# ESA SPACE WEATHER PROGRAMME STUDY

# WP 2400 REPORT

# SPACE SEGMENT DEFINITION AND ANALYSIS

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## *TABLE OF CONTENTS*

1. INTRODUCTION ..... 6
*1.1 Applicable and reference Documents ..... 7*
*1.2 Acronyms ..... 8*
2. SYSTEM ARCHITECTURE ..... 9
*2.1 System requirements ..... 9*
*2.2 System options ..... 10*
*2.3 Direct downlink to ground ..... 11*
*2.4 Data relay strategies ..... 14*
*2.5 System options summary ..... 21*
*2.6 System scenarios selection ..... 24*
3. ELEMENT DESIGN - FULL SCALE SPACE SEGMENT ..... 28
*3.1 Solar observer ..... 28*
*3.2 Upstream Monitor ..... 37*
*3.3 Radiation Belt Monitors ..... 42*
*3.4 Ionospheric monitors ..... 51*
4. ELEMENT DESIGN - MEDIUM SCALE SPACE SEGMENT ..... 67
*4.1 Combined Solar Observer - Upstream Monitor ..... 67*
*4.2 Radiation Belt Monitors ..... 71*
*4.3 Ionospheric Monitors ..... 74*
5. ELEMENT DESIGN - MINIMUM SCALE SPACE SEGMENT ..... 76
*5.1 Solar Observer ..... 76*
*5.2 Upstream Monitor ..... 78*
*5.3 Radiation Belt Monitor ..... 78*
6. POSSIBLE ADDITIONS AND FUTURE DEVELOPMENTS ..... 78
*6.1 Magnetospheric constellation ..... 78*
*6.2 Solar wind monitor upstream of L1 ..... 79*
*6.3 Solar wind monitor downstream of L1 ..... 80*

## *TABLE OF FIGURES*

FIGURE 2-1- DATA TRANSMISSION WITH GEOSYNCHRONOUS RELAY SATELLITE ... 15
FIGURE 2-2 - DATA FLOW ARCHITECTURE, DEDICATED RELAY, OPTION 1 ... 16
FIGURE 2-3 - DATA FLOW ARCHITECTURE, DEDICATED RELAY, OPTION 2 ... 16
FIGURE 2-4 - DATA FLOW ARCHITECTURE, INTEGRATED SO + RELAY ... 18
FIGURE 2-5 - GEO RELAY COVERAGE ... 18
FIGURE 2-6 - GTO RELAY COVERAGE (1) ... 20
FIGURE 2-7 - GTO RELAY COVERAGE (2) ... 20
FIGURE 3-1 - GEO SO VIEW (1) ... 30
FIGURE 3-2 - GEO SO VIEW (2) ... 31
FIGURE 3-3 - FULL SCALE SEGMENT - DATA FLOW ... 34
FIGURE 3-4 - L1 UM, SUNSIDE VIEW ... 38
FIGURE 3-5 - L1 UM, SIDE VIEW ... 39
FIGURE 3-6 - RBM ORBITS CONFIGURATION ... 43
FIGURE 3-7 - IONOSPHERIC SPACE SEGMENT (F=FULL SCALE; M=MEDIUM SCALE) ... 52
FIGURE 3-8 : SSO IM DEPLOYMENT STRATEGY ... 53
FIGURE 3-9 - POLAR CAP MONITOR ORBIT CHARACTERISTICS ... 62
FIGURE 5-1 - LEO SO ECLIPSES VS. ALTITUDE ... 77

## *LIST OF TABLES*

TABLE 2-1 - DATA TRANSMISSION REQUIREMENTS ... 9
TABLE 2-2 - SYSTEM OPTIONS ... 11
TABLE 2-3 - SYSTEM OPTIONS PEFORMANCES SUMMARY ... 22
TABLE 2-4 - NUMBER OF SPACECRAFT VS. SO OPTION ... 24
TABLE 2-5 - FULL SCALE SPACE SEGMENT ... 25
TABLE 2-6 - MEDIUM SCALE SPACE SEGMENT ... 26
TABLE 2-7 - MINIMUM SCALE SPACE SEGMENT ... 27
TABLE 3-1 : FULL SCALE SO PAYLOAD SUMMARY ... 28
TABLE 3-2 - GEO SO MASS BUDGET ... 32
TABLE 3-3 - GEO SO POWER BUDGET ... 33
TABLE 3-4 - GEO SO BATTERY SIZING ... 35
TABLE 3-5 - GEO SO SOLAR ARRYAS SIZING ... 35
TABLE 3-6 -GEO SO PROPELLANT BUDGET ... 36
TABLE 3-7 - GEO SO LAUNCH MASS BUDGET ... 37
TABLE 3-8 - L1 UM PAYLOAD SUMMARY ... 37
TABLE 3-9 - L1 UM MASS BUDGET ... 39
TABLE 3-10 - L1 UM POWER BUDGET ... 40
TABLE 3-11 - L1 UM SOLAR ARRAYS SIZING ... 41
TABLE 3-12 - L1 UM PROPELLANT BUDGET ... 41
TABLE 3-13 - L1 UM LAUNCH MASS BUDGET ... 42
TABLE 3-14 - RBM PAYLOAD REQUIREMENTS ... 42

TABLE 3-15 - RBM ORBITAL ELEMENTS ..... 43
TABLE 3-16 - RBM MASS BUDGET (DEDICATED LAUNCH) ..... 45
TABLE 3-17 - RBM POWER BUDGET (DEDICATED LAUNCH) ..... 46
TABLE 3-18 - RBM POWER SUBSYSTEM SIZING ..... 46
TABLE 3-19 - RBM PROPELLANT BUDGET (DEDICATED LAUNCH) ..... 47
TABLE 3-20 - RBM DEDICATED LAUNCH MASS BUDGET ..... 48
TABLE 3-21 - MICROSAT RBM MASS BUDGET ..... 49
TABLE 3-22 - MICROSAT RBM POWER BUDGET ..... 50
TABLE 3-23 - MICROSAT RBM POWER SUBSYSTEM SIZING ..... 51
TABLE 3-24 - MICROSAT RBM LAUNCH MASS BUDGET ..... 51
TABLE 3-25 - SSO IM PAYLOAD REQUIREMENTS ..... 53
TABLE 3-26 : SSO IM DEPLOYMENT SEQUENCE DURATION ..... 54
TABLE 3-27 - SSO IM (FULL SCALE) MASS BUDGET ..... 55
TABLE 3-28 - SSO IM (FULL SCALE) POWER BUDGET ..... 56
TABLE 3-29 - SSO IM (FULL SCALE) PROPELLANT BUDGET ..... 56
TABLE 3-30 - SSO IM (FULL SCALE) POWER SUBSYSTEM SIZING ..... 57
TABLE 3-31 - SSO IM (FULL SCALE) LAUNCH MASS BUDGET ..... 57
TABLE 3-32 - HIGH INCLINATION IM PAYLOAD REQUIREMENTS ..... 58
TABLE 3-33 - HIGH INCLINATION IM MASS BUDGET ..... 59
TABLE 3-34 - HIGH INCLINATION IM POWER BUDGET ..... 60
TABLE 3-35 - HIGH INCLINATION IM PROPELLANT BUDGET ..... 60
TABLE 3-36 - HIGH INCLINATION IM POWER SUBSYSTEM SIZING ..... 61
TABLE 3-37 - HIGH INCLINATION IM LAUNCH MASS BUDGET ..... 61
TABLE 3-38 - POLAR CAPS MONITOR PAYLOAD REQUIREMENTS ..... 62
TABLE 3-39 - POLAR CAP MONITOR MASS BUDGET ..... 64
TABLE 3-40 - POLAR CAP MONITOR POWER BUDGET ..... 64
TABLE 3-41 - POLAR CAP MONITOR PROPELLANT BUDGET ..... 65
TABLE 3-42 - POLAR CAP MONITOR POWER SUBSYSTEM SIZING ..... 66
TABLE 3-43 - POLAR CAP MONITOR LAUNCH MASS BUDGET ..... 66
TABLE 4-1 - SO/UM PAYLOAD SUMMARY ..... 67
TABLE 4-2 - SO/UM MASS BUDGET ..... 68
TABLE 4-3 - SO/UM POWER BUDGET ..... 69
TABLE 4-4 - SO/UM PROPELLANT BUDGET ..... 70
TABLE 4-5 - SO/UM SOLAR ARRAYS SIZING ..... 70
TABLE 4-6 - COMBINED SO/UM LAUNCH MASS BUDGET ..... 71
TABLE 4-7 - MEDIUM SCALE RBM MASS BUDGET ..... 72
TABLE 4-8 - MEDIUM SCALE RBM POWER BUDGET ..... 73
TABLE 4-9 - MEDIUM SCALE RBM POWER SUBSYSTEM SIZING ..... 74
TABLE 4-10 - MEDIUM SCALE RBM LAUNCH MASS BUDGET ..... 74
TABLE 5-1 - LEO SO PAYLOAD SUMMARY ..... 76
TABLE 6-1 - ELECTRIC PROPULSION SYSTEMS CHARACTERISTICS ..... 80
TABLE 6-2 - UM USING ELECTRIC PROPULSION ..... 81

# 1. INTRODUCTION

This document presents the space segment definition and analysis performed in the Work Package 2400, in the frame of the Space Weather Programme Study by the Alcatel Consortium.

It is based on inputs from all the members of the consortium, and specifically on the Work Package 2200-2300 report compiled by Andrew Coates, Norma Crosby and Bob Bentley.

Chapter 2 describes the Space Weather system architecture selection, driven by observation and data real time availability requirements, as well as complexity, cost and launch opportunities. The following chapters are dedicated to the platform design of the different elements, in the three cases of full scale, medium scale and minimum space segment.

## 1.1 Applicable and reference Documents

### 1.1.1 Applicable documents

[AD1] WP 2200-2300 report : Space segment : measurement and system requirements, Space Weather programme study.

[AD2] WP 4000 report : First iteration synthesis

### 1.1.2 Reference documents

[RD1] STORMS Assessment Study Report - ESA-SCI(2000)7

[RD2] Ariane 5 ASAP User's Manual - Issue 1 - Revision 0 - May 2000

[RD3] Rockot User's Guide, EHB-0003, Issue 2, Rev.1

[RD4] Cosmos Launcher System - User's Guide, March 1998

[RD5] PSLV User's Manual, VSSC:PSLV:PM:65:87/4, Issue 4, December 1999

[RD6] SOYUZ User's Manual, ST-GTD-SUM-01, Issue 3 Revision 0, April 2001

## 1.2 Acronyms

| | |
|---|---|
| AOCS | Attitude and Orbit Control System |
| AsGa | Gallium Arsenide |
| ASPI | ALCATEL SPACE INDUSTRIES |
| BOL | Beginning of life |
| EOL | End of life |
| ESA | European Space Agency |
| FDIR | Failure detection, isolation and recovery |
| GEO | Geosynchronous Orbit |
| GS | Ground Station |
| GTO | Geosynchronous Transfer orbit |
| IM | Ionospheric monitors |
| ISL | Inter-Satellite Link |
| LEO | Low Earth Orbit |
| OBDH | Onboard Data Handling |
| PCDU | Power Conditioning and Distribution Unit |
| RAAN | Right Ascension of Ascending Node |
| RBM | Radiation Belt Monitor |
| S/C | Spacecraft |
| SA | Solar Array |
| SO | Solar Observer |
| TM / TC | Telemetry / Telecommand |
| UM | Upstream Monitor |
| WP | Work Package |

# 2. SYSTEM ARCHITECTURE

## 2.1 System requirements

The two main drivers of the system architecture are the need for continuous observations and the need for near real-time data from the spacecraft.

