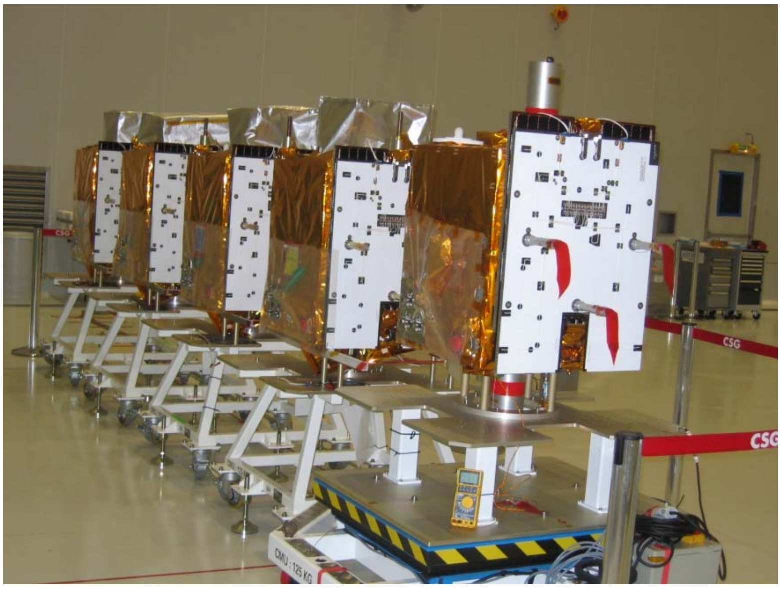
DOCUMENT

document title/ titre du document

RECURRING SERVICE MODULES

FOR FUTURE SCIENCE MISSIONS

A PRELIMINARY REVIEW



prepared by/*préparé par* Frederic Payot

reference/*réference*	
issue/*édition*	1
revision/*révision*	0
date of issue/*date d'édition*	19/12/2006
status/*état*	Draft
Document type/*type de document*	Final report
Distribution/*distribution*	

European Space Agency
Agence spatiale européenne

A P P R O V A L

Title *titre*	Recurring service modules for future science missions	issue *issue*	1	revision *revision*	0

author *auteur*	Frederic Payot	date *date*	19/12/2006

approved by *approuvé by*	Nicola Rando	date *date*	19/12/2006

C H A N G E   L O G

reason for change /*raison du changement*	issue/*issue*	revision/*revision*	date/*date*

C H A N G E   R E C O R D

Issue: 1 Revision: 0

reason for change/*raison du changement*	page(s)/*page(s)*	paragraph(s)/*paragraph(s)*

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LIST OF ACRONYMS

AAD	Attitude Anomaly Detector
AAS	Alcatel Alenia Space
ACC	Attitude Control Computer
AIT/V	Assembly Integration and Tests/Verification
AME	Absolute Measurement Error
AOCS	Attitude and Orbit Control System
APE	Absolute Pointing Error
ASAP	Ariane Structure for Auxiliary Payloads
ASI	Agenzia Spaziale Italiana
avg	average
AVM	Avionic Model
CDH	Command and Data Handling
CDMS/U	Central Data Management System/Unit
CNES	Centre National d'Etudes Spatiales
COTS	Commercial Off The Shelves
CRS	Coarse Rate Sensor
DoD	Department of Defence
DOD	Depth Of Discharge
EEE	Electrical, Electronic and Electromechanical
EGSE	Electric Ground Support Equipment
EM	Engineering Model
EO	Earth Observation
EOL	End Of Life
ERS	European Remote Sensing
ESA	European Space Agency
ESTEC	European Space Research and Technology Centre
FDIR	Failure Detection Isolation and Recovery
FEEP	Field Emission Electric Propulsion
FM	Flight Model
FOG	Fiber Optic Gyro
GEO	GEostationary Orbit
GMM	Geometrical and Mathematical Model
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Centre
GTO	Geostationary Transfer Orbit
HGA	High Gain Antenna
HW	HardWare
I/F	InterFace
JWST	James Webb Space Telescope
L2	2^{d} Lagrange Point of the Earth Sun System
LEO	Low Earth Orbit
LEOP	Launch and Early Operation Phases

LGA	Low Gain Antenna
LLI	Long Lead Items
LOS	Line Of Sight
MEO	Medium Earth Orbit
MGA	Medium Gain Antenna
MGSE	Mechanical Ground Support Equipment
MM	Mass Memory
MSG	Meteosat Second Generation
MTL	Mission Timeline
NASA	National Aeronautics and Space Agency
OBC	On board Computer
OBCP	On Board Computer Program
OBDH	On Board Data Handling
OBMM	On Board Mass Memory
ORS	Operationally Responsive Space
OSR	Optical Solar Reflector
PCDU	Power Control and Distribution Unit
PCS	Power Control Subsystem
PDE	Pointing Drift Error
PFM	ProtoFlight Model
P/H	Planck/Herschel
P/L	PayLoad
PLM	PayLoad Module
PRE	Pointing Reproducibility Error
PTTS	Platform Tank Support Structure
PWM	Pulse Width Modulation
QM	Qualification Module
RCS	Reaction Control System
RF	Radio Frequency
RFDN	Radio Frequency Distribution Network
RPE	Relative Pointing Error
RSDO	Rapid Spacecraft Development Office
RWA	Reaction Wheel Assembly
SA	Solar Array
SAA	Sun Aspect Angle
SAR	Synthetic Aperture Radar
SAS	Sun Acquisition Sensor
S/C	SpaceCraft
SCC	Sorption Cooler Compressor
SCS	Sorption Cooler System
SDE	Software Development Environment
SLT	Static Load Test
SSMM	Solid State Mass Memory
SSTL	Surrey Satellite Technology Limited
STM	Structural and Thermal Model
STR	Star TRacker

SVM	SerVice Module
SW	SoftWare
TBC	To Be Confirmed
TCS	Thermal Control System
TM/TC	TeleMetry/TeleCommand
TT&C	Telemetry Tracking and Control
TWTA	Travelling Wave Tube Assembly
w/o	without
WU	Warm Unit
XMM	X ray Multi Mirror observatory
XPND	Tranponder

INTRODUCTION

Over the last few years, due to a diminishing purchasing power and difficulties in increasing the available budget, cost effectiveness has become a great concern within ESA. Indeed, missions cost have noticeably increased (especially for science missions) and many initiatives have been undertaken to control and limit the expenditure by streamlining processes and resources, especially in order to implement a mission within a more restrictive budget.

In addition, the scientific community requirements are more and more challenging: demanding mission objectives lead to more complex mission concepts. Moreover, a quicker response time from approval to launch would be desirable, whilst keeping a very high overall level of reliability.

The main objective of this study is to review the application of recurring service modules as a potential answer to the challenges listed above.

Recurring service modules have not frequently been used in science missions mainly due to the specificity of the mission requirements, naturally leading to a fully customized design. Reuse of common service modules is also hindered by the relatively long time interval between similar missions. Additionally, science missions are very specific with each mission different and mission opportunities are generally too few to make this concept viable.

In the first part, the objectives of the study will be explained and an overview of recurring platforms will be given, showing the different concepts in the different domains of application. Then, a review of existing common platforms will be done to be followed by considerations about the common platform design. Finally, programmatic consequences of the use of generic platforms will be detailed in the last chapter of the first part.

The second part will present the study cases which will illustrate the ideas and the issues raised in the previous part. It will consist in the common service module development for Herschel and Planck and the proposals to reuse it for Gaia and Eddington.

OBJECTIVES OF THIS STUDY

This study provides with a preliminary analysis of recurring platforms in preparation for potential future studies. The goals of this work are:

- to identify the different types of reuse and related motivation in the different applications
- to review the existing reusable platforms (with a special emphasis on European activities)
- to analyse the level of maturity of this approach
- to analyse technical solutions allowing the definition of a versatile generic service module
- to analyse programmatic issues applicable to the use of generic platforms such as cost, risk management and schedule
- to identify the benefits and the drawbacks of such an approach especially for science missions
- to identify the ideal conditions applicable to platform reuse to be the most efficient
- to identify future investigations on the subject

PART 1:

RECURRING SERVICE MODULES
FOR FUTURES SCIENCE MISSIONS

1 RECURRING PLATFORMS: AN OVERVIEW

This section provides basic explanations on the concept of "recurring platforms" explaining the different trends. In order to avoid any misunderstanding, some terms need to be discussed.

"Recurring", "reusable", "generic", "standard" and "common" will indiscriminately be used in this document. In the same way, platform, bus as well as service module will indiscriminately be used as well, although the latter is generally the dedicated term when there is a clear distinction between payload module and service module.

The spacecraft platform includes all the subsystems (structure, mechanisms, thermal control system, attitude and orbit control system, telemetry and telecommand, on board data handling system) necessary to support the payload, providing services for mission success.

1.1 Different levels of reuse

Reusability is a generic term which is used to indicate different situations that need to be explained. In particular, reuse can be applied at different levels, from unit to module level:

- Unit level:

Individual items (solar cells, batteries, sensors, actuators, thrusters, electronic units and so on) can be reused from a mission to another. The advantage is that these devices are space proven (sometimes cumulating years of flight experience, which is the case for devices used in Earth observation and in telecommunications as example) and most of the times, qualified at unit level. Performance of the unit is therefore very well known and generally does not require technology development thus lowering risk of sliding schedule. Nevertheless, these considerations apply to the unit level and do not take into account system accommodation: layout, interfaces, operating modes and integration at higher level need to be adapted for mission requirements and most of the times, delta engineering is required. This type of reuse, which is not discussed further in this study, is widespread and applied in all type of space missions.

- Subsystem level:

Sometimes, it is possible to reuse whole subsystems of a platform, as it is the case for the Rosetta, Mars Express and Venus Express for instance. Venus Express chemical propulsion system is inherited from Mars Express with minor modifications whereas thermal control had to be changed.

Generally speaking, the electrical and functional architecture is mostly reused, whereas structure, thermal control and AOCS hardware generally needs adaptations. These adaptations require considerable work unless the subsystems are well decoupled.

- Module level:

Finally, when the mission profile suits the platform specifications, a full module reuse can be considered. The typical examples are the different spacecraft bus product lines available on the market for low Earth orbit and telecommunication missions, with relatively fixed performances in terms of mass, orbit and power supply, although some extensions are been studied and also common service module development, explained further in the next section. Despite a high level of reuse, there will always be some delta activities due to payload accommodation, as a minimum concerning software adaptation. An exception to this assertion is the new programme being developed by the American Department of Defence, called Operationally Responsive Space (ORS) which is meant to provide a rapid and low-cost satellite planning with plug and play payloads and subsystems. This concept allows a rapid integration or replacement of hardware and therefore multiple configurations (and reconfiguration), without any extra development activities.

1.2 From reuse opportunity to serial product

It is possible to make out different types of reuse according to the different planning philosophy and according to the scope of the space mission.

When we realize that a platform can be reused for one or several missions in addition to the first mission initially planned, without being aware that this multiple use could be possible right at the beginning, we will refer to **reuse opportunity**. The typical example of such a reuse is Mars Express/Venus Express missions. In 2001, a couple of year before Mars Express launch, ESA issued a Call for Ideas to react to the possibility of a low cost mission based on the reuse of the platform developed for Mars Express. Three scientific missions were assessed and Venus Express was finally selected. Some instruments were even reused from Mars Express. The low cost constraint and a very strict time schedule lead ESA to proceed this way. The mission was implemented within three years from approval to launch, for 82.4 M€ (EC 2003) including the design and the development of the spacecraft. The Mars Express spacecraft bus could accommodate Venus Express experiments with little modification. This approach mainly resort to subsystem reuse and can be efficient only for missions which have very similar profiles (similar resources, constraints and operations). Indeed, when a platform has been optimized for one specific mission, it is difficult to carry out major design modifications afterwards, since an adaptation of a subsystem has often an impact at system level.

Another type of reuse, specific to science missions, occurs when a consistent approach has been planned to develop a **common service module for a few missions** (typically two), with very similar profile (e.g. observatories for astrophysics missions). Even if the service module is used for only two missions, cost reduction can be ensured by avoiding duplication of efforts and resources (see the study case). Another example is represented by the case of XMM and Integral.

Then, real platform **product lines** have been developed by space industries. The idea is to provide reusable platforms able to be adapted to a more or less wide range of missions. Telecommunications missions have first taken benefit from this concept as early as in the seventies, with a huge number of similar satellites in geostationary orbit for instance. More recently, a specific market appeared for missions in low Earth orbit. This approach consists in making an easier access to space, thus increasing mission opportunities but it requires a consistent approach from the development of the generic platform to the implementation for different missions to be effective. Proteus and Prima are two examples of recurring platforms for medium sized satellites in low Earth orbits. This type of common platform is flexible enough (in its design but also in all its segments of a space mission architecture) mainly thanks to modularity.

Finally, when several identical spacecrafts need to be launched (constellations such as Iridium for mobile telephony for instance), then we can describe this reuse as **serial products** and serial production principles can even be applied for assembly, integration and tests of the satellites. The reuse is close to 100 % and is not only at the platform level, but also at spacecraft level. As far as constellations are concerned, launching a high number of satellites within a very short period of time is an additional constraint to consider. Sometimes, several launchers are required and the spacecraft compatibility with them needs to be considered.

Eventually, the higher the reuse level is, the less flexible the platform becomes (see Table 1): a unit can be reused in many cases with different configurations but an entire service module can only be reused in specific cases, with little modifications. Indeed, any major modification in the service module design can have tremendous impacts on the other subsystem design and of course on cost.

Level of reuse	Type of reuse	Flexibility: adaptations at system level	Cost reduction
Unit level	Any	+++	+
Subsystem level	Reuse opportunity	++	++
Module level	Product lines	+	+++
Entire spacecraft	Serial products	no	++++

Table 1: Reusability and related flexibility

1.3 Reuse in different application domains

It is interesting to study how the different types of space missions deal with recurring platforms. Table 2 summarizes the main trends which are discussed further below.

Application	Earth Observation	Navigation	Telecom (GEO)	Science (planetology, fundamental physics, astrophysics)
Typical spacecraft mass	> 50 kg	> 100 kg	> 1000 kg	> 500 kg
Type of reuse	From unit to module levels, with several generic bus available	Serial products (full reuse)	From unit to module levels with standard platforms available	From unit to subsystem level, fewer cases of reuse at module level
Examples	Proteus, Myriade, Minisat 400 (SSTL), BCP series	Galileo (under development), GLONASS, GPS	Alphabus, Eurostar, Spacebus	LM 900, Surrey Interplanetary Platform, Rosetta-Mars/Venus Express, XMM/Integral service module Herschel/Planck service module

Table 2: Overview of recurring platforms

1.3.1 EARTH OBSERVATION

Missions dedicated to Earth observation are usually placed low Earth orbits which altitude ranges from 400 to 2000 km with any inclination (quasi equatorial to polar). The environmental conditions (air density, dust and particles sharing, rays) are well defined and the conditions are somehow constant.

This application is characterized by a wide range of spacecraft mass: it goes from nanosatellites (see SNAP platform from SSTL) to several ton satellites (such as MetOp1 with its 4 tons for instance) but the typical spacecraft mass is of order hundreds of kg.

The cost of such missions is rather affordable given the LEO (reduced launch cost), the mature S/C design and the market size (commercial image providers, hazard prevention, security, etc.). This situation is favorable to the development of numerous missions with similar requirements by taking advantage of cost savings in recurring platform hardware in particular. Actually, many common platforms have been developed by European industries for this type of missions such as Flexbus, Leostar, Mita, Myriade, the Polar Platform, Prima, Proteus, SARsat… (See next section for more details).

The availability of these platforms points out the fact that Earth observation is well suited for platform reuse.

1.3.2 TELECOMMUNICATION

Telecommunication missions generally have a very long lifetime (sometimes over 15 years), with a high mass (several tons for some missions in geostationary orbit), high power (10 kW or more) and support very big antenna. They represent the more frequent launched missions with satellites in low Earth orbit (constellations for mobile telephony) or geostationary orbit (for direct television broadcasting).

LEO:

This type of orbit is currently used for constellations of identical spacecrafts for instance. Designing a constellation raises the development of the inter-satellites link and many aspects of the conception, launch, positioning and station keeping are specific. Manufacturing is also a major issue since a high number of satellites have to be built in a very short period of time (for Iridium constellation: a satellite produced every 22 days). Several constellations were under study in the last years but severe problems of financing stopped many of them.

GEO:

Since the price of GEO communications satellites is generally very high and requires long term investments, a common bus design approach has been used for this type of satellites since the early 70's. Several common platforms are manufactured by the main space industries. While companies do not stock GEO spacecraft buses in their factories waiting for orders to materialize, they are able to quickly produce a spacecraft bus for a given GEO mission with only minimal changes to their basic bus design, manufacturing and integration procedures. Typically, the major variations in a GEO bus design involve only power levels and stability requirements. The reuse tends to be as full as possible, with flight proven equipment which can also be used in other domains.

The success of a common spacecraft bus design for GEO communications satellites is due to the large and increasing number of missions at the GEO orbit, the commonality of missions' requirements, and uniform orbit geometry (from mission to mission). In Europe, EADS Astrium and Alcatel provide each a family of reusable platforms (respectively Eurostar and Spacebus) while Alphabus is being developed by ESA, CNES, EADS Astrium and Alcatel Alenia Space as major contributors. This standard platform will be able to support 1.2 ton payload and deliver up to 18 kW for an overall mass of 6-8 tons.

1.3.3 NAVIGATION

As far as constellations are concerned, navigation missions requirements are quite close to telecommunications' ones with respect to the high production rate and the necessity to launch numerous satellites in a short period of time. Another particularly demanding challenge is the control of a multiple satellite system. Generally, satellites are in different orbit planes in MEO. The typical spacecraft mass is hundreds of kg and costs generally very high for constellations.

The trend for this type of application is the creation of a dedicated spacecraft with subsystems available on the market.

Galileo is the European initiative to settle a constellation of 30 satellites to provide navigation services preceded by 2 tests satellites (Giove A and B). At the moment, the first has been successfully launched and is currently carrying out tests. GLONASS (Russian GLObal Navigation Satellite System) is a constellation of 24 satellites initially, with 16 in operation now. The American version is the well known GPS with 24 satellites in orbit.

1.3.4 SCIENCE

Science missions, which are the main subject of this analysis, are a very broad classification encompassing very different missions: planetary missions (both remote sensing and in situ science), astrophysics missions with observatories, missions to study the magnetosphere, fundamental physics missions. They all have very specific mission requirements that can differ considerably, in terms of orbit payload and operation:

- Orbit geometry: from low orbits to interplanetary trajectories to orbits at the vicinity of Lagrange points of the Sun-Earth system. This design driver also determines the environment of the mission (flux from the Sun, the Earth, particle, electromagnetic environment).
- Instrument type from various disciplines: astrophysics, environment study, fundamental physics, and planetology
- Mission operations: different operation concepts (e.g. observatory or planet exploration)

Therefore, it is more difficult to consider recurring platform for this type of missions since the mission requirements to be satisfied can be very different and the flight opportunities are less numerous than any other type of mission.

Science missions cost is generally relatively high since a science mission typically requires large amounts of technology development and also because of the high level of performance required. A highly adaptable and flexible core spacecraft bus design would be necessary to respond effectively to the broad range of science requirements, while minimizing cost, fabrication schedules and development risks. The question will be to know whether cost reduction is still effective for this type of application: if the platform has to be too much tailored with many modifications from the core platform, time and cost savings may not be as high as expected.

Science missions and recurring platforms:

A number of science missions have made use of recurring platforms. In particular in the case of **product lines** available from the industry, we can mention the following examples:

Mission name	Reusable platform	Mission objectives
Agile	Mita	γ-ray sources observation
Champ	Flexbus	Earth gravity field study
Corot	Proteus	Astroseismology and Earth like planets searching
Microscope	Myriade	Test of the equivalence principle
Picard	Myriade	Sun observation

Table 3: European science missions using recurring platforms

Another type of recurrence can appear when a **common development for the service module** has been planned for a few missions. There are two typical examples among ESA missions: XMM/Integral and Herschel/Planck (see Table 4).

<table>
<tr><th>Mission name</th><th>Launch</th><th>Commonality</th></tr>
<tr><td>XMM</td><td>1999</td><td rowspan="2">XMM service module design as reference to be adapted to Integral requirements</td></tr>
<tr><td>Integral</td><td>2002</td></tr>
<tr><td>Herschel</td><td rowspan="2">2008 (dual launch)</td><td rowspan="2">Symmetric role in the design development</td></tr>
<tr><td>Planck</td></tr>
</table>

Table 4: Examples of ESA science missions with a common service module development

For XMM/Integral, the service module commonality approach has ultimately proved highly successful in terms of savings in development costs, with the sharing of flight spares and the re-use of thermal and electrical models, as well as ground-support equipment.

Some benefits were also obtained in the operations area, although commonality was not implemented as systematically there as on the satellite-development side.

The idea of reusing XMM service module had to be implemented as quickly as possible in order to benefit from the potential commonality savings. Many options had been considered, ranging from having one prime contractor build both service modules in series, to two different prime contractors sharing a common design. This last option was finally adopted.

Had the schedule of the two projects drifted apart by more than a year during this period, then the cost savings would not have been so great. Close coordination between the ESA project teams was essential to ensure timely procurement of the long lead items.

For Herschel/Planck programme, further information can be found in the study case in the second part of this document.

A last illustration, already mentioned, is the **reuse opportunity** with Rosetta/Mars Express/Venus Express example and with Solar Orbiter whose heritage from Bepi Colombo is being studied.

We can notice that in both cases, the launch dates are very close to benefit from the reuse of parts.

Finally, we should highlight that the type of reuse also depends on the type of the mission. For instance, there is no clear distinction between a payload module and a service module for planetary missions, therefore the reuse is generally limited to subsystems. On the contrary, astrophysics missions are better suited to support an entire module reuse

2 EXISTING PLATFORMS AND ON GOING DEVELOPMENT

2.1 General review

The following table provides some information on existing generic platforms. Only European platform are mentioned: the blue written being dedicated to Earth observation and science missions. A more complete database, including non-European platforms, can be found in annex.

Name	Manufacturer	Application	Orbit
GeoBus (Italsat Bus)	Alcatel Alenia Space	Telecom	
ECS (OTS)	British Aerospace	Telecom	GEO
Spacebus 1000 (formerly called 100)	Alcatel Alenia Space	Telecom	GEO
Spacebus 200	Alcatel Alenia Space	Telecom	GEO
Spacebus 300	Alcatel Alenia Space	Telecom	GEO
Spacebus 400	Alcatel Alenia Space	Telecom	GEO
Spacebus 2000	Alcatel Alenia Space	Telecom	GEO
Spacebus 3000 B or C 1 to 4	Alcatel Alenia Space	Telecom	GEO
Spacebus 4000 B or C 1 to 4	Alcatel Alenia Space	Telecom	GEO
Eurostar 1000	EADS Astrium	Telecom	GEO
Eurostar 2000	EADS Astrium	Telecom	GEO
Eurostar 2000+	EADS Astrium	Telecom	GEO
Eurostar 3000	EADS Astrium	Telecom	GEO
Eurostar 3000 GM	EADS Astrium	Telecom	GEO
Eurostar 3000 S	EADS Astrium	Telecom	GEO
Alphabus	EADS Astrium/ Alcatel Alenia Space	Telecom	GEO
Proteus	Alcatel Alenia Space	Science, EO, telecom	LEO
PRIMA	Alcatel Alenia Space	Science, EO, (Telecom, navigation)	LEO, MEO (GEO)
MITA	Carlo Gavazzi Space	Science, EO, validation new technologies, (Telecom)	LEO

MiniFlex	EADS Astrium	EO, Science	
Flexbus	EADS Astrium	EO, Science	LEO
Leostar 200 (1.2 m fairing)	EADS Astrium	EO, Science, Telecom	LEO
Leostar 500 (2 m fairing)	EADS Astrium	EO, Science, Telecom	LEO
Leostar 500 XO	EADS Astrium	EO, Science, Telecom	LEO
Polar Platform (also called Spot Mk 1, 2, 3)	EADS Astrium	EO	LEO SSO
Myriade	EADS Astrium, ASPI, Latecoère (custumizer)	Science, EO, service demonstration for telecom, validation of new technologies	LEO
SAR-sat	OHB system	EO	LEO
MegSat	Meggiorin Group	Science, EO	LEO
Surrey Interplanetary Platform	SSTL	Science	
SNAP	SSTL	remote-inspection and formation flying missions	LEO
Minisat 400	SSTL	Earth Observation, communications and technology demonstration	LEO
MicroSat 100	SSTL	EO	LEO
MicroSat 70	SSTL	EO, communications and technology demonstration	LEO
Constella	SSTL	EO, Telecom, Navigation	LEO
Surrey Lunar Microsatellite (Moonshine)	SSTL	Science	NEO
Tubsat	Technical University of Berlin	EO, validation of new technologies	LEO

It should be noted that the information provided in the table is based on available sources as from December 2006. Changes are expected depending on specific mission requirements (lifetime, orbit, etc.).

2.2 European common platforms for Earth observation and science missions

There are several reasons for developing recurring platforms for Earth observation and science missions. One possibility for space industries is to develop a common platform "on their own" to lower the recurring cost and to be more competitive. This is the example of SSTL.

2.2.1 SSTL SERIES

Surrey Satellite Technology Limited is a company created in 1985 by the University of Surrey to commercialise the results of its innovative small satellite engineering research. Among its products, SSTL provides standard platforms for small satellites (from 6 kg up to 500 kg).

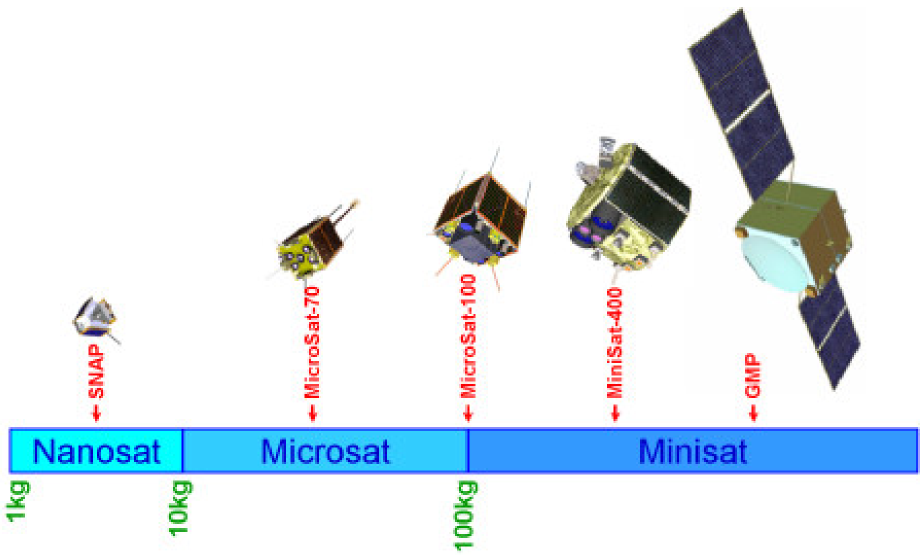


Figure 1: SSTL standard platform for small sat

SSTL was the first professional organisation to offer low-cost small satellites with rapid response. Since 1985, SSTL has been involved in 23 missions with different level of service (from know how transfer to turnkey service) and now employs 200 people.

