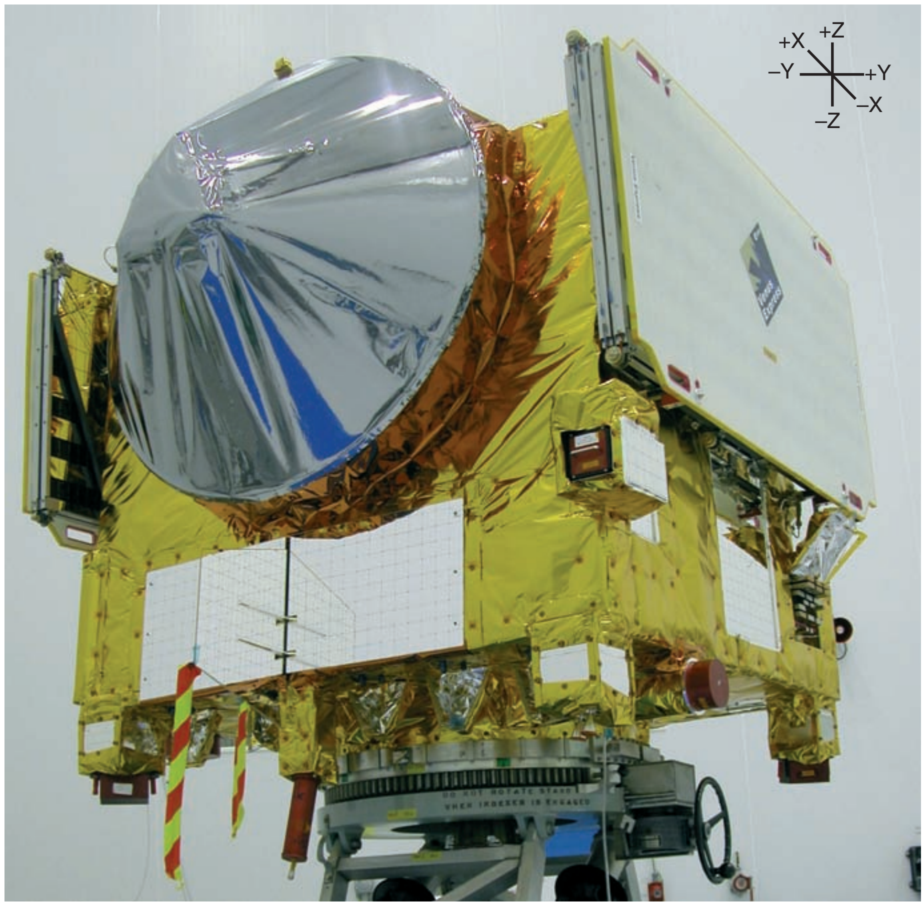


Fig. 1. Apart from solar array covers and a few minor items to be removed before encapsulation, Venus Express is seen here in its near-flight configuration in Baikonur, shortly before fuelling.

biggest challenge was VIRTIS, because the low temperature demanded by its infrared detectors was a very stringent requirement. It was achieved by coupling VIRTIS to a dedicated radiator on the cold face of the spacecraft, always turned away from solar illumination.

2.3 Thermal flux

The thermal flux at Venus is four times that at Mars – 2600 W m^{-2} (or double that at Earth). The main impact was to increase radiator power. Conversely, a consequence is that more heating is necessary during cruise and Earth-pointing. This led in turn to minor changes to the power subsystem, in particular the increase of the battery discharge regulator capability from 250 W to 300 W. The other important consequence was the major design change to the solar array. It was proved that silicon cells, as used on Mars Express, was not suitable for Venus Express. The Venus thermal environment imposes a very wide temperature range on solar cells, and thus a wide voltage range that could not be handled by the existing Power Control Unit. Gallium arsenide cells were substituted, since they are much less sensitive to temperature and to the radiation environment.

In addition, external coatings were modified in order to minimise the thermal flux entering the spacecraft. The new selection of coatings, although based on existing flight-proven materials, called for an extensive set of ground tests to validate their use at Venus.

To increase flexibility in adjusting the heater power before launch on critical elements of the propulsion system, heater distribution units were added, with resistors for fine tuning.

Table 1. Venus Express principal features.

End nominal mission	September 2007 (after 2 Venus days or ~486 Earth days)
End extended mission	January 2009 (additional 2 Venus days or ~486 Earth days)
Baseline operational orbit	pericentre 250 km x apocentre 66 600 km, inclination 90°, argument of pericentre 101°
Ground stations	Cebreros, Spain (X-band), New Norcia, Australia (X/S-band, mainly for radio science)
Operations Centres	Mission Operations Centre: ESOC, Darmstadt (D); Science Operations Centre: ESTEC, Noordwijk (NL)
Payload	ASPERA, MAG, PFS, SPICAV, VERA, VIRTIS, VMC
Mechanical	
Structure	box-like 1.7 m length x 1.7 m width x 1.4 m height
Total launch mass	1245 kg (of which 94 kg payload)
Propellants	570 kg (354 kg MON, 216 kg MMH)
Structure stiffness	first eigenfrequencies at 17 Hz lateral X, 15 Hz lateral Y, 34 Hz axial Z
Thermal	
Radiative area	1.7 m^2 on Y panels, 0.3 m^2 on X panel + payload-dedicated radiators
Heating system	780 W heater power distributed over 16 redundant heater lines using bimetallic thermostats or ON/OFF regulation via software control
Multi-Layer Insulation	mostly 23 Kapton layers, embossed on external side
AOCS/Propulsion System	
Concept	3-axis stabilised, sensors and actuators connected to AOCS Interface Unit (internally redundant) with core AOCS software residing in one processor of the CDMU
Observation pointing performance	absolute pointing error < 0.06°, relative pointing error < 0.006° over 30 s
Sensors	2 cold-redundant startrackers with 16.4° circular FOV, sensitivity to magnitude 1.7–5.5, tracking up to 9 stars at 0.17°/s tracking rate; 2 inertial measurement units, each with 3 gyros (angular input rate ±15°/s) and 3 accelerometers (acceleration input ±25 g); 2 Sun-acquisition sensors with 18x18° FOV, accuracy 1°
Actuators	4 reaction wheels, 12 Nms angular momentum at max 6000 rpm; 2 solar array drive mechanisms, with 8 speed levels up to 1.5°/s, harness and signal lines routed via twist capsule allowing ±180° motion
Propulsion	bipropellant (NTO, MMH); high-pressure helium (276 bar) regulated to 17 bar for capture operations using a main engine (414 N thrust, 317 s specific impulse); 4 primary, 4 redundant 10 N thrusters also operating in blowdown mode
Power	
Generation	2 stowable solar wings, each with 2 panels of 1.4 m width, 4 hold-downs released by pyro cutting of rods, deployment by spring-actuated hinges with synchro cable. Total of 1056 triple-junction GaAs cells over a total cell area of 2.6 m^2 per wing; performance BOL near Earth 820 W, EOL Venus 1400 W
Storage	3 Li-Ion batteries, 24 Ah capacity; 16 parallel strings of 6x1.5 Ah cells in series per battery
Power Conditioning Unit (PCU)	fully regulated 28 V (±1%) power bus, using 3 domain-control system (Array Power Regulation, Battery Charge Regulation, Battery Discharge Regulation); performance 1500 W in APR mode, 900 W in BDR mode; max charge current 3 A or 0.95 A (selectable)
Power Distribution Unit (PDU)	39 nominal and 39 redundant power lines fitted with Latching Current Limiters; 3 nominal and 3 redundant power lines fitted with Foldback Current Limiters; 32 nominal and 32 redundant pyro lines fitted with safety barriers; 750 W maximum power handling capability
Heater Distribution Unit (HDU)	6 HDUs (3 nominal, 3 redundant) for adjustment of propulsion system heater power
RF Communications	
Uplink	S- or X-band, NRZ/PSK/PM modulation on a 16 KHz sine subcarrier; telecommand bit rates 7.8–2000 bit/s
Downlink	S- or X-band, PCM/PSK/PM modulation for info rates up to 22.5 Kbit/s on square wave subcarrier, for higher info rates, direct modulation on carrier; telemetry bit rates from 9 bit/s to 228 Kbit/s; Reed Solomon/convolutional coding
High Gain Antenna (HGA)	HGA1: dual-band Cassegrain system antenna, 1.3 m diameter; half 3 dB beamwidth 3° S-band, 0.8° X-band; power handling 5 W S-band, 65 W X-band HGA2: single band (X), single offset parabolic reflector, 0.3 m diameter and corrugated horn; half 3 dB beamwidth 4° X-band; power handling 65 W X-band
Low Gain Antenna (LGA)	2 x hemispherical quadrifilar S-band omnidirectional antenna; right-hand circular polarisation, coverage 0–95°; power handling 10 W
Other RF equipment	waveguide interface unit and switches to transfer uplink/downlink to the right transponder/antenna combination
Data Handling	
Command & Data Management Unit (CDMU)	set of 2 cross strapped CDMUs containing: 2x2 processor modules, 2x1 transfer frame generator, 2x2 reconfiguration modules, 2x1 high power command modules, 2x1 memory modules with 512 Kword PROM and safeguard memory (with 64 Kword RAM and 64 Kword EEPROM)
Remote Terminal Unit (RTU)	2x146 command lines (high and extended high power commands); 2x304 acquisition lines (temperatures, digital, relay status, analogue); pulse and clock functions
AOCS Interface Unit (AIU)	internally redundant unit containing: 2 interface modules to all AOCS sensors (1355, RS422 and MACS I/F) and CDMU; 2 interface modules to command thrusters, main engine and acquire internal/external AIU telemetry (temperatures, pressures, status)
Solid-State Mass Memory (SSMM)	common kernel made of 3x4 Gbit memory modules using 64 Mbit SDRAM memory devices; 2 redundant controllers

2.4 Planetary configuration

From Mars, the Earth is always within 40º of the Sun, which conveniently allows antennas to point at Earth and the solar array to the Sun at the same time as the cold side of the spacecraft faces deep space. At Venus, this is not possible. As seen from the spacecraft, the Sun and Earth can be at any angle, so special measures are needed to hold the cold side away from the Sun. To achieve this, a second High Gain Antenna (HGA) was added, facing the opposite direction to the main HGA. Their alternate use, combined with an optimised attitude guidance law, restricts Sun illumination to only two faces (+X and +Z) during steady-state communications with Earth.

