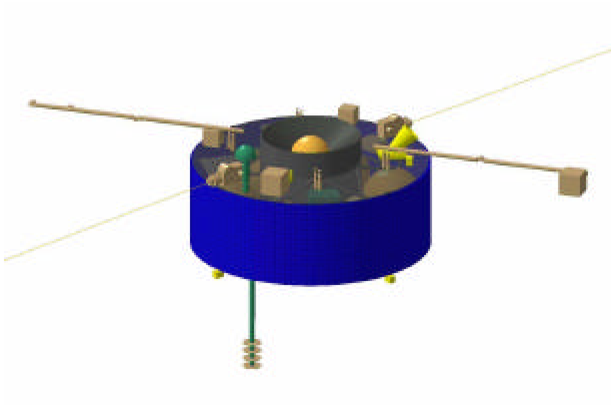
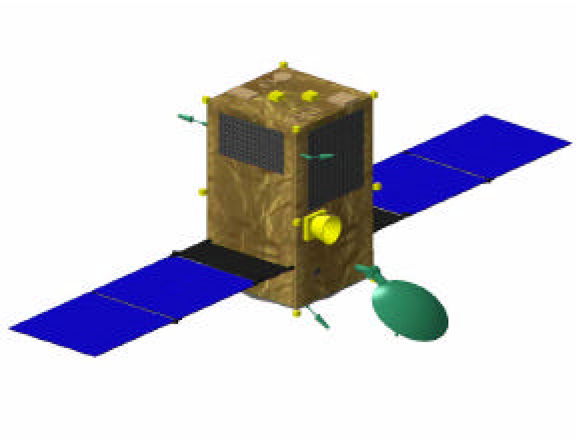
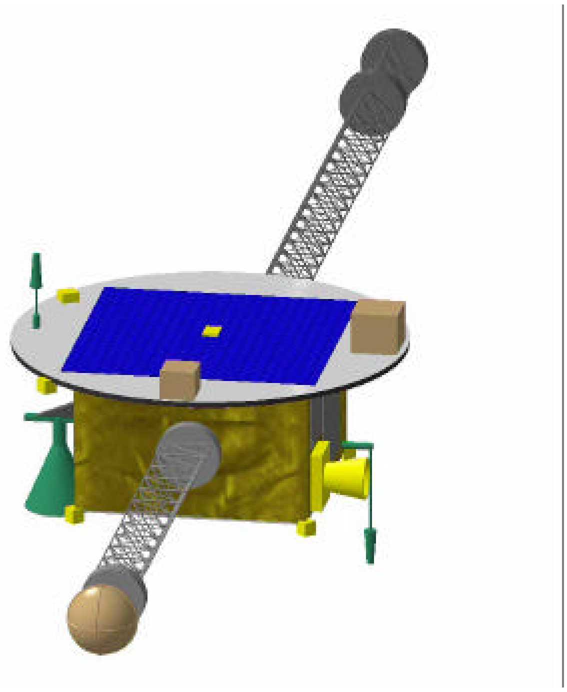
CDF Study Report

Space Weather







FRONT COVER

The front cover gives an impression of how the three spacecraft designed during the the Space Weather study will look in flight.

The inside cover shows the three designs in their deployed configurations.

STUDY TEAM

This Study was performed in the ESTEC Concurrent Design Facility (CDF) by the following interdisciplinary team:

Team Leader	A Santovincenzo	TOC-MCT			
AOCS	G Bolognese	TOS-ESC	Power	P Rueda	TOS-EPS
Communications	E Daganzo	TOS-ETT	Programmatics/AIV	M Braghin	TOS-MTC
Configuration	S Mangunsong	TOS-MCS	Propulsion	R Schonenborg	TOS-MPC
Cost	T Bieler	IMT-ICE	Pyrotechnics	N Cable	TOS-MCS
	H Joumier	IMT-ICE	Radiation	H Evans	TOS-EMA
Data Handling	C Monteleone	TOS-ESD	Risk	L Oliefka	TOS-QQD
Ground Systems & Operations	K Nergaard	TOS-OSF	Simulation	L Dwedari	TOS-EMM
	JL Pellón-Bailón	TOS-OSF	Structures	D de Wilde	TOS-MCS
Instruments	A Gálvez	TOS-PID	Systems	R Biesbroek	TOS-PID
Mechanisms	L Gaillard	TOS-MMM		JF Martín-Albo	TOS-PID
Mission	G Janin	TOS-GMA	Thermal	H Ritter	TOS-MCT

Study Manager	E Daly	TOS-EMA
Scientific Coordinator	A Hilgers	TOS-EMA
Scientific Advisor	A Glover	TOS-EMA
Report Editor	K Fletcher	TOS-PID

The study team would like to thank the following for their kind contributions:

D Berghmans (SCI-SO)
L Eliasson (IRFK, Sweden)
P Hyvönen (Orbitum, Sweden)
H Laakso (SCI-SO)
A Masson (SCI-SO)
P Nieminen (TOS-EMA)
P Wintoft (IRFL, Sweden)

A Coates (MSSL, UK)
S Eckersley (Astrium, UK)
M Hapgood (RAL, UK)
B Huet (Alcatel, France)
F Lefeuvre (LPCE, France)
M Pick (Meudon Obs., France)
O Poinsard (Alcatel, France)
F Primdahl (DSRI, Denmark)

Further information and/or additional copies of this report are available from:

E Daly
ESA/ESTEC/TOS-EMA
Postbus 299
2200 AG Noordwijk
The Netherlands
Tel: +(31)-71-5653828
Fax: +(31)-71-5654696
Eamonn.Daly@esa.int

For further information on the Concurrent Design Facility please contact:

M Bandecchi
ESA/ESTEC/TOS-PID
Postbus 299
2200 AG Noordwijk
The Netherlands
Tel: +(31)-71-5653701
Fax: +(31)-71-5656024
Massimo.Bandecchi@esa.int

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

This page intentionally left blank

1. Introduction

In carrying out this space weather study, the Concurrent Design Facility (CDF) has been used to consider possible elements of a future space segment for an operational, service-oriented, European space weather system.

1.1 Background

ESA embarked on studies of space weather in 1999, under the General Studies Programme (GSP). Parallel contracts were placed with consortia consisting of engineering, science, and effects experts. These consortia were led by Alcatel Space and Rutherford Appleton Laboratory (RAL). The studies performed wide-ranging analyses of the need for a European space weather programme and the possible content of such a programme. The specific activities included:

- Analysis of space weather effects
- Analysis of requirements of a space weather system
- Definition of a service including prototyping of aspects of the services
- Definition of the space segment
- Analysis of programmatic and organisational issues

These studies were supported by a space weather working team, who provided inputs to the studies, analysed the work of the consortia, and advised ESA on the future strategy.

1.2 Objectives and Scope of the CDF Study

The two consortia discussed above have proposed many interesting options for a space weather system [RD1], [RD2]. It was decided that some of the proposed options should be analysed and further developed by ESA through the CDF, to establish the feasibility and cost of the various options.

While a comprehensive space weather system should include elements beyond the ones studied, it was decided on the advice of the consortia to study the following three demonstrator elements:

- An element for continuous monitoring of solar features which are important in ultimately causing space weather hazards near and on the Earth
- An element for continuously monitoring the solar wind upstream of the Earth
- A fleet of inner magnetospheric monitoring spacecraft which would observe changes taking place within the terrestrial radiation belts, the magnetosphere and partly the ionosphere

For each of those three, the study team was requested to

- perform mission conceptual design and trade-offs
- prepare a preliminary spacecraft design including budgets, configuration, subsystem designs with required performance and sizing
- define space and ground operations and facilities
- define the programmatics of the mission development

- perform risk assessment and cost analysis

Important elements not studied include:

- An ionospheric/atmospheric monitoring function
- Hitch-hiker payloads or spin-off on other spacecraft (e.g. radiation or plasma monitors on science spacecraft, use of radio reception techniques for ionospheric monitoring)

1.3 Document Structure

For Space Weather, essentially three separate studies were performed within one study. To reflect this, this document is structured as follows:

- Chapter 1 gives the background to the whole study
- The Executive Summary, chapter 2, describes the flow of the whole study and gives a mission summary for each of the three elements
- Chapter 3, Mission Objectives, defines the customer expectations
- Chapter 4 gives an overview of the Space Weather Architecture

After that, each of the three element studies is described individually in full, including:

- System and subsystem designs (chapters 5, 6 and 7)
  Details of each domain addressed in the study are contained in the specific subsections.
- One chapter, chapter 8, describes the Ground System and Operations approach to the whole Space Weather study
- Chapter 9 describes the Simulation domain view of the study
- The report ends with a common Conclusions chapter, chapter 10, plus appendices (references and acronyms)

Note: Due to the different distribution requirements, only cost assumptions (excluding figures) are given in this report. The costing information is published in a separate document.

2. Executive Summary

2.1 Study Flow

The assessment study of a potential Space Weather programme using the ESA Concurrent Design Facility (CDF) was initiated by the Space Environment Effects and Analysis section of the Technical and Operations Support Directorate (D/TOS) of the European Space Agency.

The study was conducted from the kick-off on October 2nd 2001 through to the Internal Final Presentation on November 27th 2001. It involved 13 technical sessions of the interdisciplinary study team.

The original intention was to study each of the three elements in turn, but once the first element was studied it became clear that the mission, ground systems and operations of the three would be inter-related, particularly the second and third. As a result of this, an overall ground system and operations strategy was defined, which narrowed down the options to be discussed for the remaining two elements.

2.2 IMM Mission Summary

Mission Objective	Operational Space Weather programme • To provide near-real time monitoring of the Earth magnetosphere and particles	
Payload	The instruments: • Hi-Energy Particle Monitor (HEM) • Thermal Plasma Monitor (TPM) • Mid-Energy particle Monitor (MEM) • Magnetometer (MAG) • Waves Instrument (WAVE) • GPS receiver (GRIS)	
Launcher	• 4 satellites on one stack using Indian GSLV • Performance: 4050 kg to 200 km orbit	
Spacecraft		**Baseline**
	Design Lifetime	5 years
	Attitude control	Spin stabilised at 15 rpm Spin axis perpendicular to Earth equator
	Total mass	1004 kg
	Spacecraft main body dimensions	2200 mm diameter, 1050 mm height per S/C
	Pointing accuracy	1 arcsecond, 1σ
	Pointing stability	1 arcsecond over 40 seconds
	Pointing knowledge	0.25 arcsecond
	Solar array	GaAs, 1.4 m^2
	Power	28V fully regulated bus (PCU, PDU, TCU)
		Li-Ion battery 400 Wh
		282 W EOL
	Antennas (X-Band)	Toroidal antenna + 2 LGAs
	Data download	2 kbps (LGA) or 13 kbps (Toroidal)
Mission	**Orbit**	650 x 39717 12 hr orbit at 10° inclination
	Nominal mission period	5 years
	DV	2.5 km/s Apogee raise
		0.212 km/s Perigee raise & inclination change
Operations	**Ground stations**	Kourou & Perth
		LEOP using ESA LEOP ground-stations

Programmatics	**Phase A start**	2002
	Phase B start	2003
	Launch date	2007
	Model philosophy	STM, ATB & PFM
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability	0.9 after 3.9 years

2.3 SWM Mission Summary

Mission Objective	Operational Space Weather programme • To sample the local properties of the solar wind ahead of the Earth's magnetosphere	
Payload	The instruments: • Magnetometer (MAG) • Thermal Plasma Monitor (TPM) • Mid-Energy particle Monitor (MEM) • Low-frequency Radio Spectrometer (CRS)	
Launcher	• Dual launch with SAM using direct injection towards L1 on the Soyuz-Fregat launcher • Performance: 1600 kg to direct L1 transfer • Option: dedicated launch on Rockot with STAR37FM	
Spacecraft		**Baseline**
	Design lifetime	5 years
	Attitude control	Spin stabilised at 15 rpm
	Total mass	208 kg
	Spacecraft main body dimensions	1600 x 1600 x 1000 mm
	Pointing accuracy	1 arcsecond, 1σ
	Pointing stability	1 arcsecond over 40 seconds
	Pointing knowledge	0.25 arcsecond
	Solar array	GaAs, 1.4 m^2
	Power	28V fully regulated bus (PCU, PDU, TCU)
		Li-Ion battery 400 Wh
		140 W EOL
	Antennas (X-Band)	MGA + 2 LGAs
	Data download	100 bps (LGA) or 9 kbps (MGA)
Mission	**Orbit**	L_1 halo orbit
	Nominal mission period	5 years
	DV	40 m/s launcher dispersion 5 m/s halo orbit insertion 10 m/s orbit maintenance for 5 years
Operations	**Ground stations**	Perth, Villafranca, Goldstone
		LEOP using ESA LEOP ground-stations

Programmatics	**Phase A start**	2002
	Phase B start	2003
	Launch date	2006
	Model philosophy	STM, ATB & PFM
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability	0.8 after 5 years

2.4 SAM Mission Summary

Mission Objective	Operational Space Weather programme • Near-continuous imaging of the Sun disc and its Corona	
Payload	The instruments: • EUV Imager (EUVI) • X-ray Photometer (XRP) • Cosmic Radiation Monitor (CRM) • Coronagraph (WLC)	
Launcher	• Dual launch with SWM using direct injection towards L1 on the Soyuz-Fregat launcher • Performance: 1600 kg to direct L1 transfer • Option: Dnepr-Varyag if available	
Spacecraft		**Baseline**
	Design lifetime	5 years
	Attitude control	3-axis stabilised
	Total mass	538 kg
	Spacecraft main body dimensions	1300 x 1300 x 1900 mm
	Pointing accuracy	7 arcsecond, 3σ
	Pointing stability	15 arcsecond over 15 min
	Pointing knowledge	1 arcsecond
	Solar array	2 wings of Si-BSR cells, 2.6 m^2 per wing
	Power	28V fully regulated bus (PCU, PDU, TCU)
		Li-Ion battery 400Wh
		488 W at EOL
	Antennas (X-Band)	HGA + 3 LGA's
	Data download	100 bps (LGA) or 35 kbps (HGA)
Mission	**Orbit**	L_1 halo orbit
	Nominal mission period	5 years
	D**V**	40 m/s launcher dispersion 5 m/s halo orbit insertion 10 m/s orbit maintenance for 5 years
Operations	**Ground stations**	Perth, Villafranca, Goldstone
		LEOP using ESA LEOP ground-stations

Programmatics	**Phase A start**	2002
	Phase B start	2003
	Launch date	2006
	Model philosophy	STM, EM & PFM
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability	0.8 after 5 years

3. Mission Objectives

3.1 What is Space Weather?

The US National Space Weather Programme has established a widely accepted definition of space weather as:

> "*Conditions on the sun and in the solar wind, magnetosphere, ionosphere, and thermosphere that can influence the performance and reliability of space-borne and ground-based technological systems and can endanger human life or health*"

This indicates that while it is concerned with phenomena occurring throughout the solar-terrestrial system which are of keen scientific interest, it is those aspects which ultimately lead to effects which are important for a service.

Naturally the high-profile effects are the most widely known. These include large-scale disruption to the north American power system during the March 1989 major geomagnetic storm; telecommunications satellite losses; and communication disruptions experienced during the Desert Storm military operations.

However, space weather effects are not always dramatic.. Space weather effects are wide ranging and much more frequent phenomena than the above would have us believe. In practice, these effects occur on a much more regular basis with smaller scale effects, such as temporary disruption to a single communications satellite. These smaller magnitude effects have an increased impact on society as we become increasingly reliant on space-based communications, and as satellite technology becomes more advanced, incorporating smaller and smaller components, which may be more susceptible to radiation damage in space. Services based on improved monitoring and simulation of the solar-terrestrial system and the resulting changes in radiation levels at satellite orbital altitudes might be a cost-effective contribution to solving some of the problems caused by space weather.

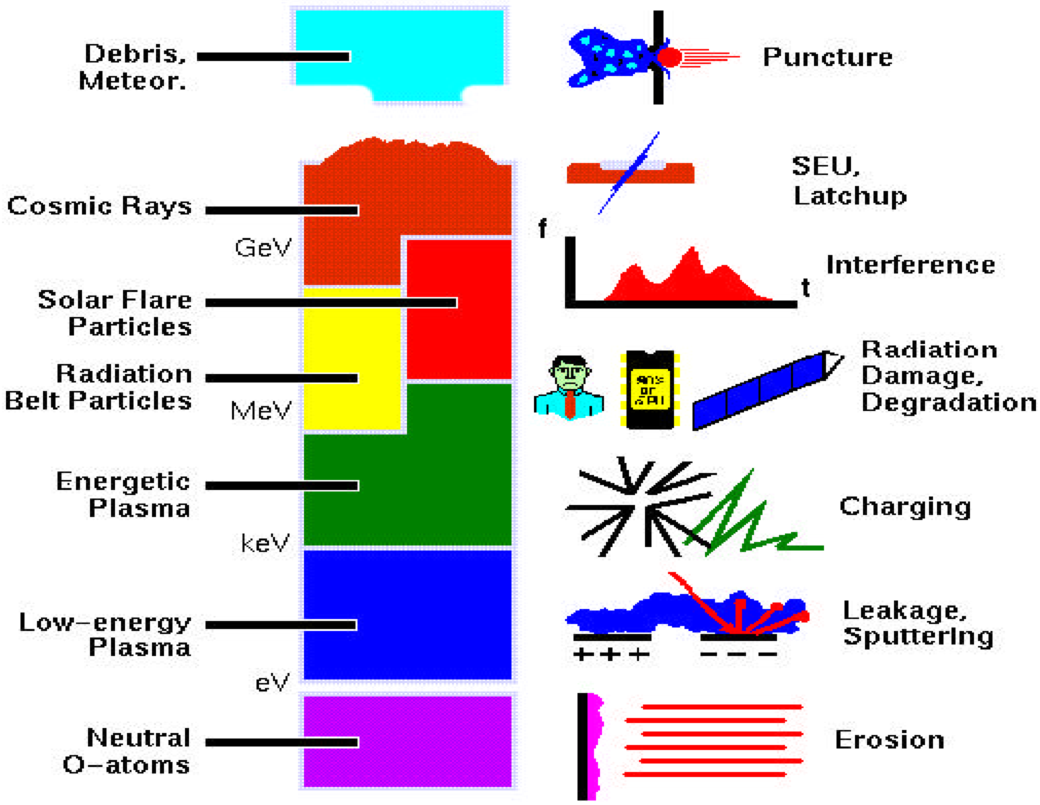


Figure 3-1: Space Weather Effects

The following provides a list of some of the main areas where space weather effects may critically disturb operations:

- *Terrestrial power distribution networks*
  Electricity grids are affected by very rapid large current-flows in cables induced by currents in the ionosphere. The current surges can destroy equipment hooked on to the grid, necessitate operational system reconfiguration, or require special designs.

- *Terrestrial Communications*
  Some terrestrial communication systems are seriously affected by changes in the structure of the ionosphere induced by space weather. Changes may affect the clarity of signal transmission, or information that the signals carry such as navigational position. Many of these systems are of military use.

- *Space-based radio services across the ionosphere*
  Radio propagation through the ionosphere can be perturbed by the influence of Space Weather. Ground-based communications and navigation services can be disrupted as well as radar-based remote sensing.

- *Oil and mineral prospecting and operations*
  Geomagnetic field variations caused by Space Weather effects can perturb magnetic readings routinely used by these industries. For example, oil drill heads navigate at the end of long, flexible pipes, with reference to the Earth's magnetic field. If the Earth's magnetic field is changed, due to a 'magnetic storm,' the heads will go off-track.

- *Defence*
  The defence sector makes increasing use of communications and navigation services, which are affected by Space Weather. Space systems are particularly important to this sector. As mentioned above, space weather effects were experienced during Desert Storm operations.

- *Airlines and aircraft*
  Advanced avionics systems are becoming susceptible to cosmic radiation hazards. Furthermore, aircrew are exposed to doses of cosmic radiation which European legislation now requires to be monitored as a potential health hazard, particularly as aircraft fly increasingly higher, above the shielding effects of the atmosphere.

- *Commercial space systems*
  Space systems are subject to numerous types of serious radiation damage and interference. Radiation hazards for astronauts are significant. The solar panels of spacecraft (from which they get their electricity supply) are degraded by space weather radiation. Spacecraft can also suffer electric discharge following plasma-induced charging, causing anomalies. Rapid atmospheric variations can affect spacecraft orbits and stabilisation through increased atmospheric drag.

Some of these effects are illustrated in Figure 3-2 below.

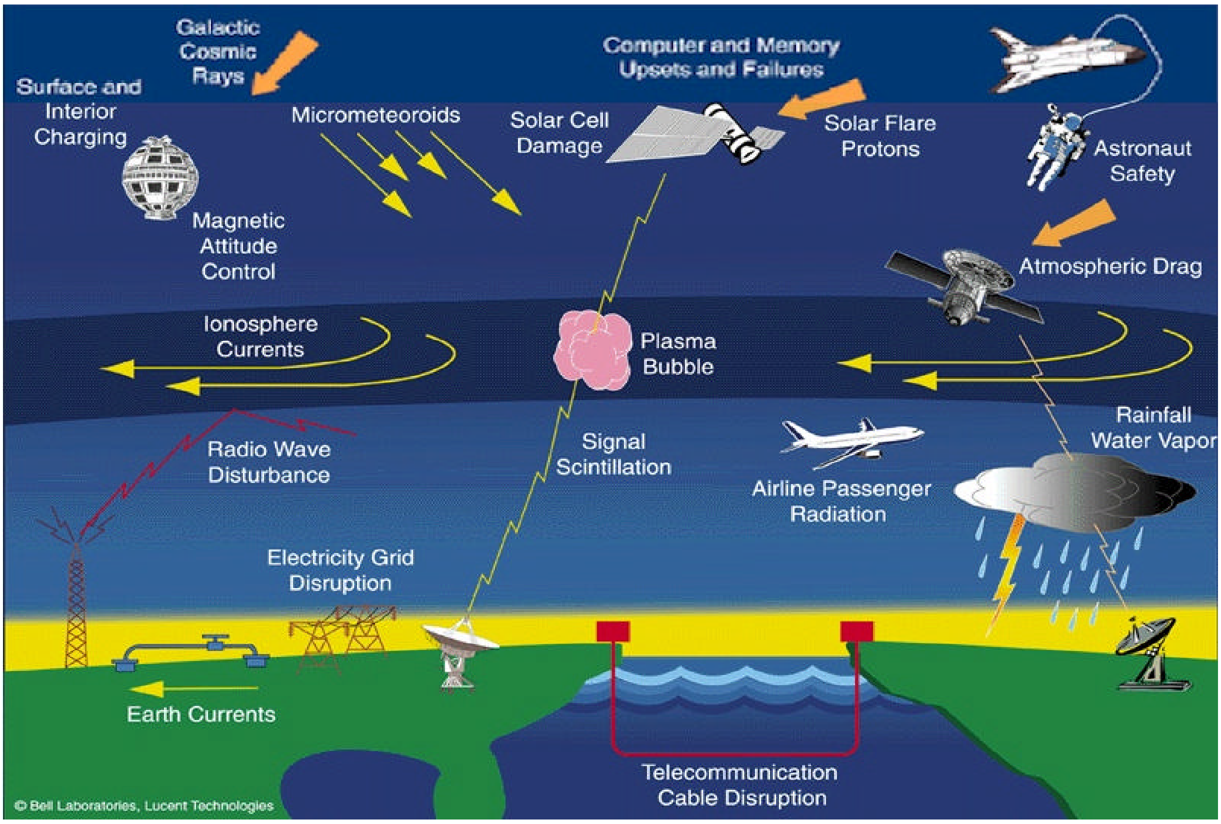


Figure 3-2: Space Weather Effects (Courtesy of Bell Laboratories, Lucent Technologies)

Identifiying the long term economic impact of space weather is difficult and requires good knowledge of the space environment whilst large losses caused by a specific high profile event are more easy to gauge. For example, because satellite problems in orbit are difficult to diagnose from the ground, there may have been cases where large-scale satellite failures are falsely attributed to space weather effects. Conversely, it is also highly probable that a large number of small anomalies due to space weather effects might go unnoticed.

For more information see [RD3].

3.2 Mission Justification: Science versus Service

Science requirements are **not** necessarily compatible with those of a service, since a monitoring service will require real-time or near-real-time data downlink, together with a negligible delay in data processing and supply of data to the relevant users.

Science-based missions will not necessarily be geared towards providing continuous coverage of the intended observation target. The NASA Solar Dynamics Observatory (SDO) provides an example of this type of mission. While it will provide observations of the sun in unrivalled detail, it will be situated in Geostationary orbit, thus causing it to spend a considerable proportion of its time in eclipse.

Other potential problems include periods during which a satellite may be out of contact with the available ground stations, necessitating data storage until a ground station comes within antennae range and the data can be downlinked. This does not allow for continuous monitoring owing to periods where data is not available in real time.

Initial data processing may also delay supply of the data to customers. Availability may also be a problem when instrument teams with scientific goals wish to restrict data access prior to publication of science results. In the context of space weather monitoring and prediction, the availability of real time data is of paramount importance in order to allow users to take appropriate action if a space weather event is predicted/detected.

A scientific study might also require instrument performance to be in excess of that required for a monitoring system, thus whilst providing excellent data, this might slow the rate of data acquisition to a rate lower than that needed for effective monitoring.

Continuity of observations and longevity of the mission might also be considered less important in terms of scientific mission objectives, whereas these would be of key importance in terms of a monitoring mission. Indeed, science missions are normally unique and have a specific expected lifetime. In contrast, a space weather monitoring system should have a system of replacement built in, such that as one element begins to fail, a replacement can be called into operation at short notice, thus maintaining an operational system.

Furthermore, whilst there is currently considerable interest in scientific study of the solar-terrestrial system, this will **not necessarily** remain a priority for scientific missions in the future. Consequently, space weather services in the long run should not rely on science missions.

There are established and developing space weather activities in the US and Japan. US initiatives and data are currently widely available. However, it is perhaps not wise for Europe to rely entirely on US initiatives. The results of these initiatives are currently available to the general public. However, in the current climate the possibility exists that the provision of this service to the US military may lead to restrictions being imposed.

4. Space Weather Architecture

4.1 High Level Requirements and System Constraints

The high level requirement for the CDF study was:

To design a minimum set of S/C, missions and associated Ground Stations to perform continuous monitoring of Space Weather phenomena and near real time downlink of the data to Earth.

This minimum set is defined in order to carry out the highest priority space-based measurements for a Space Weather service.

The set of missions should be considered as a pre-operational system in view of a future continuous service in analogy with existing meteorological services.

The following constraints and assumptions were also defined:

- Although the service must be intended to be continuously operational for several years, each mission should be designed to have a lifetime of at least 5 years. After this time, replacement S/C must be foreseen.
- The system should be considered as European-only, and no synergy with similar initiatives (e.g. in the US) should be sought at this stage.
- The system should be independent and, given the different aims, should not rely on presently operational or planned scientific missions.
- The target cost of the pre-operational system should be below 300 M€.
- The target date for deployment of the pre-operational system is 2006.

At the beginning of the study, three dedicated missions were defined as core missions to be analysed in detail by the CDF study:

Name	Mission	Main Objective
IMM	Inner Magnetospheric Monitor	To provide near-real-time monitoring of Earth's magnetic field and particles
SWM	Solar Wind Monitor	To provide near-real-time monitoring of the solar wind upstream from Earth
SAM	Solar Activity Monitor	To provide near-real-time monitoring of the solar disc (for solar flare detection) and corona

Table 4-1: Space Weather Missions Studied by CDF Study

According to the above objectives the following requirements on number of S/C and orbital locations have been defined for each mission:

Name	Number of S/C	Orbital location
IMM	Constellation (min 3)	Around the Earth Orbital plane close to the equatorial plane Eccentric orbits in order to sweep several altitudes
SWM	1	Inside the Solar Wind streamlines Between Earth and Sun and sufficiently ahead of Earth Unobstructed view of Sun
SAM	1	Sun pointing Unobstructed view of Sun Possibly pointing direction at an agle with the Sun-Earth direction

Table 4-2: Requirements on the Missions

4.2 System Architecture Trade-Off

In order to proceed with the design of the S/C and associated ground system, a baseline system architecture was selected. To this aim, several system options were defined, taking into account the requirements of Table 4-2, the possible launch strategies, the number and locations of associated ground antennas, and the type and class of relevant S/C.

Concerning the IMM mission, it appeared clear that the only possible option is to have a constellation of S/C in a GTO-like orbit, opportunely phased with each other in order to maximise the Earth magnetic field coverage. Although from a user point of view the minimum acceptable number of S/C is three, following the analysis reported in section 6.3, it was found that a constellation of four would give a significant improvement in terms of data return as compared to three. For this reason it was decided to assume four S/C as the baseline IMM constellation in the study.

For the orbital location of the SWM mission, as evident from Table 4-2, there is no real competitor to an orbit around the Lagrangian point L_1, in terms of fulfilment of user requirements.

Therefore, all the system architecture options presented hereafter have been based on the various possible choices of the SAM orbital location and number of S/C.

Six main options have been analysed, and these are described in the following subsections.

4.2.1 Option A: Dual Launch to L_1 Halo Orbit

4 IMM in GTO-like orbits (90 deg apart)
1 SWM in L_1 halo orbit
1 SAM in L_1 halo orbit

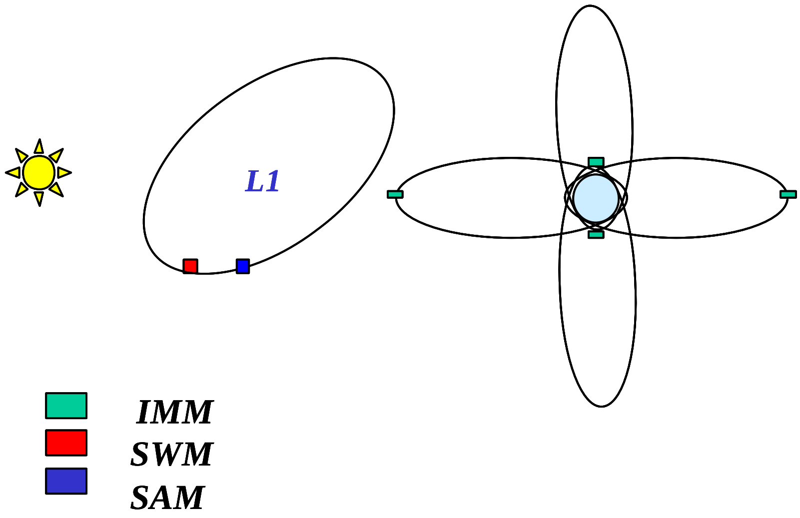


Figure 4-1: Space Segment Architecture: Option A

In this option SAM is in the same halo orbit around L_1 and close to SWM (to allow use of the same ground antennas).

The appeal of this option is the reduced number of launches required to deploy the whole system. As shown in the mission chapter, in this case only two launches would be needed because SWM and SAM can be accommodated together.

The high number of dedicated ground antennas for satellite downlink represents the main drawback of this option. In fact, a total of seven antennas would be required, located at four different stations and connected by lines to the central operation centre.

As far as satisfaction of the user requirements is concerned, this option performs well, with only two exceptions:

- A gap in coverage of the IMM constellation of max 30 min
- Non-optimal detection of CMEs because of the small angle of the Sun pointing direction with respect to the Earth-Sun direction.

A detailed summary of option A is reported in the following table:

Option		A
IMM		4 S/C in 650x39717 equat
SWM		1 S/C in L1 Halo
SAM		1 S/C in L1 Halo
Number of launches		2 (4IMM on GSLV, SWM+SAM on Soyuz)
Total number of S/C		6
Groundstations		
No. of Ground antennas		7 (4 IMM, 3 SWM&SAM)
No. of Ground locations		4
Costs		
launch		Minimum number of launches Cheap launchers
S/C		3 different types of S/C (IMM spin stab, SWM spin stab, SAM 3-axis stab)
Ground Station		High number of antennas and lines
Complexity		
S/C		3 different designs but optimised for the payload accommodation
Requirements		
User req. Fulfilment		Satisfied with 2 exceptions: 1. gap of max 30 min for data from IMM, 2. CME seen from the front

Table 4-3: Summary of Option A

4.2.2 Option B: Combined SWM-SAM Spacecraft

4 IMM in GTO-like orbits (90 deg apart)
1 combined SWM&SAM in L_1 halo orbit

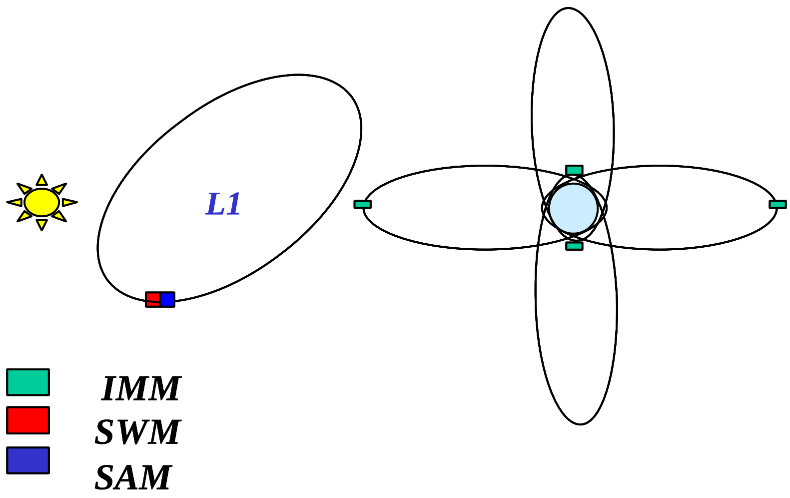


Figure 4-2: Space Segment Architecture: Option B

This option is the logical extension of option A for cost reduction: only one S/C would be needed for solar monitoring with a significant reduction in the verification programme, the overall management, and the schedule. All the considerations on launches, ground antennas and user requirements would be unchanged. However, the combined S/C would imply either some design complexity or some reduction in efficiency of the payload in the attempt to group instruments better suited for a spinning platform (SWM-carried) with instruments requiring the pointing accuracy of a 3-axis stabilised platform (SAM-carried). Further considerations for the design of such a combined SWM & SAM S/C is reported in section 7.4.5.1.

Since option A offers a better accommodation of the payload, the combined S/C option should be further investigated only if overall cost reduction of the programme is sought.

A summary of option B is reported in the following table:

Option	B
IMM	4 S/C in 650x39717 equat
SWM	1 S/C in L1 Halo
SAM	Combined with SWM in L1 Halo
Number of launches	2 (4IMM on GSLV, Combined SWM&SAM on Soyuz)
Total number of S/C	5
Groundstations	
No. of Ground antennas	7
No. of Ground locations	4
Costs	
launch	As Option A
S/C	Only 2 types of S/C (IMM spin stab, SWM&SAM 3-axis stab)
Ground Station	As Option A
Complexity	
S/C	The combined SWM&SAM is more complex
Requirements	
User req. Fulfilment	Can be satisfied but in addition to the exceptions as in Opt 1 the SWM instruments must be adapted to a 3-axis platform

Table 4-4: Summary of Option B

4.2.3 Option C: Data Relay

4 IMM in GTO-like orbits (90 deg apart)
1 SWM in L_1 halo orbit
2 SAM in GEO working also as Data Relays for the other satellites

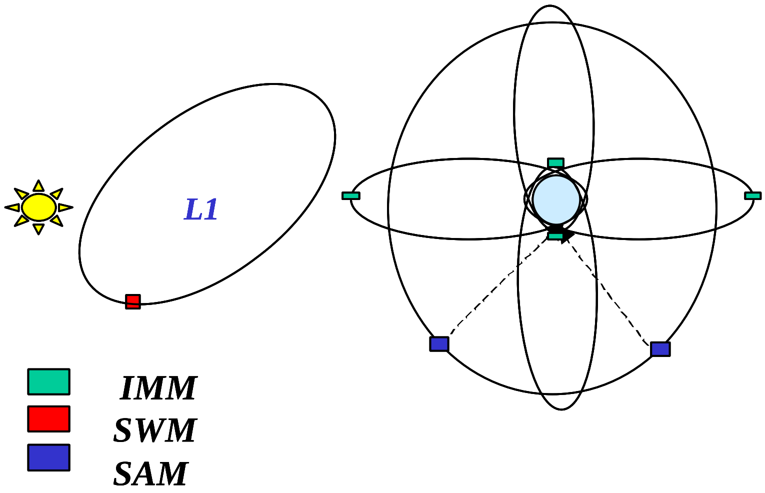


Figure 4-3: Space Segment Architecture: Option C

The advantages of this option are that the number of ground antennas is reduced to only two (in one location, as discussed in section 8.3.1) and that there would be no gap in the IMM constellation monitoring. Furthermore, this option may facilitate the future addition of more space segment components, such as LEO spacecraft for atmospheric, ionospheric and auroral phenomena monnitoring.

However, the complexity, reduced in the ground stations, is somehow shifted to the Space Segment that now requires two identical SAM satellites (to cope with the gap in Sun monitoring during the GEO eclipse) instead of only one. In addition, the design of such S/C must take into account the double function of Solar Activity Monitor and data relay, and therefore, although not unfeasible, looks more complex as compared with the design for SAM in L_1 orbit.

A major drawback of this option, in the context of the reduced set of missions investigated in this study, is the need for one additional launch to GEO, which is expensive even if carried with a companion spacecraft.

This option, therefore, while potentially very interesting in the context of a full-blown space weather programme, requires more in-depth analysis at programmatic level, which is beyond the scope of this study.

A summary of option C is reported in the following table:

Option	C
IMM	4 S/C in 650x39717 equat
SWM	1 S/C in L1 Halo
SAM	2 S/C in GEO (separation in long > 17 deg)
Number of launches	3 (4IMM on GSLV, 1 SWM on Rockot + 2 SAM on GSLV or Soyuz or A5)
Total number of S/C	7
Groundstations	
No. of Ground antennas	2
No. of Ground locations	1
Costs	
launch	Highest number of launches Launch to GEO expensive Launch of SWM to L1 with Rockot requires a STAR 37 motor
S/C	3 different types of S/C (IMM spin stab, SWM spin stab, SAM 3-axis stab)
Ground Station	Simplest Ground architecture
Complexity	
S/C	Design of SAM more complex than Opt1 because it works both as data relay and service Design of SWM more complex because a STAR 37 motor must be accommodated
Requirements	
User req. Fulfilment	Satisfied with 1 exception: 1. CME seen from the front

Table 4-5: Summary of Option C

4.2.4 Option D: Trailing Orbit

4 IMM in GTO-like orbits (90 deg apart)
1 SWM in L_1 halo orbit
1 SAM in a 10-deg Trailing orbit

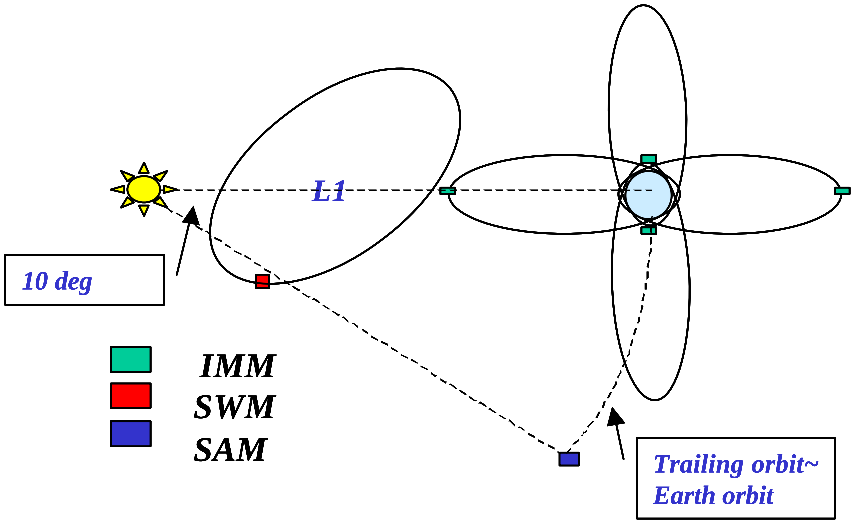


Figure 4-4: Space Segment Architecture: Option D

This option is the only one that satisfies the requirement of pointing SAM to the Sun from an angle (10 deg) with respect to the Sun-Earth direction.

In general, it is very similar to option A, because, as shown in chapter 7.2.2, a dual launch of SWM and SAM is still possible with an opportune strategy and with little propulsion impact on the SAM design.

However, the requirement of continuous coverage here calls for an additional ground antenna (in total 8 are needed) and a more demanding telecomms system on board SAM, due to the large distance from the Earth (~26 million kilometres) and the relatively long cruise phase with very variable Sun-S/C-Earth angle (Figure 7-5).

Summarising, this option is not superior to option A and should be considered only if the requirement on the monitoring of the CME's structure is not negotiable.

For more information, see section 7.2.2.

A summary of option D is reported in the following table:

Option	D
IMM	4 S/C in 650x39717 equat
SWM	1 S/C in L1 Halo
SAM	1 S/C in 10-deg trailing orbit
Number of launches	2 (4IMM on GSLV, SWM+SAM on Soyuz and later separated)
Total number of S/C	6
Groundstations	
No. of Ground antennas	8
No. of Ground locations	4
Costs	
launch	As option A
S/C	3 different types of S/C (IMM spin stab, SWM spin stab, SAM 3-axis stab)
Ground Station	Highest number of antennas
Complexity	
S/C	SAM more complex than in opt. A.: TT&C More complex, Propulsion must be carried to perform the transfer to the 10-deg TO A penumbra phase during transfer must be dealt with
Requirements	
User req. Fulfilment	Satisfied with 1 exception: 1. Gap of max 30 min for data from IMM

Table 4-6: Summary of Option D

4.2.5 Option E: Balloons

4 IMM in GTO-like orbits (90 deg apart)
1 SWM in L_1 halo orbit
Several SAM as long duration polar balloons

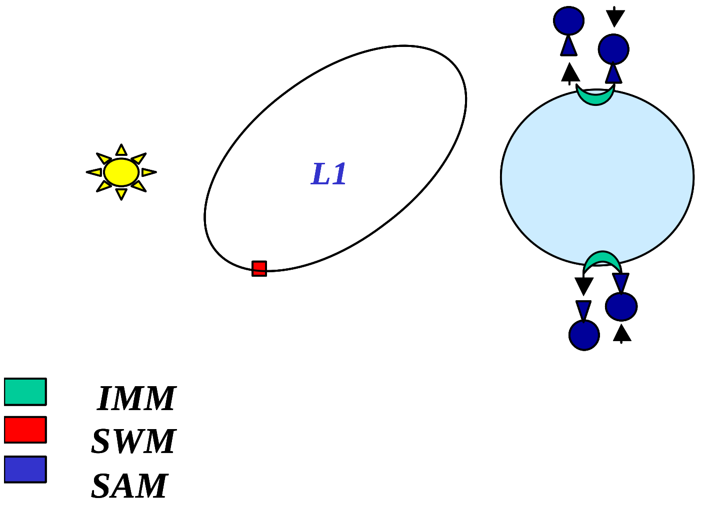


Figure 4-5: Space Segment Architecture: Option E

Balloons have been already used to carry instruments for solar observation and are very cheap and simple devices. In addition, the launch is inexpensive and the payload can be recovered and re-used.

However, due to the specific Space Weather requirements, several technology improvements and some infrastructure would be required. First of all, to avoid too frequent launches and recovery operations the balloons should guarantee a long duration (in the order of 100 days) at a defined high altitude location. Present balloons have mission duration of the order of 10 days, and although technology programmes are going on to increase the lifetime, no definitive result has been yet achieved.

Secondly, the balloons should be launched alternately in the winter and summer seasons in the two polar regions, thus avoiding eclipses but making the recovery operations much more difficult.

Thirdly, the requirement of continuous downlink to Earth can only be achieved by dedicated Polar Stations (present balloons use Data Relay Satellites) that would be difficult to build and operate.

Finally, due to atmospheric absorption, only hard x-ray and visible observations are feasible from balloons.

For all these reasons, this option has not been recommended.

Some more detail on the design of long duration balloons is in 7.4.5.4.

A summary of option E is reported in the following table:

Option	E
IMM	4 S/C in 650x39717 equat
SWM	1 S/C in L1 Halo
SAM	Several high altitude polar balloons
Number of launches	2 (4IMM on GSLV, 1 SWM on Rockot) + balloon launches
Total number of S/C	5 S/C + at least 6 ballons
Groundstations	
No. of Ground antennas	7 + 2 polar for the balloons
No. of Ground locations	4+2
Costs	
launch	As option A but launches and recoveries of balloons to be added
S/C	Only 2 types of S/C (IMM spin stab, SWM spin stab) + 1 balloon
Ground Station	High number of antennas and lines + need for 2 additional stations at the poles
Complexity	
S/C	Designs of IMM and SWM as in option A. Long duration balloons in principle simple but a reliable technology is still not available
Requirements	
User req. Fulfilment	Satisfied with 1 exception: 1. Gap of max 30 min for data from IMM

Table 4-7: Summary of Option E

4.2.6 Option F: SAM in Sun-Synchronous Orbit

4 IMM in GTO-like orbits (90 deg apart)
1 SWM in L_1 halo orbit
1 or 2 SAM in SSO

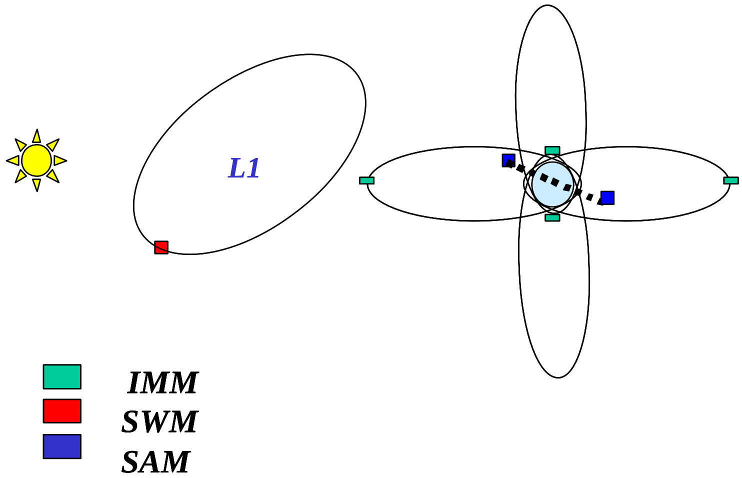


Figure 4-6: Space Segment Architecture: Option F

Two identical SAM S/C opportunely phased would satisfy the requirement of continuous monitoring of the Sun. A SSO orbit without eclipse could also be selected.

However attractive this option may seem (the S/C can re-use a commercial LEO platform and the launch would be cheap), a very high number of Ground Stations would be required to guarantee the near-real-time downlink. Because of the cost impact of building such a large system this option has not been considered further.

A summary of option F is reported in the following table:

Option	F
IMM	4 S/C in 650x39717 equat
SWM	1 S/C in L1 Halo
SAM	1-2 S/C in SSO
Number of launches	3 (4IMM on GSLV, 1 SWM on Rockot, 1-2 SAM on Soyuz or PSLV)
Total number of S/C	6 or 7
Groundstations	
No. of Ground antennas	very high
No. of Ground locations	very high
Costs	
launch	As option C but launch of SAM cheaper
S/C	3 different types of S/C (IMM spin stab, SWM spin stab, SAM 3-axis stab)
Ground Station	Very High number of antennas and lines
Complexity	
S/C	Comparable to Option A
Requirements	
User req. Fulfilment	Full coverage cannot be guarantee within a reasonable cost

Table 4-8: Summary of Option F

4.2.7 Conclusions

In conclusion, the option of a constellation of 4 IMM S/C in GTO-like orbits with SAM and SWM in L_1 as separate S/C has been considered the simplest concerning the S/C design, the one with the minimum launch cost, and which provides a satisfactory outcome from the Space Weather monitoring point of view.

Therefore Option A has been taken as baseline for the S/C design.

The Data Relay option is a possible alternative, especially if an additional constellation of ionosphere monitoring satellites is added in LEO (not considered in the present study), but the design of SAM in this case must be investigated in detail.

The combined SAM & SWM option could be considered if cost reduction is required.

The Trailing Orbit option should be only considered if emphasis is to be put on CME monitoring.

Balloons and the SSO option are not recommended for the given set of study requirements.

This page intentionally left blank

5. Inner Magnetosphere Monitor (IMM)

The Inner Magnetosphere Monitor is designed to provide near-real-time monitoring of the near-Earth magnetic and electric fields and particles.

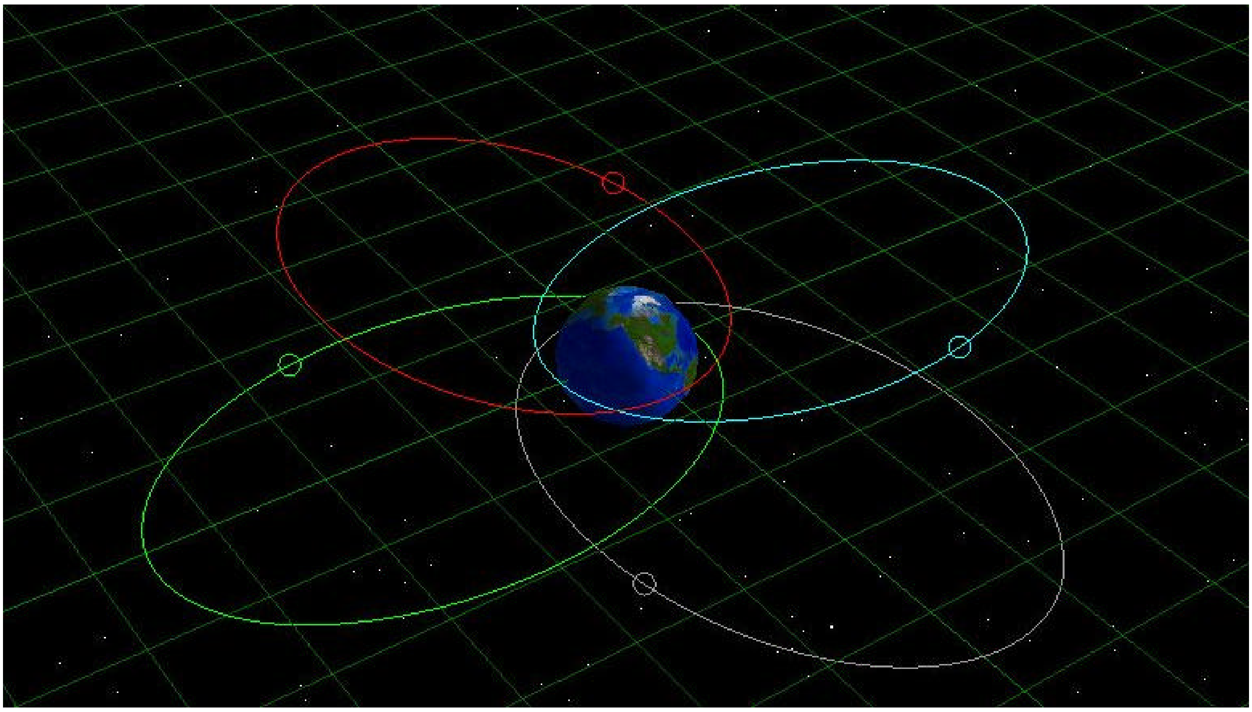


Figure 5-1: Orbits of the IMM Spacecraft

5.1 Payload

Monitoring of the inner magnetosphere is one of the key elements for tracking the propagation of geomagnetic storms and anticipating their effects on Earth-orbiting satellites, the ionosphere, and ground infrastructure. In order to achieve these objectives, the IMM satellite will carry a suite of observation instruments, providing the required information on charged particle distributions, magnetic fields and magnetospheric waves.

The following instruments have been selected:

- Thermal Plasma Monitor (TPM)
- Mid-Energy particle Monitor (MEM)
- High-Energy particle Monitor (HEM)
- Magnetometer (MAG)
- Waves Instrument (WAVE)
- GPS Receiver Ionospheric Sounder (GRIS)

5.1.1 Payload Requirements

A spacecraft design policy considering a strict (‘CLUSTER-grade’) electromagnetic, magnetic and electrostatic cleanliness is a mandatory condition for all the particle and field detectors. Other specific instrument requirements are given below.

5.1.1.1 TPM, MEM and HEM Instruments

The requirements for both the TPM and MEM are very similar. The TPM should be able to detect both ions and electrons in the low energy range, between a few eV and 40 keV. Particles in this range can lead to surface charging on spacecraft surfaces. The MEM will cover a different energy range, enabling characterisation of the distributions of ions and electrons whose energies are between 40 keV and 2 MeV. These particles are responsible for the deep dielectric charging effect, which can lead to severe malfunctions on orbiting spacecraft. Both the TPM and MEM should sample most of the 4π solid angle with a 45° angular resolution or better.

For the HEM requirements, the stress has been put on its ability to detect energetic protons with energies between a few and some hundred MeV. This instrument should be suitable to measure fluxes of particles that are responsible for both Single Event Upsets (SEUs) in the satellite on-board electronics and solar array performance degradation, among other effects.

The time resolution should be better than one minute for all three particle monitors.

5.1.1.2 MAG Instrument

Determining the local magnetic field topology is important in helping to diagnose the structure and dynamics of the magnetosphere. The suggested full scale ranges for operation in the inner magnetosphere are 0-±64 and 0-±256 nT with a minimum 1-minute time resolution. Measurements are required in three orthogonal directions to determine the magnetic field direction.

5.1.1.3 WAVE Instrument

In-situ measurements of plasma waves are important in monitoring the process of particle acceleration in the magnetosphere. They play an important role in the radiation belt dynamics, and radio waves are emited by auroral electron precipitations. In addition they can help to identify the position of certain plasma boundaries such as the plasmasphere, which in turn depend on the local Space Weather conditions. The suggested frequency ranges were 1 Hz-100 Hz using 3 magnetic and 1 electric antennae, and 100 Hz–30 kHz with 1 magnetic and 1 electric antenna.

5.1.1.4 GRIS Experiment

Radio occultation limb soundings can be used to obtain vertical profiles of electron density in the ionosphere and plasmasphere. In order to be able to carry out differential mesurements of propagation delays, the instrument should be able to receive in both L1 and L2 standard GPS frequency bands. The receiver should be adapted for operation in GTO. Compared with previous experiences in which the instrument was operated in LEO, the GTO radiation environment will be harsher, the visibility of the GPS constellation visibility will be poorer, and the GPS signals will experience an increased free space attenuation.

5.1.2 Payload Description

The requirement for most of the detectors to have a good angular coverage in almost every direction has been one of the main drivers leading to a spin-stabilised spacecraft platform. The the combination of the FOV of the instruments and the spin of the spacecraft about its axis make it possible to sample most of the solid angle sphere in a simple and effective way, and this is assumed in the description of the instrument design given below.

5.1.2.1 Thermal Plasma Monitor

This instrument will be a Top-Hat electrostatic analyser, similar to previous designs flown on WIND (3D PLASMA), Equator-S (3DA), or Cluster II (PEACE). The sensor heads will be integrated in the same box as the control electronics. This single unit will be mounted on the spacecraft side so that the pointing direction is perpendicular to the spin axis (i.e. the z axis). The instrument has an FOV of $180° \times 15°$ (polar $\times$ azimuthal angles), and this leads to a full 4π solid angle coverage after one spacecraft revolution.

The preferred instrument has the following specifications:

- mass 5 kg
- power 8 W
- telemetry rate 2 kbps
- dimensions 250 x 200 x 200 mm
- (design) temperature operating range -10/+20°C
- non-operating -30/+60 °C

5.1.2.2 Mid-Energy particle Monitor

The detector uses an array of solid state detectors in six energy channels and eight (22.5°×20°) angle channels at a time. The instrument FOV will be 180° x 20° (polar × azimuthal angles). The preferred instrument has the following characteristics:

- mass 2 kg
- power 4 W
- telemetry rate 2 kbps
- dimensions 150 x 150 x 150 mm
- (design) temperature operating range -10/+20°C
- non-operating -30/+60 °C

5.1.2.3 High Energy particle Monitor

The selected design will be able to measure protons in the 500 keV- 160 MeV range, and can also be used to detect electrons with energies in the range 30 keV - 5 MeV. The instrument is made up of silicon detectors arranged in stacks, or ‘telescopes’, three of which are mounted on the same box at angles of -15°, +15° and +45° with respect to the zenith. Each of these telescopes has a FOV of ±15°, i.e. a total FOV of approximately 90° in the spin axis plane. The box with the three solid state telescopes and the CPU is mounted on the spacecraft top (+Z) platform. Heritage is from UARS-HEPS.

5.1.2.4 Magnetometer

A well-established 3-axis flux-gate magnetometer will be used. This will be made up of an electronics box containing the DPU and two separate sensors, one mounted at the end of a 2 m boom and the other 0.5 m further inboard. The use of the boom will reduce the amount of interference from the spacecraft, while the combined use of the two sensors will enable determination of its magnitude. The sensor duplication will at the same time provide some degree of redundancy.

The selected design is based on an off-the-shelf instrument having the following specifications:

- mass 1.5 kg (1 kg CPU, 2 x 0.1 kg sensors, 0.3 kg harness)
- power 2 W
- telemetry rate 0.2 kbps
- dimensions:
  - CPU 200 x 100 x 150 mm
  - sensors 40 x 40 x 40 mm

5.1.2.5 Waves Instrument

This instrument uses electric antennae and a tri-axis search coil magnetometer. It is based on a similar design being considered for the BepiColombo MMO element, using only a pair of electric antennae, enough to meet the mission objectives.

This configuration is suitable for carrying out electric field measurements in the 0.1 Hz-16 MHz range. The spectral range for magnetic measurements is 0.1 Hz - 1 MHz.

The instrument mass breakdown is as follows:

- Electronics 1.8 kg
- 2 x wire antennae + deployers 1.5 kg each
- Search coils (with pre-amps) 1 kg
- Total 5.8 kg mass

Other specifications:

- Power 4 W
- Telemetry rate 2 kbps nominal (utilisation of 20 kbps burst mode would provide improved resolution but has not been considered in our baseline)
- Dimensions:
  - Electronics 200 x 100 x 50 mm
  - Antenna mechanism 350 x 100 x 50 mm (each)
  - Antennae 2 x 30m wires extending radially away from the spacecraft body

5.1.2.6 GPS Receiver Ionospheric Sounder

Two possibilities have been explored:

1. **A dual frequency L1 and L2 receiver** based on the GAGE instrument on the STRV-1d spacecraft.

   A small patch antenna giving it a FOV of 90° would be positioned on one of the spinning faces, so that it points towards the Earth once per spin. The electronics would include an RF sampling device which would only be turned on for a short period when the antenna was pointing in the correct direction. As opposed to the GAGE, which did not include any GPS processing (large raw samples were directly transmitted to the ground), GRIS would use on-board processing for determination of the required parameters before transmission. GAGE was a collaborative project between DERA, JPL and the US DoD.

2. **The MosaicGNSS**, a commercial GPS receiver currently under development at Daimler-Chrysler Aerospace (Astrium GmbH).

   This is actually based on an ESA-developed chip called AGGA that is able to provide GPS/GLONASS capability at L1 and L2 frequencies. This chip has been used by several European companies to develop space receivers, in particular by Laben, Saab and Astrium. These receivers have a mass of about 5 kg and a consumption around 12 W, including front end, antennas and DC-DC converters. The MosaicGNSS design shares hardware and software resources with the AOCS, including program code, memory, CPU, housing, and power supply.

The system configuration in the first option has been assumed to fulfil the Space Weather goals, though the MosaicGNSS receiver and an alternative antenna location (maybe using two or more of them) should be investigated when the design becomes more mature.

Specifications:

- mass 5 kg
- power 12 W
- telemetry rate 1 kbps
- dimensions 60 x 60 x 60 mm

5.1.3 Payload Budgets Summary

Instrument Name	Acronym	Mass (kg)	Power (W)	Telemetry rate (kbps)	Remarks
Thermal Plasma Monitor	TPM	5	8	2	
Mid-Energy particle Monitor	MEM	2	4	2	
High-Energy particle Monitor	HEM	6.1	6.25	1.5	
Magnetometer	MAG	1.2	2	0.2	2 sensors on a 2m rigid boom
Waves instrument	WAVE	5.8	4	2	Includes 1 pair of 30m wire antennas, search coil on 1.3m boom
GPS Receiver Ionospheric Sounder	GRIS	5	12	1	
Totals		25.1	36.25	8.7	

Table 5-1: Payload Mass and Power Budgets

5.2 Mission Analysis

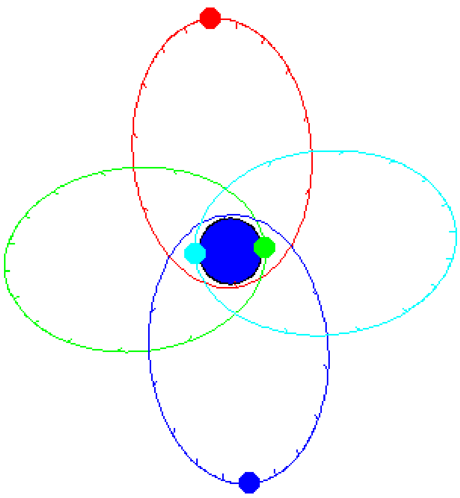
5.2.1 Orbit Selection

The IMM basic requirements for orbit selection are:

1. Optimum coverage of the Earth magnetosphere
2. Optimum visibility from a minimum number of ground stations

These requirements are respectively met by:

1. A constellation of four satellites on nearly equatorial elliptic orbits with 0° argument of perigee and apsidal lines at 90° intervals (Figure 5-2).
2. The individual orbits are synchronous



Note: The scale of the orbits (with tick marks every hour) and Earth, and the phase of the satellites, are respected.

Figure 5-2: IMM Constellation in the Equatorial Plane

The perigee height is at low altitude, but sufficiently high that the orbit is stable with respect to luni-solar perturbations. An altitude of 650 km is optimum from an orbital mechanics point of view.

The apogee height should be in the range of 6 Earth radii. A 12-h synchronicity is achieved by selecting the apogee height at 39717 km.

The phase of each satellite on its orbit is such that the best use of ground stations is made.

The IMM orbital parameters are summarised in the following table, relative to the following reference epoch: 2005-06-01 @ 00:00:00.

Parameter	S/C 1	S/C 2	S/C 3	S/C 4
Perigee	650 km			
Apogee	39717 km			
Inclination	0 to 10°			
Argument of perigee	0°			
Right ascension of ascending node	4.81°	94.81°	184.81°	274.81°
True anomaly	180°	0°	180°	0°
Orbital period (12h)	43082 s			

Table 5-2: IMM Orbital Parameters

5.2.2 Launch and Early Orbit Phase (LEOP)

The goal of the LEOP is to place the satellites in their final orbit as defined in the preceding section.

5.2.2.1 Launch

As the final orbit is almost equatorial, the use of a low latitude launch pad prevents the need for a high inclination correction during orbit acquisition. Suitable low latitude launch pads are Kourou (lat. 5.2°) and Sriharikota (lat. 13.9°). Corresponding low-cost launchers are VEGA and PSLV respectively. Unfortunately, the performances of these two launchers are insufficient for a direct single launch of the constellation. However, India offers a more powerful launcher: GSLV (Geostationary Satellite Launch Vehicle). Although this launcher is optimised for GEO launches, it can be used for LEO launches with a mass performance gain of about 1.4 compared with the PSLV.

The baseline is therefore to launch the four IMM satellites as a single launch by a GSLV on a circular LEO.

- Altitude: 250 km
- Inclination: 18°
- GSLV estimated performance: 4050 kg
- Launch site: Sriharikota (SHAR), India (long. 80.4° E, lat. 13.9° N).

On this LEO, the rotation of the nodal line due to the J_2 perturbation is -7.96°/day. Rotation of the apsidal line of the final orbit is 0.32°/day. The differential rotation rate of the apsidal line is therefore -7.64°/day. After 11.8 days, the node has rotated by 90°.

5.2.2.2 Constellation Deployment

The deployment of the constellation is achieved by the following sequence of injections into elliptic intermediate orbit:

1. Using its on-board propulsion system, satellite 1 is injected into an elliptic orbit with apogee height at 39717 km by a sequence of perigee manoeuvres totalling 2.500 km/s.

2. 11.8 days later, satellite 2 is injected into the elliptic orbit, 90° separated from the apsidal line of satellite 1.
3. 23.6 days after the first injection, satellite 3 is injected into the elliptic orbit, 180° separated from the apsidal line of satellite 1.
4. 35.4 days after the first injection, satellite 4 is injected into the elliptic orbit, 270° separated from the apsidal line of satellite 1.

To reduce gravity loss to below 3%, the injection manoeuvre is divided into a sequence of ten consecutive perigee manoeuvres.

5.2.2.3 Insertion into Final Orbit

For each of the satellites, at first apogee passage after reaching the intermediate orbit, a combined apogee manoeuvre of 0.212 km/s is performed for

- Raising the perigee height from 250 km to 650 km
- Decreasing the inclination from 18° to 10°.

The proper phase of the satellite on its orbit is achieved by staying on a slightly asynchronous orbit (having an apogee height slightly above or below the nominal value) for a few days.

A total of 48 orbit manoeuvres are required for deploying the constellation to its final configuration.

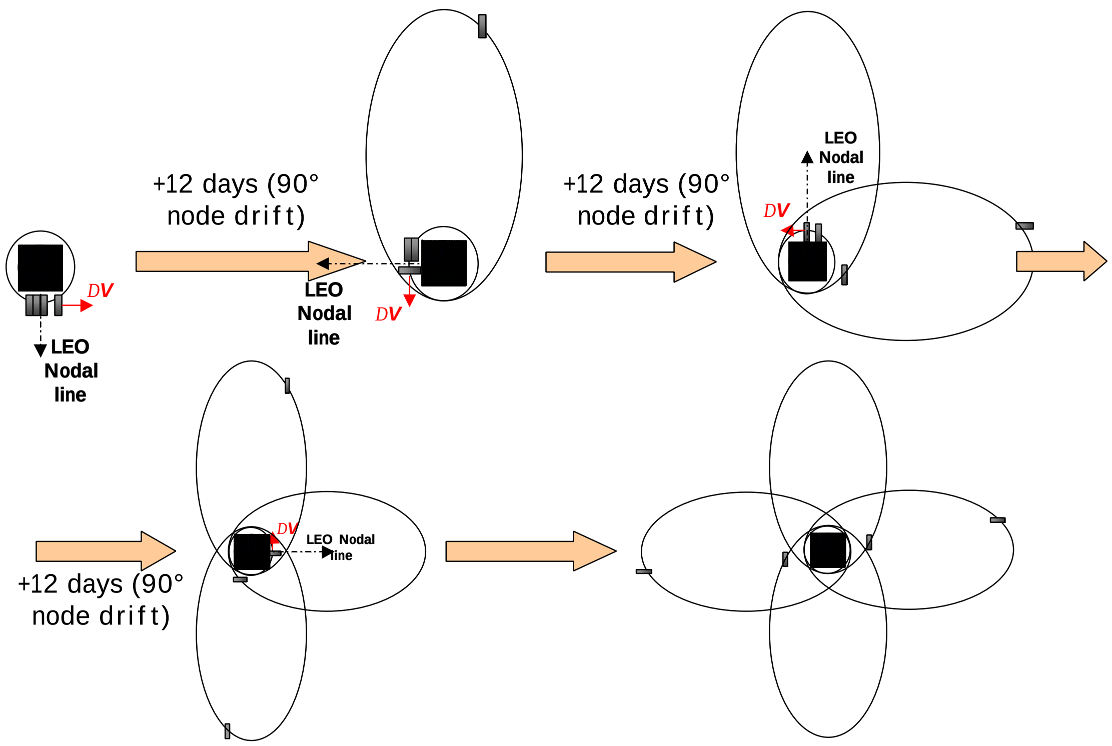


Figure 5-3: IMM Insertion into Final Orbit

5.2.3 Eclipse

A typical eclipse profile on the operational orbit is shown on Figure 5-4. Maximal eclipse duration is 2 h 22 mn, occuring sometimes during equinox. There is no eclipse during solstice.

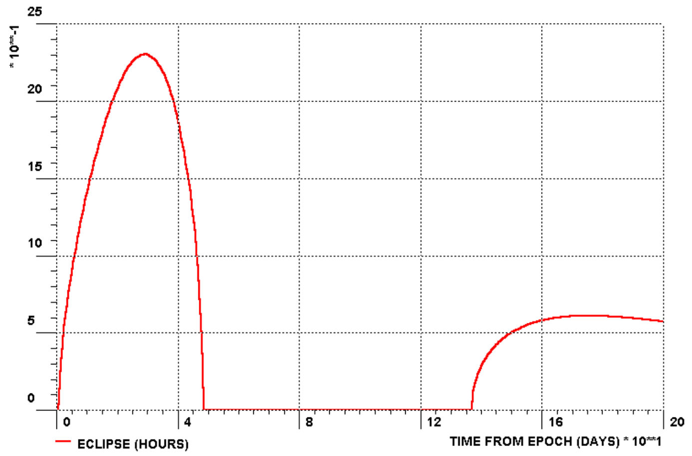


Figure 5-4: Typical Eclipse Duration Profile

5.2.4 Ground Station Coverage

Four ground stations are required for covering the part of the orbit above 3000 km height. The coverage outage around perigee is 33 mn (4.7 % of the time). These ground stations have to be located by pair at opposite longitudes. The best choice among the ESA stations is Kourou (long. E –52.8°) and Perth (long. E 115.8°) (see section 8.1). A coverage profile and a detailed coverage table are given in Figure 5-5.

2005/ 6/ 1	0: 0: 0.0	0.00	BEGIN		
2005/ 6/ 1	0:16:10.4	0.27	AOS	3	KOUROU
2005/ 6/ 1	0:16:31.6	0.28	AOS	1	PERTH
2005/ 6/ 1	5:42:48.0	5.71	LOS	4	PERTH
2005/ 6/ 1	5:48:11.3	5.80	LOS	2	KOUROU
2005/ 6/ 1	6:15:11.2	6.25	AOS	4	KOUROU
2005/ 6/ 1	6:15:32.2	6.26	AOS	2	PERTH
2005/ 6/ 1	11:41:49.0	11.70	LOS	1	PERTH
2005/ 6/ 1	11:47:12.2	11.79	LOS	3	KOUROU
2005/ 6/ 1	12:14:12.0	12.24	AOS	1	KOUROU
2005/ 6/ 1	12:14:32.4	12.24	AOS	3	PERTH
2005/ 6/ 1	17:40:49.2	17.68	LOS	2	PERTH
2005/ 6/ 1	17:46:13.0	17.77	LOS	4	KOUROU
2005/ 6/ 1	18:13:12.9	18.22	AOS	2	KOUROU
2005/ 6/ 1	18:13:33.6	18.23	AOS	4	PERTH
2005/ 6/ 1	23:39:49.9	23.66	LOS	3	PERTH
2005/ 6/ 1	23:45:13.8	23.75	LOS	1	KOUROU
2005/ 6/ 2	0: 0: 0.0	24.00	END		

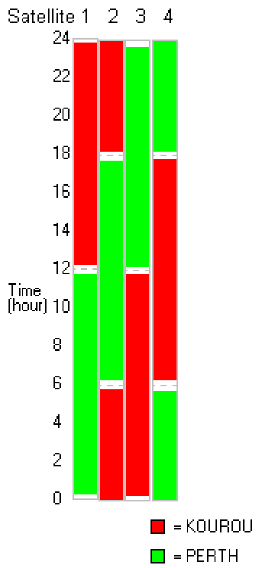


Figure 5-5: Coverage of the IMM Constellation from Kourou and Perth over 24h

5.2.5 Orbit Maintenance

Orbit maintenance is needed for keeping synchronicity. The corresponding total *DV* does not exceed 1 m/s for the duration of the mission. As natural perturbations are acting in a similar way on the four satellites, no other type of orbit maintenance is needed and disturbance in the spacing of the apsidal lines will not exceed 20° at the end of the lifetime.

5.2.6 Orbit Lifetime

Orbit stability with respect to luni-solar perturbations is guaranteed for at least 12 years.

De-orbit at the end of the mission can be accomplished by an apogee manoeuvre of the order of 70 m/s to reduce the perigee height to an altitude within the dense atmosphere.

5.2.7 Replacement Strategy

Should one of the satellites of the constellation suffer from a failure, it would not be possible to re-orient the apsidal line of two of the other satellites in order to have a regular distribution of the orbits (120° apart, a configuration which would be very close to satisfying the mission requirements). A satellite replacement strategy should be therefore be foreseen.

One possible replacement strategy is to have the spare satellite waiting in LEO in order to have the flexibility to inject it into a final orbit of any orientation. However, LEOs are decaying, and therefore the satellite would have to be raised to an altitude of 400-500 km in order to survive the atmospheric drag.

Another replacement strategy would be to have the fifth satellite ready on the ground and to launch it on demand, using a small launcher such as Rockot.

In any case, the replacement strategy implies additional launch costs to be taken into account in the overall programme budget.

5.2.8 Launch Option: ASAP 5

With its ASAP 5 (Ariane 5 Structure for Auxiliary Payload), Ariane 5 offers the possibility of launching four auxiliary payloads of maximum 300 kg mass each on a GTO, together with a main passenger.

Using this option the four satellites would be delivered into the same GTO. To create the necessary angular separation in apsidal line, three of the satellites would have to be moved into a waiting orbit of apogee higher or lower than the GTO apogee height. This means that the satellites would have to be equipped with an orbit propulsion system.

Using an orbit higher than the GTO would endanger the life of the satellite: the luni-solar perturbations would become strong, causing the perigee height to descend inside the dense atmosphere, and thus burning the satellite. Therefore, a lower orbit is recommended.

Figure 5-6 gives the waiting time needed by satellite 4 to achieve a 270° apsidal line rotation by waiting in an orbit with a given apogee height. The rotation is calculated relative to the 12-h orbit, where satellite 1 would be injected first. Satellites 2 and 3, to be rotated by 90° and 180° respectively, would undergo a shorter waiting time.

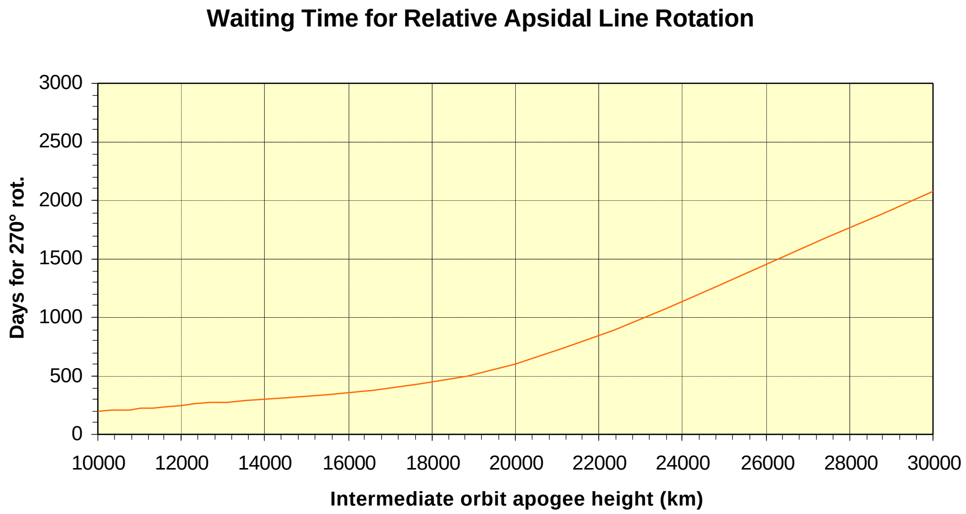


Figure 5-6: Time to Achieve a 270° Apsidal Line Rotation (ASAP5 option)

Assuming a 20000 km apogee height for the Drift Orbit, the differential rotation rate is 0.46°/day, leading to a waiting time of about 200, 400 and 600 days respectively for satellites 2, 3 and 4. The *DV* needed to decrease the GTO apogee height from 35890 km to 20000 km is 397 m/s (assuming the Ariane 5 GTO perigee height of 560 km). After the waiting time, a *DV* of 457 m/s is needed to raise the apogee height to its synchronous value of 39807 km. Total *DV* is 854 m/s.