SSTL philosophy is to provide low cost though reliable platforms using COTS items with many years of flight heritage and using its own facilities (cleans rooms, integration hall and mission operation centre). This approach has been successful and three of the SSTL standard platforms are now available on RSDO catalogue (see 2.3 for further explanations). Minisat 400 (used for UoSat 12) is one of them, the main characteristics of this platform are detailed hereafter:

Type of missions	Earth Observation, communications and technology demonstration
Orbits	LEO, altitude from 400 to 1400 km, any inclination
Launch vehicles	Dnepr, SS-18, Taurus,Athena, Delta II, Eurockot, Ariane-5, Cyclone, Zenit
Applications	UoSat 12 (launched in April 1999)
General shape	Cylinder with 9 lateral panels
Mass	Up to 400 kg wet mass
	Up to 200 kg P/L mass
Power	28 V bus voltage
	100 W EOL for P/L
	Single junction AsGa solar cells, body mounted SA on 9 panels
	NiCd battery, 20 Ah
Propulsion	Cold gas (N2) for propulsion
	10 thrusters (0.1 N)
	5.3 kg propellant mass
	ΔV: up to 15 m/sec
CDH, Telemetry and Telecommand	RS422/485 and CAN
	3 Gb mass memory
	Downlink: S band, 2 Mbps for P/L
	Up link 16/128 kbps
ADCS	3-axis stabilization
	Inertial, star or nadir pointing
	Pointing accuracy 360”
	Attitude knowledge: 72”
	GPS
Delivery time	18 months from approval to launch
Lifetime	1 year (nominal)

Table 5: Minisat 400 characteristics

Another possibility is for national space agencies to take the decision to develop a new recurring platform, using national industries for the manufacturing and testing of the platform. The idea is to increase mission opportunities (especially science missions) within national space program and to increase the competitiveness of national industries. In case of success, applications beyond the national space program can be considered. Two of the most significant examples of such strategy is the Proteus/Myriade product line developed by CNES and Prima/Mita developed by ASI.

2.2.2 CNES INITIATIVE

At the end of the 90's, CNES decided to develop two generic platforms (called "filières") for minisatellites in low Earth orbits: Proteus and Myriade. These two buses share the same modular philosophy with two cubic modules one upon the other: the service module and the payload module. They were designed to adapt to a wide range of missions, such as science and their main goal is to reduce the cost of space access and to increase mission opportunities within CNES program. Cooperation contracts have been signed with the different organizations so that each upgrade performed on the platform can benefit to all of them. This strategy has been a success and the platforms are now used for missions beyond CNES program.

2.2.2.1 Proteus

Proteus (Plateforme Reconfigurable pour l'Observation, les Télécommunications et les Usages Scientifiques) is a reusable platform for small sat (500-700 kg spacecraft mass). The development started in 1996 with Alcatel as prime contractor. The following table gives a quick insight of the performances of the platform. Most of the information is taken from the user's manual available on the internet: http://smsc.cnes.fr/PROTEUS/Fr/A_documentation.htm

Organisation	Collaboration between Alcatel and CNES
Type of missions	Science, EO, Telecom
Orbits	LEO (from SSO to almost equatorial, inclination from 20 to 145 deg), altitude from 500 to 1500 km
Launch vehicles compatibility	Compatible with all launchers in the 500 kg class (fairing diameter >1.9m): Ariane 5, LMLV2 (Athena), Cosmos, Delta2, Delta3, LM-2D, PSLV, Rockot, Taurus (depends also on the P/L)
General shape/configuration	1 m cubic shape, upper part: simple mechanical interface for P/L fixation
Mass	Up to 670 kg wet mass
	Up to 360 kg P/L mass
Power	300 W avg for P/L, 28 V unregulated
	Silicon cells
	Two symmetric wing arrays attached near to the satellite centre of mass with two single-axis stepping motor drives.
	Li Ion battery, 78 Ah
Propulsion	Hydrazine monopropellant (blow down system)
	Four 1N thrusters
	ΔV: 130 m/s (for a 450 kg spacecraft)
CDH, Telemetry and Telecommand	Electrical, on-board command and data handling architecture centralised on one single computer, the Data Handling Unit (DHU)
	2 Gbits mass memory for P/L
	Downlink: S band, 722 kbits/s
	Uplink: S band, 4 kbits/s
ADCS	3-axis stabilisation
	Earth, anti-Earth, inertial, track pointing
	Typical pointing accuracy: 0.05° (3 σ) on each axis
	Pointing stability: from 3.10^{-4} deg/s to 10^{-2} deg/s (depends on the frequency of the perturbation)
	GPS
Mechanical interfaces	4 surfaces which allow to mate the P/L on the platform
Delivery time	24 months
Lifetime	3 to 5 years depending on the orbit
On going developments	Enhanced performance and an even wider range of missions - Small science/ observation satellites - Agile satellites - Large science satellites - Telecom applications (LEO / small GEO) - Navigation MEO
Cost of the platform	15-20 M€ for SVM HW procurement 15-20 M€ for SVM and S/C AIT

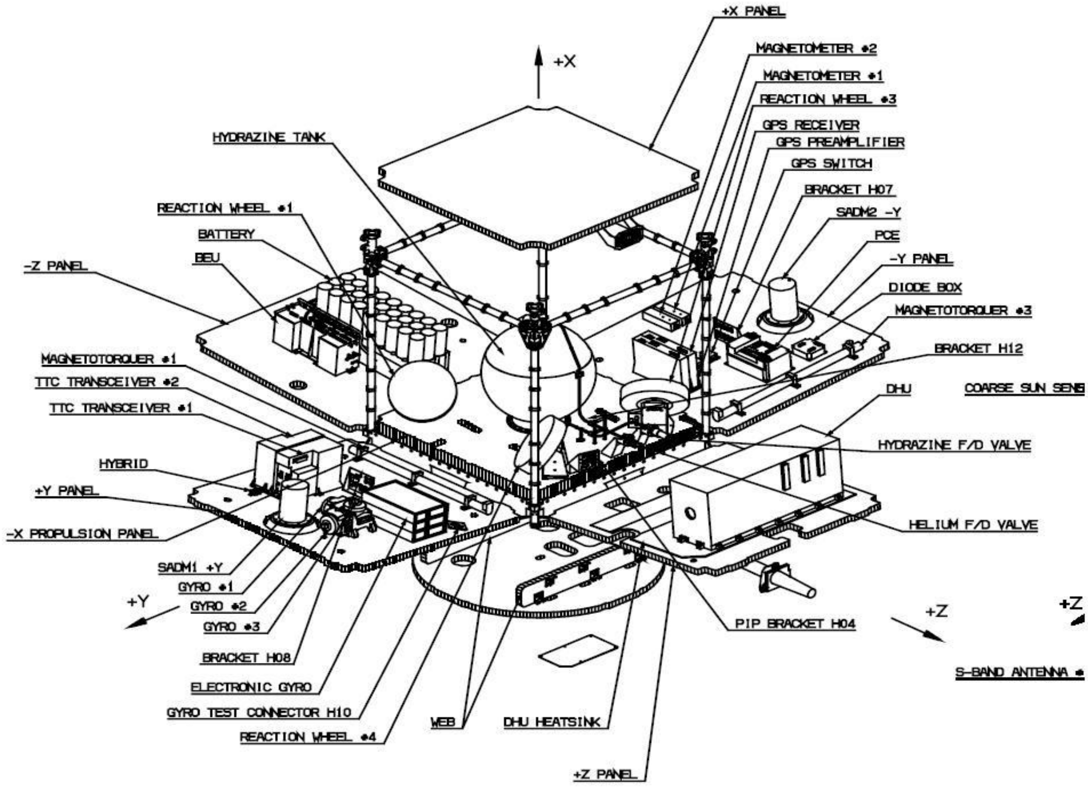


Figure 2: Proteus general layout

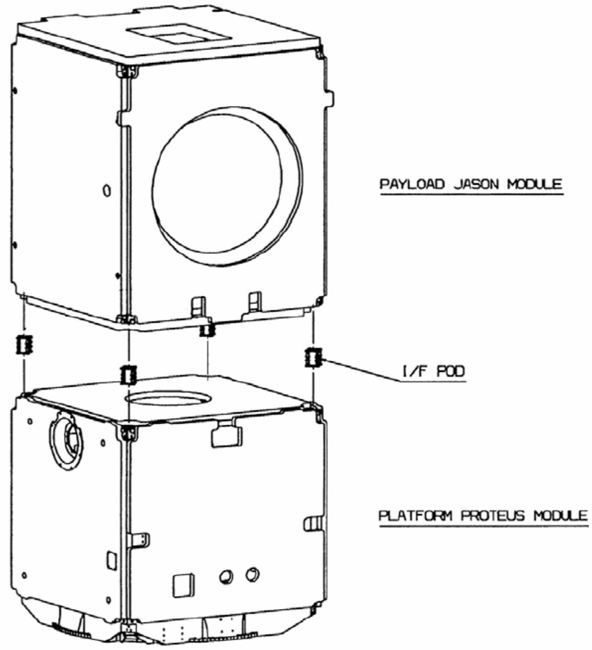


Figure 3: Payload integration on Proteus (Jason 1)

Mission	Application	Launch
Jason 1	Altimetry	Dec 2001
Calipso	Remote sensing	April 2006
Corot	Helioseismology, Earth like planet detection	Dec 2006
SMOS	Soil moisture, ocean salinity	End 2007
Jason 2	Altimetry	2008

Table 6: Proteus' missions

At the end of the 90's, Alcatel signed an agreement for the procurement of 5 platforms corresponding to the 5 first missions (see Table 6). Beyond CNES program, Alcatel can also use Proteus for other customers. A contract has recently been signed for the renewal of 48 satellites for the Globalstar constellation to be launched within 2009-2010, which will use some subsystems of the Proteus platform. Of course, CNES strongly encourages this kind of initiatives.

2.2.2.2 Myriade

Contrarily to Proteus, Myriade is rather a multi purpose set of functional chains that allow to design a 100-150 kg spacecraft. This means that the design is not fully frozen and that a larger variety of performance requirements can be met. The development started in 1999 with Alcatel and Astrium as industrial partners. The following table gives a quick insight of the performances of the platform.

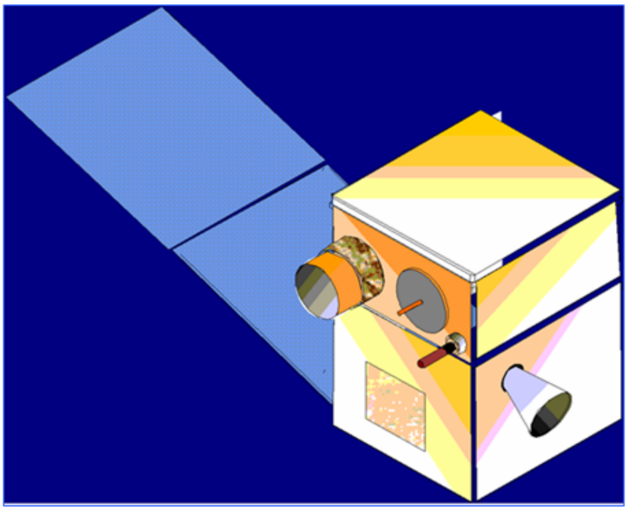


Figure 4: Myriade overview (payload module on top)

Organisation	Collaboration between ASF, AAS, and CNES
Type of missions	Science, EO, service demonstration for telecom, validation of new technologies
Orbits	LEO: Equatorial to SSO, from 500 to 1000 km (710 nominal)
Launch vehicles compatibility	ASAP 5 (up to 2008-9), PSLV, Dnepr, Soyuz, Vega
General shape/configuration	Modular concept: SVM: 60 cm x 60 cm x 100 cm PLM: 60 cm x 60 cm x 30 cm
Mass	Up to 140 kg wet mass (120 kg nominal)
	P/L: 56 kg (ASAP 5), 66 kg (Dnepr), w/o propulsion system
Power	175 W BOL supply @ 75°C
	50 W avg for P/L
	AsGa solar cells
	Li Ion battery, 15 Ah
Propulsion (optional) + 11 kg	Hydrazine monopropellant (blow down system)
	Four 1N thrusters
	ΔV: 80 m/s for a 120 kg S/C
CDH, TM & TC	Central OBC made of COTS items
	1 Gbits mass memory (16 Gbits optional)
	Downlink: 10 to 400 kB/s in S band, up to 16.8 Mb/s in X band (optional)
	Uplink: 4 to 20 kB/s in S band
ADCS	3-axis stabilisation
	Nadir (validated), Sun, inertial and V pointing (need for delta engineering)
	Pointing accuracy: < 5° (coarse mode), < 5 10^{-3} ° (fine mode)
	Pointing stability < 3'/sec
	GPS in option
Mechanical interfaces	Interface with the launcher: generic rigid lower plate with a shock damper system
Cost of the platform	6 M€ for SVM HW procurement 6 M€ for SVM AIT
Delivery time	24 months
Lifetime	3 years (nominal)
On going developments	Lower inclination, GTO (Spirale)

Mission	Application	Launch
Demeter	Detection of Electro-Magnetic Emissions Transmitted from Earthquake Regions	June 2004
Parasol	Remote sensing	Dec 2004
Essaim	Analysis of the electromagnetic environment (for French DoD)	Dec 2004 (4 sat)
Spirale	Early detection of missiles plumes (for French DoD)	End 2008 (2 sat)
Picard	Sun observation	Beginning 2009
Alsat 2	Earth observation for Algeria	2009
Elisa	Demonstration for formation flying (for French DoD)	2010 (4 S/C)
Microscope	Test the equivalence principle	2010

Table 7: Myriade's missions

Although these two product lines seem to be real success stories and as far as science is concerned, we must keep in mind that 4 over 5 of the Proteus' missions are dedicated to Earth Observation and 3 over 7 of Myriade's missions are French DoD missions (see Table 7).

Whereas no user manual is available for Myriade, a very detailed user manual is available on the web for Proteus, defining very precisely the interfaces, the payload envelope, taking advantage of numerous updates and feed back from previous missions. This is mainly due to the fact that Myriade is a set a functional chains to be accommodated to the mission needs (more flexible that Proteus design), hence the difficulty to settle a user manual with nominal performances.

2.2.3 ASI INITIATIVE

Part of the "Piccole Missioni" ASI programme consisting in developing common platforms for low cost access to space in collaboration with Italian industries, PRIMA and MITA are two different standard platforms for science missions in particular developed by Italian industries, taking advantage of the Vega launch vehicle.

2.2.3.1 PRIMA

PRIMA stands for "Piattaforma Riconfigurabile Italiana Multi-Applicazione" and is being developed by Alenia Spazio (phase C completed in February 2005).

Type of missions	EO, Science, telecom, navigation
Orbits	LEO and MEO (GEO and GTO also possible)
Launch vehicles	Vega, Soyuz, Delta 2, Dnepr
Applications	Cosmo Skymed (radar part of Orfeo program) constellation of 4 satellites to be launched from 2007, Radarsat 2 (SAR) to be launched in 2007, David (data and video distribution)
Mass	400 to 1500 kg wet mass
	Up to 700, 800 kg P/L mass
Power	28 V unregulated power bus
	From 250 to 800 W for P/L
	GaAs solar cells
	NiH_2 battery
Propulsion	Hydrazine monopropellant (blow down system)
	Four 1 N thruster
	78 Kg propellant mass
CDH, Telemetry and Telecommand	300 Gb mass memory
	Downlink: S band, 1 Mbps
	ERC32 processor
ADCS	3-axis stabilisation
	Earth, Sun and inertial pointing with typical pointing accuracy: up to 0.015°
	Pointing knowledge: 0.003°
	GPS
Cost of the platform	Approximately 10 M€
Lifetime	5 years
On going developments (February 2005)	Electric propulsion for AOCS with maximum reuse of the technologies developed at the ASI Enhanced performance for the power subsystem Standardisation of the data and electric interfaces Easier integration Goal: 2 missions each 3 years

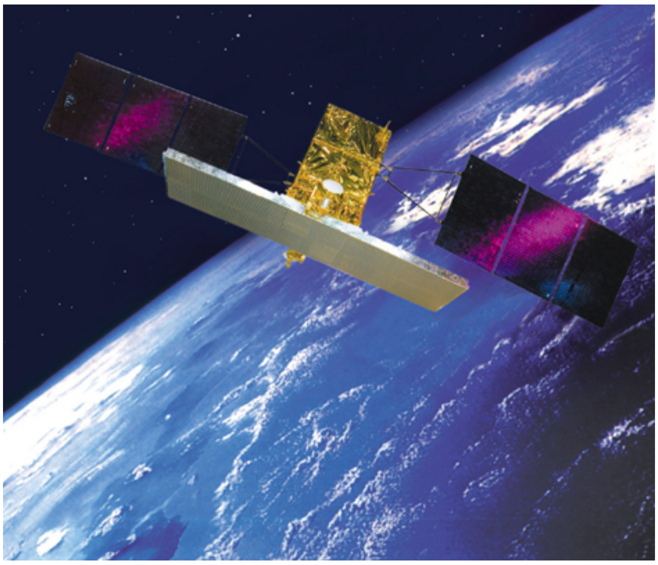


Figure 5: Cosmo Skymed on Prima platform

2.2.3.2 MITA

MITA stands for Minisatellite Italiano a Tecnologia Avanzata and Carlo Gavazzi is responsible for its development.

Type of missions	EO, Science, Telecom, validation of new technologies
Orbits	LEO
Launch vehicles	Cosmos, Soyuz, Vega, Ariane 5
Applications	Nina 2 (study of solar and galactic cosmic rays, 15th July 2000), AGILE (high-energy astrophysics studies) to be launched within a few months, HypSEO (validation in orbit of the design and technology of an Italian hyperspectral Imager for future accommodation on the Cosmo-Skymed constellation) to be launched in 2007
General shape/configuration	main body is based on a cubic shape module
Mass	150-350 kg wet mass
	Up to 70 kg P/L mass
Power	Supply: 200 W EOL
	P/L: 85 W avg, 120 W peak
	Fixed solar arrays
Propulsion type	FEEP (TBC)
CDH, TM & TC	2 computers: a multi-tasking OBDH and a P/L computer
	64 Mbytes mass memory
	Downlink: 1 Mbps in S band
	Uplink: 4 kbps in S band
ADCS	3-axis stabilisation
	Earth, Sun and inertial pointing
	Pointing knowledge: 0.1 to 1 deg / each axis
Adaptability/modularity	Modular design
Cost of the platform	5 M€ for a low requirement mission 10 M€ for a high requirement mission (P/L AIT and launch campaign excluded)
Development time	Typical phase C/D duration: 18 months
Lifetime	2 years

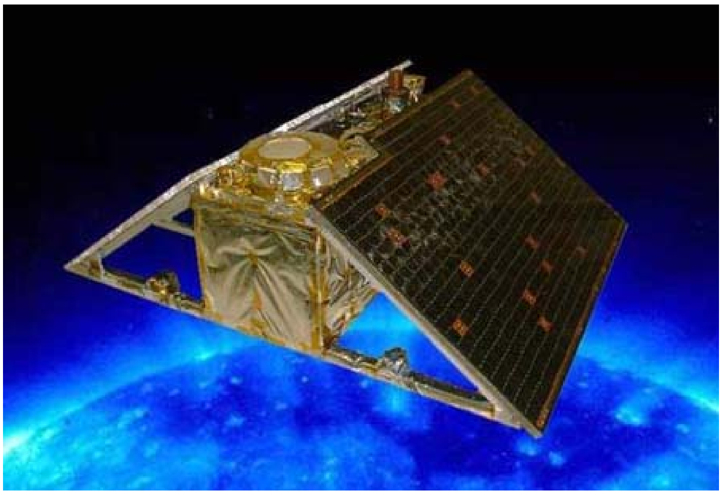


Figure 6: First Mita use with NINA 2 payload

2.3 About RSDO

RSDO, which stands for Rapid Spacecraft Development Office, is a NASA/GSFC department responsible for the management and direction of a program directing the definition, competition, and acquisition of multiple Indefinite Delivery/Indefinite Quantity (IDIQ) contracts. These contracts offer to NASA and to any other United States Government Agency an extremely fast procurement of spacecraft and payload for future missions. The Government guarantees to issue one or more orders for an amount not less than $50,000 under this contract. It is one of the most visible and successful implementations of "off the shelf" platform (core architecture and options) for science applications in particular. The current catalogue (Rapid 2) first released in 1997 gathers 17 different spacecraft buses manufactured by 8 aerospace companies (among which 2 European industries: Astrium with Flexbus and SSTL with SNAP, Microsat 70 and Minisat 400). Rapid 3 will be available within a year or so.

The US government, as the largest procurer of science mission spacecraft, recognized the benefits of standardized designs. Their vision is to streamline the procurement process and schedule by compiling a list of "space qualified" core spacecraft buses that could be purchased under a set of pre-specified contract terms. This approach has been tried previously by other agencies with mixed results. However, NASA Goddard's RSDO has been very successful with 7 spacecraft awards in the past 5 years. The RSDO provides services to all space-faring federal agencies and their affiliates, as well as university-based principal investigators. The catalogue allows virtually any government agency or entity (corporation, university, etc...) to procure a spacecraft utilising RSDO resources for a nominal fee to cover expenses.

In addition, the standard contract can be specifically tailored to meet the procuring organization's requirements. All catalogue spacecraft core bus entries must demonstrate flight qualification by showing direct heritage to a spacecraft that, at minimum, has been integrated to a launch vehicle.

The database is updated every 6 months to add new core buses with technology improvements or new options: Proteus for instance has just been added.

3 COMMON PLATFORM DESIGN

Using a generic bus for a mission also tackles several issues and brings noticeable change in the overall design approach. Compared with the classical way to design a mission (customized mission with dedicated platform), two processes will have to be taken into account:

- the development of the common bus itself, potentially designed for several missions and
- the adaptations of the platform to various missions

Designing a reusable platform requires to define its flight envelope and its potential applications domains. Obviously, since there are some uncertainties at the beginning of the bus development process about future mission profiles, the platform shall take into account some contingencies and then shall be flexible enough to accept modifications.

Therefore, the “ideal” platform would be a platform with the highest versatility, able to accommodate a wide range of missions, which implies a design as independent from the payload as possible. On the other hand, the payload is the single most significant driver of spacecraft design: payload operations and support are key requirements for the platform subsystems. This is one of the numerous issues that must be tackled. We will also notice that the concept of flexibility has its own limits in terms of design.

3.1 Design drivers

When analysing the design approach for common platform design it is important to understand what are the typical design drivers for a common platform. The objective of this section is to identify such design drivers and to analyse their impact on the overall common platform definition.

3.1.1 MASS

Causes of variation

The complexity and amount of equipments (i.e. payload and subsystems and the size of the spacecraft are one of the major sources of mass increase. A change in the propellant mass is also another reason for mass variation (e.g. as a consequence of different trajectories or mission profiles). More generally, a change in the mission profile usually implies a variation in mass.

Impacts on performances

The total spacecraft mass drives the choice of the launch vehicle which has its specific launch site(s) and its specific orbit envelope. For instance, for low Earth orbit applications, some launch vehicles do not have the capacity to reach very high or very low inclination. This characteristic has a direct impact on the performance of the mission.

More mass also means more inertia and therefore less agility: repointing speed is affected, it becomes more difficult to make the satellite rotate. On the other and more inertia means better stability: the spacecraft is less affected by perturbations (noise from the sensors, micro vibrations from the actuators, disturbing torques for instance, drag)

Mass is generally linked with size: the bigger the spacecraft is, the heavier it is. Pointing performance (knowledge and stability) can be affected by the size due to thermo-elastic deformations: for an earth observation spacecraft for instance, these deformations can affect the alignment and the focal distance. Thermo elastic deformations are critical for missions with high pointing requirements (space telescope for instance).

Potential flexibility of the driver and impact on the design

- *Structure:*

Since the structure usually represents from 10 to 20 % of the total spacecraft mass, an increase in the overall mass generally has an impact on the structure mass (both primary and secondary structure).

The role of the primary structure is to transfer the load to the launch vehicle interface and provide adequate stiffness to decouple the spacecraft's modes from those of the launch vehicle. By increasing the overall mass, modifications shall be performed such as reinforcements (addition of longerons) or thicker materials to ensure stiffness whilst fundamental frequency shall be within the requirements of the launch vehicle.

The secondary structure provides the mounting area for all equipment and thermal hardware giving protection against the launch and in orbit environment. If we want to develop a platform capable of accommodating variations in the number of units and change in unit mass, the secondary structure would need brackets (panels) with many fixation locations which can be adaptable to various mass of equipments and types of interfaces.

If the mass of the spacecraft will vary by a factor of more than 2, then a stack design may be conceivable. This is the case with the ESA highly modular Polar Platform with a unique service module, but with incremental Payload Modules (from 2 to 5)

An important aspect to take into account is the location of equipments. What ever the level of flexibility we desire for the platform, we shall keep the same general layout as much as possible because changing it would modify the mass properties of the spacecraft, the wiring network, harness, the electromagnetic compatibility/interferences requirements … and then would require additional verification for the adaptation to a new mission.

- *AOCS:*

Actuators:

More mass means more inertia so changing the attitude of the spacecraft requires highly capable actuators. Consequences are expected on:

- reaction wheels
- momentum wheels and
- control moment gyros

They shall have a bigger moment of inertia. The increase in operating speed for momentum wheels and control moment gyros is also a possibility. One thing we must not forget is that if we increase the size of the wheels or their rotation speed, the off-loading process may be more restrictive for the measures campaign (more frequent or longer off-loading events).

- magnetic torque rods (only for missions in low/medium Earth orbit missions) can be modified: either by the size (bigger moment created for the same current), either by the current

- thrusters performance shall be enhanced: a higher level of thrust is required to have a bigger moment. Actually, most of the thrusters used for attitude control are based on the liquid propulsion with which it is possible to control the level of thrust. On the other hand, more mass generally means bigger size: thrusters can be more efficient with a bigger moment arm, things being equal in other respects.

Actually, we have the choice either to select actuators capable of delivering variable moment either to change the actuators for each specific mission. In this case, the attitude control system shall be highly modular to withdraw versatile actuators.

- *Propulsion:*

For a same Δv required, if we consider a bigger spacecraft, either a more powerful thruster is needed, or a longer operating time is required. Indeed, assuming that the thrust is constant and the loss of total mass is negligible:

$\Delta v = \dfrac{F \cdot \Delta t}{M}$ where F is the thrust, Δt is the operating time and M the overall mass of the spacecraft.

So we need a type of propulsion with controllable performances (level of thrust and operating time), which is the case for the liquid and ionic propulsion. A heavier spacecraft also implies more propellant (things being equal in other respects) and so bigger tanks. One solution would consist in taking the upper value of mass the platform can support to choose the right size for the tanks even if it means that the tank are not filled for a smaller mission.

Finally, we must not forget the system which controls the propulsion system: it shall be reconfigurable to adapt to a wide range of missions (frequency and duration of the manoeuvres).

