# A Satellite for the Galileo Mission

J.C. Chiarini / C. Mathew[1], H.P. Honold / D. Smith[2]

[1]Galileo Industries GmbH, Lise Meitner Strasse 2, 85521 Ottobrunn, Germany
Phone: +49 89 88 984 – 44 441, Fax: +49 89 88 984 – 44 455
e-mail: jean-claude.chiarini@Galileo-Industries.net /
e-mail: Colin.mathew@Galileo-Industries.net
[2]EADS Astrium Germany, Ludwig Bölkow Allee, D-85521 Ottobrunn, Germany
Phone: +49 89 607 28154, Fax:+49 89 607 28546
e-mail: hans.peter.honold@astrium.eads.net /
e-mail: david.smith@astrium.eads.net

**Abstract.** The Galileo System is based on a 30 spacecrafts constellation in MEO orbits, controlled and commanded in S-band link by a Ground Control Centre. The navigation service is achieved via transmission to the user of L-band signals comprising ranging codes and timing information.

The timing signals are provided by precision on-board atomic clocks, implemented as two redundant pairs per satellite. Two different technologies are implemented - passive hydrogen maser and rubidium.

In addition to the navigation services, a Search & Rescue service is provided, which is implemented by a dedicated payload.

The specified lifetime for the Galileo satellites is 12 years while the overall specified lifetime of the Galileo System is 20 years. This means that a full replenishment of the satellite constellation will be required. In the frame of the IOV phase, 4 satellites will be produced and deployed in two different orbit planes in two dual launches using the Soyuz.

## 1 General Information

The Galileo constellation comprises of 30 satellites placed in MEO orbit, with 10 satellites placed in each of 3 orbital planes distributed evenly round the equator. The active constellation comprises of 27 satellites, with each plane containing a spare satellite which can be moved to replace any failed satellite within the same plane, thereby reducing the impact of failures upon quality of service. All satellites are identical in terms of design, performance capability and fuel load. The satellite flight configuration is shown in Fig. 1. The earth-pointing face is defined along the +Z axis. In launch configuration the solar arrays are stowed on the +/– Y sides of the satellite. The volume and outer shape of the satellite is compatible with the shroud dimensions of the selected launchers. Clearly visible on the earth-facing side of the satellite are the search and rescue and navigation payload antennas.

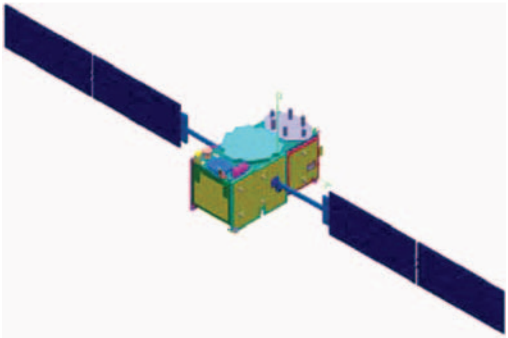


**Fig. 1.** Galileo satellite flight configuration.

The satellite is composed of the following subsystems:

- Payload Subsystem including the navigation payload and the SAR payload
- Structure Subsystem
- Thermal Control Subsystem (TCS)
- Electrical Power Subsystem (EPS) with the following units:
  - Solar Arrays (SA)
  - Solar Array Drive Mechanisms (SADM)
  - Battery
  - Power Conditioning and Distribution Unit (PCDU)
- Harness
- Avionics Subsystem with
  - on-board computer (Integrated Control and Data Handling Unit, ICDU)
  - Attitude and Orbit Control System, AOCS (based on earth sensors, sun sensors, gyros, reaction wheels and magnetic torquers),
  - Software (SW)
- Telemetry, Tracking and Command (TTC) Subsystem (with S-Band Transponder and two low-gain, omni-directional antennas)
- Propulsion Subsystem (mono-propellant system with one tank and 8 thrusters)
- Laser Retro-Reflector (LRR)
- Platform Security Unit (PFSU)

**Launchers for IOV**

For IOV Launchers the launchers already selected is Soyuz, with dual launch. The configuration under fairing is shown in Fig. 2.

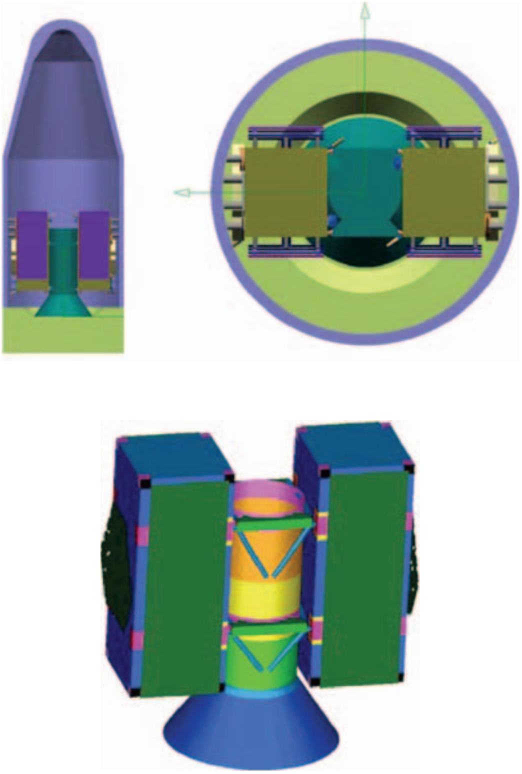


**Fig. 2.** Soyuz configuration for IOV.

## 2 Payload Architecture

The Galileo satellites include two payloads, the Navigation payload and the Search and Rescue payload.

The overall payload block diagram is presented in Fig. 3, here under.

### Navigation Payload

The main functions of the navigation payload are:

– Provision of on-board timing signals
– Receipt & storage of up-linked navigation message data
– Receipt & storage of up-linked integrity data
– Assembly of navigation message in the agreed format
– Error correction coding of navigation message
– Generation of ranging codes