The GTO inclination is 7°. To save propellant, the inclination could be left as it is. In this case, due to the rotation of the argument of perigee, the direction of the apogee of the 12-h orbit will undergo a periodic motion of ± 7° above the equatorial plane with a period of 560 days.

If reduction of the inclination to 0° is requested, this would be performed in combination with the final apogee raise manoeuvre, which would then cost 640 m/s, bringing the total *DV* to 1037 m/s.

Satellite 1 is injected immediately into the final 12-h orbit. Corresponding *DV* is only 60 m/s, or 243 m/s if the manoeuvre is combined with an inclination reduction to 0°.

5.3 Radiation

The IMM orbit takes the spacecraft through the heart of the trapped proton and electron radiation belts leading to a very harsh environment. Other, secondary, sources of radiation are from solar proton events and cosmic rays.

The McIllwain L-Shell is the equatorial distance, usually measured in Earth radii, to a magnetic field line. The magnetic field strength is then a measure of the magnetic 'latitude'.

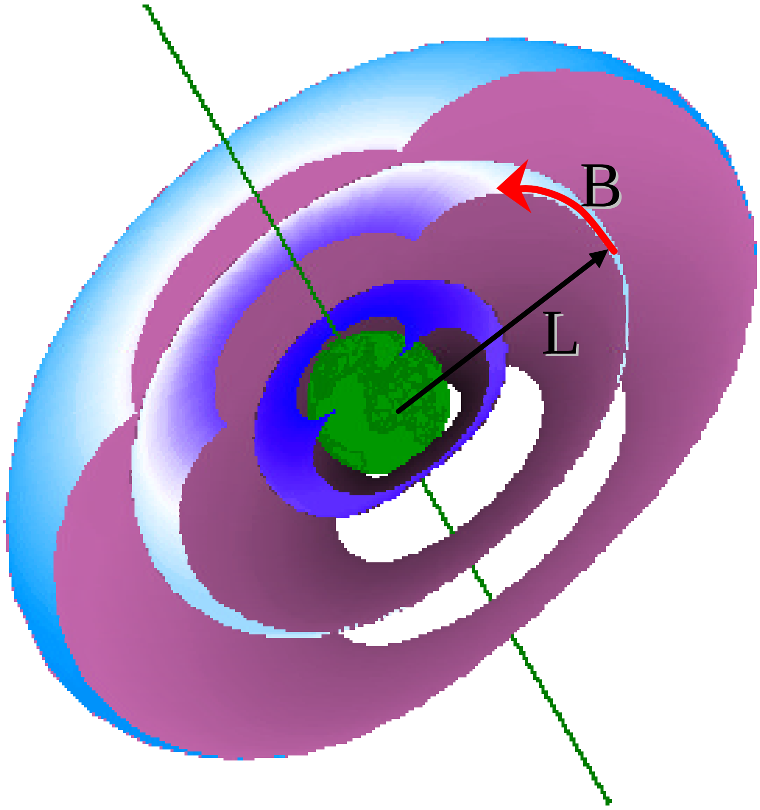


Figure 5-7: McIllwain L-Shell

5.3.1 Design Drivers

The purpose of the mission is to sample the radiation belts with good local time, and magnetic field parameter (strength, pitch angle and L-shell) coverage. The pitch-angle resolution can be resolved with either multiple detector heads or a spinning spacecraft and is not dependent on the trajectory ephemera. The magnetic field strength, local time and L-Shell coverage can only be resolved by orbit selection and synchronisation of the satellites in mean anomaly.

5.3.2 Baseline Design

Figure 5-8 & Figure 5-9 are based on orbit elements summarised in the Mission Analysis chapter, section 5.2.1. The baseline mission duration is five years.

Good L-Shell and local time coverage is achieved by separating the apsides of the four satellite orbits by 90º in the equatorial plane and distributing the mean anomaly by 90º. Figure 5-8 shows local time coverage over a 24-hour period for constellations of 3 and 4 satellites separated by 90º in mean anomaly and 90º in apsides. Note that if a constellation of three satellites was planned, these would be separated by 120°, not 90°. This figure illustrates the situation in which one of the four original satellites has been lost.

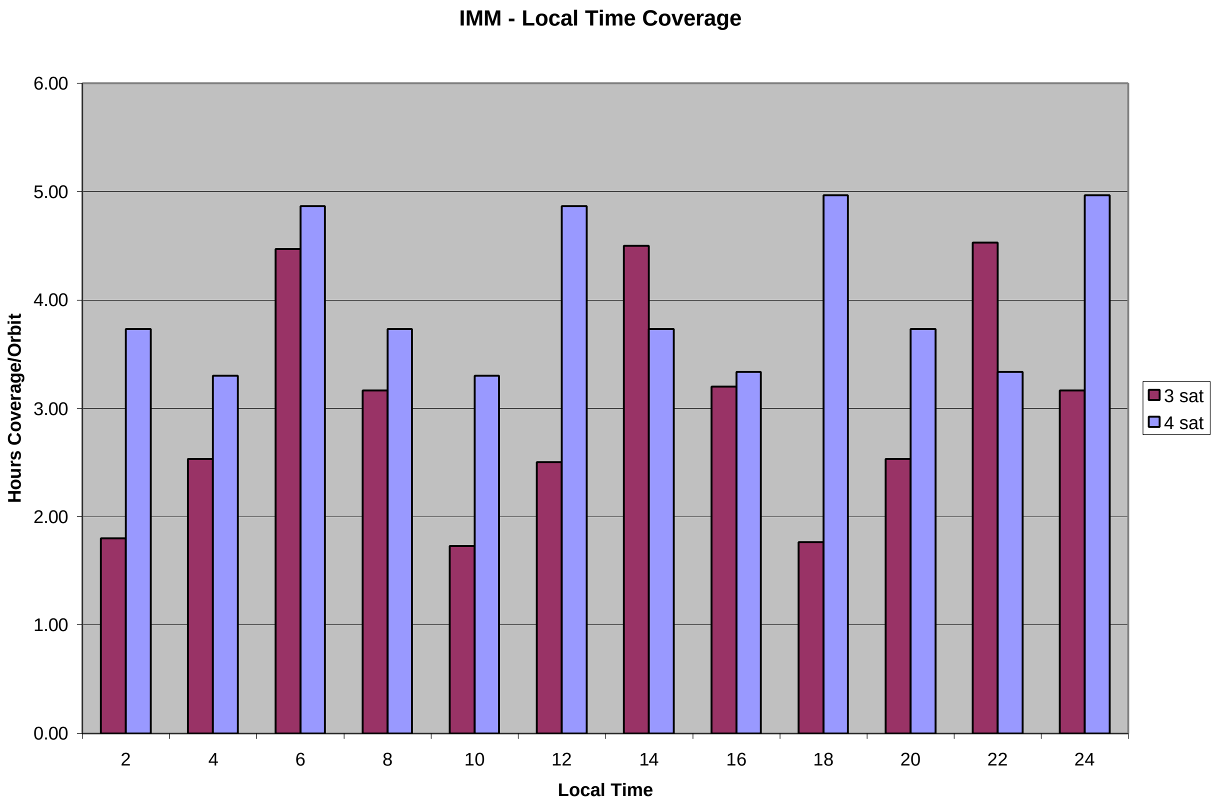


Figure 5-8: Local Time Coverage for Constellations of 3 and 4 Satellites

Figure 5-9 shows L-shell coverage over a 12-hour orbit for constellations of 3 and 4 satellites separated by 90º in mean anomaly and 90º in apsides.

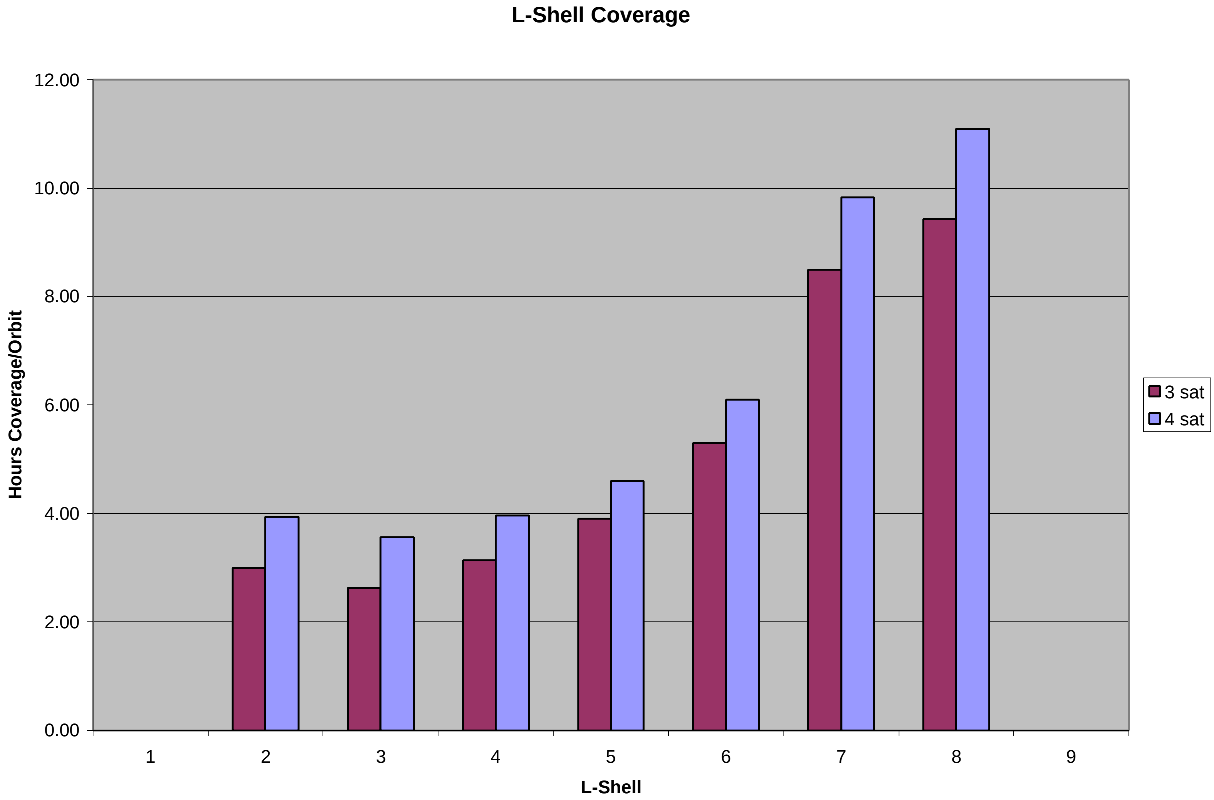


Figure 5-9: L-shell Coverage for Constellations of 3 and 4 Satellites

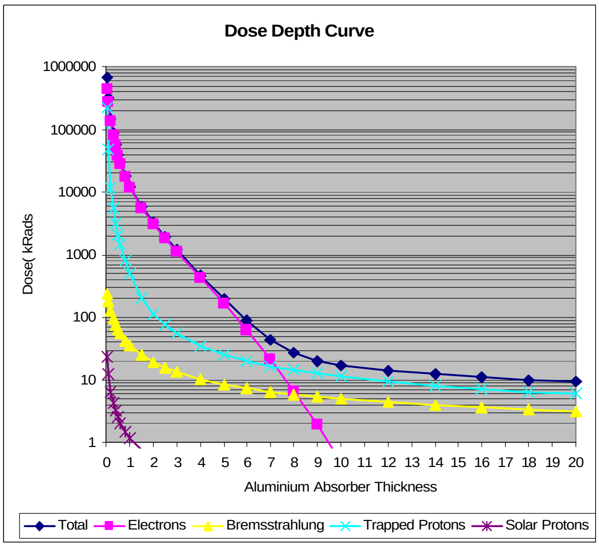
The above two figures show that, in the worst case, with four satellites a coverage of 3 hours per orbit is possible, while a constellation of 3 would only achieve 2 hours per orbit. For this reason, a four-satellite constellation has been preferred.

5.3.3 Radiation Environment

5.3.3.1 Total Ionising Dose

The traversal of the electron and proton radiation belts results in an extremely high dose. For a nominally shielded spacecraft (4mm of Aluminium shielding) this environment results in a total mission dose of 467 krad over a 5 year period. Additional shielding is effective in reducing the total dose with an resulting impact on the mass margin. For example, for the data handling subsystem, the dose has been reduced to less than 100 krad by ensuring that at least 6 mm of shielding is provided by the structure (boxes etc.).The dose is principally from the trapped electron belts, with the trapped proton belt contribution almost 10 times lower.

To avoid imposing excessive requirements, a TID calculation should be preferably be performed according to the S/C geometry.



Aluminium Absorber Thickness (mm)	Dose (kRad) Total	Electrons	Bremsstrahlung	Trapped Protons	Solar Protons
0.05	664370	438200	240	225910	23
0.10	318980	269690	183	49088	12
0.20	147110	135260	128	11719	6
0.30	86826	81233	97	5492	4
0.40	57019	53702	77	3236	3
0.50	40081	37910	64	2105	3
0.60	29629	28109	54	1464	2
0.80	18049	17199	43	806	1
1.00	12178	11642	35	500	1
1.50	5856	5627	25	203	1
2.00	3262	3130	19	113	1
2.50	1924	1833	16	75	0
3.00	1172	1104	13	55	0
4.00	**467**	**423**	**10**	**34**	**0**
5.00	197	164	8	25	0
6.00	89	61	7	20	0
7.00	44	21	6	17	0
8.00	27	7	6	15	0
9.00	20	2	5	13	0
10.00	17	0	5	11	0
12.00	14	0	4	9	0
14.00	12	0	4	8	0
16.00	11	0	4	7	0
18.00	10	0	3	6	0
20.00	9	0	3	6	0

Figure 5-10: Ionising Dose versus Aluminium Thickness

5.3.3.2 Solar Cell Degradation

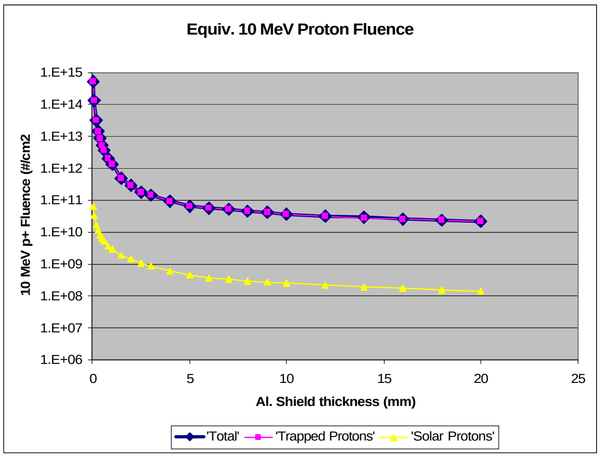
The solar cells will suffer damage principally from the trapped proton belt crossings with the total equivalent 1 MeV electron fluence for the mission exceeding the normal fluence from a GEO orbit by a factor of 250. Mitigation techniques include the use of radiation tolerant solar cells, oversizing of the solar arrays to ensure adequate power at the end of life, and the use of thick coverglass.

Coverglass Thickness (microns)	1 MeV Equivalent electron fluence (#/cm2)					
	V_{OC}		P_{MAX}		I_{SC}	
	GaAs	Silicon	GaAs	Silicon	GaAs	Silicon
0	1.1E+20	2.1E+19	7.6E+19	2.1E+19	5.21E+19	2.2E+18
76	7.3E+16	1.3E+17	5.2E+16	1.3E+17	2.72E+16	5.0E+16
152	1.7E+16	4.0E+16	1.2E+16	4.0E+16	6.03E+15	1.7E+16
305	4.4E+15	1.1E+16	3.2E+15	1.1E+16	1.58E+15	5.3E+15
509	1.7E+15	3.8E+15	1.2E+15	3.8E+15	6.26E+14	2.0E+15

Table 5-3: Solar Cell Degradation for Various Coverglass Thicknesses

5.3.3.3 Non-Ionising Dose

Apart from ionising dose, particles can lose energy through non-ionising interactions with materials, particularly through ‘displacement damage’ or ‘bulk damage’, in which atoms are displaced from their original sites. This can alter the electrical, mechanical or optical properties of materials and is an important damage mechanism for electro-optical components (e.g. solar cells and opto-couplers) and for detectors such as CCDs.



'Spherical Al Shield Radius'	'Total'	'Trapped Protons'	'Solar Protons'
0.05	5.11E+14	5.11E+14	6.48E+10
0.1	1.36E+14	1.36E+14	3.42E+10
0.2	3.16E+13	3.16E+13	1.64E+10
0.3	1.44E+13	1.43E+13	1.11E+10
0.4	8.42E+12	8.41E+12	8.33E+09
0.5	5.28E+12	5.28E+12	6.41E+09
0.6	3.75E+12	3.74E+12	5.31E+09
0.8	2.08E+12	2.07E+12	3.79E+09
1	1.33E+12	1.32E+12	2.92E+09
1.5	5.00E+11	4.98E+11	1.91E+09
2	3.01E+11	3.00E+11	1.44E+09
2.5	1.81E+11	1.80E+11	1.09E+09
3	1.46E+11	1.45E+11	8.92E+08
4	9.64E+10	9.58E+10	6.17E+08
5	6.53E+10	6.48E+10	4.60E+08
6	5.75E+10	5.71E+10	3.78E+08
7	5.10E+10	5.07E+10	3.36E+08
8	4.51E+10	4.48E+10	2.96E+08
9	4.12E+10	4.09E+10	2.74E+08
10	3.75E+10	3.72E+10	2.53E+08
12	3.12E+10	3.09E+10	2.19E+08
14	2.84E+10	2.82E+10	1.95E+08
16	2.57E+10	2.55E+10	1.73E+08
18	2.40E+10	2.38E+10	1.58E+08
20	2.21E+10	2.20E+10	1.42E+08

Figure 5-11: Non-ionising Dose versus Aluminium Thickness

5.3.3.4 Cosmic Ray

It is not expected that the cosmic-ray effects on the IMM will be any different than those for geostationary orbits. However, a higher single event upset (SEU) rate is likely during the passage of the trapped proton belts. This can result in enhanced error rates in on-board memory components and other 'soft' components.

5.3.4 Conclusions

The severity of the radiation environment will, in general, preclude the use of COTS components and may require the use of expensive rad-hard components or considerable additional radiation shielding. As the dose scales linearly with time, a further option is to provide replacement satellites at more frequent intervals and reduce the design life from 5 years. This increases the overall cost of the system.

5.4 IMM System Design

5.4.1 Requirements and Constraints

The IMM design was driven by:

- The orbits selected for the constellation operation and the strategy to achieve it
- The instrument requirements
- The Space Weather specific requirements and constraints

Accordingly, the following requirements were defined:

- The 4 S/C shall operate on highly eccentric, 12-hour period, 10° maximum inclination orbits, spaced 90 degrees apart in argument of perigee
- The 4 S/C shall all operate at the same time
- The data acquired by the 4 S/C shall be down linked to Earth in near real time, meaning that small gaps in coverage are accepted provided that any data not immediately transmitted is stored on board and sent at the earliest opportunity
- The minimum lifetime of each S/C shall be 5 years. This is particularly demanding in the case of IMM because of the very harsh radiation environment of the selected orbit
- The launch date shall be 2006
- The S/C shall feature high electromagnetic, magnetic and electrostatic cleanliness (Cluster-type)
- The S/C shall be spin stabilised

5.4.2 Design Drivers

As a consequence of the orbital requirements for IMM and the discussion on orbit acquisition strategy reported in section 5.2.2.2, two important design drivers arose:

- In order to achieve the final constellation configuration a large propulsion system is needed. In fact, either in the case of constellation building from LEO or in the case of GTO launch, a large delta-V must be applied to each S/C (2.7 km/s and 0.85 km/s respectively). This rules out the possibility of having a S/C of the microsat type, due to the large mass of propellant needed. It also reduces the possibility of re-using existing platform designs that would all have to be modified for accommodation of the large propulsion system.
- In this low inclination orbit the S/C experience very high radiation doses, as shown in section 5.3.3. This requires the selection of rad-hard components and opportune shielding for sensitive equipment. Again, this goes in the opposite direction from a simple and cheap design.

In addition, the selection of the launcher and its mass and volume performance largely constrain the design as discussed in the following section.

5.4.3 Design Options

5.4.3.1 Launcher

The complete constellation must be launched in a single launch for cost effectiveness.

A launch to GTO of the 4 S/C as main passengers would be too expensive; therefore different alternatives must be found.

Russian launchers, the cheapest available, are all launched from high latitudes. This means that large manoeuvres and therefore very large propellant masses are required to reduce the inclination to at least 10 degrees.

For the above reasons, only the following alternatives are possible, as described in section 5.2:

Launch strategy	Launcher	Mass Performance
To LEO low inclination (18° min)	PSLV	2800 kg max
To LEO low inclination (18° min)	GSLV	4050 kg max
To GTO	A5 ASAP	1200 kg max

Table 5-4: IMM Launch Options

Since the mass performance of GSLV is higher than that of PSLV, the following two design options have been retained:

Option 1:
Stack of 4 IMM satellites in the 1000-kg class, launched as the only passenger by GSLV, to a 250 km, 18° inclination circular orbit, and reaching the final constellation configuration by means of its own propulsion.

Option 2:
4 IMM mini-satellites (300 kg max) launched to GTO by Ariane 5 ASAP together with another payload and reaching the final constellation configuration by means of own propulsion with the set of manoeuvres described in section 5.2.8.

5.4.3.2 Platform

In order to keep the cost of the constellation as low as possible in both the above options, an analysis has been performed to assess if the re-use of existing designs or commercial platforms would be possible.

Table 5-5 below reports the platforms analysed.

	S/C mass	P/L	Mission	Orbit	AC	Prop	Rad hard	Note
STRV C&D	112 Kg	25 Kg	Inner Magnetosphere	300X36000 Km	Spin (? rpm)	None	YES	Propulsion needs to be accommodated
Equator-S	230 Kg	45.7 Kg	Inner Magnetosphere	500X63700 Km	Spin (40 rpm)	Solid prop	YES	Propulsion adequacy to be assessed
SSTL Emicro	up to 140 Kg	up to 45 Kg	Various	LEO	3-axis (spin possible ?)	Resistojet (for AOCS)	not specific	Stabilisation and propulsion to be checked
Proba	100 Kg (TBC)	?	Technology	Polar	3-axis	None	?	The platform requiring the biggest adptation
Freja	235 Kg	73 Kg	Magnetosphere	Polar	Spin	Solid prop	?	
Astrid-2	30 Kg	9 Kg	Magnetic field	Polar	Spin	None	?	Too small

Table 5-5: Platforms Analysed for IMM

Factors to be taken into account for possible selection include the payload mass carried, the type of stabilisation, the existence of propulsion onboard and, most of all, the type of orbit for which the platform has been designed (which defines the radiation environment).

From Table 5-5 it is evident that the only two valid candidates would be STRV 1c&d and Equator-S.

A closer look at the design of Equator-S showed it to be the one requiring the highest number of adaptations. Therefore, if re-use of a platform must be considered, STRV is the best candidate.

The STRV platform is shown in Figure 5-12.

Figure 5-12: STRV

STRV is a satellite of about 120 kg [RD4], box-shaped (~0.75 m size) and with spin stabilisation. It is designed to be launched by the A5 ASAP microsat configuration to a GTO orbit with a payload mass of about 25 kg. The orbit insertion is direct, thus no main propulsion is carried on board, only AOCS thrusters. The internal configuration is made up shelves for equipment accommodation, and the load carrying structure is represented by shear panels.

In the case of option 1 (GSLV launch), due to the peculiar structure required by the launch stack configuration and the very large propulsion system, only a custom design can be considered. As a reference, a design similar to the STORMS mission can be assumed.

The adaptation of STRV can be considered in principle for design option 2 (ASAP5 launch) as defined in 5.4.3.1.

The two S/C design options analysed are summarised in the following Table 5-6.

Mission		1	2
Number of Satellites		4.0	4.0
Orbit-type		Elliptic 12-hour (sync)	Elliptic 12-hour (sync)
Perigee (Km)		650.0	650.0
Apogee (km)		39717.0	39717.0
Inclination (deg)		10.0	7.0
Launch Date		2006	2006
System			
Satellite Type		STORMS type	STRV-adapted
Existing Platforms Identified			
Dry Mass-class (kg)		<1000	<300
Stabilisation		Spin-stabilized	Spin-stabilized
Launcher			
Launcher		GSLV	AR5 ASAP Minisat
Launch strategy		LEO+own prop	GTO as piggy-back + natural apsides drift (600 days max)+own prop
Payload			
Instrument set		nominal set (High energy Ion Spectrom, Thermal Plasma Monitor, Mid Energy particle Monitor, Magnetometer, GPS receiver, Waves instrument)	nominal set (High energy Ion Spectrom, Thermal Plasma Monitor, Mid Energy particle Monitor, Magnetometer, GPS receiver, Waves instrument)

Table 5-6: IMM Design Options

5.4.4 Trade-Offs

The options described above were traded in order to define a baseline for the S/C design.

A more quantitative look into option 2 showed that heavy modifications would be required to the STRV platform to adapt it to the IMM mission requirements:

- A pair of booms should be added on the top surface of the S/C
- A main propulsion system should be added
- The internal shelf structure should be re-designed to accommodate the propulsion system, that, in the case of a 490 N bi-propellant engine, would require a couple of tanks for about 70 kg propellant and a total height of about 0.8 m (including nozzle)
- The external panels should be stretched up to the maximum allowed in A5 ASAP minisat in order to increase the solar cell area and the power available on board, due to the requirement of continuous observation and downlink
- The payload should be re-allocated
- The Data Handling system would require upgrades
- The mechanical I/F to the launcher must be upgraded from the microsat one to the minisat one

These modifications are illustrated below.

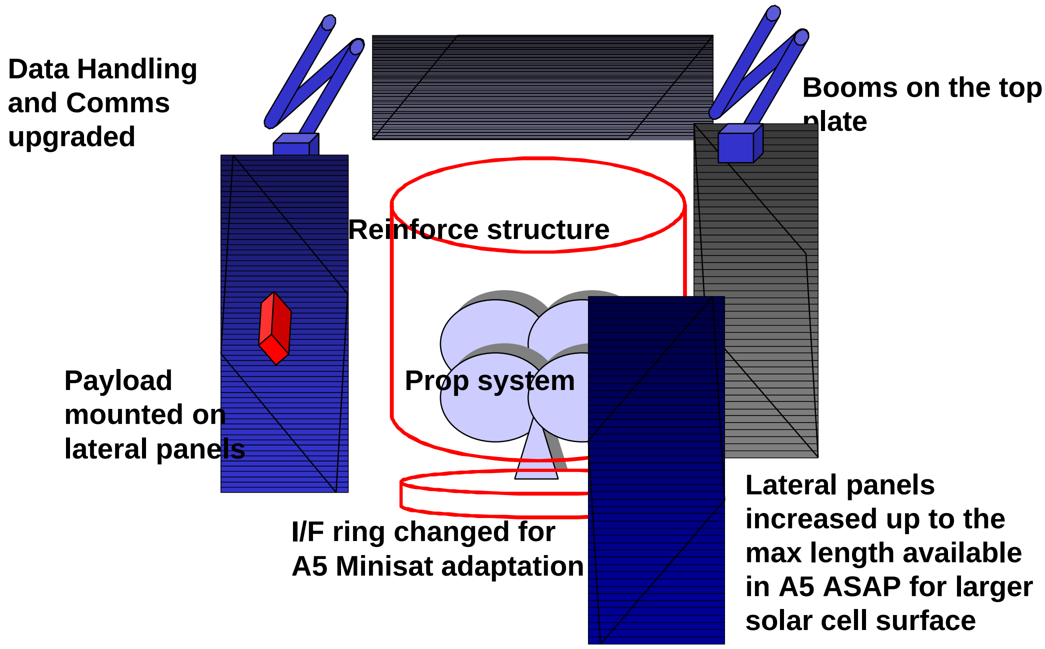


Figure 5-13: Modifications Required to STRV for IMM Use

Although these modifications are possible within the 300 kg mass available, the re-use of STRV hardware would be minimum, and thus so would the cost saving.

In addition, as shown in section 5.2.8, the time to deploy the constellation in its final configuration would be extremely long (up to 600 days), eating up a large fraction of the lifetime of the S/C and impacting the total system cost because of the more frequent S/C replacement needed.

Finally the A5 ASAP minisat launch configuration is still a rather expensive one (~40 Meuro) and the saving compared to a GSLV launch (~50 Meuro) is not very high.

Concluding, the custom design option with GSLV launch has been preferred as baseline because of its higher mass and configuration design flexibility and because of the shorter time to achieve the final constellation. The minisat option is technically feasible but its cost effectiveness could not be proven within this study.

Future evolutions in the European launcher programme, e.g. Vega or Soyuz launched from Kourou, could re-open the above trade-off.

5.4.5 Baseline Design

The baseline IMM design resembles the CLUSTER and STORMS designs for simplification and reuse of components.

As shown in Figure 5-14 and Figure 5-15 the S/C configuration is largely driven by the accommodation of the very large propellant tanks and by the required external surface area for solar cells. GaAs technology has been selected for this application to keep the external area and the S/C mass within the launcher limit. However, large margins have been taken due to the uncertainty on the cell degradation in the IMM radiation environment.

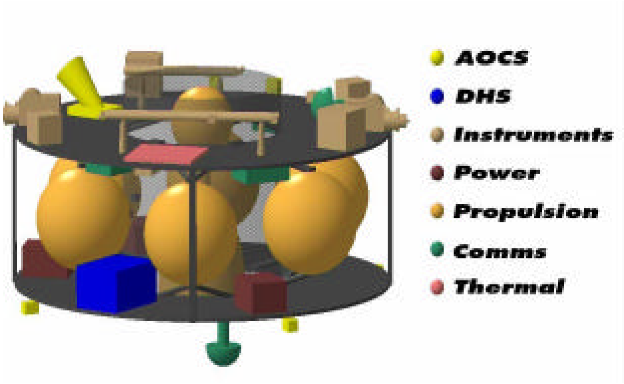


Figure 5-14: IMM Internal Configuration

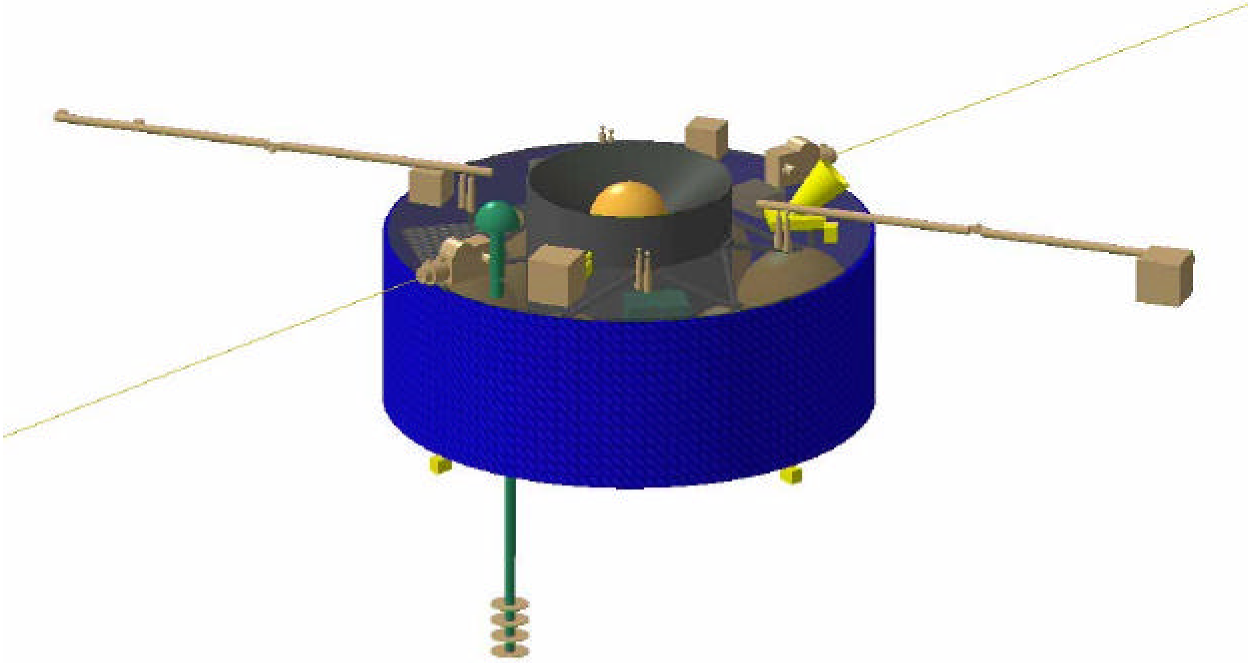


Figure 5-15: IMM External Configuration

All components are off-the-shelf to reduce the cost, including the tanks, which are taken from CLUSTER.

Local shielding has been considered in the case of the CDMU unit, for which a 6 mm Al case box has been considered. This allows a reduction of the total dose over the whole mission to less than 100 krad.

The instruments have all been accommodated on the top plate of the S/C apart, from the Magnetometer and Waves instruments, which use a pair of solid deployable booms and two wire antennas. These are also located on the top plate.

As described in more detail in section 5.14, the internal load carrying cylinder structure has been dimensioned for vibration stability at launch taking into account the unusual configuration of four spacecraft stacked inside the GSLV fairing (illustrated in Figure 5-21).

5.4.6 Modes of Operation

The system Modes of Operation for the IMM baseline mission are shown in the following table.

No.	Mode Name	Definition	Acronym
1	Launch Mode	*From lift-off until parking orbit*	LM
		All subsystems are off except essential equipment	
		Essential load = RX, CDMU + Heaters	
		Battery fully charged (charging until approx. 8 min before lift-off)	
		Launch by GSLV into 250 km 18° inclination LEO orbit	
2	Separation and Transfer Mode	*Parking orbit until final constellation acquisition*	TM
		Spin axis needs several reorientations	
		Power is from battery plus SA	
		The antenna of the top S/C is automatically deployed and it is activated	
		The top S/C separates and when at the right anomaly fires the main engine	
		The next S/C fires its main engine to achieve an elliptical orbit 90° separated from the apsidal line of the previous one (this operation is repeated by the 3rd and 4th S/C also).	
		Only TT&C, data handling and propulsion are on	
		Separation mechanism	
3	Initialisation Mode	*From final orbit acquisition to start of operations*	IM
		AOCS initialisation, spin-axis manoeuvre and sun acquisition	
		S/C spinning rate brought to 5 rpm	
		All subsystems initialised	
		Booms are deployed	
		P/L initialised	
4	Operational Mode	*Nominal operational mode*	OM
		Increase S/C spin rate up to 15 rpm	
		Standard science operation (all instruments on)	
		Continuous data communication S/C via LGA	
		Power generation and distribution to all S/S and Instruments (2 hr 22 minutes max eclipse)	
		AOCS nominal operations (spin rate and axis orientation control)	
		Passive thermal control	
		Data dumping to onboard memory during perigee passages	

No.	Mode Name	Definition	Acronym
5	Eclipse Mode	*Eclipse*	EM
		Instruments stay on also during eclipse	
		TT&C and DH on	
		Power from the battery	
		Thermal control via heaters	
6	Safe Mode	*Failure Recovery mode:*	SM
		S/C in sun re-acquisition mode	
		Instruments in survival mode	
		Non-essential functions are halted.	
		TM/TC access to DHS is guaranteed to enable failure detection and reconfiguration.	
		Failure isolation and recovery are executed by the ground.	

Table 5-7: Modes of Operation

5.4.7 Mass Budget

The mass identified in the system budget is based on the specified values of the individual units and subsystems. Depending on the maturity status of the items, contingency is applied at unit/item level. Generally, for each piece of equipment a mass margin is applied in relation to its level of development: i.e. 5% for off-the-shelf items, 10% for items qualified but requiring some modification and 20% for items to be developed.

For the payload, the margin has been calculated as a weighted average of the margins for the units making up the different instruments, according to their degree of development.

The S/C mass budget for the baseline is displayed in the table below:

Inner Magnetospheric Monitor Mass Budget					
		Target Spacecraft Mass at Launch		4050	kg
		Below Mass Target by:		6	kg
	Without Margin	Margins		Totals	% of Total
		%	kg	kg	
1. Structure	92.4 kg	18.8	17.4	109.9	10.94
2. Thermal Control	13.9 kg	10.0	1.4	15.3	1.52
3. Mechanisms	24.0 kg	10.0	2.4	26.4	2.63
4. Pyrotechnics	1.5 kg	5.0	0.1	1.6	0.16
5. Communications	10.5 kg	5.0	0.5	11.0	1.10
6. Data Handling	15.0 kg	10.0	1.5	16.5	1.64
7. AOCS	8.1 kg	10.0	0.8	9.0	0.89
8. Propulsion	72.3 kg	5.0	3.6	75.9	7.56
9. Power	40.9 kg	10.0	4.1	44.9	4.47
10. Harness	12.6 kg	20.0	2.5	15.1	1.50
11. Payload Allocation	25.1 kg	8.0	2.0	27.1	2.70
Total Dry (excl.adapter) - per sat.	316.3 kg			352.6	35.11
System Margin (excl.adapter)		17.0 %		59.9	
Total Dry with Margin (excl.adapter) - per sat.				412.6	41.08
	Propellant:	Total propellant		591.9	58.92
				0.0	
		Adapter Mass (incl. Sep. Mech.)		50.0	4.98
Total Launch Mass (single satellite)				1004.4	
Total dry mass of last satellite of the stack (excludes separation mech)				344.4	
Total dry mass of last satellite with margin				403.0	
Propellant of last satellite				577.7	
TOTAL MASS OF LAST SATELLITE				980.7	

Table 5-8: IMM Mass Budget

Three S/C are identical while the one on the top of the stack does not carry a separation mechanism on the top plate.

Noticeably, the propellant mass accounts for almost 60% of the total S/C mass. In this case a small variation of the dry mass of the S/C causes a large variation of the total mass, due to the changed propellant mass needed.

A system margin of 17% has been achieved. This is considered a bit tight for an assessment study; however, some mass gain is expected by a more optimised dimensioning of the S/C load-carrying cylinder, that can be made thinner for the higher S/C on the stack.

Clearly, a large margin would be available in the case of a constellation of only 3 S/C, while PSLV would be able to carry only 2 such S/C in their present configuration (see Table 5-4).

5.4.8 Power Budget

The six operational modes have been used to dimension the power subsystem. The corresponding S/C power demands are given in the table below.

		Instr.	Thermal	AOCS	Comms	Propulsion	OBDH	Power Cons.	Pyro	Mech	Harness (excl. PSS)	TOTAL CONSUMPTION
Mode names are linked		manual	manual	manual	manual	manual	linked	computed	manual	manual	computed	
Launch Mode	MAX	0	30	0	10	0	48	31	0	0	1.8	120
	NOM	0	30	0	10	0	32	26	0	0	1.4	99
	MIN	0	0	0	10	0	10	20	0	0	0.4	41
Separation and Transfer Mode	MAX	0	20	5	18	0	48	31	0	0	1.8	123
	NOM	0	25	5	18	0	32	26	0	0	1.6	107
	MIN	0	0	5	18	0	10	20	0	0	0.7	54
Initialisation Mode	MAX	36	20	5	18	0	48	31	0	0	2.5	160
	NOM	33	20	5	18	0	32	26	0	0	2.2	136
	MIN	33	0	5	0	0	10	20	0	0	1.0	70
Operational Mode	MAX	36	20	5	18	0	48	31	0	0	2.5	160
	NOM	33	20	5	18	0	32	26	0	0	2.2	136
	MIN	33	0	5	18	0	10	20	0	0	1.3	88
Eclipse Mode	MAX	36	25	5	18	0	48	31	0	0	2.6	165
	NOM	33	30	5	18	0	32	26	0	0	2.4	146
	MIN	33	0	5	18	0	10	20	0	0	1.3	88
Safe Mode	MAX	0	25	5	18	0	48	31	0	0	1.9	128
	NOM	0	30	5	18	0	32	26	0	0	1.7	112
	MIN	0	0	5	18	0	10	20	0	0	0.7	54

Table 5-9: IMM Power Budget

5.4.9 Conclusions and Open Points

An IMM S/C based on a STORMS-type, custom, spin-stabilised design is proposed as baseline.

The design fulfils the Space Weather user requirements apart from a gap in continuous coverage of the constellation of about 30 minutes.

The design is based on off-the-shelf components and no specific technology development is needed. The type of spacecraft is rather conventional and large expertise exists in Europe on it. Due to the peculiar orbit and constellation requirements, a commercial platform or an existing design cannot be re-used.

If a 3 S/C-only constellation, and the consequent degradation in the data return, can be accepted, then a cheaper launch with PSLV can be considered; but a redesign of the IMM structure would be needed to achieve the launcher mass performance.

The following points require further investigation in the next design phase:

- Increase of mass margin at launch for GSLV and better evaluation of GSLV performance, requirements and availability
- More detailed radiation analysis at component level
- Definition of a spare and replacement policy for the constellation. For instance, two spare S/C could be launched into parking orbits by PSLV and inserted into the constellation when needed.

This last point is especially critical from a cost point of view, because of the additional launch needed. Since the present assessment study is focussed on a pre-operational system only, the replacement strategy has not been analysed in detail. However, reliability considerations (section 5.16) suggest that a redundancy policy at spacecraft level is needed.

It is important to stress again that what is presented here is a solution to deliver a system conforming to the requirements for an **operational** system. Because this implies a set of satellites separated in local time, a solution with significant on-board propulsion is necessary. It is clear that low-cost solutions for one or several small satellites in GTO-like orbits are available, such as STRV. While these could be employed in a pre-operational or experimentation phase, piggy-back launches with Ariane 5 ASAP or similar cannot guarantee the required spread of satellites in local time. However, they might serendipitiously achieve it if the main payload launch times and dates were by chance optimal.

5.5 Configuration

5.5.1 Requirements and Constraints

The major drivers for the overall configuration can be summarised as follows :

- A spinning satellite with solar cells mounted on the outer wall;
- Accommodation of the payload instruments TPM, MEM, HEM, GRIS according to their pointing direction and field of view requirement
- Accommodation of the 6 large and heavy propellant tanks (sphere with 0.53m diameter)
- Available area for solar cells
- Accommodation of electronic boxes for Data Handling, Power, Communication
- Accommodation of the booms for antennas and payload instruments
- Stable mounting and accessibility to be guaranteed
- Compatibility with GSLV fairing envelope to accommodate a stack of 4 satellites

The spacecraft must provide accommodation to all the sub-systems and ensure compatibility between them throughout the mission. Therefore each of the constraints as listed above must be fulfilled for every operational mode and S/C attitude.

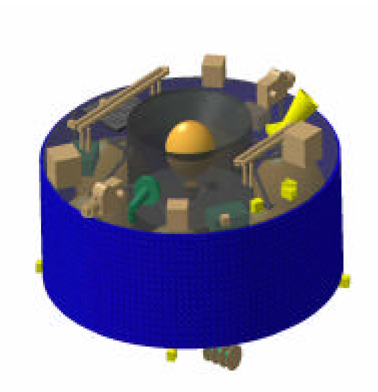
5.5.2 Spacecraft Baseline Design

The configuration is driven by the requirement for a single launch, together with the size of propellant tanks and solar cell area.

The resulting overall dimensions are:

- 4.2 m height (1.050 each spacecraft);
- 2.2 m diameter

Figure 5-16 and Figure 5-17 show stowed and deployed configurations respectively of the IMM spacecraft.



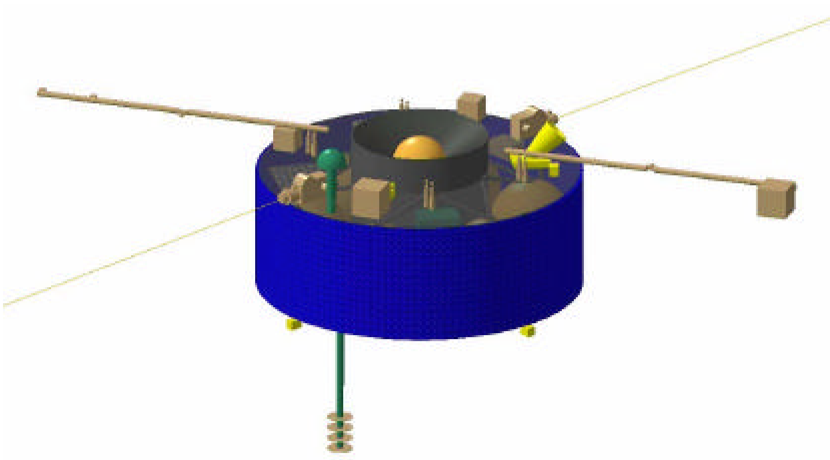


Figure 5-16: IMM Stowed **Figure 5-17: IMM Deployed Configuration**

The spin axis of the spacecraft is perpendicular to the sun-earth line. The spin axis is in the direction of the launcher axis. The solar array is mounted around the cylinder as shown in Figure 5-16. The S/C body is stiffened by two horizontal platforms that connect the central cylinder (d=900mm) and the outer cylinder as shown in Figure 5-19.

The propulsion system makes use of 3 propellant tanks, 3 oxidiser tanks, 1 pressurant tank, 1 main engine and 8 AOCS thrusters. The propellant tanks are radially accommodated inside the cylinder. The thrusters are located on the bottom part of the spacecraft. This is illustrated in Figure 5-18.

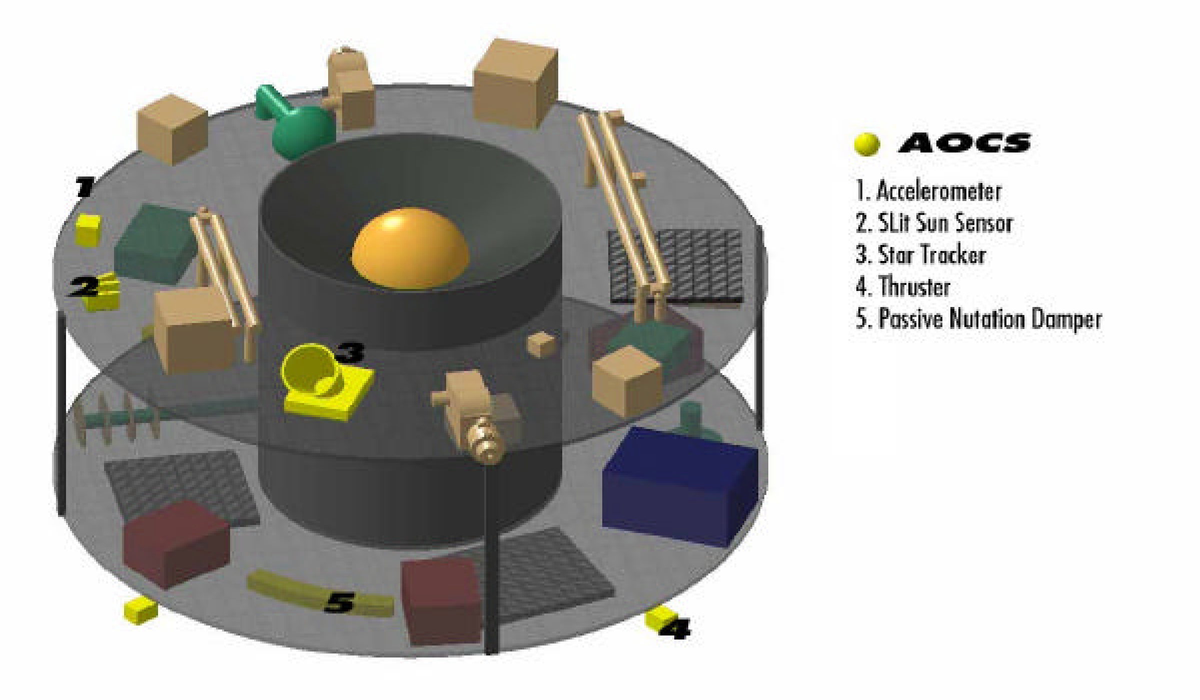


Figure 5-18: Accommodation of AOCS Units

Figure 5-19 shows the internal accommodation in the IMM spacecraft. The accommodation is summarised in Table 5-10.

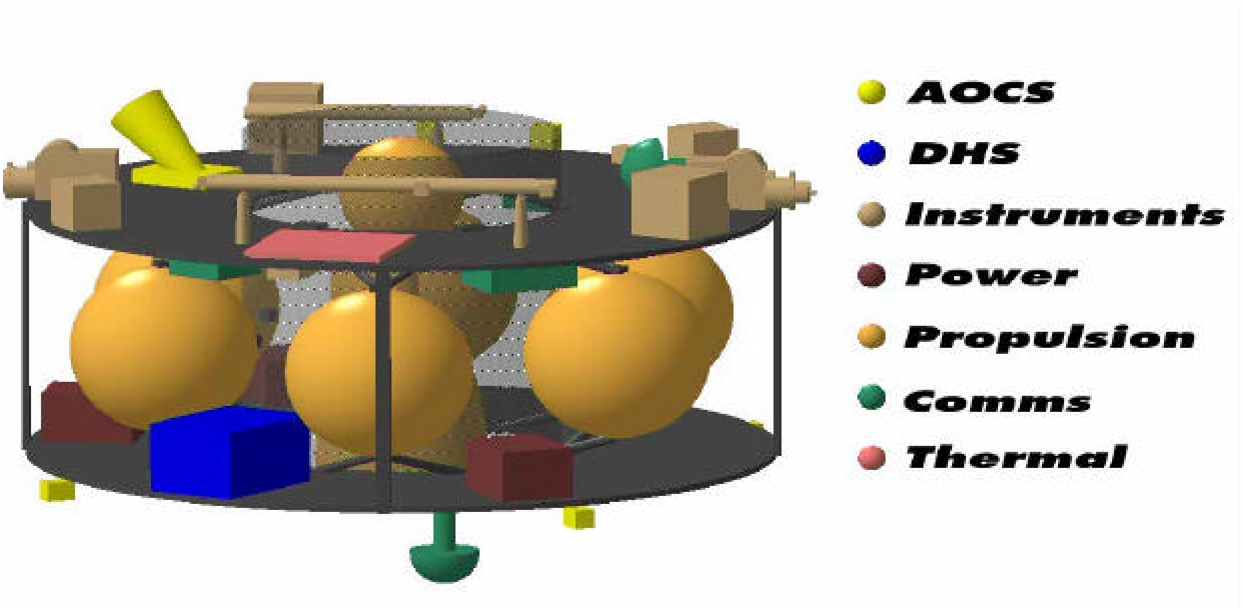


Figure 5-19: IMM Internal Accommodation

	Bottom Platform	**Equipment Platform**
AOCS:	• Thruster • Passive Nutation Dampers (2 units)	• Star tracker • Sun Sensor • Accelerometer
Telecomms:	• One deployable boom for a toroidal antenna • One fixed low gain antenna	• RFDU • Transponders (2x) • One deployable low-gain antenna
DHS:	• CDMU	
Payload		• TPM • MEM • HEM • 2 wires and a boom for WAVE • MAG • GRIS • Electronic boxes
Power:	• PDU • PCU • battery	
Propulsion:	• six propellant tanks	• Four thrusters • Pressurant tank
Thermal:	• radiators	• radiator

Table 5-10: Internal Accommodation of Units

The payload accommodation is illustrated in Figure 5-20.

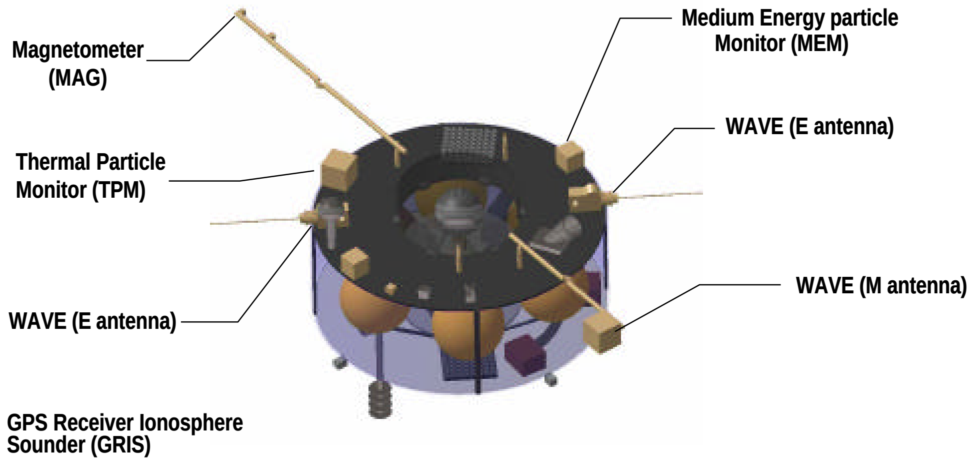
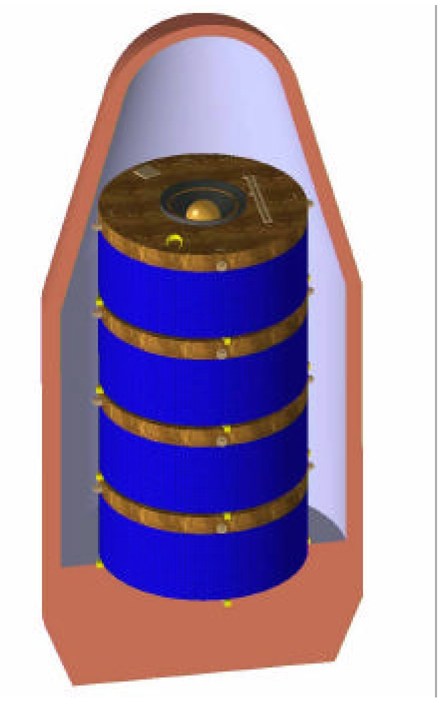


Figure 5-20: IMM Payload Accommodation

5.5.3 Launch Configuration

The following figure shows the four satellite constellation of the IMM spacecraft stacked in the GSLV fairing. The stack protrudes into the conincal part of the fairing, but sufficient clearance is available.



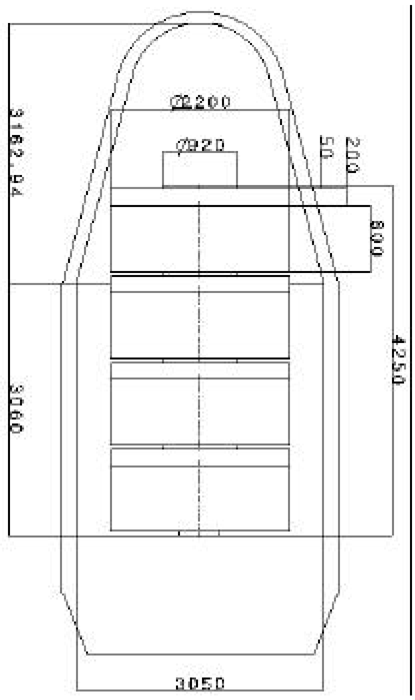


Figure 5-21: IMM satellites stacked in GSLV fairing

Figure 5-22: Dimensions of IMM in GSLV Fairing

5.6 Propulsion

5.6.1 Requirements and Design Drivers

The mission requires two large manoeuvres with a total *DV* of 2712 m/s:

- A velocity increment of 2500 m/s is required to boost the spacecraft from LEO to its mission orbit
- An additional *DV* of 212 m/s is required for inclination change from 18 to 10 degrees

Since the spacecraft is spin-stabilised, the amount of propellant required for attitude and orbit control is very small compared to the amount of propellant required for orbit transfer and inclination change. This has been included in the margin for the large engine propellant.

5.6.2 Baseline Design Description

A bi-propellant propulsion system has been chosen because of its high specific impulse. A solid propellant propulsion system was considered but was rejected because of the inertia characteristics of the satellite: the spin rate to compensate for thrust misalignment would become too high. A solid rocket motor would have required an additional dedicated small propulsion system for control of spin-rate, orbit, and attitude.

Considering all this, it has been decided to use the existing Cluster propulsion sub-system configuration as a starting point for the design of the IMM propulsion system, for cost reasons.

The 490 N bi-propellant engine will have to be fired several times in the intermediate orbit's perigee in order to reach the desired apogee height.

The number and durations of the firings and the total *DV* are within the engine's demonstrated performance.

The system comprises 6 propellant tanks and 1 pressurant tank.

The pressurant tank is filled with Helium. One high-pressure fill and drain valve is used to fill the tank for testing and mission. The Helium is isolated from the propellant by two pairs of normally-closed pyro-valves, one upstream and one downstream of the pressure regulator.

Figure 5-23 shows a block diagram of the IMM propulsion sub-system.

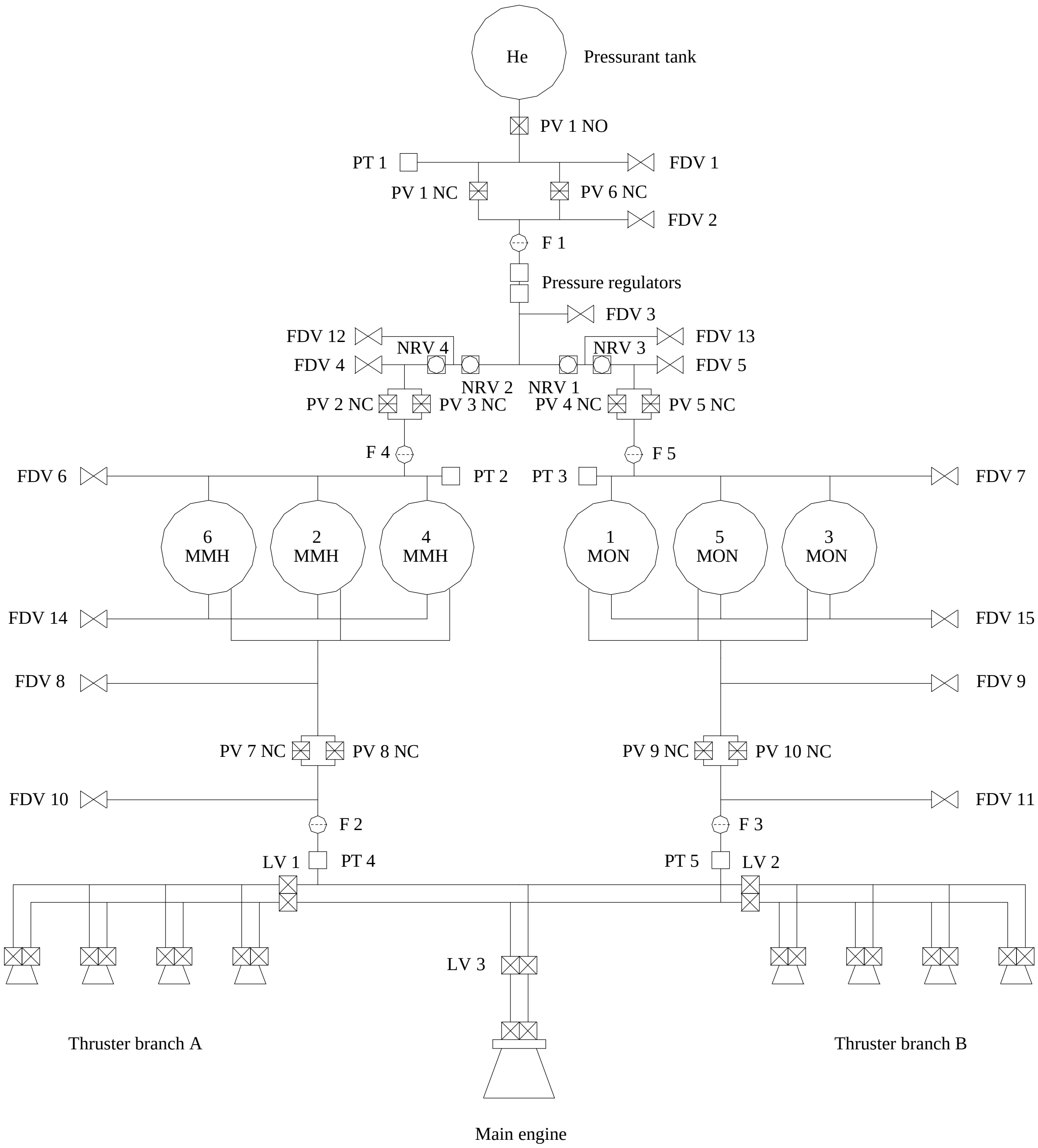


Figure 5-23: IMM Propulsion Subsystem Schematic

Two series of two non-return valves are installed in the pressurant feed lines to the propellant tanks in order to prevent mixing of oxidiser and fuel vapours in the common manifold.

The fill and drain valves are used for propellant loading and on-ground testing.

After RCS initialisation, the pressure regulator will provide regulated pressure to the propellant tanks and the main engine.

A normally-open pyrotechnic valve is provided to close the pressurant tank, to avoid the effects of possible leakage from the pressure regulators. This occurs after the final use of the main engine.

The proposed propulsion system design is, to a large extent, based on available COTS (Commercial Off The Shelf) components and not on optimised components that may need to be developed, especially if a lower number of propellant tanks were to be used. Substantially larger costs would have to be taken into account if a specific tank development were required. From a purely engineering perspective, the proposed design solution is likely not the optimum one, at least not from a dry mass point of view. The mass penalty for the non-optimised design was not traded-off at this stage against the potential cost advantage of using COTS tanks. For the time being it is assumed that the cost advantage of the proposed design outweighs the mass penalty.

Note: In recent years there has been a tendency to move away from the approach of creating branches with thrusters that could be closed in case one thrusters leaks. Using dual valve thrusters instead of mono valve thrusters does this. In this approach, latch valves 1 and 2 could be eliminated. Also latch valve 3 could be replaced by a normally-open pyrotechnic valve, which can be closed after the operation cycle of the main thruster. This approach could be followed in a later phase of the design.

Provision has to be made for propellant flow from one tank to the other when the tanks are filled and the satellite is mated to the launch vehicle in a horizontal position.

5.6.3 Mass Budgets

A mass breakdown of the propulsion system is listed in Table 5-11.

Component	Quantity	Component mass (kg)	Mass (kg)	Mass inc. margin (kg)
Propellant tank	6	6	36	
Pressurant tank	1	10.7	10.7	
Pipework and harness			7.9	
Main thrusters	1	3.4	3.4	
RCS thrusters	8	0.6	4.8	
Fill/drain valves	15	0.08	1.2	
Helium filter	1	0.09	0.09	
Propellant filter	4	0.28	1.12	
Pyro valve Normally Open	1	0.145	0.145	
Pyro valve Normally Closed	10	0.145	1.45	
Non return valve	4	0.1	0.4	
Pressure regulator	2	0.9	1.8	
Latch valve single	3	0.18	0.54	
Pressure transducer	5	0.25	1.25	
Pressurant Helium			1.5	
Propulsion system dry mass without margin			72.3	
Propulsion system dry mass with 5% margin				75.9
Propellant (incl. 1 % residuals)				591.8
Propulsion system wet mass				667.7

Table 5-11: Mass Breakdown of the IMM Propulsion System

Since the top satellite of the stack of 4 satellites does not need a separation mechanism, the propellant load for this satellite is 2.4% lower than the original propellant load, i.e. 577.7 kg.

5.7 Thermal Control

5.7.1 Requirements and Design Drivers

The spacecraft thermal control subsystem shall keep the temperatures of the spacecraft subsystems and the instruments within specified temperature limits during all expected mission phases and operation modes. An extended temperature range has been assumed for the instruments, as the limits that were originally defined were considered largely conservative enough.

The temperature limits have been assumed as follows:

	Operational	**Non-operational**
Instruments	-10°C/+40°C	-20°C/+55°C
S/C electronics	-20°C/+40°C	-30°C/+60°C
Batteries	+5°C/+35°C	0°C/+40°C
Tanks and valves	+5°C/+15°C	+5°C/+15°C

Table 5-12: Assumed Temperature Limits

5.7.2 Baseline Design

5.7.2.1 Spacecraft

The thermal design philosophy used for the IMM is based on the use of passive techniques, with the addition of heater power for special tasks (e.g. eclipse, safe mode). The design takes into account CLUSTER programme past experience, adapted and extended as necessary to meet the particular requirements of the IMM.

One difficulty concerning the thermal design of the IMM is due to the inclination of the S/C to the ecliptic plane. The top surface of the S/C cylinder (to be used to radiate the heat dissipated by the P/L to space) can be inclined by up to 23.5° towards the sun (NB assuming an orbit inclination of 0°, not 10°) and thereby absorb solar irradiation.

5.7.2.2 Payload

The thermal control design for the P/L has been defined in a way that even in an orbit with perigee in opposition to the sun and no heat fluxes on the top or bottom side, no additional P/L heating is required. Additional heating is required only for eclipse and safe modes.

5.7.2.3 Overall

Particular features of the Thermal Design are:

- Multi-Layer Insulation (MLI) Blankets and double foil trimmed as necessary to have a better heat rejection to deep space and therefore to minimise heat absorption from solar irradiation.

The blankets comprise Aluminised Mylar and/or Kapton sheets and an electrically conductive outer sheet or laminate grounded to the S/C structure in order to prevent differential electrostatic charging.

- The Solar Arrays are thermally insulated from the S/C structure to minimise the required heater power during eclipse.
- OSR radiators are placed on the top and bottom side in order to radiate the heat dissipation from the instruments to deep space. The total radiator surface needed in the hot case is 0.7m². S/S with a higher power request (e.g. communication system) shall predominantly be mounted in contact with the radiators. Since most of the electronic boxes are located on the bottom of the S/C, most of the radiators have been placed there also. Contamination effects on the OSR due to the engine need to be investigated.
- S/C internal surfaces shall generally have a high emittance finish to aid radiative heat transfer and to minimise the temperature gradients within the S/C. Therefore all internal surfaces need to be painted black.
- To maintain temperatures on the propulsion S/S (tanks and valves) and the batteries, they are thermally insulated from the S/C internal environment. For thermal control of the propulsion S/S and the batteries, several heater lines are needed, providing a total heat of about 20W.
- During eclipse mode and safe mode additional heaters controlled by thermostats provide control of minimum temperature. The required heater power is about 10W. The temperature control can be performed at element level.

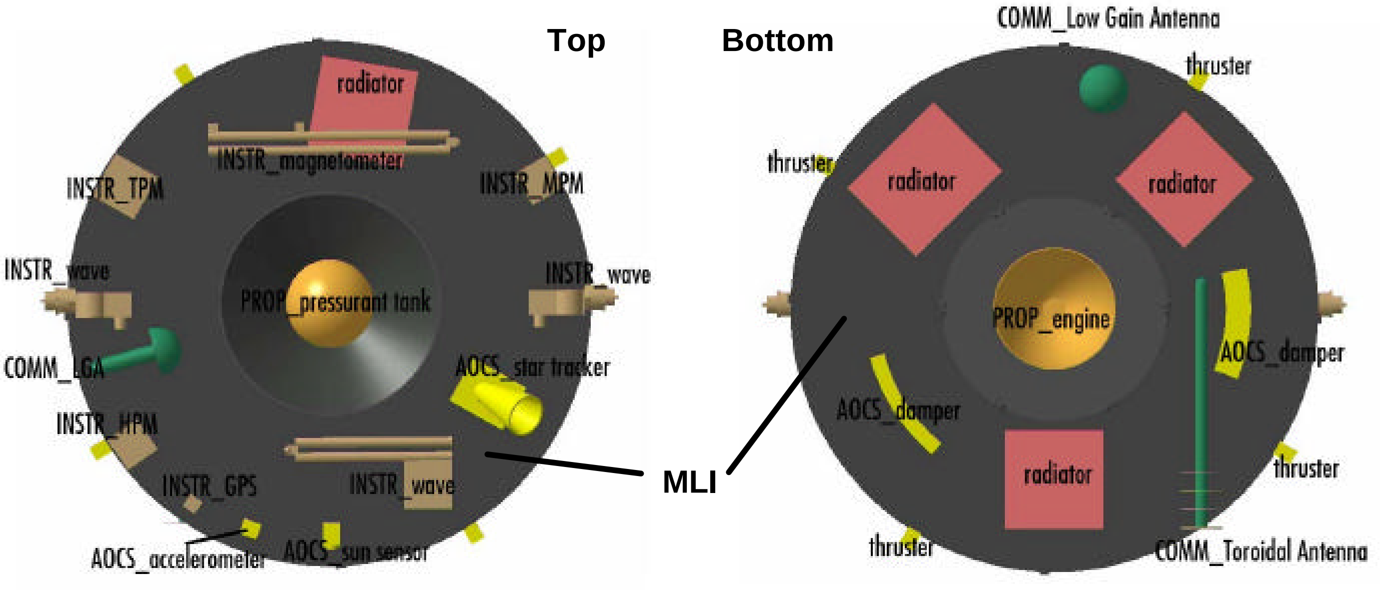


Figure 5-24: Locations of Thermal Control Elements

5.7.3 Budgets

5.7.3.1 Mass Budget

The preliminary mass budget for the IMM thermal control subsystem is provided in Table 5-13.

Item	Estimated Mass [kg]	Uncertainty 10% [kg]	Total Item Mass [kg]
MLI/Foil	7.6	0.76	8.4
Radiators (OSR)	0.5	0.05	0.6
Heaters/Thermostat/other	5.8	0.58	6.3
TOTAL	13.9	1.39	15.3

Table 5-13: Thermal Control Mass Budget

5.7.3.2 Heater Power Budget

Table 5-14 gives a summary of the preliminary power budget.

Mode	Environment	Max. Heater Power	Comment
Launch mode	LEO	30W	Transient cool-down
Separation and Transfer Mode	Sunlight	20W	
Initialisation Mode	Sunlight	20W	
Operational Mode	Sunlight	20W	
Eclipse Mode	Eclipse (2h 22mn)	30W	Transient cool-down
Safe Mode	Sunlight	30W	

Table 5-14: Thermal Control Power Budget

5.8 Power

5.8.1 Requirements and Design Drivers

Design drivers for the Power Subsystem definition have been the following:

- Attitude control is performed by spinning
- Solar array is body-mounted with a maximum projected area of 1.9 m^2
- Eclipse average power: 170W
- Sunlight average power: 150W
- Additional 10% margin considered at PSS sources
- Solar Array Temperature: 45°C
- Radiation environment as described in section 5.3.3
- Battery sized to operational needs during eclipse.
- Eclipse duration: 2.2 hours[1]
- Orbit period: 12 hours
- Maximum SA pointing deviation with respect to Normal: 23 degrees (calculated assuming an orbit inclination of 0°, not 10°; the impact of this is minor)
- 28V fully-regulated power bus for EMC cleanliness and SA minimisation

5.8.2 Assumptions and Trade-Offs

The maximum End of Life (EOL) Solar Array (S.A.) power need has been derived from the above mentioned requirements, and is 282 Watts.

For this EOL power, several parameters have been estimated for different cell technologies. Table 5-15 below sumarises the trade-off.

Configuration		Si BSR	Si HI-ETA3	GaAs/Ge	Multi-Junction GaAs
Efficiency (at AM0, 25°C)	(%)	14.0	16.5	19.5	24.7
S.A. Area	(m^2)	13.2	12.4	8.58	5.99
S.A. Projected Area	(m^2)	4.21	3.94	2.73	1.91
PVA[2] Mass	(kg)	32.2	28.4	26.5	17.5
PVA FM RECURRING COST	(%)	100	106	126	135

Table 5-15: Comparison of Solar Array Technologies

As it may be seen, Multi-Junction GaAs technology is the only one which will fit in the available projected area, and this has therefore been selected as the baseline for the solar array cells.

[1] Eclipse duration is actually 2.37h; this has a minor impact on the battery DOD.
[2] Photovoltaic Assembly (PVA) shall be understood as the Solar Array without the Panel Structure

5.8.3 Baseline Design

5.8.3.1 Power Bus

A 28V fully regulated power bus is provided to the different Main Bus users through protected power lines, as shown in the block diagram (Figure 5-25).

5.8.3.2 Electronics

Two electronic boxes, one Power Conditioning Unit (PCU) and one Power Distribution Unit (PDU), are foreseen for proper power bus regulation and distribution.

The PCU consists of:

- Two (2) 280W Battery Discharge Regulators (BDRs)
- Two (2) 200W Battery Charge Regulators (BCRs)
- Fifteen (15) Solar Array Regulator (SAR) modules
- One 2/3 Majority Voter Error Amplifier generating reliable regulator control signals

The PDU consists of:

- Latching Current Limiters for power bus protection
- Transistor switches for thermal control
- Pyrotechnic drivers

BDRs, BCRs and SAR sections operate in hot redundancy, so that the PCU is one-failure-tolerant with no reconfiguration needs. PDU failure tolerance relies on the usual cold-redundant approach.

5.8.3.3 Battery

The battery has been sized to the operational needs during eclipse: that is, for an eclipse average power of 170W and an eclipse duration of 2.2 hours. Besides the considered System margins, an additional 10% margin has been assumed with respect to the described Power requirements.

Li-Ion technology has been selected due to its energy density performance with respect to other types. To achieve maximum performance of this type of battery in a five-year mission, a relatively tight operating temperature (+20 to +30°C) is required. Being aware that such a temperature range may not be guaranteed at this stage, and due to the relatively low data available for the expected number of cycles, a conservative 40% DOD has been assumed. As a result, 1000Wh is the maximum acceptable energy consumption for Separation/Transfer phase.

As is usually done with Li-Ion batteries, battery redundancy at cell level (not at unit level) is considered.