The continuous observation aspect is addressed at spacecraft level, mostly by orbital considerations. However the data flow (continuous downlink) has to be addressed at system level. The data availability requirements are summarised in the following table for each part of the required space weather measurements.

| Element | Real time data need | Maximum acceptable gap between data downlinks |
|---|---|---|
| Solar Observer | High | Minutes |
| Upstream Monitor | High | Minutes |
| Radiation Belt Monitors (GEO) | Medium | Hours |
| Radiation Belt Monitors (GTO) | Medium | Hours |
| RBM (constellation) | Low | 24 hours |
| Ionospheric Monitors (LEO) | Medium | Hours |
| Ionospheric Monitors (polar caps) | Medium | Hours |

**Table 2-1 - Data transmission requirements**

We will assume in this discussion that the potential piggy-back payloads data flow are handled by their host platforms, and do not impact the dedicated system architecture. (i.e. we do not plan to include a full communication system within the piggy-back packages).

The architecture is therefore only dictated by the dedicated spacecraft data flows.

There are basically two ways to achieve a continuous downlink of a given data flow:

- One is to ensure the visibility of the spacecraft by a ground station at all times
- another is to use a relay on a set of spacecraft that have a continuous downlink capability, provided that the data rate can be ensured with a reasonable impact both on the observing spacecraft and the relay spacecraft.

Depending on the actual data flow requirements for each type of measurements (ionospheric, magnetospheric, etc), the different possible architectures will be examined in terms of performance, complexity and cost of the overall system.

## 2.2 System options

These system options are presented in detail in WP2200-2300 report. Here are summarised the different baseline options, not including options that could be implemented as additions to a standard system (such as a constellation of 30-50 satellites).

Considering that the solar wind in-situ measurements have to be performed outside the bow shock, two configurations are possible:

- 1 spacecraft located at L1 will constantly stay in the region of interest, at 1.5 million km from the Earth in the sun direction. A continuous observation is therefore obtained with a single spacecraft due to its particular orbit.

- Another solution is to have one or several spacecraft orbiting the Earth at a very high altitude, in order to have them pass through the region of interest. If the orbits are elliptic (with measurements in the solar wind performed at apogee), the apogee will not remain in the sun direction throughout the year, so several spacecraft will be needed to maintain coverage. If the orbits are circular, they will have a very large period (several days) meaning that several satellites will be required to maintain an acceptable gap between observation passes in the solar wind upstream of the Earth.

The option consisting of several Earth-orbiting spacecraft appears much more complicated in terms of number of spacecraft to launch, deploy and operate, for an observation and measurement potential that is less than the L1 option. From now on, this option will not be studied further, and the baseline  for solar wind monitoring will be the L1 option.

Solar observation can be performed from any location, as it is remote observation. Three representative options will be considered here (but any location in view of the sun would theoretically be suitable)

- Solar observation from an orbit around L1, which provides a continuous view of the sun. Only one spacecraft with sun observation instruments is needed.

- Sun observation from a geosynchronous orbit
- Sun observation from low Earth orbit

Those orbits are either very well suited for sun observation or for data transmission to ground. Other orbits most often are less convenient for both of these functions, while not being particularly better in terms of platform design, launch strategies and operations.

| Option | Solar Observer | Upstream Monitor | Radiation Belt Monitors | Ionospheric Monitors |
|---|---|---|---|---|
| 1 | 2 LEO | L1 | 1 GEO, 3 GTO + piggy-back | 6 LEO + 2 on eccentric orbits |
| 2 | 2 GEO | L1 | | |
| 3 | L1 | | | |

**Table 2-2 - System options**

## 2.3 Direct downlink to ground

The continuous data downlink requirement cannot be met using only spacecraft to ground links. However, a certain level of performance can be obtained. For each element of the system, what could be achieved with ground stations only is summarised hereafter:

### 2.3.1 LEO Solar Observer

The continuous downlink requirement will not be met with ground stations only, considering the size of the instantaneous area in view of the satellite.

The central angle of the spherical cap in view of the satellite (with at least 10° elevation above the horizon) is 54° for a 1500km altitude (maximum order of magnitude of the LEO satellite altitude). The number of ground stations required for continuous visibility of the satellite during one revolution is therefore 7. Considering the rotation of the Earth the total number for full time coverage is clearly unacceptable in terms of cost.

Moreover, a reduced number of ground stations would not allow a near-continuous downlink (with a delay of at most a few minutes, as required by the observations).

In this case where the requirement for continuous downlink is critical (sun data), one has to consider a satellite data relay to meet this requirement, as the direct-to-ground data flow option is excluded.

### 2.3.2 GEO Solar Observer and Radiation Belt Monitor

The requirement has no particular impact on the overall system, as the satellite will always be in view of its control ground station, whether it is geostationary or geosynchronous only. The requirement will have to be taken into account at satellite design level.

### 2.3.3 L1 Solar Observer and Upstream Monitor

The requirement can be met using three ground stations equally spaced around the equator, 120° apart. Three such ground stations would allow continuous visibility of the spacecraft with sufficient elevation for data downlink.

It makes no difference in terms of spacecraft visibility whether the two functions (sun observation and solar wind monitoring) are grouped into one spacecraft or not.

### 2.3.4 GTO Radiation Belt Monitors

The orbit is such that the satellites spend most of the orbit at a high altitude, whereas the near-perigee phase is relatively short. Obtaining a good time coverage using equatorial ground stations is possible using at least three ground stations. However the perigee phase is at very low altitude (a few hundred kilometers) making it difficult to have a good visibility of the satellites during this phase without several more ground stations.

The satellite visibility performance achieved using three equatorial ground stations separated by 120° is presented hereafter (using simplified geometric calculations, that are sufficient at this stage).

These three ground stations (latitude = 0°, longitude = 0°, 120°E, 120°W) have visibility of any spacecraft above 12000km in equatorial orbit around the Earth.

The three GTO spacecraft orbital characteristics are : perigee 650km, apogee 35786km, inclination 0°, and line of apsides separated by 120°.

Under these conditions the spacecraft are in view of one of the ground stations with more than 10° elevation at least 83% of the time. Coverage under 12000km altitude (during the remaining 17%) depends on the geometry of the Earth-satellite system. However the longest period without visibility is on the order of 20 minutes.

As a summary

- Satellites visible more 90% of the orbit on average
- Longest time without visibility : about 20 minutes

Considering the data availability requirements presented in table 2-1, such a configuration is acceptable in terms of system performance.

A configuration using only two ground stations (180° apart on the equator) is however not acceptable as it introduces gaps in the high altitude coverage of the satellites. In some configurations the satellite is not in view of any of the two ground stations at apogee, which leads to several hours of coverage gap.

### 2.3.5 Radiation Belt Monitor Constellation

This part of the space segment is considered as optional in WP2200-2300 report. It might have a substantial impact on the data relay satellite design if the relay option is selected. Two options are possible :

- Consider this constellation as a stand-alone addition to the standard Space Weather system, or
- Implement on the data relay (if any) the ability to transmit the data generated by this part of the space segment, whether it is implemented at the same time as the rest of the system, or at a later stage to enhance its abilities.

In the eventuality of a stand alone constellation, given the very high altitude and period of the satellites, it is possible to obtain long visibility periods with each of the satellites at their apogees using the existing ESA ground station network. Visibility around the perigee will be more difficult to achieve, but this is not a problem as the requirement on data transmission is not very demanding (data transmission every 24h at worst).

### 2.3.6 LEO Ionospheric Monitors

Satellites orbits required for full capability observation :

- Two sun-synchronous spacecraft, 600 km altitude, 3h/15h and 9h/21h local time
- Two 625 km altitude, ~70° inclination satellites on the same orbit
- Two 600 km altitude (near-)equatorial satellites

The same remark as for the LEO Solar Observer applies. The spacecraft altitude is lower in this case, meaning that the ground area in view of one satellite is even smaller. At most, what can be achieved at a realistic cost is a data downlink at each orbit, but not a continuous downlink.

### 2.3.7 Auroral ovals monitors

Satellite orbit required for observation :

- High inclination (~90°), eccentric orbit with apogee above the pole. (perigee 500 km, apogee 3 Re)

It seems reasonable to assume that a data link to the ground is required only when images are taken by the spacecraft, i.e. when it is in view of the polar caps to be imaged. It is therefore possible to meet this requirement with a single high latitude ground station (if only north polar cap images are taken) or two if southern observations are also required. The fact that an high inclination elliptical orbit is baselined is in favor of this approach, as the spacecraft will be at apogee above the north pole. The high altitude of the satellite apogee introduces some flexibility in the selection of the high latitude ground station used for communication with the spacecraft.