- *Launch vehicle compatibility:*

Compatibility with launch vehicle depends on the strategy adopted: either the generic platform is compatible with only one launcher, either with several launchers. The advantage of selecting several launch vehicles is to provide more launch opportunities and to allow to access to a more various types of orbit which is of importance for commercial applications. For instance, if a launcher can not access LEO with high inclination, manoeuvres will have to be performed by the spacecraft. Notice that this kind of manoeuvres uses a lot of propellant and therefore has an impact on the design of the overall system.

On the other hand, dealing with several launch organisations is more complex since different requirements have to be taken into account. The usable volume under fairing, the interfaces between the spacecraft and the launch vehicle, the environment conditions (mechanical, thermal and electromagnetic) and the management may be quite different.

3.1.2 POINTING REQUIREMENTS

Impact on performance

Pointing and measurement errors have a direct impact on mission performance. Pointing errors will directly affect the quality of the image (image blur is related with relative pointing error for instance).

Pointing drives the attitude and orbit control system and the choice of sensors and actuators in particular (and therefore affects cost). Sensors accuracy is a prerequisite to performance. Different types of sensors are available: Sun and Earth sensors (different accuracy), gyroscopes (different accuracy), star tracker (very high accuracy), actuators (magnetic torque rods, thrusters, cold gas, FEEP, reaction wheels, Momentum bias, control moment gyro)

Impact of high pointing requirements on the spacecraft design

In this part, we will focus on the impacts of high pointing requirement (which is often the case for science missions, especially observation missions) on the spacecraft design.

- *AOCS:*

Of course, high pointing requirements strongly affects the AOCS. The choice of the sensors and the actuators depends of the number and types of operating modes (coarse and fine pointing mode, slew mode, manoeuvre mode, cruise mode, safe mode).

The sensors have to be very accurate and as close to the payload as possible in order to reach a better attitude estimation, which directly impact the accuracy of the pointing.

Generally, reaction wheels are the most commonly used actuators for high pointing requirements because of the low noise produced (the spin can be stopped while observing). A precise characterisation of the static and dynamic imbalance shall be performed to avoid perturbations and sometimes damping systems are required (for both sensors and actuators). The unloading wheel management is to be considered: the time required to unload the reaction wheels and the frequency of the operations have to be discussed since it can affect the observing time. The capacity of the wheels has to be thought trough the quantitative assessment of the disturbing torques and is also a trade off between noise and unloading wheel management: highly capable reaction wheels has a high level of noise but they need to be unloaded less often, so more time for observation is available.

- *Orbit:*

The choice of the orbit has also a great impact on the capacity to meet high pointing requirements. Indeed, choosing a LEO or a L2 orbit has different consequences on the thermal control and secondly on the intensity and the type of disturbing forces. For instance, Hubble which is in LEO has to withstand eclipses, which prevents it from performing observations just after a shift from Sun light to Earth shadow or vice versa. The major perturbation to take into account for this orbit is drag. On the other hand, taking for example JWST in L2 orbit, the spacecraft always keeps its orbit within the Earth shadow in a very stable thermal environment, far enough from the Earth not to be disturbed by the albedo. Solar pressure is in this case the major contribution to disturbing torque. The design of the thermal control of these two extreme missions turns out to be radically different. Of course, the serviceability ensured to Hubble (replacing gyros or solar arrays by the Space Shuttle for instance) will not be possible for the JWST, hence the necessity to develop very reliable key components.

- *Structure:*

For more agility, a minimum moment of inertia is required. Therefore, heavy equipments shall be located as close to the centre of gravity as possible, as it is the case for Pleiades HR. On top of that, flexible modes shall be avoided by the use of material with a high stiffness or by dampers. The choice of the material shall also take into account the coefficient of moisture expansion (especially for composite materials) and the out gazing time constant which shall be minimize in order to have a stable platform very quickly.

- *Thermal control:*

Finally, very demanding pointing requirements have consequences on thermal control in terms of requirements on absolute temperature, and on temperature variations as a function of time and space. Notice that very precise probes (platinum probe used for Spot HRV) can be required for very restrictive thermal environments.

Potential flexibility

The issue is to manage to ensure flexibility for missions which have different pointing requirements. An advantage of these types of requirements is that it is possible to define common requirements for all the mission profiles selected for the platform, compatible with all the others: it will be obviously the most restrictive pointing requirements of the mission envelope. But if we decide to implement the most restrictive pointing requirements on the generic platform, missions which have lower pointing requirements will be supported by a platform which has extra performance, hence the idea to see if it would be possible to find out another way to meet different pointing requirements.

For instance, thermo elastic deformations in the interfaces between the platform and sensors or reaction wheel assembly increase PDE and PRE. A solution would consist in changing the type of material according to the mission requirements. Moreover, RPE can be due to fuel sloshing or more generally noise for instance. Thus, according to the mission need, it shall be possible to add dumpers if necessary. RPE is also linked with the performance of the control law. An adaptation of this law for each mission may be conceivable. Finally the frequency of the calibration drives the PDE, so by adjusting this frequency, it shall be possible to meet the requirements more tightly.

The trade off to consider is either we keep the same configuration so we have a platform with fixed pointing performance (rather high actually), either we try to adapt the platform to each mission (for instance foreseeing the adoption of different equipment)

	Fixed design	Flexible design
Advantages	- No delta engineering required	- More optimized design
Drawbacks	- Most restrictive requirements hence more complexity	- Delta engineering required

Table 8: How to cope with different pointing requirements

A balance has to be studied between the cost of the fixed but high pointing requirements on the one hand and the cost of the adaptations for each mission on the other hand.

3.1.3 POWER

In this part, we will only focus on the solar array technology with secondary batteries which is the most common option used.

Causes of variation

A change in the power budget can be due to a modification in the payload (higher power required or number) in the propulsion needs (in case of electric propulsion), in the thermal control philosophy (use of active control), in the items of the attitude control (actuators) and of course in the orbit (the distance from the Sun, the frequency and duration of the eclipses strongly affect the power system design)

Impacts on performance

Power may have an impact on performance through payload (actually a limited energy resource restricts the number of payloads and their operation time) and potentially through communications (data volume).

Potential flexibility of the driver and impacts on the design

- Solar arrays:

A platform adaptable to missions with different power budgets will require solar arrays able to deliver a certain range of power. This can be achievable with **various sizes of solar arrays** by adding panels for instance (The Polar Platform and Prima uses modular solar arrays, whereas the size of the solar array is fixed for Proteus, Myriade and Mita). Disturbing forces and torques may be affected by a change in the size of the solar array and shall be taken into account in the AOCS. We also must keep in mind that the power control unit shall be flexible enough to withdraw several configurations of solar arrays.

The **choice of the configuration option** for the solar array is also important: body-mounted, deployable but fixed solar arrays or deployable array equipped with a drive mechanism can be considered. Each of these solutions is a different answer to the utilisation of the surface of the solar arrays.

If we want to keep a configuration option, we shall change the size of the solar arrays to ensure different level of power. On the contrary, if we want to keep the same size of the solar arrays, we can imagine to consider the higher power budget to define the surface of the solar array with optimized configuration (with drive mechanism) and then to change the configuration for lower power budget. (Proteus and Prima propose the second option).

The two solutions have consequences on AOCS: changing the size of the solar arrays modifies the level of drag (for LEO satellites) and the effects of the solar radiation pressure whereas the second option brings disturbing moment while the mechanism is operating.

- Batteries:

Since the batteries are in charge of storing and delivering power, their role is critical in the power design. Modularity is still at stake to ensure flexibility: it is possible to adjust the number of batteries to withdraw the different levels of power budgets. An important problem to take into account is the increase of mass and volume.

A trade off to take account is the choice between parallel and individual charging. The former which is simpler and cheaper also degrades the batteries since the current is not controlled while charging. Additionally, flexibility in the vehicle integration is not possible with this type of charging. On the contrary individual charging which consists in charging batteries up to their own unique limits, is better for long missions since battery life time drives the mission life time. But using individual chargers adds impedance, requires more hardware and generate more heat.

- *Bus voltage:*

Very different voltages are required on a spacecraft (DC 5 V, DC 15 V, DC 270 V, AC 115 V rms) due to the different types of equipments. The distribution of the power is made through the bus power which makes the link between the power sources and the power "users". We can point out three types of voltage regulation:

- unregulated power subsystem for which the solar arrays are directly linked with the batteries, thus providing a variable voltage. This option is simple but it makes life more complicated for the power users to operate across a range of voltages. This architecture is generally well adapted to LEO mission with a short life time
- regulated power subsystem which provides a constant voltage (typically 28 V DC) thanks to an entire decoupling between the batteries and the solar arrays. It consists of adding a charge and discharge regulators linked to the batteries. This architecture brings flexibility in the battery design (choice of the capacity) and allows standardisation and compatibility with payload and components (just by adding some converters) but it is more complex to carry out (need for more electronic hardware).
- quasi regulated power subsystem which is unregulated only during the eclipses. This option is quite well fitted for geostationary applications

Another trade off to deal with is the distribution architecture: centralized or decentralized The decentralized approach consists in placing converters close to each equipment separately and generally is used with unregulated power bus. The advantage of the centralized approach is that it is not necessary to re-design the distribution architecture for each different applications (which can be interesting for a modular reusable platform).

- *Power regulation:*

Two types of power regulation can be considered:

- Peak power tracker which consists in using an array shunt regulator to switch sections of the array on or off (thermally speaking, it is better than a shunt dump regulator) in order to extract form the power source the exact power the spacecraft needs. A drawback of this option is to use from 4 to 7 % of the power. Missions with life time below 5 years and which require more power at BOL than at EOL are well fitted to this type of regulation.
- Direct energy transfer which consists in dissipating the extra power produced by the solar arrays with the use of shunt dump regulator. The advantage of this system is, compared with the PPT, the lower mass, the need for fewer parts and a high total efficiency at EOL.

3.1.4 THERMAL CONTROL

Cause of variation

Different type of orbits and attitude controls can have tremendous consequences on the thermal control. The type of mission also drives the thermal control: a deep space probe has radically different thermal requirements compared to mission in the vicinity of the Sun for instance.

Impacts on performance

Thermal control has direct impacts on the performance of the payload: detectors shall not support neither temperature gradients nor fast variation of temperature and their sensitivity is affected by absolute temperature. Moreover, dimensional stability shall be ensured (thermo-elastic deformations shall be controlled) and equipment require absolute temperature between certain limits.

Impacts on the design

In order to see the impact of thermal control on the spacecraft design, we will compare different types of missions with different thermal environment.

- Missions to the vicinity of the Sun (distance from the Earth < 1 AU)

The main issue with this type of mission is high temperatures. To cope with this, a sunshield may be required in order to protect the major part of the spacecraft (actually, a sunshield is also required for Gaia located in L2 in order to have very stable thermal environment). It is the case for Messenger and it will be the case for BepiColombo and Solar Orbiter. Furthermore, the spacecraft has to be always oriented to the Sun since the shield is useful for only one direction.

High temperatures have also consequences on solar cells efficiency: the hotter the cells are, the less efficient they are. There are two alternatives: either the arrays are tilted in order to reduce the received energy, either optical solar reflectors are placed between the solar cells in order to increase the reflection of the solar flux. The choice of the solar cells type is also important since they must be efficient even at high temperature. For instance, triple junction GaAs may be used for Solar Orbiter.

- Missions to outer planets (distance from the Earth > 1 AU)

The thermal environment is quite different from the previous mission profile one. The goal is to keep the heat from escaping which require much power. This is one of the main issues because of the distance from the Sun. But we must not forget that sometimes getting to deep space requires fly by or gravitational assistance manoeuvres at the vicinity of inner planets. For instance, Cassini Huygens mission required a fly by near Venus which implies extreme temperatures variations from hotter environment (Venus fly by) to colder environment (vicinity of Saturn).

- LEO missions

The main issue is the duration and the frequency of eclipses which drives the power storage and regulation design. Notice also that Earth albedo has to be taken into account in the thermal balance. A reusable LEO platform shall withstand different duration of eclipses and then be capable of support thermal cycles with different amplitudes and different frequency

- *Observatory missions*

A very stable thermal environment is required for this type of missions. Orbits around L2 provide an adequate environment: Earth eclipses are avoided (if the orbit is large enough) and the influence of the Earth and the Moon albedo and IR emissions are negligible. Generally, only one face of the spacecraft points towards the Sun and huge radiators are in the shadow.

Potential flexibility

We can first favour the heat conduction in order transfer the heat produced by the equipments or by the solar exposure to radiators, hence the importance of the choice of the material and size of the conductive paths. Heat pipes can be adapted to different thermal requirements thanks to their variable conductance.

On the contrary, we may want to insulate some parts of the spacecraft: Multi Layers Insulator (MLI) is the most common material with a good efficiency. Conductivity can be changed by changing the thickness of the blanket for instance.

Thermal balance can also be changed by the use of materials with different radiation properties (absorptivity, emissivity and reflectivity). Different paints, coatings with different surface conditions are available.

In order to get rid of the heat, radiators are used with the possibility to change their surface and therefore there evacuating capacity.

In order to produce some heat, electrical heaters can be adapted to a wide range of situations thanks to their diversity of shape (rectangular, square, circular or spiral) and the possibility to change the heat production by controlling the current delivered in it.

3.1.5 ORBIT

Impacts on performance

The orbit has a direct influence on the resolution (distance from the target) and performance may also be affected by the environment (cosmic rays, dust, Sun, Earth and Moon light).
Potential flexibility of the driver and impacts on the design

- *Thermal*

Orbit geometry and distance from the Sun affects drastically the thermal control system. To give an idea of the different power flux: the ratio between the Mercury orbit and the Saturn one is 500! Hence very different thermal designs (see section before).

- *TM and TC*

The choice of the orbit determines the distance and the visibility of the spacecraft from the ground stations. This is the reason why the size of the antennas is radically different from LEO spacecrafts to deep space missions (4m dish for Cassini). The delay time for communications between the spacecraft and the Earth also impact the autonomy of the spacecraft: in case of emergency, decisions and actions have to be taken early, therefore deep space spacecrafts require an autonomous break down detector and also need to be able to switch from one mode to another autonomously.

- *Power*

Different orbits imply different level of power available (if solar arrays are used) and different duration or/and frequency of eclipses which shall drive the choice of the voltage regulations and the type of the distribution. (see section 4.3)

- *Protection from the environment*

Depending on the orbit, the spacecraft shall have some protection from debris (for LEO) or protection from radiations if the orbit crosses the radiation belt, avoiding upset events in electronics parts.

- *AOCS*

The level of dynamic perturbations may change from several range of magnitude, depending on the orbit, as already mentioned before, hence a different attitude control approach.

- *Propulsion*

The orbit generally drives the Δv requirements and thus a propulsion module can be required in order to get into the orbit.

Orbit parameters impact heavily on S/C design and designing a platform for both LEO and deep space orbit is difficult. Too many subsystems depend critically on the orbit.
Indeed, the orbit is probably the main design driver for developing a common platform (the driver which has the biggest impact on the design).

3.1.6 AUTONOMY

Autonomy is the capacity of the spacecraft to operate on its own without the intervention of commands from the ground stations. It needs to consider the balance between ground stations work load, the link budget, and on board capacities (processor, mass memory…)

The definition of the level of on board autonomy is the result of a trade off dealing with the following questions:
1) May the data be mined, processed and archived on board or on the ground?
2) May the missions operations be planned adaptively and carried out autonomously?
3) May the manoeuvres be done autonomously?
4) Shall the housekeeping functions be analyzed and performed on board (wheels unloading, charge/discharge battery)?

5) Should the spacecraft detect faults autonomously?
6) Should the switch from a mode to another be commanded from the ground or on board?

Autonomy has a big impact on the OBDH architecture. Autonomy requires software capable of performing complex tasks and processors able to perform operations very quickly. The problem is to build a very reliable on board computer, hence the issue of validation. Since autonomous spacecrafts require complex software, the fully qualification is a tricky tasks since the number of configurations and cases tremendously increases compared with a spacecraft with low autonomy. On top of that, autonomous systems are more expensive (but maintenance cost is cheaper). On the other hand, less people are required in the ground stations which implies less labour work and less human errors.

A particular attention has to be paid when the level of autonomy is being defined: the net number and complexity of the tasks to be performed has to be checked carefully. The problem to deliver a quick diagnostic in case of alarm and the estimation of the duration in which the spacecraft can be left alone must be analyzed in terms of overall cost and design solutions.

Synthesis

Based on the discussion of the drivers, the following table gives general ideas about how to ensure a maximum flexibility for the platform.

Subsystems	Main parameters	Approach to flexibility
Structure	Strength, frequency	Increase strength or frequency Choice of the material
	Mechanical and thermo-elastic stability	Increase efficiency in stability management: identification of the parts where stability is required
	Launch vehicle I/F	Able to support different adapter diameter
	Configuration	Modularity Over-dimensioned S/C Large volume allocation for P/L
Power	Generation	Allow flexible SA size, shape and location Allow SA over sizing
	Storage	Allow for changing number and capacity of the batteries Allow for different type of charge and discharge management
	Distribution and regulation	Versatile PCDU (high number of power and heater lines) Standard I/F

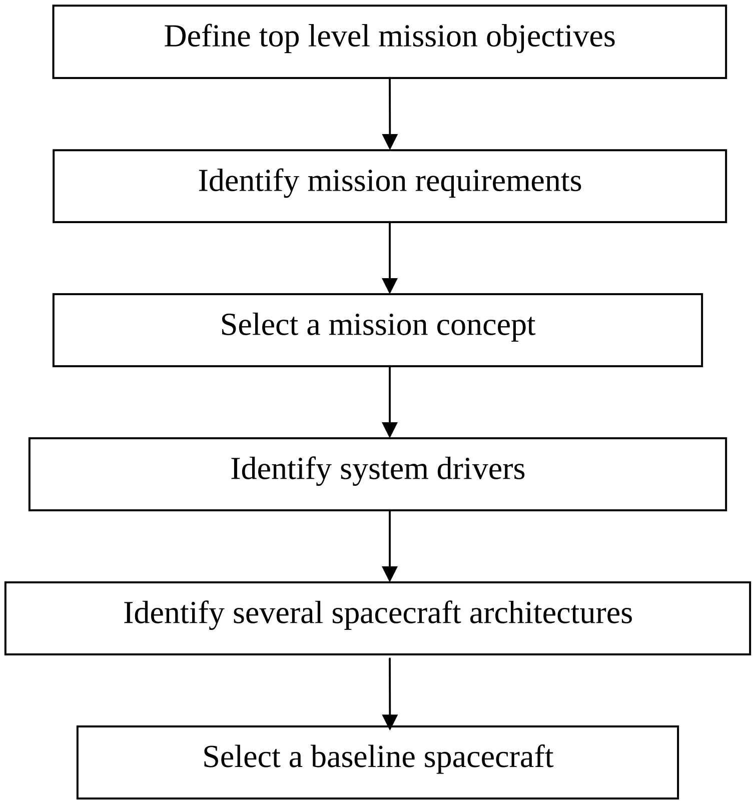
Subsystems	Main parameters	Approach to flexibility
Environment protection and control	Temperature	Decoupling Choice of MLI material and MLI thickness Variable size position of radiators, heaters Variable performance of heat pipes Mounting of equipment directly on radiators Allow for large heating power
	Radiation hardness	Able to support different radiation environment
AOCS	Sensors selection	Modular approach Allow different location Allow different sensor performance
	Actuators selection	Modular approach Allow different type of actuators
TM and TC	Frequency of communications	Compatibility with several ground stations Large on board processing capabilities
	Data rate	Allow different type and number of antenna Allow different type of modulation schemes Allow different data rate
OBDH	Architecture	Modularity Standard data bus Scalable mass memory size
	Processor	Processor able to implement open standards
	SW	Allow different AOCS modes, different FDIR functionalities

3.2 Design approach

The development of a common platform has to follow a different approach compared with the typical development of a spacecraft with a customized platform. This section will provide some ideas to understand the process.

Typical approach (top down) for designing a mission (first main steps before detailed definition) is described below:

The typical approach for designing a mission

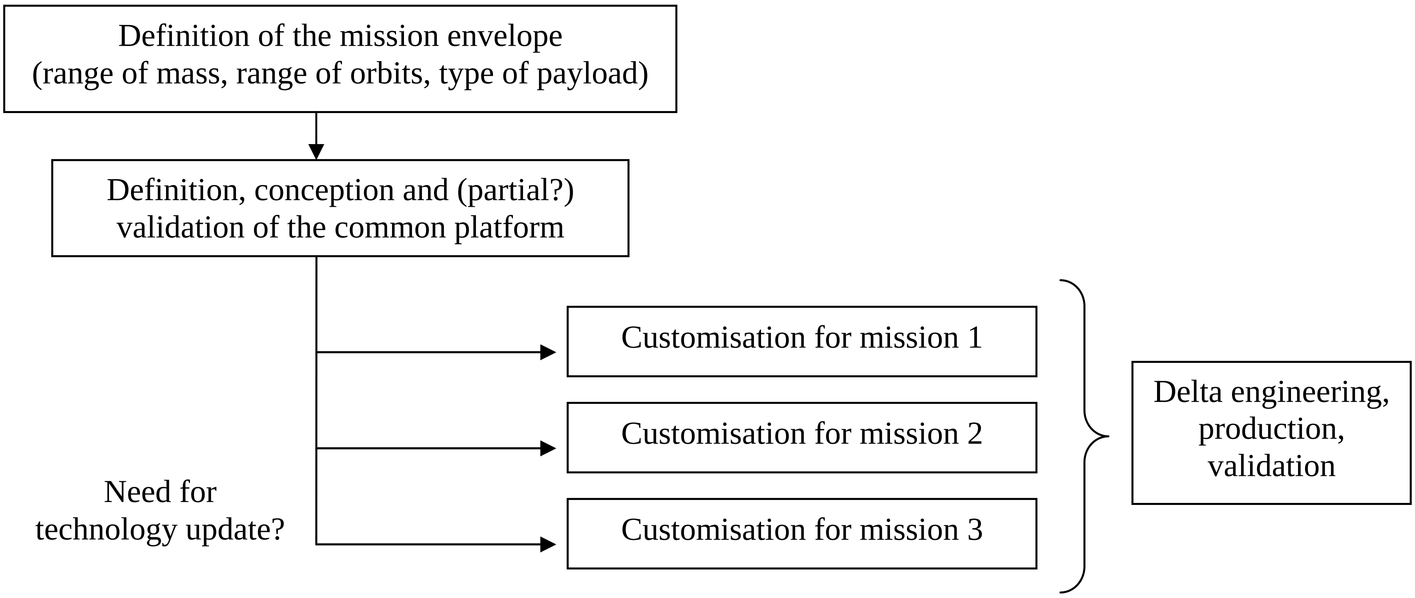


We can notice that the design of the platform and its subsystems comes later in the process and strongly depends on the preceding steps while this design activity is the primary goal in our study: designing a platform without knowing the mission objectives very clearly is the first difficulty which will influence the development of a recurring service module.

Flexibility is a critical parameter to determine and has a great impact on the design of the platform and on the adaptation to a mission both. The more flexible a bus is, the more configurations you have to study, assess, select and validate, on the other hand, the more flexible a bus is, the more missions the bus can be adapted to, hence the necessity to define a core architecture with nominal characteristics with a range of variation to define. What is often done with generic platforms is that a specific mission is the starting point of the process and defines the core design which is fully validated (e.g. Jason for Proteus, Demeter for Myriade).

Flexibility is also necessary to allow the platform to tolerate **technology updates**. Since the lifetime of the platform may be quite long (roughly 10-15 years), the technology used in some subsystems may become obsolete. During the conception phase of the platform, designers must be aware of cutting edge technologies to see at least what the major trends are and to take it into account as far as possible in the design for a later modification.

General overview of the consequences of using a common platform



A possible methodology

In this part, we will describe a possible methodology which could be applied for the design of a reusable platform.

Definition of the mission envelope

Mission envelope includes flight domain (orbit, environment), launch vehicle compatibility, resources (budgets) and mission objectives to be met.

In case a reusable platform is developed starting from a specific mission, extra mission objectives must be identified, keeping in mind that:

- They must be compatible each other (in terms of attitude control requirements or EMC requirements for instance)
- Contrary to the typical mission design, the mission objectives may change during the development phase: some will have to be removed, other modified after a feasibility study for instance.

Identification of the system drivers

Once we have an idea of what and how well the platform shall accomplish, we must identify the system drivers that have the biggest impact on the spacecraft design, cost and performance. The problem is that it is sometimes difficult to assess the impact of a driver. In order to do that properly, the following steps must be followed:

- identify the area of interest (cost, performance, risk, schedule)
- identify numerical parameters which measures the impact (e.g. figures of merit)
- analyse the impact of a modification in the driver (for flexibility assessment)

Identify several spacecraft alternatives

Several spacecraft options must be identified, each of these being a different answer to trades off:

- What is the part of space/ground processing?
- What is the level of flexibility?
- What is the level of autonomy?
- What is the level of risk? (well proven technologies or state of the art)
- Is it a central or distributed control? ...

For each alternative, a set of budgets has to be made (mass, power, TM/TC link…) in order to be able to assess all the options (see if the objectives and the requirements are met with the constraints).

Select a baseline architecture

The result of the assessment of the different alternative is the choice of a baseline architecture for which the interfaces will be clearly defined.

Many steps of the development follow the typical development phases except the first step: mission objectives come naturally for a typical mission whereas the choice of objectives to be met by the generic platform is quite challenging.

4 PROGRAMMATIC CONSIDERATIONS

Beyond technical aspects, platform reuses lead to different ways of development phases planning and in the risks management approach for instance. This chapter provides preliminary qualitative considerations concerning programmatic aspects in relation with the platform reuse concept.

4.1 Development process

If it is decided for a mission to reuse an already existing platform, some changes in the development phases are expected. We outline below a typical definition and development process.

4.1.1 DEFINITION PHASES

Phase A

The P/L definition is likely to be influenced by a SVM reuse. Indeed, since a standard design (with possible modification) is available, I/F as well as resources are well known very early in the development process. This allows to focus more on P/L to meet scientific objectives rather than SVM design. As a consequence, a special effort has to be put on the P/L definition in terms of I/F and resources budgets in accordance with the use of an existing platform. It is also expected to have a shorter P/L definition since I/F and the resources with the SVM design are already defined (although some modifications may have to be performed).

Phase B

New activities:

The compatibility of the platform to the mission shall be assessed, with a time and cost estimation of the modifications to be carried out.

For instance, in the standard Proteus' schedule a pre phase B is planned to confirm that the mission is compatible with the platform specifications. Then, the platform equipment is ordered during phase B. During this phase:

- A preliminary concept of the satellite is identified
- An identification of points out of PROTEUS platform specifications,
- An identification of critical points in the modifications of the platform

are performed.

Activities not to be performed:

Once again the definition of the SVM does not need to be done avoiding a long iterative process coupled with the P/L definition with corresponding benefits in terms of lower risk, shorter times and lower cost.

Finally, we can expect:

- Minimum engineering effort on SVM, only modifications to consider
- Focus on PLM rather than SVM
- Schedule separation between PLM and SVM
- Quicker process for PLM (only one module to be developed ex nihilo, the other already exists)

Seemingly, phase B should be faster (compared with a dedicated SVM design) since time consuming phases are avoided (analysis and assessment of SVM concepts and trades off).