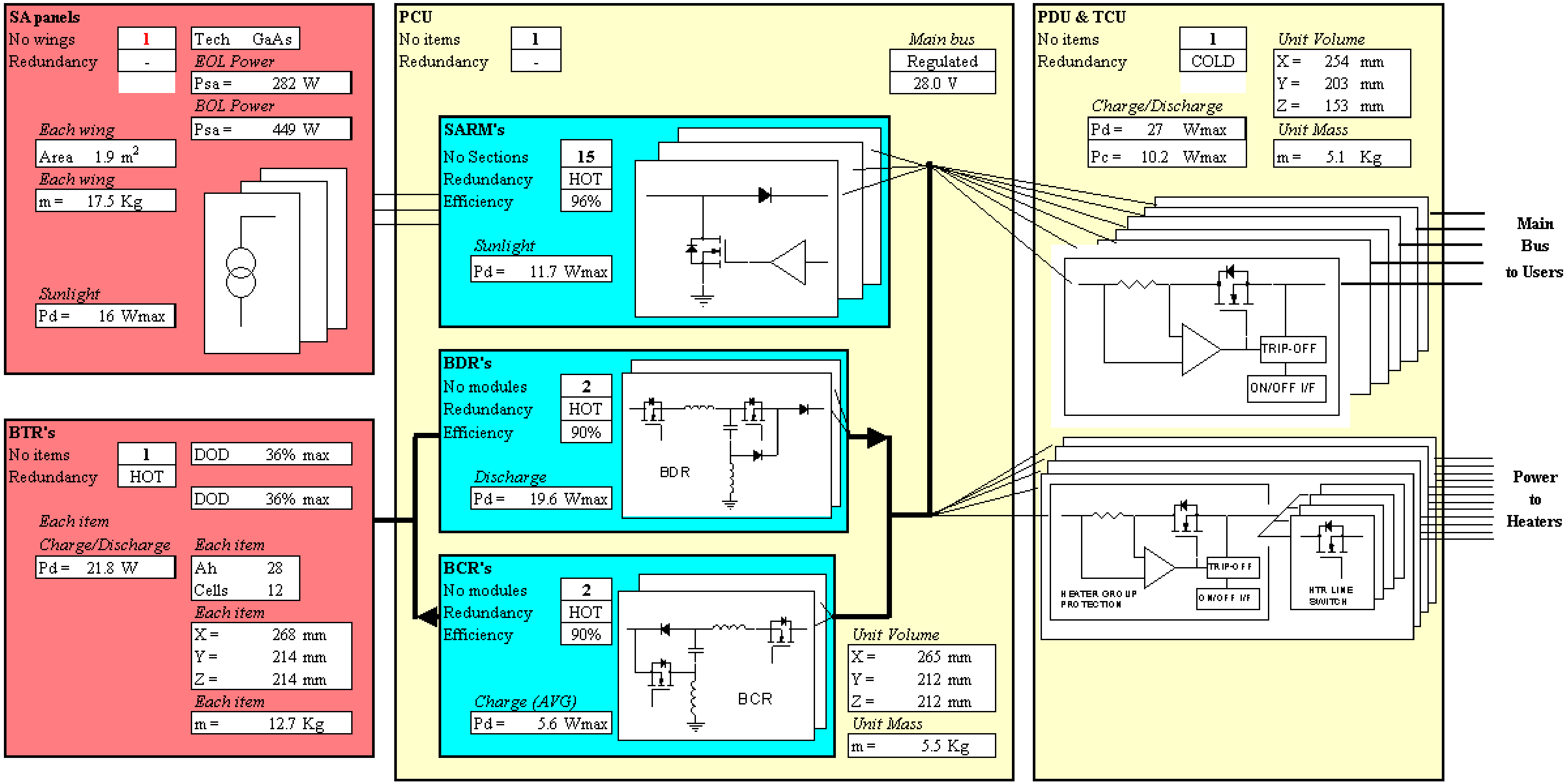


Figure 5-25: Power Subsystem Block Diagram

5.8.4 Budgets

The overall mission power consumption budget is given in section 5.4.8. The power dissipation is given below.

		PCU	PDU	TCU	BATTERY	PSS Harness	PSS TOTAL DISSIPATION	S/C TOTAL DISSIPATION
Launch Mode	MAX	24	14	11	15.9	5.0	71	160
	NOM	20	12	9	13.1	4.1	58	132
	MIN	11	8	7	5.4	1.7	34	54
Separation and Transfer Mode	MAX	25	14	11	4.2	2.2	57	150
	NOM	22	12	9	3.7	1.9	49	130
	MIN	13	9	7	1.8	1.0	32	66
Initialisation Mode	MAX	30	16	11	5.5	2.8	65	195
	NOM	25	13	9	4.6	2.4	55	165
	MIN	15	9	7	2.4	1.2	35	85
Operational Mode	MAX	29	16	11	5.5	6.6	68	197
	NOM	25	13	9	4.6	5.6	57	168
	MIN	17	10	8	3.0	3.6	41	109
Eclipse Mode	MAX	30	16	11	21.8	6.8	86	220
	NOM	26	13	9	19.4	6.1	74	195
	MIN	17	10	8	11.7	3.6	50	117
Safe Mode	MAX	26	15	11	4.4	2.3	58	156
	NOM	22	12	9	3.8	2.0	50	137
	MIN	13	9	7	1.8	1.0	32	66

Table 5-16: IMM Power Dissipation

The Power S/S mass breakdown is given in Table 5-17 below.

S/S Item	Mass (kg)
Solar Array (Multi-Junction GaAs, w/o substrate structure)	17.5
Battery (Li-Ion)	12.7
Electronics (PCU/PDU/TCU)	10.6
PS/S Total	40.9

Table 5-17: Power Subsystem Mass Breakdown

5.9 Mechanisms

The identified mechanisms for the Space Weather IMM satellite are:

- 1 toroidal antenna deployment boom
- 1 low gain antenna deployment boom
- 1 deployable boom to position the search coil magnetometer sensor far from the spacecraft (WAVE instrument)
- 1 deployable boom to position the flux-gate magnetometer sensors far from the spacecraft (MAG instrument)
- 1 separation mechanism per spacecraft to separate them from each other after launch

5.9.1 Requirements and Design Drivers

5.9.1.1 Toroidal Antenna Deployment Mechanism

The required length of this antenna deployment boom is mainly dependent on the radiated shape of the toroidal antenna with respect to ground station locations. The foreseen position of this antenna on the spacecraft and its radiated shape lead to a boom length of 0.8m. This boom needs to be deployed after each satellite separation.

This antenna shall be located on the bottom of the spacecraft.

Therefore, the foreseen mechanisms are:

- 1 deployment mechanism
- 1 hold-down and release mechanism

5.9.1.2 Low gain Antenna Deployment Mechanism

The required length of this antenna deployment boom is mainly dependent on the radiated shape of the low gain antenna. The foreseen position of this antenna on the spacecraft and its radiated shape lead to a boom length of 0.2m. This boom needs to be deployed automatically after satellite separation.

This antenna shall be located on the top of the spacecraft.

Therefore, the foreseen mechanisms are:

- 1 deployment mechanism
- 1 hold-down and release mechanism

5.9.1.3 WAVE and MAG Boom Deployment mechanisms

Two flux-gate magnetometer sensors are required. The required radial distances of the two flux-gate magnetometer sensors (MAG instrument) with respect to the spacecraft body have been estimated at 1.5m and 2m. Both sensors should be mounted on the same boom. This 2m boom shall be compacted for stowing on the spacecraft during launch. Due to the available volume on top of the spacecraft, the boom will be stowed as a two-section boom (2 x 1m booms).

The required distance of the search coil magnetometer (WAVE instrument) with respect to the spacecraft shall be more than 1m.

In order to optimise the dynamic balance of the spacecraft, the two booms (one for the search coil magnetometer and one for the flux-gate magnetometer) will be located on opposite sides on top of the spacecraft and will preferably be deployed simultaneously.

In this case the optimised length of the search coil magnetometer boom shall be 1.3 m.

This 1.3m boom shall be stowed on the spacecraft during launch. Due to the available volume on top of the spacecraft the boom will be stowed as a two-part boom (2 x 0.65m booms).

The spin of the satellite will provide the booms' main deployment motorization. To decrease the shock level at the end of the deployment, a speed deployment regulator based on low melting temperature alloy will be used at each hinge level.

Therefore, the foreseen mechanisms are (for each of the two booms):

- 2 hinge deployment mechanisms (with integrated speed deployment regulator)
- 2 hold-down and release mechanisms

5.9.1.4 Spacecraft Separation Mechanisms

In order to get a good separation between the satellites, a standard clamp band separation mechanism has been selected. The size of this clamp band separation mechanism is linked to the size of the central tube of the spacecraft.

Therefore, the separation mechanism per spacecraft is:

- 1 standard clamp band separation mechanism (including 2 pyros)

5.9.2 Assumptions, Trade-Offs and Baseline Design

The approach which has been followed to identify the conceptual design of Space Weather IMM satellite mechanisms has been to use (as far as possible) qualified, off-the-shelf equipment, in order to reduce cost, procurement time, and development risks.

In the following paragraphs a short description of the anticipated mechanisms is provided, including a preliminary estimate of mass budgets.

5.9.2.1 Toroidal Antenna Deployment Mechanism

Boom

One short boom (around 0.8 metre length), carrying the toroidal communication antenna is foreseen to be deployed in order to allow the required field of view clearance. This boom will be realised with a composite standard carbon tube.

Deployment Mechanisms

Conventional spring-based systems are foreseen at the boom roots, to actuate the rotation, while a damping system plus latching device will be implemented to reduce the end-of-travel mechanical shock. A simple damping system (crushed honeycomb-like) can be foreseen for this application, because of the limited accuracy required on the antenna final position (of the order of one degree). The deployment angle is 180°.

Hold-down and Release Mechanism

One single hold-down / release point can be foreseen for this antenna boom, in order to provide adequate stiffness and strength in the stowed configuration. A pyro-actuated device (similar to a separation nut) can be used to actuate the separation.

RF junction

The RF junction (at deployment level) should preferably be done with flexible coaxial.

5.9.2.2 Low Gain Antenna Deployment Mechanism

Boom

One short boom (around 0.2 metre length), carrying the low gain communication antenna is foreseen to be deployed to allow the required field of view.

Deployment Mechanisms

Conventional spring-based systems are foreseen at the boom roots, to actuate the rotation. No damping systems are required in this case. The antenna end position will be maintained by the deployment spring pre-load itself.

Hold-down and Release Mechanism

A single hold-down/release point is used, in order to provide adequate stiffness and strength in the stowed configuration. A pyro-actuated device (like a separation nut) or lighter pin-puller device can be used to actuate the separation.

RF junction

The RF junction (at deployment level) should preferably be done with flexible coaxial

5.9.2.3 MAG and WAVE Boom Deployment Mechanism

The only difference between the two booms is their length. Rigid half-booms are foreseen for the deployment of the magnetometers. These booms are based on qualified booms used for the Cluster or Ulysses satellites. In order to decrease the end deployment shock and to be able to analyse the deployment kinematics, a regulator device will be integrated in each of the two hinges. The use of the regulator may lead to significant mass saving and better test representativeness. The stowed configuration of the booms will be one on each side on the top of the spacecraft.

Booms

The complete 1.3m and 2m booms will be made from two 1m and 0.65m booms respectively. In order to accommodate them in the available space on top of the spacecraft. The two booms will be realised from carbon fibre tubular structure, to provide good stiffness performance with respect to mass.

Deployment Mechanisms

Two hinges are required to deploy each complete magnetometer boom. One will be situated on the spacecraft, the second one will be situated between the two half-booms. Conventional spring-based systems will be used for deployment motorisation even if the centrifugal force developed by the spinning spacecraft is enough itself to completely deploy the booms. No shock-damping systems are foreseen in this case, thanks to the use of specific qualified regulators. These regulators are integrated at hinge level and are based on fusible metal technology. Therefore, by simply heating the regulator with 10W or 15W, the deployment time could be tuned to 4 to 6 min (at 0°C starting temperature).

An additional locking device will maintain the booms in deployed configuration after their successful deployment.

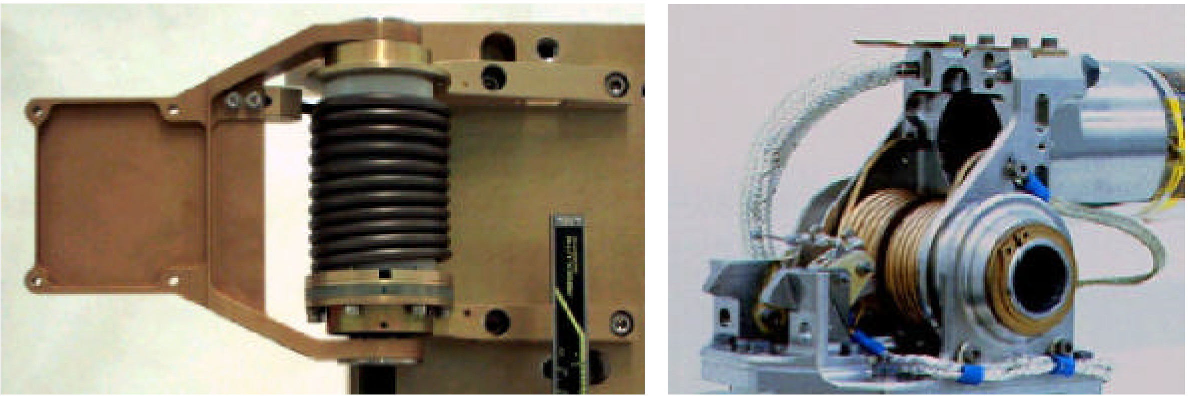


Figure 5-26: Regulated Deployment Mechanism Hinge (Source: SENER)

Hold-down and Release Mechanism

In this application, the foreseen hold down and release mechanisms might be an adaptation of those used on Cluster. The design is composed of a titanium clamp and a pyrotechnic separation nut. Two hold-down and release mechanisms will be required to clamp the booms to the structure during the launch. These hold-down and release mechanisms will be located close to each articulation's hinge, and will clamp both boom tubes together with the structure and will provide the required stiffness.

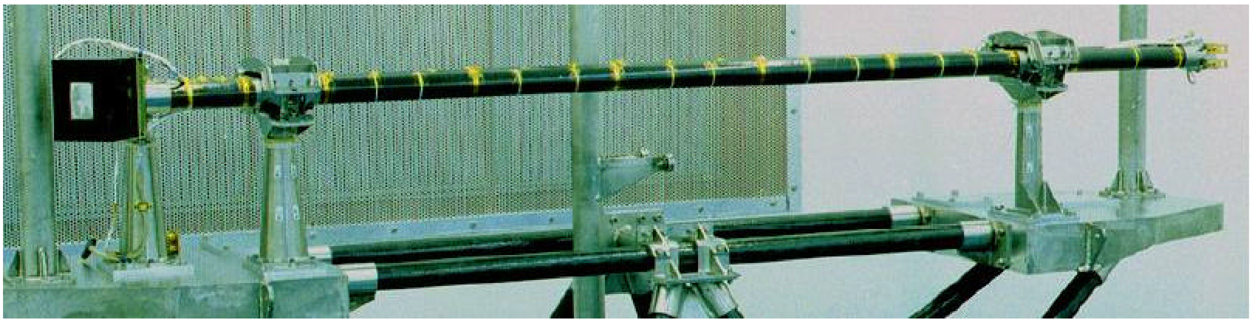


Figure 5-27: Magnetometer Boom Example (Source: SENER)

5.9.2.4 Spacecraft Separation Mechanisms

The recommended separation mechanism is a Saab Ericsson SS 937V. The clamp band technology is a well known technology for correctly separating spacecraft. This clamp band is sized for the worst case (the interface between the bottom spacecraft and the launcher) and will be used for all the four spacecraft, for similarity reasons. Note that the bottom one is not part of the spacecraft mass budget, but of the launcher.

The clamp band consists of a band with two connecting points. Band tension provides pressure on the clamp that attaches the satellite to the launcher (or the satellite underneath). Release is effected by two pyrotechnic bolt cutters. The retention set (part of the separation mechanism) secures a safe release behaviour and parks the clamp band on the launcher (or satellite underneath) after the satellite release.

The separation mechanism mass budget does not take into account the contact rings, which are part of the satellite (or launcher) structure.

Figure 5-28: Separation Mechanism (Source: SAAB)

5.9.3 Budgets

The estimated mass and power budgets are summarised in Table 5-18 below. Mass figures include the boom(s) and mechanical flanges.

Mechanism type	Number	Electrical Power	Number of Pyros	Unit Mass (no margin)	Deployment Time
Bottom toroidal antenna deployable boom with associated hold-down and release mechanisms	1 per satellite	0	1	2.1 kg	< 5 sec
Top low gain antenna deployable boom with associated hold-down and release mechanisms	1 per satellite	0	1 (or 1 pin puller)	1.6 kg	< 5 sec

Mechanism type	Number	Electrical Power	Number of Pyros	Unit Mass (no margin)	Deployment Time
Flux-gate Magnetometer boom composed of two regulated deployment hinges and two dedicated hold-down and release mechanisms	1 per satellite	20W to 30W requested during deployment for speed regulation	2	6.5 kg	4 to 6 min (at 0°C)
Search coil Magnetometer boom composed of two regulated deployment hinges and two dedicated hold-down and release mechanisms	1 per satellite	20W to 30W requested during deployment for speed regulation	2	6 kg	4 to 6 min (at 0°C)
Spacecraft separation mechanism	1 per satellite		2	7.8 kg[3]	< 1 sec
Total		**40W to 60W**	**8**	**24 kg**	

Table 5-18: Mechanisms Resource Budgets

5.9.4 Pyro Options

Pyro hold-down points used for this mission can be changed to non-pyro devices. Some solutions as based on Shape Memory Alloy, low melting temperature alloy, paraffin actuators or thermal knives are today qualified and provide good performances with significantly reduced shock.

The main drawback of these solutions is that they can not be fired with the same time accuracy as pyros.

[3] Part of the complete separation system mass (satellite rings) comes under the Structure mass budget.

5.10 Pyrotechnics

The IMM baseline is to use pyrotechnics:

- for the separation of the spacecraft from one another in the initial LEO orbit
- for toroidal antenna boom release;
- for low-gain antenna release;
- for the release of the magnetometer booms;
- for opening the branches of the propulsion system and closing the pressurant tank.

5.10.1 Requirements and Design Drivers

For all the applications listed above, cost and reliability considerations demand that qualified off-the-shelf devices are used.

5.10.1.1 Spacecraft Separation

For redundancy, two Pyrotechnic standard bolt-cutters are used in the clamp-band.

5.10.1.2 Antennae and Magnetometer Hold-down and Release

Six similar Pyrotechnic standard release-nut devices can be used to release all booms.

5.10.1.3 Propulsion System

Ten normally-closed pyrotechnic valves are needed to open the branches of the propulsion system, and one normally-open valve is needed to close off the pressurant.

5.10.2 Assumptions, Trade-Offs

Standard off-the-shelf devices reduce performance and procurement risk and allow for 5% mass-margin to be applied. The devices include redundant initiators with independent switching, command and supply, harness and electronics.

5.10.3 Budgets

The power demand per pyrotechnic device is of millisecond duration and thus negligible, particularly when fired before full spacecraft operation.

Unit masses of typical pyrotechnic actuators are in the region of 0.17 kg. Specific equipment and harness will be needed for the inter-spacecraft separation. The mass of the valves is included in the Propulsion subsystem budget.

5.11 Attitude and Orbit Control (AOCS)

Given the mission payload pointing requirements, a spin stabilised S/C design is most appropriate. The low pointing accuracy requirement (1°) makes a spinner attractive, as this option should at first reduce S/C complexity and cost.

5.11.1 Main Requirements

The functions required of the Attitude and Orbit Control Subsystem in IMM are:

- To spin the spacecraft in a direction perpendicular to the equatorial plane and maintain a 1°, 1σ relative pointing error
- To spin the spacecraft at 15 rpm
- To provide a 1°/40s pointing stability
- To determine the spacecraft attitude in inertial space with a pointing knowledge accuracy of 0.25°

5.11.2 Design Features

Figure 5-29 below illustrates the general architecture of the avionics subsystem.

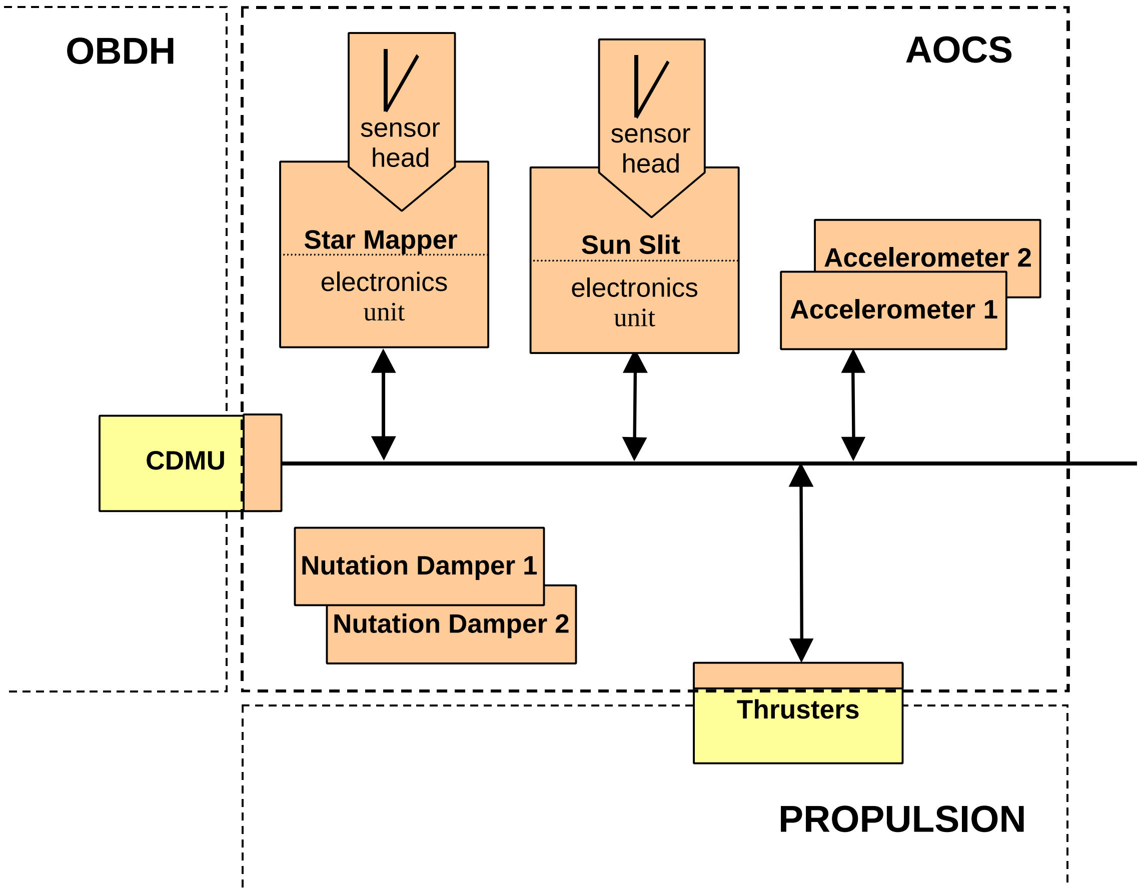


Figure 5-29: Avionics Subsystem General Architecture

5.11.3 Equipment Overview

The S/C AOCS design consists only of units which are well characterised and with relatively low mass and power requirements. Though not strictly required, accelerometers have been included for the sake of robustness, bearing in mind also that they require little extra mass and power. Similarly, two passive nutation dampers are envisaged even though one damper may be sufficient.

5.11.3.1 Star Mapper

The star mapper works in a ‘scanning’ mode on spinning spacecraft. It is placed on a side of the spacecraft perpendicular to the S/C axis of rotation. The star mapper requires that the solar aspect angle be greater than 60°. The sensors (redundant) consist essentially of a pair of V-shaped slits, a meridian slit (or a slit parallel to the S/C spin axis) and an oblique slit. The star mapper electronics unit provides S/C attitude information with respect to an inertial reference.

5.11.3.2 Sun Slit Sensor

The Sun Slit Sensor operates in a ‘scanning mode’, like the star mapper, and is placed on the top or bottom face of the spacecraft. The sensor consists basically of a pair of sensors with greycodes in a V-slit arrangement, i.e. with one slit parallel to the spacecraft spin axis and the other oblique. In the nominal operational mode of the spacecraft, the sun comes in and out of view for every spin of the spacecraft.

5.11.3.3 Accelerometers

The accelerometers, in the current configuration, i.e. without the need for active nutation damping, are not an essential part of the avionics design. Active nutation damping would involve not only the use of accelerometers, but most likely also a different thruster system design. However, given the low cost of the accelerometers in terms of mass and power requirements, it is recommended that accelerometers be included in the design as they can provide useful extra AOCS information.

5.11.3.4 Nutation Dampers

The nutation dampers are tuned to the specific characteristics of IMM, both in terms of size and location in the S/C. The size of the nutation dampers depends on the nutation frequencies of the S/C (i.e. the two dampers may be of different sizes). The dampers are inherently very reliable as they are of very simple design, and therefore redundancy has not been supplied.

5.11.4 Mass and Power Budgets

Unit	Quantity	Unit mass (kg)	Max unit power (W)
Star mapper	1	4.10	0.7
Sun slit sensor	1	0.81	0.1
Accelerometers	2	0.24	0.5
Passive nutation dampers	2	0.91	0
Total		7.21	1.8
Total with 10% margin		7.93	2.0

Table 5-19: Avionics Units' Mass and Maximum Power Requirements

5.12 Data Handling

5.12.1 Requirements and Design Drivers

The IMM command and data management system is characterised by the continuous acquisition of low rate data from payload instruments. It supports the flight software for the command and data management functions as well as for the attitude control and navigation functions.

There are six instruments housed in the spacecraft, which generate a total data rate of 8.7 kbps. The data flow is assumed to be continuous.

The ground coverage outage around perigee is about 33 mn. During this period, storage of payload and housekeeping data is required.

The highly elliptical, low inclination orbit and the long mission lifetime result in a harsh radiation environment of 467 krad over five years for a standard 4mm thickness of shielding.

5.12.2 Design Assumptions

The avionics of IMM is assumed to be built as a compact system that includes in the same box the Data Handling and Attitude/Orbit Determination and Control functions. The actual volume of data is low, and the necessary mass memory will be integrated within the same box of the CDMU. The heritage comes from present On-Board Computer Units, particularly from the PRIMA platform.

5.12.3 Baseline Design

The Data Handling System of IMM combines in a single box two main sections:

- The Command and Data Management Unit (CDMU)
- A Remote Unit (RU)

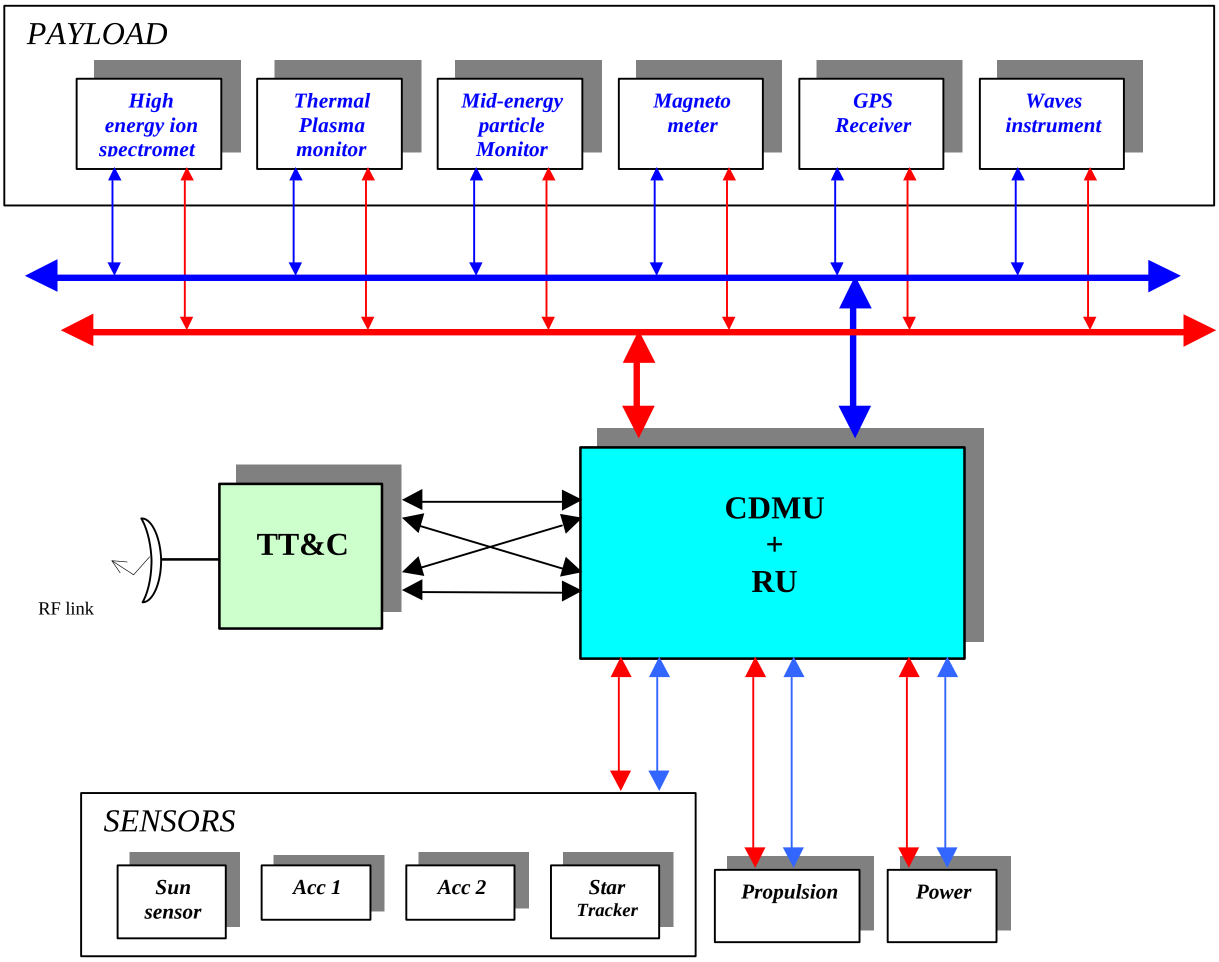


Figure 5-30: Data Handling Connection Schematic

The CDMU is a complete, redundant data management and control unit that provides the following functions:

- Telecommand support functions (decoding, validation, authentification, distribution)
- Telemetry support functions (data collection, encoding, and transmission to transponders)
- Timing functions (on-board time handling and maintenance, distribution of time, and synchronisation)
- On-board surveillance and reconfiguration functions

The CDMU will be composed of the following basic functional modules, as shown in Figure 5-31:

- Telecommand/telemetry module
- Processor module
- Reconfiguration module

The RU section is tailored to the mission, and implements interfaces with the dedicated AOCS sensors and actuators, the dedicated platform and payload housekeeping interfaces, the payload data interface, and the local mass memory. It is composed of the following basic functional modules (also shown in Figure 5-31):

- AOCS interface module
- Housekeeping module
- Local Mass Memory module

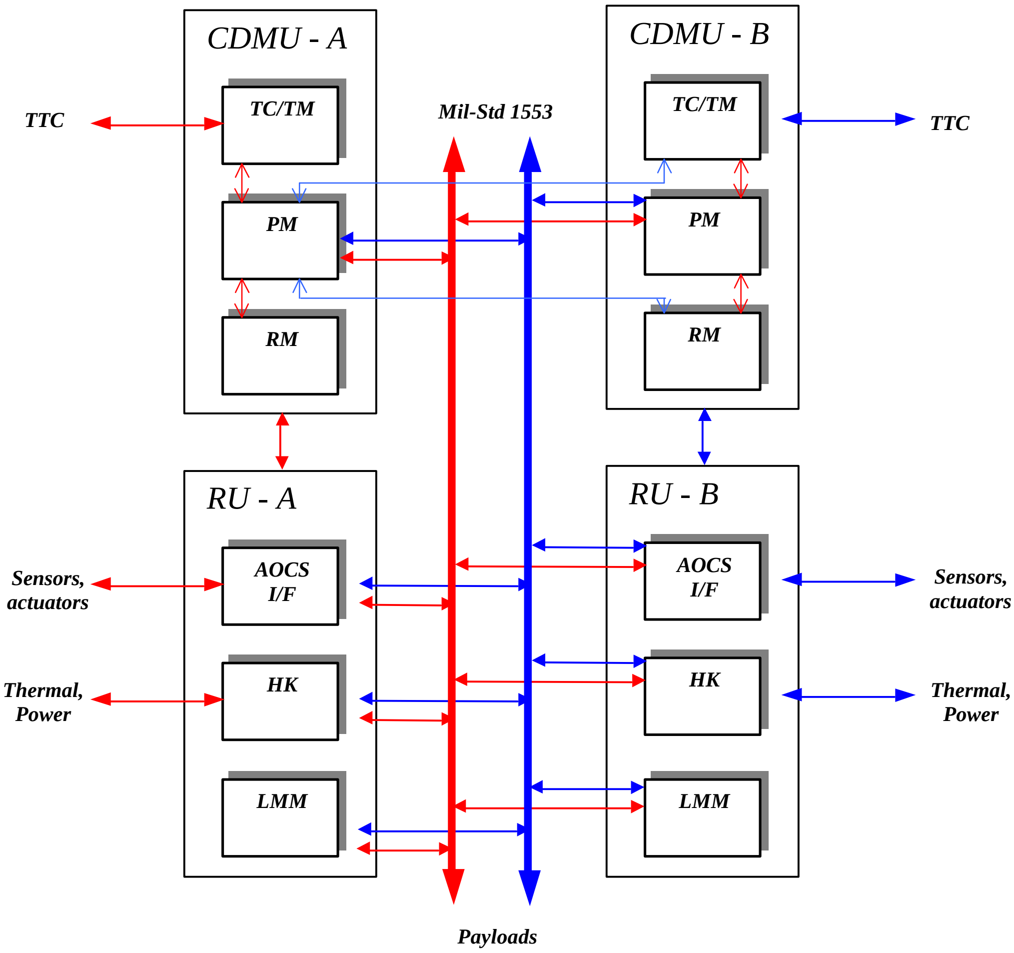


Figure 5-31: CDMU/RU Block Diagram

The CDMU is internally redundant and based on the TSC695E single chip SPARC microprocessor. There are two processor modules in cold redundancy. The frequency of the processor and the size of the application software memory will be adapted to the mission needs, to minimise unnecessary power consumption. The CDMU interfaces with the TTC subsystem via digital input/output lines for the Telemetry and Telecommand streams. In addition, it interfaces with the RU part via a redundant serial bus (Mil-Std 1553).

The RU part directly interfaces with the AOCS hardware via standard serial lines (RS422 and TTC.B.01). The housekeeping module implements a number of interfaces for the acquisition of data from the spacecraft such as analogue status, bi-level status, temperature status.

The instruments are connected with the DHS via the RU serial bus, over which commands are sent to instruments and telemetry data retrieved.

The mass memory is sized accordingly to the ground coverage and to the data generated on board. The coverage is about 95% of the orbit period, that is 12 hours, while the housekeeping and the instruments generate an average value of 11 kbit/sec. That leads to a storage of 22 Mbit of data during the longest coverage outage (about 33 mn per 24 hours). Even if the full data transmission is achievable in any orbit, it is planned to have an on-board capability of at least one day of data storage in case of contingency.

By using either 256 Mbit or 1 Gigabit DRAM memory modules, very few components will be required and the redundancy can be implemented in the same board, reducing the total unit mass and volume.

For DHS development the driving requirement is the very harsh radiation environment (467 krad with 4mm equivalent shielding over 5-year lifetime). The processor selected can withstand up to 500 krad with a 3V supply. The new radiation-tolerant technology made by TEMIC for prospective new ASICs shall withstand more than 200 krad. The radiation tolerance of new DRAM devices is not yet available and presumably is not more than 40 krad. Therefore a more in-depth selection on the components and additional shielding are required. The present design has been made assuming 6 mm Al shielding; however, trade-offs and related analysis should be carried out in the subsequent project phases.

5.12.3.1 Software

The DHS software tasks can be grouped in two main parts:

- The ‘basic software’, corresponding to a real-time operating kernel and to the low level functions such as on-board communication and configuration services, I/O and LMM drivers.
- Application S/W that covers the DHS S/W for the Mission handling, Spacecraft and Payload management functions, system level FDIR and the AOCS handling.

No particular features or criticality are foreseen for the S/W developments.

5.12.4 Budgets

The IMM DHS subsystem mass and power budgets consider that the unit is derived from a present or in-development unit. Then it has to be rebuilt and can benefit from a higher level of integration and a simpler design of the RU part.

MASS	Power	
	Nominal	Max
15 kg	32 W	47 W

Table 5-20: IMM DHS Mass and Power Budgets

5.13 Telecommunications

5.13.1 Communication Requirements

The key mission requirements for the communication subsystem of IMM satellites are as follows:

1. Continuous real time downlink of payload data (@8.7 kbps) plus housekeeping telemetry (typically 2 kbps).
2. Telecommand uplink (typically at 2 kbps) and ranging capability required but not in a continuous basis
3. In order to support contingency operations and the Ground Station coverage outage, the mass memory on board will be sized for storage of at least *one day of data* (1.12 Gbits). Therefore, a high data rate mode needs to be considered for the simultaneous downlink of stored and real time telemetry.

5.13.2 Design Drivers

The communication subsystem must ensure downlink continuity with adequate on-board antennae coverage and avoiding any antennae switching during the pass. The main design drivers in the definition of the communication architecture are the following ones:

5.13.2.1 Type of Orbit and Ranges Involved

The IMM mission is a constellation of four spinning satellites on quasi-equatorial highly elliptic orbits, with an apogee of 39717 km. Link budgets are calculated at apogee. Note that a margin improvement of up to 22 dB can be achieved for shorter distances (see Table 5-21).

Distance (km)	Attenuation (dB)	Delta Gain (dB)
3000	-180	22
5000	-184	18
10000	-191	12
20000	-197	6
30000	-200	2
40000	-203	0

Table 5-21: Variation of Propagation Losses vs. Range

5.13.2.2 S/C Attitude and Access Angles for Nominal and Safe Operations

During safe and contingency operations, the spacecraft should be able to communicate with Earth for any aspect angle. Therefore an omni-directional coverage for both transmit and receive is highly desirable.

During nominal operations the spacecraft is spinning and the spin axis is perpendicular to the equatorial plane. To achieve adequate antenna coverage, a toroidal type of antenna is needed. The worst case appears when the spacecraft is closer to Earth (3000 km height) and data may be transmitted up to +/- 40 deg referred to the equatorial plane.

Semi-hemispherical antennas, typically located on the top and bottom of the S/C body, may have phase cancellations (gain holes) in the equatorial plane when transmitting simultaneously. Continuous real-time downlink being a key mission requirement, the activation of antenna switches will complicate operations and will reduce safe mode capabilities.

Another option would be to use a phase array of antennae with a lobe rotating at the same rate as the S/C (Meteosat type). However, these antennas are complex, expensive and power consuming, and therefore have not been considered for IMM.

The chosen solution is thus switches.

5.13.2.3 Frequency of Operation

The frequency band candidates for supporting the communication functions are:

- S-band (2025 – 2110 MHz uplink, 2200 – 2290 MHz downlink)
- X-band (7190 - 7235 MHz uplink, 8450 – 8500 MHz downlink)

S-Band is shared by the Space Operations (SO), Space Research (SR) and Earth Exploration Services (EES). X-Band is allocated to the SR service. X-Band is proposed for IMM based on the following rationale:

1. The implementation of the toroidal antenna in S-Band is very big and heavy (four times bigger than in X-Band). The required size can be hardly allocated in the S/C structure
2. The use of X-Band is a must for the SWM and SAM missions. Additional savings and operational advantages can be achieved using the same band for this Space Weather mission also

5.13.3 Design Assumptions

The following sections describe the main assumptions considered in the architectural design of the subsystem and the link budget evaluation.

5.13.3.1 Ground Station

The ground stations will be located at Kourou and Perth (see 8.1.1). The ground stations shall have the capability to transmit in both RHCP or LHCP (selectable by switch) and to simultaneously receive and combine both polarisations. Two types of stations are considered, a 15 metre antenna with performance as per the ESA ESTRACK network and an 8 metre antenna, as the minimum size able to support the mission requirements.

	15 metre (ESA station)	8 metre ground station
Transmission (7190 – 7235 MHz)	300 Watts- RF TWT 82 dBW EIRP	80 Watts- RF TWT 71 dBW EIRP
Reception (8450 - 8500 MHz)		
G/T @ 10° elevation (Kourou Weather conditions)	38.0 dB/K clear sky 35.6 dB/K > 99% average year 34.2 dB/K > 99.9% av. year	30.1 dB/K > 99% av. year
Eb/No for PFL = 1e-5[4]	12.5 dB (No coding) 2.8 dB (Concatenated coding) 1 dB (Turbo Coding)	

Table 5-22: Ground Station Performances

5.13.3.2 Transponder

The transponder required to support IMM shall be X/X Near Earth type, with Tx/Rx coherency and ranging capability. As baseline, concatenated coding has been considered. An improvement of 1-2 dB could still be achieved with Turbo Encoders (presently under development).

The transponder requirements are as follows:

Transmit Power	33 dBW (2W) at diplexer common port
Tx Modulation	NRZ/BPSK/PM for 2 kbps and 13 kbps data rates SPL/PM or BPSK for 170 kbps Data rates selectable by TC
Encoder	Concatenated coding (RS included in transponder, Viterbi implemented in the Data Handling Unit)
Rx Threshold	-128 dBm (carrier acquisition) -118 dBm (Telecommand operation)
Noise Figure	2.5 dB (receiver input)
Rx modulation	NRZ/BPSK/PM @ 2 kbps
Mass	3 kg. Goal: 1.5 kg
Dimensions	275 x 110 x 197 mm^3
Power Bus	From 21 to 50 V
Consumption Rx	3 Watts
Consumption Tx	12 Watts

Table 5-23: IMM Transponder Characteristics

[4] Probability of Frame Loss for Interleaving Depth I=5 and Frame length 1275 Octets

5.13.3.3 Low Gain X-Band Antennae

Each low gain antenna has a nearly hemispherical coverage, with an absolute gain >-5 dBi, and opposite polarization (RHCP and LHCP). The estimated mass of one low gain antenna is 300 gram. This type of antenna will be needed for most future X-Band missions and is already under development.

Optimisation of the antennae location taking into account the FOV blocking effect of appendages like the solar arrays and other structural elements was not analysed in this phase of the study.

5.13.3.4 Toroidal X-Band Antennae

Toroidal antennae are in development for C to Ka Band. Toroidal coverage is achieved combining two bicone antennas feeded in circular waveguide. Dimensions are 17cm x 40 cm (C-Band, X-Band 400gr). The antenna gain considered over the complete coverage is +2dBi.

The toroidal antenna needs to be mounted on a boom parallel to the spin axis. The boom length required to allow +/-40deg FOV is 1.6 metres, too large for the IMM configuration. The boom length can be reduced to around one metre, and in this case an asymmetrical coverage would be required for the antenna (e.g. +15deg to -40deg ref. to equatorial plane). This restriction is acceptable provided the ground station(s) are in the Southern Hemisphere (case of Perth) or around the equator (case of Kourou).

5.13.4 Downlink Data Rate Evaluation for Data Dump

The ground coverage presented for IMM mission ensures visibility when the spacecraft altitude is higher than 3000 km. There is approximately 30 minutes of coverage outage (23.4 Mbits) that, if required, may be downloaded to ground as soon as visibility conditions permit. In fact the coverage outage is 33 minutes, but 23.4 Mbits correspond to 30 minutes, and the following table is computed on the basis of 30 minutes $\rightarrow$ 23.4 Mbits. 33minutes $\rightarrow$ 25.7 Mbits.

The data rates are as follows:

Housekeeping typical rate	2.00 kbps
Payload real-time data collection	11.00 kbps
Total real-time data rate	13.00 kbps

Table 5-24 presents an analysis of data rate requirements for different conditions of on-board storage.

Recording time (h)	Data (Mbit)	Data rate (kbps)			
		39	**65**	**169**	**325**
0.5	**23.4**	15.0	7.5	2.5	1.3
1.0	**46.8**	30.0	15.0	5.0	2.5
12.0	**561.6**	360.0	180.0	60.0	30.0
24.0	**1123.2**	720.0	360.0	120.0	60.0

Recording time (h)	**Data (Mbit)**	**Data rate (kbps)**			
		39	**65**	**169**	**325**
48.0	**2246.4**	1440.0	720.0	240.0	120.0

Table 5-24: Download Time against Data Dump Rate

Data collected over 30 minutes could be downloaded in 15 minutes at 40 kbps or just 2.5 minutes at 170 kbps. The data collected in one orbit (12 hours) would require 1 hour at 170 kbps.

5.13.5 Link Budget Evaluation

Range: 39700 km @ 10° elevation (Slant range 44540 km)

TC (2 kbps)	**8m station** **71 dBW EIRP**
LEOP and Safe Mode (via LGAs)	4 dB margin
Nominal Operation (via toroidal antenna)	11 dB margin
Notes: • Margins evaluated for 3 dB extra propagation losses due to rain. • TC margins for 15m antenna are increased by 11 dB due to higher EIRP	

Table 5-25: Uplink Budget

@ 39700 km, 10° elevation (=44540 km)	**8m station**
HK@ 2 kbps via LGA	4 dB margin – clear sky -
RT data @ 13 kbps via toroidal (nominal operations)	8 dB margin – clear sky - 4 dB margin – 99% time average year at Kourou - 2 dB margin – 99.9% time av. Year at Kourou -
Data dump @ 170 kbps	Supported for spacecraft altitude between 3000 and 15000 km, or 15m antenna required and S/C height below 30000 km (see 5.13.2)
Notes: • TM margins for 15m antenna increase by 5 dB due to higher G/T • G/T increase 2-3 dB at zenith • 2 Watts RF power on board, that could be increased to 5 Watts to support high data rate downlinks with the 8 metre antenna.	

Table 5-26: Downlink Budget

5.13.6 Communications Baseline Design

The communication subsystem consists of the following elements:

- Two Low Gain Antennae
- One Toroidal Antenna
- One RF Distribution Unit
- Two transponders. The transponder integrates the transmitter (plus modulator), the receiver (plus demodulator) and the diplexer that combines both units in a single port towards the antennae.

The architectural design proposed for the communication subsystem is depicted in Figure 5-32.

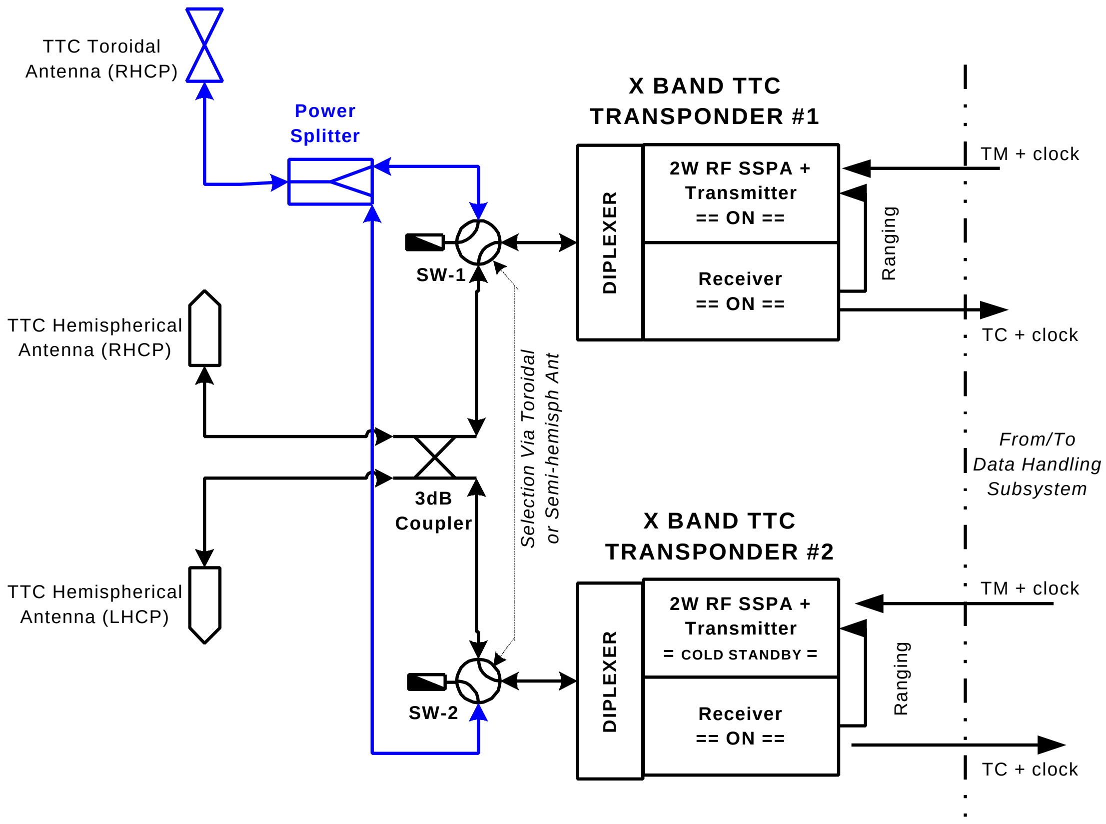


Figure 5-32: IMM Communication Subsystem

During LEOP and in Safe Mode, the spacecraft is transmitting and receiving simultaneously via both Low Gain Antennae (LGA). Signals received by LGA-1 (RHCP) and LGA-2 (LHCP) are combined in a 3dB-hybrid and routed to both receivers. The receivers operate in hot redundancy; the demodulated data is sent to the OBDH subsystem where one chain will be selected for further process. The transmitters operate in cold redundancy.

During nominal operations, the switches SW-1 and SW-2 route both transponders towards the toroidal antennae. Note that with action of only one of these switches, it would still be possible to receive with the 2nd transponder the uplink received via the LGAs.

Figure 5-33 shows the antenna locations considered and the coverage provided

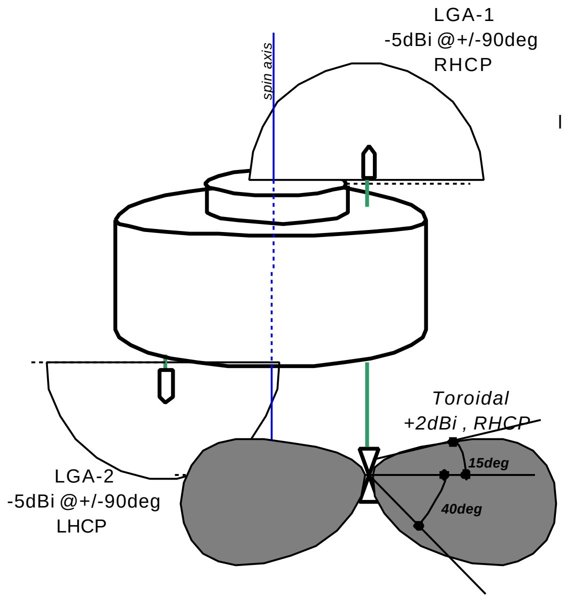


Figure 5-33: IMM Antenna Layout and Coverage

	Coverage	TC uplink	TM downlink	Ground Station
LEOP and Safe Mode	Omni via 2 LGAs (simultaneous Tx/Rx from both)	2 kbps	2 kbps	8 metre 71 dBW EIRP 30 dB/K G/T
Nominal	Via toroidal antenna	2 kbps	13 kbps	
High Data Rate	Via toroidal antenna	2 kbps	170 kbps	8m antenna @ H<15000km or 15m antenna @ H<30000km

Table 5-27: Modes of Operation

5.13.7 Budgets

Items	Number of units	Nominal Mass per unit (kg)	Total Nominal Mass (kg)
X-Band Transponder including receiver, 2W SSPA transmitter and diplexer:	2	3.50	7.00
RF Distribution unit -including power combiners, switches and harness)	1	2.50	2.50
X-Band semihemispherical LGA	2	0.50	1.00
X-Band Toroidal antenna	1	0.50	0.50
Total Mass (kg)			10.50

Table 5-28: Telecomms Mass Budget

Item	Number of units	DC power per unit (Watts)	
X-Band Transponder including:			
- Receiver	2	3.00	hot redundancy (6 Watts)
- 2W SSPA + Transmitter	2	12.00	cold redundancy
- Diplexer	2	0.00	Passive device
RF Distribution unit + Harness	2	0.00	Min. consumption (switch control)
Antennae	3	0.00	Passive device
Power Consumption		18.00	2 Receivers + 1 Transmitter On

Table 5-29: Telecomms Power Budget

5.14 Structures

5.14.1 Requirements and Design Drivers

The Inner Magnetosphere Monitor structural design is driven by the volumetric budget requirements for the fairing of the reference launcher (GSLV) available for the four stacked satellites, and by the dimensional stability requirements of the individual satellites and the stacked configuration.

The instruments do not pose a stability constraint on the structure.

For all the equipment supporting the instruments and the S/C operations the structure needs to provide:

- A platform for electronic equipment, propulsion, power & harness;
- Easy access for AIV activities.

A sketch of the structural design for the IMM mission is shown in Figure 5-34.

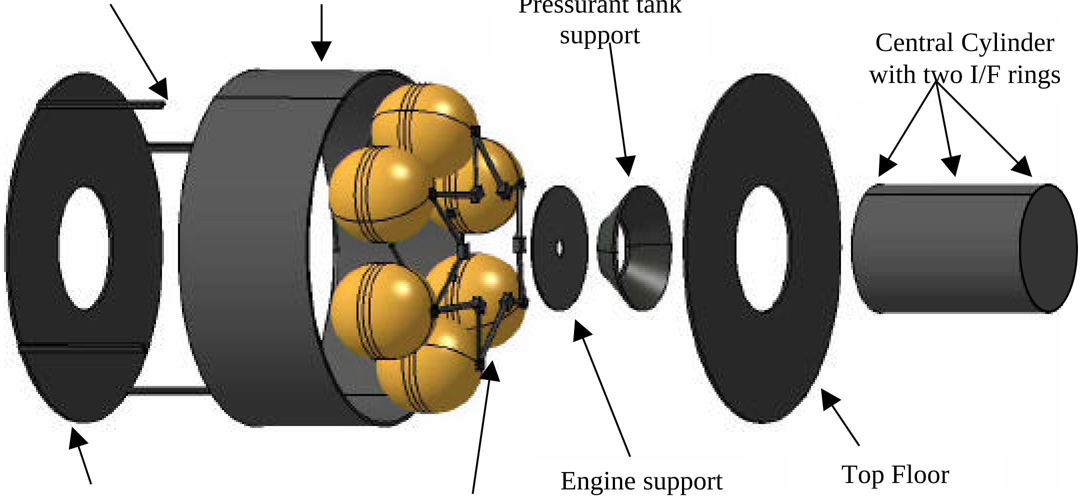


Figure 5-34: Primary S/C Structure

The load-bearing structure is based on a central cylinder and a propellant tank support structure, while the top and bottom floors are used to accommodate other equipment.

The spacecraft structure is not a recurrent item. It is a completely new structural design, but the architecture and the related technology are recurrent from many proven spacecraft bus designs.

5.14.2 Assumptions and Trade-Offs

The available volume for the GSLV launcher and the requirement to launch four equal spacecraft simultaneously resulted in a stack of four spacecraft. The shape of the S/C, cylindrical, uses the launcher fairing optimally and provides a simple shape for the required spin-stabilised platform.

The S/C interfaces with the GSLV launcher through a 937 standard interface adapter. Each S/C cylinder in the stack interfaces through the same adapter to the next.

5.14.3 Baseline Design

The structural baseline will be sandwich panels with an Aluminium core and CFRP face sheets. The face sheet selection was made for mass-saving reasons. Aluminium face sheets would be cheaper, but result in a higher total mass for the structure.

5.14.3.1 Budgets

Table 5-30 shows the mass breakdown of the primary structure.

Item	No.	Mass [kg]	With margin [kg]
I/F Ring	2	3.6	3.78
Top Floor	1	12.92	15.50
Bottom Floor	1	12.92	15.50
Central Cylinder	1	26.15	31.38
Outer Wall	1	12.07	14.48
Struts for outer wall support	4	1.60	1.92
Tank Brackets	7	1.40	1.68
Inserts and Miscellaneous	1	5.00	6.00
		TOTAL:	**109.9**

Table 5-30: Primary Structure Mass Budget

5.14.3.2 Frequency Requirements

For the selected configuration, four satellites stacked in one fairing volume, it will be important to assess the lateral stiffness of the primary structure. The central cylinders will act as the core of the stack to provide lateral stiffness. With the currently proposed design a first lateral frequency of about 10 Hz is feasible, yet needs to be analysed in detail in the following phase. Also due to incomplete information at present about the selected launcher, a complete trade-off needs to be made in the next phase.

5.15 Programmatics

5.15.1 IMM Master Schedule

The project Gantt chart below in Figure 5-35 indicates the major mission phases of the IMM system, consistent with the following key milestones:

- Start of the project Phase A in July 2002
- Launch in July 2007
- A 4-month ± 2 months Transfer Phase to L-1
- A nominal Operational Phase of 4 years until mid 2011

The second mission operation phase is foreseen from July 2011 for 4 additional years. This extended phase is not considered as part of the baseline mission and has not been costed for.

ID	Task Name	Duration	Start
2	ATP	0 d	01.07.02
3	Phase A	138 d	01.07.02
4	Phase B	230 d	09.01.03
5	Phase C/D	847 d	27.11.03
6	Launch Campaign	92 d	26.02.07
7	Launch	0 d	03.07.07
8	LEO+Transfer	120 d	04.07.07
9	Operational	1044 d	19.12.07
10	Extended Oper.	1044 d	20.12.11
11			
12	**System Milestones**	**1076 d**	**08.01.03**
13	ATP	0 d	08.01.03
14	PDR	0 d	26.11.03
15	SDR	0 d	18.01.05
16	CDR	0 d	08.07.05
17	QR/AR1	0 d	12.06.06
18	FAR	0 d	23.02.07

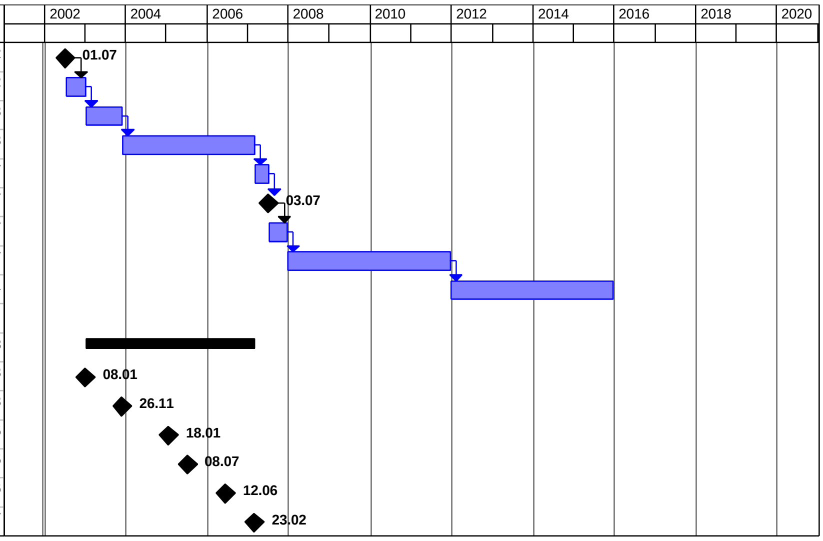


Figure 5-35: Project Master Plan

5.15.2 Development and AIV

Before going into the details of the IMM specific development, a brief introduction is necessary. The Space Weather system is a constellation of different satellites, i.e. the IMM (with four flight units operating together on equispaced orbits), the SWM, and the SAM. Each type of spacecraft requires a dedicated design, development and testing. For all types, the launch date is currently required in 2006.

Note that this date cannot be met, because there is too little time to perform phase A, B development and test is C/D of four satellites, unless they go through integration and test in full parallel. However, this would increase the cost in an unacceptable way.Even with this date, from a purely development perspective, three industrial development teams have to work in parallel on IMM, SWM and SAM, to meet the required launch date. Other possibilities considering one team only for the three types of satellites have not been evaluated, but in this case one should consider a lag of about one and a half to two years between launches, which is not acceptable. Therefore the planning will consider the three types of satellite separately.

The IMM spacecraft includes the following main building blocks:

- A cylindrical body carrying the Solar Arrays on the external surface.
- An upper platform carrying both the experiments and the spacecraft subsystem electronics (Avionics, Telecom, AOCS, Power S/S) and supporting the low gain / toroidal antenna boom.
- A lower platform carrying the Propulsion system, Propellant and Pressuring tanks, and the 490 N main engine. A low gain antenna is supported by this platform on a short boom.
- A payload of scientific instruments including a set of deployable wires and booms.

The development of the spacecraft strongly relies on existing designs and available technology. Precautions need to be taken in order to ensure the high level of magnetic cleanliness required by the mission.

The radiation environment is considered critical, due to the altitude and high eccentricity of the selected orbits. The situation is particularly critical for the experiments, due to their requirements on the field of view and their consequent location on the exterior of the S/C, where effective shielding cannot be achieved.

The project development and, more specifically, the cost estimates have assumed a streamlined industrial team whereby the Prime Contractor is responsible for the:

- Overall mission analysis
- Overall design development and procurement of the spacecraft
- Detailed spacecraft design at system and subsystem level
- Direct procurement of the spacecraft units, equipment and major assemblies (hardware and software)
- Overall spacecraft Assembly, Integration and Verification (AIV) activities
- Definition and control of the technical and operational interfaces of the Instruments

5.15.2.1 Model Philosophy

Considering the moderate development risk identified in most aspects of the spacecraft design, a Protoflight approach has been selected at spacecraft level, based on a 4-model philosophy:

1. Structural and Thermal Model (STM)
   Will ensure the mechanical and thermal qualification of the spacecraft design. Most of the unit assemblies will be represented by thermal and structural dummies.
2. Avionics Test Bench model (ATB)
   Will ensure verification of the overall electrical, functional and software interfaces. Breadboard units (BBs) will be used most of the time; exceptionally, Interface Simulators could be used for the Payload Units. Elegant BB units (EM like with commercial components) or modified EM could be used if cost-effective, e.g. in the case of recurring units with EM available, or off-the-shelf equipment.

3. Protoflight Model (PFM)
   Built to full flight standard, this will be subject to qualification test levels with acceptance duration.

4. Flight Model (FM)
   Built to full flight standard, this will be subjected to acceptance test levels for acceptance duration.

As a programmatic approach, the use of Mil-grade EEE parts has been assumed. However, the reliability level of EEE parts must be carefully assessed, due to the impact this selection necessarily has on the risk and cost of the project.

For the costing, procurement of European Hardware has been assumed in general whenever a design and the technology are available. Provision of spares kits is foreseen for all units. These could be specifically procured or available as heritage of recurring units from past projects.

5.15.2.2 AIV Approach

Taking into account the given model philosophy and the expected development time of the Instruments, an overall AIV planning is outlined in Figure 5-36.

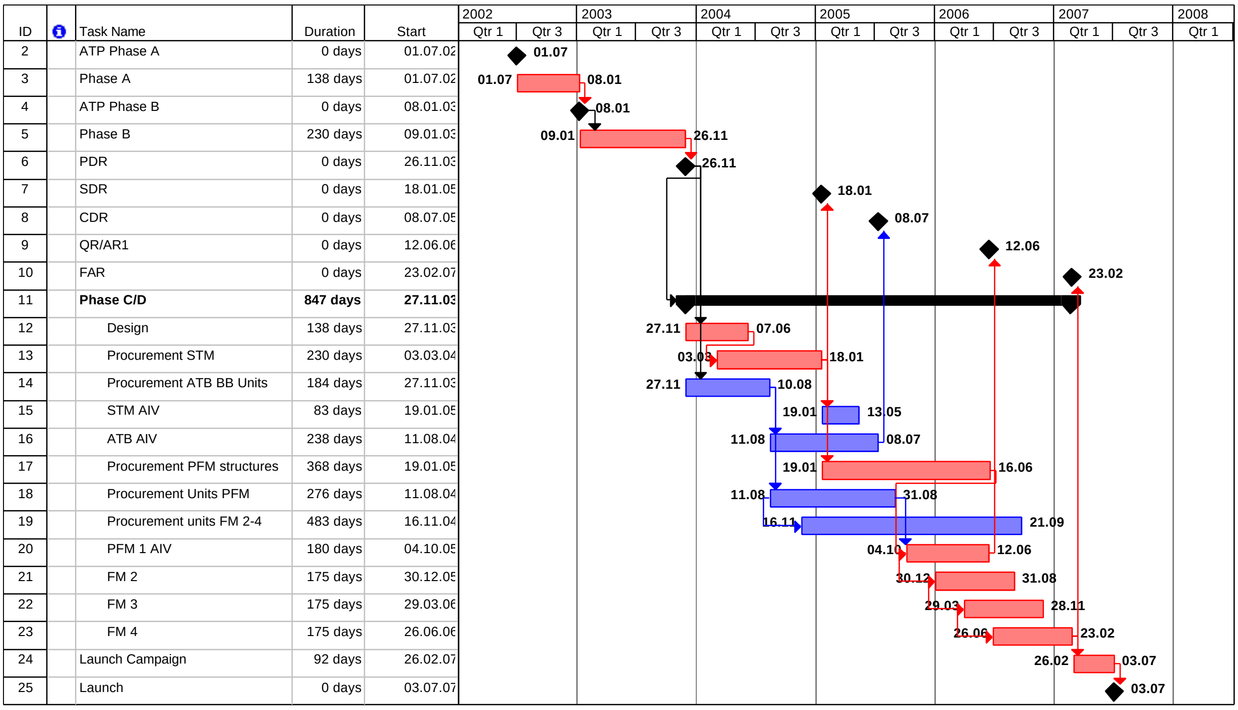


Figure 5-36: AIV Planning Chart

The validity of the planning is based on the following key assumptions:

- The mechanical and thermal designs of the instruments are qualified separately. This includes verification of inner optical alignment stability. Final qualification will be achieved on the spacecraft PFM.

- Instrument STMs will be delivered at a build standard compatible with the spacecraft STM programme. Where required, mechanical alignment of instrument boxes with the spacecraft structure will be tested at spacecraft level.
- Instrument BBs will be delivered at a build standard which, at a minimum, has to be representative of the electrical and functional interfaces. No calibration activities are envisaged on the ATB.
- The Prime Contractor will install the instrument PFM and FM on the satellites. Functional and interface tests will be performed to certify their proper on-board accommodation. Calibration activities will be performed on all flight satellites.

5.15.2.3 Critical AIV Aspects

The integration of a constellation of four satellites imposes some critical assumptions.

The first is that to perform a timely delivery of the four flight units, their test activities have to be performed partially in parallel, as shown in the schedule.

It is possible to minimise the effect of parallel work by applying the following criteria:

- The integration of the flight models is performed in sequence. Only one integration team and one set of integration tools are therefore necessary.
- The environmental tests are performed in the same facilities, and in sequence on the four satellites. This way, one set only of test adapters is needed.
- The schedule effect brings the end of the test on “PFM 1” at the beginning of the test on “FM 4”. This means that there will be a peak of two satellites being environmentally tested together. There is an additional criticality, in that any delay will create the need either for the third satellite to go into parallel testing, or for a delay on the start of testing on the third satellite itself.
- A consequence of the above is that two sets of EGSE could be sufficient instead of 4, depending how the schedule is arranged. A margin should be applied to allow this reduction to be evaluated in terms of pros and cons.
- Duplication of MGSE is envisaged to handle the four flight satellites. Four containers and four integration dollies are needed. Some savings are expected on the other MGSE items, where the discontinuous utilisation allows for shared use.
- The same principles of satellite integration and testing are applied to structure and propulsion AIV. Assembly and test of the structure models could be done in sequence, to avoid duplication of tools and teams. The same applies to propulsion models.

Another critical aspect is the timely release of the SW versions. The on-board SW first version V 1.0 must be ready for the start of the ATB activities.

The final version, implementing the results of system testing on the ATB, must be loaded on the PFM units before they are delivered to the system for integration. This final release V 2.0 shall be also tested on the ATB. This late test will give the final confirmation of adequacy of SW implementation of the system functions.

5.15.3 Programmatic Risk Assessment

The risk elements, from a programmatic point of view can be summarised as follows:

- The proposed short development time
- The rad-hard component technology currently under development (see next section).

5.15.4 Critical Technology

No critical technological aspects have been identified in the IMM platform design, except for the strong radiation environment.

Rad-hard components will be needed all across the spacecraft. It is assumed that in this respect the project will strongly benefit from the on-going technology developments initiated in the frame of the Galileo program. It deals in particular with the availability of a rad-hard version of the Data Handling SPARC processor and its related components. For the mass memory, heavy shielding is envisaged, since the availability of high-density memory chips in rad-hard version is not likely.

5.15.5 Links to Other Projects

Other ESA projects (Cluster I and II, Mars Express) have been used as a reference for costing purposes. The proposed spacecraft concept is a dedicated design for the IMM mission, and does not rely on parallel developments.

5.16 Risk Assessment

The Space Weather Service (SWS) is, from a risk point of view, to be handled differently from usual scientific missions, because the strongest requirement driving this analysis is the requirement for a continuous service of near real-time data. By this definition, success of the service depends on a successful delivery of near real-time data to the user Secondary benefits are not considered.

In the scope of this study the risk assessment is limited to the risk of loss of service availability of the IMM space segment built up of several single spacecraft.

5.16.1 Requirements and Design Drivers

For the entire SWS no requirement for the service availability is yet defined and mapped into space segment dependability requirements. Therefore the risk assessment will highlight the dependency of the service availability from the availability of the space segment for various combinations of operational spacecraft and instruments.

5.16.2 Assumptions and Trade-Offs

The baseline configuration is defined by 4 satellites each carrying an identical set of 6 instruments.

Various configurations of instruments and spacecraft required to be operational to provide the service of the space segment are analysed, as shown in the following figure.

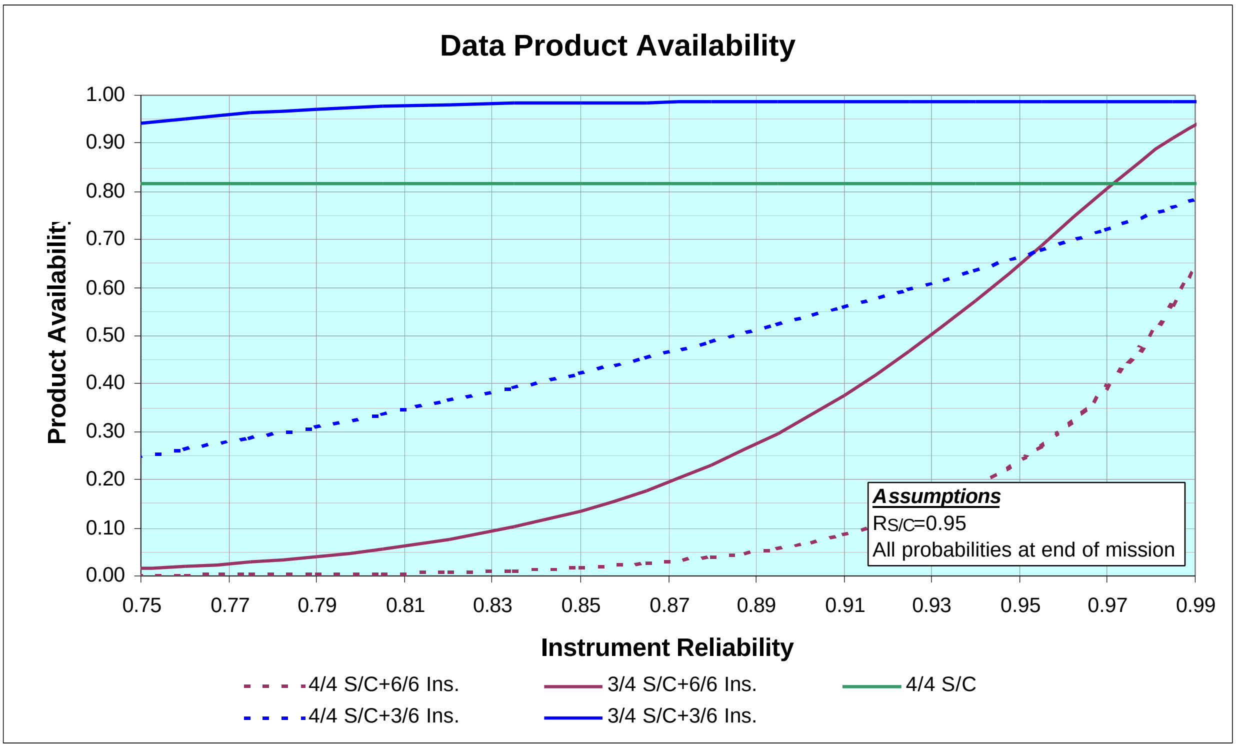


Figure 5-37: Data Product Availability

The reliability of each spacecraft is assumed to be 0.95 at the end of its intended lifetime.

Based on this assumption, the following can be found:

- In case data (full or partially) has to be obtained from all four satellites, the probability to maintain the service is defined by the spacecraft reliability and is in the best case 0.81
- If data shall be obtained from all 24 instruments, the service probability is only 0.63 (3/4 S/C + 6/6 Ins.)
- If 3 out 6 instruments per satellite need to be operational on all satellites to provide the service, the probability is in the best case 0.79 even if the instruments reliability is 0.99 (3/4 S/C +3/6 Inst.)

As it is indicated by the customer to be able to tolerate some instrument failures, it is obvious that the major driver for the availability of the space segment service is the reliability of the spacecraft. As 0.95 is already a high figure assuming to aim for life times of at least 3 years, failure tolerance in the spacecraft constellation has to be considered. The next figure shows the dependency of spacecraft reliability versus reliability of the satellite constellation.

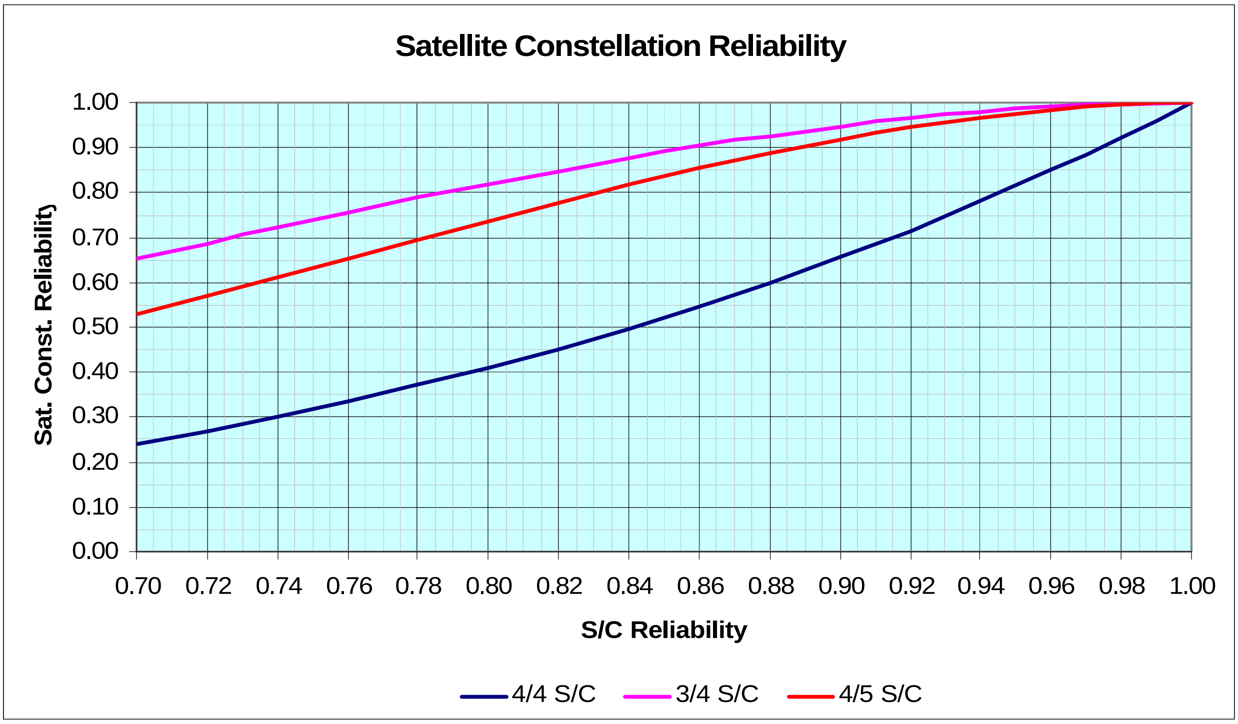


Figure 5-38: Satellite Constellation Reliability

It can be easily seen that a 3/4 or 4/5 configuration (i.e. 4 out of 5 satellites are operational) allows operation of the spacecraft in orbit down to a reliability of 0.9 (3/4) or 0.92 (4/5) respectively, to maintain a satellite constellation reliability of 0.95. In view of the harsh environmental conditions this can be mapped in an extended mission time, thus increasing flexibility with respect to a replacement strategy and its impact on cost.