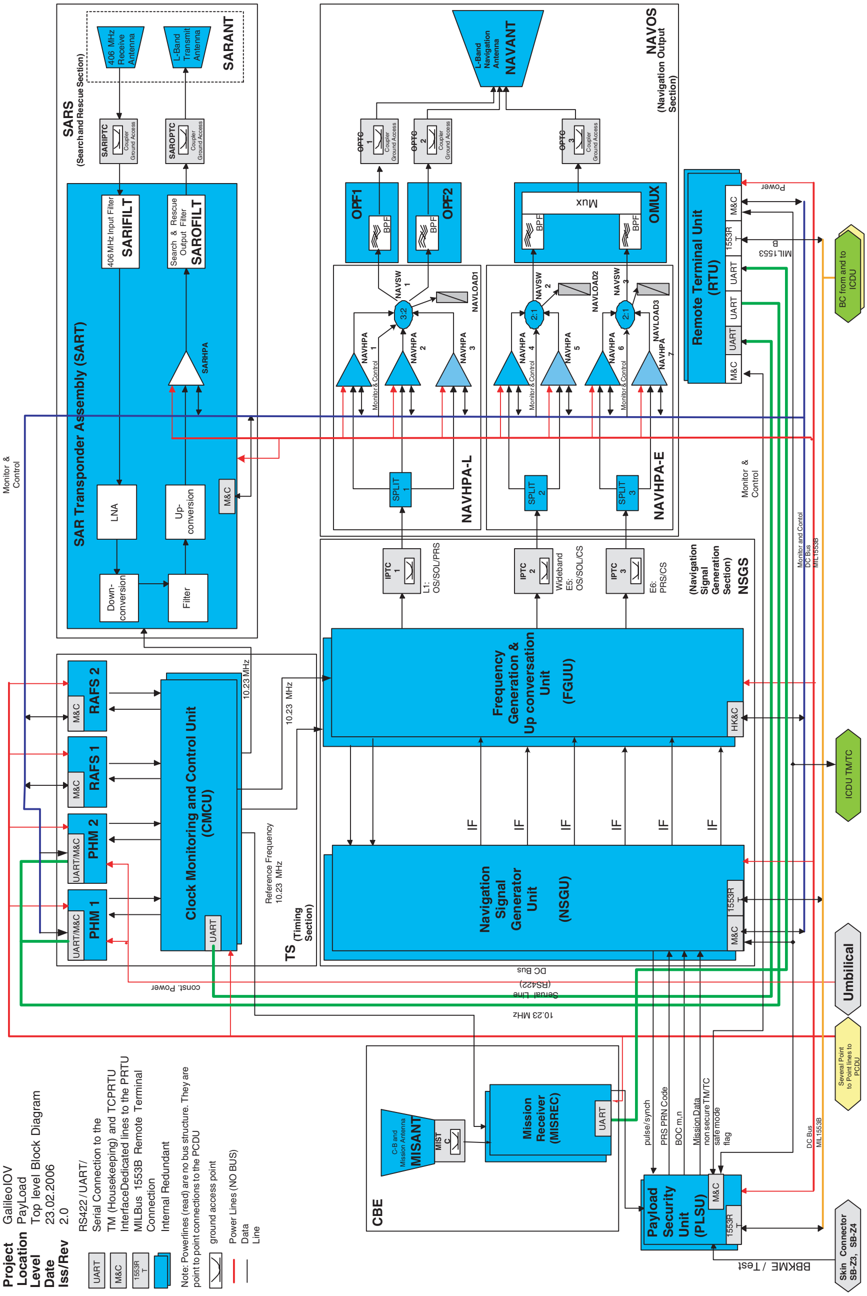



Fig. 3. Navigation payload.

– Encryption of ranging codes as required
– Generation and modulation of L-Band carrier signals
– Broadcast of navigation signals

The timing signals are provided by high precision on-board clocks, implemented as two (cold) redundant pairs per satellite, each pair including two different technologies, the Passive Hydrogen Maser (PHM) which is the primary reference clock and the Rubidium Atomic Frequency Standard (RAFS), both of them being operated simultaneously. Due to the highly stable frequency stability requirements the clocks are mounted on a separate radiator panel, which is kept facing deep space using a yaw steering law controlled by the Avionics.

The whole clock ensemble is under the control of a dedicated (internally cold redundant) Clock Monitoring and Control Unit (CMCU) which performs the monitoring and switching functions (selection under Ground control) and generates a highly stable on-board reference frequency of 10.23 MHz which is distributed to the other payload units.

The navigation data (including integrity data, Search and Rescue data and other mission data) are contained in the C-band spread spectrum uplink signal which is received via the Cband mission receive antenna operating in RHCP polarisation (baseline solution is a small aperture axially corrugated circular waveguide horn). The Mission Receiver (MISREC) which includes the Mission Processor function (MISPROC), performs the receive function, the despread and demodulation functions in order to provide a data stream which is routed to the Payload Security Unit (PLSU) which performs COMSEC treatment of the incoming signal, and passed to the Navigation Signal Generator Unit (NGSU). The MISREC is an internally cold redundant unit.

The Navigation Signal and Generator Unit (NSGU) which includes internal cold redundancy, receives the up-linked navigation data and uses them to generate the navigation signals in the appropriate format, performs the PRN encoding and the modulation of the 3 navigation signals (E5a + E5b, E6 and L1) and passes them to the Frequency Generation and Upconversion Unit (FGUU) which performs the up- conversion into L-band of the 3 signals.

The FGUU includes internal cold redundancy.

The 3 L-band navigations signals are then routed to a 2:1 (E5a + E5b and E6 with respectively about 65 W and 70 W SSPA RF output power) or 3:2 (L1 with parallel amplification at about 50 W SSPA RF output power) SSPA redundancy ring. The two amplified L1 signals are routed to the navigation antenna and combined in free space while the two other signals are multiplexed (within the NAVOMUX) before being routed to the navigation antenna.

The Navigation transmit Antenna (NAVANT) which is operated in RHCP polarisation consists of a high and a low band beam-forming network and a dual band array of radiating elements which provides a global coverage iso-flux radiation pattern.

Test couplers are included in order to allow the testing of the different navigation payload sections during the payload and satellite AIT cycle.

The payload Remote Terminal Unit performs the communication functions between the payload and the avionics subsystem via the 1553B data bus as well as the acquisition of all payload units telemetry and the distribution of all commands to the payload units except the On/Off commands and the acquisitions of the corresponding status of the payload units connected to the 1553B data bus (in order to allow these units to be switched off in case of anomalies occurring on the 1553B data bus).

The Navigation Payload consists of the following units:

- Ultra Stable Oscillators
  - Passive Hydrogen Maser (PHM),
  - Rubidium Atomic Frequency Standards (RAFS),
- Clock Monitoring and Control Unit (CMCU),
- MISANT antenna (C-band)
- Mission Receiver (MISREC)
- Payload Security Unit (PLSU),
- Navigation Signal Generator Unit (NSGU),
- Frequency Generator & Up Converter Unit (FGUU),
- NAVANT antenna (L-band),
- NAVHPA (SSPA),
- Output Multiplexer (OMUX) and filters,
- Remote Terminal Unit (RTU).

**Search & Rescue Payload**

The Search And Rescue (SAR) payload is principally based on a transparent transponder receiving the uplink signal in the 406.0–406.1 MHz band and retransmitting it in the 1544.05–1544.15 MHz band. The SAR payload includes no redundancy at satellite level.

Redundancy at mission level is obtained via the satellites of the constellation.

The SAR antenna performs both the receive function at UHF-band (RHCP polarisation) and the transmit function at L-band (LHCP polarisation). The baseline solution consists of a ring of 6 quadri-filar helix radiating elements for the UHF receive function, with a central 'splashplate' feed acting as the L-band transmit antenna.

The receive signal is routed to the Search and Rescue Transponder (SART).

The SART performs low pass filtering of the received signal, amplification via a Low Noise Amplifier (LNA), output filtering and then down-conversion into IF, close-to-band filtering using crystal filtering (switchable in order to provide a wideband (90 kHz) or narrow band mode (50 kHz)) and up-conversion into L-band.

The signal is then amplified with a nominal RF output power of 5 W, filtered and routed to the SAR antenna transmit port for transmission.

Test couplers are included in order to allow the testing of the SAR payload during the payload and satellite AIT cycle.

The Navigation Payload consists of the following units:

- Search and Rescue (SAR) Transponder,
- SAR Antenna (at UHF for receive and L-Band for transmit).

The SAR payload uses the 10.23 MHz signal from the navigation payload for the frequency up-conversion which is the only interface between the two payloads.

## 3 Electrical Architecture

The overall satellite electrical architecture is presented in Fig. 4.

The satellite power is distributed to the electrical units via a 50 V fully regulated power bus delivered by the Power Conditioning Unit (PCDU).