2.5 Venus gravity

Venus gravity is stronger than Mars gravity: 0.81 Earth gravity *vs.* 0.38. As a consequence, greater delta-V is needed for orbit capture (1251 m/s *vs.* 814 m/s), in turn requiring 100 kg more propellant than on Mars Express. However, the Mars Express tank capacity and structure were sufficient.

3. Design Overview

3.1 Overview of main features

A summary of the main features of the mission and spacecraft design is given in Table 1.

3.2 Mechanical design

The mechanical design was driven by the following considerations:

— reuse of the Mars Express mechanical bus as far as possible;
— the specific constraints of the Venus Express mission;
— the need to minimise the spacecraft dry mass and optimise the location of the centre-of-mass.

The reuse of the Mars Express mechanical bus (structure and propulsion system) minimised the development risks and helped to secure the programme's very tight schedule. Drawing on the Mars Express qualification, the core structure design remained basically unchanged, which allowed qualification by similarity. The modifications to the secondary structure were strictly limited to accommodation of the new or modified units. Reusing the bus also meant that most of the Venus Express units have the same mechanical environment as on Mars Express. The main mechanical design modifications relate to the payloads, the additional HGA and the constraints from the thermal design.

As on Mars Express, the overall spacecraft mass was close scrutinised. The maximum mass of the spacecraft (including propellants) allowed by the launcher was agreed at 1270 kg. Through careful mass management, the propellant tanks could be filled to their maximum capacities.

Close attention was paid to the centre-of-mass and the alignment of the main engine. During the main engine burn for Venus orbit insertion, the disturbing torques imposed on the spacecraft were directly linked to the offset between the direction of thrust and the location of the centre-of-mass, which migrated as propellant was depleted. To improve the control torque margins, strict control of the centres-of-mass of individual units was maintained and balance masses were added for fine adjustment.

3.3 Thermal control

The thermal control design is robust and passive with maximum commonality with Mars Express. However, some specific modifications were made to cope with the hot environment at Venus. The key thermal control features are presented in Figs. 2 and 3, and are summarised as:

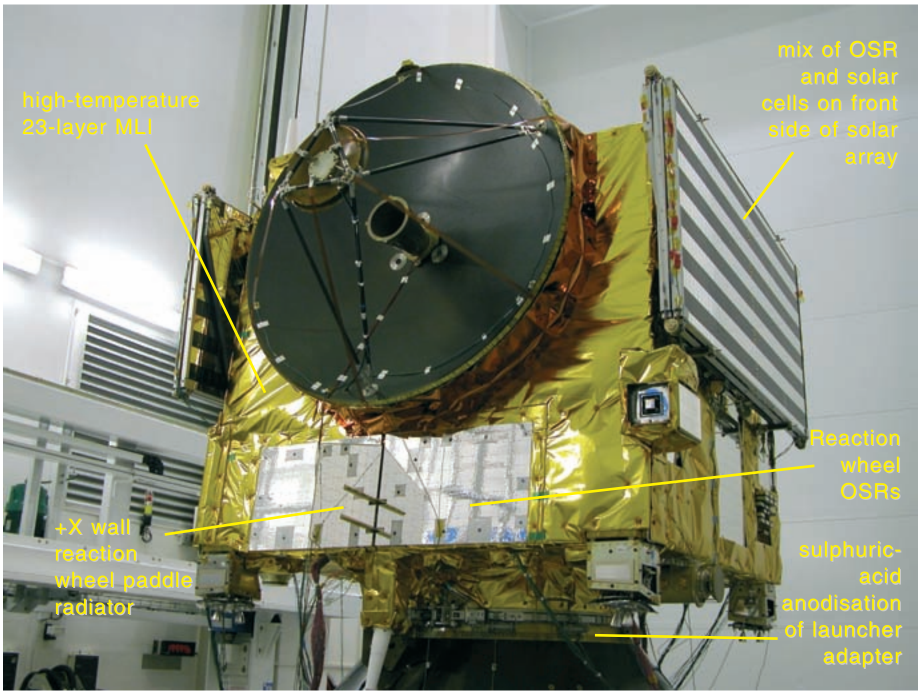


Fig. 2. Venus Express during integration at Intespace (Toulouse, F), showing the thermal control of the +X wall and the solar array. The MultiLayer Insulation (MLI) and Optical Solar Reflectors (OSRs) are clearly visible.

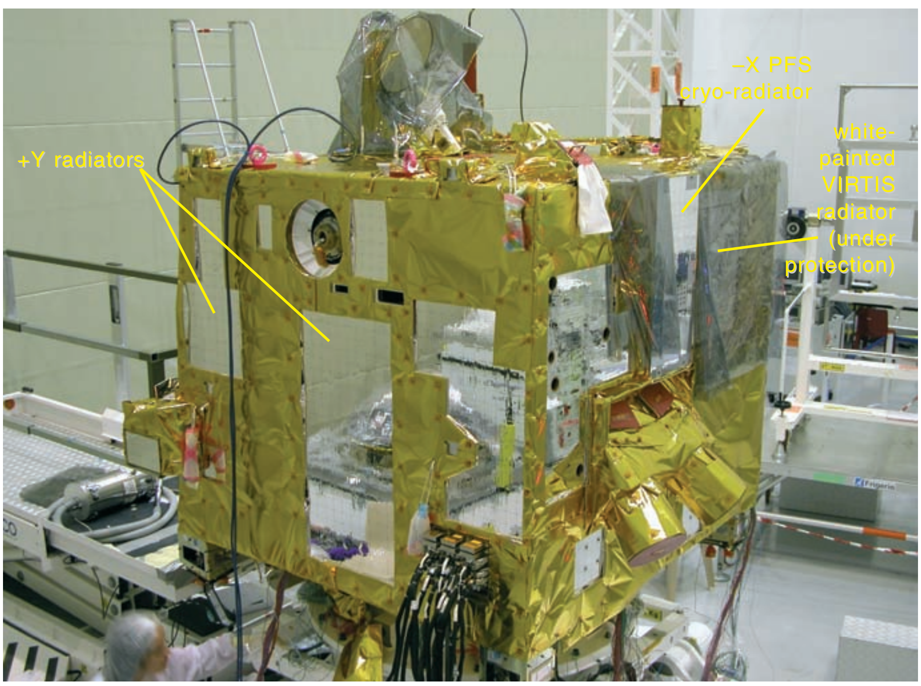


Fig. 3. Venus Express during integration at Intespace (Toulouse, F), showing the thermal control of the +Y and –X walls.

— the heat rejection towards space is via radiators mainly on the ±Y panels for the internal units and on the –X panel for the payload equipment. These sides are the most favorable areas, as they are protected from direct Sun most of the time. The rest of the spacecraft has Multi-Layer Insulation (MLI) blankets to minimise heat exchange and temperature fluctuations.

— VIRTIS and PFS have dedicated radiators because of their need for low operating temperatures. Radiative areas of the +X face exposed permanently to the Sun and through which reaction wheels reject heat were redesigned to cope with the solar flux at Venus.

— high-dissipation units are mounted directly behind the radiators in order to provide good conductive paths. Where dissipation was insufficient, the qualification temperature of some units was raised to cope with the hotter environment.
— heat pipes were added under the Power Conditioning Unit (PCU) and Power Distribution Unit (PDU) to spread the high PCU thermal dissipation evenly.
— thermal straps connect the reaction wheels and those payload units with dedicated radiators (PFS, SPICAV,VIRTIS).
— cells and Optical Solar Reflectors (OSRs) were mixed on the front side of the solar array panels. The rear side was completely covered with OSRs.
— the Alodine treatment of the launch vehicle adapter ring was replaced by a clear sulphuric anodisation to minimise the temperature under solar illumination.

Significant emphasis was placed on the selection of external coatings and MLI. The MLI has up to 23 layers, with no spacer material on the external layers for better resistance at higher temperatures. Care was taken with the fitting overlaps to avoid damage through Sun-trapping. On the external side, low solar absorptance and low-ageing coatings are used. To avoid multiple reflections and to diffuse the direct sunlight, most of the blankets have an embossed external layer. Despite these constraints, only flight-proven materials were used because it was not possible to embark on new developments in the time available. However, extensive accelerated testing was performed on the existing materials to verify their behaviour in the extreme Venus environment.

3.4 Attitude and Orbit Control System

3.4.1 AOCS overview

The key to the complex operability of the Venus Express spacecraft shown in Fig. 4 is the Attitude and Orbit Control System (AOCS, Fig. 5). The characteristics of the mission and the fact that the spacecraft has fixed HGAs and a single rigidly-mounted main engine means that there are demanding manoeuvring requirements. The autonomous startracker with its extensive star catalogue ensures that accurate attitude estimation can be achieved in almost any position. This has to be supported by miniature Inertial Measurement Units (IMUs) when the startracker field of view is obscured by a planet or the Sun. Four reaction wheels are used for most of the attitude manoeuvres for specific activities such as Earth communication and scientific observations; they provide flexibility and reduce overall propellant consumption. The wheels' angular momentum has to be carefully managed from the ground; regular off-loading keeps the wheel speeds within performance limits.

All of the sensors and actuators are connected to the AOCS Interface Unit (AIU) through various links: an IEEE 1355 bus for the startracker, a MACS bus for the solar array drive mechanism, an RS 422 link for the IMU, and direct hardwire links in the case of the Sun Acquisition Sensor and the reaction wheels.