5.16.3 Baseline Design

A driver for the baseline selection is the need to have the spacecraft equally spaced in orbit.

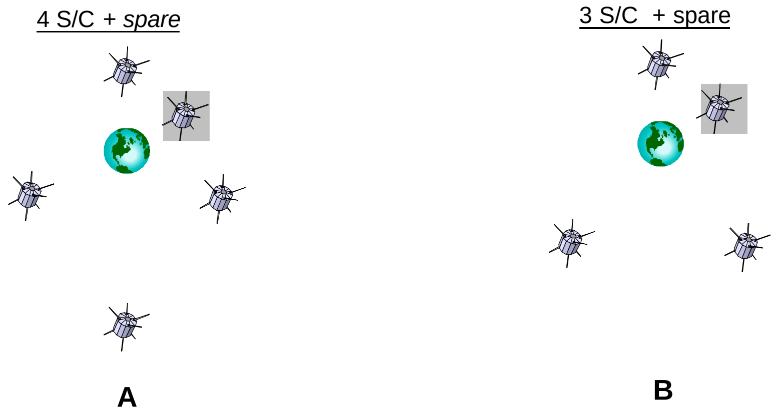


Figure 5-39: Constellations

The present baseline is to launch 4 satellites. Thus there are some options for initial constellations and service recovery strategies. Additionally the option of 4 satellites + 1 spare is indicated (spares are marked in grey).

1. To place the satellite spaced to provide continuous monitoring capability without any fault tolerance capabilities (A)
2. To place 4 satellites spaced to provide continuous monitoring capability with the fifth satellite as a spare, to be positioned when the first satellite fails (A + *spare)*
3. To place 3 satellites spaced to provide continuous monitoring capability with the fourth satellite as a spare, to be positioned when the first satellite fails (B)

The only options capable of providing a space segment with a sufficient reliability are 2 and 3. Assuming a constellation of 3 satellites is capable to collect the appropriate data for the SW service –to stay with the presently baselined 4 satellites- the data from the space segment can be provided with a probability of 0.95 as long as the instrument reliability is not less than 0.85 and the satellite reliability is not less than 0.90.

In case 4 satellites are needed, a fifth satellite would provide sufficient in orbit reliability. In this case the data from the space segment can be provided with a probability of 0.95 as long as the instrument reliability is not less than 0.88 and the satellite reliability is not less than 0.92.

It should be mentioned that in the scope of CLUSTER a reliability model was developed taking into account dependencies between the lifetime distributions of individual satellites. Depending on the level of dependency, the reliability of a 4/4 configuration can significantly increase, thus not justifying a spare satellite anymore. However, evidence on the suitability of this dependency model based on in-orbit observations is not yet available. Therefore the ‘classical’ assumption of independent satellite lifetime distributions is used in the scope of this study.

5.16.3.1 Feasibility

A first assessment of the feasibility of the present design baseline of the CDF–type IMM satellite with respect to the required reliability is shown in the following table. It gives a first hint on the time to replacement for the options considered before.

Reliability	Elapsed Mission Time
0.95	≈22000 h (≈2.5 years)
0.92	≈28000 h (≈3.2 years)
0.9	≈34000 h (≈3.9 years)

These figures are an optimistic estimate and can only take into account a ‘typical’ design for the various units of the subsystems. A decrease in reliability can be expected when the prediction is based on the more detailed design of the IMM type satellite in future.

The reliability block diagram of the present baseline is shown below (shadowed blocks indicating internal redundancy, e.g. on cell level for the battery). Instruments are indicated in the diagram but not considered in the estimations given before.

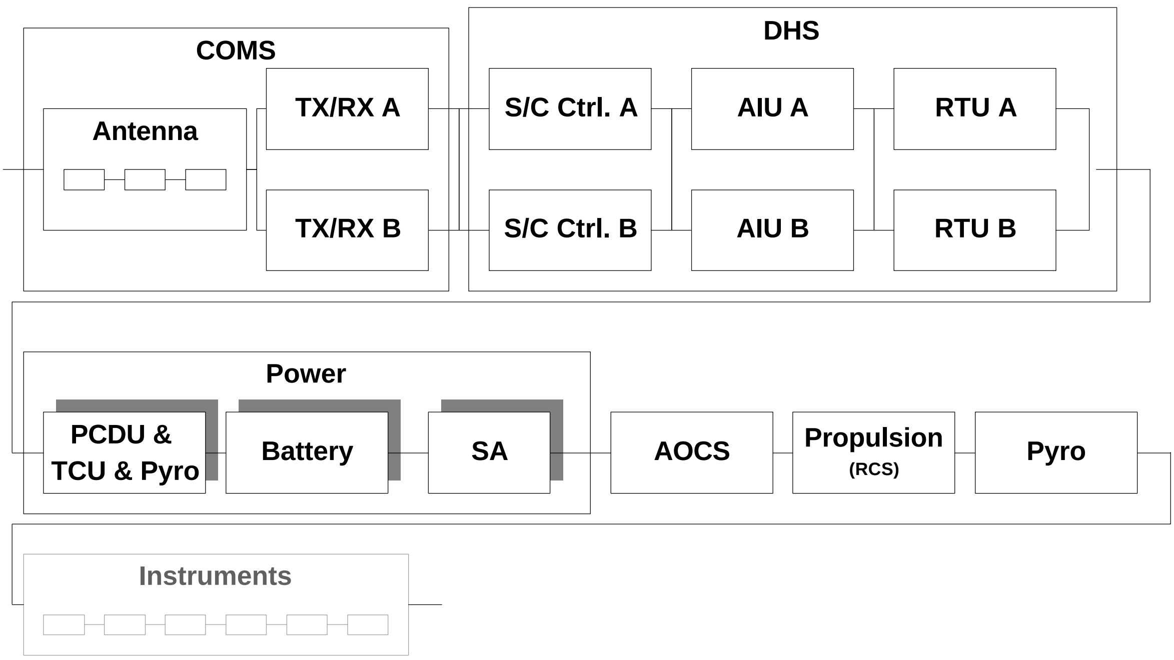


Figure 5-40: Reliability Block Diagram

An alternative design that was studied was based on the STRV C&D platform. This platform is designed for an orbital lifetime of 4 years. A constraint on this alternative is the very long time needed to place the 4 IMM satellites in their final positions. A remaining nominal lifetime of only two years is the consequence. While the instruments can be maintained at a relatively high reliability level when starting the measurements, assuming they are switched off during the transfer phase, the spacecraft subsystems will already be significantly degraded.

Early in-orbit failures of STRV C&D were caused by a specific design flaw and this can be considered to be solved for future STRV applications.

Although, from a dependability point of view, the option is feasible, no benefits of an STRV-based IMM segment can be identified compared with the baseline.

5.16.4 Conclusions

Various options are possible. Before further evaluation the customer has to decide on the best constellation needed to provide the SW space segment service. This addresses the number and position of satellites, the type and configuration of a minimum set of instruments needed on board of each satellite and the targeted service availability.

In any case a non-redundant space segment constellation seems to be not acceptable due to the low reliability of the constellation that would cause a very early and frequent replacement of satellites in orbit.

5.17 Cost Estimate

5.17.1 Main Costing Assumption

It is assumed that the industrial organisation for the IMM platform project is led by a Prime Contractor, handling the detailed design at platform level. The Prime is also assumed to be responsible for Assembly, Integration and Test (AIT) activity support at Spacecraft System level.

Most IMM platform and payload components are based on off-the-shelf equipment or on existing and available technology, modified (where applicable) for the environmental conditions (high level of radiation). All cost estimates are based on references and cost estimating methods in line with the above general hypothesis. It is considered that the spacecraft design activities and equipment selection and validation will be commensurate with the operational nature of the mission.

No geographical distribution constraints are included.

5.17.2 Cost Estimate Methodology

The following methods have been used, in descending order of preference:

- Reference to similar ESA missions;
- Reference to similar equipment/system level costs, taking into account the amount of new development required;
- Expert judgement from technical specialists in combination with similar equipment references, in the case that the amount of new development is extensive;
- Expert judgement from technical specialists only, if references are not available;
- Equipment cost models;
- The ESA internal, system level cost model RACE;
- System level cost relationships (for the Prime and Payload/Payload Contractor activities), based on recently observed relationships for relevant references.

5.17.3 Scope of the Cost Estimate

In accordance with the study requirements, the cost estimate covers:

- the IMM Platform
- Instruments (as far as information is available)
- Phase B and C/D costs of the mission
- Launch

Excluded are:

- Operational costs (ground segment)

Furthermore, the cost estimates are for the industrial costs only.

The IMM Platform Phase B and C/D cost estimate includes:

- A provision for the Phase B development costs
- The phase C/D costs, which are split into two parts: up to PFM and remaining 3 FMs
- Phase C/D equipment, software and platform level costs including Ground Support Equipment costs
- Spacecraft system level activity cost (Management & Control, Engineering, PA, AIT)
- Launch vehicle dispenser interface costs
- Platform Design Maturity provision

For the Payload, the Phase B and C/D cost estimate includes:

- A provision for the Phase B equipment development costs
- The phase C/D costs, which are split in two parts: up to PFM and remaining 3 FMs

The industrial cost is considered to be as the Prime Contractor offering a firm fixed price would see it. It covers the supply of the flight unit(s) with the associated development models when applicable, the spares, the specific GSE and the user manuals. It also covers the Project Office cost of the equipment suppliers.

5.17.4 Phase B Cost Assumptions

The Phase B costs have been estimated based on the Phase B versus Phase C/D cost ratios for projects with a strong prime contractor involvement at subsystem level, similar to what was assumed for STORMS. The Phase B costs do not cover the pre-developments assumed to be part of Phase C/D.

5.17.5 Phase C/D Cost Assumptions

For the cost estimates the platform development and Assembly, Integration and Test (AIT) is regarded as being a complete project on its own handled by the Prime Contractor at satellite level. All platform subsystem Project Office (PO), AIT and Ground Support Equipment (GSE) costs are therefore included at platform level.

5.17.5.1 AOCS

- The AOCS design is derived from Cluster.
- Prices have been estimated based on this reference but are adjusted with today’s market price trends.
- All equipment is assumed to be off-the-shelf, possibly with simple modifications.

5.17.5.2 Propulsion

The cost estimate for the propulsion system is mainly based on Cluster. Necessary adaptations have been taken into account. Further Project Office costs on sub-system level are presented, based on ratios observed on previous projects.

5.17.5.3 Electrical Power

- Solar Array costs are based on ESA internal CERs. Although the GaAs solar cells will be off-the-shelf equipment, the panel configuration will be unique. The cost estimate therefore assumes that a normal Solar Array development effort including development models (STM and PFM) will be required.
- The PCU and PDU costs have been derived from Mars Express.
- The Rosetta battery was the reference for Li-Ion Battery cost.

5.17.5.4 Harness

Harness costs were determined using ESA internal CERs. Since the harness architecture has to be newly developed, this has been taken into account in the cost estimate.

5.17.5.5 TT&C

For the TT&C sub-system the procurement is proposed to demand not only PFM and STM but also EM equipment to assemble an Avionics Test Bench (ATB).

The costs are derived from references from prior ESA missions, where minor equipment modifications were taken into account.

5.17.5.6 Data Handling

- The Data Handling System consists of a single box combining CDMU (Command and Data Management Unit) and RTU (Remote Terminal Unit).
- The CDMU is internally redundant.
- The data rate can be supposed to be low.
- For the cost estimate a partly customised off-the-shelf CDMU has been assumed.
- An additional EM has been considered for the ATB.

5.17.5.7 Structure

The envisaged structure is similar to Cluster. Therefore this has been used as a reference but the costing is based on the ESA "low cost mission" internal cost model.

5.17.5.8 Thermal Control

The Thermal Control Equipment includes only passive hardware such as paint and MLI. The Thermal Control Subsystem engineering activities such as thermal control analysis and configuration design are included in the Engineering cost at payload level. Specific instrument thermal hardware is included within the payload costs.

5.17.5.9 On-Board Software

Both the Data Management Software and the AOCS Software are considered to be based on existing on-board software, with only the payload management being specifically developed for IMM.

The cost estimates for the IMM Data Management and AOCS Software are based on the costs for modified existing software on other ESA missions.

5.17.5.10 Ground Support Equipment (GSE)

The cost estimate for GSE covers the costs for all Electrical and Mechanical GSE required for the platform. It has been taken into account that the GSE will be mainly based on existing hardware and designs. Accordingly a standard ratio observed on past projects has been applied. Extra GSE equipment has been assessed for FM 2-4, to take into account parallel testing in order to meet the planning constraints.

5.17.5.11 Platform Assembly, Integration and Test

The platform AIT cost estimate includes the costs for all platform mechanical and electrical integration activities and tests, as well as the mechanical mating of the platform and the payload. The cost estimate is based both on a cost estimate relationship and on an independent AIT planning assessment performed within the CDF, with which the results are in close agreement.

5.17.5.12 Project Office Activities

The Project Office costs at Subsystem and platform level include the costs for

- Management and Control (including overheads on subcontracts)
- Product Assurance
- Engineering and documentation including payload interface engineering both at system and subsystem level, except for propulsion

The overall proportion of the prime activities for the FM2-4 satellites has been lowered to take into account the overlap of the supervision activities between:

- non-recurring and recurring phases
- each recurring model

5.17.5.13 Payload

The IMM payload for each spacecraft consists of the following instruments:

- Thermal Plasma Monitor (TPM)
- Mid-Energy particle Monitor (MEM)
- High Energy particle Monitor (HEM)
- Magnetometer (MAG)
- Waves Instrument (WAVE)
- GPS Receiver Ionospheric Sounder (GRIS)

The instrument cost assessment is characterised by the rather limited amount of available reference material and technical data on the instruments.

It has been assumed that institutes rather than industry will procure the instruments.

The cost estimates are based on similar instruments or equipment with matching technology.

The monitor cost estimates are rooted in equipment and sensors on XMM. Magnetometer costs are modified data from Cluster. GPS costs are mainly customized data from METOP.

To adapt the different costs, various ESA internal cost models have been used. It has to be noted that for more detailed estimates, further hypotheses are necessary.

5.17.5.14 Design Maturity Margins

The Design Maturity Margins account for unknown design aspects not yet identified at the level of this feasibility study. These provisions are not risk margins (i.e. cost impacts due to the realisation of a stochastic event) and must be considered as part of the total industrial cost as well as of the payload cost.

Design Maturity Margins:

- 10% for platform
- 20% for payload

5.17.5.15 Launch Dispenser

The Launch Vehicle Dispenser is derived from an existing concept, therefore minor development activities are taken into account in the cost estimate.

This estimate is based on a fully competitive environment with an optimised industrial architecture.

5.17.5.16 Launcher

The cost for the GSLV launch presented is based on a price quotation found on the Internet.

5.17.6 Cost Risk Estimate

No specific cost risk estimate has been performed. This will have to be accounted for as part of the ESA level contingencies.

5.17.7 Insurances

- Satellite:
  Due to the operational nature of the mission, an insurance amount of 7.5% has been considered. This value is based on recurrent market prices.
- Launcher
  Launcher insurance cost is assumed to be 7.5% of the launch cost, by similarity to satellite insurance.

5.17.8 Qualitative Cost Assessment

This estimate is based on a fully competitive environment with an optimised industrial architecture.

No Geographical Distribution effect is accounted for.

Some clarifications will be needed with respect to the specific operational nature of the IMM mission. In particular, some reliability and availability figures will have to be set as part of the requirements definition of such an operational mission. Some further refinements in order to cope with such figures may lead to consideration of an on-ground satellite spare.

5.17.9 Cost Breakdown

Due to the different distribution requirements, cost figures are not included in this report but in a separate document [RD5].

This page intentionally left blank

6. Solar Wind Monitor (SWM)

The Solar Wind Monitor is designed to provide near-real-time monitoring of the solar wind upstream of the Earth.

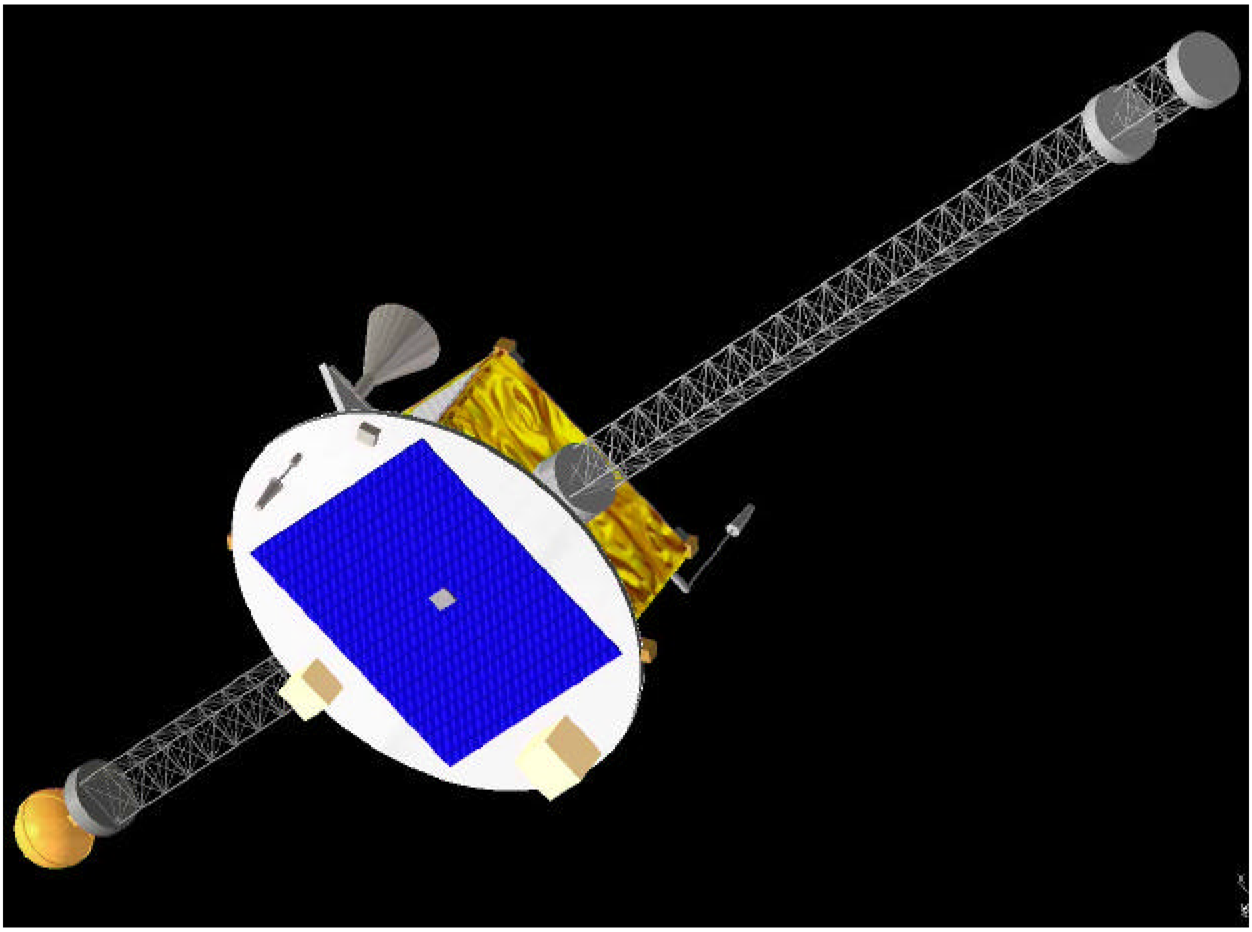


Figure 6-1: Solar Wind Monitor in Flight

6.1 Payload

Measurements of the solar wind conditions upstream of the Earth's magnetosphere are essential to enable tracking of solar wind disturbances propagating in the interplanetary medium towards the Earth. By performing such measurements from the Lagrangian L_1 point between the Sun and the Earth, SWM will be able to provide the Space Weather user community with data on the solar wind structures prior to their arrival to the Earth's vicinity, and as such a valuable early warning of upcoming geomagnetic events. The SWM payload will serve this purpose by monitoring the local magnetic field conditions and charged particle distributions. In addition it will observe the radio signatures of the propagating disturbances in the outer solar corona and interplanetary medium.

The following instruments have been selected:

- Thermal Plasma Monitor (TPM)
- Mid-Energy particle Monitor (MEM)
- Magnetometer (MAG)
- Coil Radio Spectrograph (CRS)

6.1.1 Payload Requirements

The SWM spacecraft will operate in the solar wind where the magnetic field is much weaker and the plasma environment is more tenuous than inside the Earth's magnetosphere. This is why the SWM payload requires excellent electromagnetic, magnetic and electrostatic cleanliness spacecraft characteristics, even more demanding than in the case of IMM. Other specific instrument requirements are given below.

6.1.1.1 Thermal Plasma Monitor (TPM)/Mid-Energy Monitor (MEM)

The requirements for both the TPM and MEM are very similar to those of the IMM payload, allowing reuse of the design and thus saving some development effort. The TPM should be able to detect both ions and electrons in the low energy range, between a few eV and 40 keV. The MEM will cover a different energy range, enabling determination of ion and electron distributions whose energies are between 40 keV and 2 MeV. Both the TPM and the MEM should sample most of the 4π solid angle with a 45° angular resolution or better. The time resolution should be better than one minute for both particle monitors.

6.1.1.2 Magnetometer (MAG)

Determining the local magnetic field topology is important in helping to diagnose the propagation of geomagnetic disturbances, such as the magnetic clouds associated with some CMEs. To avoid interference with the measurements, the spacecraft DC magnetic background should be lower than 0.3 nT at the magnetometer boom tip. The two suggested full scale ranges for operation upstream of the solar wind are 0 - ±64 and 0 - ±256 nT with a minimum 1-minute time resolution (either range is acceptable for SWM[5]). Measurements are required in three orthogonal directions to determine magnetic field direction.

No specific requirements for the instrument sensitivity or resolution were provided, though they are expected to be demanding, and might be an issue when deciding whether to use OTS equipment.

6.1.1.3 Coil Radio Spectrograph (CRS)

Remote sensing measurements of radio waves are useful for monitoring the propagation of disturbances in the solar atmosphere. This role would be fulfilled by an instrument able to analyse the time and space variation of the solar radio bursts – especially the so-called Type II bursts, which are connected with CMEs. This instrument would therefore ideally be a radio spectrograph, in the range 40 kHz - 300 MHz[6].

6.1.2 Payload Description

The requirement for most of the detectors to have a good angular coverage in almost every direction has been one of the main drivers leading to a spin-stabilised spacecraft platform. The the combination of the FOV of the instruments and the spin of the spacecraft about its axis make it possible to sample most of the solid angle sphere in a simple and effective way, and this is assumed in the description of the instrument design given below.

6.1.2.1 TPM Instrument

This instrument will be a Top-Hat electrostatic analyser, similar to previous designs flown on WIND (3D PLASMA), Equator-S (3DA), or Cluster II (PEACE). The sensor heads will be integrated in the same box as the control electronics. This single unit would be mounted on the spacecraft side so that the pointing direction is perpendicular to the spin axis (z axis), the instrument having a FOV of 180° × 15° (polar × azimuthal angles), this leading to a full 4π solid angle coverage after one spacecraft revolution.

The proposed instrument specifications are as follows:

- mass 5 kg
- power 8 W
- telemetry rate 2 kbps
- dimensions 250 x 200 x 200 mm
- (design) temperature operating range -10/+20°C
- non-operating -30/+60 °C

[5] This instrument is proposed for use on IMM as well, in which case the range must be the broader one because the magnetospheric field is much stronger.
[6] Note that the proposed instrument does not meet this frequency requirement, which is very demanding, but also that waves with frequencies above 40 MHz can be detected by ground-based facilities.

6.1.2.2 MEM Instrument

The detector uses an array of solid state detectors in 6 energy channels and 8 polar angle channels at a time. The instrument FOV will be 180° × 20°. This FOV should not be obstructed by any booms, and as a consequence, the current location of the instrument should be reconsidered in future studies.

The instrument proposed has the following characteristics:

- mass 2 kg
- power 4 W
- telemetry rate 2 kbps
- dimensions 150 x 150 x 150 mm
- (design) temperature operating range -10/+20°C
- non-operating -30/+60 °C

6.1.2.3 MAG Instrument

A well-established 3-axis flux-gate magnetometer will be used. The selected design is the same used as baseline for IMM, as this will reduce the cost.

This consists of an electronics box containing the DPU and two separate sensors, one mounted at the end of a 4 m boom and the other 0.5 m further inboard. The use of the boom will reduce the interference from the spacecraft, while the combined use of the two sensors will enable determination of its magnitude. This is particularly important considering the small magnitude of the interplanetary magnetic field (IMF). The sensor duplication will at the same time provide some degree of redundancy.

It is an off-the-shelf instrument having the following specifications:

- mass 1.5 kg (1 kg CPU, 2 x 0.1 kg sensors, 0.3 kg harness)
- power 2 W
- telemetry rate 0.2 kbps
- dimensions:
  - CPU 200 x 100 x 150 mm
  - sensors 40 x 40 x 40 mm

6.1.2.4 CRS Instrument

The L1 spacecraft location is suitable for low frequency solar radio wave detection without being affected by terrestrial and magnetospheric perturbations. A novel instrument concept using three orthogonal magnetic loop antennae, being studied at CNRS/LPCE in Orléans, France, would be able to make measurements in the 50 kHz-30 MHz range while keeping the mass and power figures at a reasonable level. The characteristics of a single magnetic loop antenna are as follows:

Bandwidth	**50 kHz to 30 MHz**
Sensitivity threshold	$<1.0\times10^{-6}$ nT.Hz$^{-1/2}$ in [700 kHz – 20 MHz]
	4.5×10^{-6} nT.Hz$^{-1/2}$ at 100 kHz
	9.0×10^{-6} nT.Hz$^{-1/2}$ at 50 kHz
Output impedance	50 Ω
Power supply	230 mW at ±7V (regulated)
Mass	450 g (antenna + preamplifier)
Dimensions	300 mm in diameter

Table 6-1: Single Magnetic Loop Antenna Characteristics

Three of these magnetic loop antennas arranged orthogonally – together with three preamplifiers – will be enclosed in a spherical case and mounted on a boom.

DPU/electronic box specifications have been estimated as follows:

- dimensions 20 x 10 x 5 cm
- mass 2 kg
- power 5.7 W
- telemetry rate 2.5 kbps
- Total instrument mass 3.7 kg, including
  - 1.35 kg for the antennae assembly (excluding the boom)
  - 0.3 kg for the harness
- (design) temperature operating range -20/+50°C,
- non-operating temperature range -30/+60 °C.

6.1.3 Payload Summary Budgets

Instrument name	Mass (kg)	Mass inc. 15% margin (kg)	Power (W)	TM rate (kbps)
Thermal Plasma Monitor	5.0	5.8	8.0	2.0
Mid-Energy particle Monitor	2.0	2.3	4.0	2.0
Magnetometer	1.5	1.7	2.0	0.2
Coil Radio-Spectrograph	3.7	4.2	5.7	2.5
	12.2	**14.0**	**19.7**	**6.7**

Table 6-2: Instrument Mass and Power Budgets

6.1.4 Options for Future Study

A new operational technique allows for forecasting of solar indices up to one solar rotation in advance. It is based on the analysis of interplanetary UV background maps obtained once every two days, which can be observed with a small UV photometer (around 5 kg) placed on a spinning spacecraft near the L1 point, i.e. far from contamination by Earth's exosphere (geocorona). It is therefore suitable for a platform such as SWM.

This technique has been successfully tested on the SWAN/SOHO UV background data [RD6]. The procedure is effective for the forecasting of solar indices in the range of one to two weeks, which is not possible by any other method. Accurate results have been obtained for such indices as the Solar Mg II index, the 10.7 cm radio flux (and the corrected E10.7 index), the full-disk Lyman alpha flux (121.6 nm) and correlated solar UV and EUV fluxes. These forecast values can be applied to predict the exobase temperature and atmospheric drag effects on satellites. A preliminary study by CLS has shown that the 14-day forecast of the 10.7 cm flux obtained from the SWAN data in 2000 is more accurate than the forecast values obtained by other methods.

6.2 Mission Analysis

6.2.1 Orbit Selection

The SWM basic requirements for orbit selection are:

1. In the solar wind, upstream of the Earth
2. Uninterrupted ground contact

Requirement 1 can be met on the following orbits:

- Orbits near Earth-Sun libration (or Lagrange) points L_1, L_4 or L_5
- Marginally, interplanetary orbits, in particular the orbit of the Earth around the Sun with a given phase difference with the Earth

Figure 6-2 shows the location of the five Lagrange points of the Sun-Earth system.

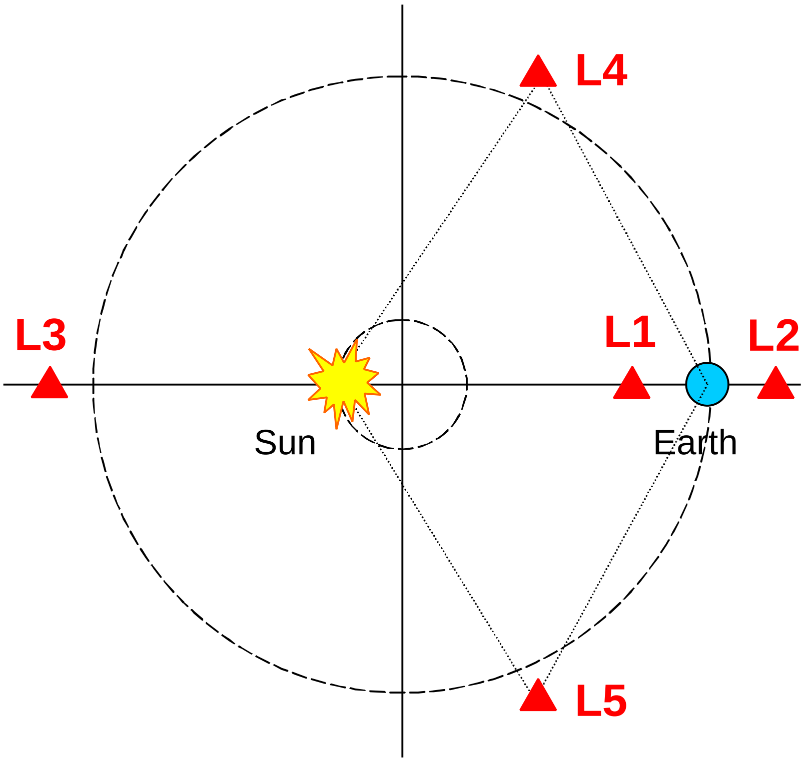


Figure 6-2: The five Lagrange Points in the Earth-Sun System

The second requirement can be met

- on LEO only with a very large number of ground stations,
- ideally on GEO,
- on Lagrange and interplanetary orbits only by selecting 3 stations approximately 120° apart in longitude.

A trade-off analysis, including also the launch energy needed for reaching the operational orbit, led to selection of an orbit around L_1, covered by a set of three ground stations.

6.2.2 Launch and Early Orbit Phase (LEOP)

The goal of the LEOP is to place the satellites in their final orbit as selected in the preceding section. In the case of this mission, the final orbit is reached only after about 3 months. However, as some of the instruments will already be activated during transfer, the LEOP will cover only the first hours of the mission.

6.2.2.1 Launch

The target orbit is 1.5 million km away from the Earth. The initial part of the transfer orbit is very close to a parabola, therefore the orbital energy to be communicated to the spacecraft by the launcher is close to the Earth escape velocity.

As SWM is to be launched together with SAM in a dual launch configuration (see section 4.2.1), the class of small launchers (such as Rockot or PSLV) cannot provide sufficient performance; a medium class launcher has to be selected. In this class, the most cost effective launcher is Soyuz with a Fregat upper stage, or alternatively Dnepr-M with a Varyag upper stage. For both launchers, the performance for parabolic orbit is about 1600 kg. Such a launcher can therefore inject the spacecraft directly into a transfer orbit to L_1.

Before injection into transfer orbit, in order to acquire the proper trajectory orientation, the launcher will coast on a low Earth circular parking orbit for no more than 1.5 hours.

6.2.2.2 Transfer Orbit

The selected final orbit is a halo orbit around L_1. It is possible to be captured into such an orbit without an insertion manoeuvre. The only manoeuvres needed during transfer are therefore:

- One manoeuvre, between 1 and 10 days after launch, for removing the launcher's dispersion. For this, the *DV* does not exceed 40 m/s
- One or two trajectory adjustment mid-course manoeuvres, and possible halo insertion adjustment manoeuvre: *DV* of 5 m/s.

6.2.3 Operational Orbit

6.2.3.1 Orbits Around Libration Point L_1

Orbits around the co-linear point L_1 are unstable, but when including orbit maintenance, they can be looked at as orbits around a central body. There are two classes of orbit around L_1, characterised by their 'amplitude' (size):

- An orbit with a small amplitude which crosses the Earth-Sun line of sight. This is called a *Lissajous* orbit.
- An orbit with a large amplitude (about 800,000 km) which, seen from the Earth, keeps in a tube around the Earth-Sun line of sight. This is called a *halo* orbit.

To ensure good communication with the Earth, an orbit that does not cros the Earth-Sun line, such as a halo orbit, is preferable. This was also the orbit selected for ESA solar observatory SOHO. However, on a Lissajous orbit, the time between two crossings of the Earth-Sun line is quite long (about 6 years) and it is even possible to perform a manoeuvre to 'jump' the undesired crossing. Therefore, if it is critical that the direction of the spacecraft does not extend too far from the Sun direction, a small amplitude Lissajous orbit can be contemplated.

A typical halo orbit and the Earth to halo transfer are shown in a rotating ecliptic reference frame centred on the Earth with the *x*-axis toward the Sun and the *z*-axis normal to the ecliptic on Figure 6-3 (*x*-*y* plane). The halo orbit is shown over five years. During transfer and the first revolution in halo orbit, tick marks are shown every 10 days. The orbital period is about six months.

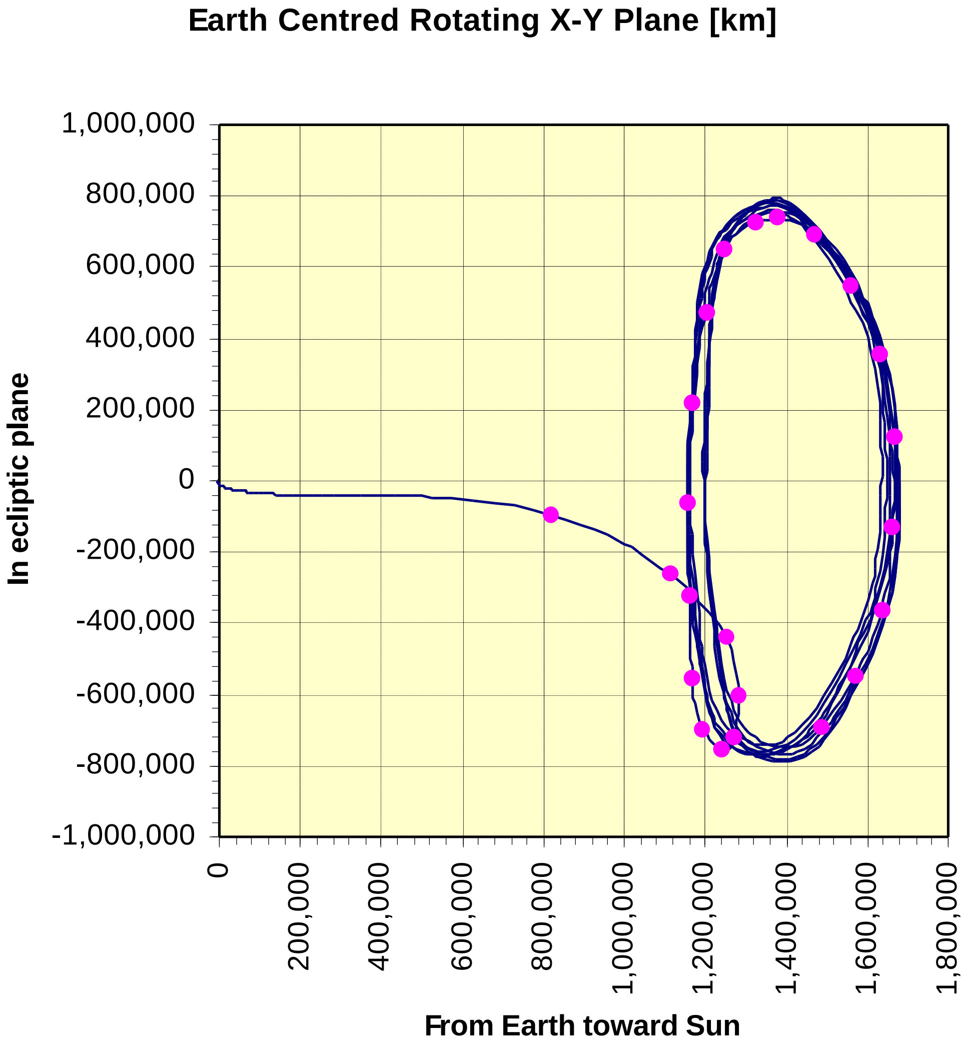


Figure 6-3: Transfer and Halo Orbit in the *x*-*y* Rotating Frame Centred on the Earth

Figure 6-4 shows the same orbit in the *x*-*z* plane (with tick marks every 10 days for the first year of the mission).

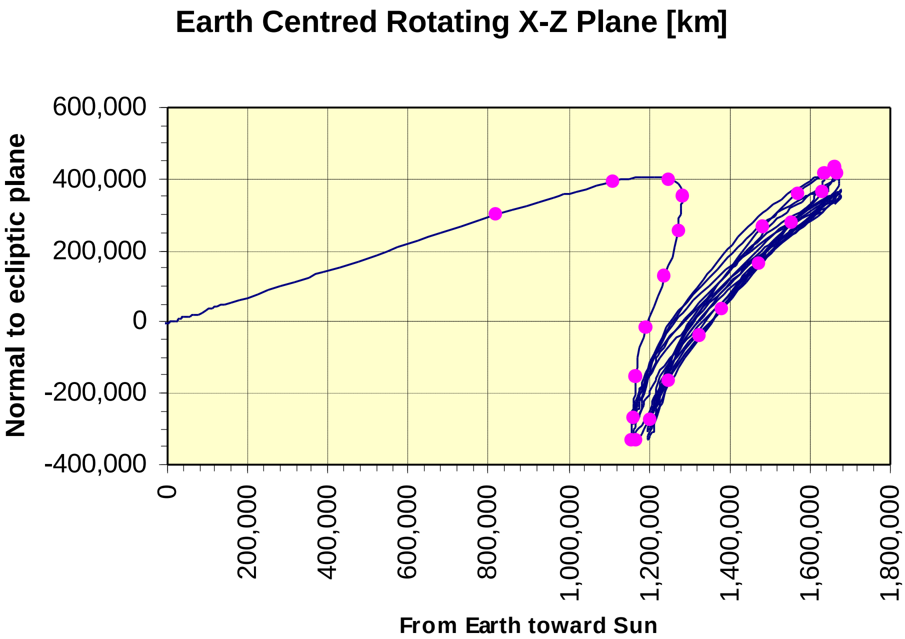


Figure 6-4: Transfer and Halo Orbit in the *x-z* Rotating Frame Centred on the Earth

Figure 6-5 visualises the *y-z* plane, showing the halo orbit staying within a tube around the Earth-Sun direction. Tick marks are shown every 10 days over the first year of the mission.

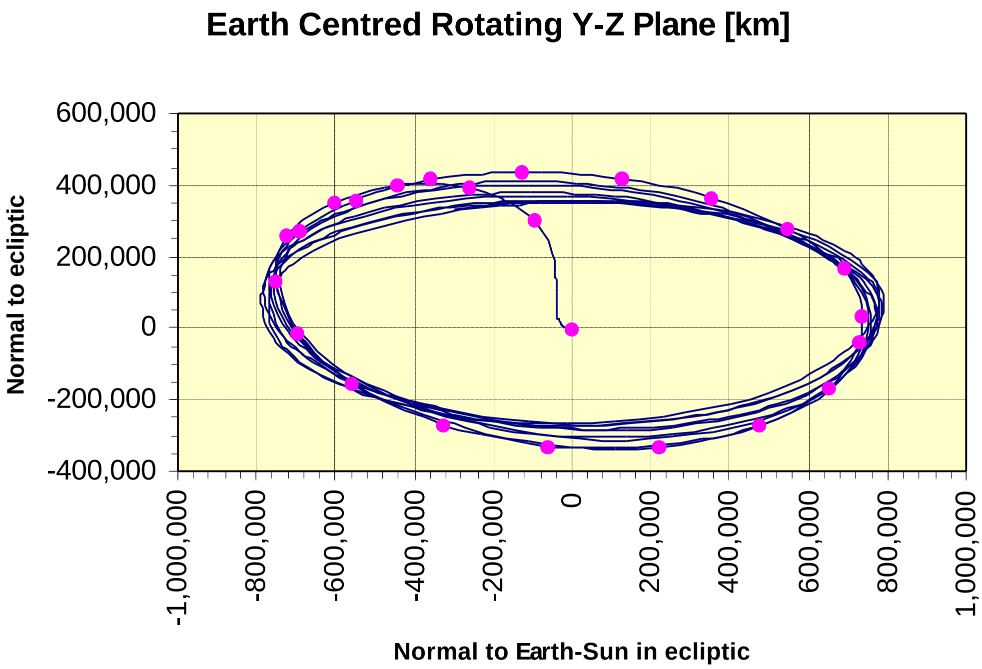


Figure 6-5: Transfer and Halo Orbit in the *y-z* Rotating Frame Centred on the Earth

6.2.3.2 Orbit Maintenance

As the halo orbit is inherently unstable, orbit maintenance is needed. Efficient strategies can be used, which lead to a total *DV* of about 2 m/s per year, with manoeuvres undertaken at a frequency of about one every month.

6.2.4 Ground Coverage

Ground coverage during transfer orbit and operational orbit is continuous when three stations approximately 120° apart in longitude are selected. The ESA stations do not offer a set with such properties, and therefore one non-ESA station has to be included. The best selection is

- Perth (longitude 115.9°)
- Villafranca (-4.0°)
- NASA station Goldstone (-116.9°)

With these stations, continuous coverage can be guaranteed.

The halo orbit is such that the viewing direction of the spacecraft from the station is always separated from the Earth-Sun direction by at least 11° (Figure 6-6). This allows communication undisturbed by Sun interferences.

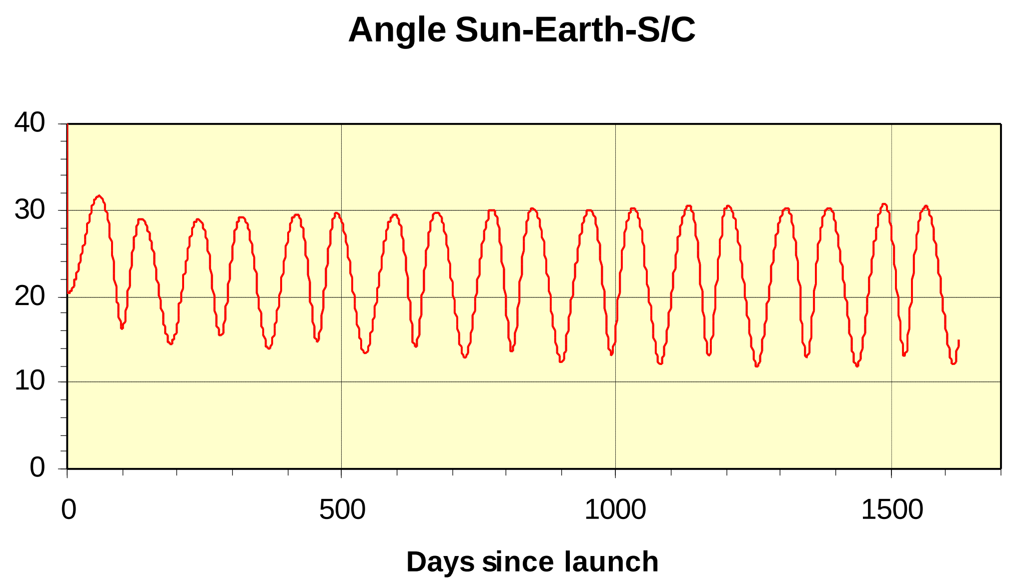


Figure 6-6: Angle Sun-Earth-Spacecraft during the 5-year Mission

Figure 6-6 also shows that the halo orbit is also such that the viewing direction of the Earth from the spacecraft is never separated from the Sun-Earth direction by more than 31°. This allows the use of a non-steerable wide lobe antenna for assuring communication with the Earth.

The same is already true during transfer, except for the first hours after launch, as shown in Figure 6-7, which shows the angle over the first five hours of the transfer. It also shows the distance to the Earth centre.

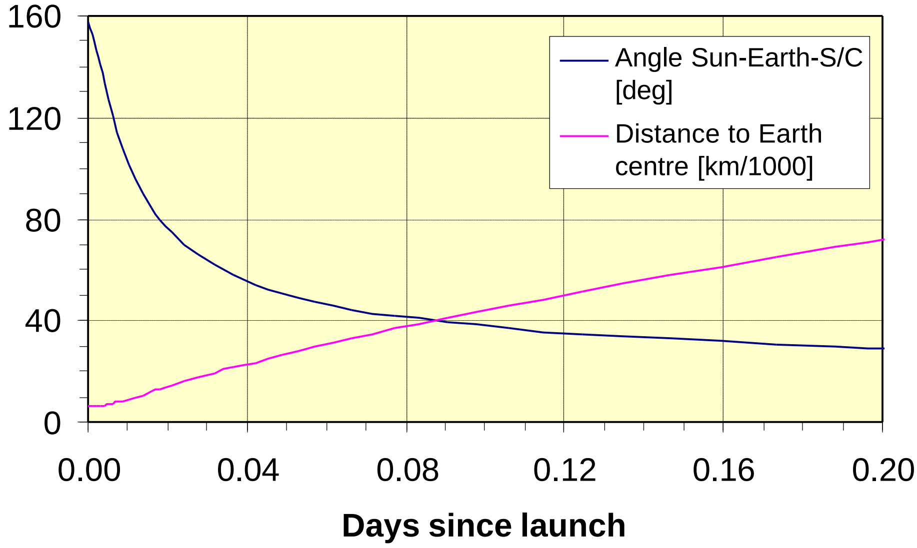


Figure 6-7: Angle Sun-Earth-Spacecraft and Distance to the Earth Centre

6.3 Radiation

The SWM mission (and likewise SAM) is predominantly in interplanetary space, and therefore is only affected by solar proton events and cosmic rays. There will be some radiation dose accumulation during the transfer from LEO to L_1, but this will be negligible compared with the radiation dose received from solar proton events, and so has been ignored in this analysis.

The baseline mission assumes the following orbital ephemera:

Orbit type	L_1
Launch date	1-Jan-06
Launch hour	0:00
Mission duration	5 years

The JPL-91 model[7] was used to calculate the solar proton fluence for a 5 year period with a confidence level of 90% that the fluences would not be exceeded. While the mission starts in the middle of a period of solar minimum activity, where no solar proton events are expected, the full 5 year lifetime was assumed for the mission (rather than the 2.5 year period of solar maximum activity) as a worst case for possible replacement satellites that would have to survive such an environment. This fluence spectrum was then used to calculate the total ionising dose, non-ionising dose and solar cell degradation electron fluences.

Figure 6-8 shows the SWM mission agains the solar cycle.

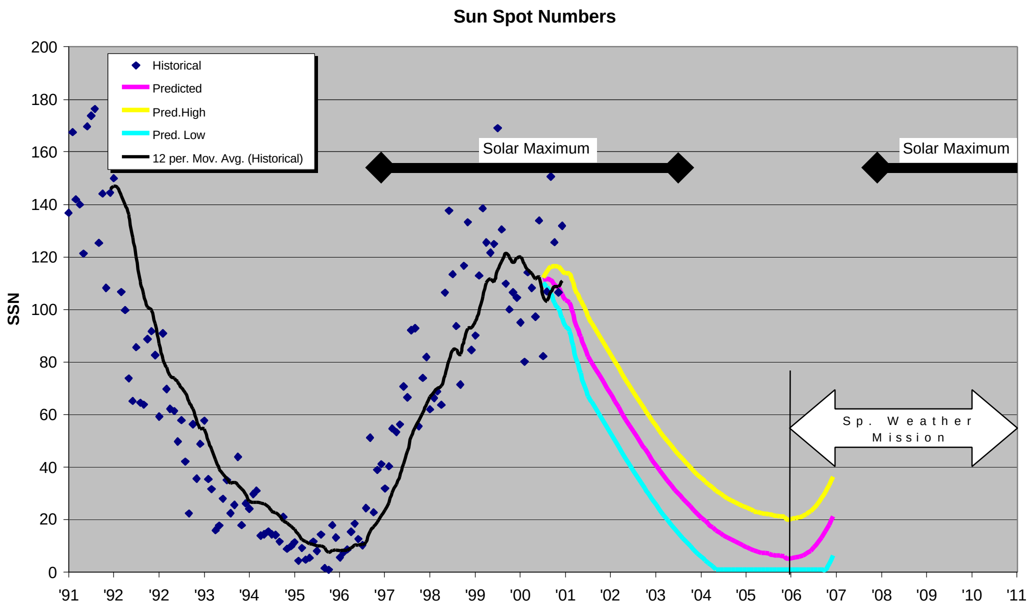
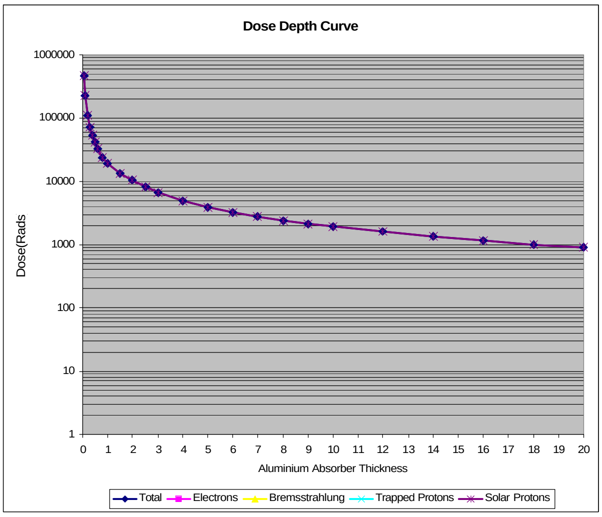


Figure 6-8: SWM Lifetime against Solar Cycle

[7] The ECSS reference model for solar proton fluence calculations.

6.3.1 Total Ionising Dose

The total ionising dose received will be principally from solar proton events and is not expected to exceed 5 krad for a nominally shielded (4 mm of aluminium) component. Figure 6-9 shows the ionising dose versus aluminium thickness for a 5-year interplanetary mission during solar maximum activity. Doses are due entirely to solar proton events.



Space Weather 2, v0.1

Aluminium Absorber Thickness (mm)	Dose (Rad) Total	Electrons	Bremsstrahlung	Trapped Protons	Solar Protons
0.05	466080	0	0	0	466080
0.10	230450	0	0	0	230450
0.20	110730	0	0	0	110730
0.30	73256	0	0	0	73256
0.40	54006	0	0	0	54006
0.50	41674	0	0	0	41674
0.60	33354	0	0	0	33354
0.80	23868	0	0	0	23868
1.00	19153	0	0	0	19153
1.50	13316	0	0	0	13316
2.00	10366	0	0	0	10366
2.50	8216	0	0	0	8216
3.00	6778	0	0	0	6778
4.00	**4942**	**0**	**0**	**0**	**4942**
5.00	3876	0	0	0	3876
6.00	3211	0	0	0	3211
7.00	2748	0	0	0	2748
8.00	2430	0	0	0	2430
9.00	2137	0	0	0	2137
10.00	1916	0	0	0	1916
12.00	1600	0	0	0	1600
14.00	1342	0	0	0	1342
16.00	1146	0	0	0	1146
18.00	1010	0	0	0	1010
20.00	907	0	0	0	907

Figure 6-9: Ionising Dose versus Aluminium Thickness

6.3.2 Solar Cell Degradation

The solar cells will suffer damage principally from the solar proton events. Mitigation techniques include the use of radiation tolerant solar cells (e.g. GaAs), oversizing of the solar arrays to ensure adequate power at the end of life, and the use of thicker coverglass. Table 6-3 shows Solar Cell Degradation for Various Coverglass Thicknesses for a 5 year interplanetary mission during solar maximum activity.

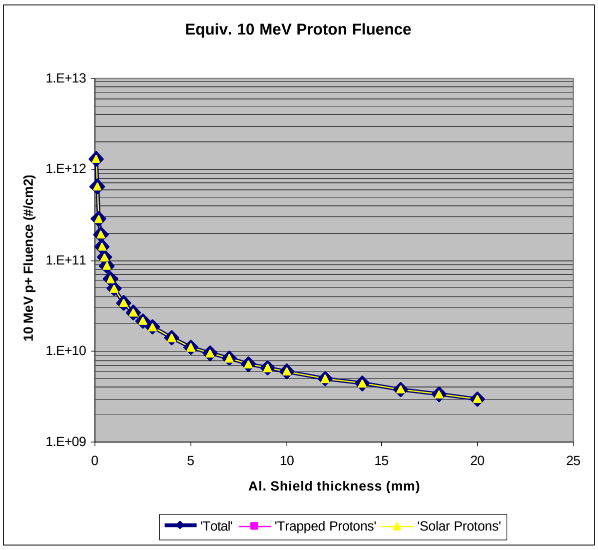
Space Weather 2, GaAs v0.1, Si v0.1

Coverglass Thickness (microns)	1 MeV Equivalent electron fluence (#/cm2) VOC GaAs	VOC Silicon	PMAX GaAs	PMAX Silicon	ISC GaAs	ISC Silicon
0	8.8E+15	4.9E+15	6.3E+15	4.9E+15	3.82E+15	1.4E+15
76	3.3E+14	7.5E+14	2.4E+14	7.5E+14	1.15E+14	3.3E+14
152	1.5E+14	3.9E+14	1.1E+14	3.9E+14	5.00E+13	1.9E+14
305	7.2E+13	1.9E+14	5.1E+13	1.9E+14	2.19E+13	1.1E+14
509	4.2E+13	1.0E+14	3.0E+13	1.0E+14	1.19E+13	6.5E+13

Table 6-3: Solar Cell Degradation against Coverglass Thicknesses

6.3.3 Non-Ionising Dose

As well as ionising dose, particles can lose energy through non-ionising interactions with materials, particularly through 'displacement damage' or 'bulk damage', in which atoms are displaced from their original sites. This can alter the electrical, mechanical or optical properties of materials and is an important damage mechanism for electro-optical components (e.g. solar cells and opto-couplers) and for detectors such as CCDs. Figure 6-10 shows non-ionising dose versus aluminium thickness for a 5-year interplanetary mission during solar maximum activity.



'Spherical Al Shield Radius'	'Total'	'Trapped Protons'	'Solar Protons'
0.05	1.31E+12	0.00E+00	1.31E+12
0.1	6.49E+11	0.00E+00	6.49E+11
0.2	2.88E+11	0.00E+00	2.88E+11
0.3	1.92E+11	0.00E+00	1.92E+11
0.4	1.42E+11	0.00E+00	1.42E+11
0.5	1.07E+11	0.00E+00	1.07E+11
0.6	8.83E+10	0.00E+00	8.83E+10
0.8	6.23E+10	0.00E+00	6.23E+10
1	4.83E+10	0.00E+00	4.83E+10
1.5	3.37E+10	0.00E+00	3.37E+10
2	2.70E+10	0.00E+00	2.70E+10
2.5	2.18E+10	0.00E+00	2.18E+10
3	1.86E+10	0.00E+00	1.86E+10
4	1.40E+10	0.00E+00	1.40E+10
5	1.11E+10	0.00E+00	1.11E+10
6	9.42E+09	0.00E+00	9.42E+09
7	8.27E+09	0.00E+00	8.27E+09
8	7.23E+09	0.00E+00	7.23E+09
9	6.60E+09	0.00E+00	6.60E+09
10	6.02E+09	0.00E+00	6.02E+09
12	5.06E+09	0.00E+00	5.06E+09
14	4.41E+09	0.00E+00	4.41E+09
16	3.82E+09	0.00E+00	3.82E+09
18	3.42E+09	0.00E+00	3.42E+09
20	3.02E+09	0.00E+00	3.02E+09

Figure 6-10: Non-ionising Dose against Aluminium Thickness

6.3.4 Cosmic Rays

The cosmic ray environment will be the same as encountered by other interplanetary missions, such as SOHO and Rosetta. Extreme single event effects will only be encountered during solar proton events.

However, as it is principally during such solar proton events that the monitoring of the space environment is crucial for the mission success, it is necessary to ensure that the spacecraft systems are capable of operating during such intense events as shown in the M=8 LETspectrum, below. This might require the exclusion of COTS parts from the design.

Note that this is shown for solar minimum, since this is the worst case for the cosmic ray environment. (The solar wind and magnetic field provide shielding from cosmic rays during solar maximum.)

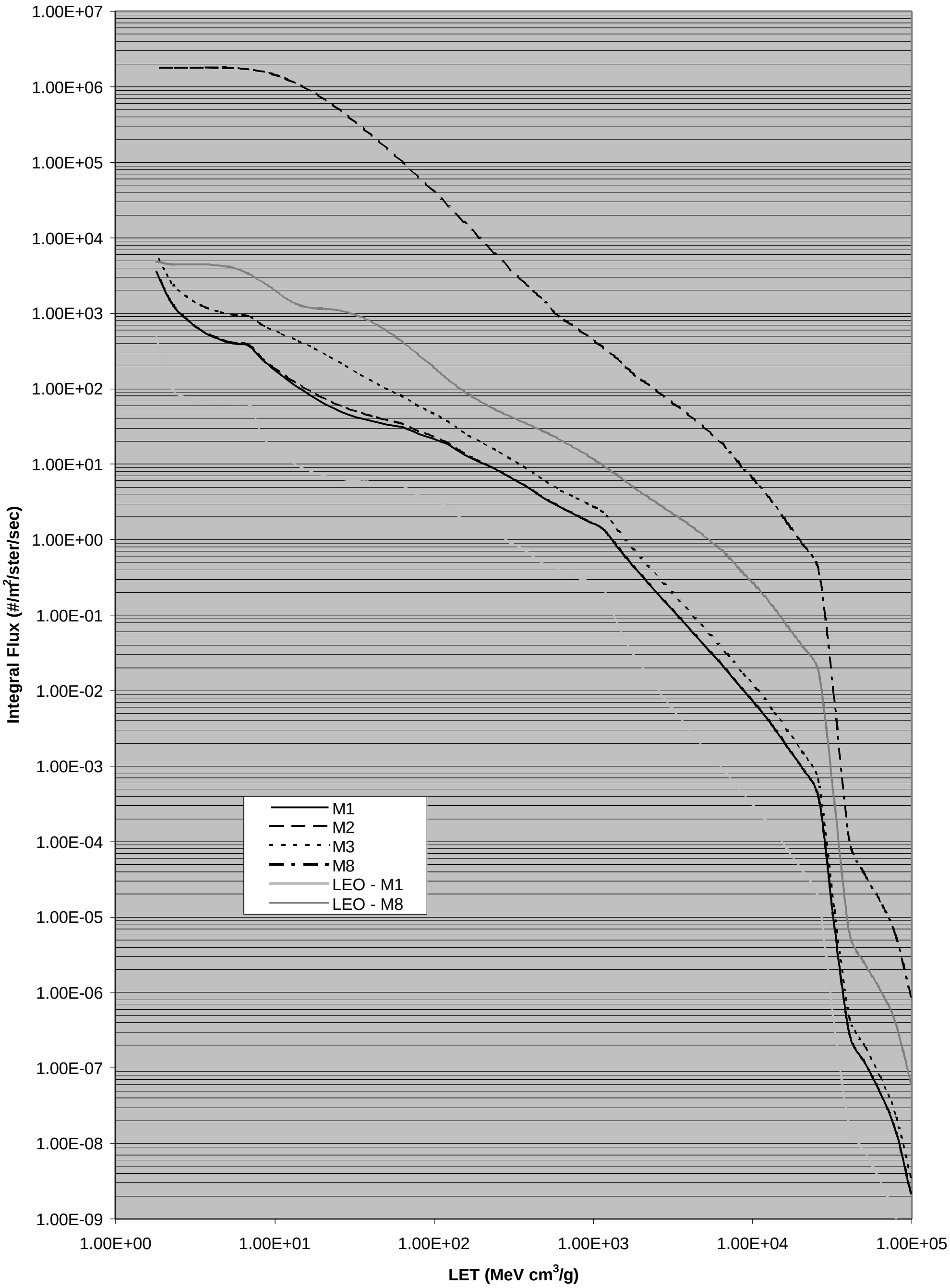


Figure 6-11: Cosmic Ray Linear Energy Transfer Spectra (Solar Minimum, 1g/cm^2 shielding)

6.3.5 Conclusions

The nominal radiation environment to be encountered by the mission at L_1 is benign with only background cosmic ray upsets to consider. However, during solar proton events the success of the mission requires that it continue with nominal operations, and so it will require protection against service interruptions due to single event upsets and high background levels in the systems, e.g. AOCS startrackers. The overall radiation damage, though, is relatively low, with a 5 krad dose to be expected for nominally shielded components.

6.4 Systems

6.4.1 Requirements and Design Drivers

The primary Solar Wind Monitor objective is to sample the local properties of the Solar Wind ahead of the Earth's magnetosphere. The observation requirements include:

- Bulk flow of ions
- Magnetic field
- High energy ions
- Low frequency radio waves to remote-sense interplanetary plasma structures
- Mid energy particles

The mission objectives and the operational characteristics of the Space Weather programme drive the system requirements and the constraints for the SWM:

- An orbital location within continuous and unobstructed flow of the Earth-incident Solar Wind
- Near real-time data flow
- Continuous coverage
- Launch of the first SWM in 2006
- 5-year lifetime

The design drivers for the system are:

- Launch together with SAM (see section 4.2)
- A halo orbit around the L_1 point between Earth and Sun (see section 6.2.3)
- Directional sensing of plasma and fields
- Cluster-type EMC, electrostatic, and magnetic cleanliness

6.4.2 Assumptions and Trade-Offs

A preliminary analysis showed that the SWM should be a low-mass satellite orbiting the L_1 Lagrange point, spinning at 15 rpm.

The halo orbit requires very little *DV* for insertion and maintenance. These manoeuvres can be performed with a low-thrust propulsion system and with little propellant mass, allowing for a simple and lightweight spacecraft design of the minisat class. In addition, the orbit is very stable from a power generation point of view, with no eclipse and battery required only for contingency.

The attitude control can be simply based on spinning the satellite with its spin axis coarsely sun-pointing, allowing for a sufficient frontal surface continuously exposed to the Sun

The thermal design can be completely passive, and does not need to cope with eclipses.

The avionics hardware can be reused from existing platforms, due to the benign radiation environment.

Although low mass allows the use of cheap launchers, such as Rockot or Ariane 5 mini-sat, the orbital requirements require either a more powerful launcher such as PSLV or the addition of a solid booster to the satellite, thus increasing the total satellite mass.

The following set of system options were evaluated:

Trade Offs	Current Baseline	Study Option 1	Study Option 2	Study Option 3
Mission				
Number of Satellites	1.00	1.00	1.00	1.00
Orbit	L1 Halo	L1 Halo	L1 Halo	L1 Halo
Launch Date	Jan.06	Jan.06	Jan.06	Jan.06
System				
Satellite Type/Platform	Custom design	Custom design	Custom design	platform ?
Dry-mass class	400.00	300.00	150.00	30-300
Stabilisation	spinner	spinner	spinner	spinner
Payload				
Instrument Set	magnetometer, thermal plasma mon., mid-energy particle monitor, low-frequency radio-spectrometer	magnetometer, thermal plasma mon., mid-energy particle monitor, low-frequency radio-spectrometer	magnetometer, thermal plasma mon., mid-energy particle monitor, low-frequency radio-spectrometer	magnetometer, thermal plasma monitor
Launcher				
Launcher	Soyuz or PSLV	Rockot+STAR37	Ar5 min-sat	Rockot dual
Launch Strategy	direct injection	direct injection		
Propulsion				
Type of Propusion	no propulsion	no propulsion	solid	depending on platform

Table 6-4: System Options Evaluated

A study baseline and a study option were chosen according to the trade-off tree shown below:

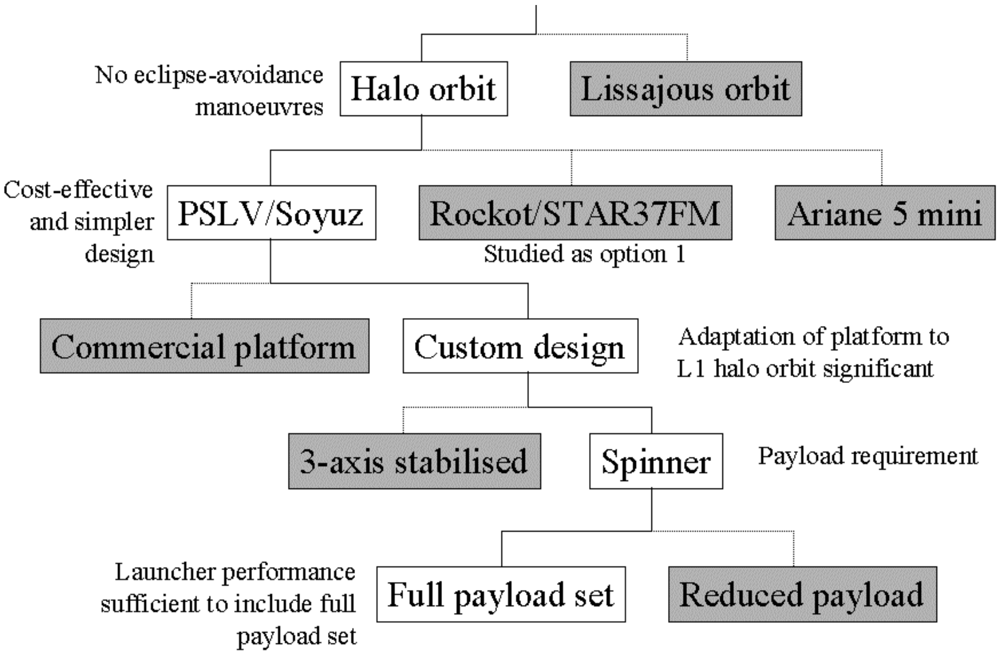


Figure 6-12: Trade-off Flow Chart

6.4.3 Launch Configuration

The choice of SWM launcher was strongly influenced by the mass, size and orbit of the SAM. Preliminary calculations showed that SAM (mass >500 kg) would be too heavy for Rockot or PSLV. If the SAM orbit were the same as the SWM orbit, both could be launched on a single

Soyuz-Fregat or Dnepr-Varyag (if available). Both have a large over-performance (1600 kg) with respect to the launch mass of SAM. The SWM could be included in the SAM launch, therefore saving the cost of a dedicated launcher for SWM (see section 4.2).

Table 6-5 reports the mass performances of various launchers to L_1.

Launcher	Performance to L_1 (kg)
Rockot	≥ 300
PSLV	400
Soyuz	1600
Dnepr	1600

Table 6-5: Launcher Mass Performances

As a baseline, SWM would be designed for a dual-launch with SAM on a Soyuz-Fregat. However, in case a dual launch would not be possible for programmatic reasons, the baseline design should be suitable for a single dedicated launch on PSLV.

It was decided during the study to perform a design iteration for the option of a launch on Rockot. Although the attachment of the solid motor complicates the design, the cost savings in comparison with a dedicated PSLV launch could justify this backup option.

The study flow led to two versions of the design model being created, namely:

- Version 1.11: Completed iteration of the baseline design.
- Version 0.20: First iteration of Option 1 (launch with Rockot + STAR37FM engine)

Option 1 was a ‘back-up’ option for the baseline design in case the total spacecraft mass of SWM was within the capabilities of the Rockot launcher.

Option 2 depends on the Ariane 5 mini-sat launch opportunity. Due to the higher costs with respect to a dedicated Rockot launch, and the need to find a co-passenger for an opposite mini-sat slot on Ariane 5, this option was not studied further.

Option 3 is a reduced payload option on a commercial platform. The total reduced mass would allow a shared launch on Rockot with the Solar Activity Monitor. However, the analysis showed that the SAM launch mass alone would exceed the Rockot performance. This option was abandoned.

6.4.4 Baseline Design

As already mentioned, the combination of a very stable orbit and a simple payload allows for a very lightweight and straightforward design. Emphasis has been put on cost reduction, minimising the operational and AOCS modes (the software is simplified) and selecting a box-like structure for quick and cheap assembly (see section 6.14). Three instruments out of four are almost identical to instruments on IMM.

Due to the EMC requirement, the CRS instrument and magnetometer sensor package need to be mounted on booms for which an original design has been selected. The attitude selected (spinning with spin-axis to the sun) minimises the complexity of the power and thermal subsystems.

6.4.4.1 Modes of Operation

The system Modes of Operation for the SWM mission are shown in the following table.

No.	Mode Name	Definition
1	Launch Mode	*From lift-off until upper stage separation*
		Battery fully charged (charging until 8 min before lift-off)
		Payload instruments switched off
		Power S/S (PCU, PDU, TCU), OBDH S/S (CDMU) switched on
		Comms S/S for TC switched on
2	Transfer mode	*From stage separation until halo orbit*
		Payload instruments switched off
		AOCS initialisation - coarse mode
		Power from solar arrays
		TT&C active
3	Initialisation Mode	*From halo Orbit acquisition until normal operation*
		Mode entered after transfer phase or during recovery from safe phase
		Detumbling and stabilisation
		AOCS initialisation - fine mode
		Power from solar arrays
		Payload instruments initialisation
4	Operational Mode	*SSOOperations - Fine Pointing Mode*
		Payload operational
		S/C sun pointing
		AOCS active and satisfying the pointing requirements
		TT&C active via medium gain antenna (MGA)
5	Safe Mode	*Failure Recovery Mode*
		S/C attitude automatically set to sun pointing
		Payload Instruments switched off
		Failure detection, isolation and recovery to normal mode are executed by the ground.
		TT&C active via low gain antenna
		TM/TC access to OBDH is guaranteed to enable failure detection and reconfiguration

Table 6-6: SWM System Modes of Operation

6.4.5 Budgets

6.4.5.1 Mass Budget

The mass identified in the system budget is based on the specified values of the individual units and subsystems. Depending on the maturity status of the items, contingency is applied at unit/item level. Generally, for each piece of equipment a mass margin is applied in relation to its level of development: i.e.

- 5% for off-the-shelf items
- 10% for items qualified but requiring some modification
- 20% for items to be developed

A system-level mass margin of 20% is placed on the spacecraft dry mass (dry mass including sub-system margins). The S/C mass budget for the baseline is displayed in the table below:

Solar Wind Monitor Mass Budget					
		Target Spacecraft Mass at Launch		300	kg
		Below Mass Target by:		92	kg
	Without Margin	Margins %	Margins kg	Totals kg	% of Total
1. Structure	55.7 kg	20.0	11.1	66.8	32.20
2. Thermal Control	7.5 kg	10.0	0.7	8.2	3.97
3. Mechanisms	9.9 kg	10.0	1.0	10.9	5.26
4. Pyrotechnics	0.0 kg	0.0	0.0	0.0	0.00
5. Communications	18.7 kg	10.0	1.9	20.6	9.91
6. Data Handling	9.5 kg	5.0	0.5	10.0	4.81
7. AOCS	9.0 kg	10.0	0.9	9.9	4.75
8. Propulsion	4.6 kg	5.0	0.2	4.8	2.34
9. Power	16.3 kg	10.0	1.6	18.0	8.65
10. Harness	5.0 kg	0.5	0.0	5.0	2.42
11. Payload Allocation	12.2 kg	15.0	1.8	14.0	6.73
Total Dry (excl.adapter)	148.39 kg			168.2	81.04
System Margin (excl.adapter)		20.0 %		33.6	
Total Dry with Margin (excl.adapter)				201.9	97.25
		Propellant:	Total propellant	5.7	2.75
				0.0	
			Adapter Mass (incl. Sep. Mech.)	0.0	0.00
Total Launch Mass				208	

Table 6-7: SWM S/C Baseline Mass Budget

In case a dedicated single launch with PSLV is selected, 4 kg of adapter mass between SWM and SAM should be subtracted.

6.4.5.2 Power Budget

Five operational modes have been considered as dimensioning for the design of the power subsystem. The corresponding S/C power demands are given in the table below.

		Instr.	Thermal	AOCS	Comms	Propulsion	OBDH	Power Cons.	Pyro	Mech	Harness (excl. PSS)	TOTAL CONSUMPTION
Launch Mode	MAX	0	0	0	30	0	14	20	0	0	0.9	65
	NOM	0	0	0	30	0	13	17	0	0	0.9	61
	MIN	0	0	0	30	0	13	14	0	0	0.9	57
Transfer mode	MAX	0	15	2	70	0	14	20	0	0	2.0	123
	NOM	0	15	0	70	0	13	17	0	0	2.0	117
	MIN	0	0	0	70	0	13	14	0	0	1.7	98
Initialisation Mode	MAX	20	15	2	70	0	14	20	0	0	2.4	143
	NOM	20	15	0	70	0	13	17	0	0	2.4	137
	MIN	0	0	0	70	0	13	14	0	0	1.7	98
Operational Mode	MAX	20	15	2	70	0	14	20	0	0	2.4	143
	NOM	20	15	0	70	0	13	17	0	0	2.4	137
	MIN	0	0	0	70	0	13	14	0	0	1.7	98
Safe Mode	MAX	0	15	2	70	0	14	20	0	0	2.0	123
	NOM	0	15	0	70	0	13	17	0	0	2.0	117
	MIN	0	0	0	70	0	13	14	0	0	1.7	98

Table 6-8: SWM S/C Power Consumption Budget

6.4.6 Options

The Rockot launcher delivers both the SWM with the STAR37FM engine attached to it into a 200 km parking orbit. The total mass performance of Rockot to a 200 km parking orbit is 1840 kg. Subtracting the wet mass of STAR37FM and the adaptor between the Rockot/Breeze upperstage and the STAR37FM system leads to a mass of 320 kg available to SWM [RD7].

Since the equations used for this calculation were based on analytical formulas instead of simulation, and given the fact that different reference sources have indicated slightly different results, a conservative value of 300 kg has been chosen as the maximum launch mass for SWM in this option.

Figure 6-13 shows a sketch of the STAR37F solid rocket motor. Figure 6-14 shows the option 1 SWM configuration inside the Rockot fairing.