The power sources include the solar array which delivers approximately 1700 W at EOL and a Li-ion battery (baseline is a 9s3p SAFT battery design with G5 cells).

The sizing requirement for the battery is the energy required to supply the satellite during the long duration launch sequence used for direct injection. During this phase the satellite is supplied (nearly) only by the battery. In the cruise phases where the launcher does not perform any manoeuvre, the attitude of the launch vehicle can be optimised for the satellite requirements. It is therefore foreseen to follow a barbecue type attitude law with the sun direction being normal to the rotation direction. This allows both a minimum amount of power to be provided via the external panel of the stowed array (external face is the cell face).

Most satellite units are switched off during launch. However a significant amount of heating power is required to maintain the satellite units in their applicable temperature limits (mostly non operating limits).

Most part of the radiator area is covered by the stowed solar array wings, which limits the heat loss but the clock radiators and a part of the platform radiators are uncovered.

Furthermore, all external units (sensors, thrusters, etc.) require heating. The battery capacity which is maximised at launch via a specific charge procedure (at a higher cell voltage than the one used for cycling operations) has been sized to be compliant to all launch.

However, the current baseline is based on conservative assumptions in terms of heating power and since no power is assumed to be delivered by the solar array.

The SADM performs the solar array orientation function and the transfer of the solar array power to the satellite. Due to the yaw steering law, the SADM is used in normal operation around the –X axis within an angular range which depends on the sun elevation (w.r.t. the orbit plane) and which can reach up to ± 90° for zero sun elevation. However, the SADM offers the possibility to orient the SA in any position around the Y axis in order to provide the capability to perform long duration orbit-keeping manoeuvres (which require solar array power) with the necessary flexibility (w.r.t. the sun direction) to achieve the repositioning duration requirements.

The battery management function is performed by the PCDU. The 50 V power bus is distributed to most electrical units except to the propulsion units (pressure transducer, thrusters and latch valves) which are supplied via a dedicated 28 V power supply (also provided by the PCDU) and to some of the avionics units such as the sun sensors and the magnetic torque rods, which are supplied by the ICDU.

The PCDU performs the power bus regulation function via shunt switching, the distribution and the protection functions according to a star topology using SSPC as protection devices to the permanent lines (S-band transponder Rx, PFSU, ICDU) protected by Foldback Current Limiters (FCL) and to the switchable lines

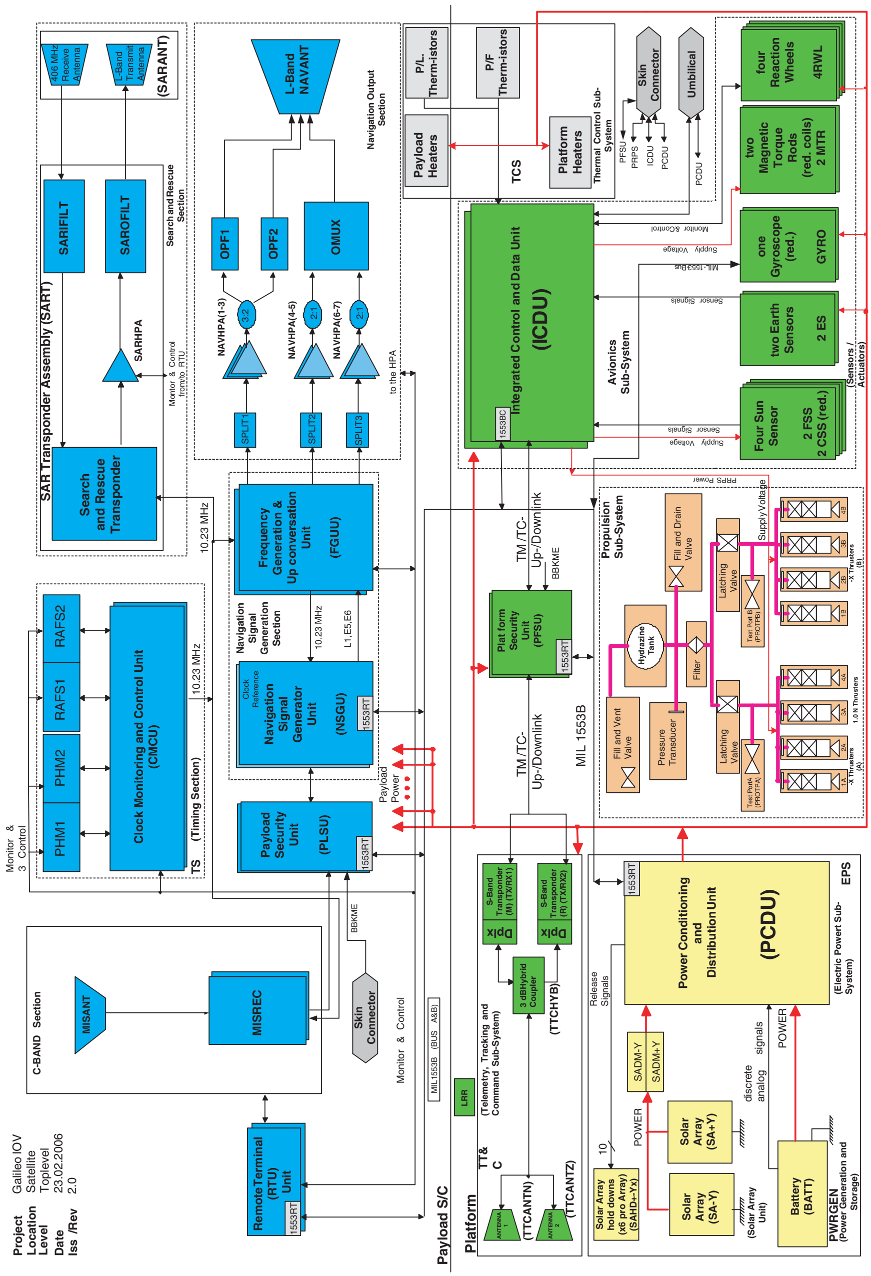



**Fig. 4.** Electrical architecture

(all other units) protected by Latching Current Limiters (LCL). This approach leads to a high cleanliness of the power bus supply in comparison to a fuse protection approach which allows the units converters to be optimised.

All switchable lines may be switched On or Off at PCDU level under ICDU control (in addition to the nominal switch On and Off function). The PCDU also provides the pyro and heaters interface functions.

The battery can be isolated from the PCDU via a dedicated strap located on a skin connector. The pyro commands can also be armed/disarmed via dedicated plugs to be connected on a skin connector.

The ICDU which is the core unit of the avionics subsystem includes in the baseline the 3 following modules:

- The TMTC-RM module which performs the following hot redundancy functions: TC decoding, monitoring and reconfiguration, context memory (Safe Guard Memory), distribution of HPC commands, and the OBT including the synchronization mechanism, as well as the following cold redundancy functions: TM encoding, TM storage (TM memory),
- The PM-BC modules (operated in cold redundancy) combining the ERC-32 processor with the related memory and the 1553 Bus Controller,
- The IOM modules (operated in cold redundancy) which deliver all the input/output signals necessary to control and monitor the platform equipments and some payload equipments and performs the thermistors acquisition. The IOM modules provide the interface functions to both nominal and redundant AOC units
- The last module is the Converter module, based on 3N and 3R independent DC/DC converters each of them dedicated to one module (TMTCRM, PM or IOM).

A Space-Wire network provides the interface between each module and a full cross-strap allowing each PM to communicate with both Main and Redundant other modules.