Phase C

At SVM level, only mission specific activities (adaptations) have to be performed during these phases:

- platform harness definition,
- system data base update,
- flight software updates,
- OBSW modifications,
- satellite simulator upgrade

These modifications are to be implemented as necessary for the adaptation of the core spacecraft, options to meet the mission specific delivery order requirements. The extent of the modifications may include the addition of performance parameters, changes to any performance parameter, changes to AIT plans, the addition of new specifications, requirements, analyses, tests, reports, hardware, software or support; adaptation of baseline hardware or software configurations; changes to baseline schedules, reviews, funding profiles, and milestones.

Finally, as it is the case of Proteus, procurement of platform LLI can start as soon as the beginning of phase C, while non LLI are procured during phase C.

4.1.2 AIT PHASES

Generally speaking, SVM AIT is expected to be much shorter (e.g. 3 months AIT for Proteus) thanks to the use of a common platform, the verification effort shall be reduced since no full qualification test programme is necessary and the verification and qualification at equipment, subsystem and module levels can start earlier allowing early resolution of potential problems. For instance, some critical activities for Venus Express (structure manufacturing) started a few months before the program was officially approved, and was continued in parallel with design activities. This risky approach allowed to save time and to start very early the structural integration of the PFM. Moreover, AIT/V activities are already known and their duration and criticality well established.

The modularity configuration (SVM + PLM) with a separate design and development also allows a flexible integration and verification approach (SVM qualified in parallel with PLM). Moreover if a generic SVM is used, tests are simpler and HW needs are minimized since verification can be performed more by similarity and analysis (especially on the structural elements) and the system control tests can be performed on in open loop configuration as it is the case for Myriade. When tests cannot be avoided, it is nevertheless possible to use test results on one S/C model for another. For instance, acoustic testing on Herschel STM provided information regarding acoustic environment response of Planck SVM.

Generally, the most important heritages are at unit level but the layout of these units necessitates adaptations that need to be qualified for the new configuration. However, the unit qualification may be considered as already achieved.

Another important point to take into account is engineering data base from the verification process: every mission reusing the same platform can benefit from these data thanks to correlation between the successive test operations. On the other hand, to be fully effective, it requires for each mission to update the engineering data for the next missions. The same applies for mathematical (mechanical and thermal) models which can take advantage of SVM reuse by being upgraded after test campaigns.

The **verification approach** will be also different from the typical way to design a spacecraft. One of the issues will be to know how the verification will be carried out (which philosophy to choose? How many models to be manufactured?) and how it will be shared between the conception of the platform and the adaptation to the missions. This will also depend on the level of flexibility: a highly flexible platform will require much more extra qualification than a less flexible one.

4.1.3 MODEL PHILOSOPHY

Using a common platform leads to changes in the model philosophy in terms of number and type of models since the platform is already qualified for some configurations. The goal is to minimize hardware need whilst keeping a level of risk acceptable.

Units that are part of the platform are usually already space qualified and a direct flight model (FM) can be manufactured. As far as module level is considered, the service module generally follows a protoflight approach with a Structural and Thermal Model (STM) and a ProtoFlight Model (PFM) which is the flight model. The STM is required only if there is a change in the structural and thermal requirements (e.g. change of launcher, different pointing configuration, different orbit). Therefore, there is no need for manufacturing and testing an Engineering Model (EM) and a Qualification Model (QM) before the Flight Model (FM)

On the contrary, the Payload Module (PLM) needs to follow a full qualification process, since there are rarely some heritage cases for payload.

Finally, at spacecraft level, the protoflight approach is generally chosen, with only one entire spacecraft manufactured.

4.1.4 HARDWARE REUSE

As described previously, minimizing hardware need can be ensured first by adopting an adequate model philosophy or by reusing hardware from a mission to another. For instance, it is possible to reuse some models, especially the avionic model which shall be reconfigurable from a mission to another. The MGSE/EGSE may also be reused and need to be interchangeable to adapt to mission specificities. Finally, spare models of a platform can be usefully not only for one mission but also for several missions adapted on the same platform.

4.2 Project team

Using a common platform is all the more efficient in the avoiding of recurring effort that people working on it remain the same. Nearly every case of platform reuse mentions the idea to keep the same personnel (in agencies as well as industries) from a mission to another in order to transmit knowledge, know how, and experience of the platform technology and of its adaptation to the missions.

If a common platform has to be developed, we can expect a reduction in the project team size. For instance, we can also assume that the number of people involved in the design phases (such as subsystem experts) would decrease.

An innovative development method is applied for Myriade at Astrium based on a small team of multi disciplinary engineers, with fast communication channels and close links between the AIT and design teams, while minimizing the procurement efforts. There is also a dedicated team in Alcatel for Proteus who works on each platform adaptations and the CNES itself has also has a management structure consistent with the "products" with dedicated project managers.

For Mars and Venus Express missions, a strong heritage in team management has been identified in each organisation implied: in the industry, many people worked for Rosetta, Mars and Venus Express missions. In particular, AIT/V operations for Mars and Venus Express were successfully performed by the same team. In ESTEC, people who had worked for Mars Express brought their experience to Venus Express mission as well.

Another example is Herschel and Planck program: in order to take benefit from SVM AIT (lessons learnt) for Satellite AIT and thus to increase efficiency, it was proposed to carry out both SVM and satellite AIT with mixed teams from SVM Contractor and from respective Herschel and Planck AIT Contractors.

A limitation of this approach occurs when there is a gap between two missions using similar platform, people may not more be available to work on the following project, hence the risk to lose development and AIT expertise.

A way to avoid this problem is to put a special attention to documentation to allow the transmission of the platform system knowledge. As example, the user manual of a platform allows to the principal investigator to know whether his payload is compatible with the platform in terms of resources, interfaces, and environment.

Finally, on top of getting a better level of documentation within an organization, it is also important to facilitate the documentation exchange between the different organizations. The typical example is the XMM/Integral common service module with the selections of two different prime contractors for both missions which required to exchange data from one industry to another.

4.3 Cost issues

Cost reduction is one of the main objectives expected to be met for reusable platforms. Actually, we have several examples of considerable cost savings when missions share the same service module design or more generally share some similarities in the mission profile.

NB: all the figures mentioned in this section come from: <u>Standardized mission operational methodologies</u> from D. Andrews and E. Soerensen (ESA/ESOC)

- ERS, MSG, and Metop series:

For these missions, the development costs are divided by 10 approximately and the AIT costs by 2 due to nearly identical platform design (just a few upgrades from a mission to another), and due to a very similar payload within series. The reference cost is the first mission for each case.

- Rosetta - Mars Express - Venus Express and Cryosat - GOCE – Aeolus:

These are two families of missions which have quite similar missions operations that allows some commonalities in the avionics. A minimum division by 2 or 3 for the costs of mission control, flight dynamics or spacecraft simulator is reported for this type of missions' family.
The same applies for Smart-1 for instance, 6 % of a total of 2462 software modules for the mission control system are new (heritage from Rosetta, Integral, MSG-1). Therefore, the total development effort is 4 to 5 times cheaper.

- Similarly, considering Herschel/Planck programme as the reference (see study case), Eddington would have benefited from a division by 2 or 3 for the costs of the design, the development and AIT of the mission control, flight dynamics or spacecraft simulator, partly due to the commonalities in the service module.

More generally, thanks to commonalities in the platform design, it is expected to reduce cost thanks to:

- little hardware equipment (less models, common spare for several S/C)
- batch procurement of units allow to reduce considerably the price per unit
- the reduction of technology development cost (at SVM side)
- the implementation of schedule compression
- the reduction of project team size
- the reduction of mission operation team size
- the reduction of mission operation team size (in case of missions with similar operation)
- the reuse of procedures
- the reuse of S/C simulator equipment

For instance, Alcatel forecasts to divide by two the price of Proteus platform for Globalstar constellation compared with its current price thanks to savings on scale.

It is also expected to have a much more precise cost assessment, and then a lower level of contingency.

On the other hand, to ensure cost effectiveness, there must be a minimum number of missions.

4.4 Risk management

Examples of technology upgrades: replacement of obsolete parts and advances in technology or production practices

Risk management is defined as "a systematic approach to support the program management and the optimization resources with the purpose to:

- identify,
- assess,
- reduce,

- prioritize,
- control,
- document and
- communicate the risks involved in a program with reference to cost, schedule and performance".

Considering this definition, we will analyse the consequences of platform reuse.

4.4.1 REDUCED RISKS

Using a common platform allows better identification, assessment and control of the risks, indeed:
- Design is well known and I/F are predefined
- There is a higher confidence in the performance analysis, including resource budgets, hence smaller margins (especially contingency margins)
- Platform testing has already shown pro's and con's of the design
- There is no need for technology development on SVM side (so no potential delays)
- Experience from AIT/V is capitalized: decreasing number of problems during integration
- SW maintenance is well known
- In case of modular configuration for the spacecraft, SVM integration is separated from PLM: any problem on one does not strongly interact with the other
- Technical risk is mitigated
- Work load is better known, delays are less likely and planning is easier
- Same test facilities are used (if possible)

As a consequence, we know where the risks are and what the level of risk is.

4.4.2 INCREASED RISKS

However, there are still some issues to consider:
- technology obsolescence (impact on the platform lifetime)
- procurement problem (LLI procurement, long period without mission)
- loss of expertise if no regular mission
- no SVM technology evolution
- biggest impact in case of failure: for instance if a family of missions has to be launched in a very short period of time, there is the risk (for the first missions) of not having the possibility to react in case of a unit failure for instance.

These issues can be coped by adopting specific counter measures:

- Technology obsolescence and SVM technology evolution:
The common platform needs to have evolution, updates in its design on a regular basis. For instance, it is planned for Proteus to perform some modification after the first 5 missions.

- Procurement problem:
Generally, long term agreements are signed with the equipment suppliers (to guaranty the durability of the product line) and strategic stocks of elementary parts were made to avoid being obsolescence-sensitive. Strategic stocks of some obsolescent EEE (Electrical, Electronic and Electromechanical) parts have been made to secure the platform availability in the next years. Moreover, batch procurement to each supplier can help reducing costs and providing equipment to all the partners involved in the contract (CNES, Astrium and Alcatel for Myriade, ASI and Alenia for PRIMA…)

- Loss of expertise
Need to document the PF (which is the case for Proteus) and to make upgrade in the test tools.

4.5 Industrial policy issues

4.5.1 LAUNCH CONSTRAINTS

The outcome of ESA's Council of Ministers of the European Space Agency in December 2005 in Berlin results in the advice to use European vehicles (Ariane 5, Vega, and Soyuz once it begins operations from Kourou) for the launch of its spacecrafts, with Rockot as a back-up, hence the necessity to develop platform compatible at least with these launchers. CNES for instance is studying new launch configurations with Soyuz and Vega for Proteus and Myriade platforms. Beyond 2008, it is probable that there will be no more launch with ASAP 5 thus deleting launch opportunity for Myriade (Spirale program with 2 S/C in GTO will likely be the last launch with ASAP 5). On the other hand, Vega will be ready or nearly ready by that time.

It is important to notice that there are several standard adapter diameters available on Ariane 5, Soyuz and Vega, one of them being common to all these launchers: the 937 mm, hence the possibility to develop a platform with a standard launch I/F compatible with European launchers without modification.

4.5.2 GEOGRAPHICAL RETURN

Geographical return plays an important role within the ESA procurement approach. The aim of geographical return is:

- To ensure that all Member States participate in an equitable manner, having regard to their financial contribution, in implementing the European space programme and in the associated development of space technology; in particular the Agency shall for the execution of the programmes grant preference to the fullest extent possible to industry in all Member States, which shall be given the maximum opportunity to participate in the work of technological interest undertaken for the Agency
- To ensure fairness of competition at all levels and in particular between Prime Contractors and subcontractors (one of the top priorities to be achieved by the Agency)
- To ensure a fair allocation of activities among industrial firms, in particular among Prime Contractors and equipment suppliers

Considering these constraints, geographical return can be a burden for generic platforms since major portions of work would be performed at a given location. For instance, all the European generic platforms units are provided by fixed suppliers from different countries. If the suppliers had to change, to meet geographical return requirements, the design of the units may change and modifications may have to be carried out. As a consequence, the concept of a common platform would be less efficient or even meaningless.

Moreover, if a common platform was developed by ESA, the problem is to select the prime contractor and ensuring a fair return for the European space industries.

4.6 Schedule

Schedule and frequency of flight opportunities are also important factors to be considered. In order to take full advantage from the platform reuse, both ESA and industry would need to have confidence on a number of missions within a given period of time. That's why the concept of reusability rather applies for small missions rather than long programmes (telecommunications excluded). The problem is that long programmes need long lead-times to obtain approvals: it is a big problem for the industry with the interruption of development and manufacturing processes. This is an additional risk to take into account.

4.7 An optimized scenario for platform reuse

After having raised the different programmatic issues, it seems interesting to provide considerations for an optimum efficient of platform reuse.

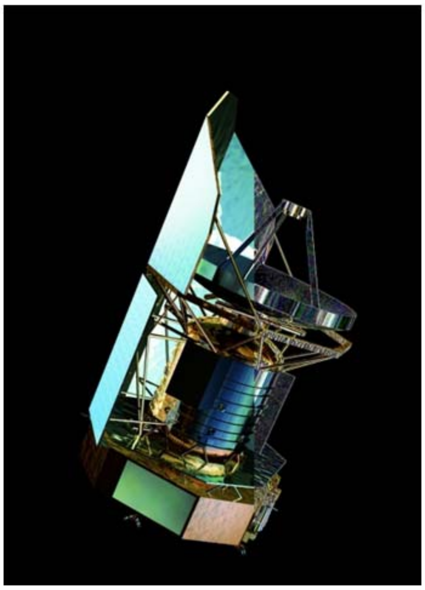
Considering the perspective for the life time of a platform (10 to 12 years for Proteus/Myriade), we can easily deduce from this that long term planning is necessary. Furthermore, reuse of the platform itself is not sufficient: this strategy shall be considered at a higher level to be fully efficient.

M/EGSE	Same as far as possible or minimum modification Quick reconfiguration
Ground station	Platform compatible with a fixed number of ground stations
Launch compatibility	Define compatibility before the design of the platform
Project management	Keep the same personnel from one project to another
Relationship with suppliers and SME	Sign long term contracts to ensure the supply and an interesting price for hardware
Versatility of the service module	Define a dedicated platform for a specific type of science mission
ESA policy	Long term planning for science missions

Table 9: Some ideas for an optimized scenario

PART 2:

STUDY CASES



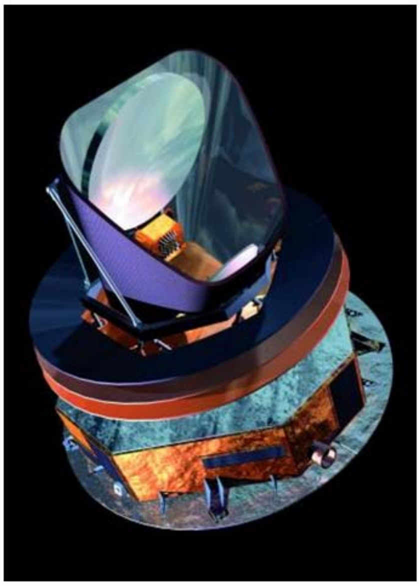


Figure 7: Herschel and Planck spacecrafts

This second part will consider Herschel/Planck programme as a study case since a common service module design has been planned for these two missions. From the comparison of the requirements analysis, we will see how it has been possible to have a common service module design and how programmatic aspects were treated. Finally, proposals to reuse Herschel/Planck SVM for Gaia and Eddington will be discussed as well. This will illustrate the first part showing how a common platform design could have been ensured and what the consequences were in terms of programmatic aspects.

1 HERSCHEL/PLANCK SERVICE MODULE DESIGN

The data presented in this chapter is based on Herschel/Planck design report issued in July 2004 and on the Design and Development plan issued in November 2004 by Alcatel Alenia Space.

1.1 Missions' overview

1.1.1 PLANCK

Planck is the Third Medium Sized Mission of ESA's Horizon 2000 Scientific Program. It is a survey mission which consists in measuring the anisotropies and polarization of the Cosmic Background Radiation Field over the whole sky over a wide frequency range (9 frequency bands), with unprecedented sensitivity and angular resolution. It is the third mission (after NASA COBE and MAP missions) to map the sky at submm wavelengths.

1.1.2 HERSCHEL

Herschel (formerly called FIRST: far infrared sub millimetre telescope) is the 4^{th} cornerstone of ESA's Horizon 2000 Scientific Program. It is an observatory mission to well identified targets in the far infrared and sub-millimetre part of the electromagnetic spectrum (wavelength range from 60 to 670 μm).

1.2 Requirements comparison

Even if the mission objectives are quite different, Herschel and Planck have similar mission profiles. For each of them, the payload consists of a telescope with instruments to be cooled in the focal plane.

1.2.1 LAUNCH

The launch for P/H will be a dual launch on A5 ECA. Sharing the same launch vehicle implies the same launch environment: mechanical (steady state acceleration, random and acoustic vibrations, shocks), thermal and electromagnetic requirements are similar for both S/C.

We will see in the section 6.4 that launching two spacecrafts at the same time has several programmatic consequences (procurement, risk management, verification philosophy).

1.2.2 ORBIT

A5 upper stage will inject Herschel and then Planck into separate transfer trajectories to reach the same orbit site (Lissajou orbit around Earth Sun L2) which implies that they will have the same on orbit environment in terms of radiations.

	Planck	**Herschel**
Transfer trajectory	Similar	
Injection	Specific manoeuvre	Free
Operational trajectory	Small Lissajou	Large Lissajou

Table 10: Orbit comparison

Herschel will perform science operations in a large Lissajou orbit (1.5 10^6 km diameter) whereas Planck will be placed in a small Lissajou orbit (350 000 km diameter).

1.2.3 LIFETIME

	Planck	**Herschel**
Nominal lifetime	21 months	3.5 years
Including degradable items	2.5 years	4.5 years

Table 11: P/H lifetime (including from 4 to 6 months transfer to L2)

1.2.4 ΔV REQUIREMENT

Manoeuvres	**Planck**	**Herschel**
Orbit injection and eclipse avoidance	195	0
TOTAL	265	73.7

Table 12: ΔV requirements (m/s) for P/H

Unlike Herschel, Planck requires an amplitude reduction manoeuvre at the end of the transfer phase to inject the S/C in a small Lissajou orbit, which makes a big difference in terms of required propellant mass.

1.2.5 COMMUNICATION WITH THE EARTH

Planck and Herschel shall communicate with Earth only 3 h per day maximum (out of maximum 8 h visibility), they have the same ground stations: New Norcia (nominal) and Kourou (back up). It is noteworthy to point out the fact that Planck and Herschel use the very close frequencies (in the X band) and also have the same data rate requirements.

	New Norcia	Kourou
Uplink bit rate	4 kbps	125 bps
		4 kbps
Downlink bit rate	5 kbps	500 bps
	150 kbps	150 kbps
	1.5 Mbps	

Table 13: TM/TC data rate common requirements

These similar requirements, on top of the fact that the Earth to spacecraft distances of the two S/C during operation (1.8 106 km for Herschel and 1.6 106 km for Planck) are comparable, leads to similar TM/TC architecture and hardware units.

1.2.6 AUTONOMY

P/H both have the same autonomy requirements, which will influence the FDIR functionality and OBC capabilities (size of the MM, for instance).

	Autonomy time
Nominal mode	48 h
Survival mode	7 days

Table 14: Common autonomy requirements

48 h autonomy in nominal mode corresponds to the worst case when one daily ground contact is lost.

1.2.7 AOCS CONFIGURATION, POINTING REQUIREMENTS

Before going into the details, let’s define a reference frame linked to the S/C:

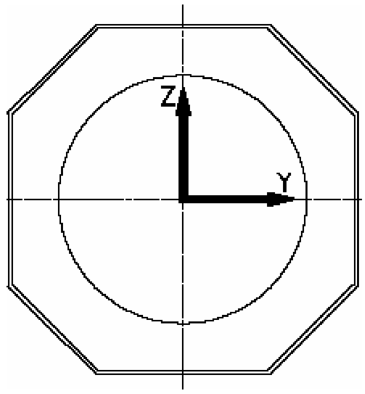


Figure 8: Upper view of the SVM (PLM would be in front)

The X axis is such that (X, Y, Z) defines a direct reference frame.

Since Planck is a survey mission, the S/C is spin stabilised at 1 rpm and normally operates with its spin axis pointing directly away from the Sun. The line-of-sight of the telescope is positioned at an angle of 85 ° to the spin axis. In order to view the celestial poles, the spin axis can be periodically moved up to 10 ° away from the anti-Sun direction and out of the ecliptic plane.

As Planck orbits L2, it makes one rotation about the Sun per year and thus theoretically allows two full sky coverages. The spacecraft spin axis has to be rotated at the same rate in order to remain Sun pointed, which is achieved by making regular maneuvres.

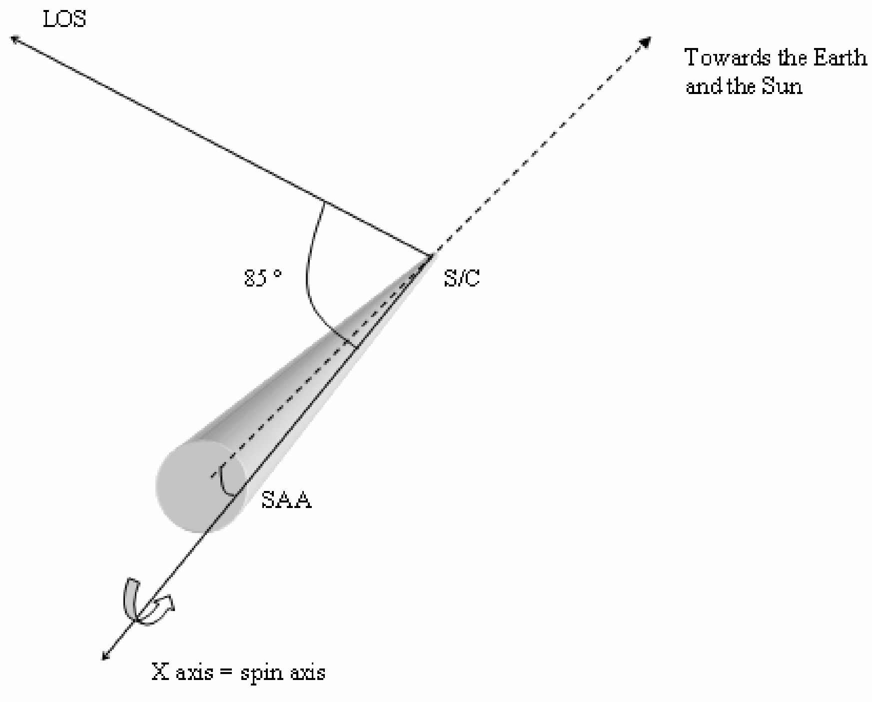


Figure 9: Planck pointing configuration

As a three-axis-stabilised observatory, Herschel follows the tradition of many recent astronomical space missions and presents no new problems in the control of the dynamics of the spacecraft. However, the pointing requirements are much less stringent for Planck than for Herschel (a difference of more than two orders of magnitude), as showed below:

	Planck	Herschel
AOCS configuration	**Spin stabilized (1 rpm)**	**3-axis stabilized**
Angle between LOS and +X axis	85 °	0 °
SAA	10° half-cone centred on –X	+/-30° about Y, +/- 1° about X
Type of pointing	Scanning	Inertial or Solar System Object referenced pointing, rasters, scanning
LOS attitude reconstruction	On the ground	On the ground
APE of LOS (short term)	**37'**	**3.7"**
RPE of LOS	**1.5'**	**0.3"**
AME of LOS	**0.5'**	**3.1"**
Mission specific	X axis = axis of major inertia = spin axis Rotation rate stability Need for nutation dumping	Slew requirement (41 sec for 8')

Table 15: P/H pointing requirements

Major differences, highlighted in red, will affect the choice of some sensors, actuators, but will also add structural requirements (need to know the mass distribution with a good level of accuracy for spin stabilisation) and modify the AOCS architecture (type of AOCS modes, control algorithms…).

1.2.8 STRUCTURAL REQUIREMENTS

P/H share the same structural concept: the SVM primary structure is an octagonal box with a central cone providing the launcher interface. The octagonal box contains upper and lower closure panels, lateral panel (supporting equipments), and shear panels.

The structure has to support all the SVM subsystem units, as well as the warm units related to P/L. The following table shows the PLM and SVM mass budget:

	Planck	Herschel
PLM mass	**400**	**2400**
ACMS	**30**	**72**
CDMS	14	14
Harness	84	75
PCS	36	36
RCS	78	56
SA	45	
Structure	387	314
TCS	**53**	**23**
TT&C	24	23
Total SVM dry mass (separation system and system margin excluded)	751	615

Table 16: P/H mass specification (kg)

CDMS, PCS, TT&C and harness mass are remarkably very similar for both missions.

Notice that the solar arrays are part of the PLM for Herschel and thus are not included in the SVM budget; and that TCS mass takes into account the Sorption Cooler System (SCS) for Planck but does not take into account the cryostat, which is part of the Herschel PLM.

The structural design takes into account the need for Planck to know the mass distribution very accurately for spin axis control: the X axis (= spin axis) must correspond to the axis of maximum inertia and moreover, the tensor of inertia shall be as diagonal as possible, thus avoiding nutation movement.

1.2.9 THERMAL CONTROL

Both S/C require:
- cryogenic temperatures for the instruments in order to have a very good sensitivity
- high level of thermal stability, despite the fact that the warm units have different level of dissipation according to the mode in which they operate (SAA is also a critical parameter)
- very low heat flux to PLM

While the temperature requirements of the SVM equipment are the typical ones for scientific satellite (room temperatures), the thermal requirements of the PLM units are much more stringent (below 20 K).

	Planck	**Herschel**
Instruments in the focal plane	from 0.1 K to 20 K	from 0.3 K to 2 K
SVM units (operating range)	Approx. from 250 K to 320 K	Approx. from 250 K to 320 K
SVM thermal stability *	1 K per hour	1 K per hour

Table 17: P/H thermal requirements

* In addition, specifically stringent stability requirements apply to HIFI units (Herschel) and SCC (Planck).

1.2.10 POWER REQUIREMENTS

The power available from SA exceeding the system demand shall be left in Solar Array. The common P/H battery is designed to provide electrical power during launch, during partial moon shadowing, in the event of an attitude loss in support of bus transient and to supply peaks if necessary.

	Planck		**Herschel**	
	Science	Telecom	Science	Telecom
SVM (including losses)	370	430	456	496
PLM	**1000**	**1000**	**550**	**550**
SA power	1913	1913	1480	1480

Table 18: P/H power budget at EOL (W)

While the Herschel instruments require 550W from the spacecraft, the Planck instruments and the Sorption Cooler require 1000 W. For Herschel it is required that the SVM has a power demand limited to 520 W.

The different power requirements leads to completely different solar arrays design.