## 2.4 Data relay strategies

Several possibilities exist for a satellite data relay permitting a continuous or near-continuous downlink of the data for the different elements of the space segment. Three options are considered in the following discussion:

1. Two geostationary satellites dedicated to the data relay
2. Two Solar Observers in geostationary (or geosynchronous) orbit, also acting as data relays
3. Three Radiation Belt Monitors in GTO orbits, used as data relay

### 2.4.1 GEO relay

#### 2.4.1.1 Dedicated relay satellites

This option consists of two geostationary or geosynchronous satellites dedicated to the relay of data from the elements of the space segment. The considered data sources are the potential elements of the space weather system, namely:

- L1 Upstream Monitor
- L1 Upstream Monitor + Solar Observer, if combined
- LEO Solar Observer
- GTO Radiation Belt Monitors
- Radiation Belt Monitors Constellation
- LEO Ionospheric Monitors
- Polar caps Monitors

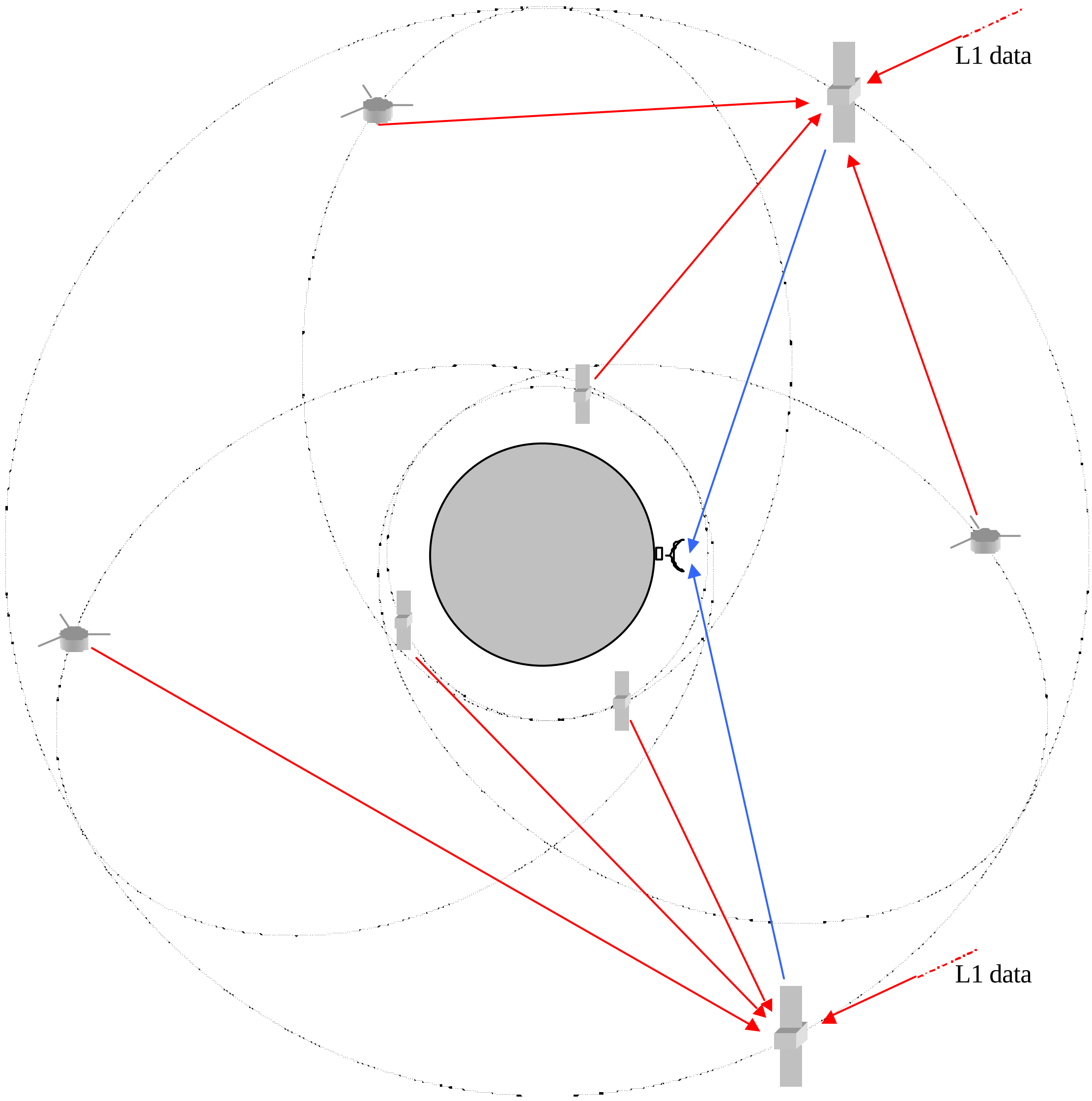



**Figure 2-1- Data transmission with geosynchronous relay satellite**

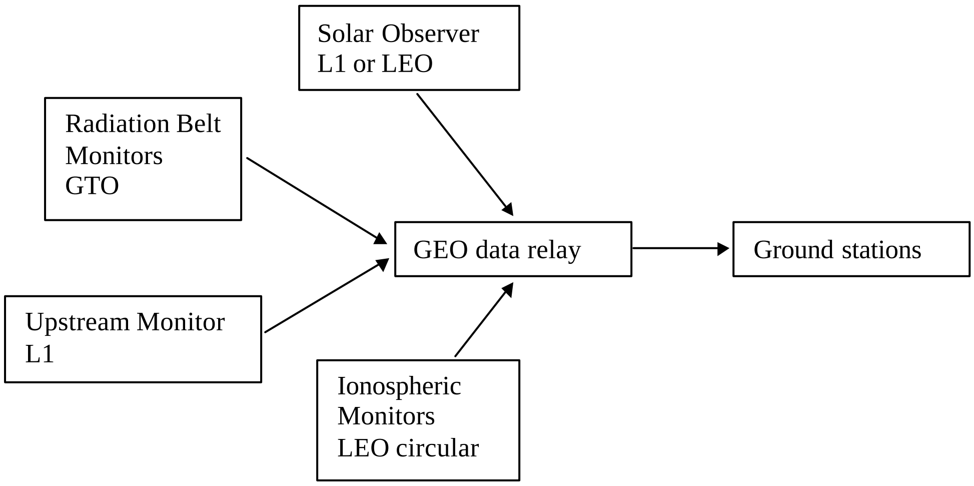



**Figure 2-2 - data flow architecture, dedicated relay, option 1**

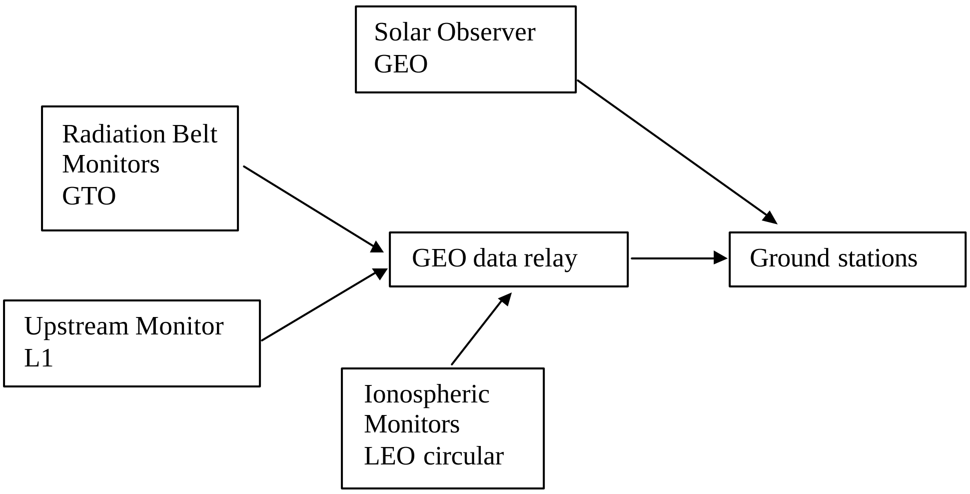



**Figure 2-3 - data flow architecture, dedicated relay, option 2**

The advantage of this option is that it only requires at most two ground stations, and possibly only one, for the whole system for telemetry (TM), and they can be standard control segment for geostationary satellites.

On the TC side there are two options : either it is also handled by the geosynchronous relays, or directly from the ground to the spacecraft. This depends on

- The level of autonomy of the different spacecraft
- The additional functions that must be implemented on the GEO relays to support a full TC capability for the whole system (multidirectional transmit capability)

- The opportunities of using existing ground stations for TC periods. Using dedicated ground stations is not desirable because it represents a loss of the advantage gained by using relay satellites.

#### 2.4.1.2 GEO combined Solar Observer and relay

This option combines the Sun observation payload and the data relay function on a single satellite. The satellite orbit might be dictated by payload pointing requirements (sun pointing) or data relay requirements. Several options are being considered at this time

- Any orbit inclination is theoretically possible as long as the two satellites are not in eclipse simultaneously, and that a good coverage is achieved for the relay function. It is better to have the two satellites on the same orbit for this reason. A possibility is a 7° inclination, meaning that no inclination correction will be required if the spacecraft are launched on Ariane 5.

- a strictly geostationary orbit can be considered. The fact that the satellite is sun-pointed will introduce seasonal changes into the satellite attitude with respect to the Earth. On the other hand it will remain fixed with respect to the Earth surface

- a 23.4° inclined geosynchronous orbit is also an option, so that the orbit is in the ecliptic plane. The will not be seasonal changes in the attitude of the satellite, but it will not be fixed with respect to the Earth (north/south variations on a one orbit time scale)

The inclination selection however does not impact greatly the system architecture and will be examined at platform level in the corresponding section of this document.

The overall system availability might be reduced in case of a GEO Solar Observer failure. In the case of an L1 to GEO relay, at worst 70 minutes of visibility per day are lost, the remaining satellite being in view of the sun almost 23 hours a day (In fact the loss of coverage is exactly the same as for the Solar Observer instruments). However for the potential LEO to GEO relay, half the coverage disappears if a GEO satellite is lost, as a full Earth hemisphere is not in view of the remaining GEO satellite.

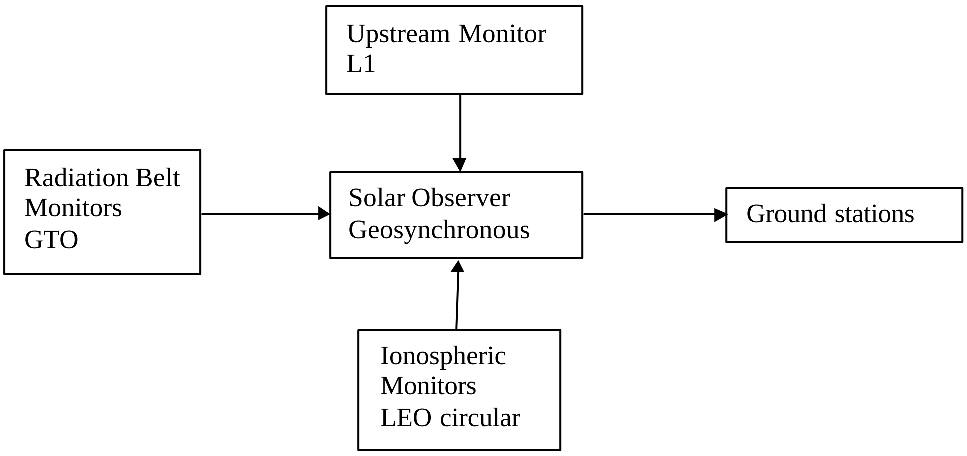



**Figure 2-4 - data flow architecture, integrated SO + relay**

#### 2.4.1.3 orbit requirements

If the two relay satellites are located 180° apart on their geosynchronous orbit, two ground stations are needed. These satellites maintain a continuous coverage of the LEO satellites at the altitudes considered (600 km).

Geostationary satellites located less than 143° apart can be controlled by a single equatorial ground station with 10° elevation. This would reduce the ground infrastructure and operations costs, while still satisfying the observation requirement. However the coverage of the 600 km LEO sphere is not complete : there is a small area up to 16° wide at the equator that is not covered.