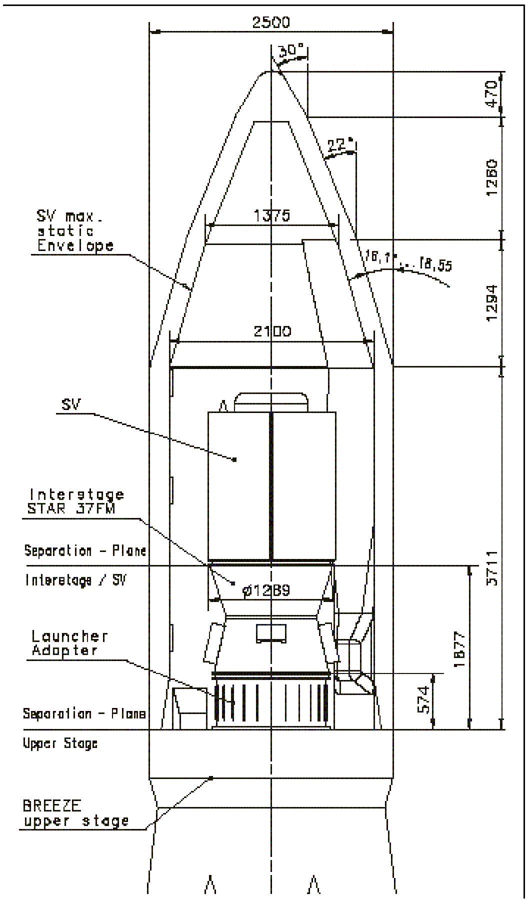


Figure 6-13: STAR37F SRM

Figure 6-14: SWM Inside Rockot Fairing

At spacecraft design level, major modifications with respect to the baseline would include:

- Antenna relocation and interfacing as a consequence of the different launcher and no longer attaching SWM to SAM.
- Reinforcement of the structure.
- AOCS re-design (faster spin-up, inclusion of nutation dampers, and more propellant for dispersion compensation of the solid engine)

These modifications, from the baseline S/C mass of 211 kg, would not exceed the target of 300 kg.

6.4.7 Conclusions and Open Points

The proposed SWM design is a minisatellite in L_1 halo orbit. This design fulfils all the requirements with a simple and reliable architecture. The design can be adapted to dfferent options such as single launch with Rockot, and SAM in GEO. The present baseline of dual launch of SAM and SWM with Soyuz-Fregat still presents a very large mass margin and can allow for additional payload.

6.5 Configuration

6.5.1 Requirements and Constraints

The major drivers for the overall configuration can be summarised as follows :

- A spinning satellite with solar cells mounted perpendicular to the spin axis
- Accommodation of the payload instruments TPM, MEM according to their pointing direction and field of view requirement
- Accommodation of a propulsion tank of 0.53m diameter
- Accommodation of the booms for antenna and payload instruments
- Accommodation of electronic boxes for Data Handling, Power, Communication
- Area needed for solar cells
- Stable mounting and accessibility to be guaranteed
- Compatibility with Soyuz-Fregat fairing type S payload envelope to accommodate a stack of SWM together with SAM

The spacecraft must provide accommodation to all the sub-systems and ensure compatibility between them throughout the mission. Therefore each of the constraints as listed above must be fulfilled for every operational mode and Sun-Earth S/C attitude.

6.5.2 Spacecraft Baseline Design

The configuration is driven by the system requirements together with the size of propellant tank and the solar cells area.

The resulting overall dimensions of SWM are:

- 1 m height (z-direction, from separation line to tip of LGA)
- 1.6 m in x-direction (from tip of MGA to tip of AST)
- 1.6 m in y-direction (from tip of Radio spectrograph to Magnetometer sensors.

Figure 6-15 and Figure 6-16 show stowed and deployed configurations respectively of the SWM spacecraft.

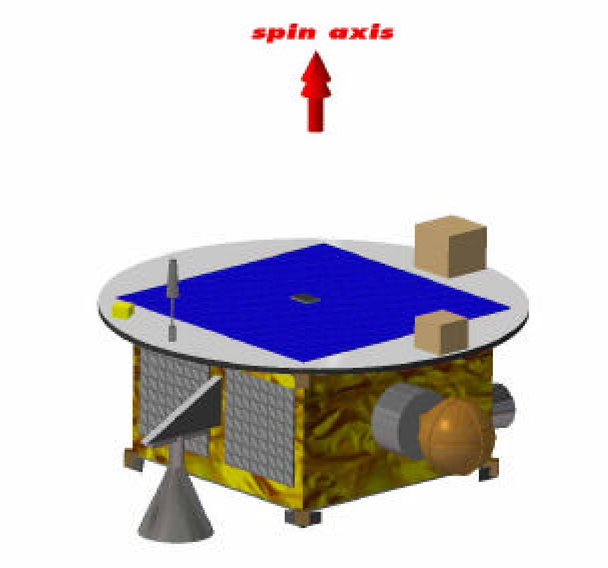


Figure 6-15: SWM Stowed Configuration

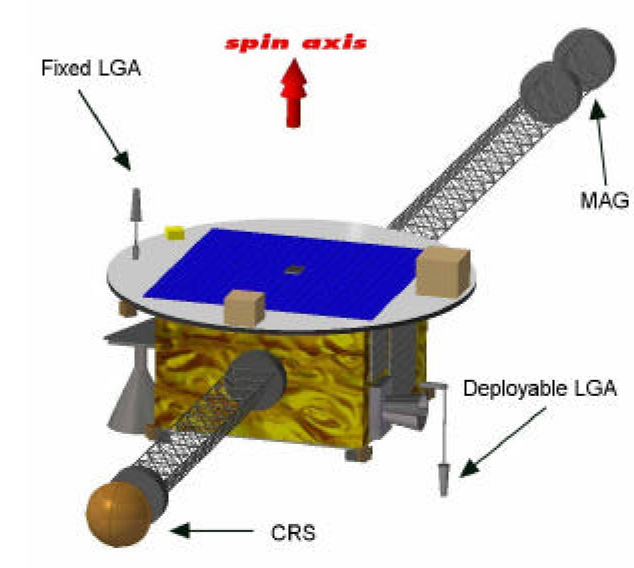


Figure 6-16: SWM Deployed Configuration

Figure 6-17 gives an exploded view of the S/C, showing the internal and external accommodation of the subsystems and units.

The S/C body is stiffened by a solid bottom plate (1 m by 1m) and 8 struts to hold other panels (0.6 m height). The solar cells are mounted on a support plate with diameter 1.9m, as seen in Figure 6-15 and Figure 6-16.

The propulsion system consists of one monopropellant tank and 8 thrusters. The propellant tank is centrally mounted on the bottom platform. The thrusters are located on the bottom and top platforms of the spacecraft.

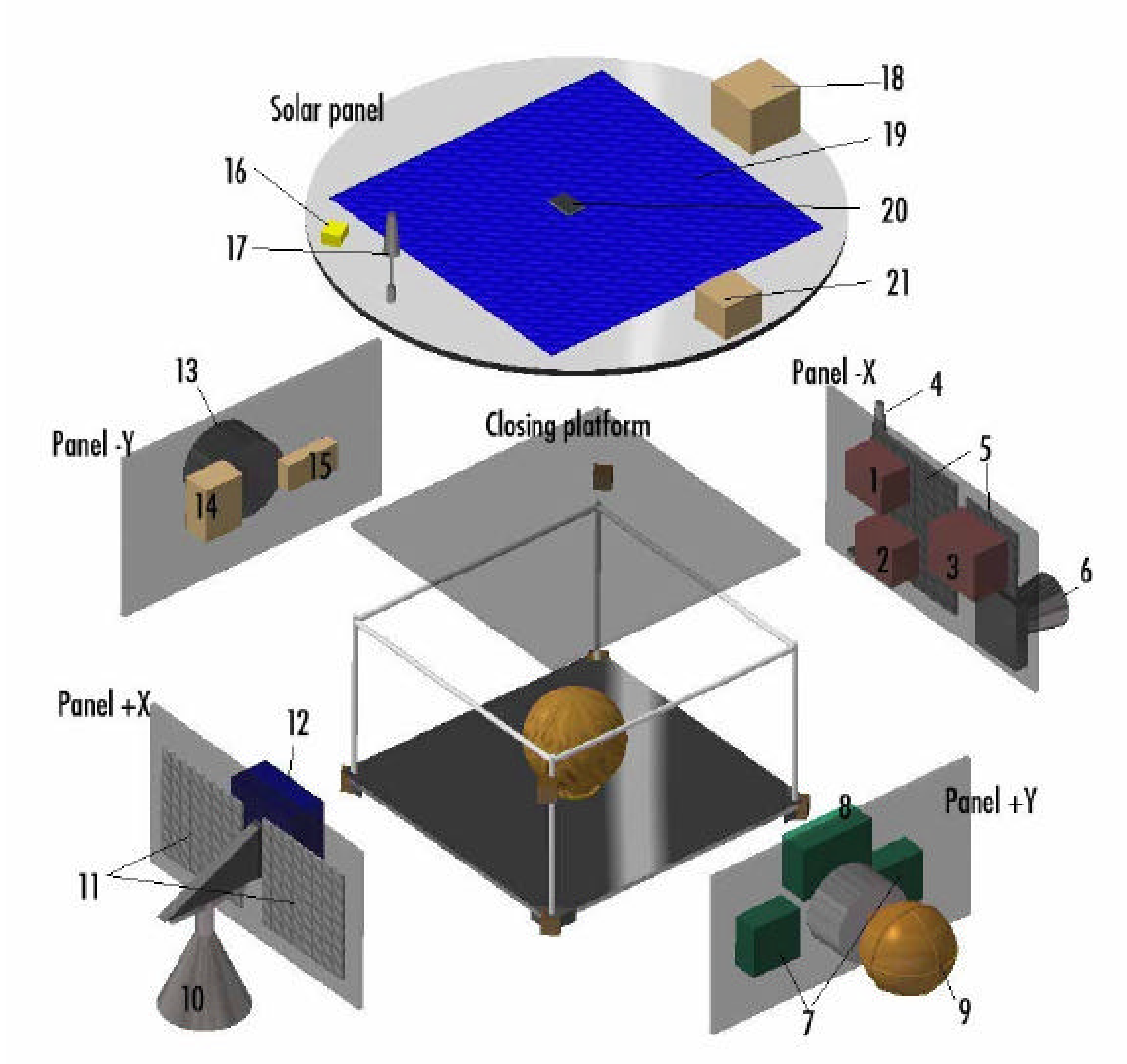


Figure 6-17: SWM Unit Accommodation

The accommodation is summarised in Table 5-10.

Domain	No.	Unit	Location
AOCS	6	Star Mapper	Outer part of Panel -X
	20	Sun Acquisition Sensor	Top platform
	16	Accelerometer	Top platform
Comms	8	RFDU	Inner part of Panel +Y
	7	2 transponders	Inner part of Panel +Y
	17	Fixed Low Gain Antenna	Top platform
	10	Fixed Medium Gain Antenna	Outer part of Panel +X
	4	Deployable Low Gain Antenna	Outer part of Panel -X

Domain	No.	Unit	Location
DHS	12	CDMU	Inner part of Panel +X
Power	1	Battery	Inner part of Panel -X
	2	PDU	Inner part of Panel -X
	3	PCU	Inner part of Panel -X
	19	Solar cells	Top platform
Thermal	5, 11	Radiator	Outer part of Panel -X/ +X
Instruments	18	TPM	Top platform
	21	MEM	Top platform
	13	Magnetometer sensors	On a coilable boom mounted on Panel -Y
	9	Radio spectrograph (CRS)	On a coilable boom mounted on Panel +Y
	14	Electronic boxes for magnetometer (MAG)	Inner part of Panel -Y
	15	Electronic boxes radio spectrograph (CRS)	Inner part of Panel -Y

Table 6-9: Unit Accommodation

The payload accommodation is illustrated below.

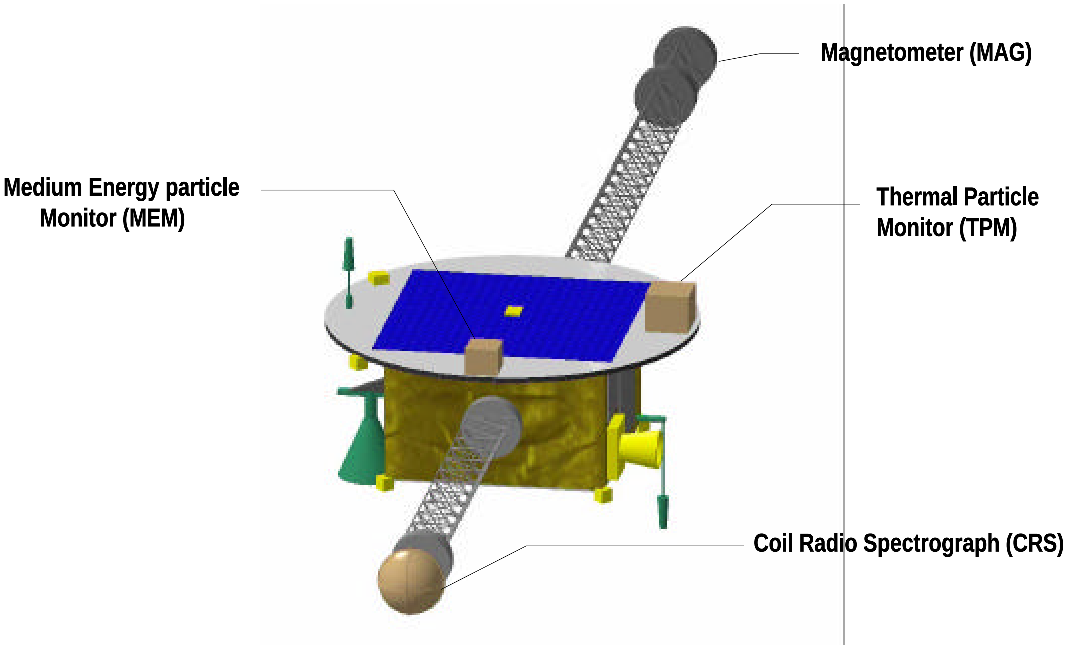


Figure 6-18: SWM Payload Accommodation

6.5.3 Launch Configuration

The following figure shows the SWM spacecraft stacked on top of SAM. The stack inside the fairing is illustrated in Figure 7-14.

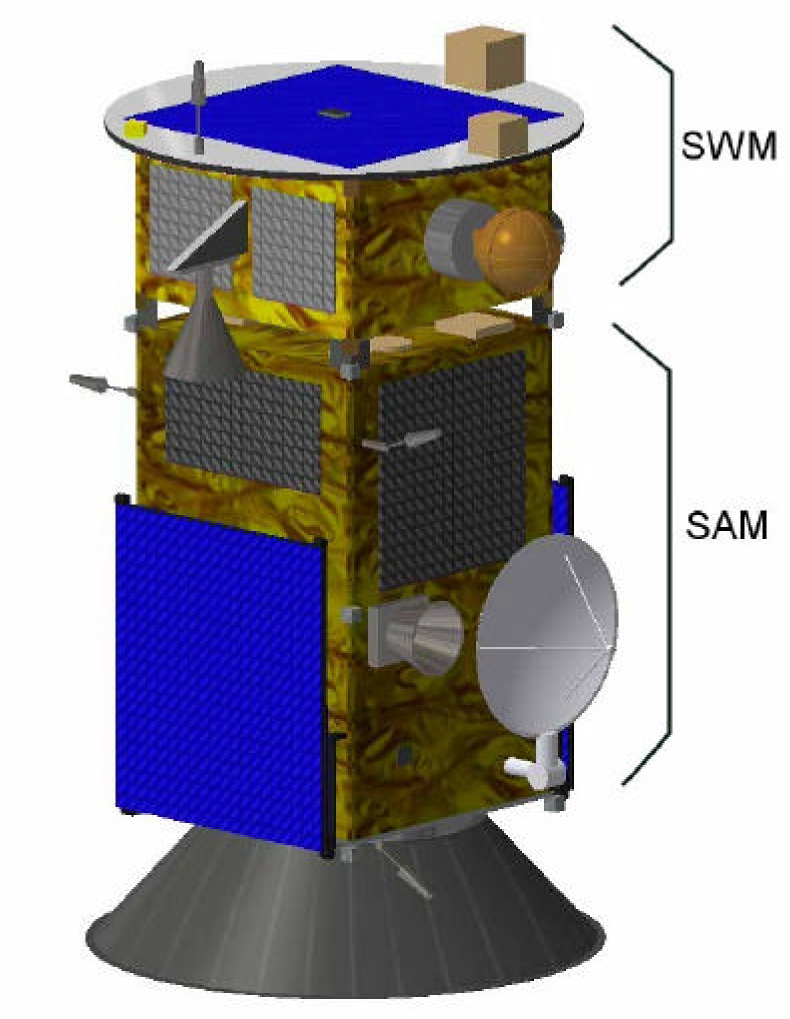


Figure 6-19: SWM Stacked on top of SAM

6.6 Propulsion

6.6.1 Requirements and Design Drivers

The Soyuz-Fregat launcher is able to put the SWM spacecraft and a second spacecraft directly into the L_1 orbit. The launcher will provide the initial spin-rate. Small adjustments to the spin rate will be required for spacecraft attitude management.

The direct injection to the L_1 orbit by the launcher leaves only a very small DV requirement on the spacecraft. The propulsion system remains necessary, though, for this launcher dispersion correction, plus spin control (664 Nms) and halo-orbit maintenance.

A 40 m/s velocity increment is required for correction of launcher dispersion (section 6.2.2.2), plus another 5 m/s for mid-course manoeuvres and possible halo insertion adjustment. Another 10 m/s velocity increment is required for orbit maintenance (section 6.2.3.2).

6.6.2 Baseline Design

A simple mono-propellant hydrazine system has been selected, with eight thrusters positioned and angled to provide the necessary thrust and rotation. The selected thruster system is able to accomplish orbit manoeuvres along any direction at any time during transfer (launcher dispersion corrections and mid-course manoeuvres) and on operational orbit (orbit maintenance).

The system comprises a propellant tank (D= 0.254 m) from which the propellant is expelled by centrifugal force (first motion initiated by the launcher). The system operates in blow-down mode with Helium as pressurant gas.

Furthermore the system comprises some thermistors and line heaters, two fill and drain valves, a propellant filter, a latching valve, a propellant isolation valve, a pressure transducer and eight 1 N thrusters.

Two barriers between propellant tank and thrusters are used, assuming that this complies with the Soyuz-Fregat launch vehicle requirements. If three barriers are required, the use of dual-valve thrusters could provide a solution, at the cost of some extra mass. Use of dual-valve thrusters also increases the reliability of the system in the event of leakage in one thruster.

The proposed propulsion system design is, to a large extent, based on available COTS components and not on optimised components that may need to be developed. Substantially larger costs would have to be taken into account if a specific tank development were required. From a purely engineering perspective, the proposed design solution is likely not the optimum one, at least not from a dry mass point of view. The mass penalty for the non-optimised design was not traded-off at this stage against the potential cost advantage of using COTS tanks. For the time being it is assumed that the cost advantage of the proposed design outweighs the mass penalty.

During the last burn of the launch vehicle and the spin manoeuvre of the launch vehicle, to expel the propellant in the spacecraft's tank to the drain hole, pressurant gas could become trapped in the area of the propellant filter. Therefore the position of the filter could be a point of debate.

Figure 6-20 depicts the schematic diagram of the propulsion system.

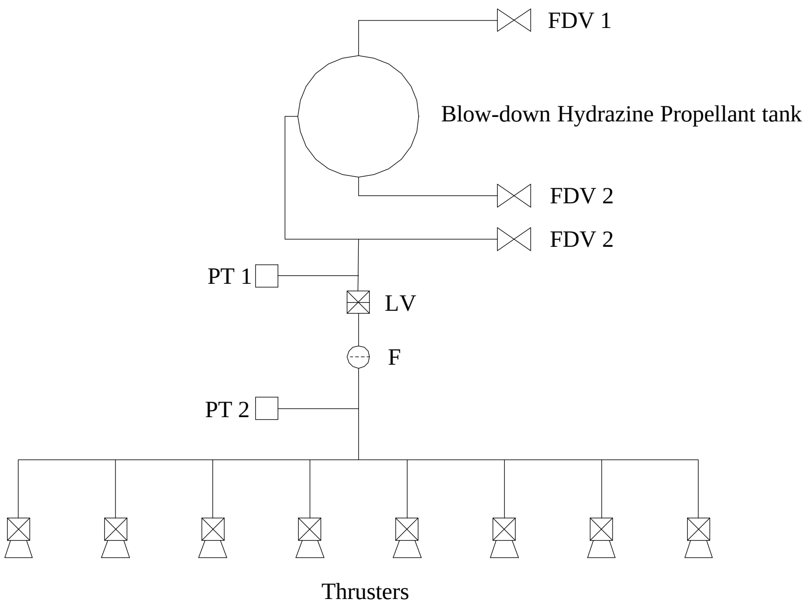


Figure 6-20: SWM Propulsion Subsystem Block Diagram

6.6.3 Budgets

Table 6-10 shows the mass breakdown of the components of the propulsion system.

Component	Number	Component mass (kg)	Mass (kg)	Potential Supplier
Propellant tank	1 x	0.85	0.85	ARDE
Pipework			0.55	
Thrusters	8 x	0.2	1.6	MBB-ERNO CHT 2.0
Fill/drain valves	2 x	0.2	0.6	Polyflex
Latch valve	1 x	0.18	0.18	ERNO
Filter	1 x	0.28	0.28	Vacco
Pressure transducer	2 x	0.25	0.5	SEP
Pressurant Helium			0.015	
Propulsion system total dry mass			4.58	
Propulsion system dry mass with 5% margin			4.80	
Propellant incl. 1% residuals			5.7	
Propulsion system wet mass			10.5	

Table 6-10: Propulsion sub-system mass budget

6.7 Thermal Control

6.7.1 Requirements

The spacecraft thermal control subsystem keeps the temperatures of the spacecraft subsystems and the instruments within specified temperature limits during all expected mission phases and operation modes. The temperature limits have been assumed as follows:

	Operational	Non-operational
Instruments	-10°C/+20°C	-20°C/+55°C
S/C electronics	-20°C/+40°C	-30°C/+60°C
Batteries	+20°C/+35°C	0°C/+40°C
Propulsion system	+5°C/+15°C	+5°C/+15°C

Table 6-11: Assumed Temperature Limits

The instrument operational temperature limit is conservative, and would probably be relaxed following more detailed study.

6.7.2 Assumptions

The solar fluxes during the operation in the L_1 halo orbit are almost constant at 1400W/m^2. Earth and albedo fluxes in the operational orbit are negligible.

6.7.3 Baseline Design

The Thermal Design philosophy used for the SWM is based on the use of passive techniques (MLI, OSR, etc.), with the addition of heater power for the propulsion system in order to keep the hydrazine propellant above the required minimum temperature.

The features of the Thermal Design (see Figure 6-21) can be summarised as follows:

- Multi-Layer Insulation (MLI) Blankets and double foil trimmed as necessary to have a better heat rejection to deep space and therefore to minimise heat absorption from solar irradiation. The blankets comprise Aluminised Mylar and/or Kapton sheets and an electrically conductive outer sheet or laminate grounded to the S/C structure in order to prevent electrostatic potential difference.
- The Solar Array plate is thermally insulated from the S/C structure to minimise heat input into the S/C structure.
- Radiator surfaces located on the side surfaces of the SWM main body are covered in black paint in order to radiate the S/C internal heat dissipation to deep space. The total required radiator surface is 0.45 m^2. The radiators are provided by cut-outs in the MLI. Subsystems with a higher power request shall preferably be mounted in contact with the radiators.
- S/C internal surfaces have a high emittance finish to increase radiative heat transfer and to minimise the temperature gradient within the S/C. Therefore all internal aluminium surfaces are black painted.

- To maintain low temperatures on the propulsion S/S (tanks and valves) and the batteries, they are thermally insulated from the S/C internal environment with MLI.
- The required minimum temperatures of the propulsion S/S and the batteries are maintained by heaters providing a heater power of 15W. Heater control is performed by thermostats at element level.

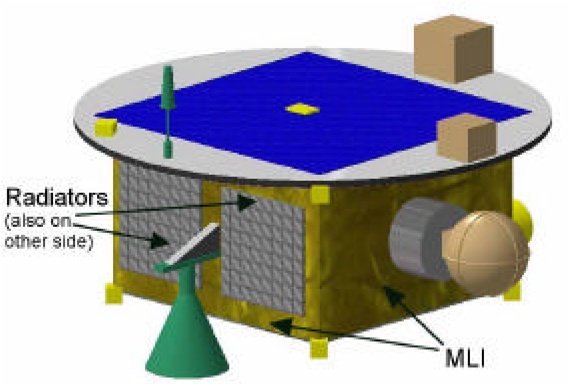


Figure 6-21: Location of Thermal Subsystem Elements

6.7.4 Budgets

6.7.4.1 Mass Budget

The preliminary mass budget for the SWM thermal control subsystem is provided in Table 6-12.

Item	Estimated Mass (kg)	Uncertainty 10% (kg)	Total Item Mass (kg)
MLI/Foil	2.8	0.28	3.1
OSR	0.7	0.07	0.8
Heaters/Thermostat/other	4.0	0.40	4.4
TOTAL	7.5	0.75	8.3

Table 6-12: Thermal Control Mass Budget

6.7.4.2 Heater Power Budget

Table 6-13 gives a summary of the preliminary power budget in different modes.

Mode	Heater Power	Comment
Launch mode	0 W	
Transfer Mode	15 W	For propulsion S/S and batteries
Initialisation Mode	15 W	For propulsion S/S and batteries
Operational Mode	15 W	For propulsion S/S and batteries
Safe Mode	15 W	For propulsion S/S and batteries

Table 6-13: Thermal Control Power Budget

6.8 Power

6.8.1 Requirements and Design Drivers

Design drivers for the Power Subsystem definition are the following:

- Spinning S/C with flat solar array attached to S/C structure with a maximum available projected area of 1.4 m^2
- 10 degrees maximum SA pointing deviation with respect to normal throughout the mission
- Sunlight average power: 140W
- Solar Array Temperature up to about 130 °C
- L_1 orbit means that battery is only used in launch and transfer mode or in case of contingency
- Cluster type EMC cleanliness

6.8.2 Assumptions and Trade-Offs

With the above mentioned requirements, the maximum End of Life (EOL) Solar Array (SA) power need has been derived at 140W. Table 6-14 below provides the estimated area and mass required to produce that EOL power for different cell types:

SA Technology	Estimated SA Area	PVA Mass
Si	3.1 m^2	7.6 kg
Si Hi-Eta	2.2 m^2	5.4 kg
GaAs	1.3 m^2	4.2 kg

Table 6-14: Comparison of Solar Array Types

Photovoltaic Assembly (PVA) shall be understood as the Solar Array without the Panel Structure.As it may be seen, since the maximum allocated area for the Solar Array is 1.4 m^2, GaAs cells are the only ones which can provide the required power within the available area, and these have therefore been baselined. Given the estimated area and the maximum allocated area, an extra 10% margin or power growth capability may be considered without significant S/C configuration changes.

6.8.3 Baseline Design

6.8.3.1 Power Bus

A 28V fully-regulated power bus is provided to the different Main Bus users through protected power lines, as shown in the block diagram.

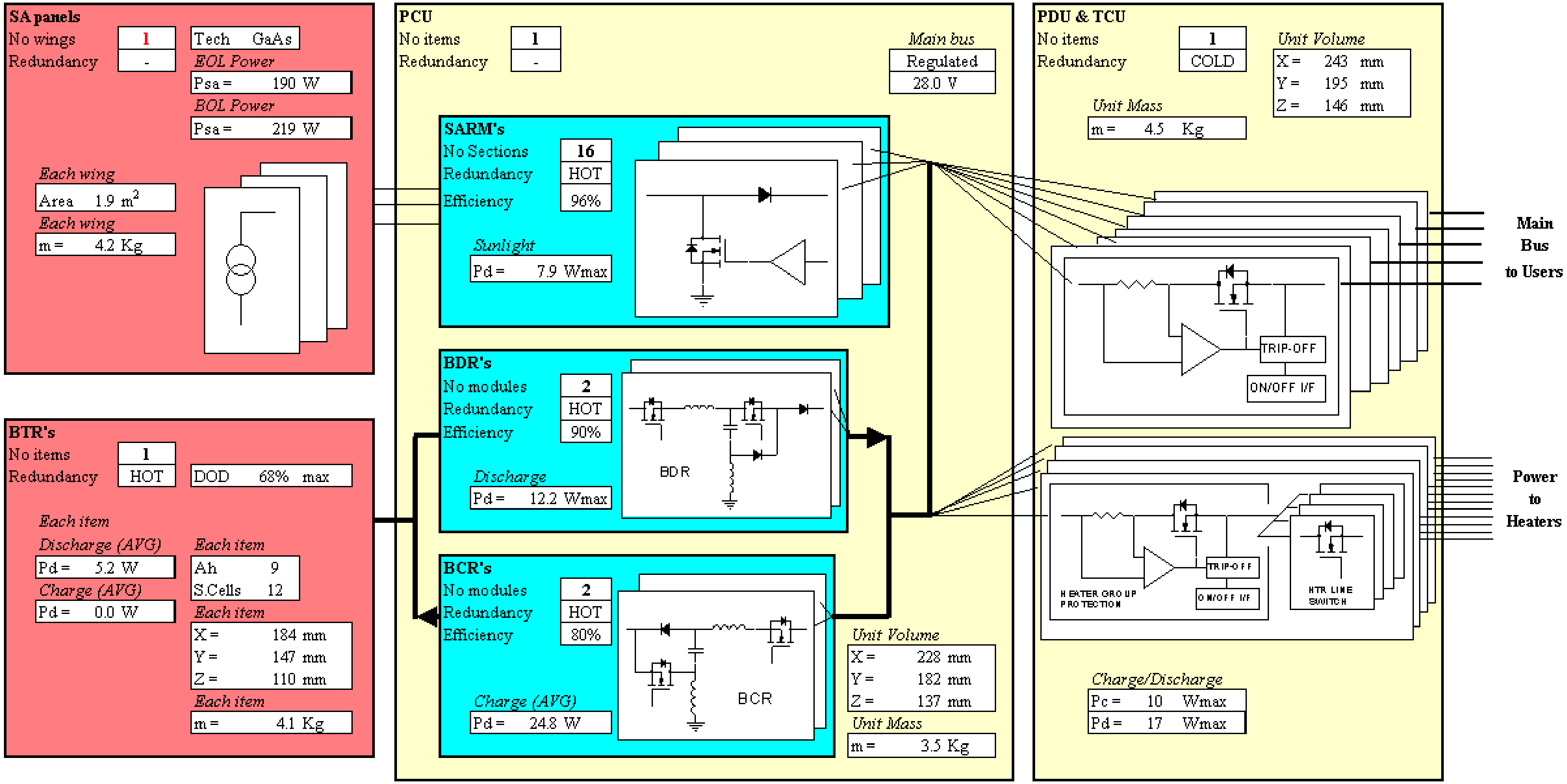


Figure 6-22: Power Subsystem Block Diagram

Two electronic boxes, one Power Conditioning Unit (PCU) and one Power Distribution Unit (PDU) are foreseen for proper power bus regulation and distribution. The PCU consists of two 200W Battery Discharge Regulators, two 200W Battery Charge Regulators, 16 Solar Array Regulator (SAR) modules, and one 2/3 Majority Voter Error Amplifier generating reliable regulator control signals. The PDU consists of Latching Current Limiters for power bus protection, transistor switches for thermal control and pyrotechnic drives, as required.

The BDRs, BCRs and SAR sections operate in hot redundancy, so that PCU is one failure tolerant with no reconfiguration needs. The solar array has been sized so that the loss of one section still satisfies the Mission requirements at End of Life. PDU failure tolerance relies on the usual cold redundant approach.

6.8.3.2 Battery

The battery is nominally used only in Launch mode and Initialisation (before sun pointing acquisition). A battery energy allocation 400Wh has been considered to power budget needs with a maximum battery depth of discharge (DOD) below 75%. At L_1, the battery will only be used in case of attitude control loss.

Stringent EMC cleanliness requirements (Cluster type) shall be expected for this type of mission. Although AgCd has been used in the past, difficulties in manipulation and loss of battery energy with lifetime have been drawbacks related to the use of these type of batteries.

Currently, Li-Ion technology is becoming the leading technology for space applications. This technology happens to be not specially bad in terms of EMC cleanliness. Since a TRP activity is currently ongoing on Li-Ion technology for Cluster type missions, for its potential use in BepiColombo MMO, this is the proposed concept for SWM.

As is usually the case with Li-Ion batteries, battery redundancy is foreseen at cell level (not at Unit level).

6.8.4 Budgets

The overall Mission power consumption budget is given in section 6.4.5.2. The power dissipation is given below.

PSS & SPACECRAFT DISSIPATION vs MODE

Mode		PCU	PDU	TCU	BATTERY	PSS Harness	PSS TOTAL DISSIPATION	S/C TOTAL DISSIPATION
Launch Mode	MAX	26	12	5	3.4	1.8	48	93
	NOM	24	10	4	3.2	1.7	43	87
	MIN	21	9	4	3.0	1.6	38	82
Transfer mode	MAX	25	15	5	0.0	1.9	47	120
	NOM	23	13	5	0.0	1.8	43	113
	MIN	19	11	4	0.0	1.5	35	90
Initialisation Mode	MAX	28	16	6	0.0	2.2	52	144
	NOM	25	15	5	0.0	2.1	47	137
	MIN	19	11	4	0.0	1.5	35	90
Operational Mode	MAX	28	16	6	0.0	2.2	52	144
	NOM	25	15	5	0.0	2.1	47	137
	MIN	19	11	4	0.0	1.5	35	90
Safe Mode	MAX	41	17	6	6.4	3.5	73	146
	NOM	38	15	5	6.1	3.3	67	137
	MIN	31	13	4	5.1	2.8	56	111

Table 6-15: SWM Power Dissipation

The Power subsystem mass breakdown is given in Table 6-16 below:

S/S Item	Mass
Li-Ion Battery Mass	4.1 kg
GaAs Solar Array	4.2 kg
Electronics (PCU/PDU/TCU)	8.0 kg
PS/S Total	16.3 kg

Table 6-16: SWM PSS Mass Budget Breakdown

6.9 Mechanisms

The identified mechanisms for the Space Weather SWM satellite are:

- 1 low gain antenna deployment boom
- 1 boom for the magnetometer instrument (MAG)
- 1 boom for the radio spectrograph instrument (CRS)

Note that the mechanism for separating SWM from SAM is covered under the SAM chapter (section 7.9.2.3).

6.9.1 Requirements, Trade-offs and Design Drivers

6.9.1.1 Low Gain Antenna Deployment Mechanism

The required length of the antenna deployment boom is mainly dependent on the radiated shape of the low gain antenna and its position on the spacecraft. For SWM, the antenna will be used during flight while attached to SAM and after separation. In these two configurations, the antenna shall be located in different positions.

- During flight, attached to SAM satellite:
  In this configuration, the antenna will be positioned perpendicularly to the main satellite axis, with a clear field of view from both spacecraft (see Figure 6-19).
- After separation from SAM satellite:
  In this configuration, the antenna will be positioned parallel to the spin axis in the direction of Earth (see Figure 6-16).

In practice, the boom needs to be stored along the spacecraft during launch, then to be deployed a first time (90°) for telecommunication transmission during the flight to L_1 while attached to SAM, and finally to be deployed 90° further for its final configuration once detached from SAM. Therefore the design would be optimised by using a mechanism with a deployment capability of two 90° steps.

The foreseen boom length is 0.5 m long.

Therefore, the foreseen mechanisms are:

- 1 deployment mechanism with specific hinges to be able to deploy in two steps
- 1 hold-down and release mechanism

6.9.1.2 Magnetometer Boom Deployment Mechanism

The required radial distance of the two magnetometer sensors (MAG instrument) from the spacecraft has been given as 3.5m and 4m respectively. The 4m boom shall be compacted on the spacecraft during launch. Due to the small available volume on the side of the spacecraft (the top of the spacecraft is dedicated to the solar array) the boom should preferably be a coilable boom.

The same coilable boom can be used to deploy the two magnetometer sensors simultaneously (at 3.5m and 4m from the spacecraft).

The spin of the satellite shall be decreased for deployment. The boom is deployed under its own stored strain energy. To regulate the deployment speed and decrease the shock at the end of the deployment, a regulator is integrated into the coilable boom deployment mechanism.

Therefore, the foreseen mechanisms are

- 1 complete coilable boom mechanism (with integrated speed regulator and hold-down devices)

6.9.1.3 Radiospectrograph Boom Deployment Mechanism

The radio spectrograph antennae (CRS instrument) is required to be distant from the spacecraft body by at least 1.5 m. To optimise the dynamic balance of the spacecraft with respect to the magnetometer boom of 4 m, the radio spectrograph boom shall be 1.8m long. The 1.8m boom shall be compactly stowed on the spacecraft during launch. Due to the available volume on the side of the spacecraft (and because the top of the spacecraft is dedicated to the solar array) the boom will preferably be a coilable boom.

The spin of the satellite shall be decreased for deployment (to perhaps 4 or 5 rpm – to be determined at a later stage). The boom is deployed under its own stored strain energy. To regulate the deployment speed and decrease the shock at the end of the deployment, a regulator is integrated to the coilable boom deployment mechanism.

Deployment of the two coilable booms (for MAG & CRS) may be done simultaneously.

Therefore, the foreseen mechanisms are

- 1 complete coilable boom mechanism (with integrated speed regulator and hold-down devices)

6.9.2 Assumptions, and Baseline Design

The approach which has been followed to identify the conceptual design of Space Weather SWM satellite mechanisms has been to use qualified, off-the-shelf equipment as far as possible, in order to reduce cost, procurement time, and development risks.

In the following paragraphs a short description of the foreseen mechanisms is provided, including a preliminary estimate of mass budgets.

6.9.2.1 Low Gain Antenna Deployment Mechanism

Boom

One short boom (around 0.5 m long), carrying the low gain communication antenna is foreseen to be deployed 180° in two steps of 90° each. The boom length is sized to provide good antenna clearance in the first 90° position during the attached phase with SAM.

Deployment Mechanisms

A specific spring-based system is foreseen at the boom hinge level, to actuate the rotation. The first rotation is stopped at 90° thanks to a pin puller end stop. Activation of the pin puller (the first end stop is retracted) will allow the boom to deploy to 180°. A specific regulator based on low melting temperature alloy will be used to avoid any shock at each of the two stop positions. Thus no damping system or latching device is required.

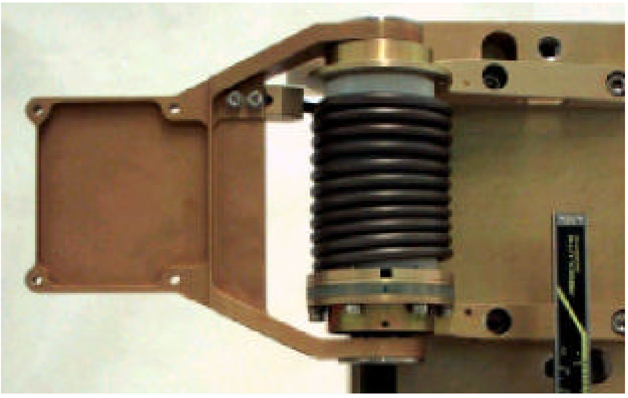


Figure 6-23: Speed-Regulated Deployment Hinge

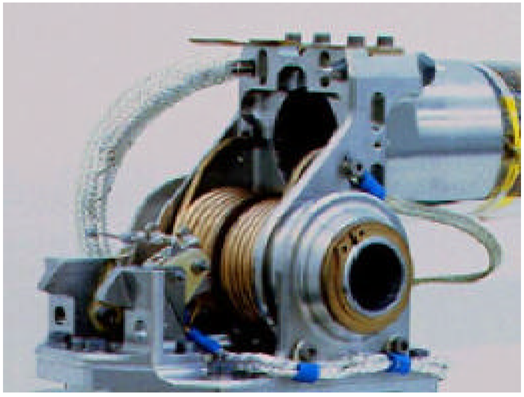


Figure 6-24: Standard Deployment Hinge

Hold-down and Release Mechanism

One single hold-down and release point is foreseen for the low gain antenna boom, in order to provide adequate stiffness and strength in the stowed configuration. A pyro-actuated device (separation-nut-like) can be used to actuate the separation operation (see section 6.10.1.1).

RF Junction

The RF junction (at deployment level) could be a flexible coaxial.

6.9.2.2 MAG & CRS Boom Deployment Mechanisms

These two booms will be of the same type. The baseline solution is based on coilable boom types. Nevertheless, this choice shall be confirmed against the, as yet unknown, boom-bending requirement during and after deployment.

As a back-up solution, a coilable boom with deployer or a collapsible tubular type of boom can be proposed, but this has an unfavourable mass impact.

The description hereafter corresponds to a coilable boom manufactured by AEC-able, but is still valid for European coilable boom solutions.

Coilable Boom

The continuous-longeron coilable boom is used for applications that require high dimensional stability and/or a high ratio of bending stiffness to weight.

The main structural elements of a coilable boom are: longerons, diagonals and battens. The longerons are continuous over the boom length and are connected to the batten frames by pivot fittings. Six relatively inextensible diagonals provide shear strength and stiffness to each bay. For this application, the longerons, diagonals and battens are all made out of unidirectional

S-glass/epoxy. The unidirectional S-glass/epoxy is used in the longeron and battens due to its high strain capability, dielectric properties, thermal stability, and strength properties.

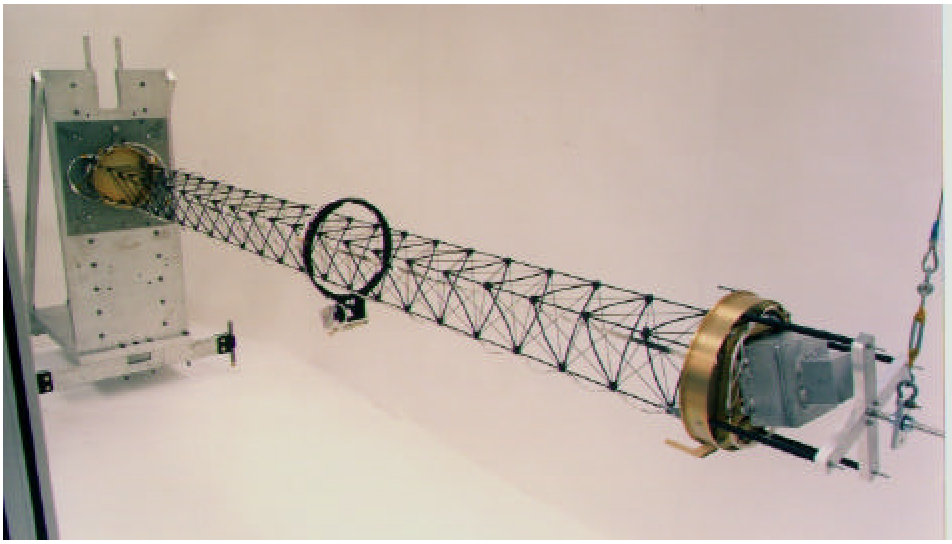


Figure 6-25: Coilable Boom Example

Hold-down and Release Mechanism

The lanyard-deployed boom is restrained during launch by a set of three pins that are spaced equally around the outer rim of the outboard end of the base canister. The pins, which are spring loaded outward, are held inward radially by a cable that is secured by a High Output Paraffin (HOP) actuator until the release command is given.

Regulator

The regulation is achived by a rotary friction damper. This centrifugal rate limiter restricts the rate of deployment of the boom. This rate limiter involves a rotating set of weights that are spun up by the lanyard that is wrapped around a reel. When spinning, pad made out of Poly-Ether-Ether-Ketone (PEEK) that are attached to the weights press outward radially against the inside of an aluminum drum. As the tension of the lanyard increases, the reel will rotate faster which causes more centrifugal force on the weights and thus more damping force. The benefits of this damper are its temperature- and vacuum-insensitivity.

6.9.3 Budgets

The estimated mass and power budgets are shown in Table 5-18 below. The mass figures do not include pyros.

Mechanisms type	Qty	Electrical Power	No. of Pyros	No. pin pullers	Unit Mass excl. Margin	Deployment Time
Low gain antenna deployable boom with associated hold-down and release mechanisms	1	10 to 15W	1	1	2.5 kg	First < 2 min Second < 2 min

Mechanisms type	Qty	Electrical Power	No. of Pyros	No. pin pullers	Unit Mass excl. Margin	Deployment Time
MAG instrument boom	1	10 W requested for deployment initialisation	0	1	4.1 kg	<5 min
CRS instrument boom	1	10 W requested for deployment initialisation	0	1	3.33 kg	< 2 min
Total		30 to 35W	1	3	9.93 kg	

Table 6-17: Mechanisms Resource Budgets

6.9.4 Options

The baseline for the magnetometer (MAG) and radio spectrograph (CRS) instruments' booms is a TERMA future design of coilable boom. In case the TERMA boom does not reach the required specifications for such a mission, an AEC-Able design would still be a good backup solution.

The pyro hold-down point used for this mission could be changed to a non-pyro device. Some solutions based on Shape Memory Alloy, low melting temperature alloy, paraffin actuators, thermal knives, etc. are today qualified and provide good performances with significantly reduced shock. The main drawback of these solutions is that they cannot be fired with the same time accuracy as pyros.

6.10 Pyrotechnics

Pyrotechnic devices are highly suitable for the release of the Space Weather SWM satellite booms.

Neither AOCS nor the Propulsion system need pyrotechnic devices.

6.10.1 Requirements and Design Drivers

For all applications, cost and reliability considerations demand that qualified off-the-shelf devices are used. Known pyrotechnic devices have achieved reliability and qualification status unmatched by alternative technologies.

6.10.1.1 Low Gain Antenna

A single pyrotechnic release-nut will be used to restrain and then release the low gain antenna after launch.

6.10.1.2 AOCS and Propulsion

The simplicity of the spacecraft AOCS and Propulsion subsystems leads to no requirements for pyrotechnic valves or wheel-release devices.

6.10.2 Assumptions and Trade-Offs

Standard off-the-shelf devices reduce performance and procurement risk and allow for 5% mass-margin to be applied. The devices include redundant initiators with independent switching, command and supply, harness and electronics.

6.10.3 Budgets

The power demand per pyrotechnic device is of millisecond duration and thus negligible, particularly when fired before full spacecraft operation.

Unit masses of typical pyrotechnic actuators are in the region of 0.17 kg.

6.11 Attitude and Orbit Control (AOCS)

6.11.1 Main Requirements

For the baseline SWM design, the functions required of the Attitude and Orbit Control Subsystem (AOCS) are:

- To spin the spacecraft in a sun-pointing direction and maintain a 1°, 1σ relative pointing error
- To spin the spacecraft at 15 rpm
- To provide a 1°/40s pointing stability
- To determine the spacecraft attitude in inertial space with a pointing knowledge accuracy of 0.25.

It is assumed that the spacecraft orbit insertion is not performed with the assistance of a solid rocket motor, i.e. this discussion is for Soyuz-Fregat (the baseline).

6.11.2 Design Features

Given the mission pointing requirements (for payload, communications, and thermal subsystems) a spin-stabilised S/C design is most appropriate. The low pointing accuracy requirement (1°) and the continuous sun-pointing payload specification make a spinner attractive, as this option should at first reduce S/C complexity and cost.The S/C AOCS design consists mainly of units which are well characterised and with relatively low mass and power requirements.

Figure 6-26 below illustrates the general architecture of the avionics subsystem.

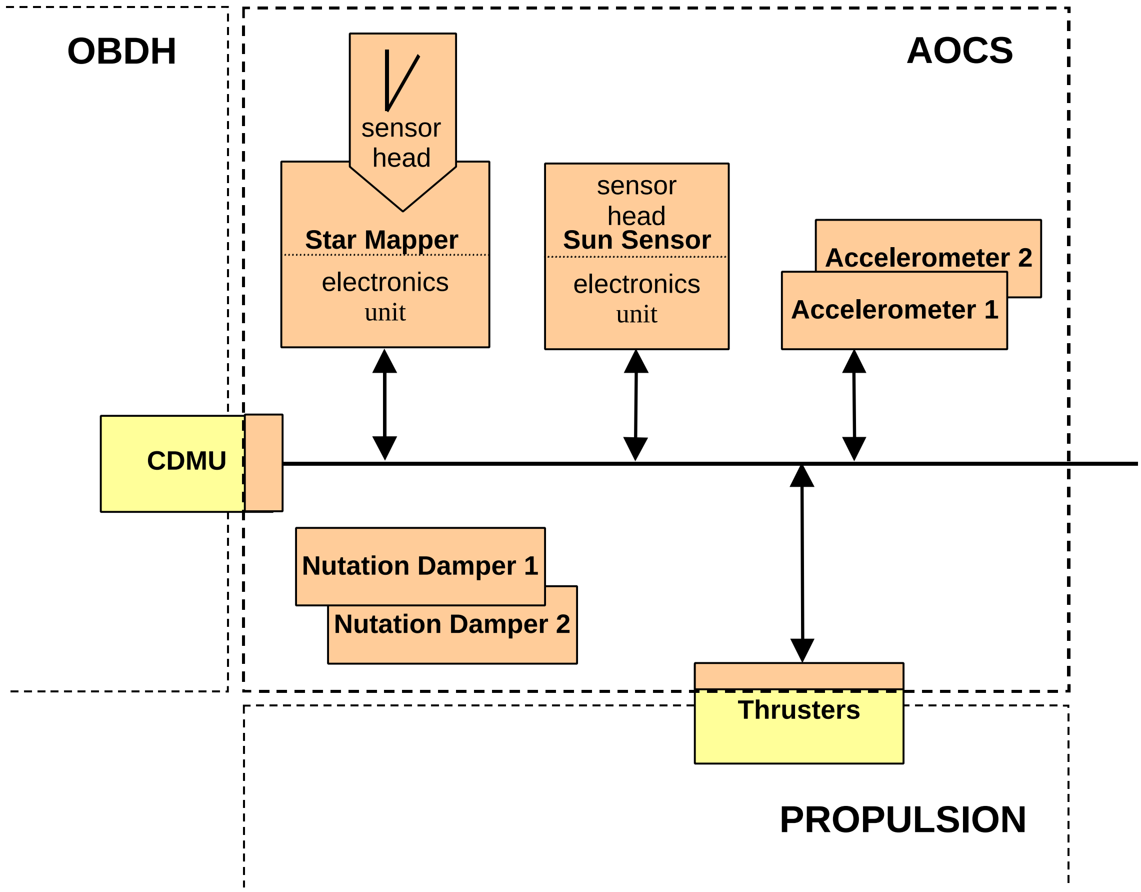


Figure 6-26: Avionics Subsystem General Architecture

6.11.3 Equipment Overview

6.11.3.1 Star Mapper

The star mapper works in a 'scanning' mode on spinning spacecraft. It is placed on a side of the spacecraft perpendicular to the S/C axis of rotation (at L_1 it should not be necessary to point the star mapper away from very bright objects). The line of sight of the optical unit may be canted with respect to the spacecraft spin axis, to ensure that the field of view sweeps over the celestial sphere. The sensors (redundant) consist essentially of a pair of V-shaped slits, a meridian slit (or a slit parallel to the S/C spin axis) and an oblique slit. The star mapper electronics unit provides S/C attitude information with respect to an inertial reference.

6.11.3.2 Sun Acquisition Sensor

A slit sun sensor, often used in spinning spacecraft, is not suitable for SWM as the sensor would be continuously in view of the sun. However, a sun acquisition sensor, more commonly used in 3-axis stabilised S/C, is a suitable sensor for the SWM spinner.

The sensor boresight should either coincide with, or be parallel to, the S/C spin axis. In order to limit the data processing requirements of the sensor, it is desirable to place it as close as possible to the S/C spin axis, i.e. in the centre of the solar panel side of the S/C (see Figure 6-17).

6.11.3.3 Accelerometers

In the baseline configuration, accelerometers are not an essential part of the design. The S/C moment of inertia around the spin axis (Iz) is greater than Ix and Iy, and hence passive nutation damping should be sufficient.

However, given the low cost of the accelerometers in terms of mass and power requirements, it is recommended that accelerometers be included in the design as they can provide useful extra information for the AOCS subsystem.

6.11.3.4 Nutation Dampers

These are tuned to the specific characteristics of SWM, both in terms of size and location in the S/C: they are of the equatorial type (as opposed to meridian) and are placed within the spacecraft in a plane perpendicular to the spin axis. The size of the nutation dampers will depend on the nutation frequencies of the S/C (i.e. the two dampers may be of different sizes). They are of very simple design and inherently very reliable, and therefore no redundancy is supplied.

6.11.4 Budgets

As shown in Table 6-18 and Table 6-19 below, the relatively simple avionics design implies very low power and mass requirements.

6.11.4.1 Unit Masses and Dimensions

unit	qty	unit mass (kg)	total mass (kg)	unit dimensions (mm)		
				Length	width	height
star mapper	1	4.10	4.10	181	151	151
digital sun sensor	1	0.81	0.81	140	120	120
Accelerometers	2	0.24	0.48	75	75	65
passive nutation dampers	2	0.91	1.82	400	60	90
		TOTAL	7.21			
		TOTAL with 10% margin	7.93			

Table 6-18: Mass and Dimensions of AOCS Avionics Units

6.11.4.2 Power

Unit	Qty	Unit power, max (W)
star mapper	1	0.7
digital sun sensor	1	0.1
Accelerometers	2	0.5
passive nutation dampers	2	0
	TOTAL	1.8
	TOTAL with 10% margin	2.0

Table 6-19: AOCS Avionics Units' Maximum Power

6.11.5 Options

If Option 1 was to be pursued (i.e. launch of SWM on Rockot with a STAR37 solid rocket motor attached), the following changes would be required:

Accelerometers would be an essential part of the design. With such a motor attached, the S/C cylinder becomes so tall that the moment of inertia around the spin axis (Iz) might not be greater than around Ix and Iy, and therefore active nutation damping would be required.

Active nutation damping would involve not only the use of accelerometers, but most likely also a different thruster system design. The mass/propellant budgets would also be considerably affected.

6.12 Data Handling

6.12.1 Requirements and Design Drivers

The SWM Data Handling System is in charge of supporting the flight software for the command and data management functions as well as for the attitude control and navigation functions.

The SWM satellite is characterised by continuous acquisition of data from four instruments at a rate of 6.7 kbps. The orbital location allows for continuous ground coverage and real time transmission to ground. However, the possibility of data storage for subsequent downlink is to be provided as far as possible. The Housekeeping data rate is assumed at 2 kbps.

Over a 5-year lifetime, the system will be exposed to a radiation environment of 5 krad with 4mm equivalent shielding.

The design drivers are simplicity and reuse of present Data Handling Systems.

6.12.2 Design Assumptions

The heritage comes from existing Data Handling Systems. In particular, the baseline is the adaptation of the DHS designed for the ESA PROBA mission.

The DHS of SWM is built around a single box and has a redundant architecture in order to be one permanent fault tolerant. Low requirements for DHS outage time are assumed so a cold duplex architecture is implemented for most modules.

As the spacecraft orbit allows uninterrupted ground contact, no payload or HK data storage is necessary. Despite this, the design provides the possibility of adding an optional Local Mass Memory in charge of storing the payload and HK data for subsequent down-link. See section 6.12.5 for more information.

6.12.3 Baseline Design

6.12.3.1 Data Handling Interfaces

The Data Handling subsystem interfaces with the Power subsystem, the AOCS sensors and actuators, the TT&C subsystem, and the payload instruments. The DHS manages the Power subsystem via direct commands from the CPDU, and acquires a collection of housekeeping parameters to allow monitoring of its status.

Figure 6-27 shows the connection of the DHS with the other satellite blocks

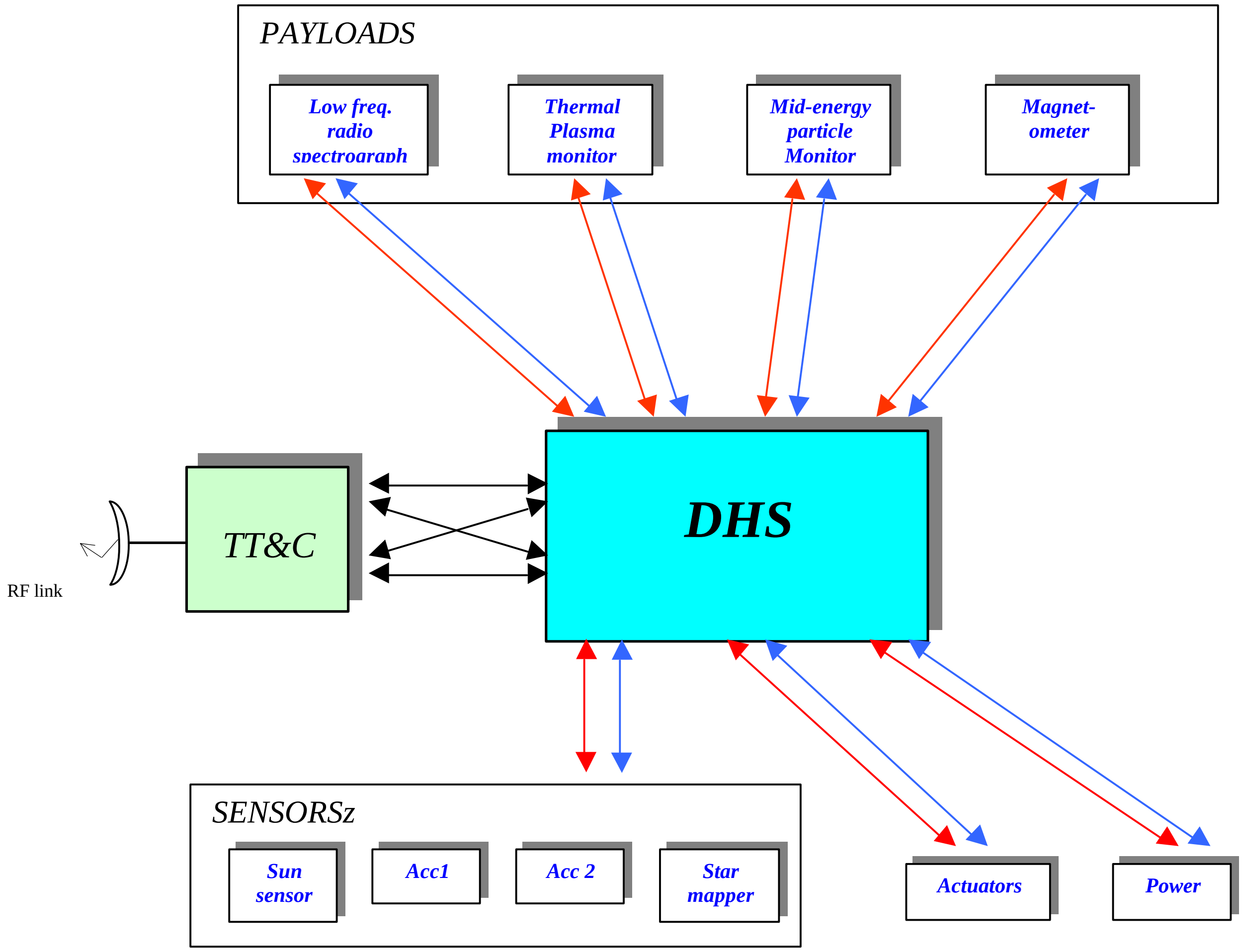


Figure 6-27: Data Handling Connection Schematic

The DHS interfaces with the TT&C subsystem via digital input/output lines for the telemetry and telecommand streams. A number of lines from CPDU are reserved for commanding the transmitter switchover and the antenna connections. Additional lines provide status information on the TT&C subsystem.

The AOCS sensors are interfaced via direct serial lines.

The four instruments are connected to the DHS through RS422 or TTC.B.01 link for command transmission and data retrieval.

6.12.3.2 Data Handling Modules

The data handling system consists of eight principal modules connected together by an internal bus. All modules are internally cold redundant except for the Telecommand and Reconfiguration modules, which have hot redundancy. The block diagram in Figure 6-28 shows the architecture of the DHS.

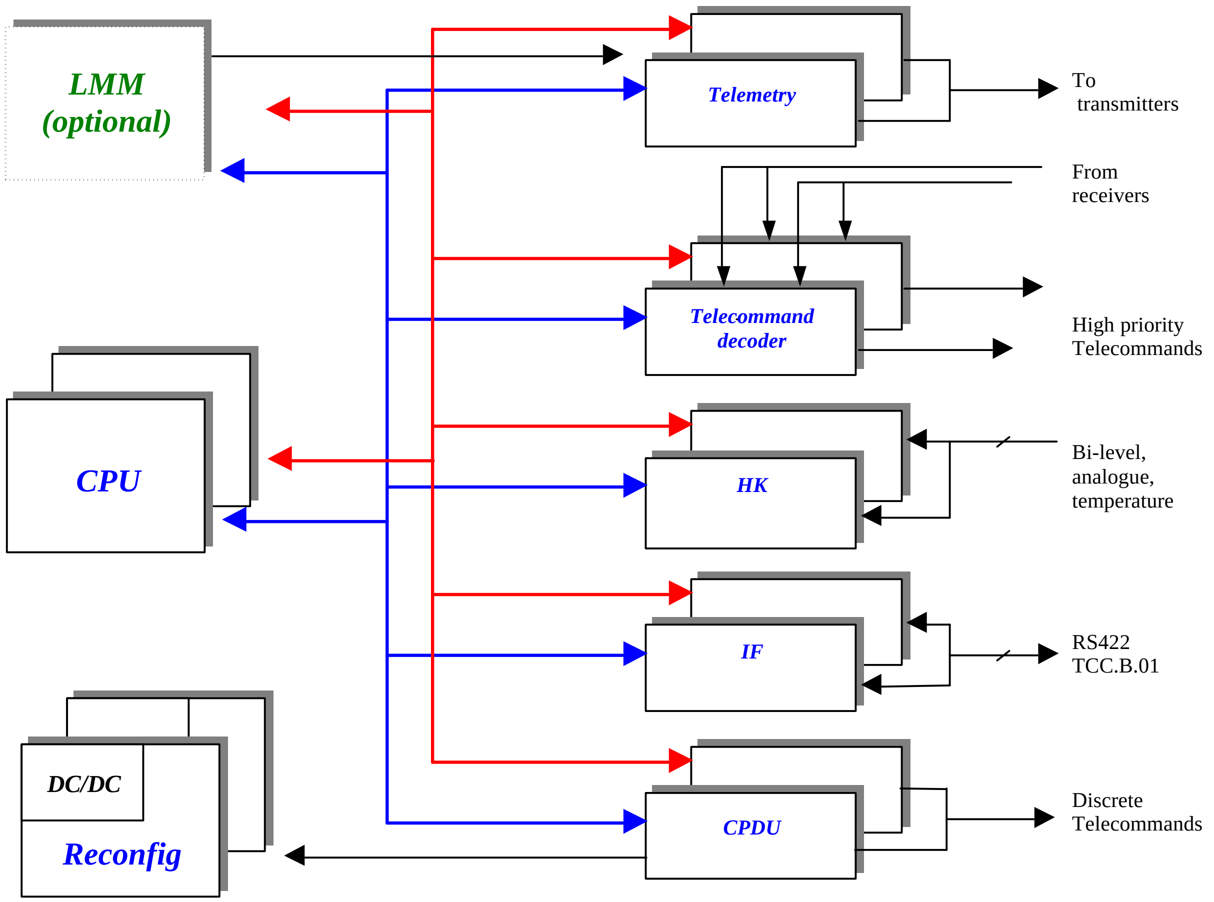


Figure 6-28: DHS Block Diagram

The functions of each module are as follows.

The *CPU Module* is based on an ERC32 processor. It also consists of the necessary program/data memory and a non-volatile memory for bootstrap and application software storage. An interface with the internal system bus is used to communicate with the other modules.

The *Telemetry Module* collects the data packets provided by the CPU module that must be encoded and transmitted to transponders. The telemetry module is also in charge of directly acquiring and transmitting the essential telemetry without CPU module intervention. The telemetry data rate varies from 8.7 kbps during operational mode to 100 bps in safe mode, when payloads are switched off and low-gain antennas are used for data download.

The *Telecommand Module* acquires, demodulates and decodes the telecommand packets sent by transponder. The high priority commands are immediately executed while the others are passed to the CPU module where they will be processed by software. The telecommand rate during nominal operation is assumed at 2 kbps, and this will be reduced to 50 bps in safe mode.

The *Housekeeping Module* is in charge of acquiring HK data from the spacecraft. It provides standard interfaces such as temperature interface, and bi-level and analogue status interfaces.

The *Interface Module* implements the RS422 and TTC.B.01 interfaces that provide a direct serial link with instruments and sensors. Commands are sent to instruments and sensors, and telemetry data is retrieved, over these interfaces.

The *Command Pulse Distribution Unit* implements the interfaces for command pulses managed by the CPU module.

The *Reconfiguration Module* performs the reconfiguration of the DHS in the event of a software or hardware failure.

6.12.3.3 Software

The operating S/W (operating system and standard services) shall be recurrent for a very large part. In addition, the continuous ground coverage reduces the autonomy requirements and therefore also the software complexity.

6.12.4 Budgets

Mass	**Power**
9.5 kg	13.5 W

Table 6-20: SWM DHS Mass and Power Budget

6.12.5 Option

In order to store data on-board during any outage periods, an optional Local Mass Memory could be provided. The data could then be dumped subsequently. In this case the stored data and real time telemetry would be simultaneously downlinked.

The Local Mass Memory Module size would be 0.75 Gb, corresponding to 24 hours of HK and payload data storage. During normal operations the mass memory can be switched off.

6.13 Telecommunications Subsystem

6.13.1 Requirements and Design Drivers

The requirements for the definition of the SWM telecommunication subsystem have been the following:

- Continuous real time downlink of payload data (@ 6.7 kbps) plus housekeeping telemetry (typically 2 kbps).
- Telecommand uplink (typically at 2 kbps) and ranging capability required, but not on a continuous basis
- halo orbit in L_1 with nominal range up to 1.7 M-km

6.13.2 Design Assumptions

The following sections describe the main assumptions made in the baseline definition and the link budget evaluation.

6.13.2.1 Frequency of Operation

For operation in L_1 it is recommended to use X-Band (7190 - 7235 MHz uplink, 8450 - 8500 MHz downlink), which is allocated to the Space Research Service. The highly interfered environment existing in S-Band could compromise the nominal performance due to the low received levels involved in missions to the Lagrangian points.

6.13.2.2 Ground Station Assumptions

The typical performance provided by the 15 metre stations of the ESA ESTRACK network has been assumed.

	7 GHz Transmit	**8 GHz Receive**
Frequency [MHz]	7145-7235	8400-8500
Polarisation	RHCP or LHCP	RHCP and LHCP
Cross polarisation [dB]	-25.00	-25.00
Sidelobes	ITU App. S7	ITU App. S7
Antenna efficiency	> 65%	60.30
EIRP [dBW]	82 (400W SSPA)	
G/T @90° [dB/K] Clear Sky		39.10
G/T @10° [dB/K] Clear Sky	-	38.00
G/T @10° [dB/K] 99% of the year for Kourou weather conditions		35.60

	7 GHz Transmit	8 GHz Receive
Rx/Tx isolation [dB]	90.00	
Pointing accuracy [dB]	< 1	

Table 6-21: Ground Station Performance

It is assumed the availability of Turbo Decoders on the station is:

Eb/No (for PFL = $1e^{-5}$) = 0.8 dB

Where: Eb = energy per bit
No = noise power density no.

6.13.2.3 Transponder and RF Power Amplifier Assumptions

The transponder required to support SWM is an X/X Near-Earth type, with Tx/Rx coherency and ranging capability. The transponder will be adapted to an external TWTA power amplifier. The requirements are as follows.

Property	**Value**
Frequency	Receive: 7190-7235 MHz, Transmit: 8450 - 8500 MHz
Transmit Power	0 dBm at the output of the transmitter within the transponder 30 Watts TWT external
TM Modulation	NRZ/BPSK/PM
TM Data Rates	100 bps (HK only) and 9 kbps (HK+RT data). TM Data rates selectable by TC
TM Coding	Turbo Encoder
Receive Threshold	-135 dBm (TC demod)
Noise Figure	2.5 dB
TC modulation	NRZ/BPSK/PM
TC Data Rates	25 bps and 2 kbps TC Data rates selectable by TC
Dimensions	Transponder : 275 x 110 x 197 mm TWT : 58 x 50 x 350 mm EPC : 100 x 80 x 360 mm
Power Bus	Transponder: From 21 to 50 V Amplifier: 22V to 37V

Table 6-22: Transponder Requirements

6.13.3 Antenna Analysis

In the definition of the antenna performance required for SWM, three different scenarios have been considered:

1. **Contingency operations and safe mode**
   The spacecraft should be able to communicate with Earth for any aspect angle. Therefore omni-directional coverage is highly desirable for both transmit and receive.

2. **Transfer orbit**
   Just after separation from the launcher, the baseline considers SWM attached to the top of the SAM S/C for several hours. During this time, both bodies are three-axis stabilised and the SWM cannot use a low gain antenna (LGA) nominally placed on the bottom side. However, the S/C attitude can be controlled to have an LGA facing the Earth.

   Following the detachment from the SAM S/C, communication with Earth is ensured by the quasi omni-directional coverage provided by two semi-hemispherical LGAs.

   As soon as the angle Sun-Earth-S/C is lower than 31°, the link can be established via a fixed medium gain antenna (MGA).

3. **Nominal operations**
   During nominal operations the spin axis is pointing towards the Sun, and the angle Sun-Earth-S/C is kept below 31°. The link will be established via the MGA

6.13.3.1 Low Gain X-Band Antenna Assumptions

Each low gain antenna (LGA-1 and LGA-2) has a nearly hemispherical coverage, with an absolute gain >-5 dBi, and opposite polarization (RHCP and LHCP). The estimated mass of one low gain antenna is 300g. LGA-2 (see Figure 5-33) will be mounted on a boom with deployment mechanism. This mechanism must deploy to an intermediate position when SWM is attached to SAM.

6.13.3.2 Medium Gain X-Band Antennas Assumptions

To provide a wide coverage over +/- 31° with +9dBi gain, a horn antenna is proposed. This Medium Gain Antenna (MGA) uses dual flared technology and supports Tx/Rx with RHCP polarisation. The mechanical interface is coaxial and the antenna is mounted on one side of the S/C aligned with the spin axis. Approx. dimensions are 25cm x 25cm x 40 cm, including the transition to coaxial, and the mass is 700g (including support brackets).

6.13.3.3 Antennae Coverage and Location

Figure 5-33 shows the proposed antenna location and the coverage provided.

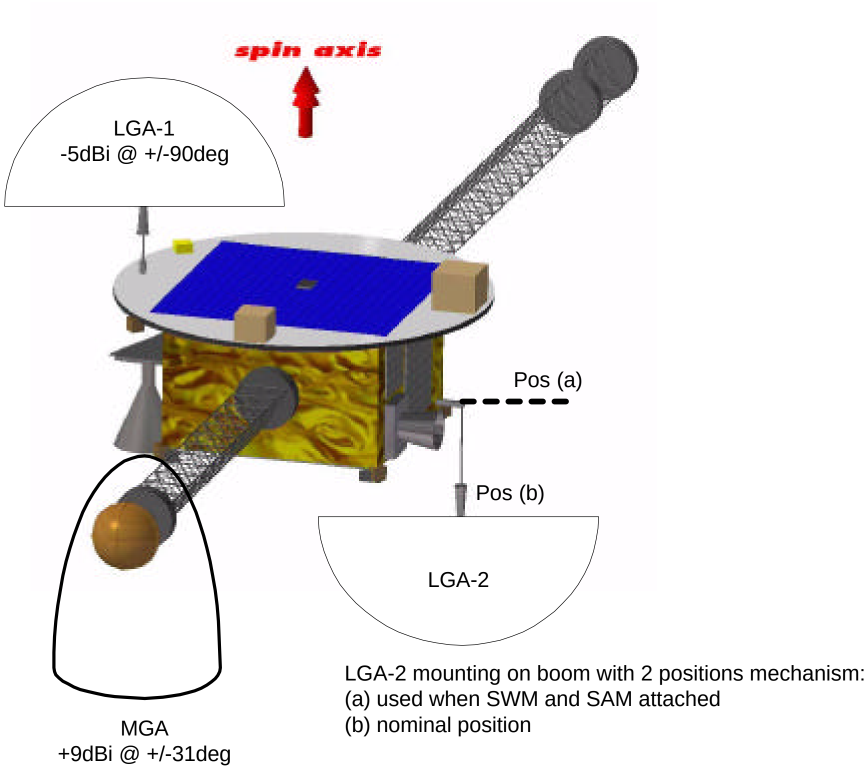


Figure 6-29: SWM Antenna Layout and Coverage

6.13.4 Link Budget Evaluation

6.13.4.1 Uplink Budget

Range: 1.7M km	**15m station** **82 dBW EIRP**
Operation via LGAs	25 bps (clear sky)
Operation via MGA	2 kbps (99.5% of year in Kourou)

6.13.4.2 Downlink Budget

Range: 1.7M km	**15m station** **G/T = 35.6 dB/K, 99% year time Kourou**
Operation via LGAs	100 bps
Operation via MGA	9 kbps

6.13.5 Communications Baseline Design

The communications subsystem consists of the following elements:

- Two Low Gain Antennae
- One Medium Gain Antenna
- One RF Distribution Unit
- Two transponders and two TWT 30 Watt power amplifiers.
  The transponder integrates the transmitter (plus modulator), the receiver (plus demodulator), and the diplexer that combines both units into a single port towards the antennae.

The design proposed for the communications subsystem is depicted in Figure 5-32.

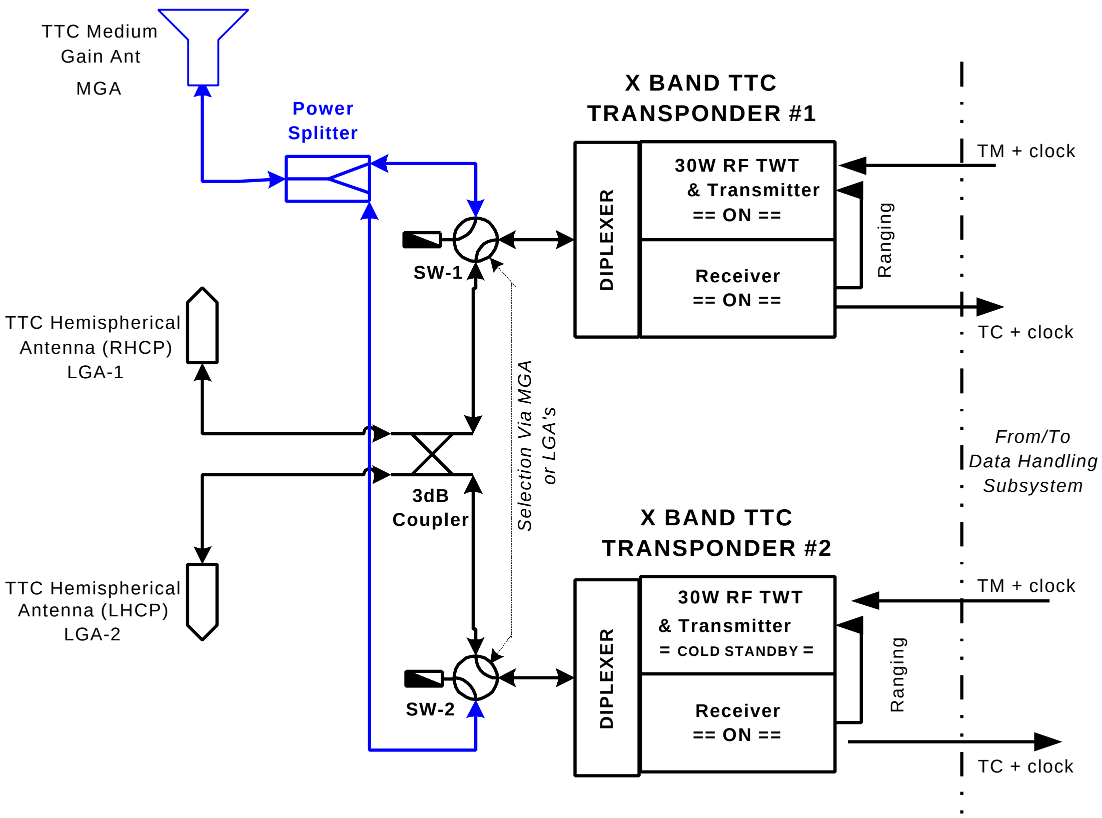


Figure 6-30: IMM Communication Subsystem

6.13.5.1 Operation with TTC Low Gain Antennas

The spacecraft transmits and receives simultaneously via both Low Gain Antennae (LGAs). Signals received by LGA-1 (RHCP) and LGA-2 (LHCP) are combined in a 3 dB-hybrid and routed to both receivers. The receivers operate in hot redundancy; the demodulated data is sent to the OBDH subsystem where one chain will be selected for further processing. The 30W TWT amplifiers operate in cold redundancy (one on and the other in standby).