The 1553B redundant data bus is used for communication between the ICDU and the following units:

- Payload: RTU, PLSU and NGSU
- Platform: PCDU and PFSU

All RT use the long stub option, i.e. transformer coupling, and are connected to both nominal and redundant buses. Each nominal and redundant BC has the capability to control both the nominal and the redundant bus.

The RTU performs the TM/TC interface function between the payload units and the ICDU via the 1553B data bus. It distributes various types of command signals (High Level Commands, extended HLC, Bi-level Commands, serial load commands) and performs the acquisition of the various payload units TM signals (thermistor, analogue telemetry, bi-level telemetry, discrete relay / switch status, serial telemetry). It includes internal (cold) redundancy and internal cross-strapping.

The other platform subsystems and their overall electrical architecture are then presented:

**The TT&C subsystem** consists of:

- two S-band conical quadri-filar helix antennas accommodated symmetrically on the satellite in order to provide a quasi omni-directional coverage which perform both the receive and transmit functions. The antennas are circularly polarised, the antenna used in nominal attitude being RHPC, the other one being LHPC,
- of a 3 dB hybrid coupler,
- of two S-band transponders including diplexers (operated in hot redundancy for the receive function and in cold redundancy for the transmit function) which perform the TC receive, the TM transmit and the ranging functions and which can be operated in 2 different selectable modes, namely ESA standard mode and spread spectrum mode.

In addition, a Laser Retro Reflector used for high accuracy ranging performed with laser ranging stations is implemented on the +Z face of the satellite.

The TC signals delivered by the TC receiver function of the TTC transponder including the data, clock and data validity signals are routed to the PFSU (the carrier lock signal being routed directly to the ICDU) which performs COMSEC processing of the data signal. The base-band TC data signal is a 1 kbps NRZ-L BCH encoded signal. The signals are then generated again by the PFSU in the same format (including BCH encoding) and are routed to the ICDU.

Similarly, the PFSU receives the TM signals (data and clock) from the ICDU in the form of a 20 kbps NRZ-L RS encoded signal. The PFSU performs the COMSEC processing of the data signal. The signals are then generated again by the PFSU in the same format (including RS encoding) and are routed to the TTC transponder. The conversion into NRZ-M and the convolutional coding is performed within the transponder transmitter function.

Since the output signals delivered by the PFSU have the same format as the input signals, it is possible to by-pass the PFSU (using an appropriate test harness) in AIT and to operate the TC/TM chain without the PFSU being present. This introduces some robustness of the AIT schedule w.r.t. a late delivery of the PFSU.

The PFSU are operated in hot redundancy and are cross-strapped for the TC function with the nominal and redundant transponders. They are cross-strapped both for the TC function and for the TM function with the nominal and redundant ICDU functions.

**The avionics subsystem** includes, additionally to the ICDU, the following AOC units (the TM/TC interface of which is performed by the ICDU) which are all cross-strapped w.r.t. the IOM module of the ICDU:

- 2 coarse sun sensors (cold redundant)
- 1 fine sun sensors (cold redundant) with power supply delivered by the ICDU
- two earth sensors (cold redundant) connected to the power bus
- two rate integrating gyros (cold redundant) connected to the power bus
- two magnetic torque rods (redundant coils) commanded by the ICDU
- four reaction wheels (4:3 redundancy) connected to the power bus

**The propulsion system** includes a hydrazine propellant tank, a filter, a propulsion transducer, two latch valves, the piping to distribute the propellant to the 8 thrusters

(4 nominal and 4 redundant) via two redundant branches and the necessary fill and drain valves and test ports. A pressure transducer allows the monitoring of the pressure in the system. The pressure transducer as well as the thrusters valves and the latch valve are supplied by the PCDU via a dedicated 28 V supply.

The thruster catalytic bed heaters are supplied via the 50 V primary bus by the PCDU via four standard heater interfaces. Four heaters are connected on each heater line (two in series and two in parallel). Each of the 8 thrusters is equipped with one nominal and one redundant catbed heater.

Thruster valve commands (not the latch valve command since the latch valve needs to be operated open shortly before launch) as well as pyro commands are inhibited in the not separated launch configuration via the umbilical connector in order to fulfil the safety requirements. These commands are indeed executed after separation during the autonomous satellite initialisation sequence and for the solar array deployment following the initial sun acquisition by the software upon separation detection.

The separation detection is obtained via three separation straps distributed on both umbilical connectors and conditioned by the ICDU via three independent acquisition channels to the maximum practical extent (use of common function is only allowed if any failure of the common part can be detected in which case a reconfiguration to the redundant IOM module would be performed). The software performs then a majority voting on the three signals to detect separation. The separation detection system is designed to be tolerant to any single failure.

Furthermore, the electrical architecture includes the necessary skin connectors used during the AIT activities. In flight configuration, the flight plugs (e.g. pyro arm, thrusters arm, SADM arm, etc.) are connected to the corresponding skin connectors while EMC covers are installed on the other skin connectors.

In order to be able to perform the key loading operations into the security units via the BBKME, dedicated skin connectors (6 in total, 2 for the PFSU, 2 for the C-band function of the PLSU and 2 for the PRS function of the PLSU) are implemented. These skin connectors are directly linked to the corresponding security units via dedicated cables which follow additional design constraint and which are included in the TEMPEST test campaign performed at unit level. All these skin connectors are equipped with seals (as well as on unit side) which are periodically controlled in AIT in order to control the integrity of the corresponding function.

The PHM ion pumps require to be permanently supplied via a dedicated low power high voltage supply except for periods of limited duration (lower than 10 days). Since this supply is not Corona free, it shall be switched off at launch and during the pump-down phase of the satellite thermal vacuum test. In order to supply the PHM ion pumps when the satellite is off, dedicated skin connectors are foreseen which will be used to connect the corresponding EGSE.

## 4 Mechanical Architecture

The mechanical architecture is based on a parallelepiped box made of aluminum sandwich panels. The size of the structure is $2530 \times 1200 \times 1100$ [mm] with extensions of the +/− Y payload panels of 100 [mm] in + Z direction and 150 [mm] in − Z direction. These extensions have been implemented to provide the necessary radiative area for the payload units.

The primary structure provides the interfaces to the directly mounted units or to the secondary structure components (brackets, etc.). Figure 5 presents an external view of the overall satellite architecture as well as the definition of the main reference coordinate system. The satellite is composed in a modular way, by a P/F and a P/L Module for separate integration and test activities at different locations. This modularity concept is shown in Fig. 6.

The body of the platform module consists of:

- the +/–Y P/F-panels, carrying the major part of the platform electronics units,
- the –X panel which is dedicated to the propulsion subsystem (thrusters, valves etc),
- the internal panel which supports the four wheels,
- and the 2 Shear Frames which serve as tank support, load path for the separation system and connecting element for the modules,
- the access panels (–Z).