** include assumption on margin wrt earth surface for optical path **

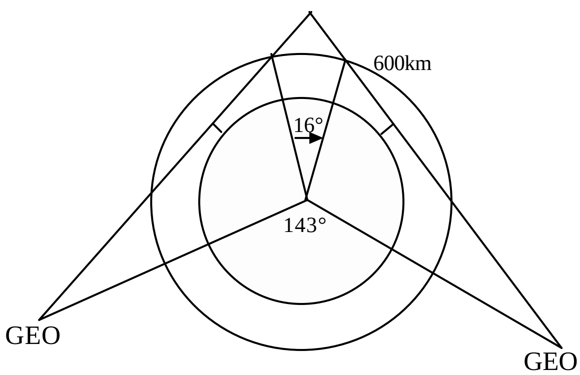



**Figure 2-5 - GEO relay coverage**

This geometrical characteristic leads to a 5 min coverage gap for an equatorial satellite at 600 km altitude, and less than half an orbit for a polar satellite which orbit happens to be in the area not covered, i.e. about 45 minutes at most, which is compatible with the data availability requirements mentioned at the beginning of this chapter.

This option of a single ground station will therefore be selected in case the two GEO relays are chosen as the baseline scenario for data transmission.

### 2.4.2 GTO relay

A possibility exists of implementing a data relay from the LEO part of the space segment to the ground via the GTO spacecraft. The performance of such a link will be less than that of a geostationary or geosynchronous relay, because both the LEO to GTO link and the GTO to ground link are not continuous. The achievable coverage is described here.

The satellite to ground station link for the GTO S/C has already been described in the previous section.

Considering the LEO to GTO link, and with no a priori knowledge of the relative positions of the different satellites, one has to consider the visibility of the 600km altitude Earth-centred sphere by the three GTO satellites.

Orbit assumptions:

- Three 0° inclined GTO orbits with perigees 120° apart, with phased perigee (simultaneous perigee crossing)
- 100 km margin above the Earth surface for the optical path, meaning that the link between two satellites shall not be closer than 100 km to the Earth

With these assumptions, the performance of the LEO to GTO link is as follows:

- full visibility is obtained 84.5% of the time. (meaning that all six LEO ionospheric monitors are visible from the GTO spacecraft)
- This value reaches 95% for the equatorial pair of ionospheric monitors.
- the minimum latitude not in view of one satellite is plotted in the following graph (all the satellites with a lower latitude than the one plotted are in view; some with a higher latitude might still be in view, depending on the geometry of the system)

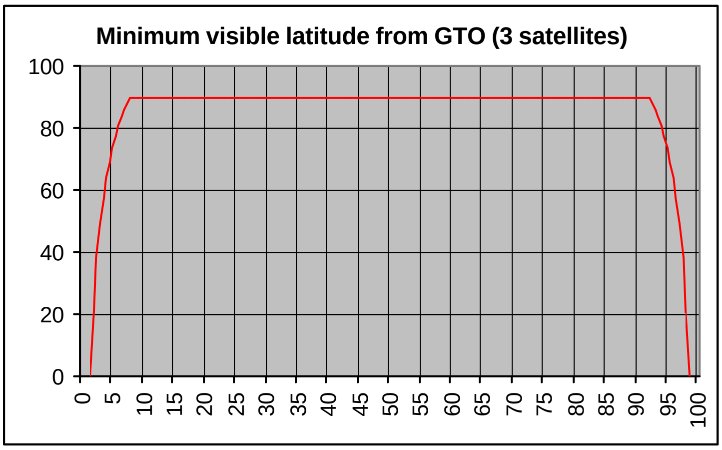



**Figure 2-6 - GTO relay coverage (1)**

- the percentage of the LEO sphere surface covered is plotted in the following graph

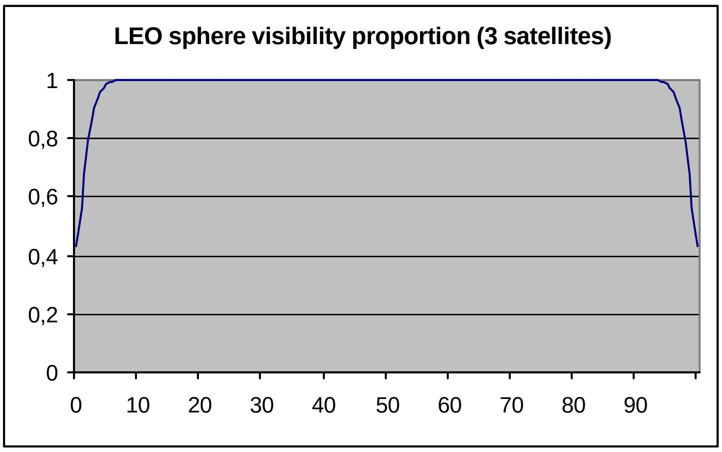



**Figure 2-7 - GTO relay coverage (2)**

The worst coverage is obviously obtained at the poles. The coverage could be increased using two ground stations (one near each pole) providing visibility at the poles, if the coverage is not sufficient. This would provide a direct LEO to ground downlink capability when the LEO spacecraft pass over the poles.

A different satellite phasing could improve these values, by reducing the overlapping areas of the fields of view.

The values given here are only for the LEO to GTO link. The performance for the full data flow must take into account the GTO to ground link as well. For real-time downlink both links need to be operational simultaneously. Otherwise, the access time is the time necessary to have a sequence of a LEO to GTO data transfer, followed by a GTO to ground data transfer.

As the visibility parameters greatly depend on the number of satellites (both GTO and LEO) and the phasing between orbits (resonance, etc), a complete mission analysis study is necessary to derive more precise statistics on the downlink periods that can be achieved.

These geometric considerations show that the option of a data relay on the GTO radiation belt monitors meets the requirements in terms of spacecraft and ground station visibility periods.

However there are several other aspects to take into account:

- The planned scientific payload is relatively simple, and the resources requirements are small (mass, power). This payload could easily be accommodated on spacecraft that would be spin-stabilised, which would even be better than 3-axis stabilisation for some experiments.

  The communication equipment for data relay, on the other hand, would lead to a much more complex satellite. The different antennas will have to be accommodated on the platform, the pointing being an issue that could even lead to require 3-axis stabilisation instead of spin-stabilisation.

- The orbit is highly elliptic, which makes it more complex to implement a data relay than on a geostationary spacecraft. This also has an impact on the antenna pointing, because the geometry of the system is much more variable.

- It is likely that from the resources point of view the driver for the platform sizing (and cost) will be the communications equipment and not the scientific payload.

- The use of standard communication equipment will be made difficult by the particular orbit (GTO), as very high radiation levels will be encountered throughout the spacecraft lifetime.

For all these reasons it appears that implementing the data relay on the GTO spacecraft will prove to be more difficult than on GEO spacecraft. This option, however, can be kept as a backup solution.

## 2.5 System options summary

| **Element** | **Direct to ground** | **GEO relay** | **GTO relay** |
|---|---|---|---|
| LEO Solar Observer | Performance insufficient | Ok | Performance insufficient |
| GEO Solar Observer | Ok (1GS for 2 S/C) | Onboard / N/A | N/A |
| L1 Solar Observer | At least 3 ground stations necessary | Ok | Performance insufficient (GTO=>ground link) |
| Upstream Monitor | At least 3 ground stations necessary | Ok | Performance insufficient (GTO=>ground link) |
| RBM (GEO) | Ok (1 GS) | Onboard | N/A |
| RBM (GTO) | Ok (3 GS) | Ok | Onboard |
| RBM (constellation) | Ok | Ok | Ok (but not required by visibility conditions) |
| IM (LEO) | Performance insufficient (with small number of GS) | Ok | Depends of required performance |
| IM (polar caps) | 1 ground station per covered pole, high latitude | Ok | Ok |
| Number of ground stations | 3 for GTO RBM and GEO or L1 SO<br>IM not adequately covered.<br>1 high latitude for auroral imager<br>RBM constellation can be covered by the aforementioned GS. | 1 GS for GEO to ground | 3 GS for GTO to ground<br>3 GS for L1 to ground (same as previous but additional antennae) |
| System robustness<br>(spacecraft failure eventuality – impact other than loss of onboard instruments data) | Each spacecraft is independent of the rest of the space segment | Coverage loss is<br>- 39% for the LEO satellites<br>- low for GTO satellites (worst case = no GTO apogee - GEO relay link)<br>- at worst 70mn per day for L1 spacecraft | Coverage loss is about 1/3 for the LEO spacecraft |

**Table 2-3 - System options peformances summary**

From this discussion it follows that a geostationary or geosynchronous data relay is desirable

The possible system implementations are

1 - L1 Solar Observer, GEO relay

2 - Combined GEO Solar Observer and relay

3 - Separate GEO Solar Observer and relay

4 - LEO Solar Observer and GEO relay

The following table presents the available options in terms of number of spacecraft. This is of course a biased estimation of the development cost, as the cost the satellites depends on their operational orbit choice, payload mass, required performances... However this can give an order of magnitude as there are many fixed costs to take into account in the development, production launch and operations of a spacecraft.

The number of spare spacecraft is simply taken to be the number of spacecraft types, to be able to respond to one random failure among all the considered spacecraft.

The numbers presented in this table also take the Upstream Monitor into account, as there is a configuration where it can be combined with the Solar Observer.

A critical point that it is interesting to mention is the number of instrument sets that have to be produced, given their high cost (estimated 80 MEuros from WP2200-2300 report, the recurrent cost being estimated to about 40 MEuros).

| Option | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| Number of Spacecraft (SO+UM+relay) | 3/4<br>combined/separate SO and UM | 3 | 5 | 5 |
| Number of replacement spacecraft | 2/3 | 2 | 3 | 3 |
| Total spacecraft | 5/7 | 5 | 8 | 8 |
| Number of Solar Observer instrument sets (including spares) | 2 | 3 | 3 | 3 |
| Launches<br>(without spares) | 2 | 2/3<br>dual / separate SO launches | 3 | 3 |

**Table 2-4 - Number of spacecraft vs. SO option**

Another element to take into account is the fact that the GEO relay is the key point for data transmission, and that system availability is critical. Therefore this element is a top priority in terms of robustness considerations. It might be necessary to have an on-orbit spare satellite in order to ensure the continuity of the service.

## 2.6 System scenarios selection

The different Space Weather programme scenarios have been selected by the entire Alcatel consortium according to observation priorities, cost and programmatics aspects, as well as the space segment performances described in the previous chapters.

As the purpose of this document is to describe the space segment of the Space Weather programme, only the system scenarios that imply the deployment of dedicated spacecraft will be highlighted in this report. The individual spacecraft description will then be done in the next chapters dedicated to the elements design.