6.13.5.2 Operation with TTC Medium Gain Antenna

The switches SW-1 and SW-2 route both transponders towards the MGA. Note that with theuse of only one of these switches, it would still be possible to receive the uplink received via the LGAs with the 2nd transponder.

6.13.6 Budgets

6.13.6.1 Mass Budget

Items	**Number of units**	**Nominal Mass per unit (kg)**	**Total Nominal Mass (kg)**
X-Band Transponder	2	3.50	7.00
30W TWT amplifier	2	3.75	7.50
RF Distribution unit (including power combiners, switches and harness)	1	2.50	2.50
X-Band LGA	2	0.50	1.00
X-Band MGA	1	0.70	0.70
Total Mass (kg)			18.70

Table 6-23: Mass Budget of the Telecomms Subsystem

6.13.6.2 Power Budget

Item	**No. of units**	**DC power (Watts)**	
X-Band Transponder	2	6.00 x 2 = 12 W	Both transponders ON
X-Band 30W TWTA	2	60.00 W 7.50 W	TWTA ON TWTA Stand-by
Power Consumption		79.50	With TWTA in Cold Redundancy

Table 6-24: Power Budget of the Telecomms Subsystem

6.14 Structures

6.14.1 Requirements and Design Drivers

The baseline launch configuration is for the Solar Wind Monitor (SWM) satellite to be launched on top of the Solar Activity Monitor satellite (SAM), as a composite. The I/F design is through four points, using pyrotechnic devices for separation (discussed in the SAM structures section, section 7.14). For the configuration of the stacked spacecraft, see Figure 6-19.

The structural design is not specifically driven by a volumetric budget requirement for the fairing of the reference launcher (Soyuz + Fregat).

The main overall stiffness requirements for the S/C will be governed by the stack design with the SAM.

The basic requirements for the satellite for the Soyuz-Fregat launch are:

- The first lateral frequency for each S/C separately should be >15 Hz
- The stack of the two S/C should have a first lateral frequency >12 Hz

For all the equipment supporting the instruments and the S/C operations the structure provides:

- A platform for electronic equipment, propulsion, power & harness
- Easy access to the inside of the spacecraft for AIV activities

An overview of the structural design for the SWM mission is shown in Figure 6-31.

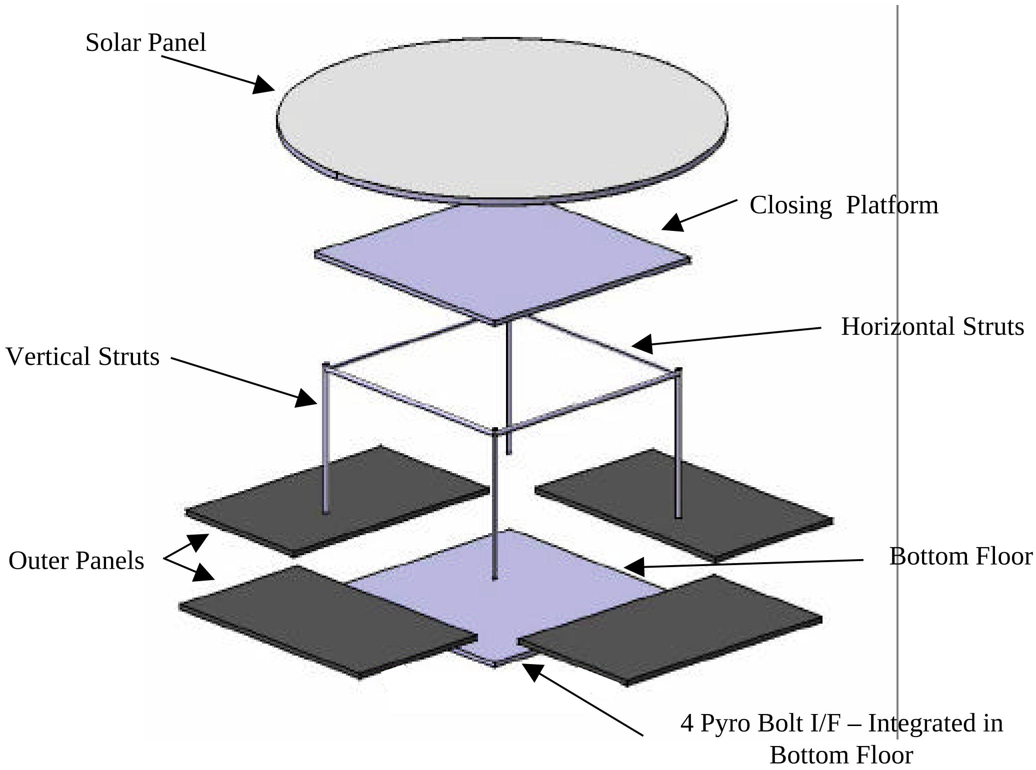


Figure 6-31: Primary Structure for the S/C

6.14.2 Assumptions and Trade-Offs

For cost reduction, an existing commercial platform was analysed first. Proteus was considered suitable from a requirements point of view. However, it is too small in volume for the required spacecraft equipment and instruments. Therefore, the structural concept of the Proteus design was taken and sized to the needs of the mission.

6.14.3 Baseline Design

For the baseline design the spacecraft interfaces to the SAM S/C via 4 I/F bolts with pyrotechnic separation devices. These I/Fs are integrated in the bottom floor of the satellite, which is the main load-bearing structure and is machined from one piece of aluminium. The S/C builds up from this platform with four vertical struts, connected via four horizontal ones, topped by a sandwich panel.

The four outer panels are hinged from the bottom plate. These panels will support the equipment and instruments, hence the stiffness requirement for these panels. The panels close the box and are then attached to the truss structure.

The solar panel is a separate plate attached to the top of the S/C and thermally insulated from it.

The main advantage of the structural concept proposed for SWM is the simplification of the AIV procedure, and consequent cost reduction because of the easy access to the internal components of the S/C.

6.14.3.1 Frequency Requirements

The stiffness of the structure for the selected concept for this platform is not driven by launcher requirements, but by the panel stiffness required for the equipment supported on the outer panels and solar array. The current design shows a high frequency well above the required 15 Hz for each S/C. For the optimal mass/stiffness design, the launcher requirements for the satellite stack and the local panel requirements for instrument support will need to be balanced.

6.14.4 Mass Budget

Table 5-30 shows the mass breakdown of the primary structure.

Item	Qty.	Unit Mass (kg)	Unit mass with margin (kg)	Total mass with margin (kg)
Bottom Floor	1	28.0	33.6	33.6
Outer Panels	4	3.2	3.8	15.2
Vertical Struts	4	0.28	0.33	1.32
Closing Platform	1	2.11	2.53	2.53
Horizontal Struts	4	0.17	0.2	0.8
Solar Panel	1	6.1	7.3	7.3
Inserts and Miscellaneous	1	5.0	6	6
			TOTAL:	**66.75**

Table 6-25: Primary Structure Mass Budget

6.14.5 Options

If the SWM is to be launched on a separate launcher, a standard 937 interface adapter could be accommodated, with an estimated mass of 3.6 kg. However, the required configuration would not change, since the envisaged bottom floor could be designed to interface with this adapter ring.

6.15 Programmatics

6.15.1 Master Schedule

The project Gantt chart below in Figure 5-35 indicates the major mission phases consistent with the following key milestones:

- Start of the project Phase A in July 2002
- Launch in October 2006
- A 6-month (maximum) Transfer Phase to L_1
- A nominal Operational Phase of 5 years until first quarter 2012

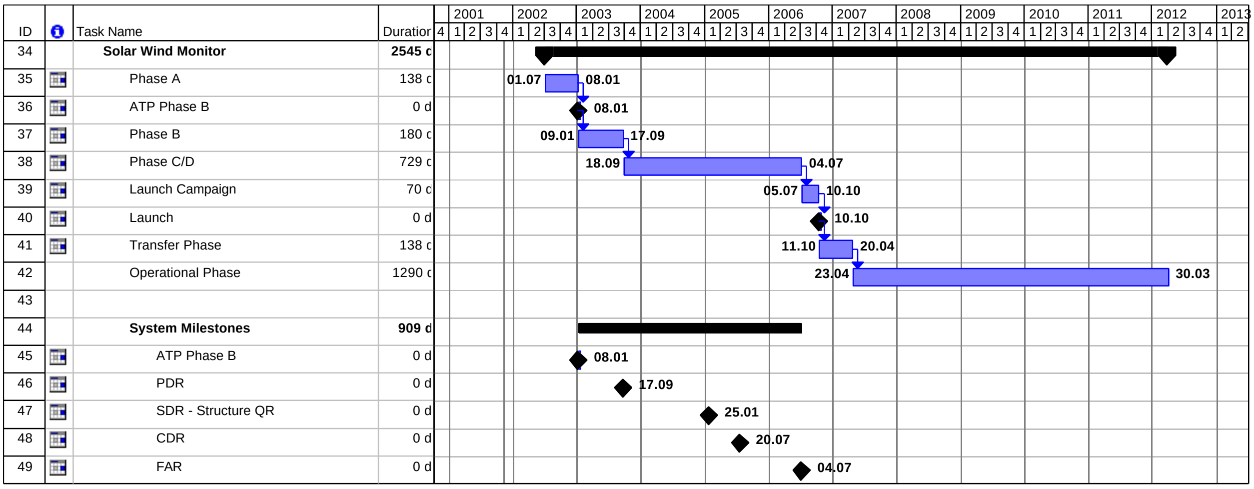


Figure 6-32: Project Master Plan

6.15.2 Development and AIV

The SWM spacecraft includes the main building blocks:

- A body like that of the Proteus spacecraft, carrying a fixed Solar Array on the 'top' surface.
- Four side panels carrying the experiments and the spacecraft subsystem electronics (Avionics, Telecom, AOCS, Power S/S).
- A deployable low gain antenna is fixed to one side panel on a short boom.
- A lower panel carrying the Propulsion system, Propellant and Pressurant tank and supporting one low gain antenna boom.
- A payload of scientific instruments including a pair of deployable booms for the magnetometers.

The development of the spacecraft relies on existing design and available technology. The structure is a Proteus-like frame of increased size, for which qualification testing is required. Precautions need to be taken in order to ensure the high level of magnetic cleanliness required by the mission.

The radiation environment is considered mild, and heritage from past projects like e.g. SOHO is available, for the needed components. Shielding should be adapted to the specific SWM design configuration.

The project development and more specifically the cost estimates have assumed a streamlined industrial team whereby the Prime Contractor is responsible for:

- Overall mission analysis
- Overall design development and procurement of the spacecraft
- Detailed spacecraft design at system and subsystem level
- Direct procurement of the spacecraft units, equipment and major assemblies (hardware and software)
- Overall spacecraft assembly, integration and verification (AIV) activities
- Definition and control of the technical and operational interfaces of the instruments.

6.15.2.1 Model Philosophy

Considering the moderate development risk identified in most aspects of the spacecraft design, a Protoflight approach has been selected at spacecraft level, based on a 3-model philosophy:

- **Structural Model (SM)**
  Will ensure the mechanical qualification of the spacecraft design. Most of the unit assemblies will be represented by structural dummies.
- **Avionics Test Bench model (ATB)**
  Will ensure verification of the overall electrical, functional and software interfaces. Breadboard units (BBs) will be used most of the time, exceptionally Interface Simulators could be used for the Payload Units. Elegant BB units (EM-like with commercial components) or modified EM could be used if cost-effective, e.g. in case of recurring units with EM available, or off-the-shelf equipment. This is in particular true for the experiments.
- **Protoflight Model (PFM)**
  Built to full flight standard, this will be subject to qualification test levels with acceptance duration.

As a programmatic approach, the use of Hi-Rel EEE parts has been assumed. However, the reliability level of EEE parts must be carefully assessed, due to the impact this selection necessarily has on the risk and cost of the project.For the costing, procurement of European Hardware has been assumed in general whenever the design and technology are available. Provision of spares kits is foreseen for all units. They could be specifically procured or available as heritage of recurring units from past projects.

6.15.2.2 AIV Approach

Taking into account the given model philosophy and the expected development time of the instruments, an overall AIV plan is outlined in Figure 5-36.

ID	Task Name	Duration	Start
	Solar Wind Monitor	1117 d	01.07.02
1	Phase A	138 d	01.07.02
2	ATP Phase B	0 d	08.01.03
3	Phase B	180 d	09.01.03
4	PDR	0 d	17.09.03
5	CDR	0 d	20.07.05
6	FAR	0 d	04.07.06
7	Launch	0 d	10.10.06
8	Phase C/D	729 d	18.09.03
9	Design	138 d	18.09.03
10	Procurement SM	230 d	24.12.03
11	Procurement ATB BB Units	230 d	21.11.03
12	OBSW V0.1	0 d	02.07.04
13	P/L STM Experiments Need Date	0 d	10.11.04
14	SM AIV	55 d	10.11.04
15	SM SRM Usage with SM SWM AIT	17.3 d	05.01.05
16	EGSE SDE Need Date	0 d	28.11.03
17	EGSE - Test Sequences	165 d	28.11.03
18	EGSE Commissioning	60 d	16.07.04
19	P/L EM Experiment Need Date	0 d	07.04.05
20	OBSW V1.0 Need Date	0 d	27.10.04
21	ATB AIV	204 d	08.10.04
22	Procurement PFM structures	180 d	10.11.04
23	Procurement Units PFM	276 d	08.10.04
24	OBSW V2.0 Need Date	0 d	26.08.05
25	EGSE - Test Sequences Update + CHECK	61 d	21.07.05
26	P/L PFM Experiments Need Date	0 d	30.12.05
27	PFM AIV	177 d	31.10.05
28	SM SRM Usage with SM SWM AIT	15.3 d	07.04.06
29	Launch Campaign	70 d	05.07.06
30	SRM Final mating	0 d	15.08.06

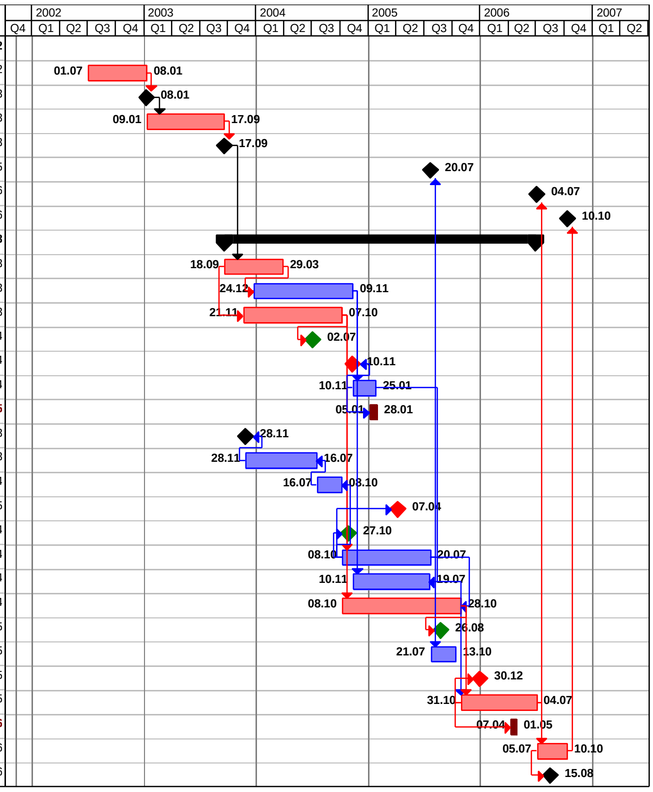


Figure 6-33: AIV Planning Chart

The validity of the planning is based on the following key assumptions:

- Final qualification of the instruments will be achieved on the spacecraft PFM.
- Instrument SMs will be delivered at a build standard compatible with the spacecraft STM programme. Where required, mechanical alignment of instrument boxes with the spacecraft structure will be tested at spacecraft level.
- Instrument BBs will be delivered at a build standard which, at a minimum, has to be representative of the electrical and functional interfaces. No calibration activities are envisaged on the ATB.
- The instrument PFMs will be installed on the satellite by the Prime Contractor. Functional and interface tests will be performed to certify their proper on-board accommodation. Calibration activities will be performed on the flight satellite.

6.15.2.3 Critical AIV Aspects

A critical aspect is the timely release of the SW versions. The on-board SW version 1.0 must be ready for the start of the ATB activities. This should be the first version fully implementing the required system SW functions.

The final version, implementing the results of system testing on ATB, must be loaded on the PFM units before they are delivered to the system for integration. This final release V2.0 shall be tested on the PFM.

6.15.3 Programmatic Risk Assessment

The risk elements, from a programmatic point of view can be summarised as follows:

- The proposed short development time is a risk factor due to the reduced possibility of recovering delays.
- In the hypothesis that a dual launch with SAM is selected, there could be the need for combined system tests of the two satellites. This implies that the test campaign of the flight SWM and SAM must be phased to meet the joint test needs. The environmental test facilities must then be carefully selected to allow both of the satellites to be processed in the same period of time.

6.15.4 Critical Technology

No critical technological aspects have been identified in the SWM platform design, except for the radiation environment, that deserves some attention.

Rad-hard components may be needed right across the spacecraft. It is assumed that in this respect the project will strongly benefit from available technology as heritage from past ESA projects. A reference for this kind of mission is SOHO.

For the mass memory, adequate shielding is envisaged, since the availability of high-density memory chips in rad-hard version is not likely.

6.15.5 Links to Other Projects

Other Agency projects (like SOHO) have been used as a reference for costing purposes. The proposed spacecraft concept is a dedicated design for the SWM mission, and does not rely on parallel developments.

6.15.6 Option: AIV for Solid Rocket Motor

In the event that a solid rocket motor (SRM) is used to propell SWM in its transfer phase to L_1 (Option 1, launch with Rockot), there would be an additional: the SRM should be procured in parallel with the SWM platform. A SM of the SRM would need to be delivered for use in sine vibration tests and acoustic tests on the system SM. The PFM SRM would be coupled to the satellite during system PFM AIV only for acoustic and shock separation tests. Then it would be charged with solid propellant (at the supplier's facilities) and separately shipped to the launch site for final coupling to the platform during the SWM launch campaign.

6.16 Risk Assessment

The Space Weather Service (SWS) is, from a risk point of view, to be handled differently from usual scientific missions, because the strongest requirement driving this analysis is the requirement for a continuous service of near real-time data. By this definition, success of the service depends on a successful delivery of near real-time data to the user Secondary benefits are not considered.

In the scope of this study, the risk assessment is limited to the risk of loss of service availability of the SWM space segment consisting of a single spacecraft.

6.16.1 Requirements and Design Drivers

For the entire SWS, no requirement for the service availability is yet defined and mapped into space segment dependability requirements. Therefore the risk assessment of the SWM space segment focuses on the dependability of the single satellite.

6.16.2 Assumptions and Trade-Offs

The baseline configuration is defined by a single satellite carrying four instruments. It is assumed that all instruments are needed over the intended in orbit lifetime of five years to provide a full service.

6.16.3 Baseline Design

The baseline design is a spinning spacecraft with a single redundant reliability structure for most of the units of the various subsystems. Designed as a spinning spacecraft, the complexity of the AOCS system is low. The power subsystem is based on a fully regulated bus design with internal redundancy. The battery and solar array provide redundancy at cell level.

The solar array is fixed, thus not susceptible to any deployment failure. Mechanisms are used to deploy instruments and antennae. The propulsion system is a monopropellant hydrazine blow-down system with RCS thrusters only. The Telecomms system is fully redundant. SWM relies on passive thermal control plus heaters on the tank and RCS thrusters.

6.16.3.1 Feasibility

The reliability block diagram of the present baseline is shown below. Assuming typical failure rates the reliability of the baseline design can be predicted to be approx. 0.8 at the end of a 5-year orbital lifetime.

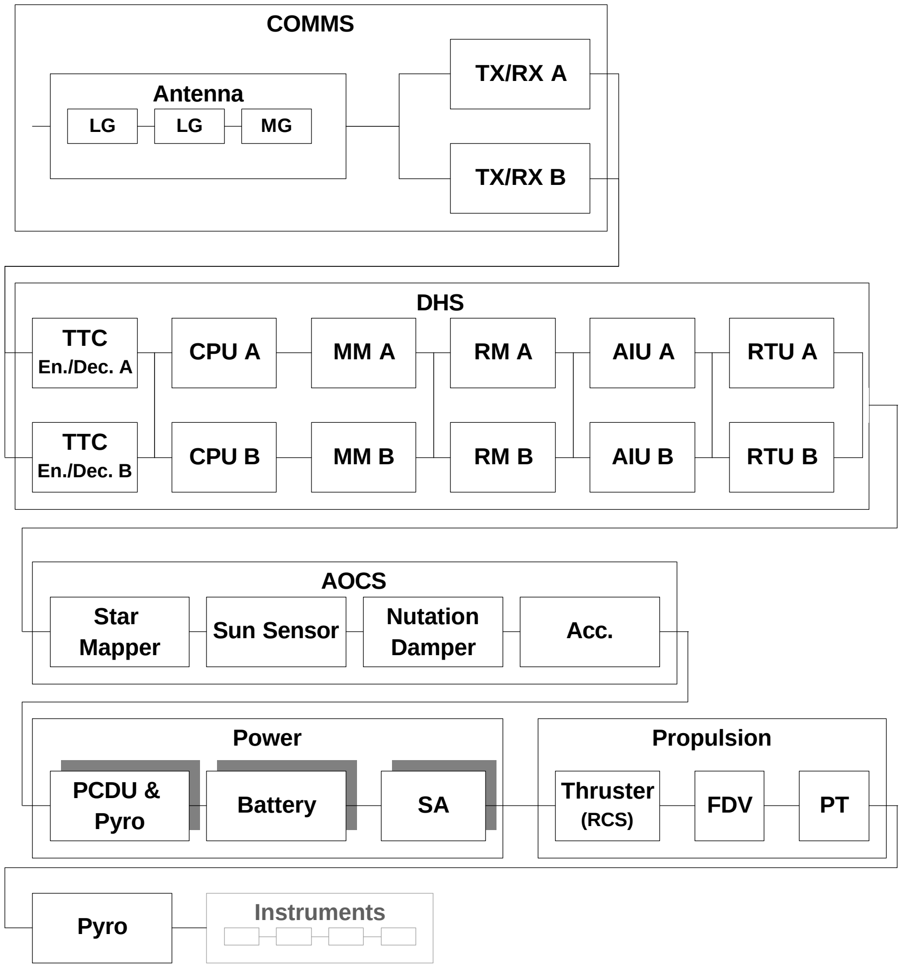


Figure 6-34: Reliability Block Diagram

A critical point is the relatively high temperature of up to 130°C at the solar cells that may have a potential impact of the available power depending on cell performance at this temperature level.

The major risk contributor to the system unreliability of 0.2 at the end of the lifetime is the DHS with ≈ 60%. However, the DHS is designed as a fully redundant system and the high percentage expresses only the relatively simple design of the SWM S/C with respect to the other subsystems.

6.16.4 Summary

The SWM mission can be considered to be close to a classical 'science' mission but with a very limited set of instruments and a simple design of the S/C. It is based on available technology for the various subsystems of the SWM spacecraft.

Besides the characteristics of the SWM as an independent space element, in the course of the ongoing activities it should be studied whether and to what extent SWM can be used to compensate for partial loss of SW monitoring capabilities of the IMM constellation.

Figure 6-35 indicates the expected availability of a SW space segment built up of IMM & SWM, assuming that SWM can compensate for the partial loss of functionality of IMM instruments (4/4 S/C + 3/6 instruments plus SWM), or the loss of an IMM spacecraft (3/4 S/C + 6/6 instruments plus SWM) as a function of instrument reliability.

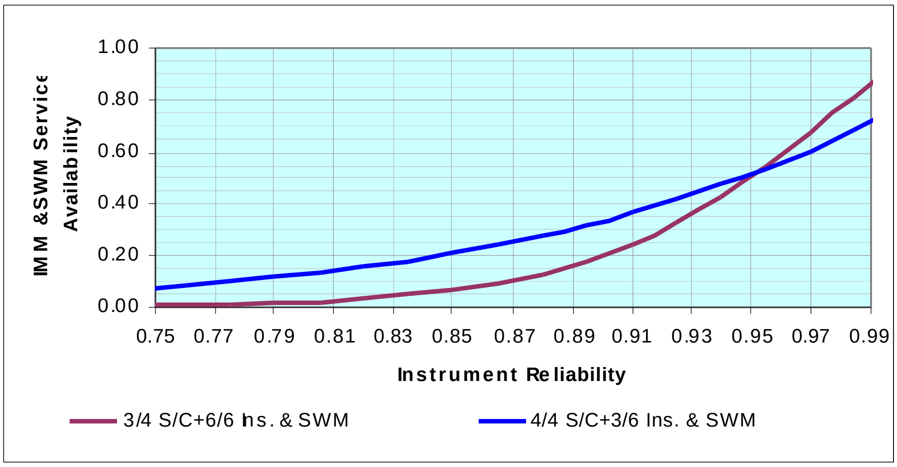


Figure 6-35: Spacecraft Availability

6.17 Cost Estimate

This chapter presents the SWM Platform and Payload Phase B and C/D cost estimate.

6.17.1 Main Costing Assumption

It is assumed that the industrial organisation for the SWM platform project involves a Prime Contractor, handling the detailed design at platform level. The Prime is also assumed to be responsible for Assembly, Integration and Test (AIT) activity support at Spacecraft System level.

Most SWM platform and payload equipment is assumed to be off-the-shelf type or based on existing and available technology, as presented in the report. All cost estimates are based on references and cost estimation methods in line with the above general hypothesis. It is considered that the spacecraft design activities and equipment selection will be commensurate with the operational nature of the mission. A 5-year lifetime was taken into account.

No geographical distribution constraints are included.

6.17.2 Cost Estimate Methodology

The following methods have been used, in descending order of preferred method:

- Reference to similar ESA missions;
- Reference to similar equipment/system level costs, taking into account the amount of new development required;
- Expert judgement from technical specialists in combination with similar equipment references, in the case that the amount of new development is extensive;
- Expert judgement from technical specialists only, if references are not available;
- Equipment cost models;
- The ESA internal, system level cost model RACE;
- System level cost relationships (for the Prime and Payload/Payload Contractor activities), based on recently observed relationships for relevant references.

6.17.3 Scope of the Cost Estimate

In accordance with the study requirements, the cost estimate covers:

- the SWM Platform
- Instruments (as far as information is available)
- Phase B and C/D costs of the mission
- the launch

Excluded:

- Operational costs (ground segment)

Furthermore, the cost estimates are for the industrial costs only.

The SWM Platform Phase B and C/D cost estimate includes:

- A provision for the Phase B development costs
- The phase C/D costs, up to PFM
- Phase C/D equipment, software and platform level costs including Ground Support Equipment costs
- Spacecraft system level activity costs (management & control, engineering, PA, AIT)
- Separation device costs
- Platform design maturity provision

For the Payload, the Phase B and C/D cost estimate includes:

- A provision for the Phase B equipment development costs
- The phase C/D costs, up to PFM

The industrial cost is considered as the Prime Contractor offering a firm fixed price would see it. It covers the supply of the flight unit with the associated development models when applicable, the spares, the specific GSE and the user manuals. It also covers the Project Office cost of the equipment suppliers.

6.17.4 Phase B Cost Assumptions

The Phase B costs have been estimated based on the Phase B versus Phase C/D cost ratios for projects with a strong Prime Contractor involvement at subsystem level. The Phase B costs do not cover the pre-development assumed to be part of Phase C/D.

6.17.5 Phase C/D Cost Assumptions

For the cost estimates, the platform development and Assembly, Integration and Test (AIT) is regarded as being a complete project on its own handled by the Prime Contractor at satellite level. All platform subsystem Project Office (PO), AIT and Ground Support Equipment (GSE) costs are therefore included at platform level.

6.17.5.1 AOCS

- The AOCS design is derived from Cluster.
- Prices have been estimated based on this reference but are adjusted with today’s market price trends.
- All equipment is assumed to be off-the-shelf with eventual simple modifications.

6.17.5.2 Propulsion

The cost estimate for the propulsion system is mainly based on different prior ESA missions. Necessary adaptations have been taken into account. Further Project Office costs on sub-system level are presented, based on ratios observed on previous projects.

6.17.5.3 Electrical Power

- Solar Array costs are based on ESA internal Cost Estimating Relationships (CERs). Although the GaAs solar cells will be off-the-shelf equipment, the panel configuration will be unique. The cost estimate therefore assumes that a normal Solar Array development effort including development models (STM and PFM) will be required.
- The PCU and PDU costs have been derived from similar items from prior ESA missions.
- The Rosetta battery was the reference for Li-Ion Battery cost.

6.17.5.4 Harness

Harness costs were determined using ESA internal CERs. Since the harness architecture has to be newly developed, this has been taken into account in the cost estimate.

6.17.5.5 TT&C

For the TT&C sub-system the procurement is proposed to demand not only PFM and STM but also EM equipment to assemble an Avionics Test Bench (ATB).

The costs are mainly derived from the reference mission Herschel-Planck, on which minor equipment modifications were taken into account.

6.17.5.6 Data Handling

- The Data Handling System consists of a single box
- The CDMU is internally redundant.
- The data rate can be supposed to be low.
- For the cost estimate a partly-customised CDMU has been assumed.

6.17.5.7 Structure

The structure cost has been based on the ESA ‘low cost mission’ internal cost model.

6.17.5.8 Mechanisms

The items concerned are instrument and antenna booms and the associated deployment mechanisms. The antenna boom cost is based on Cluster reference. Cost for instrument booms are derived from close discussion with the mechanism expert.

6.17.5.9 Thermal Control

The Thermal Control Equipment is assumed to include only passive hardware such as paint and MLI. The Thermal Control Subsystem engineering activities such as thermal control analysis and configuration design are included in the Engineering cost. Specific instrument thermal hardware is included with the payload costs.

6.17.5.10 On-Board Software

Both the Data Management Software and the AOCS Software are considered to be based on existing on-board software, with only the payload management being specifically developed for SWM.

The cost estimates for the SWM Data Management and AOCS Software are based on the costs for modified existing software from other ESA missions.

6.17.5.11 Ground Support Equipment

The cost estimate for Ground Support Equipment (GSE) covers the costs for all Electrical and Mechanical GSE required for the platform. It has been taken into account that the GSE will be mainly based on existing hardware and designs. Accordingly a standard ratio observed on past projects has been applied.

6.17.5.12 Platform Assembly, Integration and Test

The platform AIT cost estimate includes the costs for all platform mechanical and electrical integration activities and tests, as well as the mechanical mating of the platform and the payload. The cost estimate is based both on a cost estimate relationship and on an independent AIT planning assessment performed within the CDF, with which the results are in close agreement.

6.17.5.13 Project Office Activities

The Project Office costs at subsystem and platform level include:

- Management and Control (including overheads on subcontracts)
- Product Assurance
- Engineering and documentation including payload interface engineering both at system and sub-system level, except for propulsion

6.17.5.14 Payload

The SWM payload for each spacecraft consists of the following instruments:

- Thermal Plasma Monitor (TPM)
- Mid-Energy particle Monitor (MEM)
- Magnetometer (MAG)
- Coil Radio Spectrograph (CRS)

The instrument cost assessment is characterised by the rather limited amount of available reference material and technical data on the instruments. It has been assumed that institutes rather than industry will procure the instruments.

The cost estimates are based on similar instruments or equipment with matching technology.

The monitors cost estimates are rooted in equipment and sensors on XMM.

Magnetometer costs are modified data from Cluster.

To adapt the different costs, various ESA internal cost models have been used.

The cost estimates are based on similar instruments or equipment with matching technology. Different ESA internal cost models have been used.

It has to be noted that for more detailed estimates, further studies or hypotheses are necessary.

6.17.5.15 Design Maturity Margins

The Design Maturity Margins account for unknown design aspects not yet identified at the level of this feasibility study. These provisions are not risk margins (i.e. cost impacts due to the realisation of a stochastic event) and must be considered as part of the total industrial cost as well as of the payload cost.

Design Maturity Margins:

- 10% for platform
- 20% for payload

6.17.5.16 Separation Device

The Separation Device is considered to be standard off-the-shelf equipment. Therefore no development activities or development models are taken into account in the cost estimate.

6.17.5.17 Launcher

Since the launch is planned to be a Soyuz-Fregat dual launch together with the Solar Activity Monitor (SAM), the designated costs are included and presented in the SAM report.

6.17.6 Cost Risk Estimate

No specific cost risk estimate has been performed. This will have to be accounted for as part of the ESA level contingencies.

6.17.7 Insurances

Due to the operational nature of the mission an insurance amount of 7.5% was considered, where this value is based on recurrent market prices.

6.17.8 Qualitative Cost Assessment

This estimate is based on a fully competitive environment with strong involvement and motivation of the Prime Contractor.

So far no reliability and availability figures have been expressed as part of the requirements of such an operational mission. Depending on these figures, the spares philosophy and components quality level may need to be revised.

6.17.9 Cost Breakdown

Due to the different distribution requirements, cost figures are not included in this report but in a separate document [RD5].

7. Solar Activity Monitor (SAM)

The Solar Activity Monitor is designed to provide near-real-time imaging of the solar disk (for solar flare and coronal mass ejection (CME) onset detection) and the corona (for detection of expanding CME material).

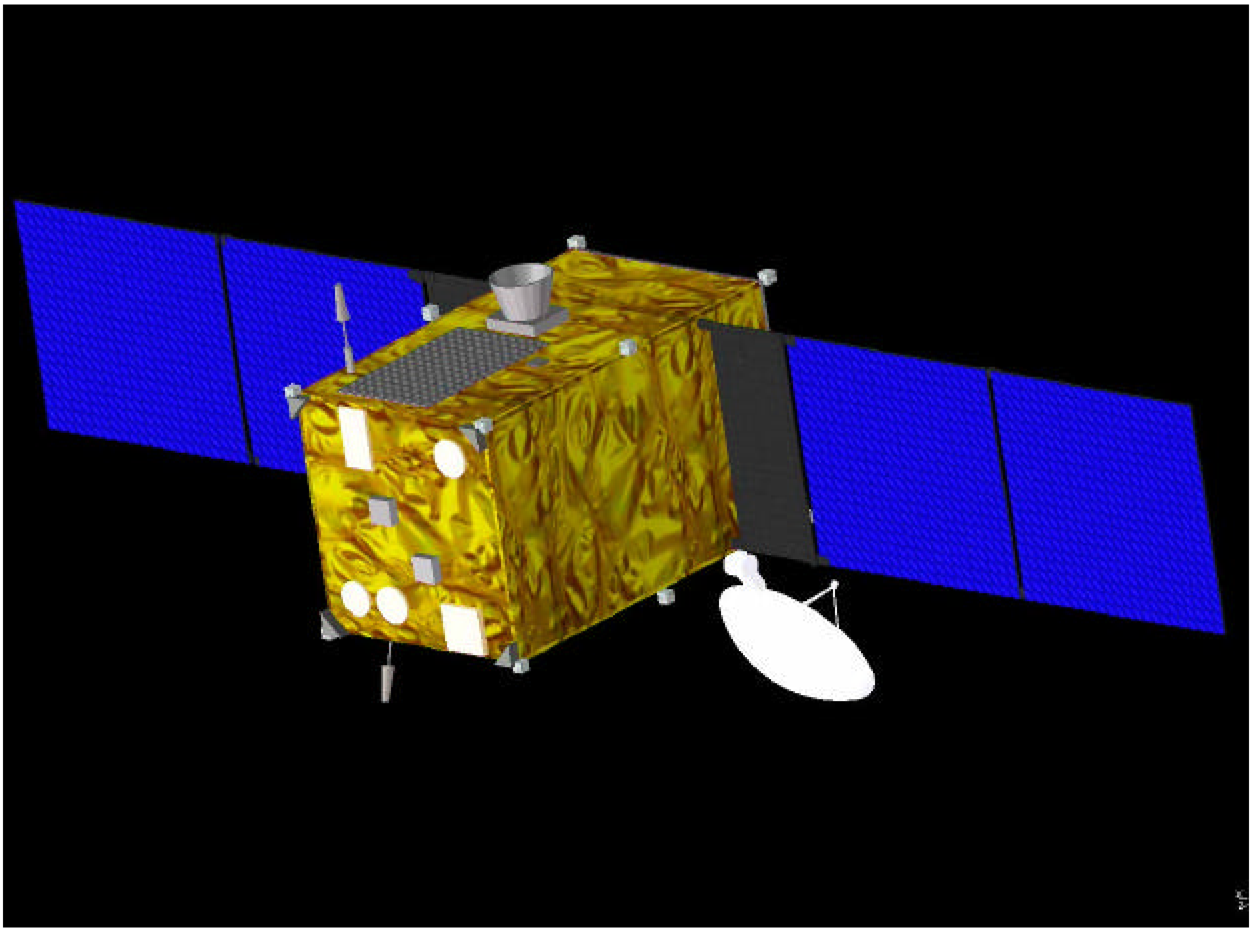


Figure 7-1: Solar Activity Monitor in Flight

7.1 Payload

The Solar Activity Monitor payload will enable identification of active regions and eruptive filaments on the Sun's surface, and follow-up of the evolution of the perturbations they generate, as these propagate through the outer layers of the Sun's atmosphere into the interplanetary medium.

The baseline SAM payload will monitor the magnetic field configuration on the solar disc through EUV imaging, and observe the visible light scattered by coronal mass ejections as they expand through the solar corona. In addition, any significant changes in the X-ray flux related to solar flares will be detected, and the energetic particle population upstream of the Earth's magnetosphere will be measured.

The following instruments have been selected:

- White Light Coronograph (WLC)
- Extreme UltraViolet Imager (EUVI)
- X-Ray Photometer (XRP)
- Cosmic Ray Monitor (CRM)

As with the IMM and SWM payloads, this set of instruments is considered to be the minimum payload sufficient to fulfil the user requirements relevant to such a space segment. The CRM could *a priori* have been installed on SWM or IMM, but it has been assumed to be more economical to have it on SAM.

Whenever possible, an enlarged set of instruments has been considered, but accommodation studies on this additional payload have not been performed.

7.1.1 Payload Requirements

7.1.1.1 White Light Coronagraph (WLC)

The cadence is required to be at least 1 image every 20 minutes, and an angular resolution of 28 arcsec per pixel is considered to be sufficient over a FOV extending out to 15 solar radii. The instrument should have an unobstructed view of the Sun at all times. The large integration times necessary for a good signal-to-noise ratio impose demanding requirements on the AOCS to prevent the blurring of the images. The required pointing accuracy is of the order of 5 arcseconds over 15 minutes.

Special precautions should be adopted while performing attitude correction manoeuvres, in order to prevent contamination of the optical system and detectors. The design should include a cover door that can be closed during such manoeuvres and during the launch and early operations phase.

The CCD detectors will require cooling to about -80°C, while the optical system will operate close to room temperature, i.e. -10/+20°C.

7.1.1.2 Extreme UltraViolet Imager (EUVI)

The EUV imager should be able to view the solar disc at all times with a 1-degree FOV. Observation in four spectral bands is required, these bands being centred in the following wavelengths:

- 195 Å (Fe XII)
- 304 Å (He II)
- 284 Å (Fe XV)
- 1216 Å (HI / Ly-α).

A cadence of one image every 2.5 minutes is required, together with an angular resolution of 10 arcsec per pixel. In addition, the minimum exposure time should be as short as 40 ms.

An instrument cover or a similar protective measure should be used when attitude correction manoeuvres take place, in order to prevent contamination of optical, thermal and detector surfaces.

The CCD detectors should be cooled to -80°C, while the rest of the system will operate at -10/+20°C.

7.1.1.3 X Ray Photometer (XRP)

The XRP detector should have a continuous unobstructed view of the solar disc. The angular FOV is less than 1 degree.

7.1.1.4 Cosmic Ray Monitor (CRM)

The CRM detector must be facing the Sun with an angular FOV of ±30 degrees, to include the so-called Parker Spiral (Interplanetary Magnetic Field configuration) direction. The detector must be in a 45 degree direction in the ecliptic plane with respect to the spacecraft axis i.e. the sun-pointing direction.

7.1.2 Payload Description

7.1.2.1 WLC Instrument

The design is partly based on that proposed for the SECCHI instrument that is due to be launched on the STEREO spacecraft in 2004. The main difference is that the Coronagraph and EUV imager units have been considered separately for SAM, responding to spacecraft configuration and structural demands. These were driven by the required stiffness of the stacked SWM/SAM configuration during launch.

The WLC instrument will be made up of two coronagraphs, COR1 & COR2.

COR1 will have a FOV covering 1.25-4 solar radii, and it will provide one image every 10 minutes. Using a 1024 × 1024 CCD will allow a high angular resolution (7.5 arcseconds) over this FOV. The mass of COR1 will be 6.2 kg (with a 20% margin, and excluding the detector) and it will be 1300 mm long and 146 mm in diameter.

COR2 will cover a radial FOV of 2 - 15 solar radii and will have a lower cadence (1 image/20 min). This element would have 7.4 kg mass (inc. 12% margin) and dimensions 1300 mm (length) x 132 mm (diameter). The angular size of the pixels would be 28 arcseconds in this case. The COR2 sensor will also be a front-illuminated 1024 × 1024 CCD. The possibility of using a polariser to improve the sensitivity (thereby reducing integration times) is also be worth considering.

The instrument specifications are as follows:

- Total instrument mass: 23 kg
- average power consumption: 20 W
- dimensions of the box containing both coronagraphs: 130 x 44 x 15 cm.

An important contribution to the mass budget will be the pointing platform, which will probably be required even if the present pointing accuracy and stability requirements for the spacecraft platform are met. Finally, with the cadences mentioned above,

- peak telemetry rate: 21 kbps.

7.1.2.2 EUVI Instrument

To enable observation in the four required spectral bands, the selected design is a TRACE-like single telescope with four mirror quadrants and optimised coatings. The optical system is a normal incidence multilayer Ritchey-Cretien telescope. A back-illuminated 256 x 256 CCD detector is enough to meet the 10” angular resolution requirement.

Resolution/cadence figures imply 10.5 kbps assuming 8-bit pixels and three (simultaneous) bands, and use of two modes of operation (quiet/active sun) to reduce the telemetry rate figure has been considered. The utilisation of these modes is however not critical as the telemetry rate figures are modest.

The instrument mechanisms include a telescope door, a quadrant selector, a filter wheel and a CCD shutter. The main reference for the design has again been the SECCHI instrument proposed for STEREO, but also TRACE and EIT-SOHO.

The spacecraft AOCS should be able to provide a pointing stability of at least 5 arsec at 0.01 Hz (1.5 min). Even if the limited instrument resolution leaves some room to relax this requirement a fine pointing system would probably be needed, especially if during later studies this pointing stability level turned out not to be feasible for the platform or not good enough for the observations.

The specifications are as follows:

- Total mass 15 kg
- power consumption 18 W
- box dimensions 100 x 20 x 20 cm

7.1.2.3 XRP Instrument

The proposed instrument will have a performance similar to – and be based upon – the GOES XRS sensor, which is part of the Space Environment Monitor package on the GOES spacecraft series of the US National Oceanic and Atmospheric Administration (NOAA). This instrument uses two ion chambers to allow real-time determination of the solar X-ray emission in two spectral bands: 0.5 - 5 Å and 1 - 8 Å.

The telemetry rate would be 0.1 kbps, considering that only two 12-bit numbers should be transmitted every 0.5 s, allowing 1% accuracy or better over the entire dynamic range. The available estimates for the mass and power figures are 16 kg and 16 W respectively.

7.1.2.4 CRM Instrument

This instrument is based on the STEREO IMPACT-SEP. A similar design was also used as baseline in the Solar Orbiter strawman payload. The original instrument design comprises five sensors, four of which look towards the nominal Parker spiral (which at 1 AU is at an angle of about 45 degrees from the Sun's direction, on the ecliptic plane), and one of which looks in the opposite (anti-spiral) direction. The latter extension to the FOV has not been considered in our baseline design.

A sensor should be added to look at higher-energy galactic cosmic ray particles (500 MeV and above). The view direction for this sensor is not as tightly constrained. Previous model instruments also include Ulysses' HET, but a lighter version would be needed.

7.1.3 Payload Mass and Power Budgets

Instrument name	Mass (kg)	Power (W)	TM rate (kbps)	Dim 1 (cm)	Dim 2 (cm)	Dim 3 (cm)	Heritage
White Light Coronagraph	23	20	21	130	44	15	Mod from STEREO-SECCHI, SOHO-LASCO
EUV Imager	15	18	10.5	100	20	20	Mod from STEREO-SECCHI SOHO-EIT, TRACE
X-ray Photometer	16	16	0.1	26	14	11	GOES-XRS
Cosmic Ray Monitor	6	4	2	20	20	20	Proposed for STEREO (IMPACT-SEP), Solar Orbiter
Totals	60	58	33.6				

Table 7-1: Power and Mass Budgets

7.1.4 Options for Future Study

Due to the great potential benefit for the Space Weather service, an extended set of instruments has also been considered. The additional instruments are the following:

- A hydrogen-alpha solar disk imager
  The H-α line corresponds to the first atomic transition in the neutral hydrogen Balmer series, and has a wavelength of 656.3 nm. This absorption line falls in the red part of the visible spectrum and is for observations of solar flares, filaments, prominences, and the fine structure of active regions.

- A soft X-ray telescope.
  This instrument would detect and locate flares for forecasting solar energetic particle (SEP) events related to flares, monitor changes in the corona that indicate coronal mass ejections (CMEs), detect active regions beyond the Sun's east limb, and analyse active region complexities for flare forecasts. Without this type of imager, only two figures representing whole-disk integrated solar X-ray activity will be obtained by SAM's X-ray photometer (XRP), rather than an image.

The accomodation of these instruments on the SAM platform and the implications for the design of the spacecraft subsystem were not studied during this exercise due to three main reasons:

1. Due to the high image cadence required for these instruments to be meaningful, their inclusion in the payload would have had a large impact on the design of other spacecraft subsystems (mainly on data handling, communications and power s/s) and on the ground segment definition. This would have lead to a major system re-design, and time constraints did not allow to study this option.

2. As the H-α line lies in the visible range, images of the Sun at this wavelength are available from ground-based facilities - though currently they cannot ensure a continuous coverage of the Sun at all weather conditions. On the other hand, no space-qualified H-α imager is presently available or has ever been flown, so this would also increase development time and costs.

3. The required soft X-ray telescope performance is very similar to that of the SXI instrument launched onboard NOAA's GOES 12 satellite, which became operational during September 2001. The images taken by this instrument are available to the worldwide user community immediately through the web.

In addition, the design and acommodation of the instruments that comprise the baseline payload would also need to be refined. For instance, it has been suggested that fine determination of the anisotropy of the high energy particle population would be desirable.This would have implications for the accommodation of the CRM, and maybe even require its deployment on a spinning platform, e.g. on SWM.

7.2 Mission Analysis

7.2.1 Mission Design

SAM is launched together with SWM and injected into the same transfer orbit to libration point L_1.

Separation between the two spacecraft occurs within a few hours after separation from the launcher upper stage. The separation mechanism will create a small difference in velocities that will prevent a collision. However, from a mission analysis point of view the two spacecraft are on the same orbit and it is intended to keep them flying in relative close formation. This allows the two spacecraft to be tracked by the same ground station dish.

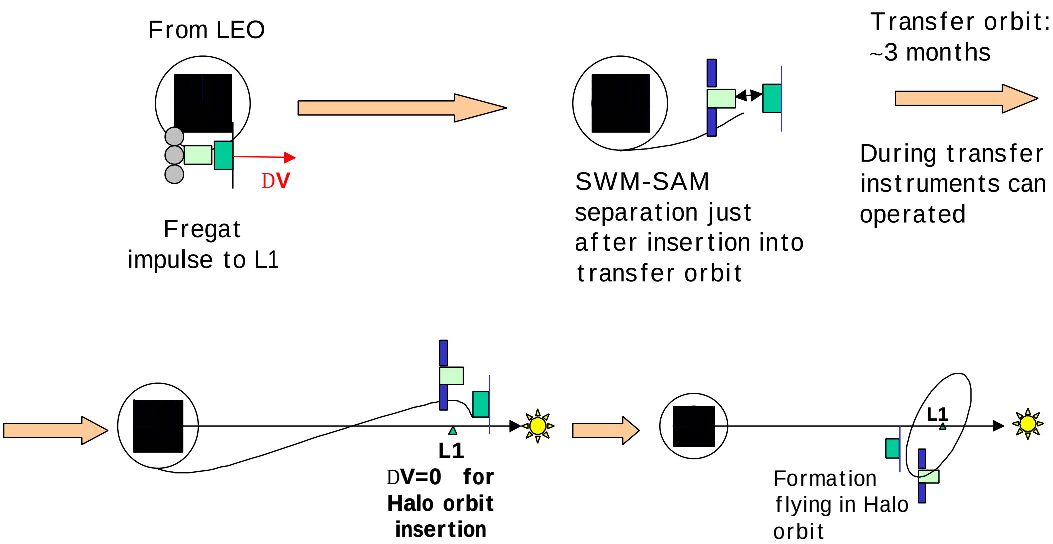


Figure 7-2: SAM Orbit Acquisition

Further mission analysis details can be found in the SWM report (section 6.2).

The remainder of this section contains details of the options discussion in chapter 4.

7.2.2 Option D: Earth Trailing Orbit

In this option SAM is located on a point trailing the Earth on its orbit around the Sun (Figure 7-3). Such a location allows a view of the Sun's surface features before they are visible from the Earth, and this slightly increases the detection efficiency of Earthward-propagating CMEs.

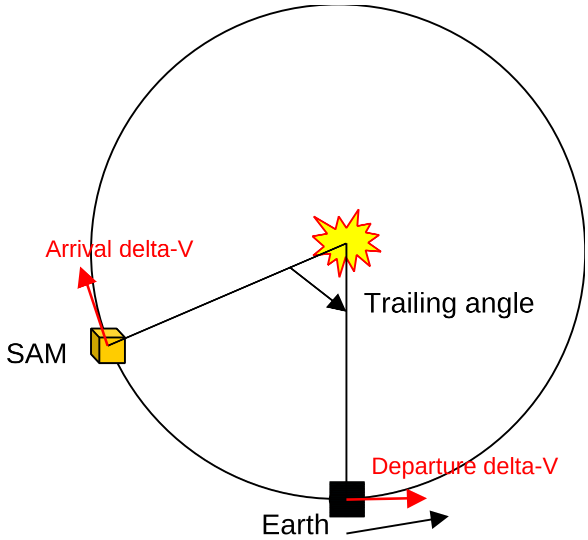


Figure 7-3: SAM Located on an Earth Trailing Orbit

Transfer from Earth to a trailing point is performed by injection into a transfer orbit that has a period of revolution slightly higher than the Earth period (one year). This is achieved by providing a velocity increment along the Earth velocity, increasing the orbit's aphelion to a value higher than 1 AU. Exactly one revolution later the spacecraft intersects the Earth orbit at the same point, which is now behind the Earth because time elapsed is more then one year. A braking manoeuvre of the same magnitude as the injection manoeuvre reduces the aphelion to one AU. The transfer orbit is represented on Figure 7-4 in a rotating coordinate system (x-axis along Sun-Earth line).

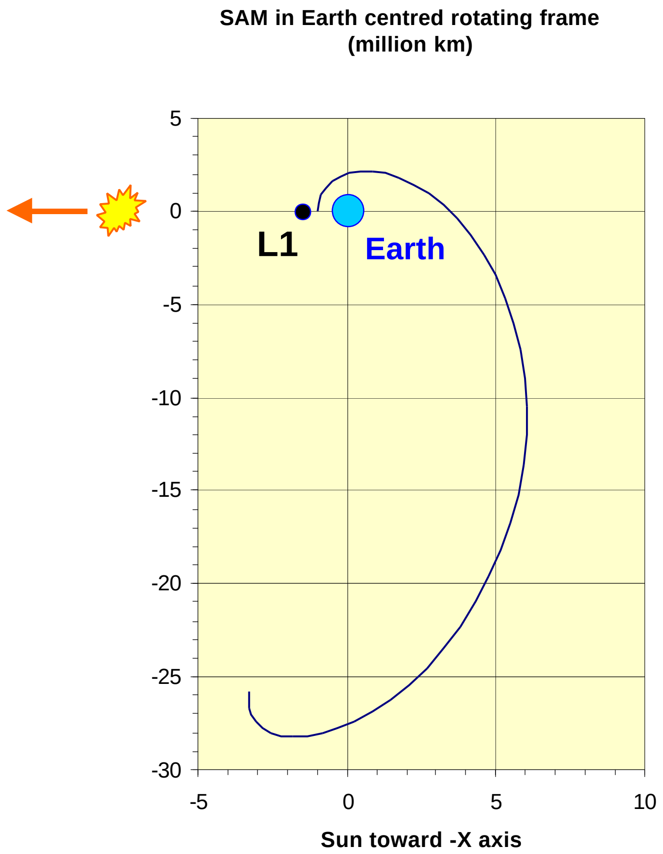


Figure 7-4: Transfer from L_1 to 10° Trailing Point on Earth's Orbit Around the Sun

The location of the trailing point is characterised by the trailing angle Earth-Sun-spacecraft. An angle of 10° is assumed for SAM. The magnitude of the *DV* for injection into corresponding transfer orbit is directly proportional to the trailing angle. For 10°, the *DV* amounts to 350 m/s, to be applied at injection and later again for insertion into the trailing point.

If SAM were directly injected into a transfer orbit to the trailing point instead of being first injected into a transfer orbit toward libration point L_1, the velocity increment would be considerably smaller. However the cost saving of launching both SWM and SAM on the same launcher would be lost. Therefore it is proposed to inject SAM into a trailing point transfer orbit in the vicinity of L_1, about one month after launch.

During transfer orbit the angle Sun-Spacecraft-Earth varies in a range between 0° and 180° (Figure 7-5). This angle is critical for the orientation of the on-board high gain antenna. It stabilises towards 90° when the spacecraft reaches the vicinity of the trailing point.

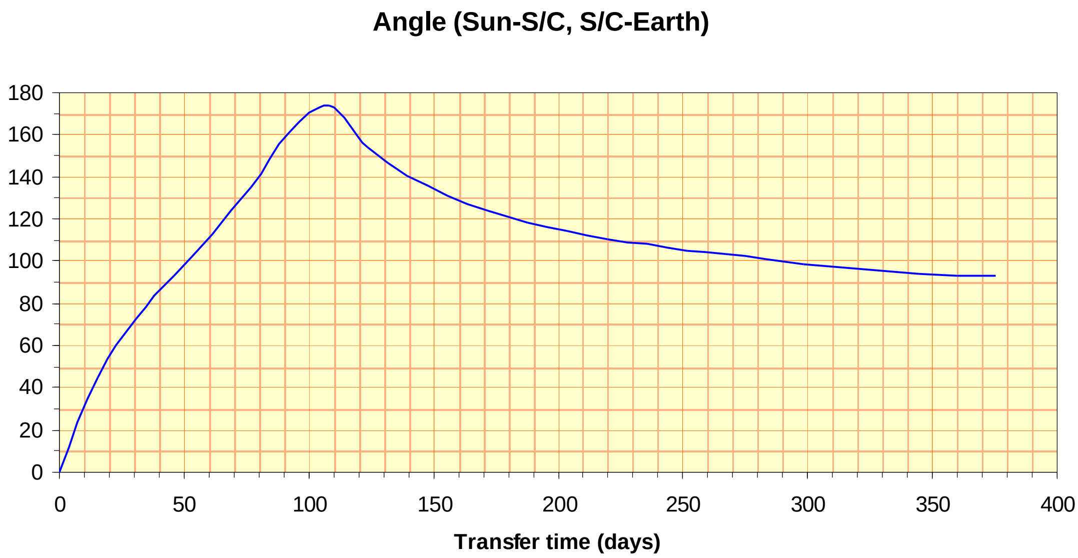


Figure 7-5: Angle (Sun-Spacecraft, Spacecraft-Earth) During Transfer to Trailing Point

The Sun-monitoring instruments can be used during transfer. However, there is a critical moment when the spacecraft is behind the Earth as seen from the Sun, namely in a penumbra region. During this period, the instruments must be switched off and the spacecraft has to be in a minimum power consumption mode.

The 10° trailing point is distant to the Earth by about 26 million km. The variation of the distance Earth-spacecraft during transfer is shown in Figure 7-6.

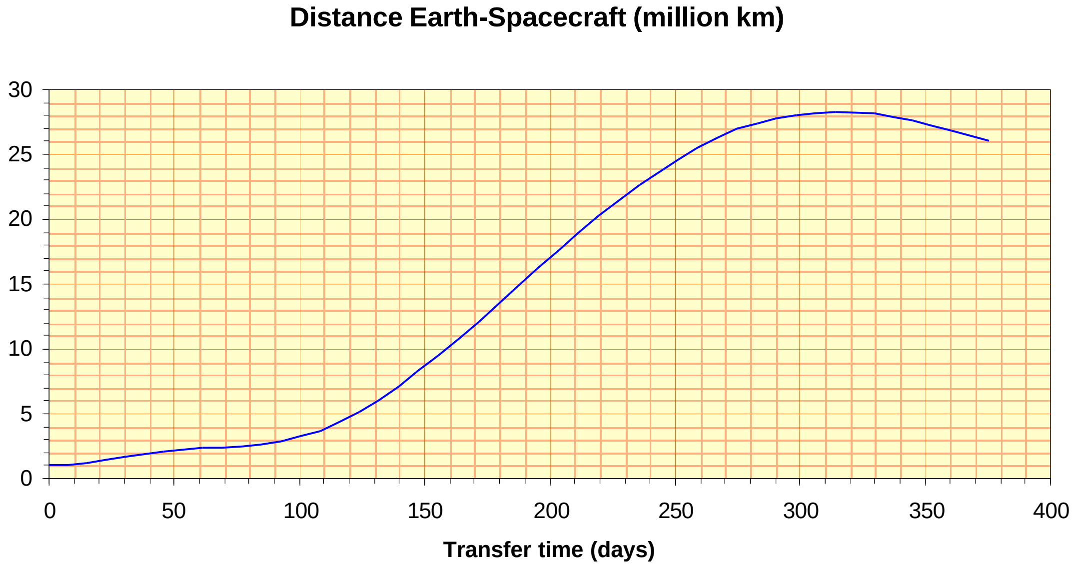


Figure 7-6: Distance Earth-spacecraft During Transfer to Trailing Point

Because of its inherently higher cost and complexity, this option has not been assumed as baseline. Among the critical mission analysis points that need further investigation are:

- Length of the penumbra phase during part of the transfer when spacecraft is behind the Earth as seen from the Sun.
- Size and frequency of the orbit maintenance manoeuvres for station keeping at the trailing location

7.2.3 Option C: SAM in GEO (Data Relay)

SAM does not necessarily need to be located on L_1 and could also be placed in a low Earth orbit environment. However, outside of high altitude Sun-Synchronous Orbits (SSO), no orbit guarantees eclipse free operations. SSO, like any LEO, cannot satisfy the requirement of continuous real time data coverage. Therefore only very high altitude orbits can be considered.

To take care of eclipses, two SAM spacecraft would have to be in orbit, with a phase difference such that one of the spacecraft is in solar visibility while the other one is in shadow. Among interesting orbits for such a 2-spacecraft constellation is the Geostationary Orbit (GEO): the two SAMs can be covered by only two dishes in one ground station. If these spacecraft are equipped with data relay functions, they could relay the data stream from IMM and SWM to the unique ground station. This means that the whole Space Weather satellite system would be covered by only two dishes located in one single ground station.

Such a neat concept would reduce the size and cost of the ground segment. The pros and cons of such a concept are listed below.

Pro:

- Simple and cost effective ground segment
- No need to have synchronous orbits for IMM
- No orbit maintenance manoeuvres for IMM
- Most of the time IMM would be covered also below 3000 km

Con:

- Two SAM spacecraft instead of one
- SAM has to include data relay functions: this is a large increase in spacecraft complexity
- Mission failure risk increase: malfunction of one of the SAM spacecraft would harm operation of all other spacecraft
- Due to higher orbit energy requirement, launch on GEO is more expensive than on L_1 orbit.

A trade-off was performed in the CDF (section 4.2.3) and the outcome was negative.

7.3 Radiation

Since SAM will be placed in a halo orbit around L_1, as will SWM, the radiation environment to which it will be exposed is identical to that of SWM.

Please see section 6.3 for more details.

7.4 Systems

7.4.1 Requirements and Design Drivers

The primary Solar Activity Monitor objective is to perform near-continuous imaging of the Sun disc and Corona in its most dynamic wavelength range.

The mission objectives and the operational characteristics of the Space Weather programme drive the system requirements and the constraints for the SAM:

- An orbital location with continuous and unobstructed view of the Sun
- Near real-time data flow
- Continuous coverage
- 3-axis stabilised Sun-pointing S/C with accuracy of 5 arcseconds
- Launch date of the first SAM in 2006
- 5-year lifetime

The design drivers for the system are:

- Payload dimensions (primarily the height of coronagraph ≈ 1.4 m)
- Halo orbit around L_1 (great distance to Earth)
- Stabilisation requirements

As already mentioned in section 6.4, the selection of the L_1 halo orbit rules out the need for a high-thrust propulsion system and simplifies the design of structures, thermal and power subsystems. However, unlike SWM, in this case the payload dimensions and attitude control requirements drive the design towards a slightly more complex architecture and a heavier spacecraft.

7.4.2 Assumptions and Trade-Offs

A preliminary analysis showed that the SAM mass would exceed the performance of both Rockot and PSLV launchers, leaving few opportunities for a cheap launch to a L_1 halo orbit. Therefore the possibility of sharing the launch with the SWM was considered in the system trade-off in the event that the orbit selected is the same for SAM as SWM.

A study baseline and some study options were chosen according to the trade-off tree shown below.

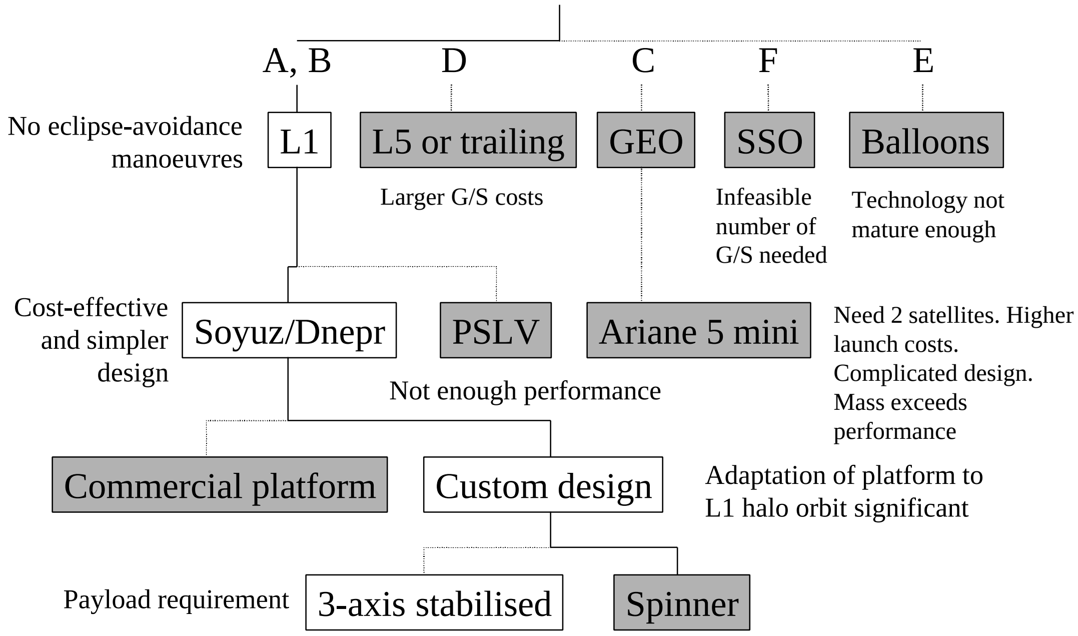


Figure 7-7: SAM Trade-off Tree

The trade-off was mainly based on Space Weather general architecture considerations and spacecraft design considerations.

A number of options were studied, but no system models for the different options were made. In particular, the following set of system options were evaluated:

	Option A	Option C	Option B	Option D	Option E	Option G	Option H
Mission							
Number of Satellites	1	2	1	1	Several balloons	1	1
Orbit	L1	GEO	L1	L1/10deg trailing	35 Km altitude - polar	L1	L1
Launch Date	2006	2006	2006	2006	2006	2006	2006
System							
Satellite Type/Platform	Custom	GEO (&data relay)	Combined SWM & SAM	Custom	balloon	LEO platform	Custom
Dry-mass class	1000	1000	1500	1000	1000	500	1000
Stabilisation	3-axis	3-axis	3-axis	3-axis		3-axis	3-axis
Payload							
Instrument Set	nominal	nominal	nominal	nominal	nominal	nominal	nominal
Launcher							
Launcher	Soyuz-Fregat dual	Ariane 5 dual	Soyuz-Fregat	Soyuz-Fregat dual		Soyuz-Fregat dual	Dnepr-Varyag dual
Launch Strategy	Direct	GTO+own prop	Direct	L1 + trailing own prop		Direct	
Propulsion							
Type of Propusion	No main prop.	bi-prop	No main prop.	bi-prop	No main prop.	No main prop.	No main prop.

Table 7-2: SAM System Options

The first five options are discussed in section 4.2. The last two are variants of Option A, considering a different S/C design or a different launcher.

Option A, an independent spacecraft at L_1, was chosen as baseline and will be discussed in detail in section 7.4.3.

Option B is a design with one spacecraft combining SAM and SWM by means of a despun platform. This option is discussed in section 7.4.5.1.

Option C involves the use of the Ariane 5 mini-sat launch opportunity and using 2 SAM satellites in GEO which also serve as data-relay satellites for IMM and SWM. This option is discussed in section7.4.5.2.

Option D was preferred in terms of observation requirements and is different from the baseline in terms of orbit: the orbit selected is a heliocentric, Earth-trailing orbit, 10° behind the Earth. This option is discussed in section 7.4.5.3.

Option E replaces the satellite by a number of balloons at 35 km altitude on both poles of the Earth. This option is discussed in section 7.4.5.4. It was discarded due to the atmospheric balloon technology being insufficiently mature.

Option F was SAM at SSO.

Option G is an L_1 spacecraft based on a commercial platform, but was not studied in detail due to the significant changes that would have to be made to platforms which are usually LEO oriented.

Option H is similar to the baseline design, but with Dnepr-M/Varyag as launcher in case this is available.

The study flow led to one version of the design model being created, namely:

- Version 1.12: Completed iteration of the baseline design.

7.4.3 Baseline Design (Option A, SAM at L_1)

The design of SAM takes advantage of the SOHO expertise, especially concerning the AOCS and propulsion subsystems. No conflicting requirements have been found among subsystems.

The thermal subsystem must cope with the requirements of cryogenic payloads (WLC and EIVI) for which passive rejection devices are used (no coolers).

The telecommunications system accounts for the largest share in the power budget of the S/C, and the payload data rate imposes the use of an HGA that must be made steerable.

As baseline, SAM would be designed for a dual-launch with SWM on a Soyuz-Fregat. The Soyuz-Fregat launcher was chosen for availability reasons; however, should the Dnepr-R/Varyag be available, this launcher could be chosen for its low launch costs. This was chosen because it satisfies the user requirements, can be launched relatively cheaply, has good communications features, and offers the required stability.

7.4.3.1 Modes of Operation

The system Modes of Operation for the SAM mission are shown in the following table:

Number	Mode Name	Definition	Acronym
1	Launch Mode	***From lift-off until upper stage separation*** ***Including LEOP*** Battery fully charged (charging until 8 min (tbd) before lift-off) Payload Instruments switched off Power S/S (PCU,PDU,TCU) , OBDH S/S (CDMU) switched on Comms S/S for RX switched on SAM solar array is deployed at the end of the parking orbit	LM
2	Transfer mode	***From stage separation until Halo orbit*** Payload Instruments may be switched on AOCS 3-axis stabilised Solar Arrays pointing to the sun TT&C two ommidirectional antenna S-band + HGA +downlink of P/L data (if any) with X-band Antenna deployment?	TM
3	Initialisation Mode	***From Halo Orbit adquisition until normal operation*** Mode entered after transfer phase or during recovery from safe phase After spinning and ejection of SWM alone SAM life starts HGA pointing to Earth Payload Instruments initialisation Fine pointing	IN
4	Operational Mode	***Fine Pointing Mode*** TT&C Active (HGA for continuous downlink) Payload Operational S/C Sun Pointing AOCS active and satisfying the pointing requirements P/L data not stored on board	OP
5	Safe Mode	***Failure Recovery Mode*** S/C attitude automatically set to Sun Pointing Payload Instruments switched off Failure Detection, Isolation and Recovery to normal mode are executed by the ground. TT&C Active via Low Gain Antenna TM/TC access to OBDH is guaranteed to enable failure detection and reconfiguration.	SM

Table 7-3: Modes of Operation for SAM

7.4.4 Budgets

7.4.4.1 Mass Budget

The mass identified in the system budget is based on the specified values of the individual units and subsystems. Depending on the maturity status of the items, contingency is applied at unit/item level. Generally, for each piece of equipment a mass margin is applied in relation to its level of development, i.e.:

- 5% for off-the-shelf items
- 10% for items qualified but requiring some modification
- 20% for items to be developed

The payload margin is zero because all figures obtained included margin.

A system-level mass margin of 20% is placed on the spacecraft dry mass (dry mass including sub-system margins). The S/C mass budget for the baseline is displayed in the table below.

Solar Activity Monitor Mass Budget					
			Target Spacecraft Mass at Launch	1300	kg
			Below Mass Target by:	762	kg
	Without Margin	Margins %	Margins kg	Totals kg	% of Total
1. Structure	90.5 kg	20.0	18.1	108.6	20.17
2. Thermal Control	12.1 kg	10.0	1.2	13.3	2.46
3. Mechanisms	20.6 kg	10.0	2.1	22.6	4.20
4. Pyrotechnics	0.0 kg	0.0	0.0	0.0	0.00
5. Communications	24.0 kg	10.0	2.4	26.4	4.90
6. Data Handling	10.0 kg	10.0	1.0	11.0	2.04
7. AOCS	42.2 kg	10.0	4.2	46.4	8.62
8. Propulsion	18.0 kg	10.0	1.8	19.8	3.68
9. Power	36.1 kg	10.0	3.6	39.8	7.39
10. Harness	8.9 kg	10.0	0.9	9.8	1.82
11. Payload Allocation	60.0 kg	0.0	0.0	60.0	11.15
Total Dry (excl.adapter)	322.29 kg			357.6	66.43
System Margin (excl.adapter)		20.0 %		71.5	
Total Dry with Margin (excl.adapter)				429.1	79.71
		Propellant:	Total propellant	59.2	11.00
				0.0	
			Adapter Mass (incl. Sep. Mech.)	50.0	9.29
Total Launch Mass				538	

Table 7-4: SAM Mass Budgets

7.4.4.2 Power Budget

Five operational modes have been considered as dimensioning for the design of the power subsystem. The corresponding S/C power demands are given in the table below.

POWER CONSUMPTION BUDGET vs MODE

Mode names are linked		Instr.	Thermal	AOCS	Comms	Propulsion	OBDH	Power Cons.	Pyro	Mech	Harness (excl. PSS)
		linked	manual	manual	manual	manual	manual	computed	manual	linked	computed
Launch Mode	MAX	0	15	10	10	0	10	33	0	0	0.9
	NOM	0	15	10	10	0	9	27	0	0	0.9
	MIN	0	15	10	10	0	9	22	0	0	0.9
Transfer mode	MAX	0	27	110	70	0	10	33	0	15	4.6
	NOM	0	27	100	70	0	9	27	0	8	4.3
	MIN	0	27	90	70	0	9	22	0	0	3.9
Initialisation Mode	MAX	58	27	110	70	0	10	33	0	15	5.8
	NOM	55	27	100	70	0	9	27	0	8	5.4
	MIN	50	27	90	70	0	9	22	0	0	4.9
Operational Mode	MAX	58	27	110	70	0	10	33	0	15	5.8
	NOM	55	27	100	70	0	9	27	0	8	5.4
	MIN	50	27	90	70	0	9	22	0	0	4.9
Safe Mode	MAX	0	15	110	70	0	10	33	0	15	4.4
	NOM	0	15	100	70	0	9	27	0	8	4.0
	MIN	0	15	90	70	0	9	22	0	0	3.7

Table 7-5: SAM Spacecraft Power Consumption

7.4.5 Options

Four options were evaluated as discussed below.

7.4.5.1 Option B: Combined SAM and SWM

Two designs are possible in the event that a combined SAM/SWM S/C is considered for cost reasons:

1. A spinning S/C with a despun platform for the fine pointing instruments of SAM
2. A 3-axis stabilised S/C with SWM payload adapted to fulfil the scanning requirements

Concerning the first, no European despun platform was found suitable for this application. Therefore a dedicated technology development would be required. In addition, due to the large dimensions of the payload, configuration problems are anticipated.

The second possibility would thus be preferred. However, a degradation in the payload performance would have to be accepted and need to be negotiated. Should cost reduction be required, this option may be further investigated.

7.4.5.2 Option C: GEO Data Relay

An option was studied in which SAM would consist of two spacecraft located at GEO altitude. Two spacecraft are needed to fulfill the observation requirements for SAMbecause each would have an eclipse phase. Both SAM spacecraft work not only as Solar monitors but also as Data Relay systems for SWM and all IMM satellites.

The spacecraft design should both include features of a Telecomms satellite (propulsion to GEO and station-keeping, large steerable antennas, large power demand) and accommodate the SAM payload (large dimensions, cryogenic radiator, demanding pointing requirements, etc.).

Though a detailed configuration study has not been performed, the combination of the two sets of requirements will result in a complex and large satellite with mass in the order of 1000 kg.

As mentioned in section 4.2.3, this option does not give real advantages from a system architecture point of view if the baseline Space Weather system includes only the IMM, SWM and SAM missions. Therefore no full design iteration was performed.

It should be noted, however, that as part of a more complex constellation of space weather measurements including low altitude polar orbiting S/C, the inclusion of a relay satellite should be reconsidered.

7.4.5.3 Option D: SAM in Trailing Orbit

Different strategies are possible for launching the SAM into a 10° Earth trailing orbit, as summarised below.

Option	Delta-V	Cruise Time	Remarks
Launch to L1 with SWM, return to Earth, swing-by and cruise to 10-deg trailing orbit	500 m/s	~2 yrs	SAM must carry own propulsion (high thrust needed)
Launch to L1 with SWM, from trajectory to L1, SAM is separated and with its own propulsion reaches the 10-deg trailing orbit	700 m/s	~1.5 yrs but during part of the transfer payload can be operated	SAM must carry own propulsion but low thrust can be enough (no need for main engine)
SAM+SWM to LEO, separation, SAM to Trailing orbit with own propulsion and SWM to L1			Not feasible because of the SAM+SWM stack on a single launch (SWM cannot be kicked to L1 By the upper stage)

Table 7-6: Launch Strategies to Trailing Orbit

The second option would be preferable since it avoids the need for a main propulsion system for SAM.