1.3 Implementation: a common service module design

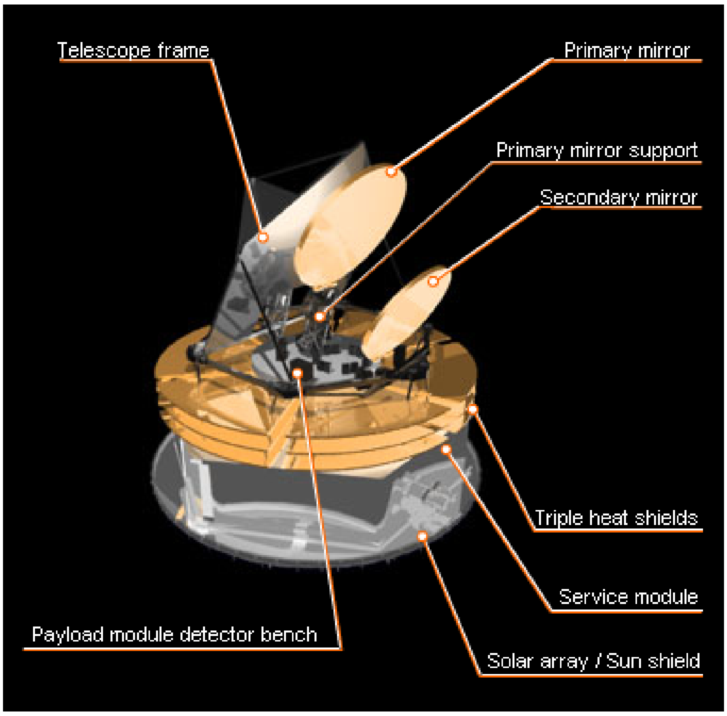


Figure 10: General layout for Planck (w/o the telescope baffle)

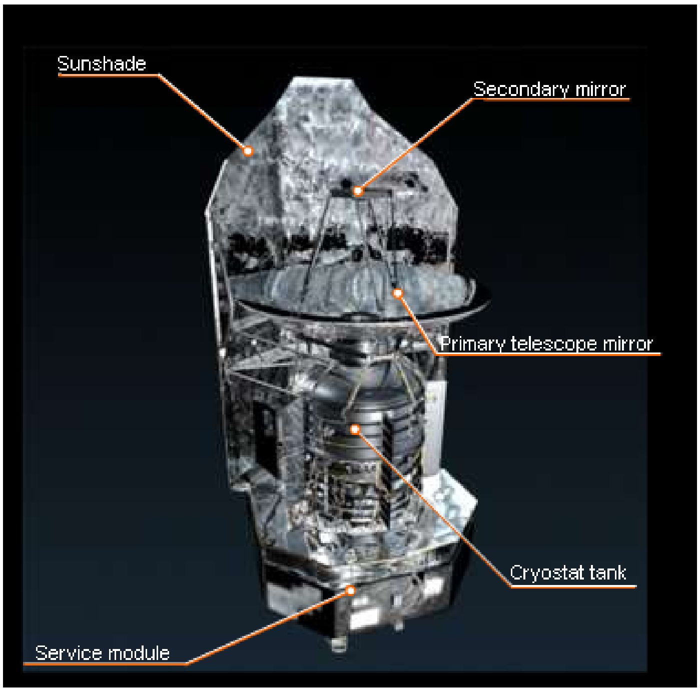


Figure 11: General layout for Herschel

1.3.1 GENERAL ARCHITECTURE

P/H share the same philosophy concerning the general configuration which is modular with a clear distinction between the PLM and the SVM. PLM contains the telescope, the focal plane assembly with the instruments whereas SVM contains all the subsystems necessary to support the P/L and also the Warm Units (WU) of the instruments which requires room temperature. Notice that for Herschel, SA, which are mounted on the sunshield, are part of the PLM.

1.3.2 STRUCTURE

General disposition:

The primary structure is made of:
- a central tube
- an octagonal box (with 8 lateral (equipment) panels, 2 closures panels and 8 radial (shear) panels)

- a subplatform
- Platform Tanks Support Structures (PTTS)

The primary structure is globally the same for both missions in terms of functions, material, shape and configuration, whereas the secondary structure is quite different since related to mission specificities (STR support for H, He tanks for P for instance).

The equipment panels accommodate the units (subsystem units and some WU) and provide them large dissipative surfaces. As far as subsystem units are concerned, dedicated panels support the different units: there is one panel for RF functions and another for CDMU and PCDU for both S/C.

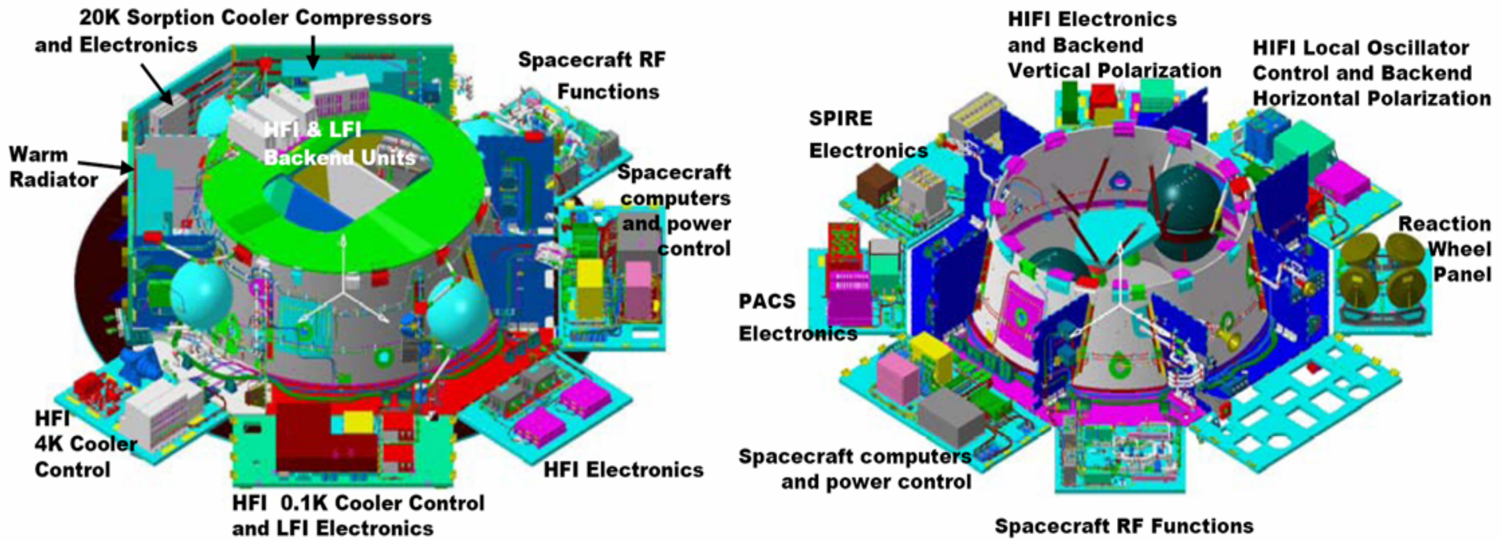


Figure 12: Planck and Herschel general lay out

		Similarities	Differences
Primary structure	Lower and upper closure panels	Shape, number material	Different cut out Upper: no support no I/F for P
	Subplatform	Material Support electrical and mechanical I/F	Different cut out Support 3 WU for P Reinforcements for P
	PTTS	Same supporting concept	3 tanks for P 2 tanks for H
	Equipment panels	Number (8), material	Different inserts location and size for P/L WU 3 SCC dedicated panels for P
	Shear panels	Number (8), shape, material	Different configuration (for 2 out of 8 panels)
Secondary structure (supports to I/F)		Material	Location, number

Table 19: Comparison of P/H structural design

1.3.3 AOCS

Type of stabilization:

Planck is spin stabilized at 1 rpm without closed loop attitude control. However it needs accurate sensors and sophisticated algorithms to manage the onboard autonomous inertial attitude determination and accurate positioning of the spin axis.

Herschel is 3-axis stabilized with closed loop attitude control and follows the tradition of many recent astronomical space missions and presents no new problems in the control of the dynamics of the spacecraft.

The different type of stabilization leads to a few differences in the choice of the sensors and the actuators and of their location.

Concerning sensors:

Star tracker:

	Planck	Herschel
Function	Autonomous S/C quaternions determination	
Operating modes	Science, orbit control and angular momentum	Science
Performance	Medium attitude determination accuracy @ 1 deg/sec "image speed"	High attitude determination accuracy @ (close to) inertial pointing
Redundancy	1 nominal, 1 cold redundant	
Localisation	External side of Equipment panel	Dedicated structure below the cryostat

Table 20: Comparison of the implementation of the STR

Notice that the STR position was a critical issue for H and that the requirements on the accuracy of spin axis repointing for P constrained its STR specification.

Concerning actuators:

Reaction control system (RCS):

	Planck	Herschel
Function	Provide linear and angular momentum for orbit and attitude control	Provide linear and angular momentum for orbit control and RW unloading
Propellant	Anhydrous hydrazine monopropellant	
Propulsion mode	Blow down with a 4:1 ratio	
Propellant mass (see ΔV requirements)	346 kg	134 kg
Number of tanks	3 same design 2	
Number (excl. redundancy) and performance of the thrusters	Six 20 N same design Six 20 N	
	Two 1N	
Redundancy	1 nominal and 1 cold redundant branch of thrusters	

Table 21: Comparison of the implementation of the RCS

Reaction Wheel Assembly (RWA):

Due to slew efficiency requirements, Herschel requires the use of reaction wheels assembly with high torque capacities during science and survival modes. The RWA is made of four reaction wheels (3 nominal + 1 cold redundant) on a dedicated equipment panel.

1.3.4 THERMAL CONTROL

The top level functions of the P/H TCS are:

- To reject the unit dissipation to the deep space (OSR, Black paint).
- To insulate the external surfaces of units and module not used for the heat rejection (MLI, Aluminized tapes).
- To increase the linear conductance for the units that need to be cooled via conduction (fillers and thermal doublers).
- To insulate conductively the units whose sink is too hot or cold (thermal washers).
- To provide power dissipation for the units or enclosures (heaters, thermostats, thermistors).

The commonality of the thermal control design mainly concerns the use of well proven design solutions and the choice of material or items:

- MultiLayer Insulation (MLI) blankets
- High and low emissivity tapes
- Paints and coating
- Heaters and thermistors
- Second Surface Mirrors (rigid and/or flexible)
- Interface fillers, washers and low conductivity stand-offs for mounting equipment and equipment supports
- Aluminium doublers

On the contrary, the sizing of thermal equipments (radiator areas, heater powers) will be different for the two satellites. A specific feature of the Planck TCS is the use of V-grooves, to obtain low temperature (60 K), heat sinks for the H_2 sorption cooler. Lower temperature coolers required by the instruments are not discussed further.

	Planck	Herschel
Thermal analysis	Different Geometrical Mathematical Model (GMM) Different dissipation of the WU Different worse cases due to different SAA Different transient modes due to different duration of attitude change and SAA	
Thermal control of the PLM	H_2 Sorption Cooler H_2 tanks, compressors and electronics on SVM	Cryostat on PLM
Thermal design principles	SVM decoupled from PLM Same thermal active regulation architecture (Pulse Width Modulation with PI regulation) but dedicated control laws Same choice of material but different sizing	
Particular attention	SCC cyclic dissipation (Heat pipes on the 3 panels)	HIFI and STR stability requirements (dedicated active thermal control law)

Table 22: Planck and Herschel thermal control comparison

1.3.5 RF COMMUNICATIONS

The allocation of L/MGA to data bit rate is exactly the same for P/H:

	New Norcia	Type of antenna	Kourou	Type of antenna
Uplink bit rate	4 kbps	L/MGA	125 bps	LGA
			4 kbps	MGA
Downlink bit rate	5 kbps	LGA	500 bps	LGA
	150 kbps	MGA	150 kbps	MGA
	1.5 Mbps	MGA		

Table 23: Common design choice for RF communications

For both missions, X band carrier frequencies were chosen for data up and downlink. TM&TC requires:
- Two transponders
- Two TWTA (Travelling Wave Tube Assembly)
- One RFDN (Radio Frequency Distribution Network)
- Two hot redundant receivers and two cold redundant transmitters
- 1 Medium Gain Antenna
- Low Gain Antennas

1.3.6 POWER

1.3.6.1 Power generation

Power is generated for both Planck and Herschel by fixed SA which are located below the SVM for Planck on the PLM for Herschel (no need for deployment for both) and which provides:
- electrical power from the sun input
- a thermal shield between the sun and the SVM/PLM.

Basically, P/H use the same solar cell technology (triple junction AsGa), same substrate, interconnections and manufacturing procedures. The only differences are the shape and the size of the SA due to different P/L demands.

	Planck	**Herschel**
Solar cell technology	Triple junction GaAs	
Surface	11.26 m^2	14 m^2, 11.71 m^2 effective
Localisation	Below SVM: Needed cut outs Impact of plume impingement	On PLM
Shape	5 panels in overall circular shape Diameter limited by fairing diameter	3 rectangular panels: Different filling factor compared with P.

Table 24: Comparison of P/H power generation

1.3.6.2 Power storage

Low capacity Li Ion battery has been chosen for both Planck and Herschel because this technology gives the highest flexibility in terms of future grown capability, compared with high capacity cells. The design is the same for both missions.

Battery cell technology	Low capacity Li Ion, derived from COTS
Architecture	24 parallel strings made of 6 cells in series
Theoretical energy (at 100 % DOD)	777 Wh
Required energy	568 Wh
DOD	76 % w/o failure 80 % with one string failure
Redundancy	1 hot redundant battery

Table 25: P/H battery characteristics

The sizing case for the P/H battery design is the launch phase for Planck before separation (50 min compared with 45 min for Herschel).

1.3.6.3 Power regulation and distribution

Power Control Subsystem (PCS) has been suitably dimensioned to be compatible with both Herschel and Planck satellites, thus providing a high level of commonality:
- Power Control and Distribution Unit (PCDU) is the same for the two satellites.
- Due to the specificity of the P/L needs, customization on distribution lines is performed at harness level thus not affecting the PCDU.

The maximum power available from the SA is not completely used by loads. This is due to the H/P topology heritage based on DET (Direct Energy Transfer) technique which dissipates power which is not used by the loads with an S3R (Sequential Switching Shunt Regulator): therefore, when the available SA power exceeds the total bus power demand, including battery recharge parts of SA sections are short-circuited.

The overall architecture, which is based on a decentralised concept, is modular and similar at a "box" level for both missions. The different units on the SVM are linked with the standard MIL 1553 data bus. We can make out 6 main modules or systems:

- PCDU: Power Conditioning and Distribution Unit
- ACMS: Attitude Control and Measurement Subsystem with the RCS and the Attitude Control Computer (ACC)
- CDMS: Command and Data Management Subsystem
- TT&CS: Telemetry, Tracking and Command System
- TCS: Thermal Control System
- P/L warm units

1.3.7 ON BOARD DATA MANAGEMENT

The SVM avionics core and architecture of the CDMS (Command and Data Management Subsystem) and ACMS (Attitude Control and Measurement Subsystem) are identical for P/H and comprises respectively two distinct control computers (CDMU, ACC for Attitude Control Computer) based on ERC-32 microprocessor. The two computers, identical both for Herschel and Planck, are connected by means of a 1553 bus.

1.3.8 SW ARCHITECTURE

SW product tree for both P/H is quite similar:

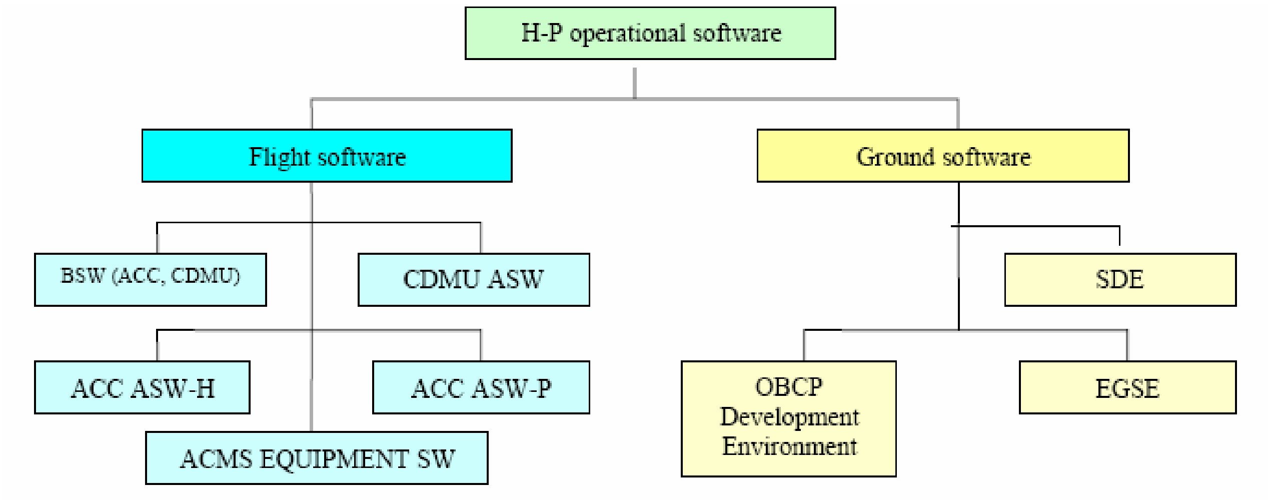


Figure 13: P/H SW product tree

1.3.8.1 Flight SW

The Basic Software (BSW) layer represents the lowest SW layer of ACC and CDMU computer and supplies basic services to interface the hardware devices and accomplishes important functionality by itself. It is basically the same for both missions, except for the I/O drivers of the units belonging to the two S/C subsystems (CDMS, ACMS).

The highest SW level is represented by the Application Software (ASW) layer which makes use of the BSW provided services. Basically the ASW handles S/C autonomous functions and provides user's services for spacecraft observability and commandability. It implements the management of the following main functions and subsystems:

- Mission (Event, MTL, OBCP, modes)
- Payload
- Power Control Subsystem
- TT&C
- Thermal Control
- FDIR,

which are very mission peculiar (different modes, different attitude and thermal control laws, different FDIR functionalities…) hence different ASW for P and H.

1.3.8.2 Ground SW

SDE (SW Development Environment) uses the same tools for both missions, EGSE has the same functional architecture (it will be used to test the P/H AVM) but OBC will be rather different, since it is run onboard through CDMU ASW supported services which are basically different.

1.3.9 EXTERNAL INTERFACES

I/F are often critical in a satellite design and failures has occurred in some past programs (e.g.. XMM).

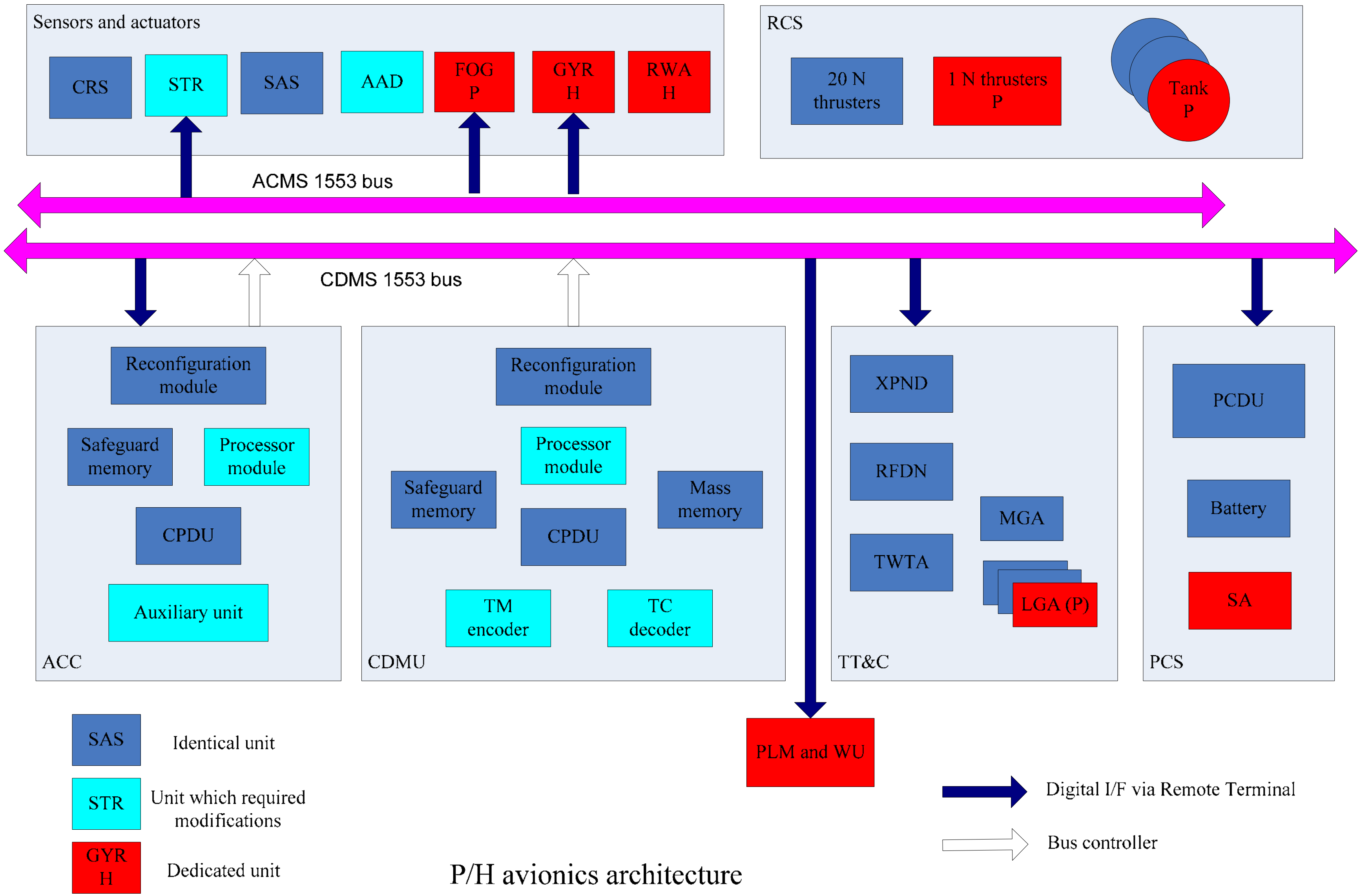
	Planck	Herschel
LV I/F	A5 2624 mm adaptor (upper diameter) for both with Sylda 5 P/L internal carrying structure	
SVM PLM I/F	6 dedicated skin connectors on shear panels and subplatform	12 dedicated skin connectors on upper closure panels
EGSE I/F	Skin connectors with different locations and different routing	

Figure 14: Comparison of external I/F

1.4 Synthesis

	Planck	Herschel
Launch	A5 ECA dual launch	
Launch mass	2000 kg	3400 kg
Dimensions	4.2 m high 4.2 diameter	7.2 m high 4 m diameter
Orbit	L2, small Lissajou	L2 large Lissajou
Propellant mass	346 kg	134 kg
Nominal in orbit lifetime	1,75 years	3.5 years
Configuration	Spin stabilized	3-axis stabilized
Pointing accuracy	0.3' RPE 8' APE	0.24" RPE 2.5"APE
P/L power need	1000 W	550 W

Table 26: P/H main specifications



P/H avionics architecture

1.5 Programmatic aspects

The combination of two missions into one programme which aims achieving economy of scale and taking advantage of the technical common developments for both spacecrafts, led to several consequences, such as:

1.5.1 MANAGEMENT STRUCTURE

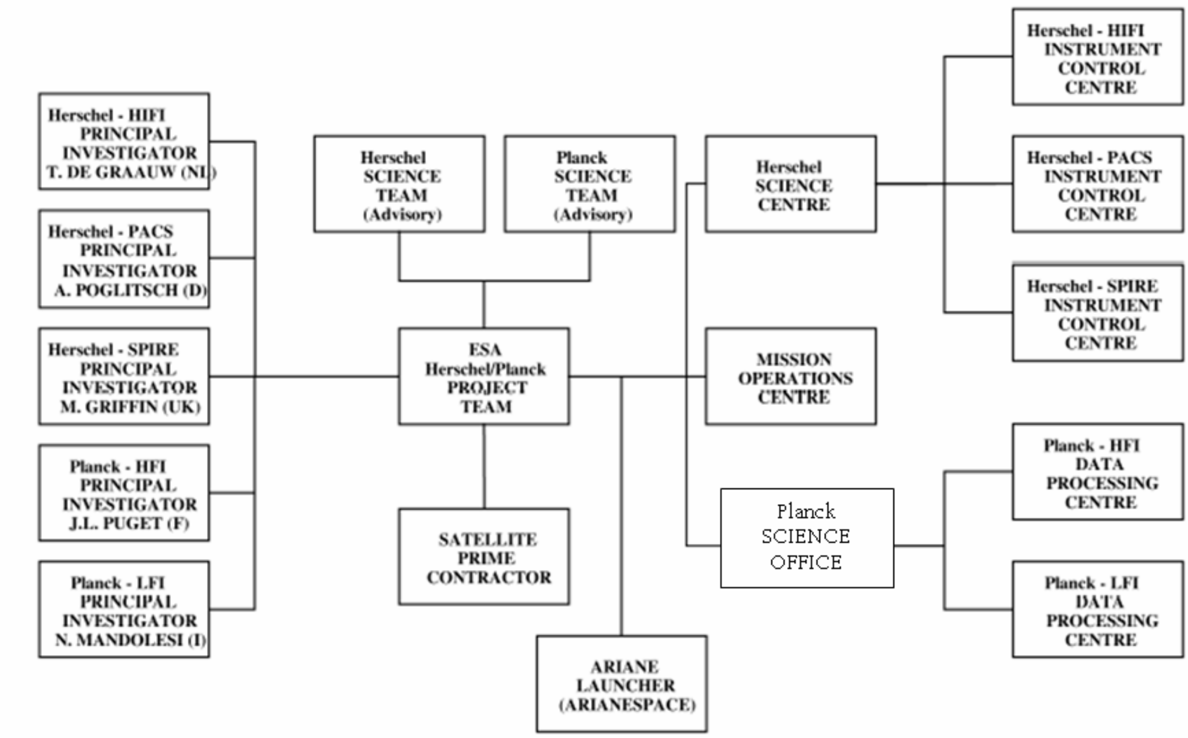


Figure 15: P/H project organization

It is noteworthy to point out the fact that there is a unique project team for both missions which plays a central role interfacing the different groups.

1.5.2 DEVELOPMENT PROCESS

P/H Assembly Integration and Verification (AIV) approach and development plan have been used as working basis to promote commonality in terms of verification activities, test sequences and Ground Support Equipment (GSE).

Globally, only one complete model of each satellite is being developed.