This is all under the control of the Control & Data Management Unit (CDMU), which contains dedicated redundant processors for the AOCS software that also controls the processing of sensors and actuators, the estimation and control algorithms, and the management of the AOCS Failure Detection, Isolation & Recovery (FDIR) function.

3.4.2 AOCS units

The startracker is the main optical sensor of the AOCS and is used at the end of the attitude acquisition following each manoeuvre to acquire the 3-axis pointing required for almost all nominal operations. The startracker includes a star pattern-recognition function and can perform attitude acquisition autonomously. It has the same basic design as that on Mars Express with

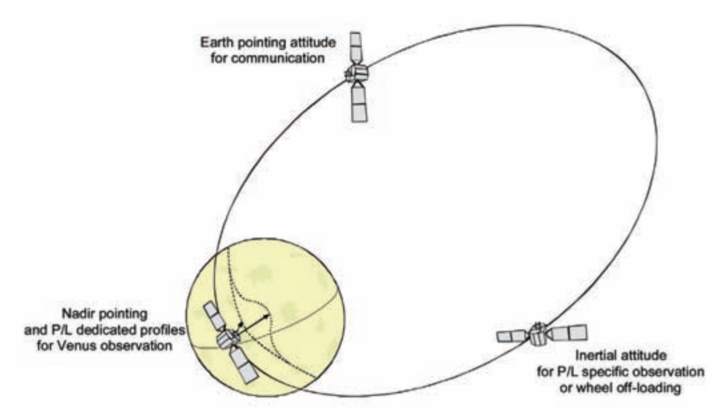


Fig. 4. The manoeuvrability of Venus Express required for Venus nadir observations, Earth pointing attitudes and inertial attitudes for specific payload (P/L) observations.

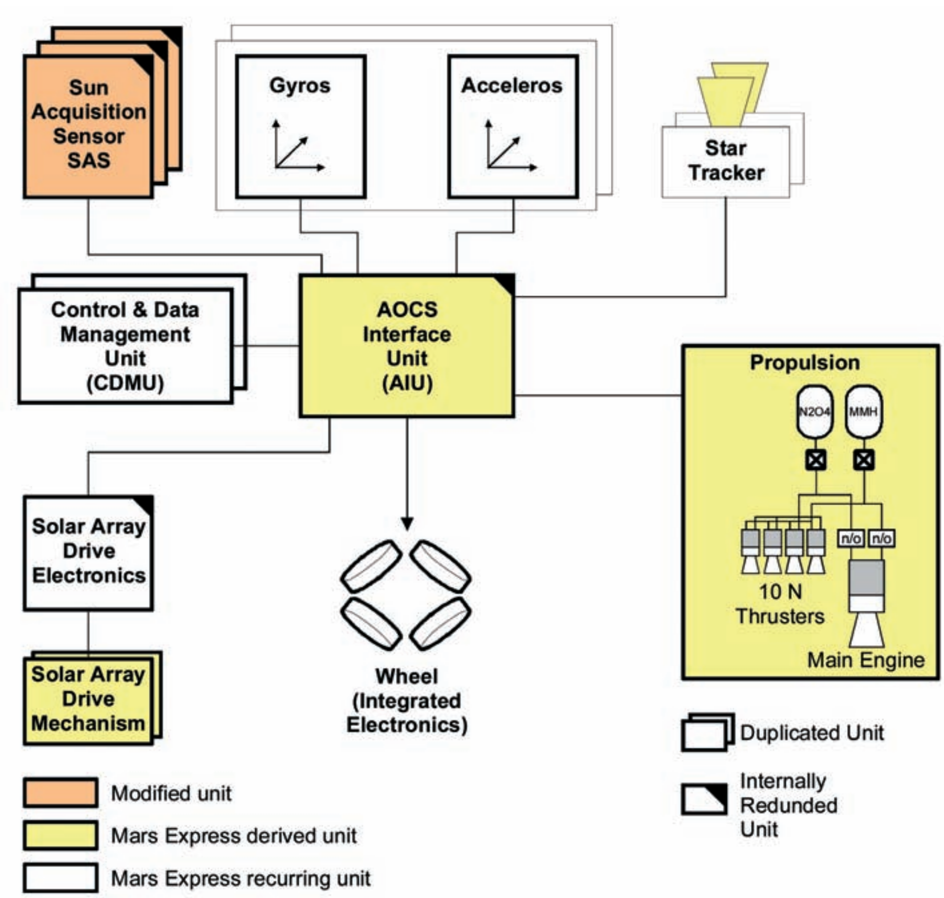
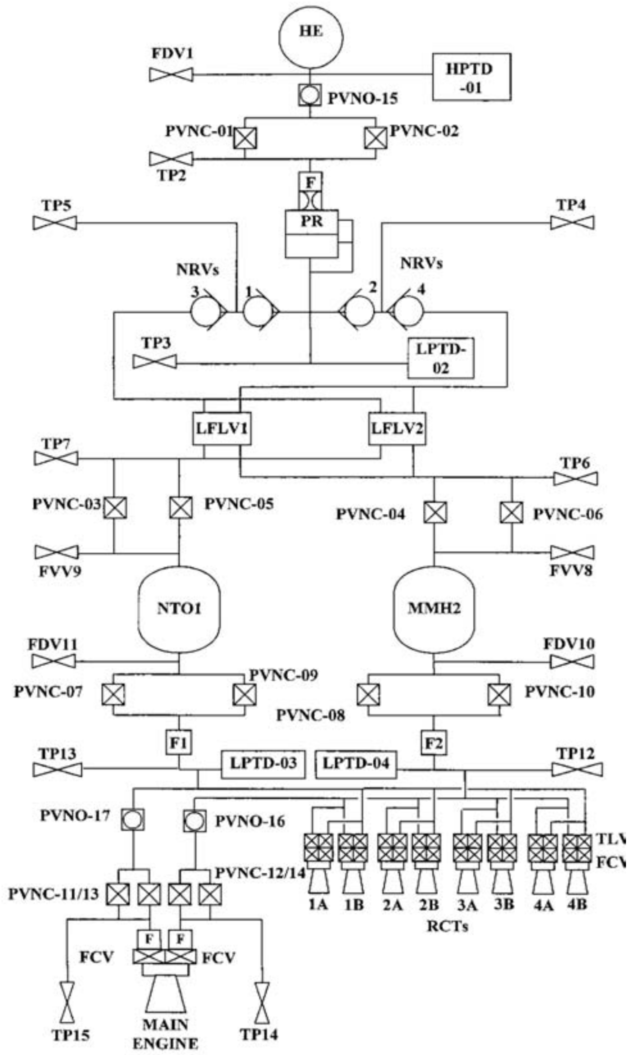


Fig. 5. The AOCS system.

modifications learned from the Mars experience. These included a new diaphragm to improve the robustness against straylight and improvements to the star catalogue. Venus Express has two startrackers aligned with an angle of 30º between their optical axes for operational redundancy and to ensure that one can always see a recognisable region of the sky.

Each of the two IMUs has a set of three gyros and three accelerometers aligned along three orthogonal axes. The AOCS can use either the three gyros of the same IMU or any combination of three gyros among the total of six. Only a full set of accelerometers from one single IMU is used; they are essential to improve the accuracy of manoeuvres performed with the main engine. The gyros are used during attitude acquisition phases for rate control, during observation and communication phases to ensure the required pointing performance and

Fig. 6. The Venus Express propulsion system.
F: Filter.
FCV: Flow Control Valve.
FDV: Fill & Drain Valve.
FVV: Fill & Vent Valve.
HE: Helium Pressurant Tank.
HPTD: High-Pressure Transducer.
LFLV: Low-Flow Latch Valve.
LPTD: Low-Pressure Transducer.
MMH2: MonoMethyl Hydrazine Propellant Tank.
NRV: Non-Return Valve.
NTO1: Nitrogen Tetroxide Propellant Tank.
PR: Pressure Regulator.
PVNC: Pyrotechnic Valve Normally Closed.
PVNO: Pyrotechnic Valve Normally Open.
RCT: Reaction Control Thruster (dual valve).
1A–4A primary RCTs;
1B–4B redundant RCTs.
TLV: Thruster Latch Valve.
TP: Test Port.



during the trajectory correction manoeuvres for control robustness and failure detection. The non-mechanical technology removes the possibility of mechanical failure during the mission.

Two redundant Sun Acquisition Sensors (SAS) mounted on the spacecraft central body ensure pointing in the Sun Acquisition Mode (SAM) at the time of first acquisition after launch or any subsequent reacquisition in the case of a safe mode following an onboard failure. The Mars Express sensors were modified, using different solar cells with ceramic backing in order to cope with the thermal environment at Venus.

The Reaction Wheel Assembly (RWA) comprises four individual wheels in a skewed configuration to manage the spacecraft momentum in all three axes. Although nominally all four wheels are used for operations, this configuration, which is identical to Mars Express, allows full performance with any three-wheel configuration.

The Solar Array Drive Mechanisms (SADMs) control the orientation of the solar array. Identical to those on Mars Express, they have stepper motors and employ twist capsule technology.

3.4.3 AOCS modes

The AOCS includes several modes for attitude acquisition/reacquisition, the nominal scientific mission and orbit control.