As far as the spacecraft design is concerned, the following modifications to the baseline have been identified:

- During cruise to the 10° trailing orbit the spacecraft experiences a penumbra phase. This has not been studied in detail but the impact on the power and thermal systems may be large, depending on the duration of this phase
- The telecomms system on board must be designed to cope with the very large distance to Earth (≈ 25 million km) and with the high variability of the angle to Earth during cruise (up to 180° steering) (see section 7.13)
- The ground-segment would be as needed for 100% availability in a deep space mission using either X- or Ka-band (see section 7.13)

Therefore, this option is feasible with modifications of the baseline design, which would increase either the complexity or the cost of the system.

7.4.5.4 Option E: Balloon Option

In this option, the payload would be carried on balloons flying at altitudes up to 35 km around the poles. Note that this platform would only be possible for x-ray, visible, and coronagraph measurements, because of atmospheric absorption.

The balloon gondola configuration consists of a rotator, structure, body-mounted solar array, Support Instrument Package (SIP) and ballast hopper (shown in Figure 7-8). The maximum allowable science payload, with the current technology, is up to 1800 kg (depending on the launch site).

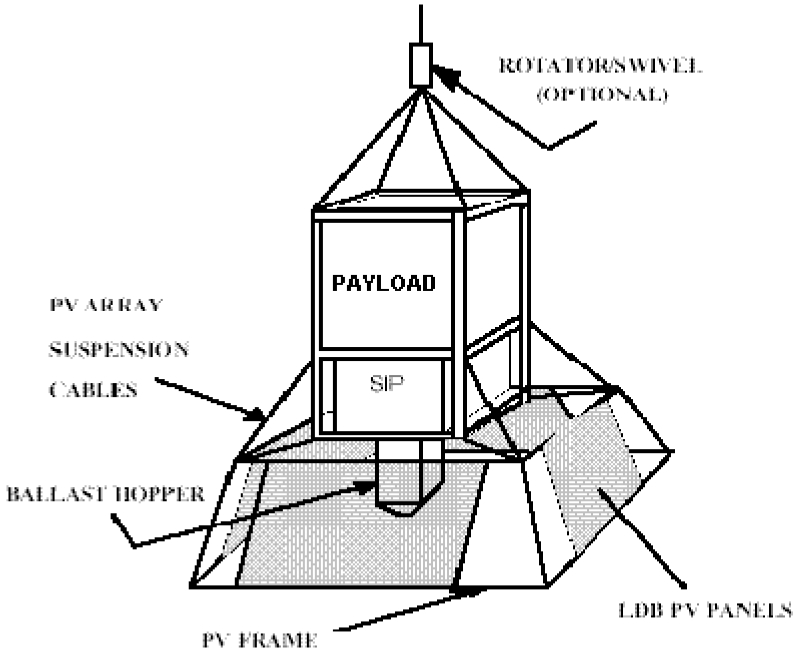


Figure 7-8: Gondola Configuration

The rotator provides one degree of freedom on the camera pointing. Further degrees, if necessary, should be contained within the payload. The Support Instrument Package (SIP) provides all functions which would by given by the Service Module in a conventional spacecraft. Power is provided by the body mounted solar array (PV panels).

The instruments would be flown for as long as the performance of the balloon allows. This performance is limited by the use of ballast that must be released in order to compensate for altitude losses due to thermal fluxes in the balloon gas. The mission is over as soon as there is no more ballast left. With the current technology, the lifetime of the mission on a regular basis is up to 12 days. Thus, in order to carry out continuous monitoring of the Sun it would be necessary to launch around 30 balloons each year.

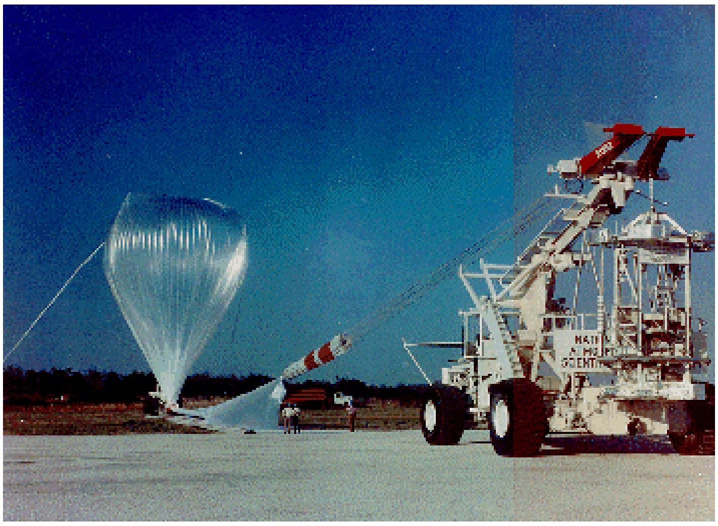


Figure 7-9: Balloon Launch

Ground operations must cope with extreme conditions during the whole lifetime, from launch till recovery.

The ballons should be recovered (in order to reuse the payload), refurbished and re-launched. The recovery of the payload for further reuse should be analysed carefully because of the severe conditions of this recovery in polar zones.

Moreover, due to the lack of suitable ground stations available at those latitudes, the downlink of data is performed through data relay satellites (INMARSAT and TDRSS). Therefore this option would require either the building of ground stations suitable for continuous downlink (at both polar regions), or the use of a data relay satellite. In this last case this implies to establish a link from a position close to the pole at low altitude to

Consequently, this option was discarded because this technology was found not to be mature enough for continuous Sun monitoring. It could be reconsidered once the technology development allows balloons lasting > 100 days.

7.4.6 Conclusions and Open Points

In order to fulfil the SAM requirements, a SOHO-type design has been proposed. Cost reduction has been the main driver in all subsystem designs, with the emphasis on reuse of components from earlier ESA missions or other Space Weather missions.

7.5 Configuration

7.5.1 Requirements and Constraints

The major drivers for the overall configuration can be summarised as follows :

- 3-axis stabilised satellite with deployable solar panels
- Accommodation of the payload instruments; Coronograph, EUV Imager pointing towards the sun
- Accommodation of 2 propulsion tanks of 0.45m diameter
- Accommodation of one High Gain Antenna using a fine pointing mechanism (2-axis rotation)
- Accommodation of electronic boxes for Data Handling, Power, Communication
- Stable mounting and accessibility should be guaranteed
- Compatibility with Soyuz-Fregat fairing type S payload envelope, to accommodate a stack of SAM together with SWM

The spacecraft must provide accommodation to all the sub-systems and ensure compatibility between them throughout the mission. Therefore each of the constraints as listed above must be fulfilled for every operational mode and Sun-Earth S/C attitude.

7.5.2 Spacecraft Baseline Design

The configuration is driven by the system requirements together with the size of payload instruments.

The resulting overall dimensions of SAM are:

- 1.9 m height (from tip of LGA length to separation line between SAM and SWM)
- 1.3 m in x-direction (stowed solar panel)
- 1.3 m in y-direction (from tip of LGA to tip of HGA stowed)

Figure 7-10 and Figure 7-11 show stowed and deployed configurations respectively of the SAM spacecraft.

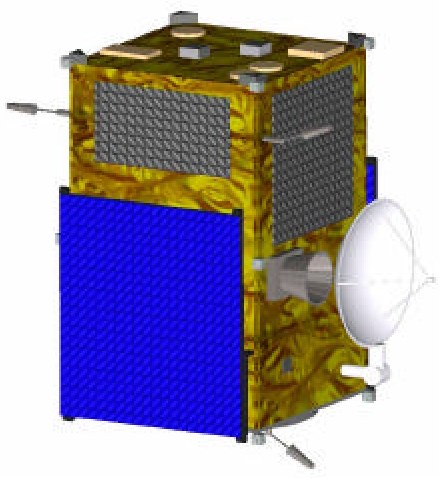


Figure 7-10: SAM Stowed Configuration

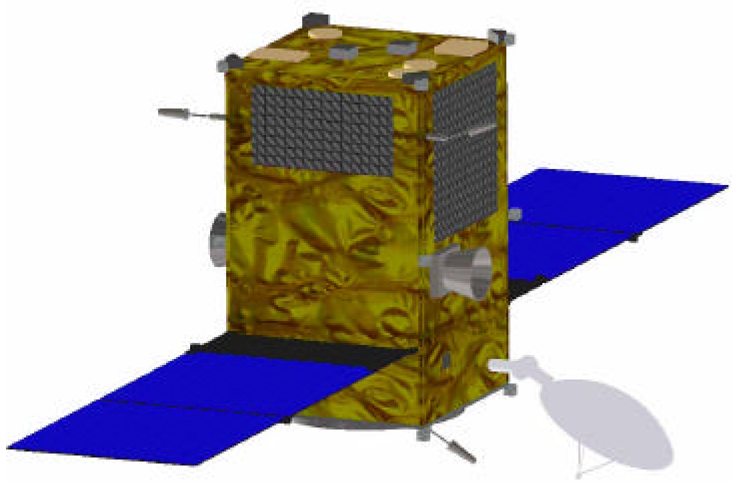


Figure 7-11: SAM Deployed Configuration

The S/C body is stiffened by a solid bottom plate (1 m by 1m), three lateral panels (1.7 m height), and four outer panels. Solar panels are mounted on the side panels (+X and –X).

Figure 7-12 shows the internal accommodation in the SAM spacecraft. The accommodation is summarised in Table 5-10.

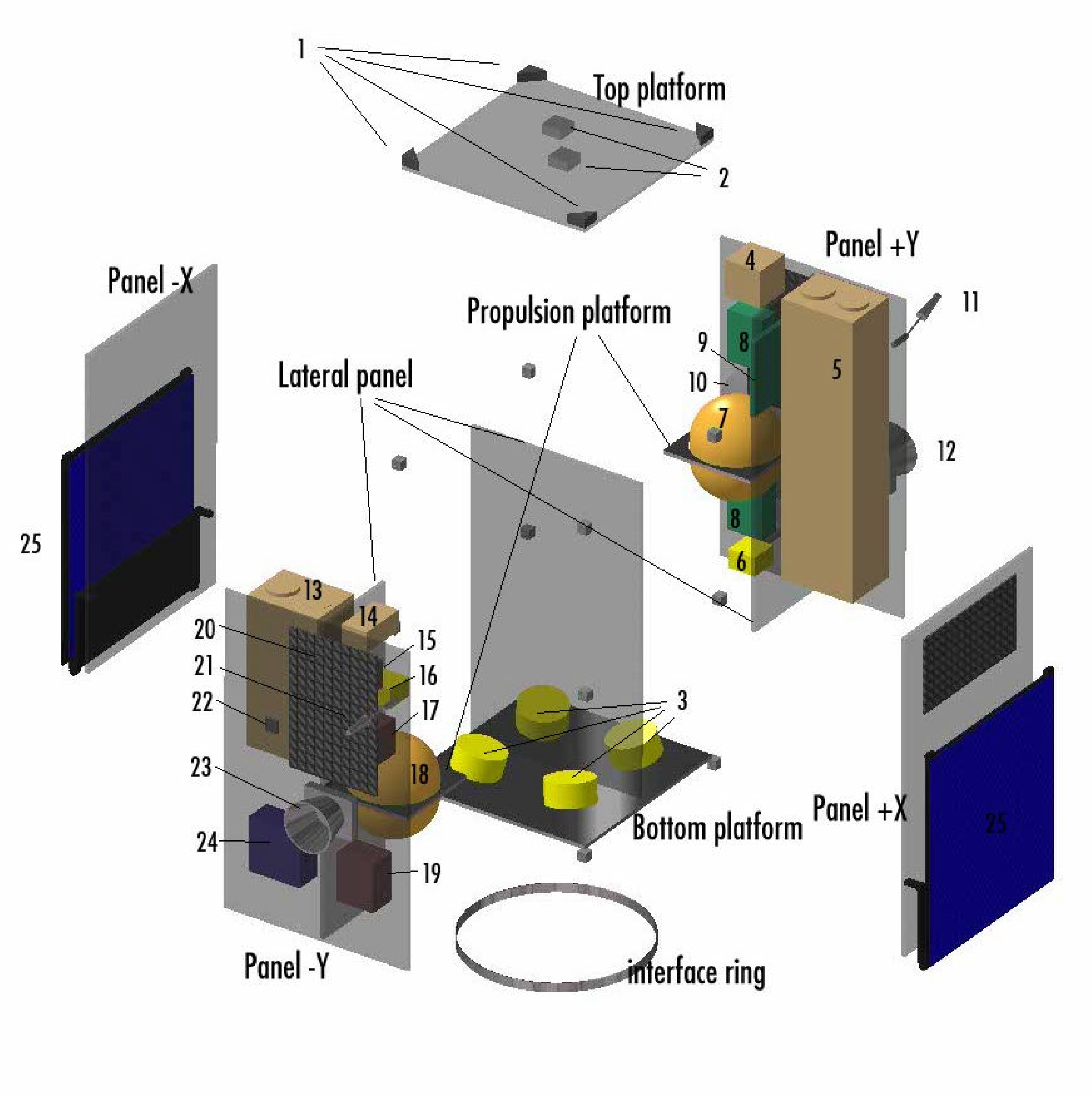


Figure 7-12: SAM Units Accommodation

Domain	No.	Unit	Location
AOCS	12, 23	2 x Star Tracker	Panel –Y/+Y
	2	2 x Sun Acquisition Sensor	Top Platform
	3	4 x Reaction wheels	Bottom platform
	6, 16	Gyro	Lateral panel

Domain	No.	Unit	Location
Telecomms	9	RFDU	Panel +Y
	8	2 transponders	Panel +Y
	11, 21, at the bottom	3 Fixed Low Gain Antennas	Panel –Y/+Y/bottom platform
	10	Deployable High Gain Antenna	Panel +Y
DHS	24	CDMU	Panel –Y
Power	15	PDU	Panel –Y
	17	PCU	Panel –Y
	19	Battery	Panel –Y
	25	Solar cells	Panel +X, -X
Thermal	20	Radiator	Panel –Y/+Y
	22	Cryogenic radiator	Panel –Y/+Y
Instruments	4	Cosmic Ray Monitor	Panel +Y
	5	Coronagraph	Panel +Y
	13	EUV Imager	Panel –Y
	14	X-Ray Photometer	Panel –Y

Table 7-7: Unit Accommodation

The payload accommodation is illustrated below.

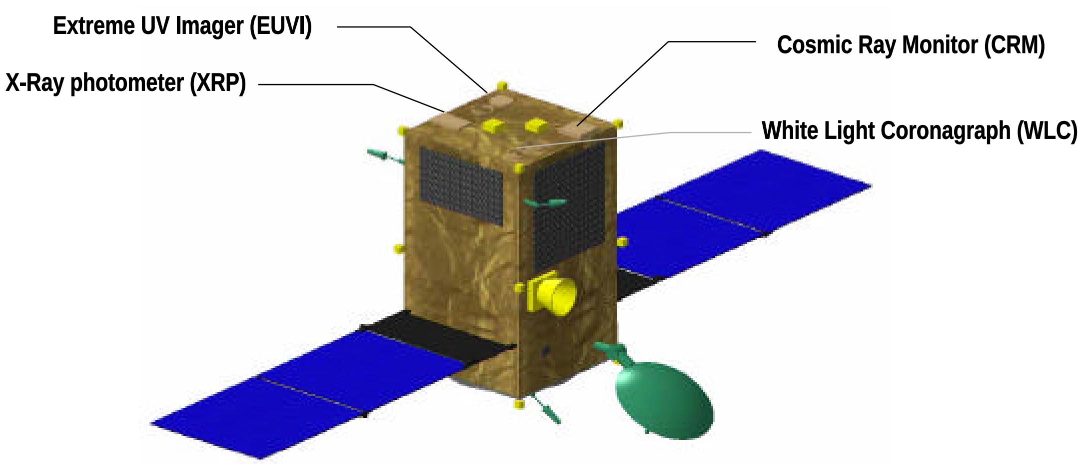


Figure 7-13: SAM Payload Accommodation

7.5.3 Launch Configuration

The following figure shows the SAM spacecraft stacked with SWM in the Soyuz fairing type S.

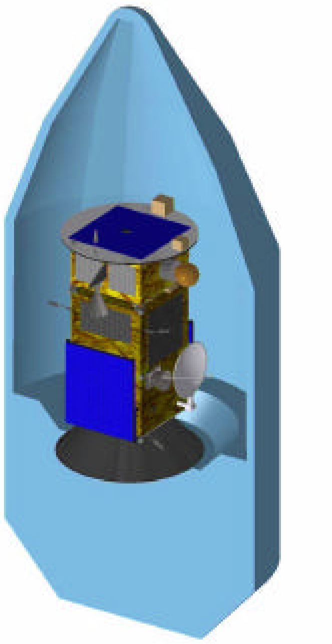


Figure 7-14: SAM and SWM Stacked in Soyuz Fairing

7.6 Propulsion

7.6.1 Subsystem Requirements and Design Drivers

The Soyuz-Fregat launcher puts the SWM and SAM spacecraft directly into the L_1 halo orbit.

The spacecraft is 3-axis stabilised and corrects its attitude upon release from the launcher. For manoeuvring the spacecraft during its operational life and unloading of reaction wheels, about 48 kg of hydrazine is required.

There will be no main engine. The propulsion system is able to accomplish orbit manoeuvres along any direction, both during transfer (launcher dispersion corrections and mid-course manoeuvres) and on operational orbit (orbit maintenance).

A 40 m/s velocity increment is required for correction of launcher dispersion (section 6.2.2.2), plus another 5 m/s for mid-course manoeuvres and possible halo insertion adjustment. Another 10 m/s velocity increment is required for orbit maintenance over five years (section 6.2.3.2).

7.6.2 Subsystem Baseline Design

The direct injection to the L_1 orbit by the launcher leaves only a very small DV requirement on the spacecraft side. The propulsion system remains necessary, though, for this launcher dispersion correction, attitude control and halo-orbit maintenance.

Since the DV requirement is small, a simple mono propellant hydrazine system is sufficient. The system used on SOHO serves as an example for the propulsion sub-system design.

In Figure 7-15 the schematic diagram of the propulsion system is depicted. The system comprises two propellant tanks in which the propellant is expelled by a propellant management device (diaphragm). The system operates in blow-down mode with Helium as pressurant gas. Two branches of eight 5 N thrusters, two fill and drain valves, two propellant filters, two propellant isolation latch valves, three pressure transducers and some thermistors and line heaters complete the system.

Two barriers between propellant tank and thrusters are used, assuming that this complies with the Soyuz-Fregat launch vehicle requirements.

The proposed propulsion system design is, to a large extent, based on available COTS components and not on optimised components that may need to be developed. Substantially larger costs would have to be taken into account if a specific tank development were required. From a purely engineering perspective, the proposed design solution is likely not the optimum one, at least not from a dry mass point of view. The mass penalty for the non-optimised design was not traded-off at this stage against the potential cost advantage of using COTS tanks. For the time being it is assumed that the cost advantage of the proposed design outweighs the mass penalty.

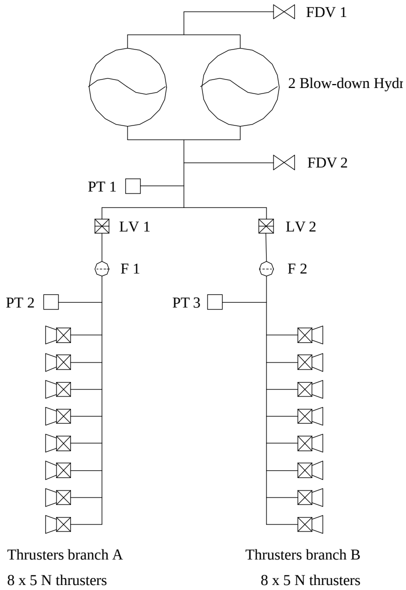


Figure 7-15: Propulsion Subsystem Block Diagram

Note: in recent years there has been a tendency to move away from creating branches with thrusters that could be closed in the event that one thruster in the branch leaks. Using dual-valve thrusters instead of mono-valve thrusters is a solution. The two latch valves could then be replaced by one latch valve, which would also create an extra barrier between propellant tank and thrusters. This approach could be taken into account in the phase A/B design.

7.6.3 Budgets

A mass breakdown of the components of the propulsion system can be found in Table 7-8.

Component	Qty	Unit mass (kg)	Total mass (kg)
Propellant tank	2	5.58	11.16
Pipework	-		1.5
Thrusters	16	0.22	3.52
Fill/drain valves	2	0.05	0.1
Filter	2	0.28	0.56
Pressure transducer	3	0.25	0.75
Latch valve	2	0.2	0.4
Pressurant Helium	-		0.03
Propulsion system dry mass			18
Propulsion system dry mass with 5% margin			18.9
Propellant (incl. 1 % residuals)			59.3
Propulsion system wet mass			78.2

Table 7-8: Propulsion Subsystem Mass Budget

7.7 Thermal Control

7.7.1 Requirements

The spacecraft thermal control is required to keep the temperatures of the spacecraft subsystems and the instruments within specified temperature limits during all expected mission phases and operation modes. The temperature limits have been assumed as follows:

	Operational	**Non-operational**
Instruments general	-10°C/+20°C	-20°C/+55°C
CCDs (WLC and EUVI)	≤ -80°C	-80°C/+55°C
S/C electronics	-20°C/+40°C	-30°C/+60°C
Batteries	+20°C/+35°C	0°C/+40°C
Propulsion system	+5°C/+15°C	+5°C/+15°C

Table 7-9: Temperature Limits

7.7.2 Baseline Design

The Thermal Design philosophy used for the SAM is based on the use of passive techniques (MLI, OSR, etc.), with the addition of heater power for the propulsion system in order to keep the hydrazine propellant above the required minimum temperature.

Particular measures are required to provide the low operation temperatures at the CCDs of the coronagraph and the EUV-monitor. For both instruments a CCD surface area of (5.5 x 5.5) mm² has been assumed which is entirely exposed to solar irradiation. For both instruments, the *effective* CCD area exposed to solar irradiation has been assumed to be 30 mm^2.

The solar fluxes during operational phases in the L_1 halo orbit are almost constant at 1400 W/m^2. Earth and albedo fluxes are negligible.

The particular features of the Thermal Design can be summarised as follows:

- Multi-Layer Insulation (MLI) Blankets and double foil trimmed as necessary to have a better heat rejection to deep space and therefore to minimise heat absorption from solar irradiation. The blankets comprise aluminised Mylar and/or Kapton sheets and an electrically conductive outer sheet or laminate grounded to the S/C structure in order to prevent electrostatic discharge.
- Radiator surfaces covered in black paint in order to radiate the S/C internal heat dissipation to deep space. The total required radiator surface is 1.5m^2, based on the power dissipation budget. The radiators do not need to be insulated from the S/C structure and are therefore provided by cut-outs in the MLI. Subsystems with a higher power request shall predominantly be mounted in contact with the radiators.
- Two separate small OSR radiators maintain the very low operational temperatures required by the CCDs (of the coronagraph and the EUV-monitor). Each of these radiators has a

surface area of 0.01 m². The two instruments are located next to the side panels of the S/C, with the radiators situated on the S/C outside wall close to the CCDs. The CCDs are connected to the radiators by means of heat pipes. The two radiators are insulated from the S/C structure.

- S/C internal surfaces shall generally have a high emittance finish to enhance radiative heat transfer and to minimise the temperature gradients within the S/C. Therefore all aluminium internal surfaces are black painted.
- The rear sides of the solar arrays are black painted in order to act as radiators.
- The Solar Arrays are thermally de-coupled from the S/C structure to minimise the heat input into the S/C structure.
- To maintain the required temperatures on the propulsion S/S (tanks and valves) and the batteries, they are thermally insulated from the S/C internal environment.
- The required minimum temperatures of the propulsion S/S and the batteries are maintained by several heater lines providing a heater power of 15W and 12W, respectively. Heater control is performed by thermostats at element level.

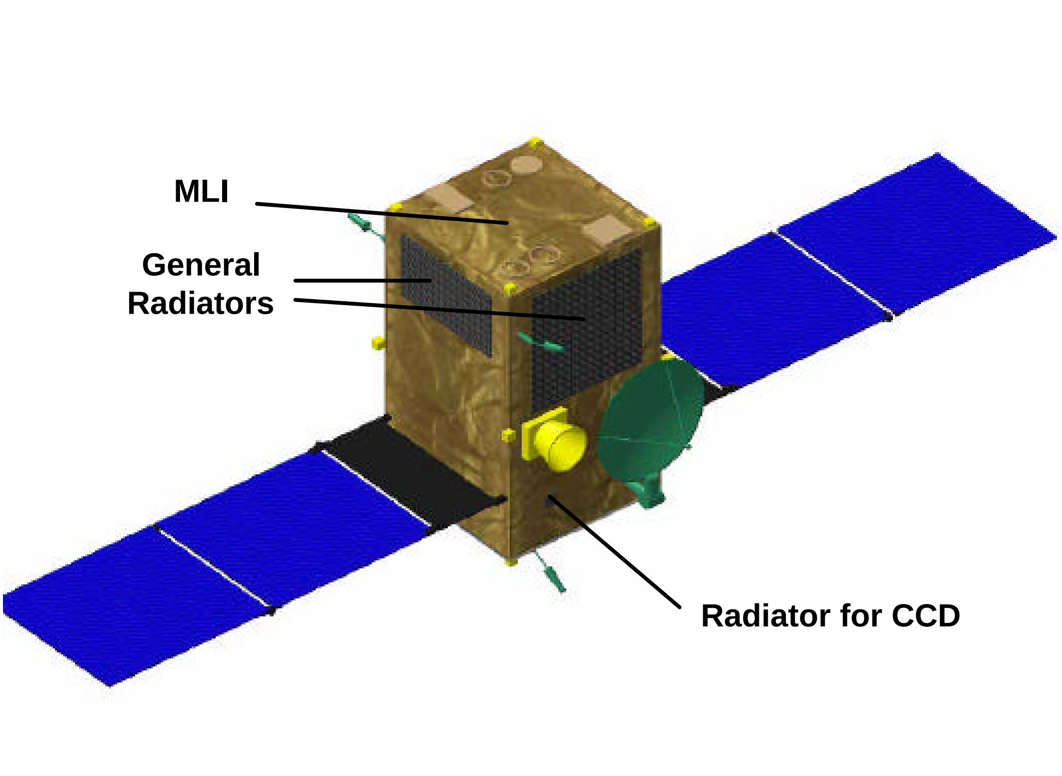


Figure 7-16: Components of Thermal Design

7.7.2.1 Mass Budget

The mass budget for the SAM thermal control mass budget is provided in Table 6-12.

Item	Estimated Mass (kg)	Margin 10% (kg)	Total Item Mass (kg)
MLI	3.92	0.39	4.31
Radiator (room temperature)	1.13	0.11	1.24
Radiator (cryogenic)	0.01	0.00	0.01
Heaters/Thermostat/other	6.00	0.60	6.60
Heat pipes for CCDs	1.00	0.10	1.10
TOTAL	12.05	1.20	13.26

Table 7-10: Thermal Control Mass Budget

7.7.2.2 Heater Power Budget

Table 7-11 gives the heating power budget.

Mode	Heater Power	Comment
Launch mode	15 W	For propulsion S/S
Transfer Mode	27 W	For propulsion S/S and batteries
Initialisation Mode	27 W	For propulsion S/S and batteries
Operational Mode	27 W	For propulsion S/S and batteries
Safe Mode	15 W	For propulsion S/S

Table 7-11: Thermal Control Power Budget

7.8 Power

7.8.1 Requirements and Design Drivers

Design drivers for the power subsystem definition are the following:

- Three-axis stabilised S/C with deployable solar array with a maximum folded area of 1 m x 1.5 m
- Sunlight average power: 350W
- L_1 orbit means that battery is only used in launch and transfer mode or in case of contingency
- 28V fully regulated power bus

Cost minimisation has been addressed as a key driver for the mission design.

7.8.2 Assumptions and Trade-Offs

The following assumptions have been taken into account for the power subsystem preliminary design:

- 10 degrees maximum SA pointing deviation with respect to Normal and no need of drive mechanism
- Solar array temperature up to about 70°C (if using the panel rear side as a radiator; otherwise 120°C)

7.8.3 Baseline Design

A 28V fully regulated power bus is provided to the different Main Bus users through protected power lines, as shown in the block diagram.

Two electronic boxes, one power conditioning unit (PCU) and one power distribution unit (PDU) are foreseen for proper power bus regulation and distribution.

The PCU consists of:

- two 200W battery discharge regulators (BDRs)
- two 200W battery charge regulators (BCRs)
- eight solar array regulator (SAR) sections
- one 2/3 Majority Voter Error Amplifier generating reliable regulator control signals.

The PDU consists of

- latching current limiters for power bus protection
- transistor switches for thermal control
- pyrotechnic and thermal knife drives, as required.

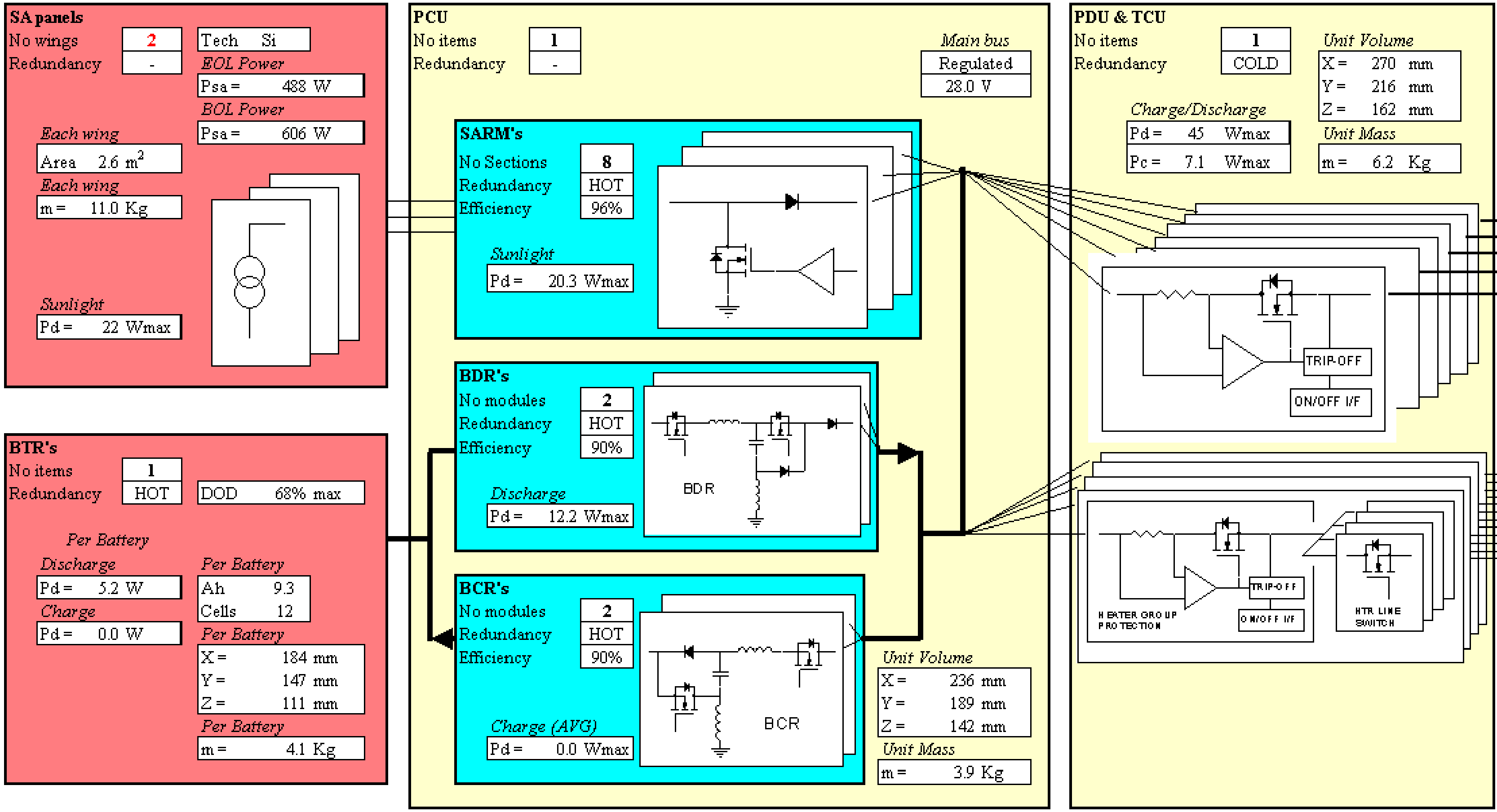


Figure 7-17: Power Subsystem Block Diagram

The BDRs, BCRs and SAR sections operate in hot redundancy, so that the PCU is one failure tolerant with no reconfiguration needs. The solar array has been sized so that the loss of one section still satisfies the mission requirements at end of life. The PDU failure tolerance relies on the usual cold redundant approach.

The battery is nominally used only in Launch mode and Initialisation (before sun acquisition). A battery energy allocation of 400Wh has been considered sufficient to cope with power budget needs. A maximum battery depth of discharge (DOD) below 75% is foreseen. At L_1, the battery will only be used in the event of attitude control loss.

A Li-Ion battery is baselined, as it was for SWM. In fact, as a potential source of cost reduction, the same SWM battery design may be used.

Furthermore, since SWM presents a significant mass margin with respect to the allocated System budget, the concept of designing common PCUs and common PDUs for SWM and SAM is strongly suggested as a potential source of cost reduction.

The solar array preliminary design assumes a power need of about 490W at end of life (EOL) at the power interface (30.5V; that is, 450W at bus level). To achieve such an EOL power, two wings consisting of two solar panels and one panel yoke each are foreseen.

For cost reasons, standard Si BSR cells are envisaged. With the maximum acceptable panel dimensions, 5 strings of 79 cells per panel may be comfortably allocated to provide the required EOL power. Thus, two sections per panel (one section with two paralleled strings, another with three paralleled strings) and a total of eight Solar Array sections are foreseen.

The panel dimensions are: 1.1m x 1.0 m. Therefore, once deployed, the expected size of each wing is 2.9 m x 1.0 m (including panel yoke and hinges).

7.8.4 Budgets

The overall mission power consumption budget is given in section 7.4.4.2. The overall power dissipation is given below.

PSS & SPACECRAFT DISSIPATION vs MODE

		PCU	PDU	TCU	BATTERY	PSS Harness	PSS TOTAL DISSIPATION	S/C TOTAL DISSIPATION
Launch Mode	MAX	26	23	12	4.1	8.4	73	119
	NOM	23	20	10	3.8	7.7	64	109
	MIN	20	18	8	3.5	7.1	56	101
Transfer mode	MAX	36	30	12	0.0	4.1	81	288
	NOM	32	26	10	0.0	3.8	71	260
	MIN	28	23	8	0.0	3.4	62	232
Initialisation Mode	MAX	41	34	12	0.0	5.1	92	358
	NOM	37	30	10	0.0	4.6	81	326
	MIN	33	26	8	0.0	4.2	71	292
Operational Mode	MAX	41	34	12	0.0	5.1	92	358
	NOM	37	30	10	0.0	4.6	81	326
	MIN	33	26	8	0.0	4.2	71	292
Safe Mode	MAX	34	29	12	0.0	4.0	79	303
	NOM	30	25	10	0.0	3.6	69	275
	MIN	27	22	8	0.0	3.2	60	248

Table 7-12: SAM Spacecraft Power Dissipation

The power S/S mass breakdown is given in Table 7-13 below.

S/S Item	**Mass**
Li-Ion Battery Mass	4.1 kg
Solar Array (Si BSR including structure)	21.9 kg
Electronics (PCU/PDU/TCU)	10.1 kg
PS/S Total	36.1 kg

Table 7-13: Power Subsystem Mass Breakdown

7.9 Mechanisms

The identified mechanisms for the SAM satellite are:

- 1 antenna pointing mechanism for the high gain antenna (HGA)
- 2 solar array deployment and associated hold-down and release mechanisms
- 1 separation mechanism (4 identical hold-down and release points between SWM and SAM)

7.9.1 Requirements and Design Drivers

7.9.1.1 Antenna Pointing Mechanism (APM)

The antenna pointing mechanism shall be used to point the antenna towards the Earth during flight to L_1 and orbit around L_1. Thus, the APM shall be able to deploy and trim the high gain antenna around two axes. The APM will be located on one side of the spacecraft. Two hold-down and release mechanisms shall be required to stow the antenna and the APM during launch.

The foreseen HGA mechanisms are:

- 1 two-axis pointing mechanism with associated electronics
- 2 hold-down and release mechanisms

7.9.1.2 Solar Array Mechanisms

The SAM satellite will be powered by two solar arrays, located one on each side of the spacecraft. During launch, these solar arrays will be stowed on each side of the spacecraft thanks to 4 hold-down and release mechanisms. The solar array shall be deployed after separation of the satellite from the launcher. Each of the two solar arrays is composed of 1 yoke and two panels, thus 6 deployment hinges are required for solar array deployment.

For each of the two solar arrays, the foreseen mechanisms are:

- 2 hinged 90° deployment mechanisms (to deploy the yoke from the spacecraft)
- 4 hinged 180° deployment mechanisms (to deploy the two solar array panels)
- 4 hold-down and release mechanisms (based on thermal knife technology)

7.9.1.3 Spacecraft Separation Mechanisms

In order to achieve a good separation between SAM and SWM satellites, a standard set of 4 identical hold-down and release mechanisms (including separation spring set) will be used, one at each top corner of the spacecraft.

Therefore, the foreseen mechanisms per spacecraft are:

- 4 identical hold-down and release mechanisms with separation spring set.

7.9.2 Assumptions, Trade-Offs and Baseline Design

The approach which has been followed to identify the conceptual design of the SAM mechanisms has been to use (as far as possible) qualified, off-the-shelf equipment, in order to reduce cost, procurement time, and development risks.

In the following paragraphs a short description of the anticipated mechanisms is provided, including a preliminary estimate of mass budgets.

7.9.2.1 Antenna Pointing Mechanism

Antenna Pointing Mechanism

The antenna pointing mechanism will be mainly composed of two identical rotary actuators powered by dedicated electronics (APME). The two actuators shall be oriented at 90° to each other. The selection of these off-the-shelf rotary actuators and associated position sensors with respect to the specification will be the driver of the mechanism. The SOHO Antenna Pointing Mechanism can be considered as baseline for this mission. The step resolution of this type of antenna pointing mechanism is commonly 0.01°. The accuracy is linked to the rotary actuator capability but also related to the design of the brackets under thermal behaviour and can be considered as 2 or 3 times the actuator resolution capability.

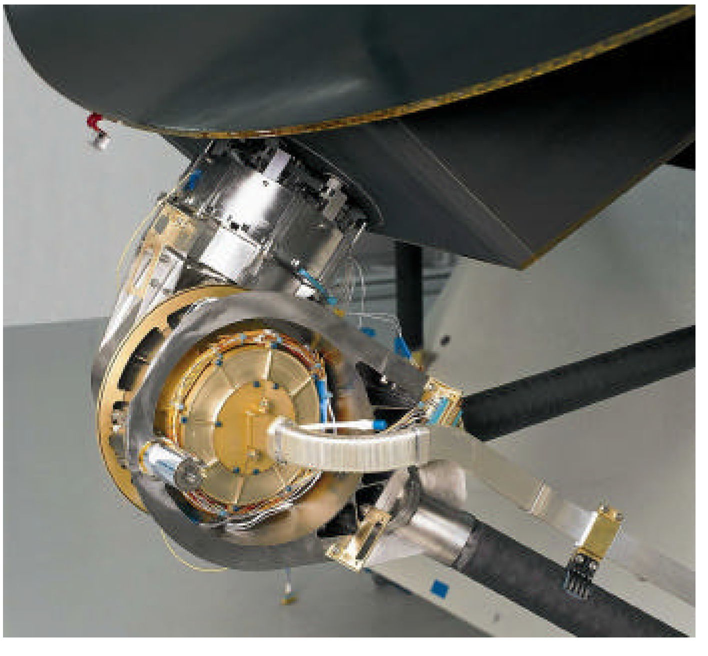


Figure 7-18: Example of 2-axis Pointing Mechanism used for Rosetta

Deployment Mechanism

The deployment of the antenna to its required operational position will be done by the antenna pointing mechanism. No additional devices are needed, and the deployment can be achieved within minutes of release of the hold-down points.

Hold-down and Release Mechanism

Two standard hold-down and release points will be used to stow the antenna and the pointing mechanism together on the spacecraft during launch in order to provide adequate stiffness and strength. Each of the hold-down points will be based on a pyro device to actuate the separation function.

RF Junction

The RF junction (at pointing mechanism level) will be done with flexible wave guide and/or an RF joint.

7.9.2.2 Solar Array Mechanisms

The solar array mechanisms are similar to those used for SOHO. The solar array sub-system is thus a known and fairly mature product. Qualified products will be used at mechanisms level for the deployment hinges and the hold-down and release mechanisms.

Deployment Mechanisms

The number of solar array panels is two per wing. Therefore, the number of standard deployment hinges shall be 6 (2 for yoke deployment, 4 for solar array deployment).

Hold-down and Release Mechanisms

During launch, one of the two solar arrays will be stowed on each side of the spacecraft thanks to 4 hold-down and release points. These hold-down points are based on thermal knife technology: Kevlar® ropes hold the array down, and these are cut by the thermal knives.

7.9.2.3 Spacecraft Separation Mechanisms

The separation mechanisms will be used to separate SAM from the SWM. SWM will be spun after separation. Therefore, the separation mechanisms shall safely connect the two spacecraft (SAM and SWM) together during launch, then separate and eject SWM from SAM.

The selected baseline is to use 4 indentical hold-down and release points, one at each corner on top of the SAM. One separation spring per hold-down point will be used to eject the SWM.

The four hold-down points shall be fired simultaneously, therefore pyrotechnic HDRMs have been selected.

7.9.3 Budgets

The estimated mass and power budgets for the SAM mechanisms are reported in Table 7-14 below. Mass figures do not include pyros.

Mechanism type	Number	Electrical Power	Number of Pyros	Unit Mass w/o margin	Deployment Time
Antenna pointing mechanism with associated 2 hold-down and release mechanisms	1	0.5W	2	9.2 kg	< 20 min
Electronics for antenna pointing mechanism (APM)	1	5W	0	4.35 kg	

Mechanism type	Number	Electrical Power	Number of Pyros	Unit Mass w/o margin	Deployment Time
Solar array hinges and hold-down & release mechanisms	6 hinges and 4 HDRM per wing	10W per hold-down point	0	[Included in power mass budget]	<10 s
Spacecraft separation mechanisms	1 (4 identical HDRM)	0W	4	7 kg	< 1 s
Total		5.5 W	6	20.55 kg	

Table 7-14: Mechanisms Resource Budgets

7.9.4 Options

Pyro hold-down points used for this mission can be changed to non-pyro devices. Some solutions such as those based on Shape Memory Alloy, low melting temperature alloys, paraffin actuators or thermal knives are today qualified and provide good performances with significantly reduced shock.

The main drawback of these solutions is that they can not be fired with the same time accuracy as pyros.

Concerning the separation mechanism, which is actually based on 4 pyro hold-down and release systems needing simultaneous release, a non-pyro solution could be found if a single activation could liberate the 4 hold-down points at the same time. This could be done by using for example a thermal knife to cut a single wire used to lock the 4 hold-down points. However, careful evaluation would be required.

7.10 Pyrotechnics

The Space Weather SAM satellite will require deployments of one communications antenna and two solar arrays, all of which are stowed for launch, plus separation of the SWM spacecraft from the composite launched to L_1. The stored energy of pyrotechnics leads to high mass- and power-efficiency without the thermal control requirements associated with alternative approaches.

For all the above applications, cost and reliability considerations demand that qualified off-the-shelf devices are used. Known pyrotechnic devices have achieved reliability and qualification statuses unmatched by alternative technologies.

7.10.1 Assumptions, Trade-Offs

Standard off-the-shelf devices reduce performance and procurement risk and allow for 5% mass-margin to be applied. The devices include redundant initiators with independent switching, command and supply, harness and electronics.

7.10.2 Baseline Design

7.10.2.1 SWM/SAM Separation

The multi-point SWM/SAM inter-satellite attachment requires discrete devices for attachment and release. Four pyrotechnic release-nut devices are particularly appropriate for this application in view of the small response-time dispersion achievable.

7.10.2.2 Communications Antenna

Two similar release-nut devices will provide the launch constraint and release of the Communications antenna.

7.10.2.3 AOCS

The four reaction wheels for AOCS each require launch-locks which are released with pyrotechnic devices. These are incorporated in the wheel design.

7.10.2.4 Solar Array

The chosen solar array employs pyrotechnics for release of the launch-locks.

7.10.2.5 Propulsion

The Propulsion subsystem does not require pyrotechnic valves.

7.10.3 Budgets

The power demand per pyrotechnic device is of millisecond duration and thus negligible, particularly when fired before full spacecraft operation.

Unit masses of typical pyrotechnic actuators are in the region of 0.17 kg.

Mechanism type	No of Pyros	Unit Mass	Total Mass
Antenna hold-down and release mechanism	2	0.17 kg	0.34 kg
Spacecraft separation mechanism	4	0.17 kg	0.68 kg
Total	6		1.02 kg

Table 7-15: Pyro Mass Budgets

7.11 Attitude and Orbit Control (AOCS)

7.11.1 Main Requirements

The main functions required (i.e. the performances required during the nominal observation mode) of the Attitude and Orbit Control Subsystem (AOCS) for SAM are:

- To keep the 3-axis stabilised spacecraft sun-pointing in observation mode with a 7 arcsecond, 3 σ pointing accuracy
- To provide a pointing stability of 5 arcsecond over 15 min
- To determine the spacecraft attitude in inertial space with a pointing knowledge accuracy of 1 arcsecond

7.11.2 Design Assumptions

The S/C AOCS design resembles that of SOHO and consists mainly of units which are well characterised.

7.11.3 Baseline Design

Given the mission high pointing requirements (payload, communications, thermal) a three-axis stabilised spacecraft design is most appropriate.

The avionics design is very similar to that of SOHO, but an important difference is the use of the Astrium fibre-optic gyroscopes. These units should be particularly advantageous in terms of reliability as well as accuracy. They will be qualified in missions such as Herschel, and offer very attractive advantages in accuracy and reliability.

Figure 7-19 below illustrates the general architecture of the avionics subsystem.

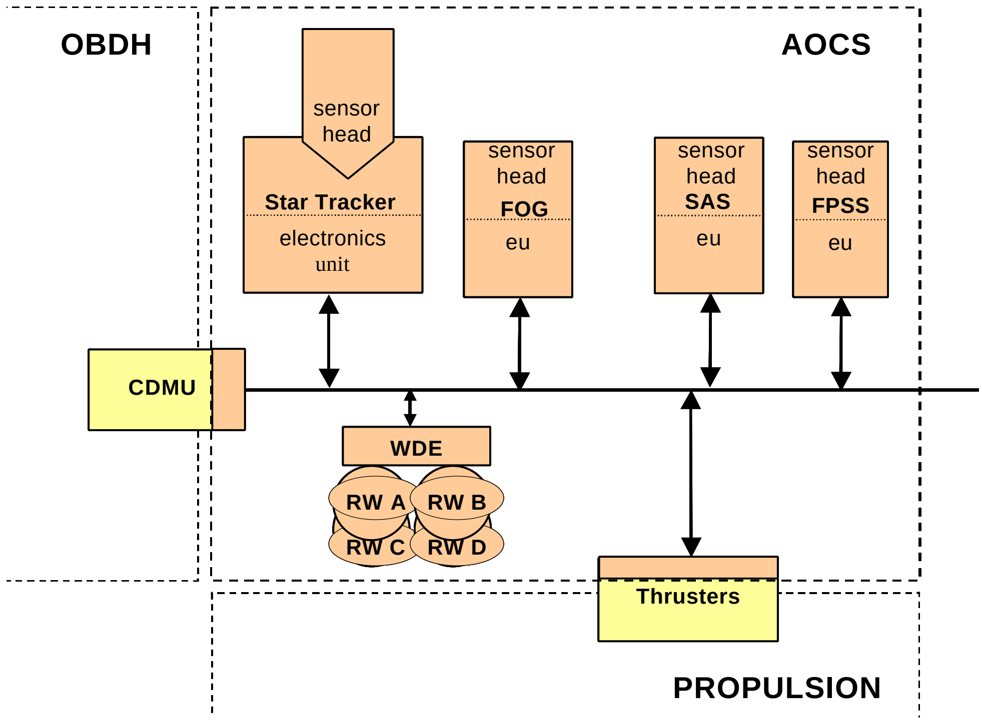


Figure 7-19: Avionics Subsystem General Architecture.

7.11.4 Equipment Overview

7.11.4.1 Star Tracker

The star trackers provide the spacecraft attitude (directly from the star tracker electronics unit as a quaternion). They must be placed on a spacecraft side panel (i.e. not the payload or opposite face) and should point away from the sun by at least 40 degrees. The use of a combination of gyros, star tracker and sun sensors offers robustness to the avionics design.

7.11.4.2 Sun Acquisition Sensor

These coarse sun sensors are used for initial sun acquisition and during failure recovery. The units made by TNO-TPD have been flown extensively on missions such as ISO, SOHO and SPOT.

7.11.4.3 Fine Pointing Sun Sensor

Initial sun acquisition is performed with the sun acquisition sensors, before using the fine sun sensors to go to a fine sun pointing phase.The Adcole fine sun sensor is envisaged, the same type of unit which was used on missions such as SOHO and XMM.

7.11.4.4 Fibre Optic Gyros

After years of development in Europe, this type of gyro is now mature. The Astrium fibre optic gyro is to be qualified with Herschel in 2004. A skewed configuration of four gyros is used, together with the electronics unit (either housed separately or in a single box).

7.11.4.5 Reaction Wheels

The MMS reaction wheels and wheel drive electronics are the units with the highest mass and power requirements in the avionics. The higher power consumptions, however, are associated only with the early orbit insertion phase of the mission, and are not demanded for the greater part of the mission (i.e. not in observation mode).

7.11.5 Avionics Mass and Power Budgets

As can be appreciated from Table 7-16 below,the power requirement for the avionics subsystem is relatively low. This is due mainly due to the fact that the reaction wheels, i.e. the units with the highest mass and power requirements, will require the most power during accelerations, i.e. when the spacecraft performs manoeuvres. As these are foreseen only for the initial operations leading up to the nominal observation mode, the operational power requirements are relatively low.

Unit	**Qty.**	**Unit mass (kg)**	**Total mass (kg)**	**Unit power, nom (W)**	**Unit power, max (W)**
Star tracker optical head	2	1.9	3.8	19.2 (both)	
Star tracker EU	2	2.2	4.4	-	-
Fine pointing sun sensor	2	0.4	0.8	-	-
Fine sun sensor EU	2	1.4	2.8	1.8	
Sun acquisition sensor	3	0.25	0.75	-	-
Reaction wheel	4	4.85	19.4	22.7 x 3[8]	< 90
Reaction wheel EU	1	2.2	2.2	6	8
Fibre optic gyros	4	2	8	20 (total)	
		TOTAL	42.2	115	
		TOTAL with 10% margin	46.4	127	

Table 7-16: Mass and Maximum Power Requirement of Avionics Units

[8] There are four reaction wheels for redundancy reasons, only three will be used at a time.

7.12 Data Handling

7.12.1 Requirements and Design Drivers

The main objective of the Solar Activity Monitor mission is to provide real-time imaging of the solar disc and corona. The orbital location allows an unobstructed view of the sun and so continuous data generation from instruments must be managed by DHS.

The housekeeping data rate is assumed to be 2 kbps, and the four instruments housed in the spacecraft generate a total observational data of 33.6 kbps. Most of the telemetry data comes from the White Light Coronograph (21 kbps) and the EUV Imager (10.5 kbps). The instrument data rate is continuous and no rate peak is assumed. The TM rate reduces to 100 bps in safe mode, when payloads are switched off and LGAs are used for data download.

The telecommand rate during nominal operation is assumed to be 2 kbps, but that will be reduced to 50 bps in safe mode.

No payload data processing is required and only raw data will be transmitted to the ground station for subsequent processing.

The ground coverage is uninterrupted in order to realize a real time downlink to earth. As an option, contingent outage periods could be managed by on-board storage of the generated data for subsequent dumping. See section 6.12.5 for more details.

The mission time is long (5 years) but the satellite will be exposed to a ‘benign’ radiation environment of 5 krad with 4mm equivalent shielding.

The design drivers are simplicity and reuse of existing Data Handling Systems.

7.12.2 Design Assumptions

The assumptions have been derived from the mission requirements (e.g. storage, data flow, operations and processing requirements), from previous DHS experiences, and from currently available components and standards. In particular, the requirements and constraints are similar to those for SWM, so the designs of the two DHS’s are very similar, and derive from the adaptation of the DHS designed and manufactured by SIL for the ESA PROBA mission.

7.12.3 Baseline Design

The mission lifetime is high, so a redundant architecture has been selected. A cold duplex architecture is implemented because of the assumption that the DH must tolerate one permanent fault and that the requirement on the maximal outage time of the DHS is low.

All the SAM DHS modules are built around a single compact unit that provides all the expected services. The other S/C units are connected with the DH box through a few simple interfaces.

These are :

- Analogue status interfaces
- Temperature status interfaces
- Bi-level interfaces
- Event interfaces
- Command pulse interfaces
- RS422 interfaces
- TTC.B.01 interfaces

The intruments and sensors are connected by RS422 or TTC.B.01 serial link to send commands and retrieve telemetry. The use of serial links simplifies the electronic design on both the DHS and instrument sides. A few additional lines directly connect the instruments and sensors to telemetry module to provide important status values in the event of severe failure of the instruments.

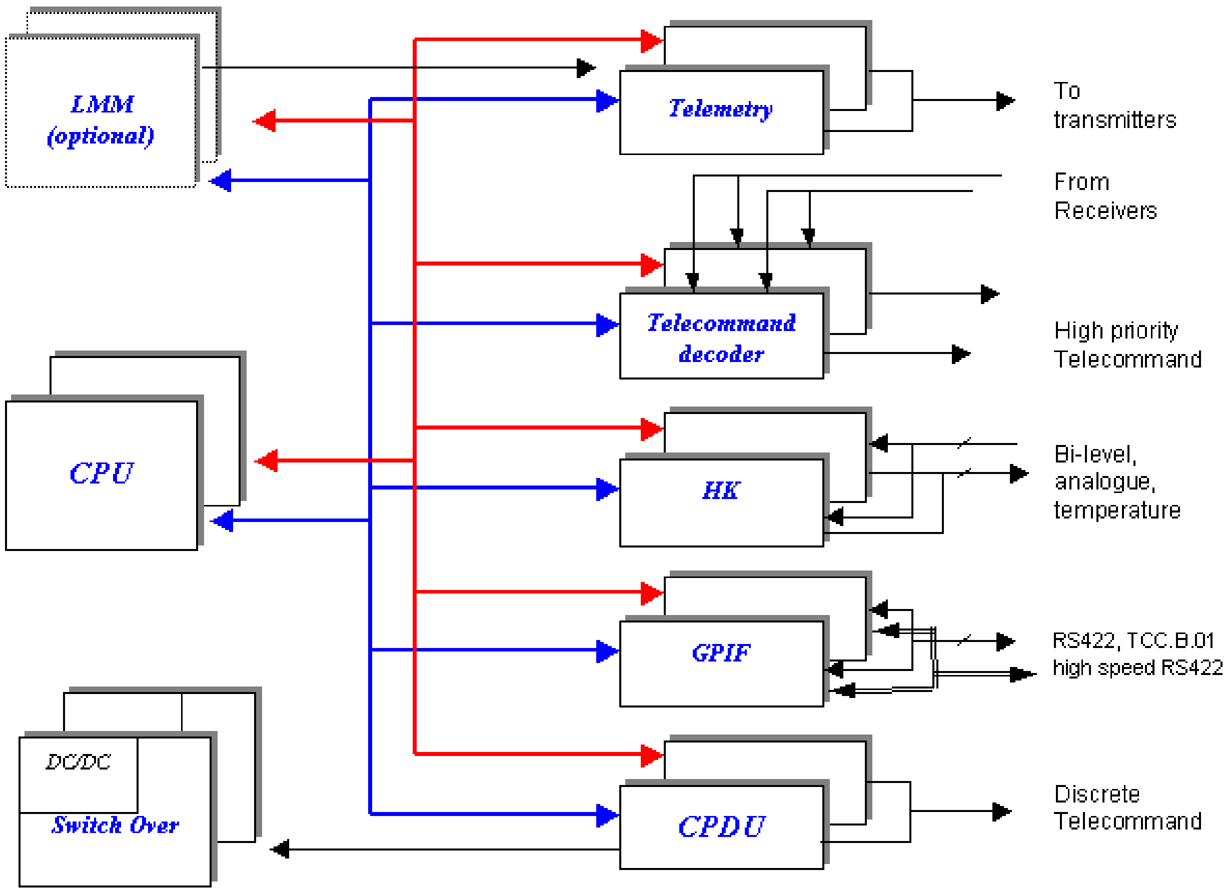


Figure 7-20: CDMU/RU Block Diagram

The data rate of the coronagraph exceeds the maximum data rate of the PROBA DHS serial link, so a dedicated high speed RS422 interface has been added.

The processor module is based on the SPARC TSC695 CPU, whose performance at 25 MHz is up to 20 MIPs and 5 Mflops. The frequency of the processor and the size of the application software memory will be adapted to the mission needs, to minimise unnecessary power consumption.

No criticalities are foreseen for the application software. The continuous ground coverage reduces the autonomy requirements and therefore also the software complexity.

7.12.4 Budgets

Mass	Power
10 kg	13.5 W

Table 7-17: SAM DHS Mass and Power Budget

7.12.5 Options

In order to store data on board during any outage periods, an optional Local Mass Memory could be provided. The data could then be dumped subsequently. In this case, the stored data and real time telemetry would be simultaneously downlinked.

The Local Mass Memory Module size would be 3 Gb, corresponding to 24 hours of HK and payload data storage. During normal operations the mass memory can be switched off.

To download the stored data (together with the real-time TM), a high data rate mode is required (350 kbps). Therefore an additional IEEE1355 interface would be required.

7.13 Telecommunications

7.13.1 Requirements and Design Drivers

The requirements for the definition of the SAM telecommunication subsystem have been the following:

- Continuous real-time downlink of payload data (@ 33 kbps) plus housekeeping telemetry (typically 2 kbps).
- Telecommand uplink (typically at 2 kbps) and ranging capability required but not on a continuous basis
- halo orbit in L_1 with ranges during nominal operations up to 1.7M km

7.13.2 Design Assumptions

The following sections describe the main assumptions considered in the architectural design of the subsystem and the link budget evaluation.

7.13.2.1 Frequency of Operation

For operation in L_1 the use of X-Band is recommended (7190 - 7235 MHz uplink, 8450 – 8500 MHz downlink), which is allocated to the Space Research Service. The highly interfered environment existing in S-Band could compromise the nominal performance due to the low received levels involved in missions to the Lagrangian points.

7.13.2.2 Ground Station Assumptions

The typical performance provided by the 15 metre stations of the ESA ESTRACK network has been assumed.

	7 GHz Transmit	**8 GHz Receive**
Frequency (MHz)	7145 - 7235	8400 - 8500
Polarisation	RHCP or LHCP	RHCP and LHCP
Cross polarisation (dB)	-25.00	-25.00
Sidelobes	ITU App. S7	ITU App. S7
antenna efficiency	> 65%	60.30
EIRP (dBW)	82 (400W SSPA)	
G/T @90° (dB/K) Clear Sky		39.10
G/T @10° (dB/K) Clear Sky	-	38.00
G/T @10° (dB/K) 99% of the year for Kourou weather conditions		35.60
Rx/Tx isolation (dB)	90.00	

	7 GHz Transmit	8 GHz Receive
Pointing accuracy (dB)	< 1	

Table 7-18: Ground Station Characteristics

It is assumed the availability of Turbo Decoders on the station is:

Eb/No (for PFL = $1e^{-5}$) = 0.8 dB

Where: Eb = energy per bit
No = noise power density no.

7.13.2.3 Transponder and RF Power Amplifier Assumptions

The transponder required to support SAM is X/X Near-Earth type, with Tx/Rx coherency and ranging capability. The transponder will be adapted to an external TWTA power amplifier. The requirements are as follows:

Property	Value
Frequency	Receive: 7190 - 7235 MHz, Transmit: 8450 - 8500 MHz
Transmit Power	0 dBm at the output of the transmitter within the transponder 30 Watts TWT external
TM Modulation/Data Rates	NRZ/BPSK/PM for 100 bps (HK only) SPL/PM for 35 kbps (HK + Payload Data RT) OQPSK for 350 kbps (Data Dump) TM Data rates selectable by TC
TM Coding	Turbo Encoder
Receive Threshold	-135 dBm (TC demod)
Noise Figure	2.5 dB
TC modulation	NRZ/BPSK/PM
TC Data Rates	25 bps and 2 kbps TC Data rates selectable by TC
Mass	Transponder: 3.5 kg 30W Amplifier: 750 g TWT, 3 kg EPC & HV cable
Dimensions	Transponder : 275 x 110 x 197 mm TWT : 58 x 50 x 350 mm EPC : 100 x 80 x 360 mm
Power Bus	Transponder: From 21 to 50 V Amplifier: 22 V to 37 V
Consumption Transponder	5 Watts (Rx & Tx at 0 dBm output)

Property	Value
Consumption TWTA	ON: 60 Watts Stand-by: 7.5 Watts

Table 7-19: SAM Transponder Requirements

7.13.3 Antenna Analysis

In the definition of the antenna performance required for SAM, three different scenarios have been considered:

1. Contingency operations and Safe mode

   The spacecraft should be able to communicate with Earth from any aspect angle. Therefore an omni-directional coverage for both transmit and receive is highly desirable.

2. Transfer orbit

   Just after separation from the launcher, the baseline considers SAM attached to the bottom of SWM S/C for several hours. During this time, both bodies are three-axis stabilised. SAM cannot use a low gain antenna (LGA), nominally placed in the top side; however a good compromise has been made by combining two LGAs (LGA2 and 3) as shown in Figure 5-33. Following the detachment from SWM S/C, communication to Earth is ensured by the quasi omni-directional coverage provided by LGA1&2 or LGA1&3, or using a pointable 70 cm dish high gain antenna (HGA).

3. During nominal orbital operations the link will be established via the HGA pointing towards the Earth (by means of the steering mechanism)

7.13.3.1 Antenna Coverage and Location

Figure 5-33 shows the antenna locations.

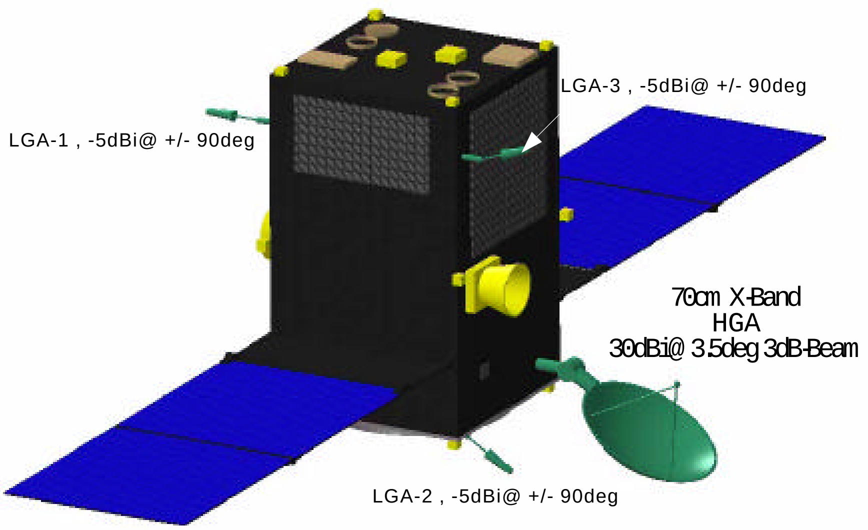


Figure 7-21: SAM Antenna Layout

7.13.3.2 Low Gain X-Band Antennas Assumptions

Each low gain antenna has a nearly hemispherical coverage, with an absolute gain >-5 dBi. The estimated mass of one low gain antenna is 300 gram. Optimisation of the antenna locations taking into account the FOV blocking effects of appendages like the solar arrays and other structural elements, should be analysed in a later phase of the study.

This type of antenna will be needed for most future X-Band missions and is already under development.

7.13.3.3 High Gain X-Band Antenna Assumptions

To support nominal operations, a 70 cm parabolic antenna is considered. This High Gain Antenna (HGA) presents +30dBi gain and a 3.5deg 3dB beamwidth. The HGA is mounted on a pointable mechanism.

7.13.4 Link Budget Evaluation

7.13.4.1 Uplink Budget

Range: 1.7M km	**15m station** **82 dBW EIRP**
Operation via LGAs	25 bps (clear sky)
Operation via HGA	2 kbps (99.5% of the year in Kourou)

7.13.4.2 Downlink Budget

Range: 1.7 M-km	**15m station** **G/T = 35.6 dB/K, 99% year time Kourou**
Operation via LGAs	100 bps
Operation via HGA	35 kbps (SPL/PM, compatible with ranging)
Operation via HGA	350 kbps (OQPSK, no ranging)

7.13.5 Communications Architectural Design

The communication subsystem consists of the following elements:

- Three Low Gain Antennae
- One 70 cm High Gain Antenna
- One RF Distribution Unit

- Two transponders and two TWT 30 Watts power amplifiers. The transponder integrates the transmitter (plus modulator), the receiver (plus demodulator) and the diplexer that combines both units into a single port towards the antennae.

The architectural design proposed for the SAM communication subsystem is depicted in Figure 5-32.

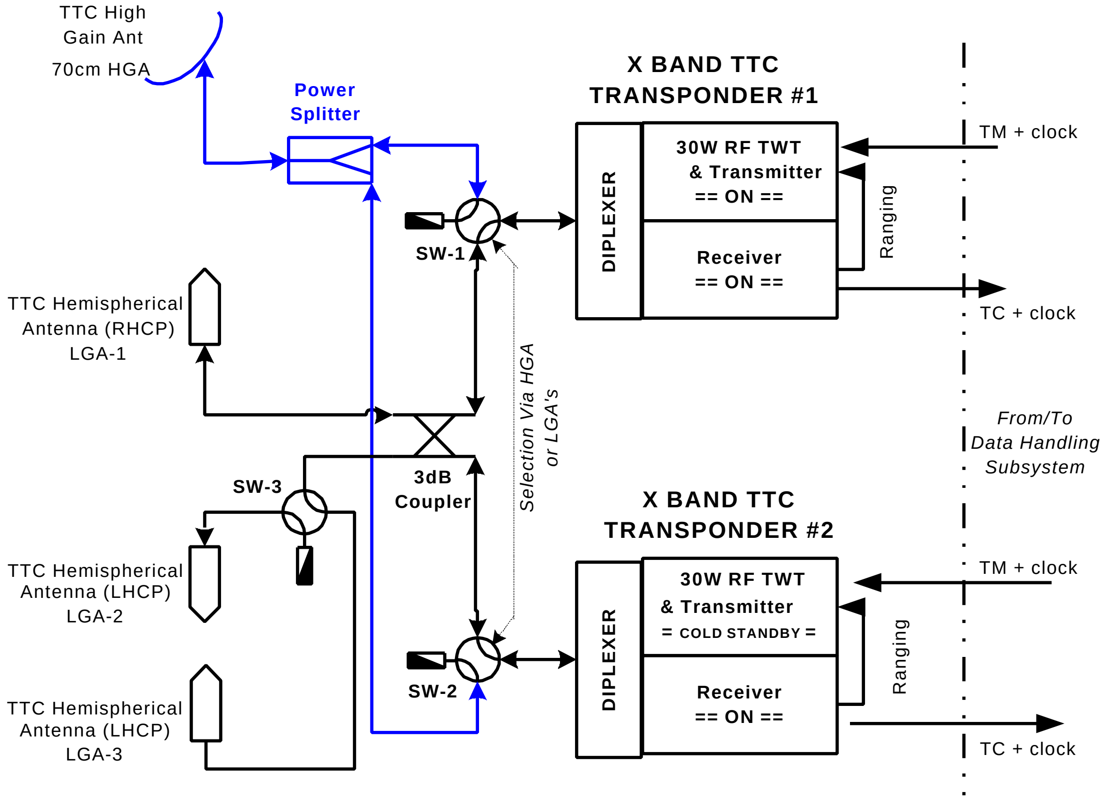


Figure 7-22: SAM Communication Subsystem

7.13.6 Budgets

7.13.6.1 Mass Budget

Items	No. of units	Nominal unit mass (kg)	Total Nominal Mass (kg)
X-Band Transponder	2	3.50	7.00
30W TWT amplifier	2	3.75	7.50
RF Distribution unit (including power combiners, switches, harness)	1	2.50	2.50
X-Band LGA	3	0.50	1.50
X-Band HGA	1	5.50	5.50
Total Mass (kg)			24.00

Table 7-20: SAM Telecomms Mass Budget

7.13.6.2 Power Budget

Item	Number of units	DC power (Watts)	
X-Band Transponder	2	6.00 x 2 = 12W	Both transponders ON
X-Band 30W TWTA	2	60.00W 7.50W	TWTA ON TWTA Stand-by
Power Consumption		79.50	With TWTA in Cold Redundancy

Table 7-21: SAM Telecomms Power Budget

7.13.7 Options

7.13.7.1 Option C: Data Relay

As an option, the possibility to use intersatellite links by means of GEO relay to support all three Space Weather missions was evaluated with the following outcome:

- The ground segment would be certainly simplified, however the relay satellites become complex, requiring dedicated TT&C developments and very high mass and power consumption. The associated costs and risk need further attention.
- The use of relay satellites to support the IMM constellation may be a solution although difficult to justify for only these four spacecraft.
- The use of relay satellites for communication with the Lagrangian points requires a complete analysis of antenna coverage and revised pointing requirements.

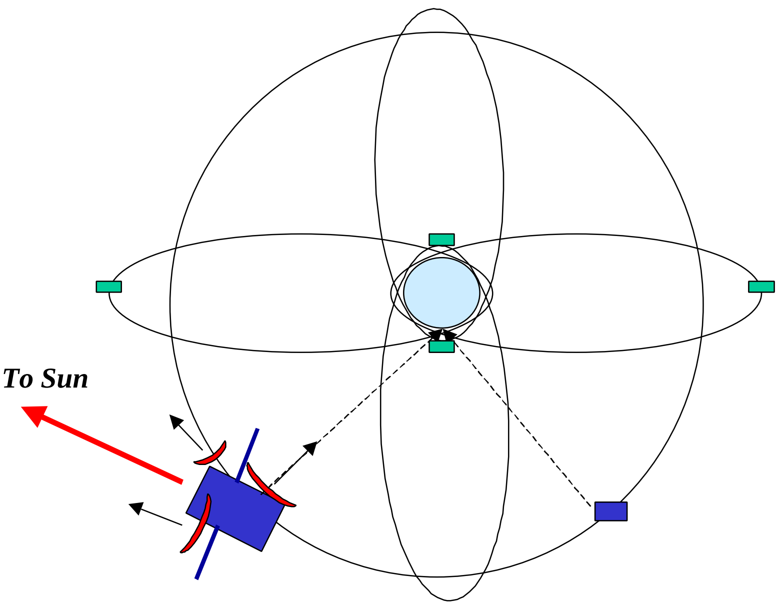


Figure 7-23: Impression of Communications for Data Relay Option

7.13.7.2 Option D: Trailing Orbit

As an option, the implications of having SAM S/C at a trailing point located 26M-km from Earth have been evaluated and found feasible, but with the following warnings:

- There are 23 dB extra propagation losses that make the support of Safe Mode (Via on board LGAs) unfeasible with a 15m antenna ground station. The use of a 35m Deep Space ground station would be required.
- The nominal operations (35 kbps using on board HGA) could only be supported with a 15m antenna if operating in Ka-Band. However, in this case and due to the higher attenuation losses due to rain, the real time downlink requirement of above 95% average year needs to be carefully evaluated depending, on the ground station location.

7.14 Structures

7.14.1 Requirements and Design Drivers

The Solar Activity Monitor (SAM) structural design, in combination with SWM on top, is not specifically driven by a volumetric budget requirement for the fairing of the reference launcher (Soyuz-Fregat). Nor is it driven by mass budget constraints for the satellite stack to be launched, given the large margin provided by the launcher.

The requirements for the stiffness of the SAM structure from the launcher are:

- The first lateral frequency for one S/C separately should be >15 Hz.
- The stack of the two S/C should have a first lateral frequency >12 Hz.

For all the equipment supporting the instruments and the S/C operations, the structure needs to provide:

- A platform for electronic equipment, propulsion, power & harness
- Easy access for AIV activities

7.14.2 Assumptions and Trade-Offs

No COTS platform was found fully suitable, due to the requirement from the instruments (particularly the telescopes) that define the height of the S/C. A trade-off was done between one integrated S/C structure and a conventional split into Service Module plus Payload module. Since the instruments will be delivered as complete units, one base structure is most suitable to provide the support for both instruments and equipment.

7.14.3 Baseline Design

For the baseline design, the spacecraft interfaces with the launcher via a standard 937 adapter. The adapter ring is connected to the bottom platform of the satellite, which is machined from one piece of aluminium. The S/C core is build up from three sandwich panels forming a cross, closed at the top by another sandwich panel. The four outer panels will close the box. These panels will contain the equipment and instruments. The tanks are supported by two horizontal platforms with brackets.

An overview of the structural design for the SWM mission is shown in Figure 6-31.

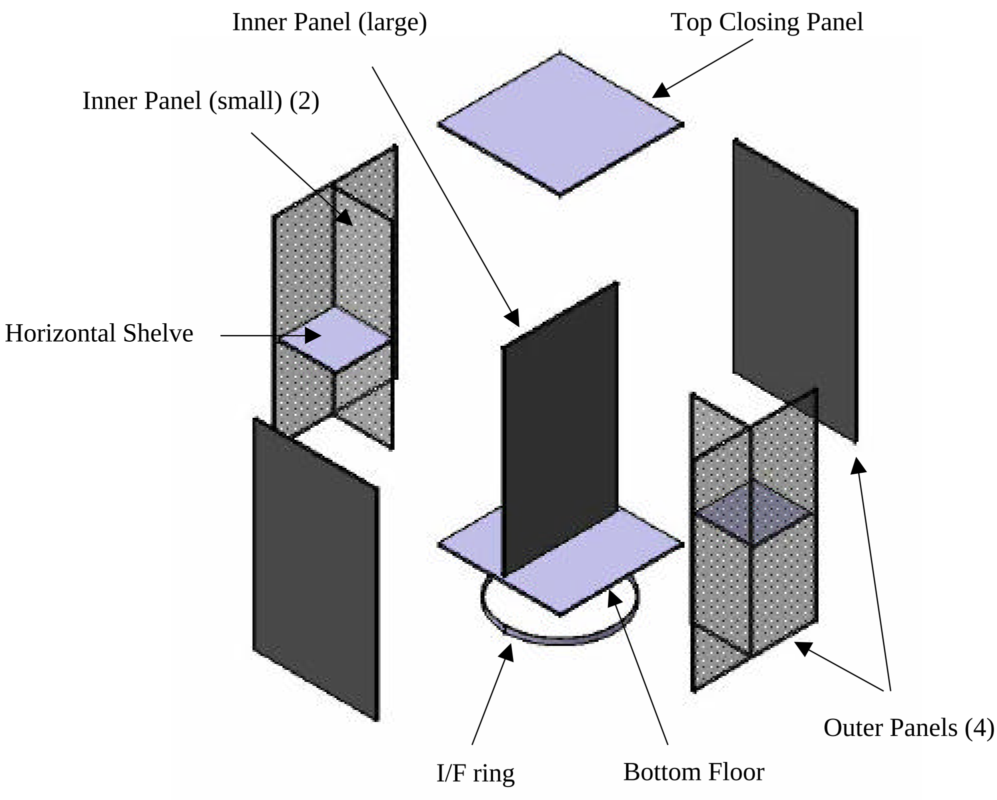


Figure 7-24: Primary Structure for the S/C

The spacecraft structure is not a recurrent item. It is a completely new structural design, but the architecture and the related technology are recurrent from many proven spacecraft bus design.