### 2.6.1 Full Scale space segment

The full scale space segment option has been selected according to the performance considerations mentioned above. Its main feature is the presence of two geosynchronous Solar Observers, also having radiation belt monitoring instruments as well as the communications functions necessary to relay to the ground the observation data from the rest of the space segment. This configuration requires a single ground station for the entire system.

| Spacecraft | Number of spacecraft | Orbit | Instrumentation |
|---|---|---|---|
| Solar Observation | 2 | Geosynchronous, ~140° angular separation | Soft X-ray imager<br>EUV imager<br>Magnetograph<br>Coronagraph<br>H-alpha imager<br>Soft X-ray & UV flux monitor<br>EUV spectrograph<br>Thermal plasma monitor<br>Mid energy particle monitor |
| Solar wind - heliosphere | 1 | L1 | Radio spectrograph<br>Solar and galactic radiation monitor<br>Solar Wind monitor<br>Thermal plasma monitor<br>Mid energy particle monitor<br>Magnetometer |
| Magnetosphere monitoring | 3 | apogee 35786km, perigee ~500km, both in the equatorial plane inclination <18° | Thermal plasma monitor<br>Mid energy particle monitor<br>Magnetometer<br>Waves A<br>Waves B |
| Ionosphere/thermosphere monitoring | 2 | SSO, 3-15h and 9-21h, 600km altitude | Low energy plasma monitor<br>E-field antenna<br>Neutral mass spectrometer<br>GPS receiver<br>Topside sounder |
| | 2 | 600km, 75° inclination | Low energy plasma monitor<br>Interferometer<br>Neutral mass spectrometer<br>GPS receiver<br>Topside sounder |
| | 2 | equatorial, 600km | Interferometer<br>Neutral mass spectrometer<br>GPS receiver<br>Topside sounder |
| | 1 | eccentric polar orbit (apogee ~3 Re) | UV imager<br>Visible imager |

**Table 2-5 - Full scale space segment**

### 2.6.2 Medium scale space segment

The medium scale space segment has been selected with an emphasis on the solar and solar wind observation. The data relay function present in the full scale system has been cancelled, meaning that the performance in terms of real time data will be less. The Solar Observer and Upstream Monitor have been grouped on a single spacecraft at L1, as the geosynchronous location was mainly driven by the data relay function. The other parts of the space segment (RBM and IM) have been reduced in scale and will require the use of separate ground stations.

| Spacecraft | Number of spacecraft | Orbit | Instrumentation |
|---|---|---|---|
| Solar Observation / Solar wind heliosphere | 1 | L1 | EUV imager<br>Magnetograph<br>Coronagraph<br>H-alpha imager<br>Soft X-ray & UV flux monitor<br>EUV spectrograph<br>Radio spectrograph<br>Solar and galactic radiation monitor<br>Solar Wind monitor<br>Thermal plasma monitor<br>Mid energy particle monitor<br>Magnetometer |
| Magnetosphere monitoring | 3 | apogee 35786km, perigee ~500km, both in the equatorial plane inclination <18° | Thermal plasma monitor<br>Mid energy particle monitor<br>Magnetometer<br>Waves A |
| Ionosphere/thermosphere monitoring | 2 | SSO, 3-15h and 9-21h, 600km altitude | Low energy plasma monitor<br>E-field antenna<br>Neutral mass spectrometer<br>GPS receiver<br>Topside sounder |

**Table 2-6 - Medium scale space segment**

### 2.6.3 Minimum scale space segment

A minimum scale space segment has also been selected, aiming at maintaining a bare minimum observation capability. Its main features are

- the presence of a capable Upstream Monitor (similar to the full scale spacecraft), as this has been considered a vital necessity
- a very reduced Solar Observer in LEO, allowing a reasonably good observation capability at low cost, but very far from satisfying the real-time data requirement
- a single Radiation Belt Monitor in GTO orbit, very good candidate for a cost effective microsat launch on Ariane 5 ASAP.

| Spacecraft | Number of spacecraft | Orbit | Instrumentation |
|---|---|---|---|
| Solar Observation | 1 | SSO, 6-18h, 950km | EUV imager<br>Coronagraph<br>Soft X-ray & UV flux monitor<br>EUV spectrograph |
| Solar wind - heliosphere | 1 | L1 | Radio spectrograph<br>Solar and galactic radiation monitor<br>Solar Wind monitor<br>Thermal plasma monitor<br>Mid energy particle monitor<br>Magnetometer |
| Magnetosphere monitoring | 1 | apogee 35786km, perigee ~500km, both in the equatorial plane inclination <18° | Thermal plasma monitor<br>Mid energy particle monitor<br>Magnetometer<br>Waves A |

**Table 2-7 - Minimum scale space segment**

# 3. ELEMENT DESIGN - FULL SCALE SPACE SEGMENT

## 3.1 Solar observer

The selected option for the Full Scale solar observation is to have two identical geostationary Solar Observers in orbit, so that at least one is in view of the sun at all times.

### 3.1.1 Payload requirements

The instruments accommodated on this satellite are the full scale solar observation instruments, and the inner-magnetospheric instruments meant to be operated in GEO orbit. The full set of instruments is summarised in Table 3-1.

| WP2200-2300 ref. | Instrument | Mass (kg) | Power (W) | Telemetry (kbps) |
|---|---|---|---|---|
| 1 | Soft X-ray imager | 25 | 20 | 70 |
| 2 | EUV imager | 28 | 20 | 28 |
| 3 | Magnetograph | 26 | 25 | 9,5 |
| 4 | Coronagraph | 25 | 25 | 50 |
| 5 | Halpha imager | 18 | 20 | 120 |
| 6 | Soft X-ray & UV flux monitor | 5 | 5 | 0,2 |
| 8 | EUV spectrograph | 5 | 5 | 1 |
| 11 | Thermal plasma monitor | 6 | 8 | 2 |
| 12 | Mid energy particle monitor | 2 | 4 | 2 |
| **Total** | | **140** | **132** | **282,7** |

**Table 3-1 : Full scale SO payload summary**

### 3.1.2 Orbital configuration and requirements

The selected orbit has been selected geosynchronous for communications reasons. The altitude of the orbit is therefore 35786km.

The inclination can be selected according to several criteria :

- Position of the satellite with respect to its control ground station ( fixed for i=0°; not fixed for any other inclination)
- Position of the sun with respect to the orbit plane (in the orbit plane for i=23.4° with proper RAAN; outside of the orbit plane, and elevation varying in time for any other configuration)
- Inclination change after launch (none for i=7° if the spacecraft are launched by Ariane 5)

The inclination is selected to be 23.4° (both satellites in the ecliptic plane) as it is expected to facilitate the relay antennae accommodation on the spacecraft.

In terms of orbital configuration of the two satellites, any Earth central angle of more than 17.7° is acceptable as it ensures that they are not in ecplise at the same time (including an arbitrary 200km margin between the line of sight and the surface, to avoid atmospheric disturbances). In other words the line of sight satellite-to-sun is never closer than 200km to the surface of the Earth for both SOs at the same time.

As far as the data relay function is concerned,

- The data received from the L1 Upstream Monitor imposes a further requirement on the separation, as the lines of sight from the SO's to the UM are not parallel. The separation should be larger than 18.2°, still with the aforementioned 200km margin.
- Considering the data links the SO's to the ground, the angular separation has to be limited in order to allow the use of a single ground station for both spacecraft. The actual angle depends on the location of the ground station. For equatorial spacecraft and ground station it has to be less than 143° (with 10° elevation of the S/C above the local horizon).