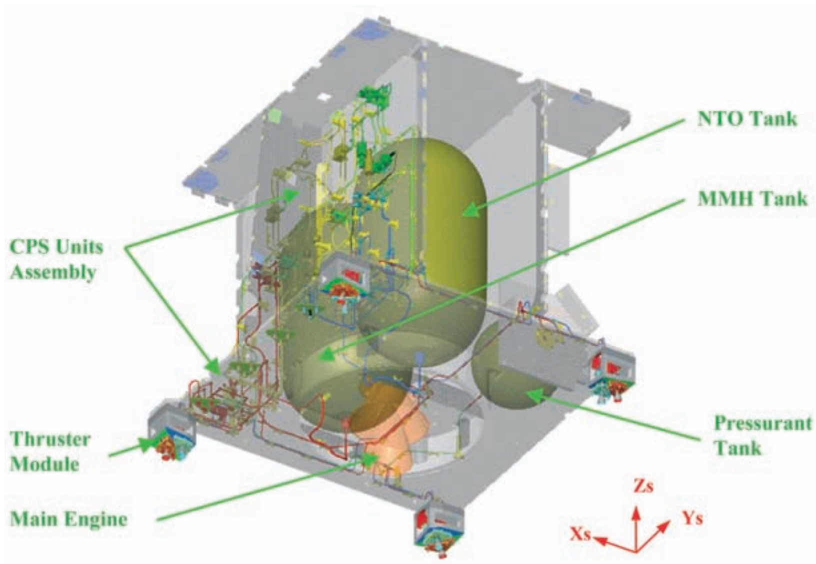


Fig. 7. Layout of the propulsion system.

The attitude acquisition and reacquisition sequence has two basic modes:

- the Sun Acquisition Mode (SAM), pointing the X-axis and the solar array towards the Sun in order to guarantee power;
- the Safe/Hold Mode (SHM), which completes the acquisition phase and provides the final 3-axis pointing, with one HGA pointing towards the Earth to guarantee telemetry and telecommanding.

Trajectory correction manoeuvres are performed using three modes:

- the Orbit Control Mode (OCM), for small corrections using the 10 N thrusters;
- the Main Engine Boost Mode (MEBM), for major manoeuvres using the 400 N engine;
- the Braking Mode (BM, so far unused) would allow aerobraking should it be necessary for emergency of scientific purposes.

Finally, to ensure a smooth transition between the thruster-controlled modes and wheel-controlled modes, there is also a Thruster Transition Mode (TTM).

3.5 Chemical Propulsion Subsystem

The Chemical Propulsion Subsystem (CPS) is a helium-pressurised bipropellant system, using monomethyl hydrazine (MMH) as the fuel and nitrogen tetroxide (NTO) with 3% nitric oxide as the oxidiser. The main engine, essential for Venus orbit insertion, has a thrust of 414 N and a specific impulse of 317 s. Four pairs of 10 N thrusters (four primary, four redundant) provide trajectory corrections and attitude control and reaction wheel unloading. Inherited from the Eurostar 2000 telecommunications platform, they are the same as those used on Mars Express (Fig. 6). The complete layout of the CPS is shown in Fig. 7. The CPS operated in a constant-pressure mode during main engine firings for the capture manoeuvre and the first part of the apocentre reduction manoeuvre, using a regulated helium supply.

The CPS comprises two subsystems: the pressurant subsystem and the propellant feed subsystem. The helium pressurant subsystem, commonly referred to as the ‘gas side’, has two sections: high-pressure and low-pressure. The high-pressure gas side comprises a 35.5-litre helium tank, normally-open and normally-closed pyrovalves, a high-range pressure transducer, a fill & drain valve, and a test port to support integration activities.

This section has a maximum expected operating pressure (MEOP) of 276 bar. During all ground operations and launch, it was isolated from the pressure regulator by a pair of normally-closed pyrovalves. These are arranged in parallel for redundancy.

The low-pressure gas side comprises a pressure regulator, non-return valves, a pair of low-flow latch valves, a low-range pressure transducer, normally-closed pyrovalves, and test ports and fill & vent valves. This section has a MEOP of 20 bar, controlled by the regulator that senses downstream pressure.

The propellant feed subsystem, commonly referred to as the 'liquid side', supplies propellants to the main engine and thrusters. It comprises a pair of 267-litre propellant tanks, normally-open and normally-closed pyrovalves, propellant filters, low-range pressure transducers, main engine, reaction control thrusters, and test ports and fill & drain valves. This section is pressurised with helium from the low-pressure gas side, and has a MEOP of 20 bar.

3.6 Power architecture

3.6.1 Power subsystem overview

The Venus Express power architecture is shown in Fig. 8. Electrical power is provided by two solar wings equipped with triple-junction GaAs cells. The array is oriented towards the Sun by two SADMs. During eclipses, power is provided by three lithium-ion batteries that recharge after the eclipse. Power management and regulation is performed by the Power Control Unit (PCU) that provides a regulated 28 V main bus. The PCU uses a Maximum Power Point Tracker (MPPT) in order to operate at the maximum power output of the solar array, which avoids the need to oversize the solar array to cope with both near-Earth and Venus orbit conditions. Battery management is performed using three Battery Charge and Discharge Regulators (BCDRs) under the control of a Main Error Amplifier (MEA) control loop. The resulting +28 V regulated bus is distributed to all spacecraft users by a Power Distribution Unit (PDU) featuring Latching Current Limiters (LCLs), which protect the bus from overcurrents at unit level. The PDU is also responsible for generating the commands for firing the pyrotechnics.

3.6.2 Solar array

The solar array consists of two identical low-weight deployable wings (Fig. 9a), each having two solar panels pointed towards the Sun by a one degree-of-freedom SADM. When stowed, each wing was clamped to the spacecraft side panel on four hold-down points and release mechanisms. For deployment (which was performed autonomously after launch as part of the separation sequence), redundant pyrotechnic bolt cutters released each wing individually. The electrical power is transferred to the spacecraft by a harness routed on the rim of the wings onto the connectors of the SADM (Fig. 9b). In order to meet the stringent requirements associated with the Venus radiation environment, the chosen solar cell technology was GaAs with 100 μm cover glass. The maximum array current is 18 A per wing. The total array power values are of the order of 820 W near Earth and 1400 W at Venus (end-of-life).

3.6.3 Power storage

Three batteries supply the spacecraft power if either the solar array is not illuminated by the Sun or if the power demand is higher than can be generated by the array. The energy is stored in three identical 24 Ah low-mass Li-ion batteries with a total capacity of around 500 Wh. Each has 16 parallel strings of six serial 1.5 Ah cells. The batteries are identical to those on Mars Express.

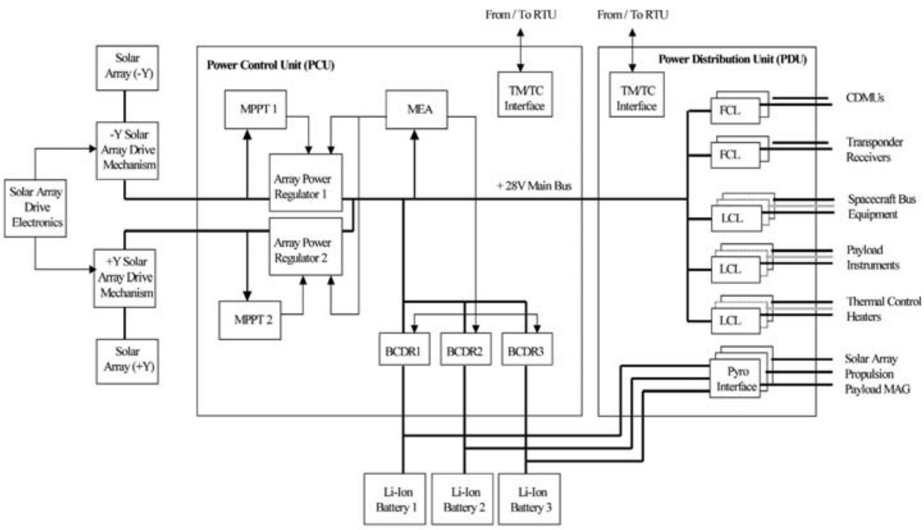


Fig. 8. Venus Express power system architecture.

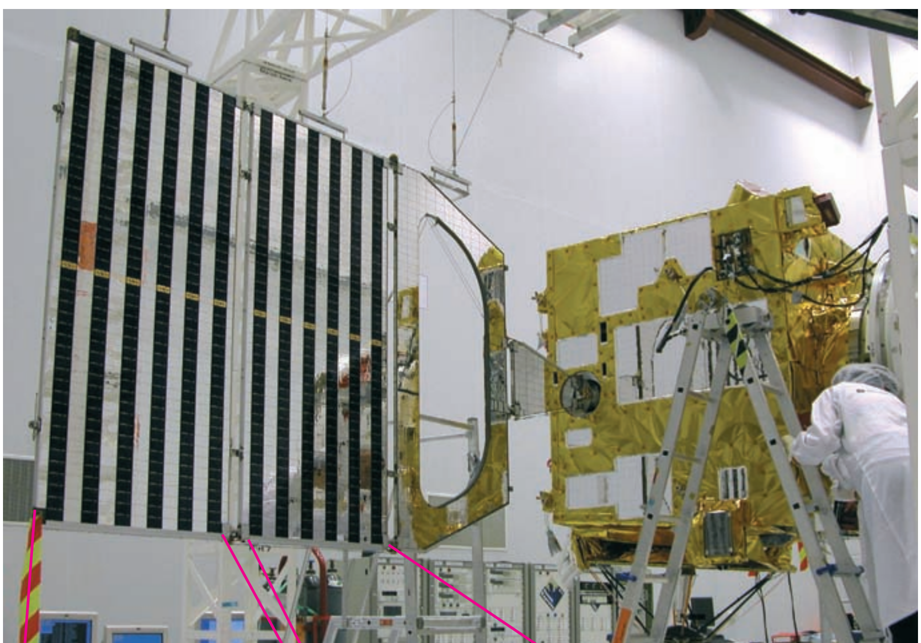


Fig. 9a. The solar array during integration and deployment testing at Intespace.