For the I/F with the SWM, four brackets provide an interface for the four bolts plus pyro-technical separation devices. (see Figure 7-25).

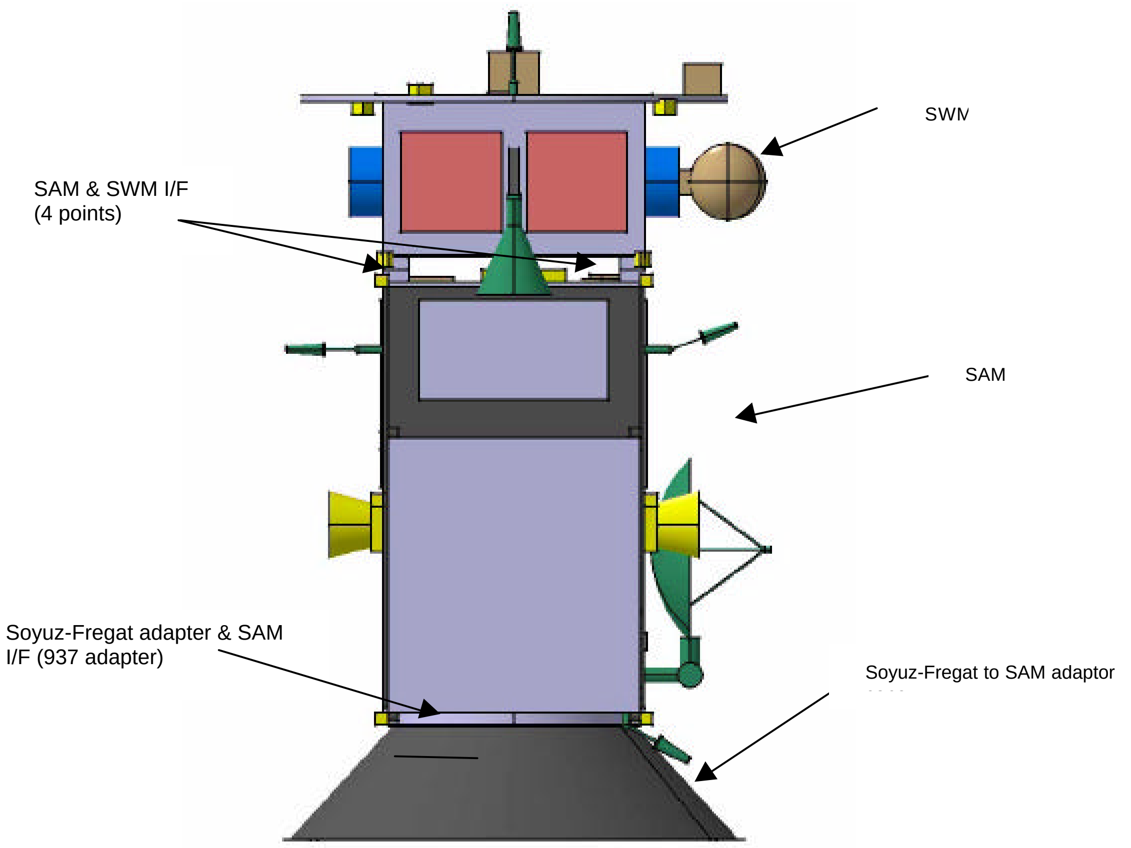


Figure 7-25: SAM & SWM Stacked in Launch Configuration

7.14.3.1 Frequency Requirements

The launcher frequency requirements have been met: the first lateral frequency of the SAM S/C is 22 Hz, well above the required 15 Hz. For the S/C stack, with SWM on top in launch configuration, a first lateral frequency of 12.8 Hz was calculated, which is above the required 12 Hz.

Figure 7-26 shows the stick model used for the first lateral mode calculation.

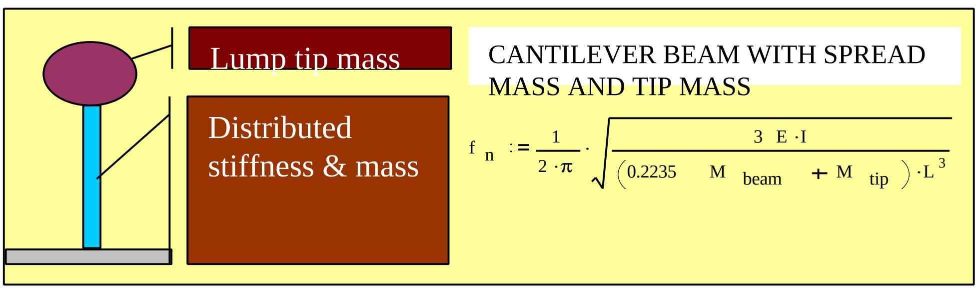


Figure 7-26: Preliminary Modal Analysis Model

7.14.4 Mass Budget

Table 5-30 shows the mass breakdown of the primary structure.

Item	No.	Unit Mass (kg)	Total Mass (kg)	Unit mass with margin (kg)	Total mass with margin (kg)
I/F Ring	1	3.0	3.0	3.6	3.6
Bottom Floor	1	28.0	28.0	33.6	33.6
Outer Panels	4	5.63	22.52	6.76	27.02
Inner Panel (large)	1	5.58	5.58	6.69	6.69
Inner Panels (small)	2	5.58	11.16	6.69	13.38
Top Closing Panel	1	5.58	5.58	6.69	6.69
Horizontal shelf	2	0.82	1.64	0.98	1.96
I/F brackets for SWM	4	0.5	2.0	0.6	2.4
SA Attachment brackets	2	3.0	6.0	3.6	7.2
Inserts and Miscellaneous	1	5.0	5.0	6.0	6.0
		TOTAL	90.48		108.54

Table 7-22: Primary Structure Mass Budget

7.15 Programmatics

7.15.1 Master Schedule

The project Gantt chart in Figure 5-35 below indicates the major mission phases consistent with the following key milestones:

- Start of the project Phase A in July 2002
- Launch in November 2006
- A 4- to 6-month (maximum) Transfer Phase to L_1
- A nominal Operational Phase of 5 years until first quarter 2012

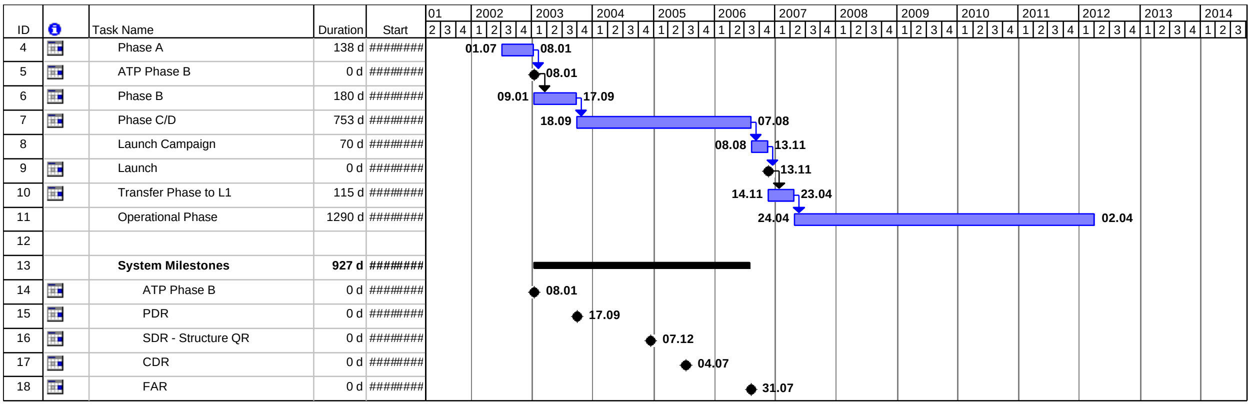


Figure 7-27: Project Master Plan

7.15.2 Development and AIV

The SAM spacecraft includes the following main building blocks:

- A SOHO-like body.
- A top platform carrying two Sun Acquisition Sensors
- Four side panels carrying both the experiments and the spacecraft subsystem electronics (Avionics, Telecom, AOCS, Power S/S). They are covered with MLI or radiators.
- Externally mounted on the side panels are the two Solar Arrays, the deployable High Gain Antenna and the two Star Trackers.
- A bottom platform carrying the Reaction Wheels.
- The horizontal panels carry the 2 propulsion tanks, with equatorial mounting. The propulsion thrusters are opportunely mounted on the external panels.

The development of the spacecraft relies on existing designs and available technology. The structure is specifically designed, with a reference to the former ESA SOHO spacecraft. Qualification testing is required.

The project development and more specifically the cost estimates have assumed a streamlined industrial team whereby the Prime Contractor is responsible for:

- Overall design, development, and procurement of the spacecraft
- Detailed spacecraft design at system and subsystem level
- Direct procurement of the spacecraft units, equipment and major assemblies (hardware and software)
- Overall spacecraft Assembly, Integration and Verification (AIV) activities
- Definition and control of the technical and operational interfaces of the Instruments

7.15.2.1 Model Philosophy

Considering the development risk identified in most aspects of the spacecraft design, a Protoflight approach has been selected at spacecraft level, based on a 3-model philosophy:

- Structural and Thermal Model (STM):
  This will ensure the mechanical qualification of the spacecraft design. Most of the unit assemblies will be represented by structural and thermal dummies (STM units).
- Engineering Model (EM)
  This will ensure verification of the overall electrical, functional and software interfaces. EM units will be used most of the time, representing the form, fit and functions of the flight units. Exceptionally, Interface Simulators could be used for the Payload Units. BB units (functionally representative, with commercial components) could be used if cost-effective, e.g. in the case of recurring units with simple design, or off-the-shelf equipment.
- Protoflight Model (PFM):
  Built to full flight standard, this will be subject to qualification test levels of acceptance duration.

As a programmatic approach, the use of Hi-Rel EEE parts has been assumed. However, the reliability level of EEE parts must be carefully assessed, due to the impact this selection necessarily has on the risk and cost of the project.For costing, procurement of European Hardware has been assumed in general whenever a design and the technology are available, and provision of spare kits is foreseen for all units. They could either be specifically procured or available as heritage of recurring units from past projects.

7.15.2.2 AIV Approach

Taking into account the given model philosophy and the expected development time of the Instruments, an overall AIV plan is outlined in Figure 5-36.

ID		Task Name	Duration	Start
2		Phase A	138 d	01.07.02
3		ATP Phase B	0 d	08.01.03
4		Phase B	180 d	09.01.03
5		**PDR**	0 d	17.09.03
6		**SDR - Structure QR**	0 d	07.12.04
7		**CDR**	0 d	22.07.05
8		**FAR**	0 d	07.08.06
9		**Launch**	0 d	13.11.06
10		**Phase C/D**	**753 d**	**18.09.03**
11		Design	138 d	18.09.03
12		Procurement STM	250 d	24.12.03
13		Procurement EM/BB Units	230 d	21.11.03
14		**OBSW V0.1**	0 d	02.07.04
15		**P/L STM Experiments Need Date**	0 d	08.12.04
16		STM AIV	88 d	08.12.04
17		EGSE SDE Need Date	0 d	31.10.03
18		EGSE - Test Sequences	185 d	31.10.03
19		EGSE Commissioning	60 d	16.07.04
20		**P/L EM Experiment Need Date**	0 d	07.04.05
21		**OBSW V1.0 Need Date**	0 d	27.10.04
22		EM AIV	206 d	08.10.04
23		Procurement PFM structures	180 d	08.12.04
24		Procurement Units PFM	276 d	05.08.04
25		**OBSW V2.0 Need Date**	0 d	23.06.05
26		EGSE - Test Sequences Update + CHECK	66 d	25.07.05
27		**P/L PFM Experiments Need Date**	0 d	27.10.05
28		PFM AIV	247 d	26.08.05
29		Launch Campaign	70 d	08.08.06

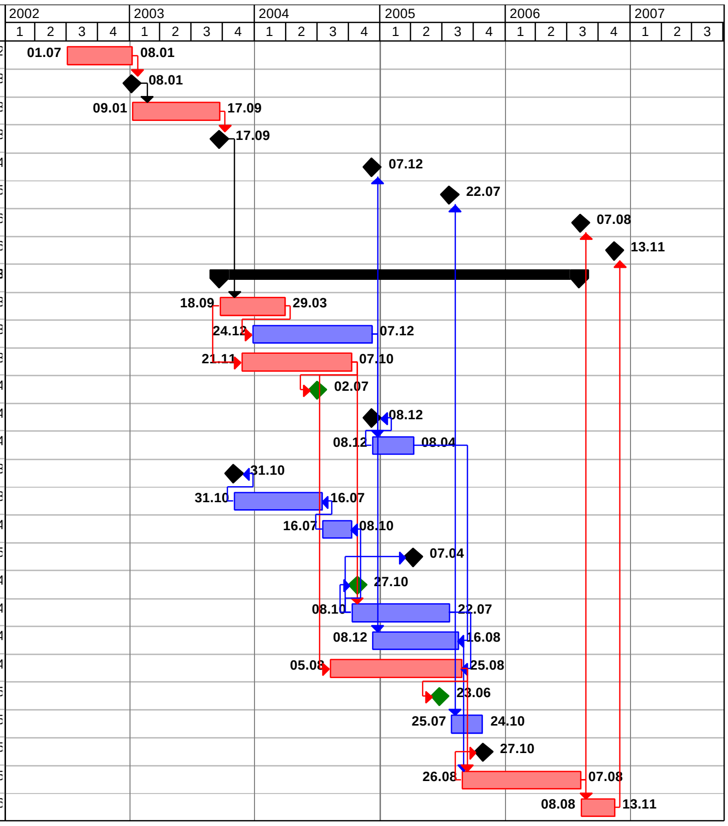


Figure 7-28: AIV Planning Chart

The validity of the planning is based on the following key assumptions:

- The suppliers qualify the instruments' mechanical and thermal designs, including verification of inner optical alignment stability. Final qualification will be achieved on the spacecraft PFM model.
- Instrument STMs will be delivered at a build standard compatible with the spacecraft STM programme. Where required, mechanical alignment of instrument boxes with the spacecraft structure will be tested at spacecraft level.
- Instrument BBs will be delivered at a build standard which, at a minimum, has to be representative of the electrical and functional interfaces. No calibration activities are envisaged on the ATB.
- Instrument PFMs will be installed on the satellite by the Prime Contractor. Functional and interface tests will be performed to certify their proper on-board accommodation. Calibration activities will be performed on the flight satellite.

7.15.2.3 Critical AIV Aspects

A critical aspect is the timely release of the SW versions. The on-board SW version V1.0 must be ready for the start of the EM activities. This should be the first version fully implementing the required system SW functions.

The final version, implementing the results of system testing on EM, must be loaded on the PFM units before they are delivered to the system for integration. This final release, V2.0, shall be tested on the PFM.

7.15.3 Programmatic Risk Assessment

The risk elements, from a programmatic point of view can be summarised as follows:

- The proposed short development time is a factor of risk due to the reduced possibility to recover delays.
- In the hypothesis that a dual launch with SWM is selected, the same constraints as for SWM shall apply.

7.15.4 Links to Other Projects

Other Agency projects (like SOHO) have been used as a reference for costing purposes. The proposed spacecraft concept is a dedicated design for the SAM mission, and does not rely on parallel developments.

7.16 Risk

The Space Weather Service (SWS) is, from a risk point of view, to be handled differently from usual scientific missions, because the strongest requirement driving this analysis is the requirement for a continuous service of near real-time data. By this definition, success of the service depends on a successful delivery of near real-time data to the user Secondary benefits are not considered.

Within the scope of this study the risk assessment is limited to the risk of loss of service availability of the SAM spacecraft.

7.16.1 Requirements and Design Drivers

No requirement for the service availability is yet defined for the entire SWS and mapped into space segment dependability requirements. Therefore the risk assessment of the SAM space segment focuses on the dependability of the single satellite.

7.16.2 Assumptions and Trade-Offs

A single satellite carrying 4 instruments defines the baseline configuration. It is assumed that all instruments are needed over the intended in orbit lifetime of 5 years to provide a full service.

7.16.3 Baseline Design

The baseline design is a 3-axis stabilised spacecraft with a single redundant reliability structure for most of the units. However, for a 3-axis stabilised spacecraft the AOCS subsystem is more complex than the one proposed for the SWM, and fully redundant.

The power subsystem is based on a regulated bus design with internal redundancy on PCDU unit. The battery and solar array provide redundancy at cell level. The solar arrays are folded along the S/C side walls and released by thermal knives. No solar array drive mechanism is foreseen.

Mechanisms are used to point the high gain antenna towards Earth, to unfold the solar arrays, and to separate SWM and SAM after they are launched as a stack. SWM–SAM separation is performed by pyrotechnic devices.

The telecommunications subsystem is fully redundant.

The propulsion system is reduced to a reaction control system (RCS) based on a single-failure-tolerant monopropellant hydrazine blow-down system.

Thermal control is implemented by means of radiators operating at ambient temperature for the S/C and instrument electronics and dedicated cryogenic radiators for thermal control of CCD devices of the instruments. Heaters controlled by thermostats are used on the batteries, tanks and thruster.

7.16.3.1 Feasibility

The reliability block diagram of the present baseline is shown below. Assuming typical failure rates, the reliability of the baseline design is approximately 0.77 at the end of a 5-year orbit lifetime.

The major risk contributors to the system unreliability of 0.23 at the end of the lifetime are the DHS and AOCS subsystems.

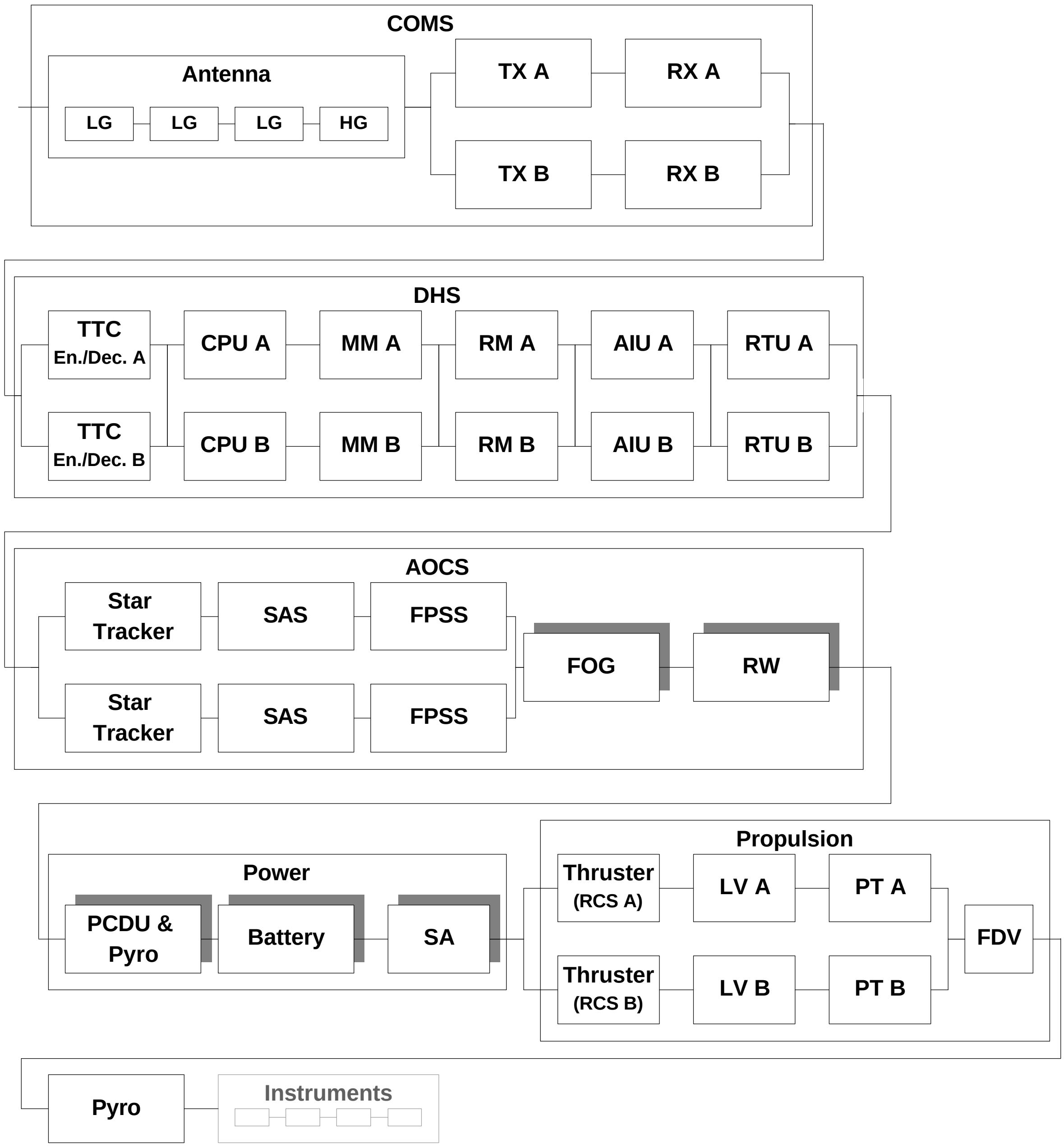


Figure 7-29: SAM Reliability Block Diagram

7.16.4 Summary

Similarly to SWM, the SAM mission can be considered to be close to a classical 'science' mission. It is based on available technology for the various subsystems of the SAM spacecraft. The proposed design baseline is capable of meeting the requirements, based on an expected reliability of about 0.8 at the end of the orbital lifetime.

In the context of the SW space segment SAM, is not considered to be able to provide a back-up capability for instruments that fail on one of the other satellites of the SW space segment, as the instruments carried on SAM are different from those on board IMM and SWM.

7.17 Cost Estimate

This document presents the SAM Platform and Payload Phase B and C/D cost estimate.

7.17.1 Main Costing Assumption

It is assumed that the industrial organisation for the SAM platform project involves a Prime Contractor, handling the detailed design at platform level. The Prime is also assumed to be responsible for Assembly, Integration and Test (AIT) activities support at Spacecraft System level.

At SAM platform level, a PFM approach has been selected. This is based on STM, EM, PFM model philosophy.

Generally platform and payload equipment is assumed to be off-the-shelf equipment or based on existing and available technology, as presented in the report. All cost estimates are based on references and cost estimation methods in line with the above general hypothesis. It is considered that the spacecraft design activities and equipment selection will be commensurate with the operational nature of the mission.

A five-year lifetime was taken into account. No geographical distribution constraints are included.

7.17.2 Cost Estimate Methodology

The following methods have been used, in descending order of preferred method:

- Reference to similar ESA missions;
- Reference to similar equipment/system level costs, taking into account the amount of new development required;
- Expert judgement from technical specialists in combination with similar equipment references, in the case that the amount of new development is extensive;
- Expert judgement from technical specialists only, if references are not available;
- Equipment cost models;
- The ESA internal, system level cost model RACE;
- System level cost relationships (for the Prime and Payload/Payload Contractor activities), based on recently observed relationships for relevant references.

7.17.3 Scope of the Cost Estimate

In accordance with the study requirements, the cost estimate covers:

- the SAM Platform
- Instruments (as far as information is available)
- Phase B and C/D costs of the mission
- the launch

Excluded are the ground system and operations costs.

Furthermore, the cost estimates are for the industrial costs only.

The SAM Platform Phase B and C/D cost estimate includes:

- A provision for the Phase B development costs
- The phase C/D costs, up to PFM
- Phase C/D equipment, software and platform level costs including Ground Support Equipment costs
- Spacecraft system level activity cost (Management & Control, Engineering, PA, AIT)
- Launch Adapter costs
- Platform Design Maturity provision

For the Payload, the Phase B and C/D cost estimate includes:

- A provision for the Phase B equipment development costs
- The phase C/D costs, up to PFM

The industrial cost is considered as the Prime Contractor offering a firm fixed price would see it. It covers the supply of the flight unit with the associated development models when applicable, the spares, the specific GSE and the user manuals. It also covers the Project Office cost of the equipment suppliers.

7.17.4 Phase B Cost Assumptions

The Phase B costs have been estimated based on the Phase B versus Phase C/D cost ratios for projects with a strong Prime Contractor involvement at subsystem level. The Phase B costs do not cover the pre-developments assumed to be part of Phase C/D.

7.17.5 Phase C/D Cost Assumptions

For the cost estimates, the development and Assembly, Integration and Test (AIT) of the platform are regarded as being a complete project on its own handled by the Prime Contractor at satellite level. All platform subsystem Project Office (PO), AIT and Ground Support Equipment (GSE) costs are therefore included at platform level.

7.17.5.1 AOCS

- The AOCS costs are primarily based on Mars Express costs.
- Prices have been estimated based on this reference but are adjusted with today’s market price trends.
- All equipment is assumed to be off-the-shelf with possible simple modifications.

7.17.5.2 Propulsion

The cost estimate for the propulsion system is mainly based on various prior ESA missions. Necessary adaptations have been taken into account. Further Project Office costs at sub-system level are presented, based on ratios observed on previous projects.

7.17.5.3 Electrical Power

- Solar Array costs are based on ESA internal Cost Estimating Relationships (CERs). Although the Si solar cells will be off-the-shelf equipment, the panel configuration will be unique. The cost estimate therefore assumes that a normal Solar Array development effort will be required.
- The PCU and PDU costs have been derived from similar items from prior ESA missions.
- The Rosetta battery was the reference for Li-Ion Battery cost.

7.17.5.4 Harness

- Harness costs were determined using ESA internal CERs. Since the harness architecture has to be newly developed, this has been taken into account in the cost estimate.

7.17.5.5 TT&C

The costs are mainly derived from the reference mission Herschel-Planck, on which minor equipment modifications were taken into account.

For the TT&C sub-system the procurement is proposed to demand PFM, EM and STM, and the EM equipment needed to assemble an Avionics Test Bench (ATB).

7.17.5.6 Data Handling

- The Data Handling System consists of a single box combining CDMU (Command and Data Management Unit) and RTU (Remote Terminal Unit).
- The CDMU is internally redundant.
- The data rate is low.
- For the cost estimate a partly customised off-the-shelf CDMU has been assumed.

7.17.5.7 Structure

The structure cost has been based on the ‘low cost mission’ ESA internal cost model.

7.17.5.8 Mechanisms

The items concerned are two solar array deployment mechanisms, an antenna pointing mechanism and the separation mechanism. The mechanisms costs were determined using ESA internal CERs.

7.17.5.9 Thermal Control

The Thermal Control Equipment is assumed to include only passive hardware such as radiators, paint and MLI. The Thermal Control Subsystem engineering activities such as thermal control analysis and configuration design are included in the Engineering cost at payload level. Other specific instrument thermal hardware is included within the payload costs.

7.17.5.10 On-Board Software

Both the Data Management Software and the AOCS Software are considered to be based on existing on-board software, with only the payload management being specifically developed for SAM.

The cost estimates for the SAM Data Management and AOCS Software are based on the costs for modified existing software on other ESA missions.

7.17.5.11 GSE

The cost estimate for Ground Support Equipment (GSE) covers the costs for all Electrical and Mechanical GSE required for the platform. It has been taken into account that the GSE will be mainly based on existing hardware and designs. Accordingly a standard ratio observed on past projects has been applied.

7.17.5.12 Platform Assembly, Integration and Test

The platform AIT cost estimate includes the costs for all platform mechanical and electrical integration activities and tests, as well as the mechanical mating of the platform and the payload. The cost estimate is based both on a cost estimate relationship and on an independent AIT planning assessment performed within the CDF, with which the results are in close agreement.

7.17.5.13 Project Office Activities

The Project Office costs at subsystem and platform level include the costs for

- Management and control (including overheads on subcontracts)
- Product assurance
- Engineering and documentation, including payload interface engineering both at system and subsystem level, except for propulsion

7.17.5.14 Payload

The instrument cost assessment is characterised by the rather limited amount of available reference material and technical data on the instruments.

It has been assumed that institutes rather than industry will procure the instruments.

The cost estimates are based on similar instruments or equipment with matching technology.

The WLC, EUVI and XRP cost estimates are based on the Solar Coronal Imaging Package on STEREO.

The monitor cost estimate is rooted in equipment and sensors on XMM.

To adapt the different cost, various ESA internal cost models have been used.

It has to be noted that for more detailed estimates, further studies are necessary.

The cost estimates are based on similar instruments or equipment with matching technology. Different ESA internal cost models have been used.

It has to be noted that for more detailed estimates, further hypotheses are necessary.

7.17.5.15 Design Maturity Margins

The Design Maturity Margins account for unknown design aspects not yet identified at the level of this feasibility study. These provisions are no risk margins (i.e. cost impacts due to the realisation of a stochastic event) and must be considered as part of the total industrial cost as well as of the payload cost.

Design Maturity Margins:

- 10% for platform
- 20% for payload

7.17.5.16 Launch Adapter

The Launch Adapter is considered to be standard off-the-shelf equipment. Therefore no development activities or development models are taken into account in the cost estimate.

7.17.5.17 Launcher

The cost for the Soyuz-Fregat dual launch (SWM and SAM together) is included in the SAM estimate.

7.17.6 Cost Risk Estimate

No specific cost risk estimate has been performed. This will have to be accounted for as part of the ESA level contingencies.

7.17.7 Insurances

- Satellite:
  Due to the operational nature of the mission, an insurance amount of 7.5% has been considered, where this value is based on recurrent market prices.

- Launcher:
  Launcher insurance cost is assumed to be 7.5% of the launch cost by similarity to satellite insurance.

7.17.8 Qualitative Cost Assessment

This estimate is based on a fully competitive environment with strong involvement and motivation of the Prime Contractor.

No Geographical Distribution effect is accounted for.

So far, no reliability and availability figures have been expressed as part of the requirements of such an operational mission. Depending on these figures, the spare philosophy and components quality level may need to be revised.

8. Overall Ground Segment and Operations Concept

The ground segment key tasks for Space Weather can be summarised as follows:

- Spacecraft command, monitoring and control;
- Receive data from instruments and distribute them to users (the Space Weather community) in real time;
- Process payload data for off-line distribution and archiving;
- Ground Segment operation in order to assure the spacecraft health and product distribution to the users in an effective manner and in the correct timeline.

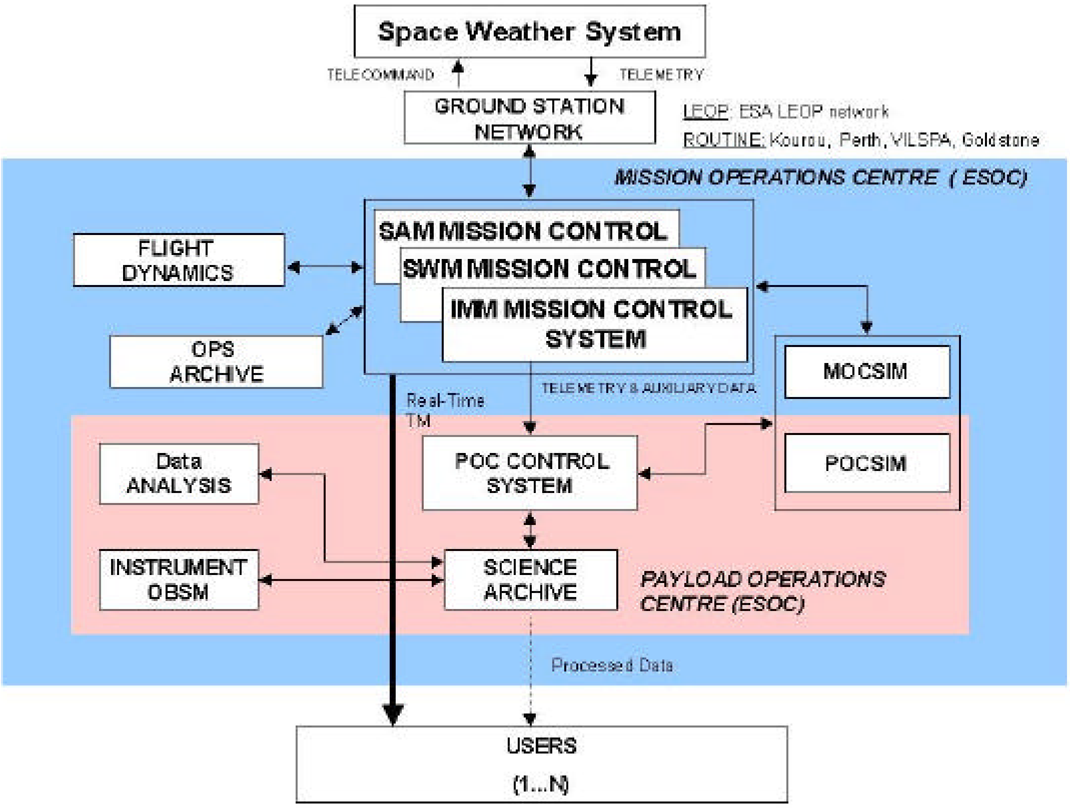


Figure 8-1: Overall Space Weather Ground Segment Architecture

The key requirements for the Space Weather operations are:

- Real-time payload data retrieval and distribution
- Perigee data to be stored and transmitted at a later time for IMM
- No strict orbit maintenance requirements for IMM – manoeuvres will be done to maintain the relative phases of the four spacecraft. Nominal halo orbit maintenance for SWM and SAM
- Relatively low data rate from all spacecraft
- X-band used for up and down link for all spacecraft

- Both SWM and SAM are in a halo orbit around the L_1 equilibrium point of the Sun-Earth system

8.1 Ground Segment Facilities and Services

For the Space Weather Ground Segment some new facilities have to be built, but the existing ESA infrastructure, conveniently adapted to the mission requirements, will be reused when possible.

The ground segment selected for the baseline Space Weather architecture will comprise:

- The ground stations network comprising new TM antenna facilities in Kourou (IMM), Perth or New Norcia (IMM, SWM/SAM), Maspalomas (SWM/SAM) and Papeete (SWM/SAM);
- The Mission Operations Centre (MOC) at ESOC consisting of the mission control system for all the satellites, the flight dynamics system, the operations simulator and the operations archive;
- The Payload Operations Centre (POC) comprising the payload operations system, the science archive, the data analysis module, the on-board software maintenance (OBSM) and the payload operations simulator.

8.1.1 Ground Stations Network

The following figure shows the ground stations (red dots denote stations that are for LEOP + routine operations, green dots are for LEOP only).

Figure 8-2: Space Weather Ground Stations

8.1.1.1 LEOP

During the Launch and Early Orbit Phases of the different spacecraft, the ground stations in Kourou and Perth will be used to track and command the satellites and to receive housekeeping telemetry. The whole ESA LEOP network will be available for the Space Weather LEOPs. ESA's LEOP network will have completed its upgrade to X-band by mid 2004.

The ESA LEOP network consists of the following Ground Stations:

- Kourou (French Guyana)
- Malindi (Kenya)
- Maspalomas (Gran Canaria, Spain)
- Perth (Australia)
- Redu (Belgium)
- Villafranca (Spain)

8.1.1.2 Operational Phase

During the operational phase, a total of seven TM receiving antennas (distributed amoung Kourou, Perth, Maspalomas and Papeete) will retrieve continuous real time service data. Due to the real-time constraints of the mission(s), dedicated TM receiving antennas shall be built at the sites. The complete Ground Station setup will be as follows (all antennas dedicated to Space Weather):

Ground Station	IMM	SWM/SAM
Kourou	8m TM	
	8m TM	
Perth/New Norcia	8m TM/TC	15m TM/TC
	8m TM	
Maspalomas		15m TM
Papeete		15m TM

Table 8-1: New Antennas Required for Space Weather

One of the IMM Perth antennas and the Perth SWM/SAM antenna need to be configured with ranging capabilities.

The ground stations will communicate with the mission operations centre via the operational network OPSNET. OPSNET is a closed wide area network for TC, TM, tracking data, station monitoring, control data and voice. The bit rates are as indicated in the following table:

	Uplink	Nominal downlink	Data dump
IMM	2 kbps	13 kbps	170 kbps
SWM	2 kbps	9 kbps	None required
SAM	2 kbps	35 kbps	None required

Table 8-2: Bit Rates for Uplink and Downlink

Due to the real time constraint, low mission planning activity is foreseen as the spacecraft will ideally operate non-stop during the lifetime. From the point of view of orbit maintenance, the

IMM constellation will only require phase correction, and for SWM and SAM in a halo orbit around L_1, the orbit maintenance is performed approximately once a month. A single TM/TC antenna for IMM and another TM/TC antenna for SWM/SAM can thus manage commanding for the whole set of satellites.

8.1.2 The Mission Operations Centre (MOC)

The complete Space Weather system will be operated by ESOC as one consolidated system. During LEOP and commissioning operations of the different missions, the Main Control Room (MCR) augmented by the Flight Dynamics Room (FDR), and the Project Support Room (PSR), will be used. During the routine phase, a Dedicated Control Room (DCR) and the FDR will be used for mission control.

The mission control team will be composed of:

- Dedicated Space Weather staff
- Experts from the different fields involved in spacecraft control on a shared basis with other missions

The interface with the users community will be two-fold:

- TM data stream will be directed in real time from the control centre to the customer
- Processed data will be delivered off-line to the payload data users

8.1.2.1 Flight Control System, Computer Facilities and Network

The Flight Control System will be based on SCOS 2000 and is in charge of all the activities related to satellite monitoring and control. The functionality of SCOS 2000 comprises:

- Telemetry reception, real time TM processing, quality checking, archiving and distribution
- Telemetry analysis facilities for status/limit checking and trend evaluation
- Command preparation and execution, and on-board software maintenance

As part of the computer facilities, a set of workstations hosting SCOS 2000 will be used for Flight Control purposes.

Three flight control software systems will be developed:

- – IMM – one system for all 4 spacecraft
- – SWM
- – SAM

These three systems will be integrated into one overall Mission Control System.

The flight dynamics software ORATOS plus an interface with SCOS 2000 will also be hosted on a workstation.

The simulators will be developed using the SIMSAT-NT infrastructure operating on Windows-XP based computers. They will be used for ground segment verification, for staff training and during operations. Similarly to the flight control system, three separate simulators will be developed for the three missions.

All computer systems in the control centre will be redundant with common access to data storage facilities and peripherals. All computer systems will be connected by Local Area Network (LAN) to allow transfer of data at sufficient speed and to allow joint access to data files.

8.1.3 Payload Operations Centre (POC)

A Payload Operations Centre will be set up in ESOC for the Space Weather mission. Existing infrastructure will be used wherever possible. The POC will interface with the MOC for telemetry and auxiliary data, and it includes the following main functions:

- Payload control system
- Science archive
- Payload simulator
- Payload data analysis
- On-board software maintenance for the instruments
- Distribution of processed data

8.2 Mission Operations Concept

The operations concept for Space Weather comprises the spacecraft and the instruments operations.

The spacecraft operations mainly consists of:

- spacecraft monitoring and control
- orbit and attitude determination and control
- reduced mission planning activities
  (as all instruments will be producing real-time data for the duration of the mission)

The instrument operations mainly consists of:

- data acquisition (science telemetry)
- real time transmission to the users

The Space Weather operations concept includes mission operations for IMM, SWM, and SAM. Mission operations start at the separation from the launcher of the first spacecraft to be launched and will continue until the end of life of the last remaining spacecraft. The activity level will reduce as the project winds down.

8.2.1 Spacecraft Operations and Mission Planning

Space Weather will be operated by the Mission Operations centre in ESOC according to the procedures established in the Flight Operations Plans (FOP) for each mission.

The spacecraft will be commanded and monitored and the payload data retrieved and distributed to the users in real time. The commanding is kept at a low level according to the station keeping concept (approximately 1 manoeuvre/month for satellites in halo orbits, and only phasing maintenance for the IMM constellation). Actions will be taken when anomalies are noticed from the analysis of the spacecraft trends.

The Payload Operations Centre will meet the needs of the data users concerning the data analysis, the data archiving and maintenance of all on-board software.

24 h/day operations, 7 days a week will be maintained from the mission operations centre with TT&C in X-band and payload downlink also in X-band during the duration of the passes (approximately 12 h/antenna/station for IMM, and a yearly average of also approximately 12 h per station for SWM and SAM).

8.2.2 Engineering Support

The engineering support will plan the orbit and attitude manoeuvres when necessary, and will manage the spacecraft subsystems and consumables according to the analysis of the spacecraft trends. The maintenance services of the Ground Stations, the flight software, and the simulator will also be part of the engineering support.

8.3 Other Options Considered

During the Space Weather Study, two alternative concepts for the SAM mission were considered as options from the Ground System point of view.

The first one considered the GEO as the operational orbit for the SAM spacecraft, combining the Sun-monitoring objective with a data relay function (option C).

The second considered placing the SAM spacecraft in an Earth-like heliocentric orbit, at a point trailing 10º behind the Earth (10º Earth trailing orbit) (option D).

8.3.1 SAM in GEO – Data Relay Concept (Option C)

According to the mission requirement of continuous Sun observation, if GEO is selected as the operational orbit for SAM, then 2 spacecraft are needed to avoid loss of coverage of the Sun during the Spring and Fall equinoxes (each eclipse season lasts 46 days, the maximum eclipse duration being 71.5 minutes). The spacecraft should be separated in GEO by an arc of at least 17° to ensure that both of them are not in the shadow area at the same time.

In this case the SAM spacecraft could be used not only to monitor the solar activity but also as Data Relay Satellites, collecting the data from SWM in its halo orbit around L_1 and from the IMM constellation and transmitting them to a single location on Earth. From the point of view of the Ground System this approach simplifies the ground stations network to one ground station with two TM antennas (one for each GEO satellite). As the commanding requirements are kept at a low level, an existing ESA TC antenna with visibility of the satellites could uplink the necessary housekeeping commands.

Apart from the simplification of the Ground segment there would be no need to keep the IMM constellation in synchronous orbits and they could have any phase in their orbits. In addition, all of the IMM spacecraft would be able to provide 100% real-time coverage without the >3000 km altitude constraint.

On the other hand the complexity and the cost of the space segment will considerably increase, and intersatellite communications will require a new and potentially expensive space architecture.

Figure 8-3 shows a schematic of the relay option.

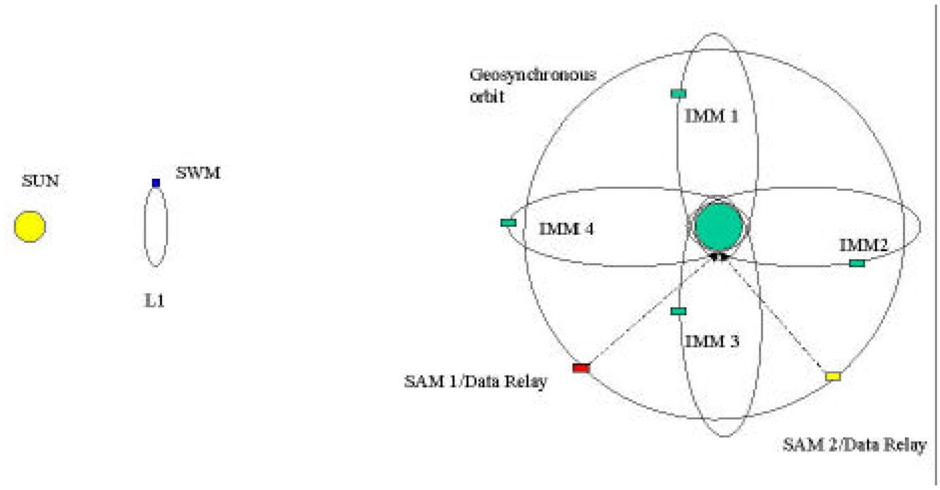


Figure 8-3: Space Weather Relay Option

8.3.2 SAM on 10º Earth Trailing Orbit (Option D)

This kind of operational orbit represents an advantage from the point of view of forecasting possible effects of the Solar Activity on the Earth as the coronal mass ejection can be better observed for a spacecraft which is not on the Earth-Sun line (it is then also possible to see the direction of the CME). The transfer orbit is shown in Figure 8-4. The transfer orbit lasts a little bit longer than 1 year and the final position of the SAM spacecraft is the 10º Earth trailing orbit.

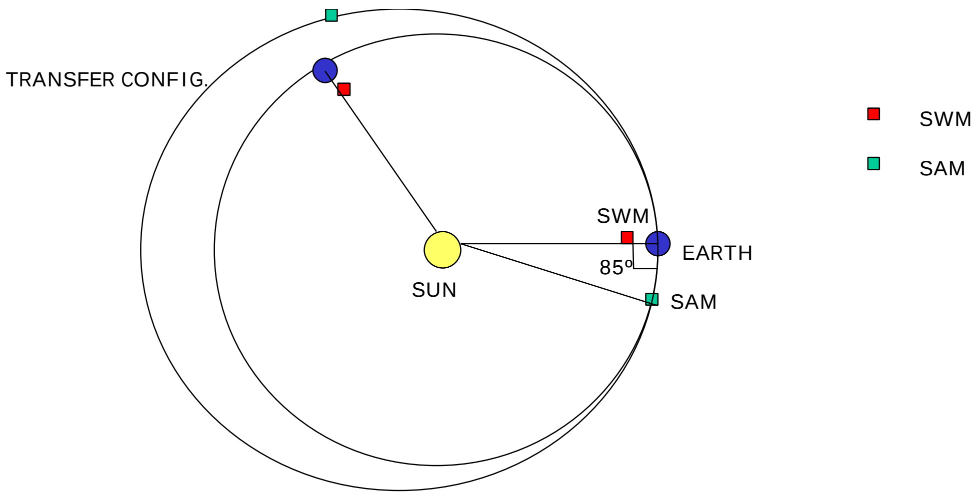


Figure 8-4: Transfer of SAM to Final Position in the 10º Trailing Orbit

From the point of view of the Ground Segment this operational orbit for SAM, in combination with the SWM's halo orbit around the L_1 point, means that four ground station locations approximately 90º apart in geographic longitude are needed to cope with the continuous data flow in real time. In each location a dedicated TM receive antenna should be located. A possible selection of the sites could be Papeete, Kourou, Malindi and Perth.

If data is downlinked in X-band, then 35m antennas will be needed. The existing Deep Space facility of ESA in New Norcia close to Perth could be used provided that time can be shared with the Rosetta and Mars Express missions. An additional three antennas would have to be built. Figure 8-5 shows the coverage from Papeete, Kourou, Malindi and Perth/New Norcia over one year of the SAM mission on the trailing orbit. The minimum elevation considered for acquisition of signal (AOS) has been 10º. The dark grey zones in the picture indicate that two stations simultaneously are covering the spacecraft. There are no visibility gaps.

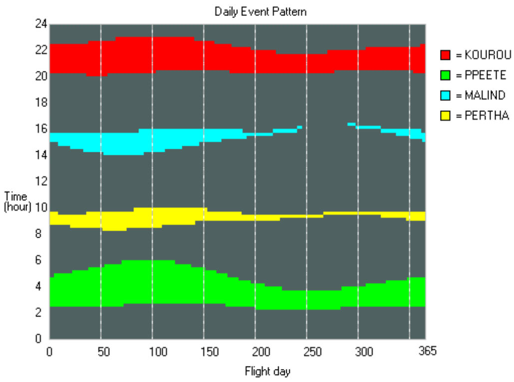


Figure 8-5: Ground Coverage from Recommended Stations in X-Band

If, however, data is down linked in Ka-band, then less expensive 15 m TM receiving antennas can be used. The minimum elevation angle considered for AOS in Ka-band is 30º which causes visibility gaps in the transition from Perth to Papeete (the white patch in Figure 8-6). An alternative is to substitute the antenna in Perth by an antenna in Darwin, as is shown in Figure 8-7.

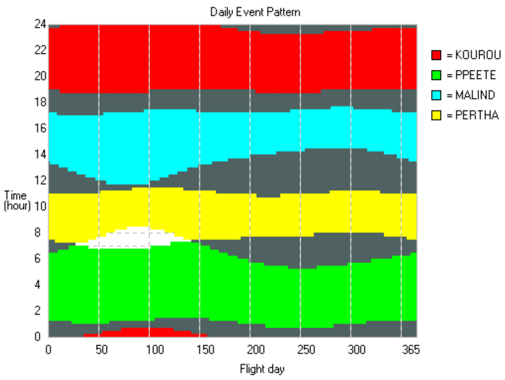


Figure 8-6: Ground Coverage in Ka-Band

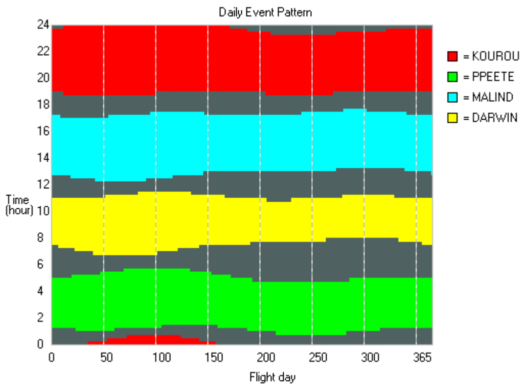


Figure 8-7: Ka-Band Coverage with Darwin

In case of contingency, if a spacecraft transmits through the low gain antenna then a 35 m dish is needed on ground, and communication is done in X-band. For this, the existing facility in New Norcia could be used.

9. Overall Simulation

A SpaceWeather prototype Project Test Bed (PTB) has been developed in the frame of the Space Weather study.

The PTB implementation consists mainly of a system simulator containing functional models of the spacecraft and their subsystems, the spacecrafts' orbital environment, and the Sun-Earth magnetic environment. Furthermore, external applications (such as a 3-D visualiser, a 2-D camera showing snapshots of the sun and a 2-D Earth map) are connected to the simulator.

The simulation has taken as input the data provided by Mission Analysis and by the customer. This data consists of various sets of Kepler elements for the satellite orbits at different phases of the mission for the IMM constellation, and the position and velocity in the GSE frame of SWM and SAM. The graphical model of the satellites has been imported from the Configuration model and has been integrated into the 3-D visualisation. The mission timeline has been taken from the system model and sample data from the customer. This data has been implemented in the simulator.

During the study the PTB has been mainly used for illustration purposes, and for developing a scenario of measurements of a space weather event from its solar origin to its geomagnetic consequences. However, tools developed for the PTB can be used for the following tasks:

- Visualisation of the mission
- Verification of the suitability of the selected Space Weather architecture to a space weather service
- Analysis of system design parameters

The following sections will describe these uses for the simulation in more detail.

9.1 Visualisation of the Space Weather Mission

The PTB is capable of simulating the SpaceWeather mission using a 3-D display. The following items are available in the 3-D visualisation:

- Satellite model as imported from Configuration (for each satellite)
- Satellite orbit (for each satellite)
- The environment: Earth, Sun, Moon and stars
- CME, magnetic field lines, magnetopause and electron injections

Because the PTB is implemented as a system simulator, it can be used to verify certain design parameters, combining inputs from various subsystems.

In the Space Weather study, this can be performed for the computation of values measured by the satellites depending on the time and their position. These values can be computed from tables and functions provided by the customer, and then plotted graphically, to simulate the potential user output.

Some examples of the 3-D visualisation are given below.

Figure 9-1 shows the nominal trajectories of the IMM and SWM spacecrafts. SWM is in an L_1 halo orbit, while IMM is a cluster of four spacecraft in elliptical equatorial orbits separated by 90 degrees, and phased two by two, 180 degrees apart. The two lower panels show simulated data in quiet time radiation
belts.

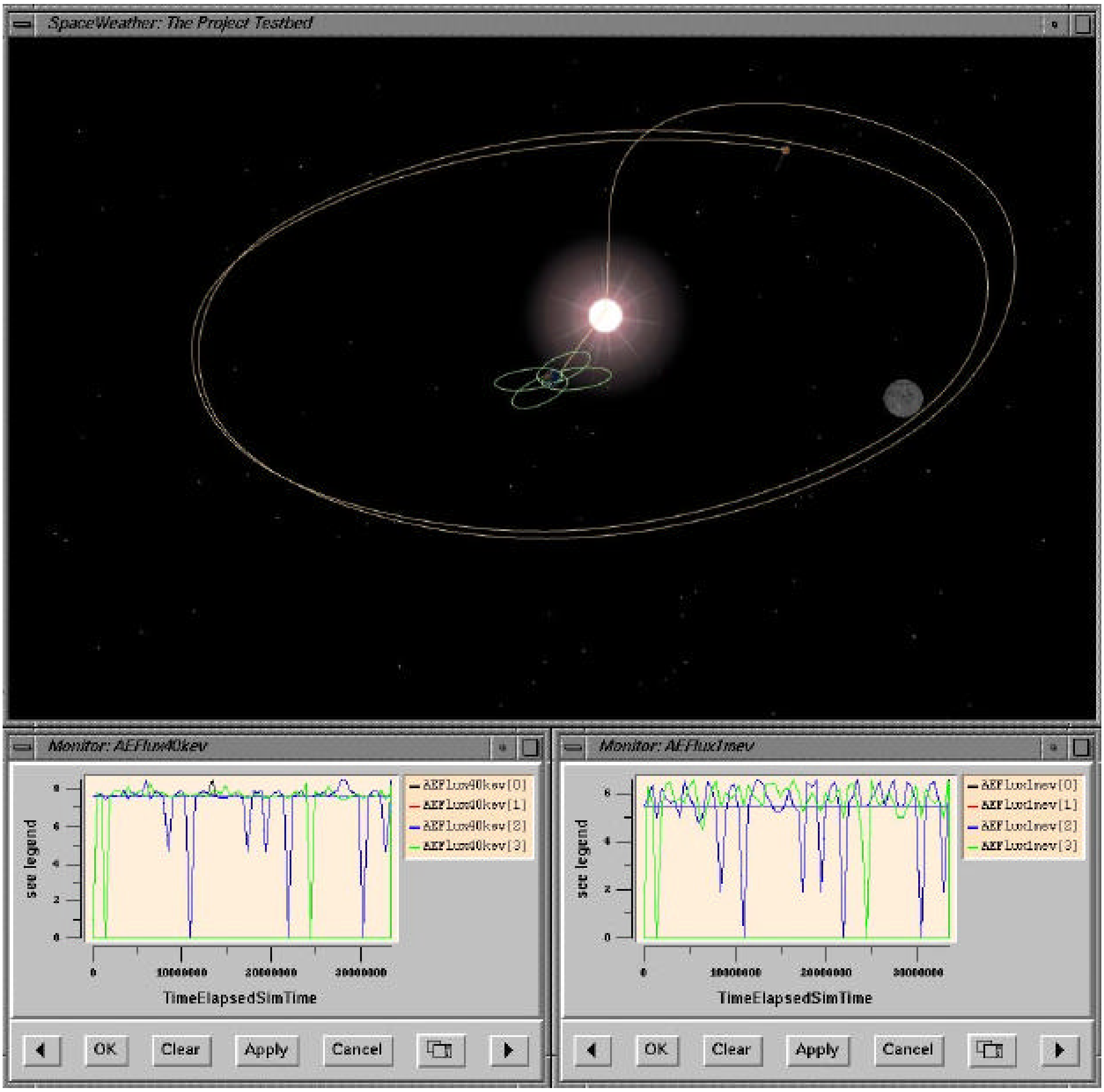


Figure 9-1: Orbits of IMM and SWM/SAM

9.2 Verification of the Spacecraft Operations

The PTB is capable of simulating the mission timeline, except for the orbit acquisition manoeuvres of the IMM satellites. Simulation of ground segment operations is not included.

In order to verify that the selected spacecraft architecture was suitable for the space weather service, most effort has been spent on implementing a timeline able to show a typical space weather event:

- A Coronal Mass Ejection (CME) is detected
- The particle emission reaches the Earth and affects the Earth's magnetic field and magnetopause
- Afterwards an electron injection belt can be detected

Figure 9-2 shows the Sun following the CME onset. The CME onset data shown originates from the SOHO EIT and LASCO cameras and is used to illustrate the type of data that would be available from the SAM spacecraft (shown in the two upper right panels). The plots in the lower two panels show measurements of x-ray flux as would be detected by the SAM x-ray monitor, and solar protons of energy >30 MeV as would be detected by IMM's high energy proton monitors. Sample data is used courtesy of SOHO and GOES.

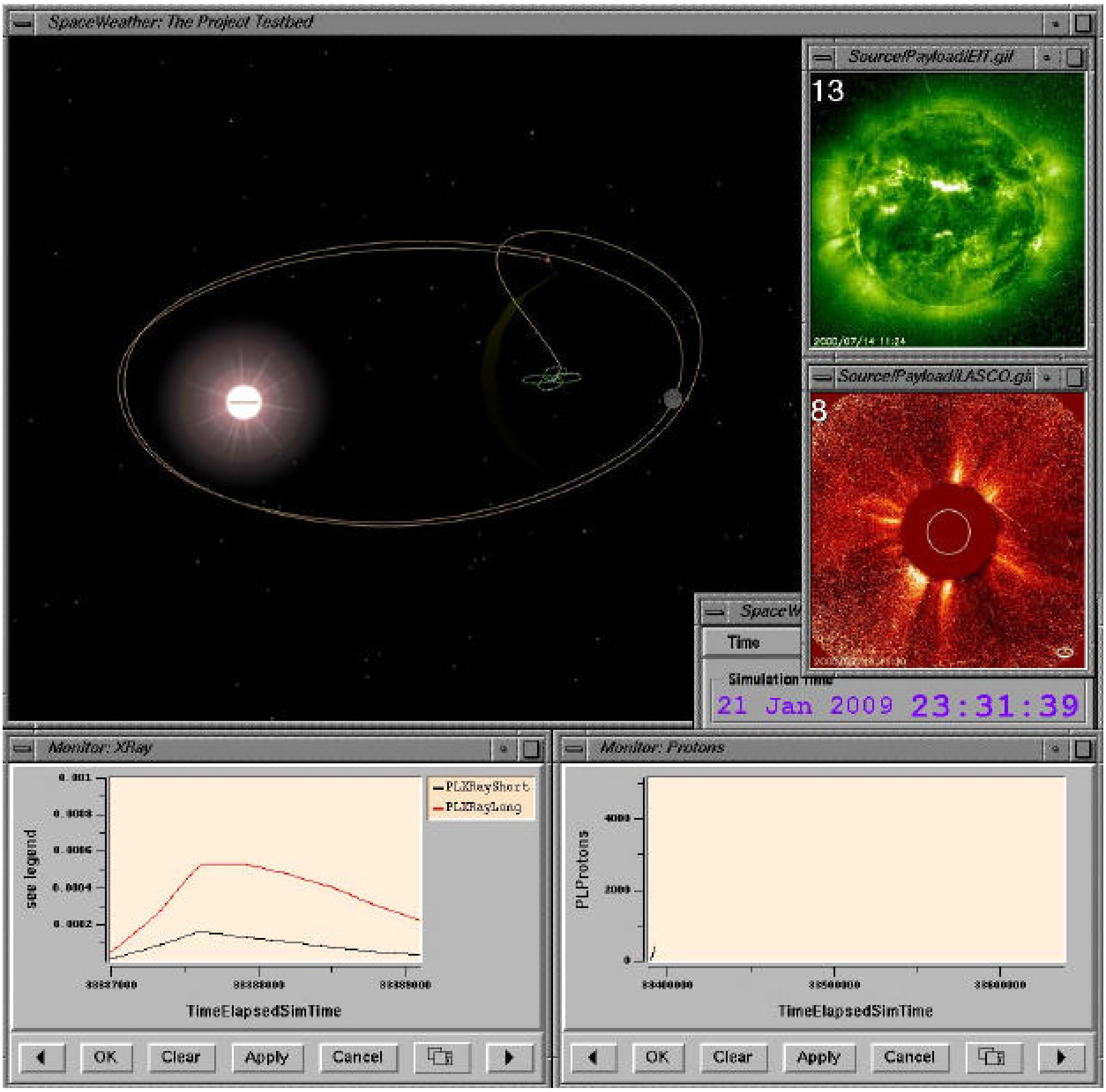


Figure 9-2: CME Captured by Cameras on SAM

Figure 9-3 shows the CME (orange) reaching the earth, and how the magnetopause (yellow) is compressed as a result. The two plots at the bottom show proton flux and solar wind pressure measurements, as would be made by IMM and SWM respectively, and how they would be seen by the user of the Space Weather service.

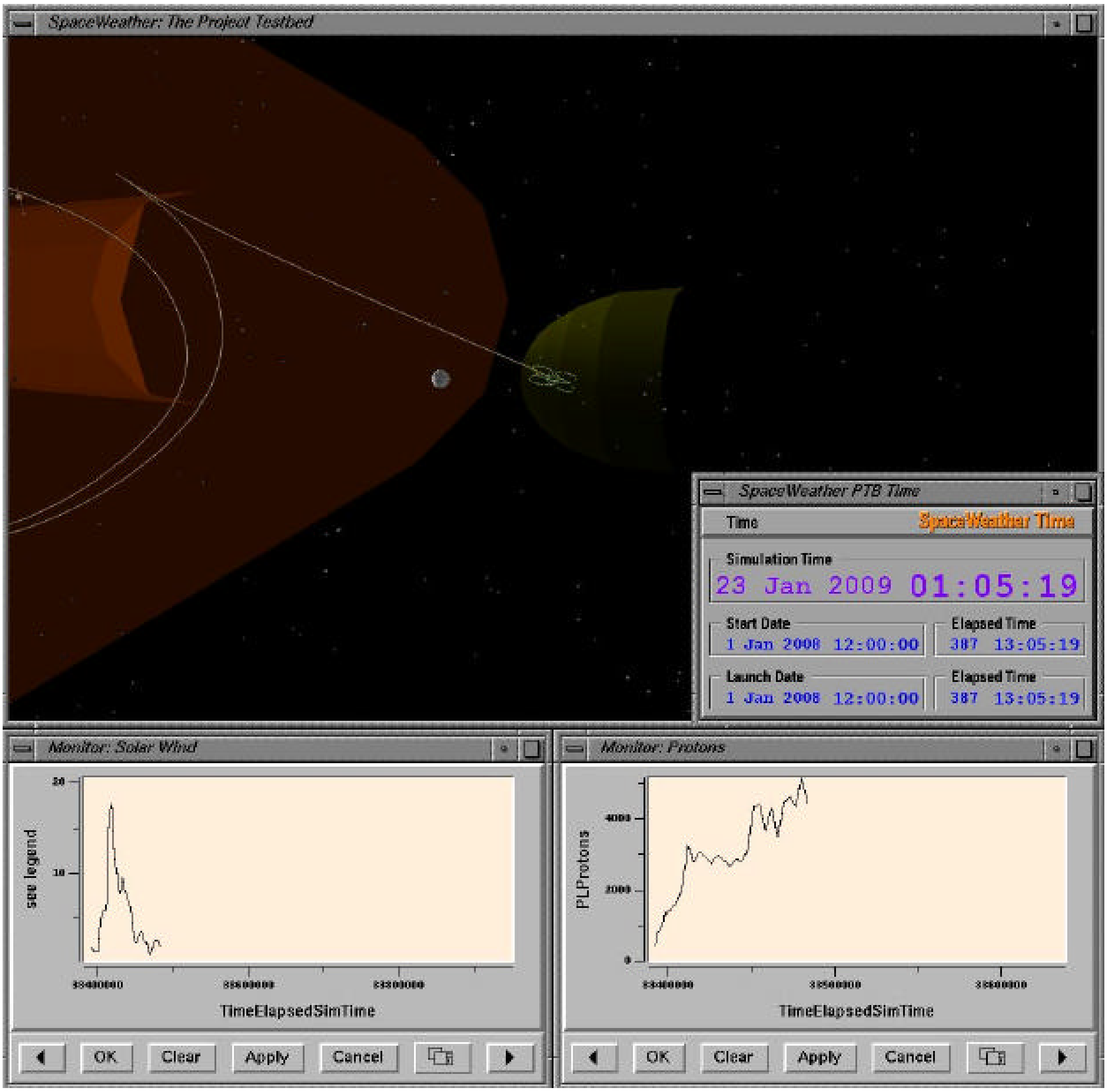


Figure 9-3: Sample Data Showing an Interplanetary CME Reaching the Earth

Figure 9-4 shows the simulation of an intensification of electron flux in the 40 keV energy range due to electron injection during a storm. The plots show the solar wind pressure and B-field (the By and Bz component of the magnetic field) as measured by SWM, and the electron flux as measured at both 40 keV and 1 MeV by IMM.

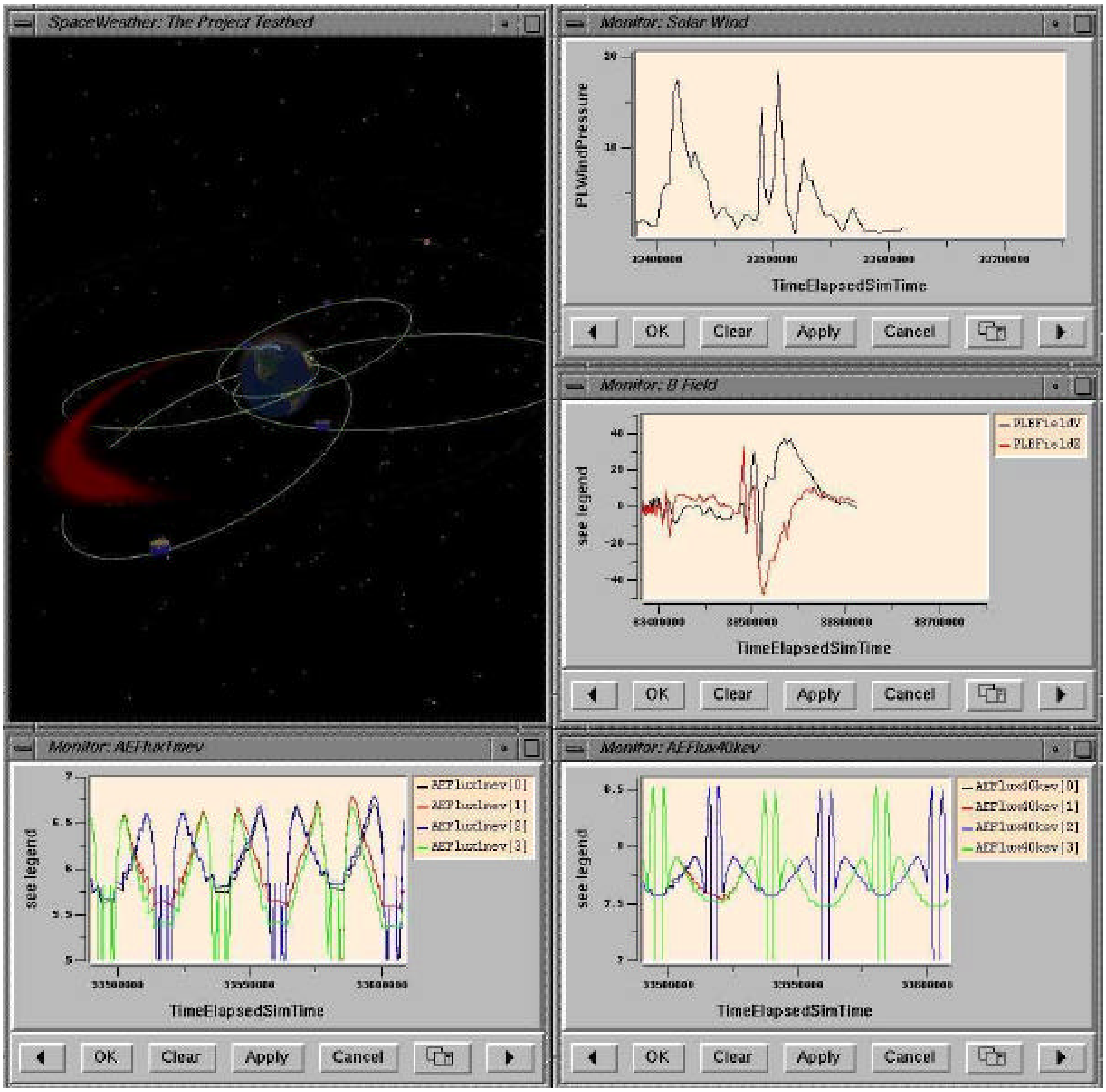


Figure 9-4: Electron Injection as would be Measured by IMM

9.3 Data Sources

- Radiation belt measurements and visualisation are simulated with a numerical model.
- Solar wind pressure, B-field, x-ray and protons are from measurements taken by the ACE satellite during July 2000.
- Magnetopause is simulated and visualised based on the Fairfield Meridian 4o model of the terrestrial magnetopause.
- Sun snapshots were taken by EIT and LASCO cameras of SOHO.

10. Conclusions

In this report, a reference architecture for a Space Weather Space Segment has been selected and analysed in detail. It represents the best compromise between design simplicity, cost, and satisfaction of a minimum set of user requirements.

Several options, which could increase either the cost effectiveness or the user requirement satisfaction, have been proposed and partially analysed to assess their feasibility.

In the proposed baseline architecture, no technical showstopper or new technology development (apart from some instruments) has been identified and it can be concluded that this Space Segment is well within Europe's capability.

In the space segment, the most complex element is the constellation of IMM, in terms of launch and operations; the simplest and cheapest is the SWM with its low mass and small number of operation modes.

The total cost, including instruments and operations, of the baseline architecture exceeds the target of 300 Meuro by about 50%. However, several countermeasures are proposed to reduce the overall cost of the programme with little impact on the user requirement satisfaction.

More investigation needs to be performed in a later phase to establish the cost effectiveness of these options.

The cost of instruments has been based on the rather scarce data available, and is therefore subject to a large error. It is estimated that the current evaluation is on the optimistic side.

From a programmatic point of view, the first feasible date for the deployment of the pre-operational system appears to be 2007.

Finally, it should not be forgotten that the starting point of this study was a subset of components of a space weather space segment proposed by the Alcatel and RAL consortia, while a complete space weather system should include other components, especially with auroral, ionospheric and atmospheric monitoring functions.

Appendix A: References

[RD1] Space Segment Definition and Analysis
ESA Contract 14070/99/NL/SB, Alcatel Space Industries, 2002

[RD2] Space Weather Space Segment Options
ESA Contract 14069/99/NL/SB, Astrium, 2002

[RD3] General information on space weather
www.estec.esa.nl/wmwww/spweather

[RD4] STRV-1c & d Satellite Architecture Design Document
1CD/MGT/SPEC/0092/4.1, A. Cant, H.B. Simpson,
DERA January 1999

[RD5] Space Weather CDF Study Cost Analysis Report
CDF-11(B), December 2001

[RD6] SWAN/SOHO UV background data
http://sohowww.nascom.nasa.gov/data/summary/swan/

[RD7] The Rockot Launch System – The Commercial Solution for Low-Cost Planetary Missions
P. Freeborn, M. Kinnersley & V.E. Viertel
IAA-L-0703

Appendix B: Abbreviations and Acronyms

A5	Ariane 5
AIV	Assembly, Integration and Verification
AOCS	Attitude and Orbital Control System
AOS	Acquisition of Signal
APM	Antenna Pointing Mechanism
APME	Antenna Pointing Mechanism Electronics
ASAP5	Ariane 5 Structure for Auxiliary Payload
ASIC	Application Specific Integrated Circuit
AST	Autonomous Star Tracker
ATB	Avionics Test Bench
BB	BreadBoard
BCR	Battery Charge Regulators
BDR	Battery Discharge Regulators
BPSK	Bi-Phase Shift Keying
CCD	Charge Coupled Device
CDF	Concurrent Design Facility
CDMU	Control & Data Management Unit
CER	Cost Estimating Relationship
CME	Coronal Mass Ejection
COTS	Commercial Off-The-Shelf
CPU	Central Processing Unit
CRS	Coil Radio Spectrograph
CTC	Cost to Completion
CTM	Collapsible Tubular Mast
DC	Direct Current
DCR	Dedicated Control Room
DHS	Data Handling (Sub)system
DOD	Depth Of Discharge
DPU	Data Processing Unit
DRAM	Dynamic Random Access Memory
EES	Earth Exploration Service
EFF	EFFiciency
EIRP	Effective Isotropic Radiated Power
EM	Engineering Model
EMC	ElectroMagnetic Compatibility
EOL	End Of Life
ESOC	European Space Operations Centre
FDIR	Failure Detection, Isolation and Recovery
FDR	Flight Dynamics Room
FM	Flight Model
FOV	Field Of View
G/T	[antenna] Gain to [system] noise Temperature (GS figure of merit)
GEO	Geostationary Earth Orbit
GPS	Global Positioning Satellite
GRIS	GPS Receiver Ionospheric Sounder

GS	Ground Station
GSE	Geocentric Solar Ecliptic
GSLV	Geostationary Satellite Launch Vehicle (Indian launcher)
GSP	General Studies Programme
GTO	Geostationary Transfer Orbit
HDRM	Hold-Down and Release Mechanism
HEM	High-Energy particle Monitor
HGA	High Gain Antenna
HK	HouseKeeping
HOP	High Output Paraffin (actuator)
IMM	Inner Magnetosphere Monitor
LEO	Low Earth Orbit
LEOP	Launch & Early Operations Phase
LET	Linear Energy Transfer
LGA	Low Gain Antenna
LHCP	Left Hand Circular Polarisation
LMM	Local Mass Memory
MAG	MAGnetometer
MCR	Main Control Room
MEM	Mid-Energy particle Monitor
MGA	Medium Gain Antenna
MGSE	Mechanical Ground Support Equipment
MLI	Multi-Layer Insulation
MOC	Mission Operations Centre
NRZ	Non-Return to Zero
OBSM	On-Board Software Maintenance
OH	Optical Head
OSR	Optical Surface Reflector
OTS	Off-The-Shelf
P/L	Payload
PCU	Power Conditioning Unit
PDU	Power Distribution Unit
PFL	Probability of Frame Loss
PFM	ProtoFlight Model
PM	Phase Modulation
POC	Payload Operations Centre
PSLV	Polar Satellite Launch Vehicle (India)
PSR	Project Support Room
PSS	Power SubSystem
PVA	PhotoVoltaic Assembly
RAL	Rutherford Appleton Laboratory
RCS	Reaction Control Subsystem
RF	Radio Frequency
RHCP	Right Hand Circular Polarisation
RU	Remote Unit
S/C	SpaceCraft
SA	Solar Array
SAM	Solar Activity Monitor
SAR	Solar Array Regulators

SEU	Single Event Upset
SIP	Support Instrument Package
SM	Structural Model
SO	Space Operations
SR	Space Research
SRM	Solid Rocket Motor
SSO	Sun Synchronous Orbit
STM	Structural and Thermal Model
SW	SoftWare
SWM	Solar Wind Monitor
SWS	Space Weather Service
TBC	To Be Confirmed
TC	TeleCommand
TID	Total Ionising Dose
TM	TeleMetry
TPM	Thermal Plasma Monitor
TWTA	Travelling-Wave Tube Amplifier
UV	UltraViolet
WAVE	Waves instrument