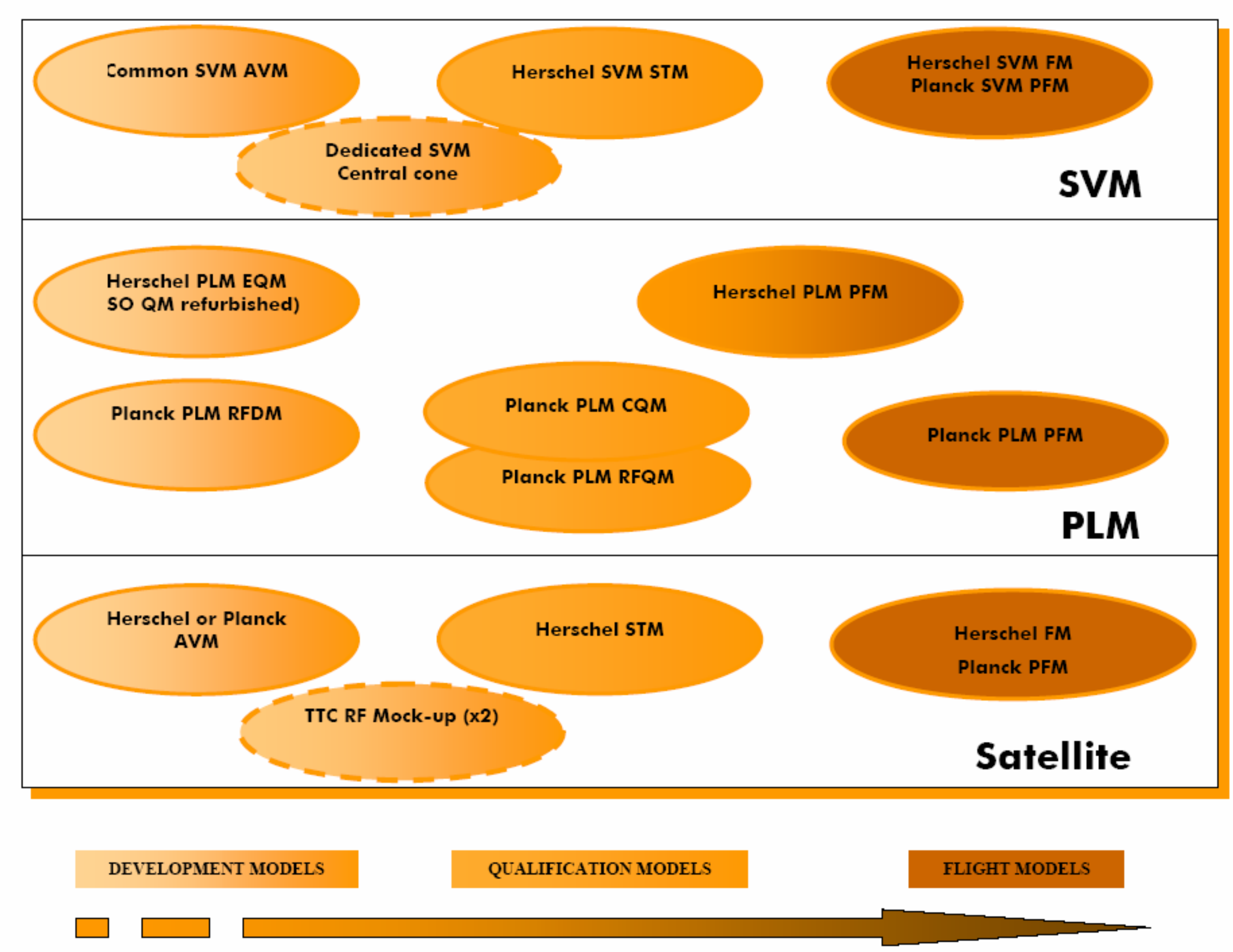


Figure 16: P/H development logic

The model philosophy applicable on Herschel is at system level based on:

- a multipurpose Structural & Thermal Model (STM)
- a Flight Model (FM).

However the reuse on Herschel FM of some STM elements such as WU panels is envisaged.

The model philosophy applicable on Planck is at system level based on:

- a Proto-Flight Model (PFM).

The choice to use a common platform thus allows to skip the manufacturing of a STM for Planck

These models are completed by:

- a common Avionics Model (AVM)
- 2 TTC RF Mock-up

In the frame of the Planck Satellite PFM approach and due to SVM commonality, for launcher compatibility, only a dedicated SVM Primary Structure is submitted to Static Load Test (SLT) combining the worst cases of Herschel and Planck configuration. This test specimen is a

H/P hybrid cone covering H/P criticality for both configurations and built from basic Planck cone SVM without the Planck reinforcement but with implementation of some Herschel specificities.

The development program considered separately dedicated test programs for
- the qualification models of the scientific instruments,
- the SVM development and
- the S/C qualification. (through the PFM)

in order to carry out tests in parallel, decreasing phase D duration.

An important point is the AVM with which the electrical interfaces and SW functional validation is performed. It can be configured either in Planck or in Herschel version. The verification on AVM is conceived to minimize the hardware needs with maximum reuse of Engineering Model equipments. The AVM will be kept operational all along the AIT sequence to be usable for potential failure analysis or for validation of software modification.

The AVM will be developed in order to allow the maximum flexibility between Herschel and Planck testing in terms to swap from Herschel to PLANCK configuration and vice-versa changing the HW and SW configurations in a short time, typically 48 hours.

For this the following capabilities will be implemented:

− Quick SW loading on Avionics on board computer, ACC and CDMU, in order to quickly modify the SW configuration from Herschel to PLANCK and vice-versa.

− I/F Connectors that allow to integrate easily the AVM Common Elements and the AVM Modular Elements.

1.5.3 RISK MANAGEMENT

All the SVM units which are common to Herschel and Planck apply the same type of redundancy and the same level of failure tolerance because the priority functions are basically the same for P and H. For instance, power transmission regulation and control is two failure tolerant for both S/C.

Moreover, the fact that some units are flight proven significantly decreases the level of risk. For instance, the 20 N thrusters are used on both Herschel and Planck RCS's. This type of thruster (and the propellant flow control components) has been designed, developed and qualified for the XMM/Integral satellites and, later, delta qualified for the MetOp program. Therefore, no delta mechanical qualification testing is needed for the 20 N thrusters (and the associated items).

2 PROPOSITIONS TO REUSE THE SVM FOR GAIA AND EDDINGTON

The information provided in this chapter is derived from different technical notes provided by industry to ESA. It should be noted that the Eddington mission was cancelled due to lack of funding, while the present Gaia design differs considerably from what is presented here.

2.1 Reuse for Gaia

2.1.1 MISSION OVERVIEW

Gaia is a survey mission which will consist in creating the largest and most precise three dimensional chart of our Galaxy by providing unprecedented positional and radial velocity measurements for about one billion stars in our Galaxy and throughout the Local Group.

Launch:
Soyuz Fregat single launch initially planned for December 2011

2.1.2 THE REASONS FOR REUSING A SVM

Following the decisions of the November 2001 Ministerial Conference, an urgent reassessment of the technical baseline of Gaia was identified in order to reduce drastically the mission cost at completion. ESA had a good level of confidence that Gaia budget could be significantly reduced down to a level of 80% to 70%.

Cost reduction was deemed to be achievable by choosing a Soyuz-Fregat launch (instead of A5), by reviewing the P/L design and by reusing as much as possible low cost and updated design busses, like the Herschel-Planck SVM, and identifying solutions with a potential cost reduction.

2.1.3 COMPARISON OF THE REQUIREMENTS AND DESIGN SOLUTIONS

2.1.3.1 Launch

	P/H (baseline)	Gaia
Launch vehicle	A 5 ECA	Soyuz Fregat
Fairing diameter	4200 mm	3800 mm
Adapter upper diameter	2624 mm	1666 mm
Configuration	Dual launch	Single launch
Expected date of launch	Beginning 2008	End 2011
Consequences		
Need to adapt the S/C dimensions to Soyuz fairing and the adaptor diameter More flexible launch opportunity for Gaia		

Table 27: Comparison of the launch opportunity between P/H and Gaia

2.1.3.2 Orbit, lifetime and ΔV requirements

The operational orbit of Gaia is a small Lissajou orbit at the vicinity of L2, similarly to Planck. Therefore, the propellant budget will be similar for these two missions. Besides, Gaia is expected to be operational during 5 years which is closer to Herschel lifetime than Planck one. As Herschel has been designed for mission duration of 3.5 years mainly due to payload autonomy (size of the cryostat), the compatibility of Herschel SVM on GAIA can be easily demonstrated: no critical lifetime constraints were reported to the P/H SVM.

	Planck	Herschel	Gaia
Orbit	Small Lissajou	Large Lissajou	Small Lissajou
Injection	Need for a demanding ΔV manoeuvre	Free injection	Need for a demanding ΔV manoeuvre
Orbit injection and eclipse avoidance	225 m/s	0 m/s	270 m/s
Orbit maintenance	4.5 m/s	8.9 m/s	Approx. 10 m/s
Nominal lifetime	21 months	3.5 years	5 years
Including degradable items	2.5 years	6 years	6 years

Table 28: Baseline choice for Gaia w.r.t. orbit and lifetime

2.1.3.3 Autonomy

Due to the orbit and in the view to achieve the maximum recurrence (OBMM size, data rate and FDIR functionalities), the autonomy requirements are the same for both P/H and Gaia.

2.1.3.4 General configuration

Gaia, as P/H S/C, is made of 2 modules:

- a PLM which provides a very stable thermal environment to the focal plane assembly,
- a SVM which carries the warm units of the PL and provides all the required functionalities to the PLM (power, AOCS, thermal control, communications, sunshadowing…)

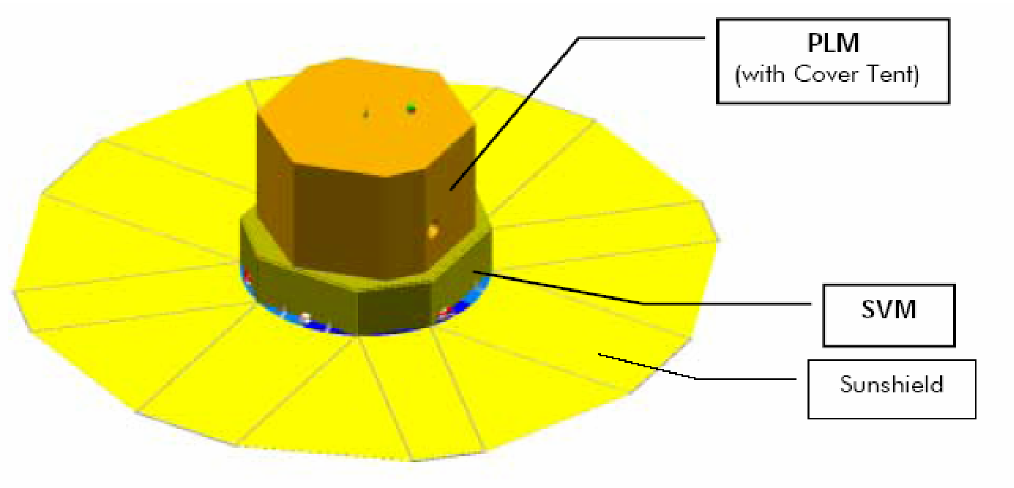


Figure 17: Gaia overview

2.1.3.5 Mission concept

Similarly to Planck, Gaia is a survey mission which goal is to map the sky. To this purpose, the S/C has to follow a specific scan law which is compared to Planck's one in the following table:

	Planck	Gaia	Consequences on the design
Spin rate	1 rpm	360 deg over 6 h	AOCS communication
Precession	No	360 deg over 70 days	
Nutation	To avoid as much as possible		Very good knowledge of mass distribution
Angle between Sun-Earth axis and spin axis	+/- 10°	50°	Impacts on: - antenna lay out - SA efficiency

Table 29: Scan law comparison

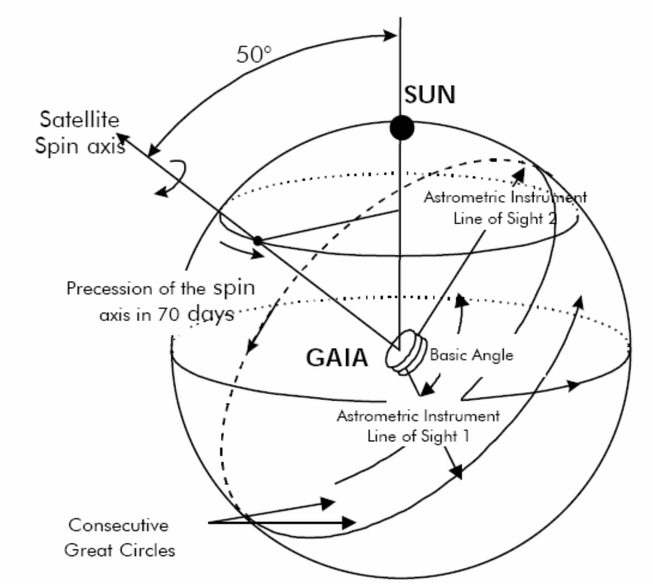


Figure 18: Gaia scan law

2.1.3.6 Structure

The Gaia SVM is P/H recurrent at different levels:

- same design: a primary structure composed of one central cone and an octagonal secondary structure
- same Planck SA concept : annular and central SA below the SVM
- same overall dimensions
- same material characteristics

	Planck	Gaia
Structural concept	Same modular concept for the octagonal box (same elements with same functions)	
Material choice	Same material	
Central cone	Baseline	Reversed
Adapter diameter	2624 mm	1666 mm
PLM accommodation	On the subplatform	3 bipods attached to central cone
SA support	Same design below the SVM SA acting as a sunshield	

Table 30: Comparison of the structural design

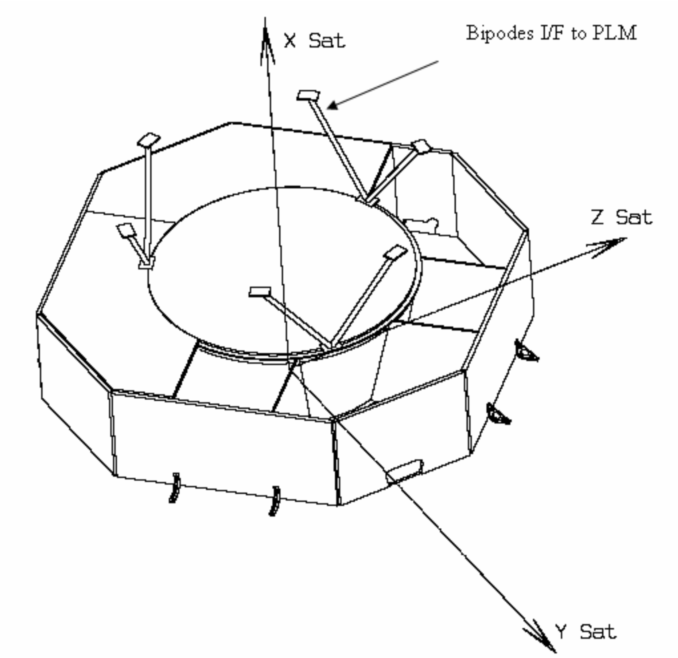


Figure 19: Gaia structural design

2.1.3.7 General lay out

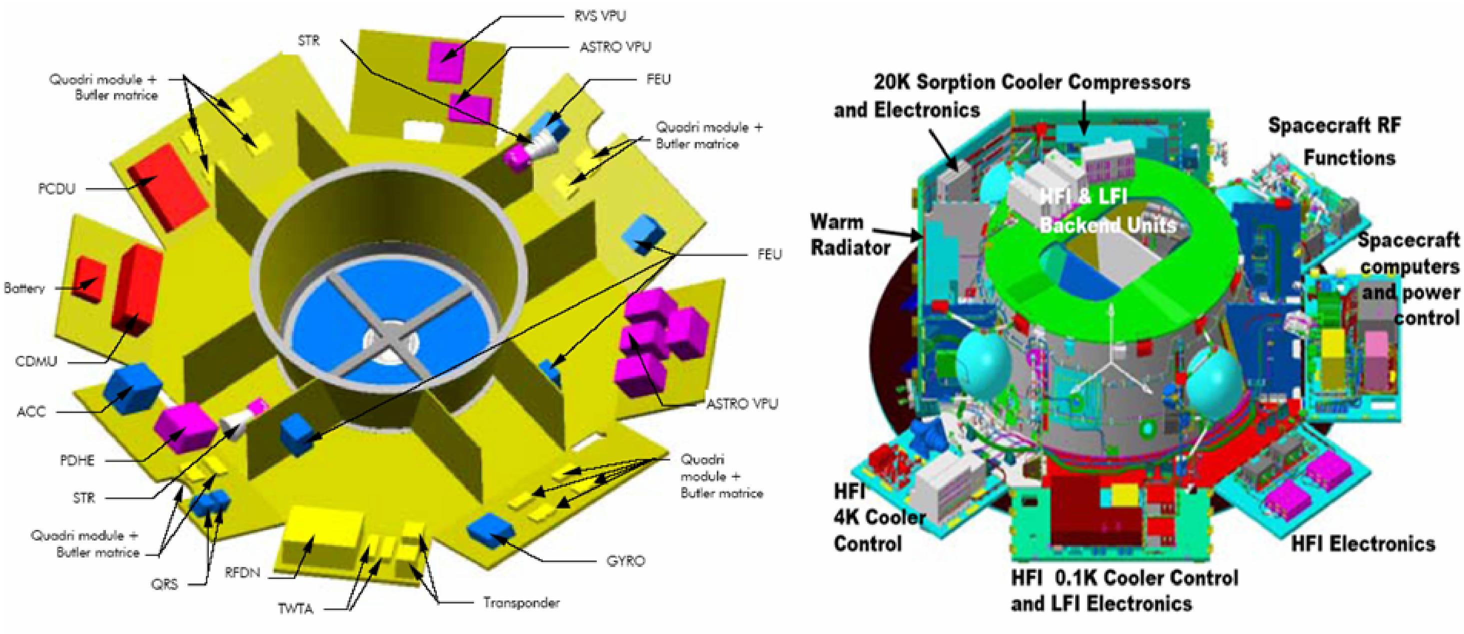


Figure 20: Gaia and Planck general layout

As Planck and Herschel, Gaia has equipment panels which are dedicated to either SVM functions, or PL WU with the same type of constraints (gathering of functional chains, spacecraft mass balancing, thermal dissipation distribution).

2.1.3.8 Sunshield

The requirement of no PLM illumination (due to extremely tight thermal stability requirements) during the mission combined with the scanning law requirement, leads to the accommodation of a 10 m wide specific Sunshield

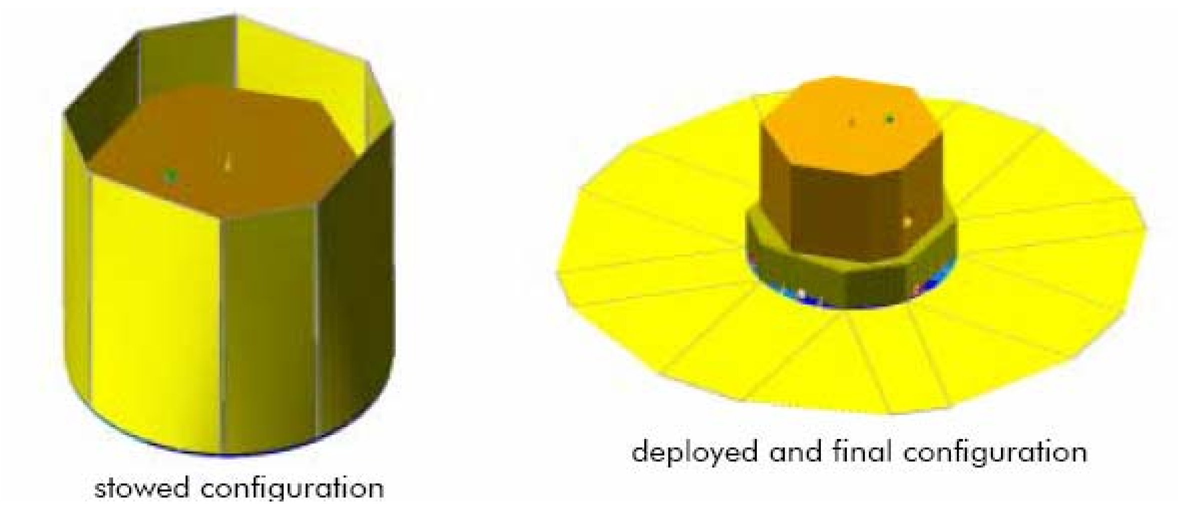


Figure 21: Gaia sunshield

The sunshield requires pyro activation devices for deployment, hence an additional energy resource, dedicated electronics for prearming, arming and firing the electro explosive device, and a further study of shock impact.

2.1.3.9 Mass budget

	Planck	**Gaia**
PLM mass	**336**	**570**
ACMS	30	12
CDMS	15,5	19
Harness	85	25
PCS	36	33,8
RCS	78	35,3
SA	45	69 (incl. SSH)
FEEP SS		36
Structure	**310**	**183**
TCS	**53**	**9,9**
TT&C	24	23,3
Total SVM dry mass (separation system and system margin excluded)	676,5	461

Table 31: Mass budget comparison

2.1.3.10 AOCS

2.1.3.10.1 Type of stabilisation

Contrary to Planck, Gaia is 3-axis stabilized, even if it is a spinner (though at a very low spin rate: 1 round every 6 h)

2.1.3.10.2 Pointing requirements

Gaia has much more stringent pointing requirements than Planck, hence the use of mN thrusters (see below).

	Planck	Herschel	Gaia
APE of LOS (short term)	1.5'	0.24"	0.18'
RPE of LOS	1.5'	0.24"	<0.02"
AME of LOS	0.48'	0.24"	0.003"
ARE	5.4'/sec		1"/sec

Table 32: Pointing requirements comparison

2.1.3.10.3 Sensors

	Recurrence	Remarks
SAS	P/H fully recurrent	
CRS	P/H fully recurrent	
GYRO	H fully recurrent	
STR	SB 4000 fully recurrent, H type Same configuration (180° one w.r.t. the other)	Different location
AAD	No AAD for Gaia	Replaced by SW detection using SAS data

Table 33: Comparison of sensors HW

2.1.3.10.4 Actuators

2.1.3.10.5 RCS

Hydrazine monopropellant thrusters are used to ensure a 3 axis torque capacity for attitude control and fast reorientations during transfer, and a 3 axis force capacity for ΔV manoeuvres (excepted orbit maintenance) without illuminating the PLM.

<table>
<tr><th></th><th>Planck</th><th>Gaia</th></tr>
<tr><td>ΔV requirement (m/sec)</td><td>319.3 [4]
341 [15]</td><td>304</td></tr>
<tr><td>Propulsion type</td><td colspan="2">Hydrazine monopropellant
Nitrogen as pressurisant gas</td></tr>
<tr><td>Propellant mass (kg)</td><td>346</td><td>217</td></tr>
<tr><td>Number of thrusters
(w/o redundancy)
and performance</td><td>Six 20 N
Two 1 N</td><td>Six 10 N</td></tr>
<tr><td rowspan="2">Tanks</td><td>3</td><td>2 (Integral type)</td></tr>
<tr><td colspan="2">Different lay out</td></tr>
</table>

Table 34: RCS comparison

2.1.3.10.6 FEEP

FEEP thrusters are required for fine orbit maintenance and attitude control during science routine operations during operational phase due to more stringent pointing requirements (compared with Planck or even Herschel). It is a completely new subsystem w.r.t. P/H design. 8 thrusters (w/o redundancy) will compensate the solar pressure and the gravitational force.

The use of this technology is completely new compared with P/H missions. As a consequence, new models will have to be taken into account for SA contamination, new power budget will have to be reassessed (+ 49 W avg w/o margins) as well as a new new mass budget (+ 34,4 kg).

2.1.3.11 Thermal control

The SVM must provide the PLM with a permanent sun shadowing for all mission phases and offer to the PLM a very stable thermal environment in order to guarantee the required thermo-elastic stability of the optical bench (µK gradients over 1 spin period of 6h) and the SVM PLM interface (1 mK for 6 h TBC). Since the operating temperatures are quite different (room temperature for SVM, 160 K average for PLM) SVM shall be thermally decoupled from the PLM as much as possible (same as P/H) in order to minimize the heat flux from SVM to PLM.

The proposed solution is based on both a regulation close to the dissipative equipments, and a thermal decoupling between the bipod interfaces and the dissipation zone. As used for P/H SVM, the active thermal regulation based on the well proven Pulse Width Modulation (PWM) control design.

To reach the required level of stability at the bipod interfaces, the lateral panels shall be thermally decoupled from the rest of the SVM, especially with the central cone (supporting the three bipods) and the closure panel by using low conductivity cleats and MLI blankets.

.

Similarities		
Design principles		PLM SVM decoupling
Passive control	Insulation	MLI, low conductive cleats
	Radiation	Equipment panels act as radiators Black painted units to minimize temperature gradients
Active control		Regulation with heaters, thermistors using PWM
Differences		
Particular attention		SCC for Planck Level of thermal stability at the PLM SVM IF for Gaia
SW development		SVM thermal analysis (a specific model is necessary to determine)
		Algorithm parameters for the thermal regulation.
Sizing		Position and number of heaters
		Size and position of MLI on external radiators.
AOCS specific		Needed power for thermal regulation and heaters according to the AOCS modes.
		Temperature of interfaces and equipments, according to the AOCS modes

Table 35: Comparison of Planck/Gaia thermal control

2.1.3.12 Communications

	P/H		Gaia
Ground visibility	3 h per day		6 h per day (?)
Max TM bit rate	**1,5 Mbps**		**5 Mbps**
Modulation scheme	**GMSK**		**8PSK**
Antennas	Planck	Herschel	2 LGA **1 HGA**
	3 LGA	2 LGA 1 MGA	

Table 36: Comparison of the communication architecture

For mission needs, a High Gain Antenna (HGA) will perform the transmission in X-band of the scientific data from satellite to Earth due to the high bit rate and the GAIA requirements for attitude stability. As a baseline, the phased array antenna is envisaged to meet RPE requirements (avoiding mechanisms). It shall be accommodated in SVM close to the satellite/launcher interface.

2.1.3.13 Power

2.1.3.13.1 Power generation

	Planck	Gaia
PLM power	1026 W	700 W
SVM power	404 W	609 W
S/C power	1430 W	1440 W
SA topology	Below SVM Body mounted w/o deployment 30 sections to be compatible with the common PCDU	
		Specific cut out in the central panel for HGA
Solar cell technology	Triple junction AsGa but different solar cell accommodation	
Average operating temperature	110° C	60° C
SA area	11,26 m^2	10,45 m^2
SAA during observation mode	10°	50°
Power provided by SA (EOL)	1900 W	1456 W

Table 37: Power budget and power generation design

Due to Gaia specific needs, only the bare solar cell can be fully reused from P/H design. Moreover, the solar cell technology obsolescence may be taken into account, and future solar cells with improved efficiency are likely to appear in the next years. It would then allow to deliver more power from the solar array.

For a given solar flux, the incident angle of 50° reduces by about 35% the incident solar flux on the solar array compared with Planck which is a significant power decrease. In the same time however, the Gaia average solar array temperature will be lower than the Planck one so the solar cell efficiency will be better, but globally for the power generation the SAA increase is a major penalty for the SA sizing.

By the way, the fact that the maximum power is lower than for Planck is favourable for electronic recurrence because the Gaia SVM would not have to provide higher levels of current.

2.1.3.13.2 Power storage

Since the overall GAIA mission is eclipse free and the SA delivers all the required power to the S/C all along the mission, except in the 2h Stand By mode (launch phase) during which only the battery can provide energy to the S/C. This phase will be the sizing case for the battery sizing. Fortunately, Planck sizing case is more stringent, therefore the H/P battery is fully compatible with the Gaia need, and 100% recurrence is expected for this equipment.