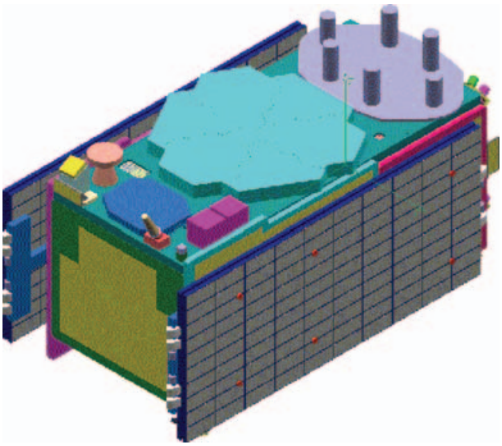


**Fig. 5.** External view of the overall satellite architecture.

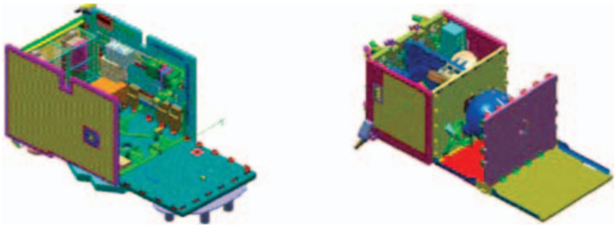


**Fig. 6.** PL Module and PF module.

The structure will be closed at the $-Z$ side by the 3 removable $-Z$ access panels which can be removed separately, allowing access to the 3 compartments (platform compartment and the two other compartments which allow the access to the tank and to the payload units).

The body of the payload module consists of:

- the +/–Y P/L-panels, carrying the high dissipating payload units,
- the +Z panel which has the length of the entire satellite body and carries the payload antennas and additional low dissipating payload equipment on its internal side,
- the +X panel, which accommodates the clocks.

## Structure Design

The sandwich layout, i.e. overall thickness of the sandwich, core density and thickness of the face sheets have been optimized w.r.t. the structural needs (strength, stiffness, load carrying capability of inserts etc) in order to optimize the structure w.r.t. mass.

Where necessary, the panels have been reinforced, mainly at highly loaded areas or cutouts.

For high-load introduction into the panels, embedded brackets are foreseen, e.g. in the areas of the separation systems.

All sandwich panels are vented by use of perforated core material.

The design of the panel assembly via brackets (cleats) is based on heritage from previous programs.

The cleats are metallic profiles, reinforced by webs as necessary to achieve the required stiffness. The cleats will be made of 7475-T7352 alloy, for the attachment of the +x-panel of Ti-6Al-4V for reduced thermal conductivity.

The design of the cleats as well as the location accounts for good accessibility for integration activities, i.e. the integration flow has been considered in the attachment configuration for the cleats. In addition, it allows easy removal of panels if requested by AIT activities.

The brackets will be assembled with the panels via face-to-face inserts for M5 bolts. For the fixation of the units, the panels provide inserts which will be the space qualified ENN398M4/M5 type. These inserts contain replaceable, self-locking Helicoil threads, which allow a limited number of operations and easily can be replaced before the number of allowable operations has been exceeded.

## Accessibility in Ait

Special attention has been paid to achieve the required access to the internal part of the satellite. The foreseen access area will be via the – Z panel. This panel is split into 3 parts, each part corresponding to one compartment, i.e. platform, tank + payload and payload. As these three panels belong to the primary structure and therefore

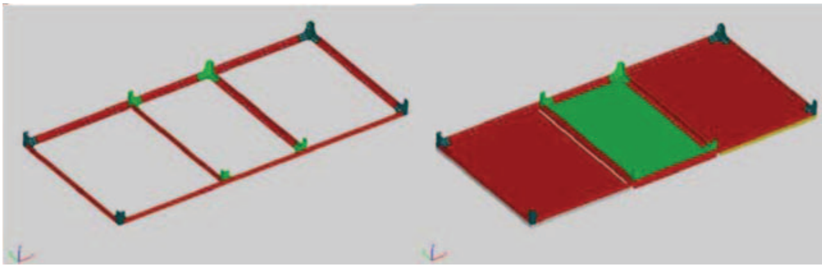


**Fig. 7.** Frame and access –Z panels (Centre Panel).

have a structural function, only one panel at a time may be removed to maintain the satellite mechanical integrity.

The panels are attached to a frame which is also part of the primary structure. Fig. 7 presents the removable −Z access panels as well as the −Z frame.

## Separation System Interfaces

The separation system will be provided by the launcher authority. The structure will provide a "standard interface" for the attachment of such a system, i.e. threaded holes in embedded brackets. In addition the surrounded structure includes the necessary reinforcement.

## Specific Interfaces

The structure shall provide specific interfaces for the the tank interfaces and the Navigation Antenna interfaces.

**Tank Interface.** The tank will be supported by two frames, via 2 polar mounted supports. The propellant tank has a significant impact on the dynamic behaviour of the satellite and on the design of the support structures due to its high mass (which may reach up to 85 kg when fully loaded with propellant).

It shall be noted that the structure, and more specifically the tank support interface, is specified to support the propellant tank loaded at full capacity. This allows both to provide margins w.r.t. the currently needed propellant mass and to provide the flexibility to fully load the propellant tank in launch scenarios where the launch mass is not limited by the launcher performance.

**Navigation Antenna Interface.** The antenna shall be attached via 6 bolts homogeneously arranged on a circle of 730 mm diameter.

## 5 Thermal Architecture

The satellite thermal control architecture consists of 6 main thermal zones as follows:

- The clocks zone (payload),
- The payload global thermal control zone, excluding clocks
- The platform global thermal control zone (excluding battery and propulsion)
- The battery zone,
- The propulsion zone,
- The external elements (antennas, sensors) specific thermal control

**Clock Zone.** The thermal control stability of the clocks is one of the major design drivers for the thermal control subsystem. In order to reach the required stability the clocks and their radiators are accommodated on the +X panel. In order to achieve the required thermal stability performance, the following measures are implemented:

– Only the clocks are accommodated on the clock panel in order to avoid any disturbance coming from other units with regard to the thermal stability.
– The clock panel is conductively decoupled from the other panels by using thermal washers.
– The clock panel is radiatively decoupled from the satellite internal environment by using two MLI tents, one for each clock pair.
– The +X clock radiator is free of any satellite protuberance as far as practicable (the only present item is the TTC antenna which impact is acceptable); in particular, no radiator extension is implemented in the +X direction of the Y payload panels.
– Each clock pair of a same technology (PHM on one hand, RAFS on the other hand) is accommodated on a doubler (one for each pair) which spreads the heat to the corresponding clocks radiator. This approach provides a homogenous heat distribution on the radiator, optimises the thermal stability (by using the thermal heat capacitance of the redundant unit) and minimises the heater consumption.
– Active thermal control of the clocks by heaters which are regulated by a PI algorithm using high accuracy thermistors (three with majority voting for failure tolerance).

**Payload Global Zone.** The payload zone corresponds to the ±YPL panels (Fig. 8) and the +Z panel.

All high dissipating payload units are located on the ±Y***PL*** panels. These S/C panels, which are submitted to very limited sun incidence and therefore to minimum external heat load, include the OSR radiators and are therefore the most suitable panels for the accommodation of the dissipating units.

However, these panels are submitted to the reflected sun radiation and infrared radiation from the solar array and the yoke onto the ±Y***PL*** panels. The influence of the solar array panels is minimised via the yoke which takes away the panels and thus minimises their factor of view to the radiator. The yoke shall be mechanically (a frame type design shall be used) and thermally (white painted) designed such as to minimise the radiative heat transfer between the yoke and the Y radiators.