### 3.1.3 Satellite configuration

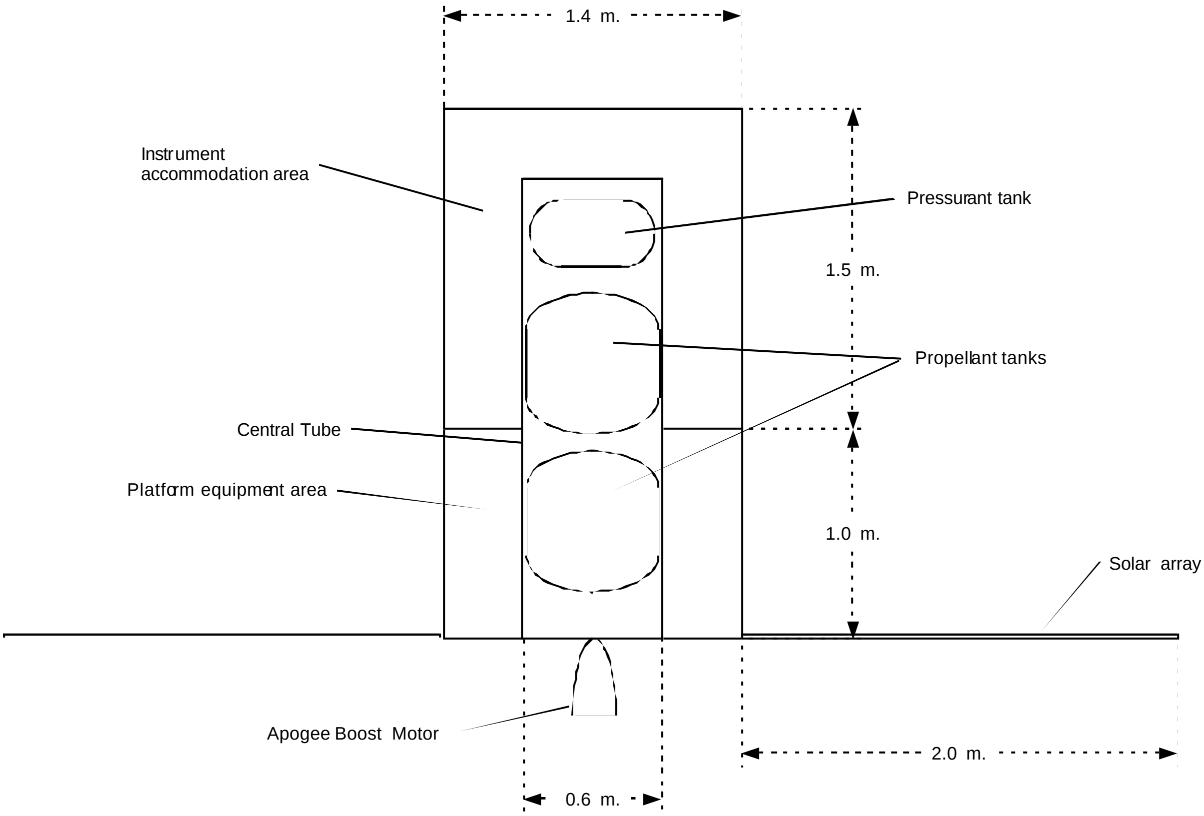



**Figure 3-1 - GEO SO view (1)**

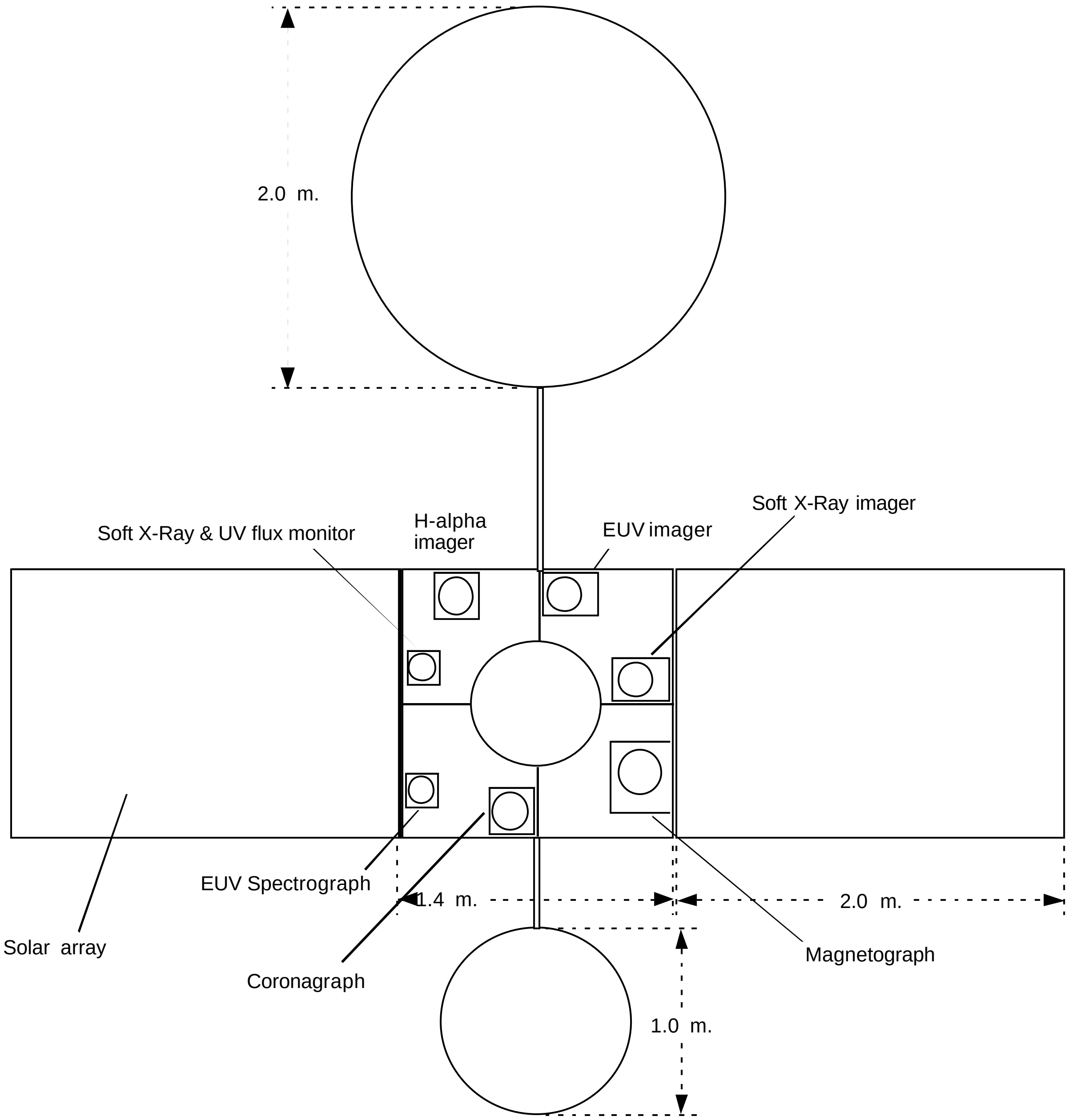



**Figure 3-2 - GEO SO view (2)**

### 3.1.4 System budgets

***Spacecraft Mass Budget***

| Subsystem | | Mass | Number | Total mass |
|---|---|---|---|---|
| **OBDH** | | 20 | 1 | 20 |
| **Power** | | | | |
| | PCDU | 10 | 1 | 10 |
| | Solar array | 20 | 2 | 40 |
| | Batteries (Li-ion) | 20 | 1 | 20 |
| | | | | |
| **Structure** | | 50 | 1 | 50 |
| **Thermal** | | 15 | 1 | 15 |
| **AOCS** | | | | |
| | reaction wheels | 5 | 4 | 20 |
| | star trackers | 3 | 2 | 6 |
| | IMU | 2 | 2 | 4 |
| | Sun sensors | 1 | 2 | 2 |
| **Comm** | | | | |
| | L1 antenna | 5 | 1 | 5 |
| | L1 electronics | 5 | 1 | 5 |
| | TMCU antenna | 8 | 1 | 8 |
| | TMCU electronics | 25 | 1 | 25 |
| | TTC equipment | 7 | 1 | 7 |
| | | | | |
| **Propulsion** | | | | |
| | tanks | 25 | 2 | 50 |
| | press tank | 15 | 1 | 15 |
| | 10N thrusters | 0,65 | 16 | 10,4 |
| | tubing | 5 | 1 | 5 |
| | apogee boost motor | 10 | 1 | 10 |
| | miscellaneous equipment | 5 | 1 | 5 |
| | | | | |
| **Payload** | | | | |
| | Soft X-ray imager | 25 | 1 | 25 |
| | EUV imager | 28 | 1 | 28 |
| | Magnetograph | 26 | 1 | 26 |
| | Coronagraph | 25 | 1 | 25 |
| | Halpha imager | 18 | 1 | 18 |
| | Soft X-ray & UV flux monitor | 5 | 1 | 5 |
| | EUV spectrograph | 5 | 1 | 5 |
| | Thermal plasma monitor | 6 | 1 | 6 |
| | Mid energy particle monitor | 2 | 1 | 2 |
| **Total Spacecraft** | | | | **472,4** |

**Table 3-2 - GEO SO mass budget**

***Spacecraft Power Budget***

| Subsystem | | Power (W) |
|---|---|---|
| **OBDH** | | 25 |
| **Power** | | 20 |
| **Thermal** | | 15 |
| **AOCS** | | |
| | reaction wheels | 28 |
| | star trackers | 17 |
| | IMU | 20 |
| **Comm** | | |
| | L1 electronics | 25 |
| | TMCU electronics | 150 |
| | TTC | 20 |
| **Payload** | | |
| | Soft X-ray imager | 20 |
| | EUV imager | 20 |
| | Magnetograph | 25 |
| | Coronagraph | 25 |
| | Halpha imager | 20 |
| | Soft X-ray & UV flux monitor | 5 |
| | EUV spectrograph | 5 |
| | Thermal plasma monitor | 8 |
| | Mid energy particle monitor | 4 |
| **Total spacecraft** | | **452** |

**Table 3-3 - GEO SO power budget**

### 3.1.5 Communications subsystem

In the selected system configuration the following communications links have to be implemented on the Solar Observer:

- Instrument Data relay links from other spacecraft of the Space Weather system to the GEO Solar Observers
- Instrument data downlink from the GEO solar observers to the ground station(s)
- TM/TC link between the GEO solar observer and the ground

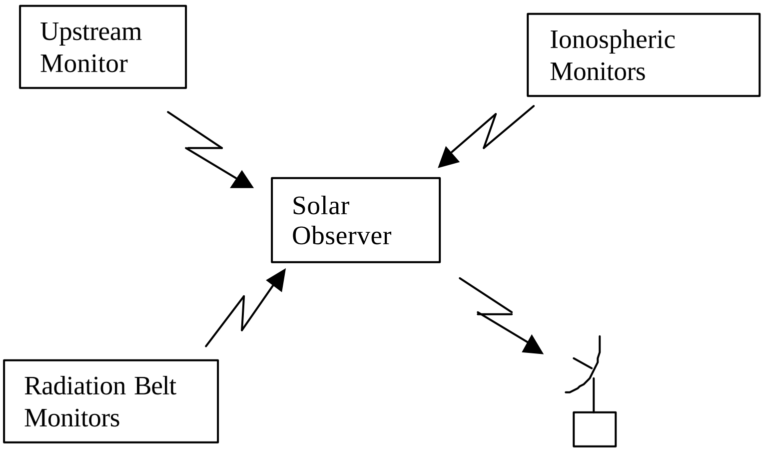



**Figure 3-3 - Full scale segment - data flow**

It must be noted that the TM/TC links for all other satellites will not relayed by the Solar Observers, (1) for system robustness reasons, (2) because the SO's would need a transmitting capability towards the other spacecraft, which would add to the complexity of the satellite.

The required communication equipment on the solar observer can be summarised as follows:

- L1 Upstream Monitor to GEO Solar Observer data link

  Electronics : 5 kg , 25 W

  1 m diameter antenna, 2 axis steerable and pointed towards the L1 spacecraft : 5 kg

- GEO Solar Observer to ground station link

  Onboard electronics : 25 kg, 150 W

  2 m diameter antenna, 2 axis steerable : 8 kg

- TM/TC subsystem

  Total 7 kg, 25 W

### 3.1.6 Power

A preliminary power subsystem sizing has been performed both for the batteries and the solar arrays.

#### 3.1.6.1 Batteries

The battery sizing has to take into account the total number of cycles over the spacecraft lifetime. The selected lifetime is 11 years. However there is some flexibility in the choice of the inclination of the orbit. Therefore, in order to keep a margin with respect to changes in potential following studies, the worst case will be taken into account here. It corresponds to the Solar Observer being in the ecliptic plane, and going through one eclipse per day over its entire lifetime.

***Spacecraft Battery sizing***

| | |
|---|---|
| Eclipse duration (mn) | 71 |
| Orbit duration (mn) | 1440 |
| Lifetime (yrs) | 11 |
| Number of cycles | 4015 |
| DOD (%) | 40,00% |
| efficiency | 0,9 |
| Battery capacity (W.h) | 1485,7 |
| Bus voltage (V) | 28 |
| **Battery capacity (A.h)** | **53,1** |

**Table 3-4 - GEO SO battery sizing**

#### 3.1.6.2 Solar arrays

The spacecraft solar arrays size estimation is based on well proven silicium cells technology, which is sufficient considering the current power demand. It allows to have only two panels of solar arrays, one on each side of the spacecraft, and the deployment mechanism will remain very simple.

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| SA -> platform efficiency | 0,8 |
| Sunlight power demand | 594,30 |
| | |
| *Solar arrays* | |
| Cells technology | Silicium |
| Efficiency (EOL, 11 years) | 10% |
| Fill factor | 0,8 |
| | |
| Solar array surface ($m^2$) | 5,39 |
| Number of arrays | 2 |
| **Required single array surface ($m^2$)** | **2,70** |

**Table 3-5 - GEO SO solar arryas sizing**

In case the power demand increases (and therefore the SA surface, keeping the same cell technology), the panels might become too large to be stored on the side of the spacecraft. In this case there will be a need to either have four panels instead of two (increased deployment complexity) or change the cells to AsGa solar cells which have a higher efficiency.

### 3.1.7 Propulsion

Considering the manoeuvres to be performed to reach the operational orbit of the Solar Observer from the GTO orbit in which it will be delivered by the launcher, a high efficiency bipropellant MON/MMH propulsion system is baselined.

In consists of

- An Apogee Boost Motor (ABM) in the 100-400N thrust range. 400N is the baseline for most GEO telecom platforms, however the dry mass of the Solar Observer is much smaller, thus requiring less thrust for manoeuvres.
- 16 (TBC) attitude control thrusters, used for reaction wheels unloading and stationkeeping manoeuvres. The fuel and oxidizer are the same as for the main engine.
- Two propellant tanks (identical volume) for storage of the propellant needed over the 11 years lifetime.
- The associated equipment (valves, tubing, pressure transducers etc)

The following propellant budget has been performed in the option of a launch by Ariane 5 into a 7° degrees inclination GTO orbit. The delta-V allocation includes the manoeuvres required to reach the operational orbit, and provisions for attitude control and stationkeeping over the spacecraft lifetime.