2.1.3.13.3 Power distribution and regulation

	P/H	Gaia
Type of regulation	Direct Energy Transfer	
Type of power bus	28 V fully regulated	
Battery charge concept	BCR via 3 sections	S4R via 6 sections
Battery discharge concept	BDR	
PCDU	Same architecture	
		Additional electronic card for pyro lines New structural box

Table 38: Comparison of the power distribution and regulation concept

Given the recurrence foreseen on H/P, the H/P SVM PCDU shall be compliant with the most demanding requirement, i.e. the Herschel one. As the Gaia requirement is less demanding, it will be fully met by the H/P PCDU. However, compared with Planck, the number of equipments (i.e. the number of power lines) increases though being lower than for Herschel.

2.1.3.14 On board data management

On board computers

The GAIA avionics would be based on the H/P one composed of 2 computers: the Command and Data Management Unit (CDMU) and the ACC (Attitude Control Computer). The GAIA AOCS equipment connection with both computers results from the maximisation of the ACC recurrence. For instance, FEEPS interface on the CDMU via Remote Terminals which enables the full recurrence of the H ACC.

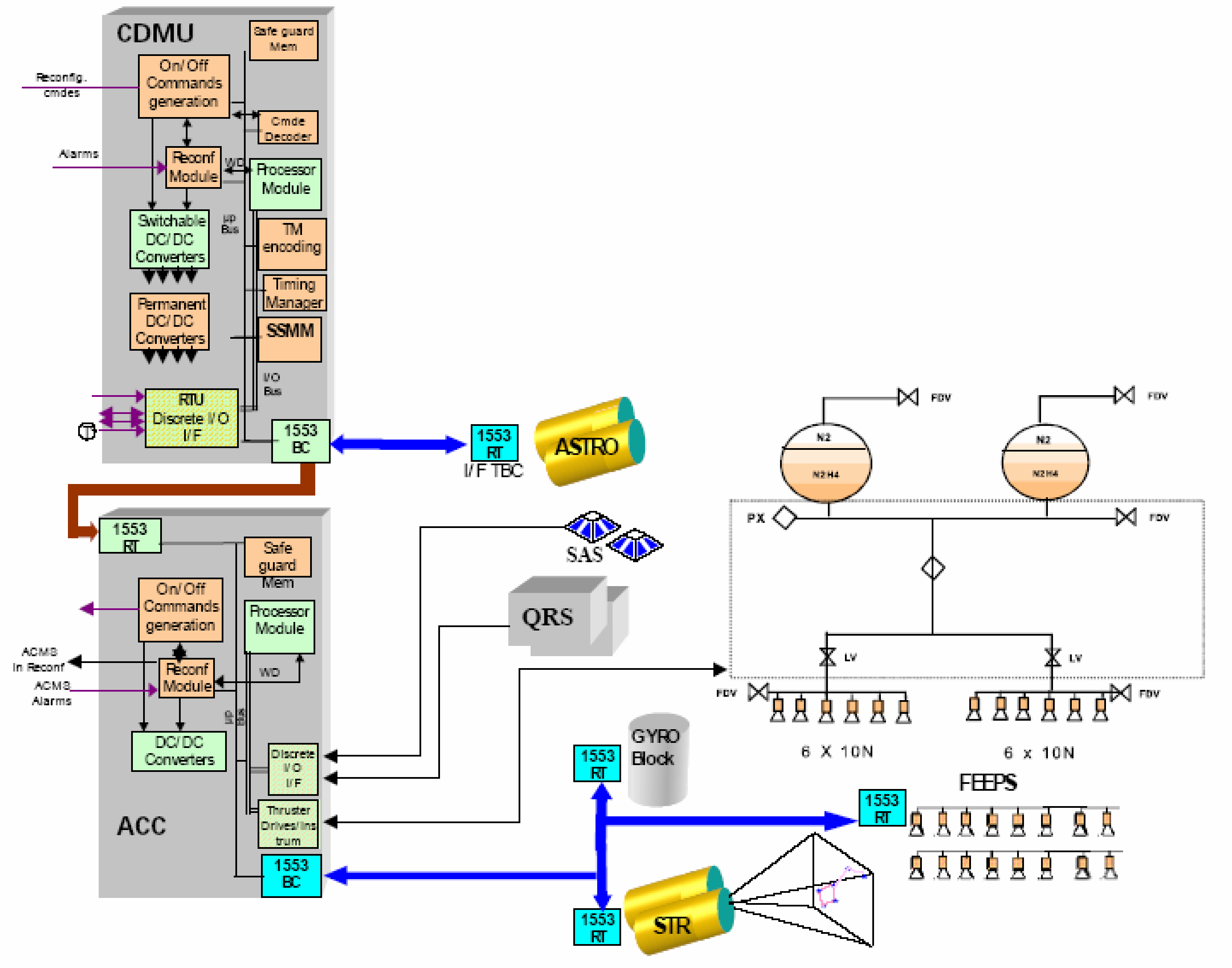


Figure 22: Gaia AOCS equipments I/F with avionics

Gaia CDMS has the same breakdown at a "box" level: it is made of the same elements with the same internal I/F to the processor (except the 1355 SpaceWire which is not used as an internal I/F for P/H), and the same external I/F.

	P/H	Gaia	Consequences on the design
Science data rate	130 Kbps avg	1 Mbps avg	CDMU accommodation
MM size (EOL)	**25 Gb**	**400 Gb**	Different storage boards breakdown Adaptation of MM DC/DC converter
Local Oscillator		Higher short term stability	Ultra Stable Oscillator required
TM user bit rate	1.5 Mbps	5 Mbps	Accommodation of TM Encoder for appropriate input

Table 39: Different DH requirements

Ground station outage to be considered is 3 consecutive days. Hence, the Solid State Mass Memory capacity must allow the storage of science data (at 1 Mbps mean rate) during 4 days. SSMM capacity to be considered is 400 Gbit EOL. As a consequence of this Mass Memory capacity increase, Power Converters must be accommodated to cope with the associated consumption increase.

SW architecture

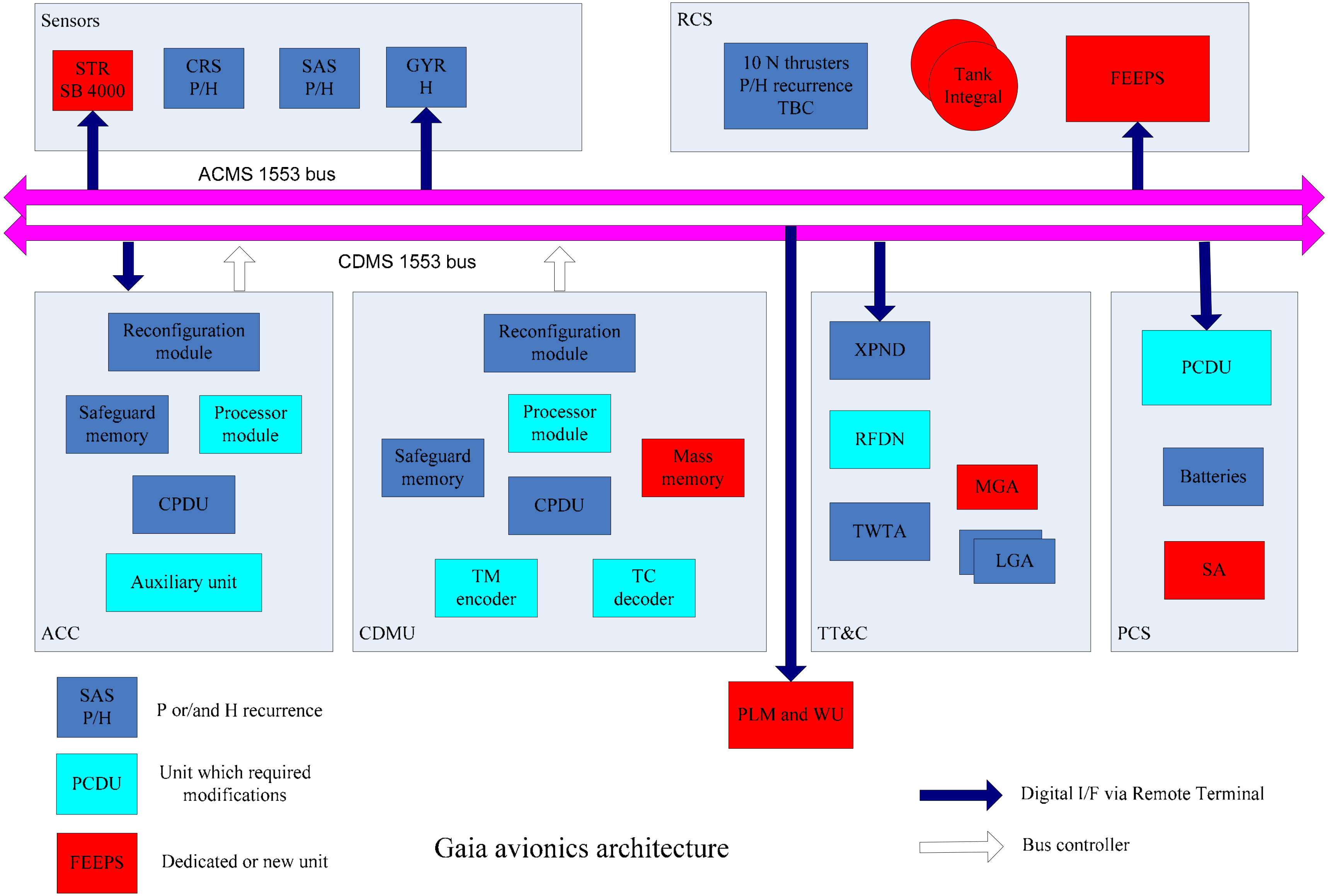
The same flight SW breakdown is applied for P/H and Gaia: BSW and ASW. An intensive reuse of the P/H BSW is assumed (despite adaptations due to different SSMM size and different TM data bit rate) whereas specific ASW (with dedicated parameters definition) has to be developed to cover Gaia need. For instance, CDH and AOCS SW need to be modified.

However, development standards (specification, validation and tests methods, programming language) used for P/H will be applicable for GAIA.

2.1.4 SYNTHESIS

	P/H	Gaia	Impacts
Launch	A 5 ECA dual passenger	Soyuz Fregat single passenger	
Fairing diameter	4200 mm	3800 mm	SA diameter
Adapter diameter	2624 mm	1666 mm	Adapter accommodation
Configuration	Spin stabilized (P)	3 axis stabilized	
Scanning law	1 rpm (P)	1 round over 6 h	AOCS, communication
Rotation axis tilting	+/- 10° (P)	+/- 50°	SA, sunshield sizing
Lifetime	2.5 years (P)	6.5 years	SA sizing
Orbit	Small Lissajou orbit (P)		
Propellant mass	346 kg (P)	217 kg	Tanks sizing
P/L power	1000 W (P)	700 W	
Data volume from P/L	130 kbps (P)	1 Mbps	Antenna and MM sizing

Table 40: Comparison of the specifications and impacts on the design



Gaia avionics architecture

	Adaptations
Structure	Different LV I/F Different PLM accommodation (w.r.t. P) Reversed central cone Sunshield accommodation
AOCS	STR SB 4000 recurrent No AAD Integral type tanks 10 N thrusters (instead of 20 N) RWA replaced by FEEPS
Thermal control	Dedicated SVM thermal analysis Different thermal items sizing Different power lines for heaters (TBC)
Communication	Accommodation of TM modulator Accommodation of a HGA
Power	Risk of solar cell obsolescence Different P like solar panels size Different accommodation of solar cells Different battery charge concept (S4R) Additional pyro lines (hence additional electronic card) on PCDU
OBDM	MM size increase Integration of an Ultra Stable Oscillator Adaptation of TM encoder Specific ASW

Table 41: SVM adaptations for Gaia mission

2.2 Reuse for Eddington

2.2.1 MISSION OVERVIEW

Eddington is a mission dedicated to sounding the interior of stars (astro seismology) and searching for habitable Earth like planets deduced by precise measurements of stellar light variations.

This mission was finally cancelled. The level of definition correspond to the one reached in April 2003

Launch:
Soyuz Fregat (with ST fairing) single launch initially planned for beginning 2008

2.2.2 THE REASONS FOR REUSING A PLATFORM

Selected in October 2000 as a reserve mission (F2/F3 flexible mission, such as Mars Venus Express), Eddington was about to be a very cost effective mission, implemented over short time scale with new approach to development. For this purpose, ESA requests the reuse of P/H SVM.

2.2.3 COMPARISON OF THE REQUIREMENTS AND DESIGN SOLUTIONS

Preliminary remark: due to requirements closer to Planck than to Herschel one's, the adaptation is rather based on Planck SVM (except pointing requirements).

2.2.3.1 Launch

	P/H (baseline)	Gaia	Eddington
Launch vehicle	A 5 ECA	Soyuz Fregat	
Fairing diameter	4200 mm	3800 mm	
Adapter diameter	2624 mm	1666 mm	1194 mm
Configuration	Dual launch	Single launch, 1640 kg max launchable mass	

Table 42: Different launch configurations

Similarly to Gaia, Eddington would be launched as a single passenger on Soyuz rocket with ST fairing and Fregat upper stage. The fairing is defining demanding restrictions to the P/H SVM for both Gaia and Eddington.

The use of a connecting cone between the Soyuz 1194 mm adapter and the SVM with the 2664 mm A5 I/F was envisaged. One of the major problems of this approach is the fact that it brings additional mass than cannot be jettisoned once in orbit.

2.2.3.2 Orbit and lifetime

	Planck	**Herschel**	**Eddington**
Orbit	Small Lissajou	Large Lissajou	Small Lissajou
Injection	Need for a demanding ΔV manoeuvre	Free injection	Need for a demanding ΔV manoeuvre
Orbit injection and eclipse avoidance	225 m/s	0 m/s	250 m/s
Nominal lifetime	21 months	3.5 years	5,25 years (2 years for astro- seismology 3 years for planets searching)
Including degradable items	2.5 years	6 years	?

Table 43: Comparison of P, H and Eddington orbit and lifetime

The first 2 years are dedicated to asteroseismology and the following 3 years to the search for Earth like planets by detecting transits.

2.2.3.3 Autonomy

Due to the orbit (3h ground contact per day) and in the view to achieve the maximum recurrence (OBMM size, data bite rate and FDIR functionalities), the autonomy requirements are the same for both P/H and Eddington.

2.2.3.4 General configuration

Eddington, as P/H S/C, is made of 2 modules:
- a PLM which provides a very stable thermal environment to the PLM
- a SVM which carries the warm units of the PL and provides all the required functionalities to the PLM (power, AOCS, thermal control, communications).

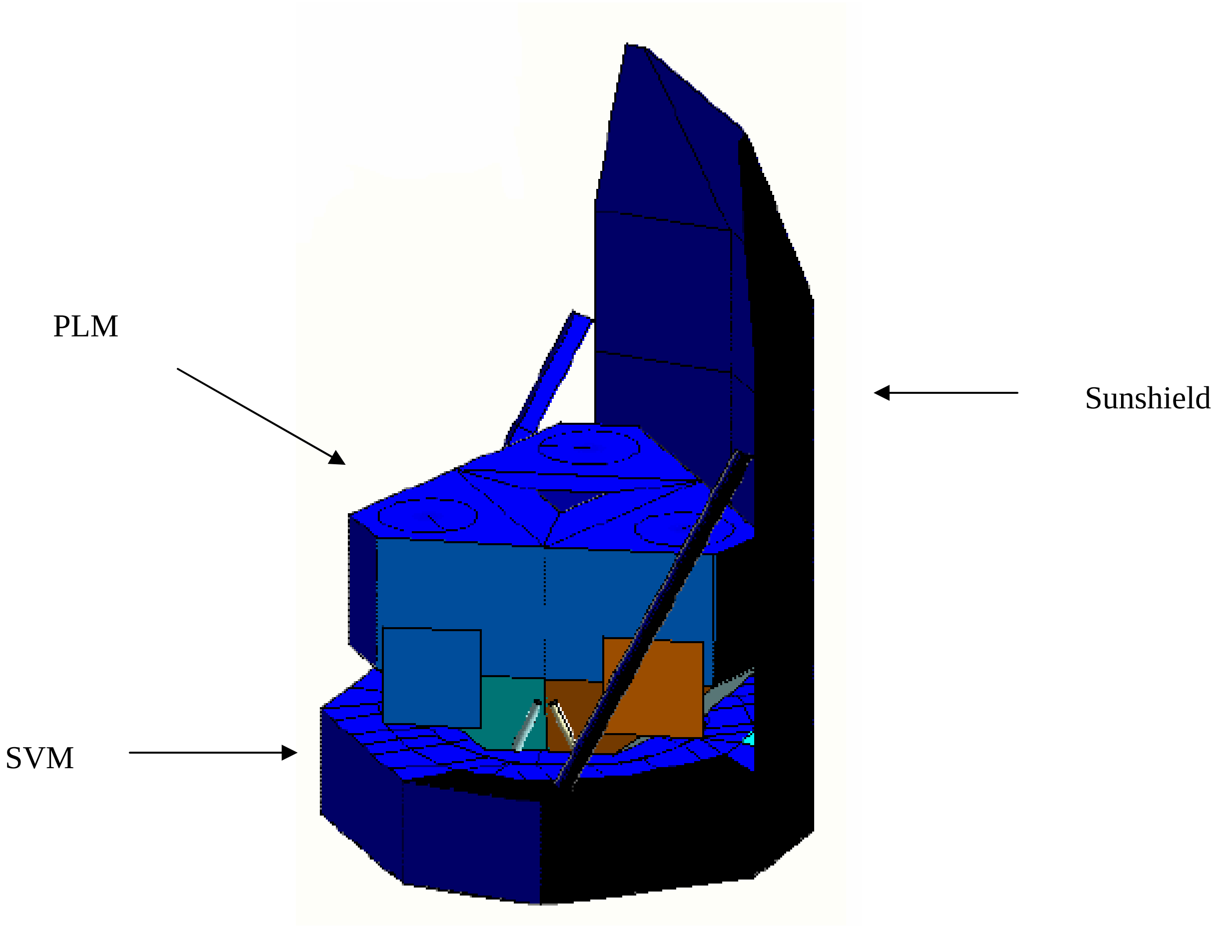


Figure 23: Eddington overview

2.2.3.5 Structure

The Eddington SVM is mostly H recurrent at different levels:
- same design: a primary structure composed of one central cone and an octagonal secondary structure
- same H SA concept : annular and central SA below the SVM
- same overall dimensions
- same material characteristics

	Herschel	Eddington
Structural concept	Same modular concept for the octagonal box (same elements with same functions)	
Material choice	Same material	
Adapter diameter	2624 mm	1194 mm
PLM accommodation	On the subplatform	3 bipods attached to central cone (as Gaia)
SA support	Same design on the top of SVM SA acting as a sunshield	

Table 44: Comparison of the structural design

The SVM houses the instruments control electronics and the bus electronics subsystems have the same allocations as P/H, the equipments panels for H P/L electronics are free for specific Eddington HW (more than sufficient surface and volume available).

A non compliance with the allowed fairing envelope was noticed: with the initial design, there is a 5.5 mm obstruction, hence the need for modification of the primary structure (modification of panel edge) or even potential modification (see the margins) of ST fairing.

	Herschel	Eddington
PLM mass	**2400**	**615**
ACMS	72	65.3
CDMS	15,5	16.8
Harness	85	106.9
PCS	36	39.1
RCS	58	62.3
SA	103	103
Structure	287	300.8
TCS	23	26.8
TT&C	23	23,3
Total SVM dry mass (separation system and system margin excluded)	648	627

Table 45: Comparison of the SVM mass budget (kg) between Herschel and Eddington

2.2.3.6 AOCS

2.2.3.6.1 General configuration

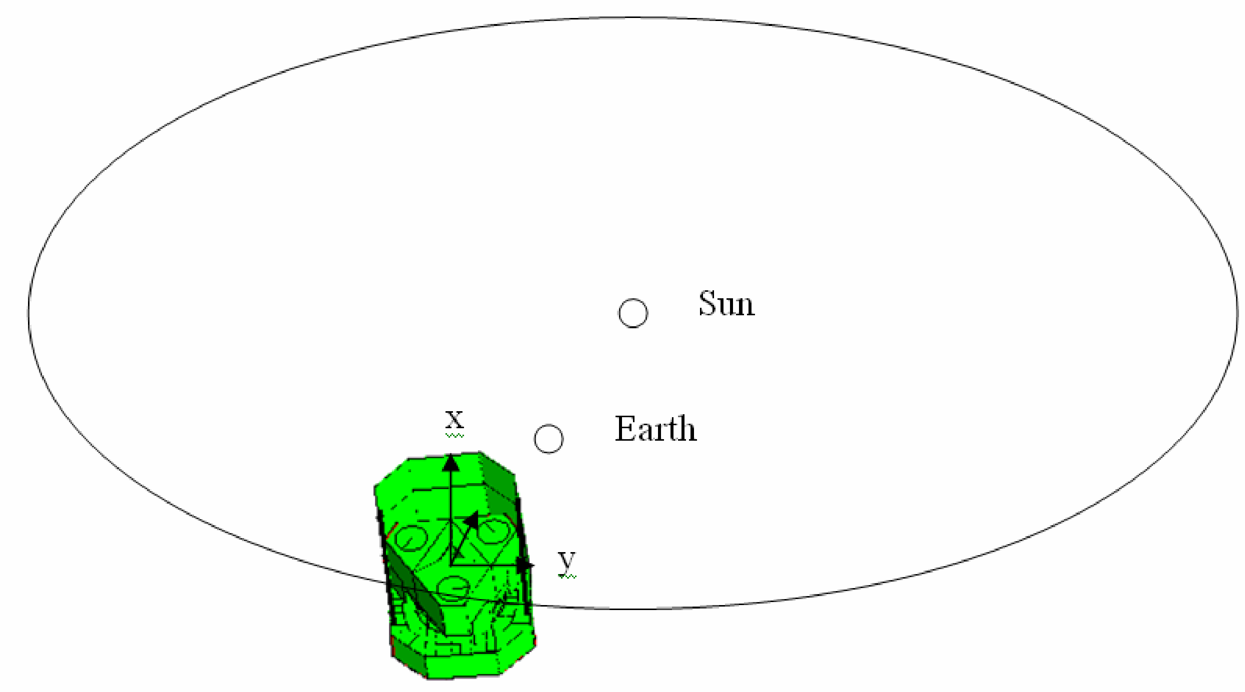


Figure 24: Eddington attitude configuration

As Herschel, Eddington is 3-axis stabilized and its nominal pointing is the following:
+Z axis of satellite (SA) is pointing to sun
+X axis is pointing north
+Y axis completes the coordinate system according to the right hand rule.

Both star observation and planet finding modes are submitted to limitations on SAA which are different from Herschel one's.

	Herschel	Eddington
About Y	+/-30°	+/- 45°
About X	+/-1°	+/-15°

Table 46: SAA comparison

Different SAA requirements (compared to Herschel) would have led to a different thermal analysis (through different worst and transient cases), different constraints on high data rate communications (need for a steerable antenna) and a different power generation assessment (due to solar rays' incidence).

2.2.3.6.2 Pointing requirements

	Herschel	Eddington
APE	0.24"	60"
RPE over 1 min	0.25"	0.15"
PDE	1.19" over 1 day	0.4" over 30 days

Table 47: Herschel and Eddington pointing requirements

APE specified for Eddington is 2 orders of magnitude less stringent than the one specified for Herschel. H ACMS is therefore able to meet this specification. On the contrary, the stability requirement is more stringent for Eddington and on top of taking into account RW micro vibrations, antenna pointing also adds some jitter. As a consequence, fine sensors would have to be mounted on the P/L and not on the SVM as for Herschel.

2.2.3.6.3 Sensors

	Herschel	Eddington
STR	2, with same location as Planck	
SAS	2, with different location	
CRS	2	
AAD	1 internally redundant unit	
GYRO	4 gyro in a tetrahedral configuration	

Table 48: Comparison of sensors HW

2.2.3.6.4 RCS

	Planck	Eddington
ΔV requirement (m/sec)	319.3 [4] 341 [15]	300
Propulsion type	Hydrazine monopropellant Nitrogen as pressurisant gas	
Propellant mass (kg)	346 kg	230 kg
Number of thrusters (w/o redundancy) and performance	Six 20 N Two 1 N	No data found
Tanks	3	No data found

Table 49: RCS comparison

2.2.3.7 Thermal control

Similarities		
Design principles		PLM SVM decoupling
Passive control	Insulation	MLI
	Radiation	Equipment panels act as radiators
Active control		Regulation with heaters, thermistors
Differences		
P/L thermal control		Less stringent thermal stability requirements for Eddington Passively cooled FPA for Eddington Actively cooled for Planck
SW development		SVM thermal analysis (a specific model is necessary to determine)
		Algorithm parameters for the thermal regulation.
Sizing		Position and number of heaters
		Size and position of MLI on external radiators.
AOCS specific		Needed power for thermal regulation and heaters according to the AOCS modes.
		Temperature of interfaces and equipments, according to the AOCS modes

Table 50: Comparison of Planck/Eddington thermal design

2.2.3.8 Communications

	P/H		Eddington
Frequency	X band		
Ground stations	New Norcia (nominal) Kourou (back up)		
Antennas	Planck	Herschel	2 LGA 1 steerable MGA
	3 fixed LGA	2 fixed LGA 1 fixed MGA	
	Different location		
Average science TM data rate	130 kbps (P)		
Modulation schemes	Identical		

Table 51: Comparison of RF architecture

The steerable antenna for Eddington requires a 2 DOF pointing mechanism. The major problem was the non compliance with the fairing envelope: the MGA protrudes through it, hence the idea to relocate the antenna at the centre of the panel, but with a modification of the radiation pattern.

TTC follows the same design rules for P/H and Eddington.

2.2.3.9 Power

2.2.3.9.1 Power generation

	Herschel	Eddington
PLM power	550 W	326 W
SVM power	533 W	505 W
S/C power	1083 W	831 W
SA topology	3 Body fixed SA on PLM serve as sun shield Different fixation locations of the struts	
Solar cell technology	Triple junction GaAs	
SA area	11.34 m^2 total (*)	14 m^2, 11.71 m^2 effective
Power provided by SA (EOL)	No data found	1170 W

(*) according to the P/H system engineer

Table 52: Power budget and power generation design

2.2.3.9.2 Power storage

As Planck, Eddington power storage is performed by Li Ion battery for LEOP.

2.2.3.9.3 Power distribution and regulation

	P/H	Eddington
Type of regulation	Direct Energy Transfer via S3R concept	
Type of power bus	28 V fully regulated	
Battery charge concept	BCR via 3 sections	
Battery discharge concept	BDR	
PCDU	Same functional architecture	

Table 53: Comparison of the power distribution and regulation concept

2.2.3.10 On board data management

The Eddington avionics is based on the H/P one composed of 2 computers: the Command and Data Management Unit (CDMU) and the ACC (Attitude Control Computer), with the same functional SW breakdown. As it is the case for the adaptation of Gaia, ACC ASW has to be modified due to different control laws (and also due to MGA articulation).