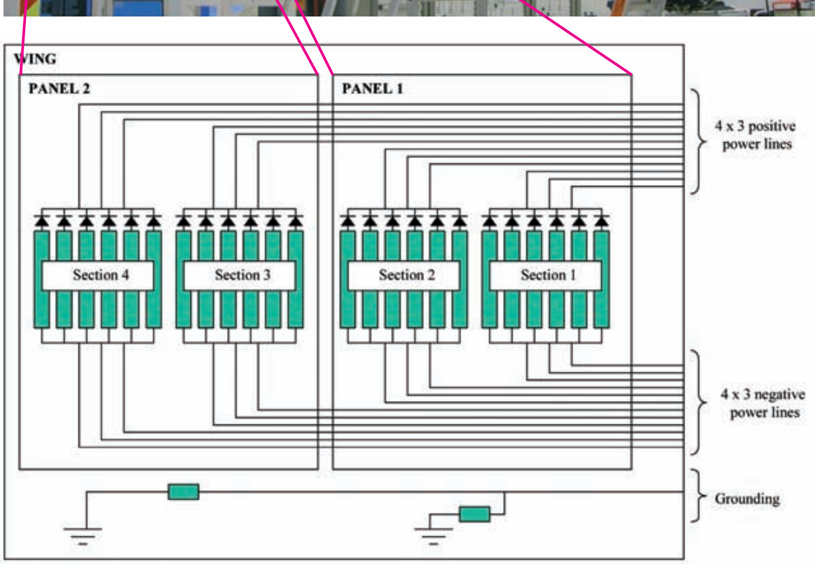


Fig. 9b. Solar wing block diagram.

3.6.4 Power control

The PCU converts the solar array and/or battery power inputs into a regulated main bus voltage of 28 V. The main bus regulation is performed by a conventional three-domain control system, based on one common and reliable MEA signal that controls the two Array Power Regulators (APRs, one for each solar wing) and the three BCDRs (one for each battery). Power management is further achieved by measurement of the power parameters within the PCU itself. When the available array power exceeds the total power demand from the PCU, including the battery power charge, the APR performs the main bus regulation based on the MEA control line signal.

In the event that the MEA signal enters either the Battery Charge Regulation or Battery Discharge Regulation domains, the MPPT tracking function automatically takes over. This drives the operating voltage of the solar array to the point where maximum power can be obtained.

3.6.5 Main bus power distribution

Power distribution is based on a centralised scheme performed by the PDU. One protected power line derived from the regulated main power bus is dedicated to each DC/DC power converter of spacecraft users. In addition, power lines are available for users that draw power directly from the power bus without internal DC/DC conversion, as is the case for the 10 N thruster flow control valves, flow control valves of the main engine coils and of the latch valves.

Each power line is switched and protected by an LCL, which is a solid-state switch that also acts as a protection device in case of overcurrent. Units that may never be switched off (CDMU and transponder) have to be able to recover autonomously. For these units, primary power is distributed through Foldback Current Limiters (FCLs). These are devices identical in principle to LCLs except that they do not have ON/OFF switching capability and an overcurrent will not lead to a disconnection when the trip-off time is exceeded.

3.6.6 Heater distribution

Each Heater Distribution Unit (HDU) consists of a small mechanical box containing printed circuit boards with adjustment resistors for fine tuning the thermal control of the propulsion system. Three HDUs were allocated to nominal propulsion system heater lines and three to redundant heater lines. The maximum power dissipated into each unit does not exceed 14 W.

3.7 Radio-frequency communications

Communications with the Earth can be performed either in S-band or X-band in accordance with conventional ESA Standards. The radio-frequency (RF) communication subsystem consists of a redundant set of dual-band transponders operating in both S-band and X-band for either the uplink or the downlink. Depending on the mission phase, the transponder is routed via RF switches to different antennas. An overview is provided in Fig. 10.

The antennas on the spacecraft are:

— two LGAs, used primarily during the Launch and Early Operations Phase (LEOP), operating in S-band for omni-directional reception and hemispherical transmission;
— the dual-band HGA1, operating in S-band and X-band for high-rate telemetry and telecommand;
— the single-band offset HGA2, operating in X-band only, for high-rate telemetry and telecommand.

The dual-band transponder (identical to that on Mars Express) performs the demodulation of the uplink signal before rerouting to the CDMU data-handling

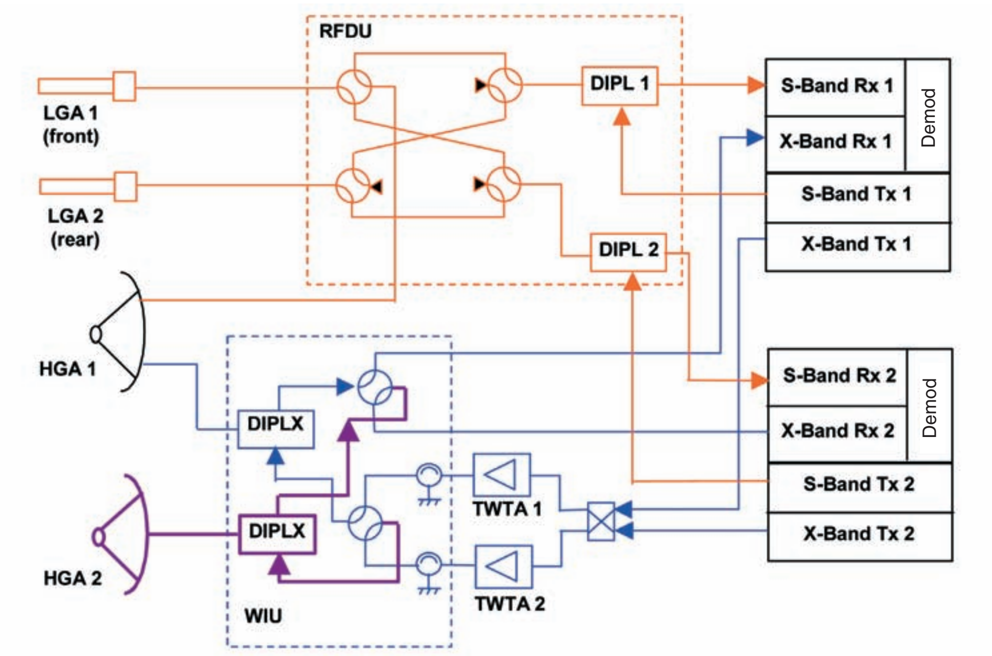


Fig. 10. RF communications block diagram. Purple denotes the hardware added to the Mars Express design: HGA2, diplexer and waveguide.

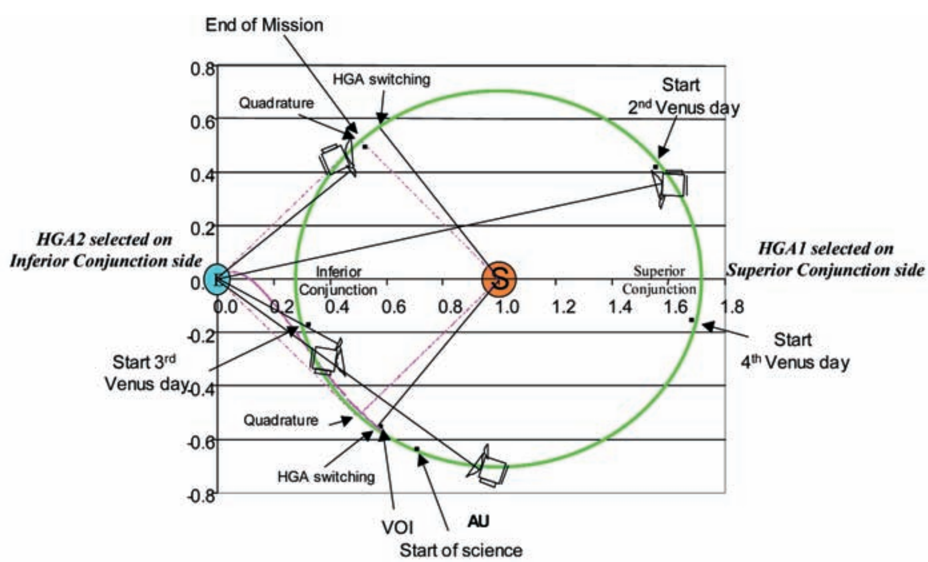


Fig. 11. HGA selection combined with an optimised attitude guidance law, shown in a fixed Earth (E)-Sun (S) reference frame.

subsystem. All data stored in the Solid-State Mass Memory (SSMM) are routed through the CDMU to the transponder for transmission to ground.

Following LEOP, all nominal operations are performed at X-band; the transponder output power is amplified using a 65 W Travelling Wave Tube Amplifier (TWTA). Selection of which HGA to use depends on the mission phase and, particularly, the relative positions of the Earth, spacecraft and Venus (Fig. 11).

To maintain thermal control for instruments, solar illumination of the –X side of the spacecraft (opposite to HGA1) is minimised. With steady-state Earth communications, the attitude guidance law ensures that the spacecraft +Z/+X plane remains in the Sun-spacecraft-Earth plane. This means that:

— no Sun impinges on the lateral sides (±Y sides);
— the solar array can be pointed towards the Sun;

— HGA1 or 2 can be pointed towards Earth;
— the cold side of the spacecraft (–X) remains facing cold space.

Before the Sun starts to impinge on the cold side, as the spacecraft and Venus orbit the Sun, the spacecraft is flipped to point the opposite antenna towards Earth. This switching between the HGAs is shown in Fig. 11.

3.8 Data handling

3.8.1 Data Management System overview

The Data Management System (DMS; Fig. 12) performs all the data-handling functions for the spacecraft, including:

— telecommand distribution throughout the spacecraft;
— telemetry data collection from the spacecraft and data storage;
— overall supervision and monitoring of the spacecraft and payload functions and health status;
— timing functions, including distribution of time and synchronisation information.

The DMS is based on a dual-processor architecture embedding standard communication links, including a standard onboard data-handling (OBDH) bus and high-rate IEEE 1355 serial data links. The OBDH bus is the data highway for data acquisition for platform units and payloads with a low data rate and for command distribution via the Remote Terminal Unit (RTU).

IEEE 1355 links are used between the Command & Data Management Unit (CDMU) processor and the Solid-State Mass memory (SSMM), the CDMU processor and the AOCS Interface Unit (AIU) and between the payloads with high data rate (VIRTIS and VMC) and the SSMM.