***Spacecraft Propellant Budget***

| | |
|---|---|
| delta V | 1700 |
| Isp | 300 |
| Dry mass | 472,4 |
| **Propellant mass** | **369,34** |
| | |
| Fuel | MON/MMH |
| density | 1,16 |
| propellant volume | 318,39 |
| number of tanks | 2,00 |
| propellant volume per tank | **159,20** |

**Table 3-6 -GEO SO propellant budget**

### 3.1.8 Launch strategy

The launch mass budget for one GEO Solar Observer is presented hereafter.

***Launch Mass Budget***

| | |
|---|---|
| **S/C dry mass** | 472,4 |
| **propellant** | 369,3 |
| **launch wet mass** | **841,7** |

**Table 3-7 - GEO SO launch mass budget**

It appears that the launch mass is well under the Ariane 5 capability (only Ariane launcher available at the mission horizon - Ariane 4 will not be available anymore). It will therefore not be cost efficient to launch both Solar Observers on the same dual launch.

A more reasonable strategy is to share two dual launches with other Ariane 5 customers, typically telecom platforms with a launch mass about 1 ton less than the Ariane 5 capability (which depends on the type of upper stage used).

## 3.2 Upstream Monitor

### 3.2.1 Payload requirements

The instruments considered for the L1 Upstream Monitor are the solar wind instruments with the addition of the radiospectrograph.

| WP2200-2300 ref. | Instrument | Mass (kg) | Power (W) | Telemetry (kbps) |
|---|---|---|---|---|
| 7 | Radio spectrograph | 12 | 6 | 0,5 |
| 9 | Solar and galactic radiation monitor | 6 | 8 | 0,1 |
| 10 | Solar Wind monitor | 6 | 5 | 2 |
| 11 | Thermal plasma monitor | 6 | 8 | 2 |
| 12 | Mid energy particle monitor | 2 | 4 | 2 |
| 13 | Magnetometer | 1 | 2 | 0,2 |
| **Total** | | **33** | **33** | **6,8** |

**Table 3-8 - L1 UM payload summary**

### 3.2.2 Orbital configuration and requirements

The Upstream Monitor will be placed on an orbit around the L1 Lagrange point, allowing a continuous view of the sun. Such orbits (Halo or Lissajous orbits) are unstable, and will require periodic stationkeeping from the spacecraft, as is being done on SoHO and ACE.

Extensive mission analysis will be required in order to select the orbital transfer and insertion strategy, select the orbit type and size, and assess the magnitude of the stationkeeping manoeuvres over the spacecraft lifetime.

For the purpose of this spacecraft dimensioning, the following assumptions have been made :

- The spacecraft will be launched into a GTO orbit on an ASAP5 mini launch (launch mass < 300kg).
- The delta V needed to reach a transfer orbit to L1 has been estimated in terms of orbit energy only. The required trajectory analysis will have to be performed in a following study.
- The insertion strategy is taken to be similar to the one used on Herschel - Planck upon arrival at L2. Therefore the same magnitude of insertion delta-V is assumed.
- The operational orbit size will also have to be studied, and the sun visibility, communication requirement, orbit stability and magnitude of the stationkeeping manoeuvres (fuel mass) will have to be traded-off against each other. The assumption here is the ACE orbit, as the two spacecraft have very similar missions.

### 3.2.3 Spacecraft configuration

A spinned spacecraft concept has been selected, both for adequation with the payload requirements and for simplicity of the design and on-orbit control. The spin axis is oriented along the sun-Earth line, and pointing at the Earth for communication reasons. The spacecraft has an octogonal shape, with deployable solar panels mounted on four of its sides (concept similar to ACE).

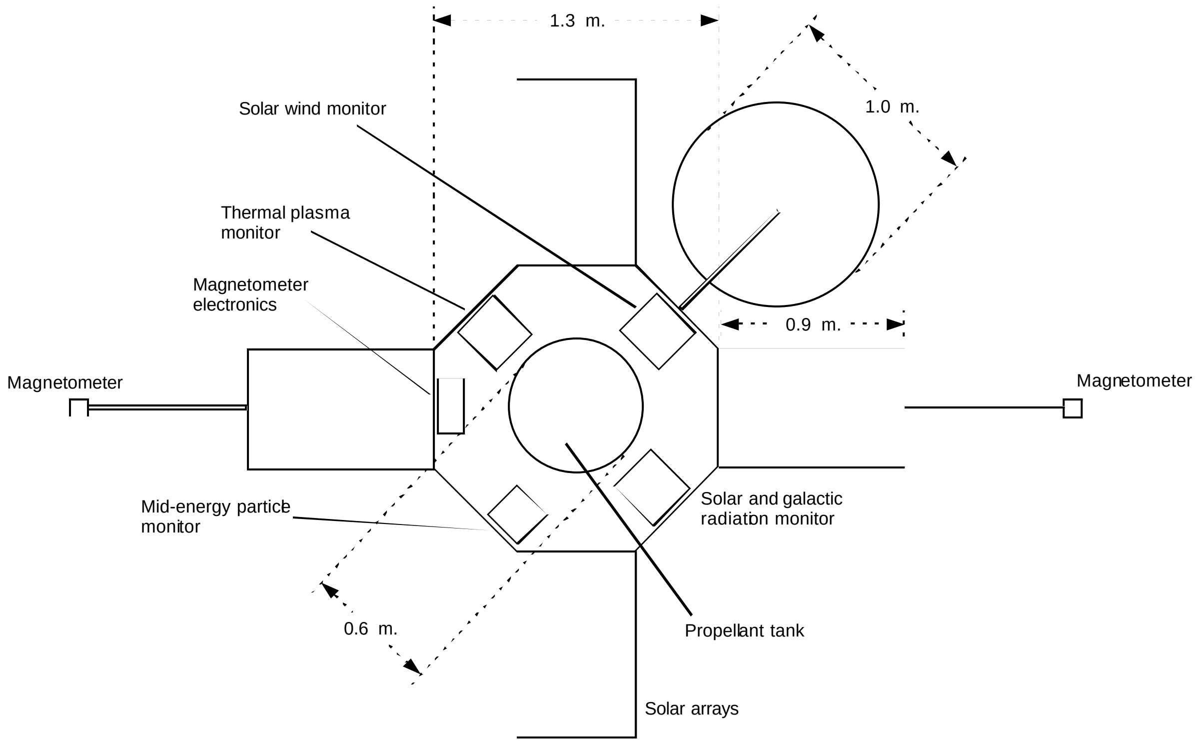



**Figure 3-4 - L1 UM, sunside view**

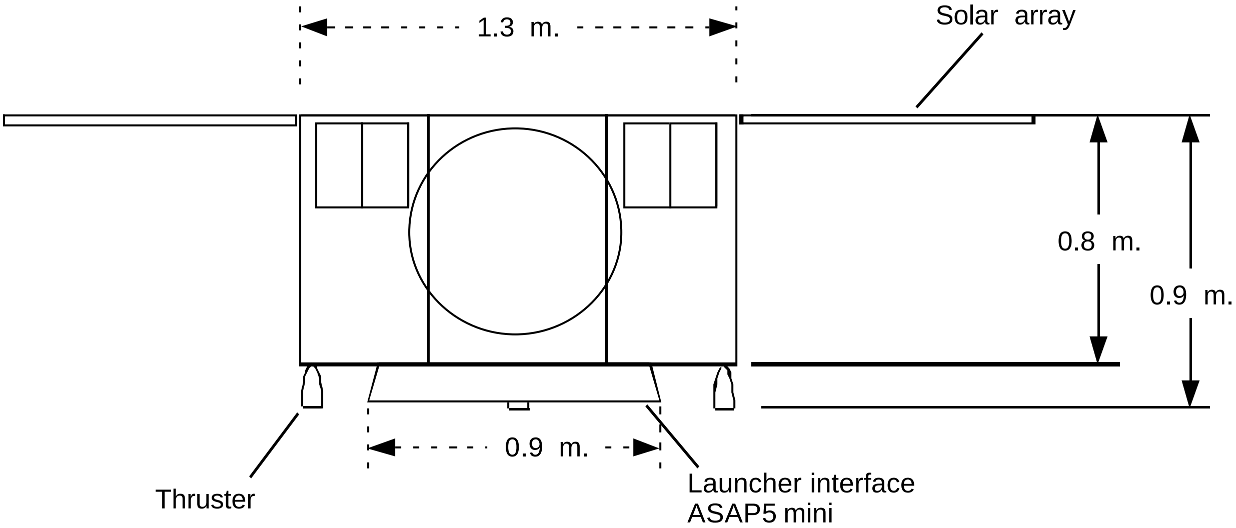



**Figure 3-5 - L1 UM, side view**

### 3.2.4 System budgets

*Spacecraft Mass Budget*

| Subsystem | | Mass | Number | Total mass |
|---|---|---|---|---|
| **OBDH** | | 10 | 1 | 10 |
| **Power** | | | | |
| | PCDU | 10 | 1 | 10 |
| | solar panels | 4 | 4 | 16 |
| **Structure** | | 25 | 1 | 25 |
| **Thermal** | | 10 | 1 | 10 |
| **AOCS** | | 12 | 1 | 12 |
| **Comm** | | | | |
| | TMCU electronics | 15 | 1 | 15 |
| | Antenna | 5 | 1 | 5 |
| | TTC | 7 | 1 | 7 |
| **Propulsion** | | | | |
| | 20N thrusters | 0,5 | 4 | 2 |
| | 1N thrusters | 0,25 | 8 | 2 |
| | hydrazine tank | 7 | 1 | 7 |
| | misc equipment | 2 | 1 | 2 |
| | tubing | 2 | 1 | 2 |
| **Payload** | | | | |
| | Radio spectrograph | 12 | 1 | 12 |
| | Solar and galactic radiation monitor | 6 | 1 | 6 |
| | Solar Wind monitor | 6 | 1 | 6 |
| | Thermal plasma monitor | 6 | 1 | 6 |
| | Mid energy particle monitor | 2 | 1 | 2 |
| | Magnetometer | 1 | 1 | 1 |
| **Total Spacecraft** | | | | **158** |

**Table 3-9 - L1 UM mass budget**

***Spacecraft Power Budget***

| Subsystem | | Power (W) |
|---|---|---|
| **OBDH** | | 15 |
| **Power** | | 10 |
| **Thermal** | | 15 |
| **AOCS** | | 20 |
| **Comm** | | |
| | TMCU | 90 |
| | TTC | 30 |
| **Payload** | | |
| | Radiospectrograph | 6 |
| | Solar and galactic radiation monitor | 8 |
| | Solar Wind monitor | 5 |
| | Thermal plasma monitor | 8 |
| | Mid energy particle monitor | 4 |
| | Magnetometer | 2 |
| **Total spacecraft** | | **213** |

**Table 3-10 - L1 UM power budget**

### 3.2.5 Communication subsystem

A preliminary sizing of the Upstream Monitor communication subsystem for the L1 to GEO transmissions and TTC equipment is as follows:

- 1m reflector antenna, 2 axis steerable, 5kg
- Onboard electronics, 15kg, 90W
- TTC subsystem : 7kg, 30W

### 3.2.6 Power subsystem

The spacecraft will not go through eclipses during its operational lifetime at L1. The nominal power susbsystem operations make use of Solar Arrays, and a Power Conditionning and Distribution Unit (PCDU) only. However batteries will be required for (a) the launch and insertion phase, during which eclipses can occur while the satellite is still in Earth orbit, and (b) contingency, i.e. in case the sun pointing attitude is temporarily lost during the operational lifetime.