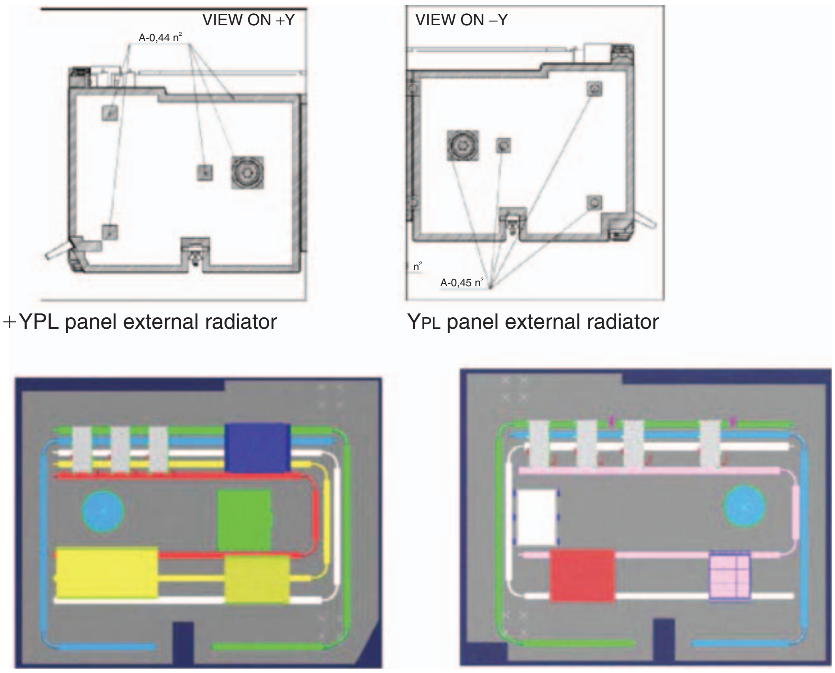



**Fig. 8.** Radiators & baseline heat pipe layout on the $+Y_{PL}$ and $-Y_{PL}$ panels.

The dissipating payload units are distributed on the ±Y***PL*** panels such as to balance the total dissipated power on the two walls.

Additionally the following requirements are fulfilled with the actual payload unit configuration:

- accessibility of the PLSU and PFSU from the satellite −Z side during AIT
- minimum RF-cables length
- no interferences of units und sufficient margin for unit connector

All external sides of the ±Y***PL*** panels are covered with OSR excluding the position of the SADM and the 3 solar array hold down points which are covered with MLI.

Most of the payload units are located on a heat pipe network to achieve a good conductive heat transfer from the unit via the heat pipes to the complete radiator area. Thermal fillers are implemented between the units and the heat pipes to achieve good thermal conductance.

U-shaped heat pipes are implemented on the ±Y***PL*** panels.

Dissipating units are black painted as well as the internal surfaces of the S/C panels to homogenise internal temperature and to facilitate heat rejection to the radiators.

On each Y***PL*** panel there are three heater lines. Most heaters are attached on the heat pipes.

Some units have heaters on the panels beside the units. Units will be heated via the heat pipe system with several survival heaters to guarantee that they are always within the acceptance temperature limits.

Three thermistors are measuring the temperatures on several positions and the output of the majority voter is used for the heater regulation.

Most units located on the +Z panel are low dissipative units (e.g. switches and couplers).

However, the OMUX and OPF which have significant dissipation, are also accommodated on the internal side of the +Z panel. They have been positioned as close as possible to the Y***PL*** panels to achieve a high view factor between the units and the Y***PL*** panels and to transfer the heat from the units to the panels via radiation.

Accommodation of the OMUX and OPF on the Y***PL*** panels has been considered in the architecture trade offs. But this solution has not been selected since the available area on the Y***PL*** panels was too limited to accommodate them together with the corresponding harness, switches and couplers. Such a solution would have led to an unacceptable length of the RF cables and therefore to unacceptable cable losses.

**Platform and Battery Zones.** The battery and the PCDU are conductively coupled to the +Y***PF*** panel to radiate its dissipated power via the external radiator to deep space.

The PCDU radiator is such that it is not covered by the stowed solar array in order to maintain the unit within its temperature limits during the launch phase until solar array deployment.

All other radiator areas of the ±Y***PL/PF*** panels are covered by the stowed solar array during launch (which allows the heating power in this phase to be minimised).

The battery zone is made of a dedicated battery radiator (conductively decoupled from the +YPF panel), a MLI including MLI support which provides radiative decoupling and the associated active thermal control.

The platform units: ICDU, PFSU, Tx/Rx S-band Transponder and Gyros are conductively coupled to the -YPF panels to radiate dissipation power via the external radiator to deep space.

A thermal doubler is used below the Tx/Rx S-band transponders to improve the thermal coupling between nominal and redundant units. Active heating with heaters on the panels will keep the units within their temperature limits. Black paint on the units and on the panels is used to harmonize the temperature within the compartment.

The reaction wheels are fixed to the M panel and on the shear frame 1. The dissipation power will be transferred radiatively to the ±YPF panels. The wheels and the panels will be black painted. Additional possible solutions to increase the heat transfer of the wheel to the ±YPF panels are:

– Thicker face sheets of the shear frame 1
– Thermal straps from the wheels to the ±YPF panels

Dedicated radiator area on the ±YPF panels where the thermal straps are connected Active heating with heaters on the panels are used to maintain the units within their temperature limits.

**Propulsion Zone.** All propulsion items (but the tank) are arranged on the –XPF panel.

The piping from the tank to the –XPF panel is routed on the M-panel. The propulsion items such as valves, pressure transducer, filter, etc. are arranged in the compartment/enclosure where the PCDU/Battery is located. This is favourable for minimising the heater power due to the PCDU which is dissipating in all phases.

Active heating with heaters implemented directly on the units (valves, pressure transducer, filter, tank and the piping system) allows the units to be maintained within their temperature limits. The units and the piping system will be thermally decoupled from the –XPF panel by thermal washers to minimize heater power consumption. Propulsion units such as valves, pressure transducer, filters and pipes will be covered with small MLI boxes.

A main and a redundant heater as well as three thermistors are mounted on the outer surface of the thruster valve.

The external side of the –XPF panel is completely covered with MLI except the required stayout area of the thrusters.

Skin connectors are positioned on the –XPF panel which are covered with MLI in flight configuration.

**External Units.** The external elements are insulated from the satellite, radiatively by MLI on their rear side and conductively by insulating washers in order to minimise mutual interaction between the satellite and the external element thermal control. This allows the heat flow between the S/C interior and the external elements to be minimised in order to achieve the required thermal stability requirements (e.g. of the payload units) and to minimise the required heating power.

All the units on the external side of the +ZPL panel will be therefore covered with MLI except the apertures of the antennas, sensors and the LRR. The remaining S/C +ZPL panel will be covered as well with MLI. The MLI of the S/C will have an overlapping with the antenna MLI blankets. Standoffs are the preferred fixation method for the purpose of several integration cycles.

Dedicated radiators are used for some external units such as the IRES and the CSS/FSS.

## 6 Command and Control Architecture

The C&C for the IOV-satellite is based on the implementation of the Packet Utilization Standard (PUS) tailored for Galileo IOV needs. This standard defines a set of services to monitor and control a satellite. Each service consists of subservices, which are either a telecommand or a telemetry report. The format of telemetry and telecommand packets are standardized to a certain extent to support re-use of existing onboard or ground software.

The following table gives an overview about the implemented services.

The hardware independent part of the PUS is implemented by a so called Core Data Handling System (CDHS), which is a library of data handling services for the support of PUS applications. The CDHS library has been used in different projects, like GSTB-V2, and TerraSarX. The hardware depended services (device commanding, memory management etc.) will be implemented to fit to the Galileo IOV ICDU design.