Notice that for SDE, Eddington could reuse H simulation model for AOCS.

	P/H	Eddington
Science data rate	130 Kbps avg (P)	
MM size (EOL)	25 Gb	
OBC	CDMU + ACC	
Processor type	ERC 32	
Data bus	Standard MIL 1553	
P/L I/F	WU for each instrument	One ICS for P/L

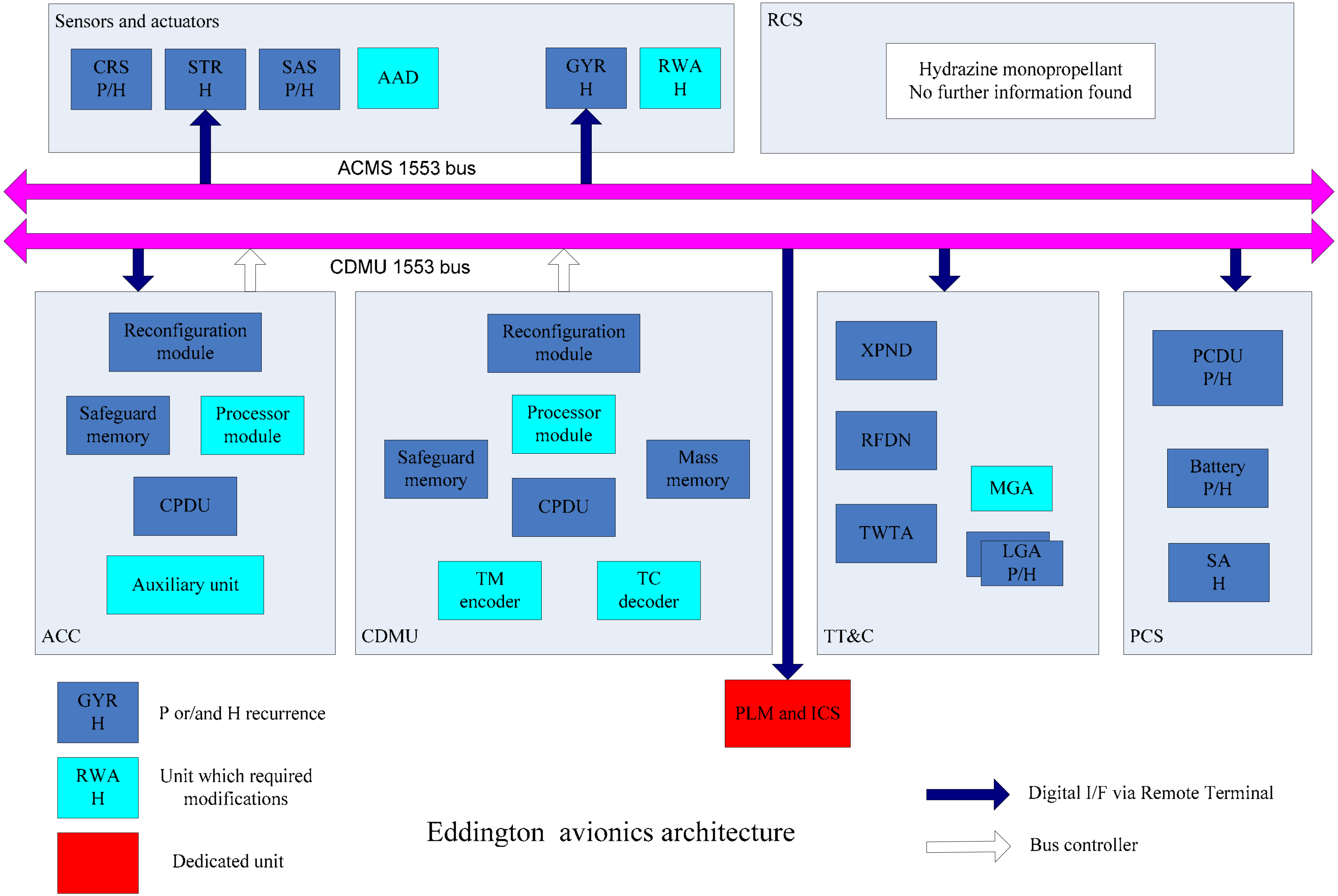
Table 54: CDH characteristics

The Instrument control system (ICS) is the I/F between the P/L and SVM with a combination of HW and embedded SW. Its main goal is to control, support and monitor the P/L by centralizing the functions needed for the different instruments.

2.2.4 SYNTHESIS

	P/H	Eddington	Impacts
Launch	A 5 ECA dual passenger	Soyuz Fregat single passenger	Max launchable mass
Internal fairing diameter	4200 mm	3800 mm	No compliance: 5.5 mm protrusion
Adapter diameter	2624 mm	1666 mm	I/F redesign
RPE	H: 0.25”	0.15”	RW dumping
Nominal lifetime	3.5 years (H)	5.25 years	
Orbit	Large Lissajou orbit (H)		Tanks sizing
SAA	About Y: +/-30° (H)	+/- 45°	Sensor locations Thermal analysis
	About X: +/-1° (H)	+/-15°	
Propellant mass	346 kg (P)	230 kg	Tanks sizing
P/L power	550 W (H)	326 W	SA sizing

Table 55: Comparison of the specifications and impacts on the design



Eddington avionics architecture

	Adaptations
Structure	Modification for different adapter
AOCS (w.r.t. H)	New smaller and less noisy RW (demanding RPE) Different sensors locations
Thermal control	Different sizing
Communication	Different location of the antenna Steerable MGA
Power	Different fixation locations of the SA struts fixations
OBDM	ICS: partial delocalisation of the CDMU functionalities

Table 56: SVM adaptations for Eddington mission

3 HOW FLEXIBILITY CAN BE ENSURED?

Following the review of the study cases presented in part 2, it is interesting to verify how the design principles discussed in part 1 were applied. This verification is performed in a synthetic manner, as illustrated in the next tables.

Subsystems	Main parameters	Approach to flexibility	P/H case and adaptations for Gaia and Eddington
Structure	Strength, frequency	Increase strength or frequency Choice of the material	Reinforcements on P subP/F
	Mechanical and thermo-elastic stability	Increase efficiency in stability management: identification of the parts where stability is required	STR accommodation Mechanical decoupling between PLM and SVM
	Launch vehicle I/F	Able to support different adapter diameter	2624 mm for P/H 1666 mm for Gaia 1194 mm for Eddington
	Configuration	Modularity Over-dimensioned S/C Large volume allocation for P/L	Modular concept Different shape and size of equipment panels with dedicated cut out and inserts for units accommodation with local reinforcements able to support different panel shape and size Surface available on equipment panels large enough for Gaia
Power	Generation	Allow flexible SA size, shape and location Allow SA over sizing	P similar to Gaia (but only reuse of bare cell), H similar to Eddington Thanks to S3R concept (P, H, Eddington)
	Storage	Allow for changing number and capacity of the batteries Allow for different type of charge and discharge management	Growth potential and low effect in case of single cell failure of the Li Ion type Same concept for each mission
	Distribution and regulation	Versatile PCDU (high number of power and heater lines) Standard I/F	Customisation on distribution lines is performed at harness level thus not affecting the PCDU Standard fully regulated 28 V power bus

Subsystems	Main parameters	Approach to flexibility	P/H case and adaptations for Gaia and Eddington
Environment protection and control	Temperature	Decoupling Choice of MLI material and MLI thickness Variable size position of radiators, heaters Variable performance of heat pipes Mounting of equipment directly on radiators Allow for large heating power	PLM/SVM, parts which require high thermal stability/rest Use of low conductivity cleats Same type of thermal control HW, different sizing Only on Planck (constant conductivity) WU mounted on equipment panels which act as radiators
	Radiation hardness	Able to support different radiation environment	H = worst sizing case
AOCS	Sensors selection	Modular approach Allow different location Allow different sensor performance	AAD with adjustable FOV Yes, thanks to modular structural design Different embedded SW for STR
	Actuators selection	Modular approach Allow different type of actuators	Capacity of growth of the tanks filling (20 % for P/H) Different localization of the thrusters thanks to the structure FEEPS on Gaia : I/F with the 1553 bus to keep maximum commonality of the ACC Need to change H RWA for Eddington

Subsystems	Main parameters	Approach to flexibility	P/H case and adaptations for Gaia and Eddington
TM and TC	Frequency of communications	Compatibility with several ground stations Large on board data storage and processing	New Norcia and Kourou for all 25 Gb MM for P/H, Eddington 400 Gb for Gaia with numerous modifications
	Data rate	Allow different type and number of antenna Allow different type of modulation schemes Allow different data rate	Flexibility ensured by RFDN thanks to the use of couplers Different antenna location Modification performed on TM modulators Same date rate
OBDH	Architecture	Modularity Standard data bus	2 OBC: one CDMU for "common" functions one ACC for specific AOCS Use of MIL 1553 data bus
	Processor	Processor able to implement open standards	Use of ERC 32 for all
	SW	Allow different AOCS modes, different FDIR functionalities	Flexible ASW, reuse of BSW Dedicated algorithms

CONCLUSION

This report provides an overview on reusable spacecraft platforms, with specific emphasis on future space science missions and on the European space industry. The document includes the analysis of the requirements driving the design of re-usable service modules, with specific attention to the needs of science missions. Programmatic considerations are also taken into consideration, highlighting the conditions that favour the application of re-usable platforms. The Herschel and Planck missions, together with potential solutions proposed for Gaia and Eddington, were analysed as study cases, representing a valid example of service module re-use. This analysis has included both technical as well as programmatic aspects, reflecting the work done in the first part of the activity.

The present review, although representing a preliminary analysis to be continued with additional work, has already allowed to identify the main benefits, drawbacks and pre-requisites applicable to the re-use of service modules for future space science missions.

On this basis, it has been showed as the adoption of a common platform to a number of different missions sharing similar mission profiles can lead to a significant reduction in the overall programme risk and therefore in the total cost. More specifically, the use of recurring spacecraft buses would allow shifting the emphasis and effort on the definition and development of the Payload Module and on any required adaptations, considerably cutting the total development time and thus leading to a faster ‘science return’. On the drawbacks side, we should quote the limitations induced by the adoption of a standard platform design (sometimes sub-optimal) and by the ageing of the equipment embarked on the bus.

Reuse of existing service modules can be envisaged in particular for astrophysics missions with similar mission profiles, thus minimising the need for adaptations and taking advantage of similar orbit design, launch and environmental conditions. This approach is also well matched to the large size and complexity of the payload required by the future astrophysics missions, with a clear separation between spacecraft bus and payload module.

In the case of planetary missions, the concept of service module (as opposed to payload module) is somehow less relevant, given that usually the instruments are located inside the S/C. Nevertheless, the re-use of the spacecraft carrier, can be envisaged in a number of missions with very similar environmental and escape velocity conditions, as recently demonstrated by Mars and Venus Express.

The availability of commercial products developed by the European industry should also be taken into consideration, in particular for smaller class missions (< 1 ton – LEO), certainly of interest to ‘niche-science’ programmes and of potential application in the context of in-orbit technology demonstration activities.

Additional analysis work on this subject would be beneficial, with specific attention to the possibility to use recurring service modules to future ESA science missions, as potentially emerging from the Cosmic Vision 2015-2025 process.

ANNEX: REVIEW OF EXISTING PLATFORMS

Name	Project initiator	Manufacturer	Application	Orbit	mass	Power
Smartbus		AeroAstro	military	LEO	nanosat	
MicroObservatory		AeroAstro	EO	LEO	150 kg SC mass	270 W EOL
NanoObservatory		AeroAstro	EO	LEO	up to 30 kg total mass	44.7 W avg (payload)
Spacebus 1000 (formerly called 100)		Alcatel Alenia Space	Telecom	GEO	130 kg (payload)	2 kW EOL
Spacebus 200		Alcatel Alenia Space	Telecom	GEO		
Spacebus 300		Alcatel Alenia Space	Telecom	GEO	250 kg (payload)	3 kW EOL
Spacebus 400		Alcatel Alenia Space	Telecom	GEO		
Spacebus 2000		Alcatel Alenia Space	Telecom	GEO	1890 kg	3.5 kW
Spacebus 3000 B or C 1 to 4		Alcatel Alenia Space	Telecom	GEO	500 kg (payload)	6.5 kW (payload), up to 13 kW supply
Spacebus 4000 B or C 1 to 4		Alcatel Alenia Space	Telecom	GEO	1000 kg (payload)	11.5 kW (payload), 15 kW supply EOL
Proteus (Plate-forme Reconfigurable pour l'Observation, les Télécommunications et les Usages Scientifiques)	CNES	Alcatel Alenia Space	Science, EO, telecom	LEO	500 kg/700 kg (SC)	200-300 W

PRIMA (Piattaforma Riconfigurabile Italiana Multi-Applicazione)	ASI	Alcatel Alenia Space	Science, EO, (Telecom, navigation)	LEO, MEO (GEO)	400-1500 kg (wet mass), 300-700 kg (payload)	250-800 W avg for payload
GeoBus (Italsat Bus)		Alcatel Alenia Space	Telecom			
BCP 600 (Ball Common Platform)		Ball Aerospace	EO	LEO	up to 90 kg (payload)	125 W avg
BCP 1000		Ball Aerospace	EO	LEO		
BCP 2000 (+ RS 2000 variant)		Ball Aerospace	EO	LEO	up to 380 kg (payload)	730 W avg
BCP 3000		Ball Aerospace	EO	LEO		
BCP 4000		Ball Aerospace	EO (SAR applications)	LEO	up to 1400 kg (payload)	1250 W avg
BCP 5000		Ball Aerospace	EO (optical and SAR remote-sensing payloads)	LEO		
Micro Mission Spacecraft		Ball Aerospace	Science	GEO, Lagrange points, Interplanetary (0.7 to 1.7 AU)	up to 45 kg (payload), launch wet mass: up to 242 kg	12 W avg (payload)
RS 300		Ball Aerospace		LEO	150 kg (payload)	120 W (payload)
Ellipso		Boeing	Telecom	big LEO		
ECS (OTS)		British Aerospace	Telecom	GEO		1260 W supply
MITA (Minisatellite Italiano a	ASI	Carlo Gavazzi Space	Science, EO, validation new technologies,	LEO	150-350 kg (wet mass)	85 W avg

Tecnologia Avanzata)			(Telecom)			
CAST 968		CAST	Science, EO, telecom, validation of new technologies	LEO SSO	approx. 300 kg (payload)	300-600 W supply
Phoenix Eye 1		CAST	EO	LEO	500 kg (payload)	700 W (payload)
Phoenix Eye 2		CAST	EO	LEO SSO	1200 kg (payload)	1700 W (payload)
DFH 1		CAST	Telecom	LEO		
DFH 2 and 2A		CAST	Telecom (defense)	GEO	900 kg (launch mass)	
DFH 3		CAST	Telecom, Navigation (Deep Space probe)	GEO	230 kg (payload)	1000 W (payload)
DFH 3A		CAST	Telecom	GEO	360 kg (payload)	2500 W (payload)
DFH 4		CAST	Telecom (defense)	GEO	600 kg (payload)	8000 W (payload)
CAST 2000		CAST	EO, Telecom, navigation, validation of new technologies	LEO, MEO, HEO	400 kg (payload)	900 W supply (EOL)
CAST Mini		CAST	Science, validation of new technologies, formation flying	LEO	50 to 120 kg wet SC mass	200 W supply
MiniFlex		EADS Astrium	EO, Science		80-250 kg (wet mass)	40 W avg (payload for a 100 kg SC
Flexbus		EADS Astrium	EO, Science	LEO	100 kg (payload)	100 W avg
Leostar 200 (1.2 m fairing)		EADS Astrium	EO, Science, Telecom	LEO	100-400 kg (35) 200-500 kg (for payload)	250 W
Leostar 500 (2 m		EADS Astrium	EO, Science, Telecom	LEO	500-1000 kg (payload)	

fairing)						
Leostar 500 XO (specific configuration?)		EADS Astrium	EO, Science, Telecom	LEO	500-1000 kg (SC)	250 - 750 W
Eurostar 1000		EADS Astrium	Telecom	GEO	up to 2000 kg (payload)	1 to 2 kW
Eurostar 2000		EADS Astrium	Telecom	GEO	400 kg (payload)	2 to 4 kW for payload
Eurostar 2000+		EADS Astrium	Telecom	GEO	550 kg (payload)	4 to 7 kW
Eurostar 3000		EADS Astrium	Telecom	GEO	1200 kg (payload)	> 10 kW
Eurostar 3000 GM		EADS Astrium	Telecom	GEO		
Eurostar 3000 S		EADS Astrium	Telecom	GEO		
Polar Platform (also called Spot Mk 1, 2, 3)		EADS Astrium	EO	LEO SSO	2000 (ERS 1) to 4000 kg (Metop)	550 W avg (ERS 1 payload) to 1800 W EOL (Metop supply?)
Myriade	CNES	EADS Astrium, ASPI	Science, EO, service demonstration for telecom, validation of new technologies		100 - 150 kg (SC)	
Alphabus		EADS Astrium/ Alcatel Alenia Space	Telecom	GEO	up to 1200 kg (payload)	up to 18 kW (payload
HS 301		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 303		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 306		Hughes Space and	Telecom	GEO		

		Communication		(spin stabilized)		
HS 308		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 312		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 331		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 333		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 335		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 351		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 353		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 356		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 371		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 373		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 376 (376 L, 376 W, 376 HP)		Hughes Space and Communication	Telecom	GEO (spin stabilized)		990 W supply
HS 378		Hughes Space and Communication	Telecom	GEO (spin stabilized)		

HS 381		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 389		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 393		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 401		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 507		Hughes Space and Communication	Telecom	GEO (spin stabilized)		
HS 601 (601 HP, 601 MEO) also called BSS 601		Hughes Space and Communication	Telecom	GEO		
HS 702 also called BSS 702		Hughes Space and Communication	Telecom	GEO		
HS Geosynchronous Mobile (GEM)		Hughes Space and Communication	Telecom	GEO		
Insat		ISRO	Telecom	GEO		
Blackbird 350		Kayser-Threde	Telecom	LEO	42 kg (total mass)	
Yakhta		Khrunichev Space Center	EO			
Yantar		Kozlov	military	LEO		
Navigator		Lavochkin Association	Science	interplanetary	757 kg (SC dry mass)	500 W (payload)

Arkon		Lavochkin Association	military, EO	LEO		
LM 100		Lockheed Martin	EO, Science	LEO, GEO, Moon	up to 24 kg (payload)	15 W avg
LM 700		Lockheed Martin	EO, Science	LEO, HEO, even interplanetary	up to 230 kg (payload)	500 W EOL
LM 900		Lockheed Martin	EO	LEO	up to 470 kg (payload) or 500 kg	300 W avg
LM A2100 (X, AX, , AX2)		Lockheed Martin	Telecom	GEO		up to 15 kW supply
LM 3000 or LM A3000		Lockheed Martin	Telecom	GEO		
LM 4000 or LM A4000		Lockheed Martin	Telecom	GEO		
LM 5000 or LM A5000		Lockheed Martin	Telecom	GEO		4850 W supply BOL
LM 7000 or LM A7000)		Lockheed Martin	Telecom	GEO		
DSCS		Lockheed Martin	Telecom (defence)			
Milstar		Lockheed Martin	Telecom (defence)			
TIROS N		Lockheed Martin	EO			
LM (AS)1000		Lockheed Martin (Astro Space)	Telecom	GEO	463 kg total mass	
LM (AS)2100		Lockheed Martin (Astro Space)	Telecom	GEO	2760 kg total mass	up to 15 kW

LM (AS)3000		Lockheed Martin (Astro Space)	Telecom	GEO	600 kg total mass	2800 W
LM (AS)4000		Lockheed Martin (Astro Space)	Telecom	GEO	1021 kg total mass	2800 W
LM (AS)5000		Lockheed Martin (Astro Space)	Telecom	GEO	2862 kg total mass	4850 W BOL
LM (AS)7000		Lockheed Martin (Astro Space)	Telecom	GEO	3415 kg total mass	
MegSat		Meggiorin Group	Science, EO	LEO	34 (M0) -55 kg (M1) (launch wet mass)	25 W avg
Spartan 400	NASA/GFSC		Science	LEO from STS	up to 1362 kg (payload)	250 W (payload)
Spartan 250	NASA/GFSC		Science	LEO from STS	up to 450 kg (payload)	100 W avg (payload)
Spartan Lite	NASA/GFSC		Science	LEO from STS	50 kg	40 W (payload)
AB 940		Northrop Grumman Space Technology (ex TRW)				
T330 (AB 1200)		Northrop Grumman Space Technology (ex TRW)	EO			
T100		Northrop Grumman Space Technology (ex TRW)		LEO	36 kg (payload)	25 W for payload
T200 A		Northrop Grumman Space Technology (ex TRW)		LEO	75 kg (payload)	72 W (apyload)

T200 B		Northrop Grumman Space Technology (ex TRW)		LEO (GEO)	95 kg (payload)	175 W (payload)
T310		Northrop Grumman Space Technology (ex TRW)		GEO (LEO)	up to 267 kg (payload)	up to 560 W EOL (payload)
TRW SSTI (+2 standard versions)		Northrop Grumman Space Technology (ex TRW)		LEO (GEO)	up to 125 kg (payload)	175 W avg (payload)
Gorizont		NPO PM	Telecom	GEO	approx. 2200 kg (launch mass)	
Express		NPO PM	Telecom	GEO		
Ekran		NPO PM	Telecom	GEO		
LEOstar		Orbital Sciences	Science, EO, validation of new technologies	LEO	up to 100 kg (payload)	110 W avg EOL (payload)
LEOstar 2		Orbital Sciences	Science, EO, validation of new technologies	LEO	up to 210 kg (payload)	118 W avg (payload)
Picostar		Orbital Sciences		LEO (spin stabilized)	20 kg (payload)	10 W avg (payload)
Pegastar		Orbital Sciences	EO, science	LEO	570 kg (payload)	60 W avg BOL (payload)
Midstar		Orbital Sciences	Science, EO	LEO	up to 780 kg (payload)	323 W avg (payload)
Microstar		Orbital Sciences	Science, validation of new technologies	LEO	up to 58.6 kg (payload)	50 W avg (payload)
Star 1 and 2		Orbital Sciences	Telecom (EO, Science, validation of new	MEO, GEO	up to 200 kg (payload)	up to 555 W (payload)

			technologies)			
MiniStar		Orbital Sciences	Science, validation of new technologies	LEO	up to 25 kg (payload)	up to 135 W (payload)
Meteor		Research and Production Enterprise Pan-Russian Research Institute for Electromechanics (NPP VNIIEM)	EO (Meteorology)	LEO		
Yamal 100		RSC Energia	Telecom	GEO	up to 1340 kg (payload)	up to 1200 W (payload)
Yamal 200		RSC Energia	Telecom	GEO	up to 1340 kg (payload)	up to 1200 W (payload)
Yamal 300		RSC Energia	Telecom	GEO	up to 1340 kg (payload)	up to 1200 W (payload)
MicroSIL G		Space Innovations Limited (SIL) now Spacedev (US)	EO	LEO	13.2 kg (payload)	20 W avg (payload)
MicroSIL S		Space Innovations Limited (SIL) now Spacedev (US)	EO	LEO	18 kg (payload)	20 W avg (payload)
MiniSIL L		Space Innovations Limited (SIL) now Spacedev (US)	EO	LEO	64-123 kg (payload)	80 W avg (payload)
MiniSIL 2L		Space Innovations Limited (SIL) now Spacedev (US)	EO	LEO	90-180 kg (payload)	200 W (payload)
MiniSIL P		Space Innovations Limited (SIL) now Spacedev (US)	EO	LEO	36-50 kg (payload)	55 W avg (payload)

MiniSIL 2P		Space Innovations Limited (SIL) now Spacedev (US)	EO	LEO	64-82 kg (payload)	200 W (payload)
SSL 1300 or LS 1300 (1300E, 1300HL, 1300S, 1300X)		Space Systems Loral	Telecom	GEO	up to 6700 kg launch mass	from 12 to 25 kW supply
LS 400		Space Systems Loral		LEO, GEO	up to 350 kg (payload)	1300 W (peak)
LS 2020 (or 20.20)		Space Systems Loral	Telecom	GEO		17 to 30 kW (supply)
Amos		Spacecom	Telecom	GEO		
Surrey Interplanetary Platform		SSTL	Science		20 kg payload mass to Mars or Venus orbit	
SNAP		SSTL	remote-inspection and formation flying missions	LEO	up to 3-4 kg (payload)	2.5 W avg (payload)
Minisat 400		SSTL	Earth Observation, communications and technology demonstration	LEO	up to 200 kg (payload)	100 W avg (payload)
MicroSat 100		SSTL	EO	LEO	up to 40 kg (payload)	
MicroSat 70		SSTL	EO, communications and technology demonstration	LEO	up to 23.8 kg (payload)	
Constella		SSTL	EO, Telecom, Navigation	LEO	10 to 60 kg (payload)	70 to 100 W supply

Surrey Lunar Microsatellite (Moonshine)		SSTL	Science	NEO	20 to 70 kg payload	
EO-SB (Earth Observer-Spacecraft Bus)		Swales Aerospace	EO	LEO	up to 236 kg (payload)	256 W EOL
Multi mission microsat (M3sat) Type A		Swales Aerospace		LEO, MEO, GEO	up to 200 kg (payload)	700 W (payload)
Multi mission microsat (M3sat) Type B		Swales Aerospace		LEO, MEO, GEO	up to 80 kg (payload)	150 W (payload)
Multi mission microsat (M3sat) Type C		Swales Aerospace		LEO, MEO, GEO	up to 25 kg (payload)	50 W (payload)
SMEX-Lite		Swales Aerospace	Science, validation of new technologies	LEO up to L points	215 kg (bus), 615 kg (total), 400 kg (payload)	159 W avg supply, 289 W avg (payload)
SA 200 B		Swales Aerospace	Science, EO, validation of new technologies	LEO, MEO, HEO, GEO, planetary	125-200 kg (launch mass) up to 100 kg (payload)	300 500 W BOL, 75 150 W avg-peak (payload)
SA 200 S		Swales Aerospace	Science, validation of new technologies	LEO, MEO, GEO	up to 200 kg (payload)	66-225 W avg-peak (payload)
SA 200 HP (High Performance)		Swales Aerospace	EO, Science	LEO, MEO, HEO, GEO, planetary	up to 800 kg (payload), 354 kg (wet bus mass)	650 W avg (payload), 2000 W supply EOL, 1 UA
SA 200 GL		Swales Aerospace				
SA 200 GM		Swales Aerospace				
Freja-C		Swedish Space	Science	LEO	less than 30 kg total mass	

		Corporation				
Tubsat		Technical University of Berlin	EO, validation of new technologies	LEO	approx 40 kg (launch wet mass)	
Lybid		Yuzhnoye	Telecom	GEO	163 kg (payload)	2894 W (repaeter consumption)
Prognoz		Yuzhnoye	Science	Earth centered, highly excentric		
AUOS (Z, SM)		Yuzhnoye	Science		310 kg (payload)	50 W (supply)
SAR-sat		OHB system	EO	LEO	Up to 40 kg (P/L)	2900 W peak power for P/L

LIST OF FIGURES AND TABLES