The DMS includes four identical Processor Modules (PMs), located in two fully redundant CDMU units. One PM is allocated to the DMS software (in charge of the management of the platform subsystems), while the other is

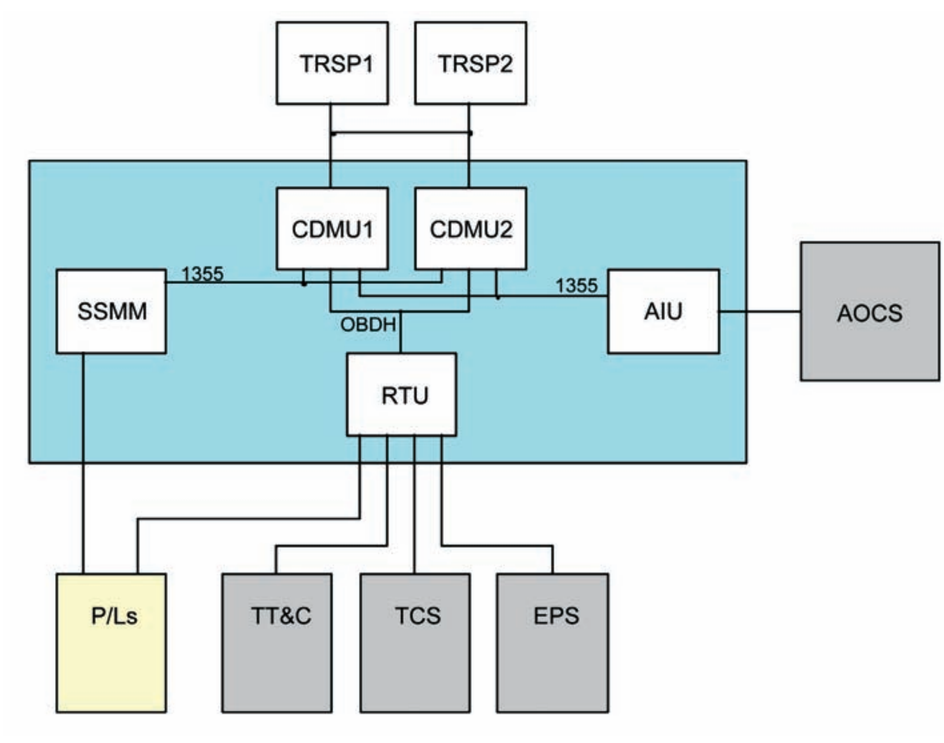


Fig. 12. The Data Management System. EPS: Electrical Power System. TCS: Thermal Control System. TRSP: Transponder. TT&C: Telemetry, Tracking & Command. Other acronyms are explained elsewhere.

allocated to the AOCS software (in charge of acquisition and control of all platform sensors, actuators and SADM via the AIU).

The data-handling architecture is organised around the two CDMUs. They are in charge of controlling ground command reception and execution, onboard housekeeping and science data telemetry storage and formatting for transmission. Onboard data management, control-law processing and execution of onboard control procedures are also part of their function. Each CDMU features two MA3-1750 PMs, each processing either DMS or AOCS software. A built-in failure operational Reconfiguration Module (RM) ensures system-level FDIR integrity and autonomously reconfigures the CDMU PMs as necessary.

Three other units are part of the data-handling subsystem:

— the RTU, connected to the OBDH bus, is the interface between the DMS PM and platform units and payloads;
— the AIU, dedicated to AOCS equipment, is the interface between the AOCS PM and the sensors, the actuators and the solar array drive electronics;
— the SSMM is a file-organised mass memory with 12 Gbit of storage that is used to store the housekeeping and science data collected by the CDMU. It also collects science data directly from VIRTIS and VMC using the IEEE 1355 bus.

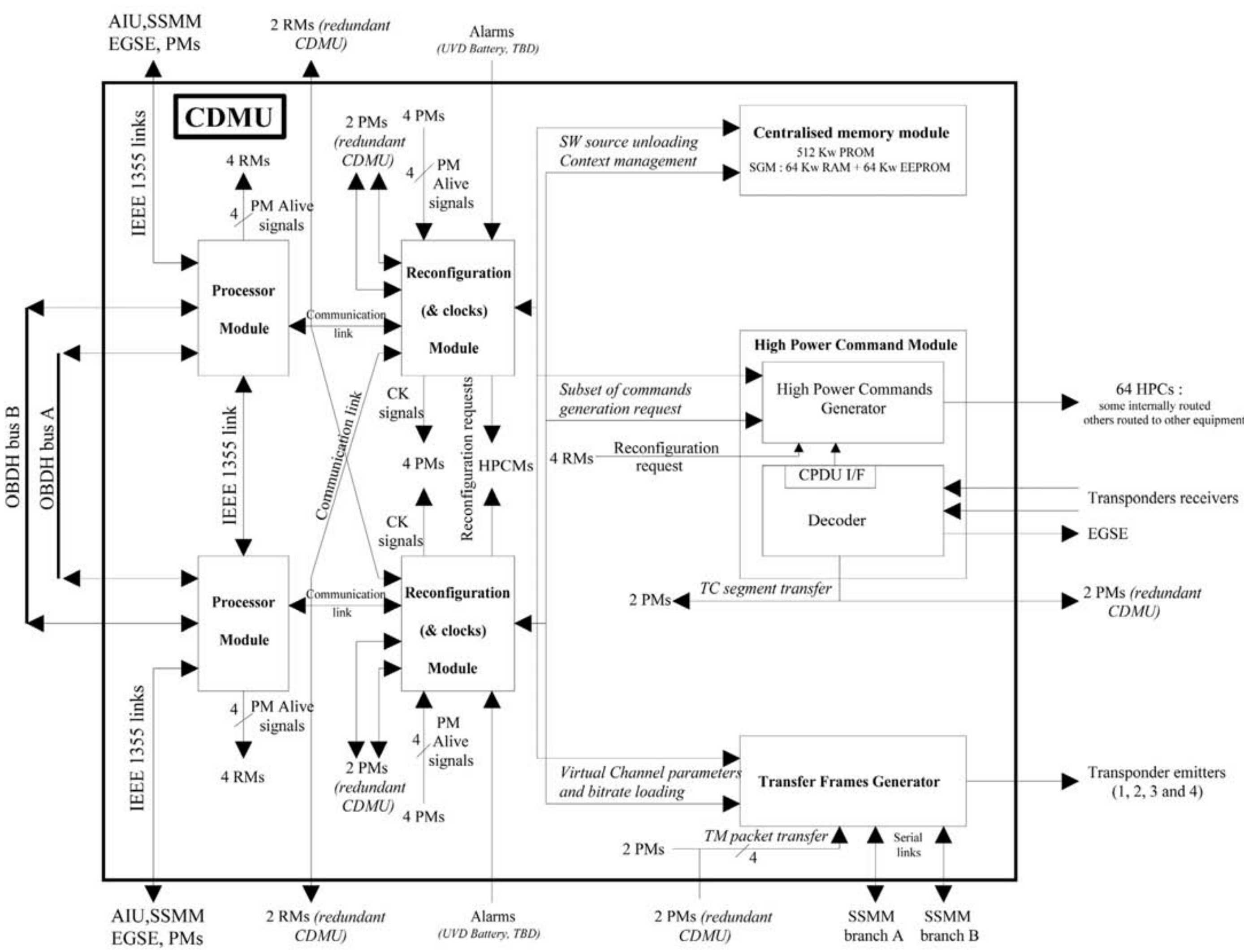


Fig. 13. The Venus Express CDMU. CK: Clock Signal. CPDU: Command Pulse Distribution Unit. EGSE: Electrical Ground Support Equipment. HPC: High Power Command. RM: Reconfiguration (and Clock Module). SW: software. UVD: Under-Voltage Detection.

3.9 CDMU

The CDMU is the core of the data-handling subsystem. Each CDMU features:

- two Processor Modules, one dedicated to AOCS software execution, the other to DMS software execution. The PM design is based on a flexible 16-bit MA3-1750 microprocessor, with 1 Mword of associated RAM and 512 Kword of EEPROM
- two Reconfiguration Modules, each containing an accurate clock function to maintain onboard timing and a watchdog function that, when triggered, sends a reconfiguration request to the High Power Command Module (HPCM);
- one HPCM containing:
  - a decoder that processes the telecommands transmitted by the transponders. Accepted telecommands will then be passed on either to the PMs or the High Power Command (HPC) generator via the Command Pulse Distribution Unit (CPDU).
  - a reconfiguration function that executes an autonomous reconfiguration of the CDMU when it receives as a minimum two of the four reconfiguration requests generated by the four RMs.
  - a Transfer Frame Generator (TFG) that contains three virtual channels (VC0 for realtime telemetry, VC1 for telemetry stored in the SSMM, and VC7 for the idle frames). It allows selection of convolutional coding and/or Reed-Solomon coding. The DMS PM delivers the subcarrier clock and the bitrate clock, which can vary from 8 bit/s to 250 kbit/s (depending on the Venus-Earth distance, the selected frequency band and on the spacecraft mode).
- one Centralised Memory Module (CMM) containing:
  - a PROM cassette including 512 Kwords of PROM accessible by the four PMs and containing the default DMS and AOCS software.
  - a Safe Guard Memory (SGM), containing 64 Kwords of RAM and 64 Kwords of EEPROM.
- two power supplies, one powering one PM and one Reconfiguration Module and the other powering the remaining parts of the CDMU.

3.10 Interface units

As for Mars Express, Venus Express uses a recognised interface unit concept that groups all data-handling interface functions with non-standard equipment into two dedicated units.