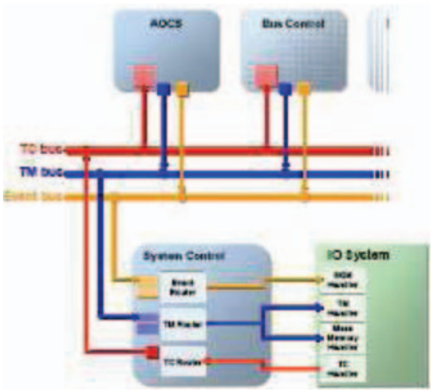



**Fig. 9.** Software buses.

The CDHS is based on the implementation of software busses (Fig. 9) for TM, TC and events. These busses provide a common interface which allows the different software applications to communicate in a standardized way.

**Telecommands.** Telecommands received by the transponder are routed to both hot redundant PFSU. If the PFSUs are in clear mode they will route the data to the TC-decoders. The TC decoder (if addressed by the VCID) will decode the CLTU, perform the frame acceptance procedure, generate the command link control word and route the TC segment to the ICDU TC interface if all checks are successfully passed. The avionics software reads the TC segment and performs the de-segmentation of a TC packet (if the command consists of more than one segment). When the telecommand packet has been reconstructed, the avionics software performs checks on the packet header and packet error control, generates the appropriate command acceptance report (TM 1,1 or TM 1,2) and routes the TC packet onto the TC-bus. The application software addressed by the application process identifier in the packet header removes the TC from the TC-bus, checks the telecommand parameters and executes the telecommand itself or routes the TC via the Mil-Std-1553B bus to the final destination to perform the TC handling. If the TC is executed by the avionics software itself, the avionics software will produce the appropriate telecommand execution report. If the telecommand is processed by an external unit (NSGU, PFSU, PLSU) this unit will generate the reports.

The following telecommand types are implemented:

- direct telecommands (DTC) are directly executed by the command pulse distribution unit (CPDU) of the TC-decoder without any software interaction
- immediate telecommands are processed and executed immediately after reception by the software
- time tagged telecommands are inserted into the master time line according to the time tag information and executed by the software when the time tag is due.

Telecommands are in addition classified according to their criticality:

- potential hazardous commands: if executed at the wrong time or in wrong configuration, this type of command could cause the loss or damage of the satellite. For this reason these telecommands are protected by a HW measures like arming by DTC or by electrical inhibits such as separation straps (for the commands which need to be performed autonomously).
- vital commands: if executed at the wrong time, this type of command could cause degradation of the mission. Therefore these commands are protected by a software arming and firing mechanism.
- non-critical commands: all other commands

**Telemetry.** The downlink data are transferred from the ICDU, from the satellite subsystems and from the payload through the TM encoder, the Security Unit until the TM transmitter. The different applications of the on-board software either generate PUS-packets itself or read PUS-packets from external units (NSGU, PLSU, PFSU). The avionic software forwards these packets to the TM-encoder of the ICDU or to the internal data storage for later downlink. The TM-encoder takes either real-time packets from virtual channel (VC0) or replay data via VC1.

If no TM data are available the ICDU will generate idle packets via VC7 automatically. The TM-frames generated by the ICDU are passed through the PFSU without processing if the PFSU is in clear mode or will be encrypted by the PSFU if it is in secure mode. From the PFSU the data are sent to the transponder.

The telemetry concept is driven by the services defined in the PUS. Each service generates defined telemetry reports. One report can consist of more than one packet.

The reports can be:

- a response to a telecommand (e.g. memory dump)
- generated periodically if enabled by telecommand (e.g. housekeeping reports)
- an asynchronous report due to an on-board event
- a telecommand verification report generated during the different steps of the telecommand processing.
- a special TM report (e.g. encrypted TM packets).

All telemetry packets are put on a so called TM-bus. A telemetry router as part of the system control application reads periodically the TM-bus and routes the packets according to their destination which can be defined by PUS-service 15 telecommand either to a packet store within the HK-memory or to the TM forward control (PUS service 14). The TM forward control allows the control of which TM packets are enabled to be sent to ground. All enabled packets are put by the TM manager to the virtual channel VC-0 hardware interface. Finally the HW TM-encoder takes the TM-packets from both virtual channels and generates TM-frames to be downlinked to ground. The HK-memory output is routed directly to VC-1 without any SW interaction, beside start and stop of the HK-memory output.

**On-board Storage and Retrieval.** The ICDU provides the capability to store 1.65 Gbits of telemetry data, which can be downlinked to ground via VC-1 on request

(TC). This allows the storage of selected telemetry data during the time between two ground contacts. It is possible to define up to 5 different packet stores, which can be managed by telecommand independently. PUS service 15 provides the necessary commands and telemetry reports to monitor and control the different packet stores.

**On-board Communication.** The main interface for data exchange between the central avionics system and the platform units and the payload is based on an on-board bus concept realized by the Mil-Std 1553B bus. Bus controller (BC) is located in the ICDU. The following units are connected to the bus: NSGU, PLSU, RTU, PFSU, PCDU,

Figure 10 gives an overview how the units are connected to the Mil-Std-1553 and how the redundancy concept is realized.

Bus controller and remote terminals are connected to the redundant bus lines via long stub transformer bus couplers. Each RTU has two Mil-bus remote terminal interfaces, where each interface is connected to both busses. The ICDU provides two bus controllers each of them being able to access to both busses.

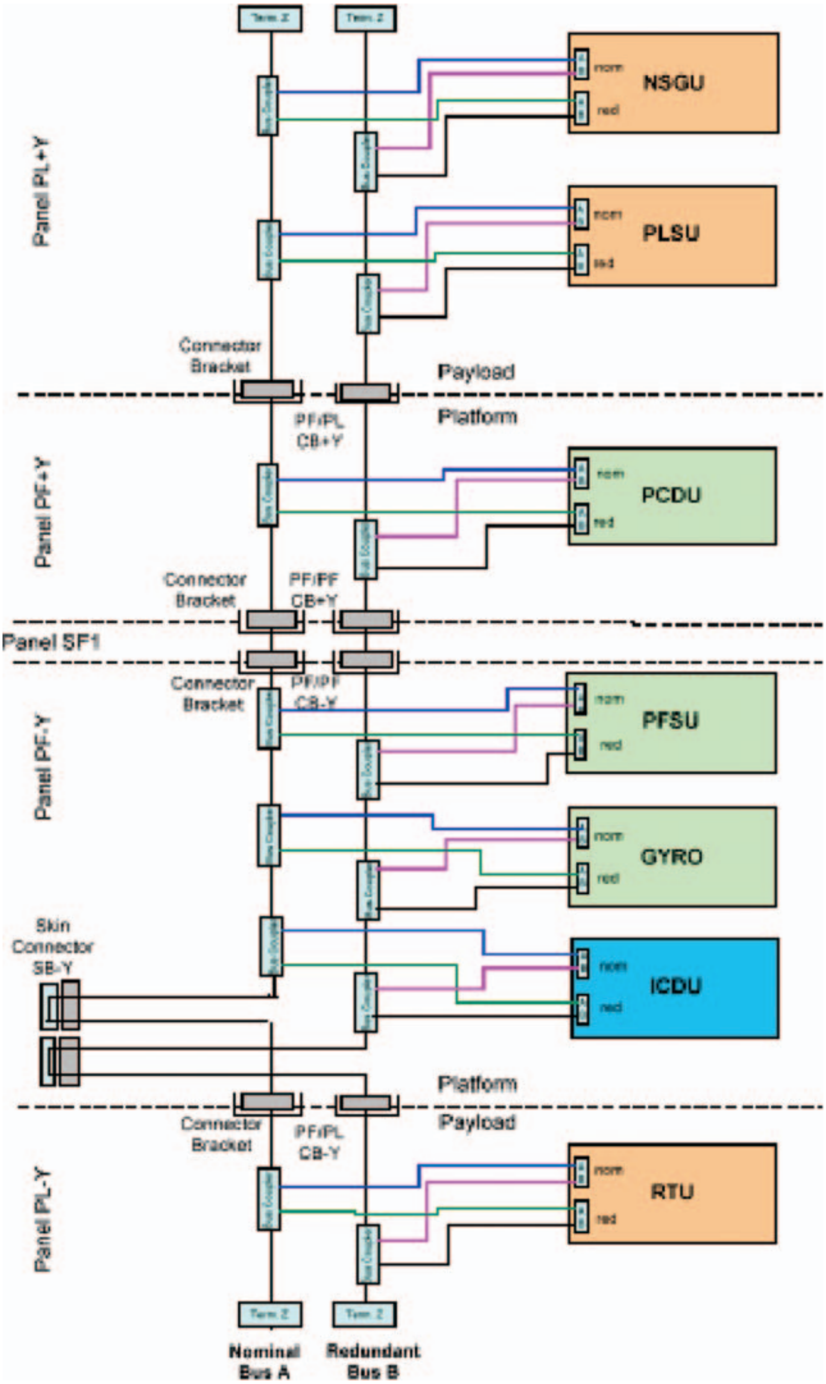
**On-board Time Management.** In the following, the baseline on-board time management architecture is described.

As shown below five different and independent time management systems exist onboard the GALILEO satellite:

– NSGU Time: This is the highly accurate time, needed for the navigation mission. The time origin is the CMCU / Atomic Clocks. It is called Local Galileo System Time (LGST).
– PLSU Time PRS: This is a separately generated time function at PLSU side synchronized by a pulse per second (PPS) and the LGST received from the NSGU.
– PLSU Time C-Band: This is a separately generated time at PLSU side, synchronised to the Galileo Time by (second level encrypted) ground command or synchronized to the PPS and LGST provided by the NSGU. This time is mainly needed for TC time tagging and TM time stamping.
– PFSU Time: This is a separately generated time at PFSU side, synchronised to the Galileo Time by (second level encrypted) ground command, mainly needed for TC time tagging and TM time stamping at PFSU side.
– ICDU Time: The ICDU Time is the time generated by the ICDU of the avionics subsystem of the GALILEO Satellite. The time is needed on-board for correct performance of the satellite (P/F) system-, subsystem management and attitude and orbit control functions, which are required for safe and autonomous operation of the satellite during all mission phases including also non-nominal situations.

## 7 FDIR Architecture

The overall FDIR is organized in a hierarchical form defined by four levels with increasing complexity in terms of recovery actions. The goal of this organization is to recover from failures on the lowest possible level. In principle an additional (w.r.t. the ones listed below) level 0 exists, which is equipment internal (e.g. EDAC) and considered transparent.



Panel PL+Y
Connector Bracket
Payload
Platform
PF/PL CB+Y
NSGU
PLSU
nom
red
Panel PF+Y
PCDU
Connector Bracket
PF/PF CB+Y
Panel SF1
Connector Bracket
PF/PF CB-Y
Panel PF-Y
PFSU
GYRO
Skin Connector SB-Y
ICDU
Platform
Payload
Connector Bracket
PF/PL CB-Y
Panel PL-Y
RTU
Nominal Bus A
Redundant Bus B


**Fig. 10.** Buses.

The FDIR functions are controllable by ground (i.e. enabled and disabled) at monitoring and recovery level.

Monitored values are filtered in order not to react on a single out of limit event.

All parameters for FDIR (e.g. thresholds and filters) are modifiable by telecommand.

When in Intermediate Safe Mode or in Safe Mode the Satellite rejects on-board stored commands (e.g. in the Master Time Line) and all ground commands (but a dedicated command allowing to recover normal command operation) and relies on the ground for the transition back to Normal Mode or to Sun Acquisition Mode. This allows the execution of telecommands which are no more suited to the new situation (entry in ISM) to be avoided.

To support ground investigation and recovery actions, the observability of the failure is a major issue.

Observability is ensured through:

– On-board storage of events in the event log buffer stored in SGM as well as in ICDU TM-Memory
– Telemetry which is stored in the ICDU TM-Memory and which provides all sensors acquisition data and actuator raw commands as well as the monitored FDIR parameters (>28 hours storage).

The general approach is to implement S/S FDIR functions directly in the related SW application (decentralised approach). This has the advantage to design, develop and validate these applications as a whole with less external interfaces. The use of OBCP will be limited to real justified needs.

**FDIR Levels.** The FDIR concept is aimed at minimizing the impact of all kinds of failures on the system performance by implementing 4 different FDIR levels with increasing complexity. Level 1 and 2 are S/W measures whereas Level 3 and 4 are H/W based.

**Level 1:** Failures allow the continuation of the current Satellite Mode. The detection and the isolation of the failure is based on equipment status information, consistency checks between units and surveillance of subsystem behaviour. After failure isolation the suspected unit(s) and/or interface(s) are switched over.

**Level 2:** Failures imply a Satellite Mode change, possibly in combination with an AOC mode or sub-mode change. These failures become apparent by violations of certain subsystem performances which require a block switching of all involved units and interfaces to be performed and/or to use another mode. As described in the chapter “Mode Dependent FDIR” the recovery mode to be applied is different in satellite NM where priority is put on the satellite mission availability to all other modes where priority is put on the satellite safety.

**Level 3:** These are malfunctions of the ASW internal to the software or caused by PM failures. Their detection is based on PM specific checks like the watchdog signal, etc. Recovery actions are Warm start, Reset, Cold start, Reconfiguration of the processor (implying a cold start of the new processor)

**Level 4:** Hardwired alarms like sun- and earth presence checks are implemented to detect major system anomalies. If one of these alarms becomes active, a Safety Sequence (SFS) is performed which consists of:

– A switchover to redundant ICDU and redundant equipments as pre-set by ground
– A transition to Satellite Safe mode and AOC safe mode
– Disabling of S/W checks and H/W alarms

**Level 0:** In addition to the FDIR levels 1 to 4 described above, also a FDIR level 0 exists. The level 0 mechanisms are part of the unit function (e.g. correction of a single bit by the EDAC) and are transparent for the FDIR system.

## 8 Satellite Main Budgets

**Mass Budgets**

| | |
|---|---|
| Overall Max. Mass: | 700 kg |
| Including Maturity and Uncertainty | |

**Power Budget**

*Safe Mode:*

| | |
|---|---|
| Max. Power in Sunlight: | 1290 W |
| Max Power in Eclipse: | 1345 W |
| Max Power Peak | 1525 W |

*Safe Mode:*

| | |
|---|---|
| Max. Power in Sunlight: | 950 W |
| Max Power in Eclipse: | 1115 W |
| Max Power Peak | 530 W |

*Safe Mode:*

| | |
|---|---|
| Max. Power in Sunlight: | 1290 W |
| Max Power in Eclipse: | 1345 W |
| Max Power Peak | 1525 W |