The RTU (Fig. 14) manages the interfaces with the instruments and all platform equipment not used by the AOCS through standard TTC-B-01 links. The RTU, which is internally redundant, contains six modules:

- two redundant core units in charge of processing the bus interrogations sent by the DMS PM on the OBDH;
- two input/output (I/O) boards interfacing with the users;
- two power supplies delivering the secondary voltages to the cores and I/O modules.

The following types of interface are implemented in the RTU:

- analogue acquisitions, such as equipment secondary voltages and currents;
- serial 16-bit digital acquisitions, such as payloads, transponders, power subsystem;
- bi-level digital acquisitions;
- relay status acquisitions;
- thermistor value acquisitions from, for example, the thermal subsystem and payloads;
- high power ON/OFF commands;

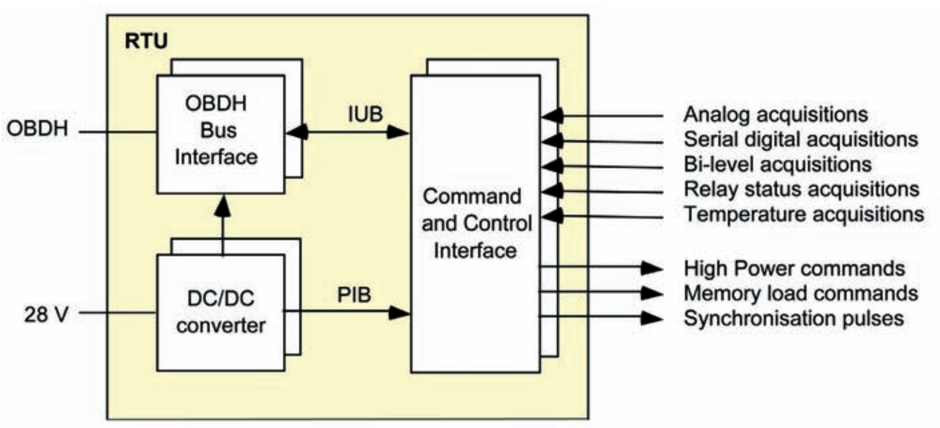


Fig. 14. The Venus Express RTU. IUB: Internal User Bus. PIB: Power Interchange Bus.

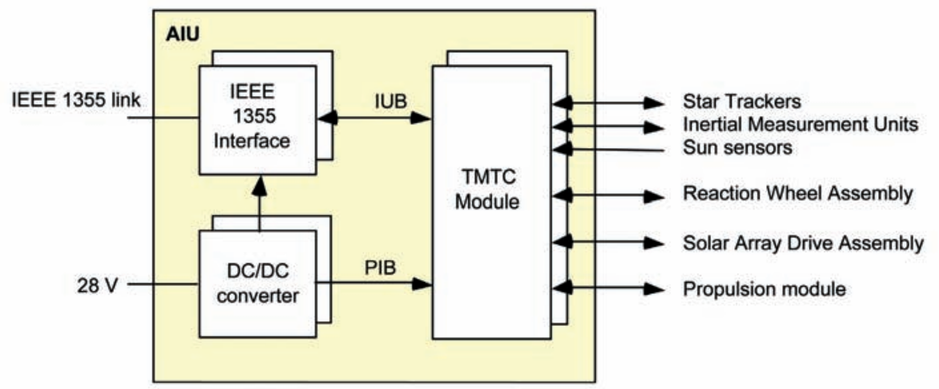


Fig. 15. The Venus Express AIU. IUB: Internal User Bus. PIB: Power Interchange Bus. TC: telecommand. TM: telemetry.

— extended high power commands (RF switches);
— memory load commands for, for example, payloads, transponders and power subsystem;
— timer synchronisation pulses for the startracker and SSMM, for example.

The AIU (Fig. 15) manages the interfaces with all the AOCS equipment. It acquires signals from the AOCS sensors and generates actuator commands according to control-law outputs that are provided by the AOCS software in the CDMU processor module. The AIU is internally redundant and contains:

— two interface modules interfacing with the AOCS PMs in the CDMU and controlling the generation of internal high-priority commands and the AIU internal bus;
— two TMTC boards that generate the commands to the thrusters and main engine, acquires the internal (secondary voltages) and external (AOCS units) telemetry and implements the interface with the reaction wheels assembly;
— two power supplies.

The following types of interface are implemented in the AIU:

— reaction wheel commands, tacho signal acquisition and analogue monitoring signal acquisition;
— programming of the startracker configuration, commanding and data acquisition;
— IMU data acquisition and power switching;
— acquisition of Sun sensor current;

- thruster and main engine commanding, and acquisition of current and temperature;
- latch valve commanding and status acquisition;
- monitoring of pressure transducer power supply and signal acquisition.

Communication between the DMS PM and the RTU is via a standard OBDH bus. Communication between the AOCS PM and the AIU is via a standard IEEE 1355 link

3.11 Solid-State Mass Memory

Venus Express uses the same store-and-forward concept employed for Mars Express, meaning that every orbit is divided into two principle phases: an observation phase, where the instruments are pointed towards Venus; and a communication phase, when an HGA is pointed towards Earth. To support this, all scientific data and housekeeping telemetry are stored in the SSMM (Fig. 16) with the following features:

- three 4 Gbit Memory Modules (MMs), providing a total of 12 Gbit user capacity. In case of failure of an entire memory module, the remaining capacity of 8 Gbit is sufficient to complete the nominal mission.
- two redundant controller paths, each providing:
  - a Memory System Supervisor (MSS) that performs the overall SSMM control and monitoring tasks;
  - a Processor module Interface Controller (PIC) that provides two bidirectional IEEE1355 interfaces to the DMS processor modules from which it receives the packets (housekeeping and science) and to which it sends the events and the housekeeping data and any other packets requested by ground.
- a User Interface Controller (UIC) that provides two bidirectional IEEE 1355 interfaces to the payloads with high data rate (VIRTIS and VMC), two interfaces with the TFGs of the CDMU and the interfaces to the memory modules.
- a File and Packet controller that controls and manages access to the MMs and also performs the file-management functions.
- an input/output Communication Switching Matrix (CSM).
- a DC/DC converter that provides the necessary voltages to the SSMM internal electronics. The controller board is powered in conjunction with the power converter. The MMs are switched on by command under control of the MSS.

3.12 Software

The Venus Express software covers the DMS and AOCS together with firmware for the CDMU, SSMM, startracker, gyros and transponder and the software of individual instruments. These software components are located in different hardware units and contribute to different aspects of the mission.

The DMS software runs on a dedicated PM in one of the two CDMUs. It comprises the Common software and the DMS application software. The Common software contains the PM hardware interface manager, the basic software services, the generic services and the OnBoard Control Procedure (OBCP) manager. The DMS application software manages the Mission Time Line (MTL), system autonomy and the system FDIR. It also manages directly or through the AOCS PM all the equipment and functions to fulfil the mission objectives:

- SSMM management, acquiring SSMM data, datapool storage and monitoring the SSMM health status;
- platform management that administers, commands and allocates all the

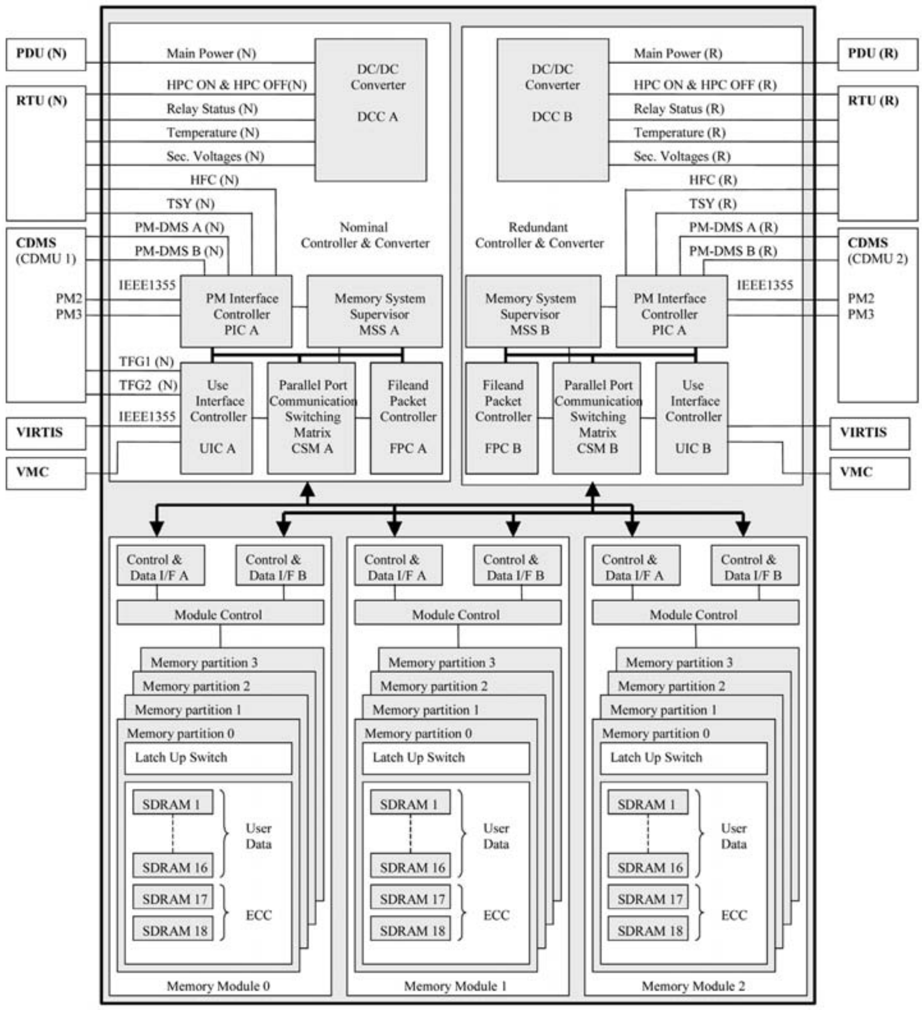


Fig. 16. The architecture of the Solid-State Mass Memory. HFC: High-Frequency Clock. TSY: Timer Synchronisation.

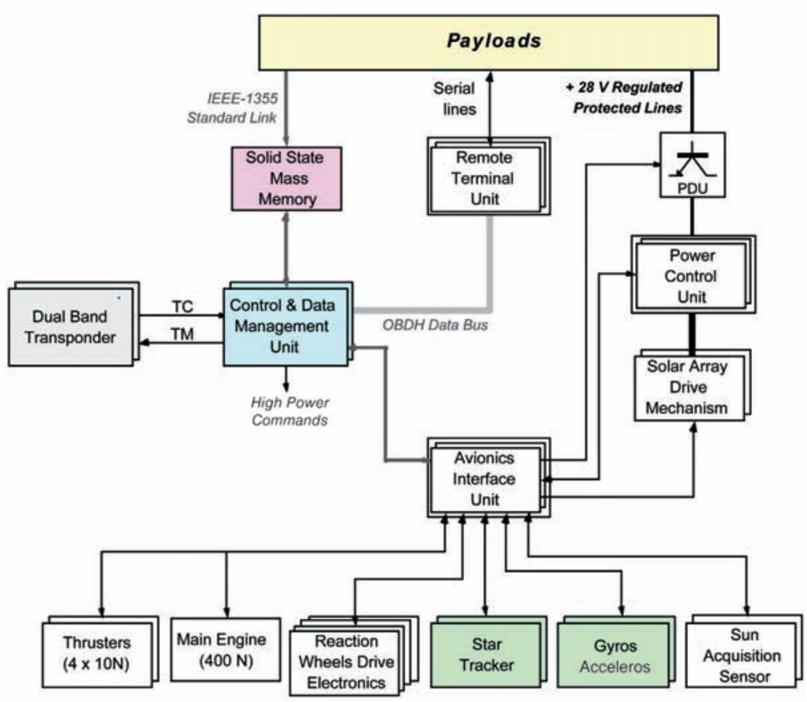


Fig. 17. Software locations.

platform bus resources (thermal control system, power system, RF system, pyrotechnics) with the exception of the AOCS units;
— payload management that sends telecommands to the payloads and receives data from them;
— remote (or Service Mode) PM management, providing the DMS software with knowledge of the condition of the remote (i.e. unused) PM state;
— mission management that sequences the nominal and contingency mission phases;
— FDIR management. Based on a hierarchical approach, the FDIR is handled at two levels by the DMS software: at DMS subsystem level monitoring the health status of equipment and managing local reconfigurations, and at system level monitoring the current functions to be fulfilled and managing functional modes reconfigurations.

The AOCS software runs on another CDMU PM. It comprises the Common software (in common with the DMS software) and the AOCS application software. The AOCS application software performs the AOCS modes and algorithm management and the AOCS unit and subsystem management. The specific functions of the AOCS application software include:

— resource management, handling all the resources to achieve the AOCS objectives, i.e. the startracker, Sun acquisition sensor, IMUs (including gyros and accelerometers), reaction wheels, thrusters and SADMs. This function performs the configuration management and commanding of these units;
— processing of sensor output and actuator input that provides the AOCS modes with specific services allowing the filtering of hardware raw resource measurements or the processing of commands computed by the attitude and control laws;
— the ephemerides propagator that provides the AOCS modes with the spacecraft inertial directions to the Earth and Sun;
— AOCS mode management, which manages the transitions between the various AOCS modes (Sun Acquisition Mode, Safe/Hold Mode, Normal Mode, Orbit Control Mode, Thruster Transition Mode, Main Engine Boost Mode, Braking Mode);
— AOCS algorithm management, which performs attitude estimation and control, and the trajectory control in each mode;
— the AOCS FDIR, which manages the FDIR at AOCS equipment and AOCS functional levels;
— mission management, to sequence nominal and contingency mission phases based on specific sets of AOCS modes;
— spacecraft management, to command and allocates all AOCS resources.

Both DMS software and AOCS software are loaded in the PM RAM and started up by the PM firmware. Each CDMU has two PMs with the firmware, which is automatically activated when the CDMU is powered on. In addition to initialising the PM, it performs, among other tasks, health status checks and software loading.

The SSMM software, running on its own processor, is designed to operate the SSMM, and has two main components:

— Initialisation software for the system controller and control interface hardware, tables, data, etc. loads the nominal software from EEPROM to RAM, handles commands and performs the transition to Operational Mode;
— Operational software executes and controls telecommands, configures and

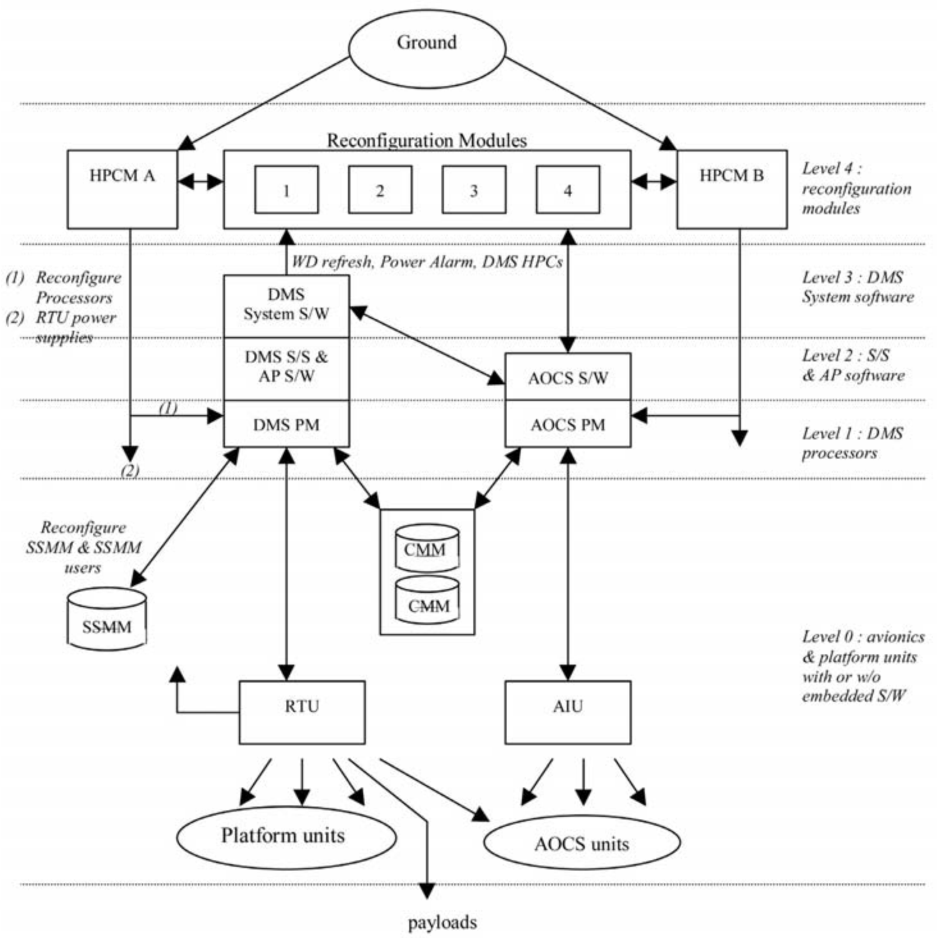


Fig. 18. FDIR hierarchy mapping. CCM: Centralised Memory Module.

tests the MMs, controls data flow from the instruments, TFG and the MMs and handles failures and failure reporting.

The startracker software runs on its own processor (in the startracker) under control of the AOCS software. It manages the startracker and delivers 3-axis autonomous attitude determination to the AOCS software. The generic part of the software provides basic services to the AOCS software for full visibility and investigation, such as health status and auto-tests. The specific part of the startracker software manages all tasks related to the application modes, and includes:

— an acquisition and measurement mode that provides 3-axis attitude restitution to the AOCS software without any initial information by performing an initial mapping and, using automatic pattern recognition, selecting stars and automatically tracking them;
— a mapping mode that provides the magnitude and coordinates of all the targets in the field of view for ground investigation;
— a calibration mode that allows the AOCS software to change default parameters.

The gyro software runs on an individual processor under the control of the AOCS software. It manages the gyros and delivers angles and velocities to the AOCS.

The transponder software runs on a processor inside the transponder and interfaces with the DMS software via serial commands and digital telemetry. The transponder software controls the functionality of the digital section and performs the signal processing operations to maintain the forward and return RF links with Earth.

There are also several other autonomous software packages running in payload instruments under the control of the DMS software.

3.13 Failure Detection, Isolation and Recovery

As a deep space mission, Venus Express requires a high level of onboard autonomy because of the time needed for ground intervention in the event of an onboard anomaly or the extended periods when communications are not possible owing to the respective positions of the Sun, Venus and Earth. The automatic FDIR function, as for many of the other functions of the spacecraft, is largely recurrent from Mars Express with some adaptations; the concept is shown in Fig. 18.

The FDIR function handles any anomalies onboard with a first goal of returning to the same spacecraft mode in order to preserve operational integrity. If this is not achievable, then the priority is to preserve spacecraft integrity and safety. In this case, the spacecraft autonomously enters Safe Mode, which ensures that power is available from the solar array, communication with Earth is available for diagnostics and recovery, and all non-essential loads have been switched off. In this way, spacecraft safety is ensured until operators at ESOC can identify and correct the anomaly and proceed with the mission.

4. Conclusions

The Venus Express end-to-end in-orbit system validation, from science planning in ESA's Venus Science Operations Centre in ESOC to scientific observation and transmission to Earth and distribution to scientists, began in May 2006. During this phase and the subsequent routine operations, the spacecraft has continued to operate smoothly, highlighting the success of tailoring the Mars Express spacecraft for Venus exploration.

The authors wish to thank all those who contributed to the design of Venus Express, especially the Astrium project team members and their counterparts in ESA.