For the purpose of this pre-sizing of the spacecraft, only the nominal case will be covered. The contingency cases should be covered in later studies, as they depend on a lot of factors such as S/C configuration, FDIR performances, AOCS performances etc.

A first cut of the solar arrays surface has been performed, and is summarised in the following table.

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| Spacecraft power | 213 |
| SA -> platform efficiency | 0,8 |
| **SA power** | **266,25** |
| | |
| *Solar arrays* | |
| Cells technology | AsGa |
| Efficiency (EOL, 11 years) | 13% |
| Fill factor | 0,8 |
| Max sun angle (°) | 20 |
| | |
| Solar array surface (m²) | 1,98 |
| Number of arrays | 4 |
| **Required single array surface (m²)** | **0,49** |

**Table 3-11 - L1 UM solar arrays sizing**

### 3.2.7 Propulsion subsystem

As was previoulsy pointed out in the introduction to the Upstream Monitor, the actual trajectory analysis for deployment at L1, and associated delta-V calculations, need to be performed to confirm this pre-sizing.

The assumed total delta-V required for the 11 years of the mission was taken to be 900m/s, including deployment from the initial orbit to the L1 orbit, and stationkeeping for the entire lifetime.

Considering this delta-V requirement and the expected launch mass of the spacecraft, it is proposed to use a monopropellant propulsion subsystem, using hydrazine. The low dry mass of the spacecraft should allow the use of four 20N thrusters used for orbit manoeuvres as well as attitude control, and of 8 (TBC) 1N thrusters for attitude control only.

The propellant budget for the mission is summarised in the following table.

***Spacecraft Propellant Budget***

| | |
|---|---|
| delta V | 900 |
| Isp | 210 |
| Dry mass | 158 |
| **Propellant mass** | **86,56** |
| | |
| Fuel | N2H4 |
| density | 1,00 |
| propellant volume | 86,56 |
| tank diameter | 0,55 |

**Table 3-12 - L1 UM propellant budget**

### 3.2.8 Launch

The selection of the launch strategy is based on cost minimisation. The objective is to use the 300kg capability of the ASAP5 mini launch to GTO, and make the transfer from GTO to L1 with onboard propulsion. The launch mass budget for this option is as follows.

***Launch Mass Budget***

| | |
|---|---|
| **S/C dry mass** | 158,0 |
| **propellant** | 86,6 |
| **launch wet mass** | **244,6** |
| **max launch mass** | 300,0 |
| **margin** | **22,7%** |

**Table 3-13 - L1 UM launch mass budget**

Provided the magnitude of the required manoeuvres for deployment and insertion into the final orbit is confirmed by accurate mission analysis, this option could prove to be a cost-efficient way of implementing a European Upstream Monitor.

## 3.3 Radiation Belt Monitors

### 3.3.1 Payload requirements

The payload considered for the Radiation Belt Monitors is presented in the table hereafter. The baseline mission lifetime is 5 years. Given the harsh radiation environment on this orbit, a longer mission duration is not considered optimal, and design issues linked to radiations would be increased.

| WP2200-2300 ref. | Instrument | Mass (kg) | Power (W) | Telemetry (kbps) |
|---|---|---|---|---|
| 11 | Thermal plasma monitor | 5 | 8 | 2 |
| 12 | Mid energy particle monitor | 2 | 4 | 2 |
| 13 | Magnetometer | 1,2 | 2 | 0,2 |
| 14 | Waves A | 1,3 | 1,2 | 2 |
| | Waves B | 8 | 0,6 | 2 |
| 15 | Neutral particle imager | 3 | 3 | 2 |
| **Total** | | **20,5** | **18,8** | **10,2** |

**Table 3-14 - RBM payload requirements**

### 3.3.2 Orbital configuration and launch strategies

The selected orbital configuration is three low inclination elliptical orbits, with a 120° separation between each line of apsides.

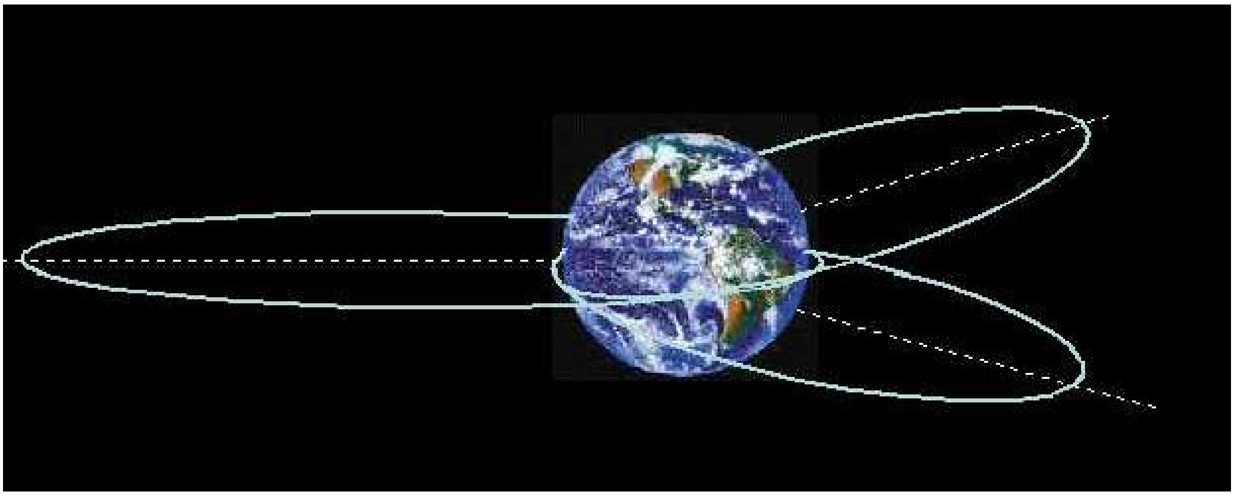


**Figure 3-6 - RBM orbits configuration**

| | |
|---|---|
| Apogee altitude | 35786 km |
| Perigee altitude | About 500 km |
| Inclination | Less than18° |
| Argument of perigee | 0° or 180° |
| Right ascension of ascending node | Separated by 120° |

**Table 3-15 - RBM orbital elements**

Two launch strategies have been considered up to now, that lead to two completely different spacecraft concepts. They will both be presented in this report.

The first option consists of a dedicated launch in low Earth orbit, in which the three satellites are delivered at once. The deployment strategy takes advantage of the natural precession of the orbits in order to obtain the required spacing (strategy similar to STORMS).

The second option consists of three separate Ariane 5 ASAP micro launches, the three RBM being delivered directly into their final orbit. The advantage of this solution is a low-cost type of satellite, with minimal propulsion capabilities. However the programmatics constraint are important, because of the requirement on the launch dates to obtain the required orbit spacing, and on the availability of ASAP5 slots on the desired launches. For this reason the dedicated launch is the baseline for this part of the space segment.

### 3.3.3 Dedicated launch option

#### 3.3.3.1 Deployment sequence

The apsidal line of the three orbits have to be separated by 120°. This separation will be accomplished using the orbit perturbations, instead of a costly manoeuvre.

The deployment sequence is as follows:

- the three satellites are launched into a 350 km 18° inclination circular orbit. The PSLV performance to this orbit is 3400 kg.

- Satellite 1 is injected into a 350x35786 km, 18° inclination, argument of perigee 0° orbit. The total delta V for this sequence of manoeuvres is 2440 m/s.

- At apogee, the perigee is raised to 650 km. If required by the mission, this manoeuvre can be combined with an inclination change to 0°.

On this orbit the right ascending node precession is -7.86°/day. Therefore the time required to achieve a 60° rotation of the orbit plane is 7.5 days.

- 7.5 days later satellite 2 is injected into a 350x35786 km, 18° inclination, argument of perigee 180° orbit. The same sequence of manoeuvres follows to put satellite 2 on its final orbit.

- 15 days later the same sequence is applied to satellite 3,with argument of perigee 0° at injection.

Advantages of this solution:

- The constellation deployment only takes 15 days from launch, due to the precession rate of the ascending node on this orbit.

- In the case of inclined orbits, two satellites have an argument of perigee of 0°, while the value is 180° for the other one (for equatorial orbits this does not apply). The minimum time in LEO can be obtained this way. However a deployment with the same argument of perigee can be obtained, taking 28 days instead of 14.

Drawbacks of this solution

- The required total delta V is high, implying the use of a high thrust engine, to reduce gravity losses. The manoeuvres might have to be performed sequentially to minimise gravity losses.

- Large propellant tanks will have to be included on the three spacecraft.

### 3.3.3.2 Spacecraft configuration

The spacecraft configuration is also similar to STORMS. A cylindrical conguration has been chosen for the three satellites, allowing them to be stacked at launch and allowing a reasonable mechanical behaviour of the stacked structure.

### 3.3.3.3 System budgets

***Spacecraft Mass Budget***

| Subsystem | | Mass | Number | Total mass |
|---|---|---|---|---|
| **OBDH** | | 15 | 1 | 15 |
| **Power** | | | | |
| | solar arrays | 8 | 1 | 8 |
| | battery | 10 | 1 | 10 |
| | PCDU | 5 | 1 | 5 |
| **Structure** | | 100 | 1 | 100 |
| **Thermal** | | 10 | 1 | 10 |
| **AOCS** | | | | |
| | star trackers | 3 | 2 | 6 |
| | IMU | 2 | 2 | 4 |
| | sun sensors | 0,2 | 2 | 0,4 |
| **Comm** | | | | |
| | TMCU antennas | 1 | 2 | 2 |
| | TMCU electronics | 10 | 1 | 10 |
| | TTC equipment | 5 | 1 | 5 |
| **Propulsion** | | | | |
| | tanks | 10 | 4 | 40 |
| | press tank | 10 | 1 | 10 |
| | 10N thrusters | 0,7 | 8 | 5,6 |
| | tubing | 5 | 1 | 5 |
| | apogee boost motor | 5 | 1 | 5 |
| | miscellaneous equipment | 5 | 1 | 5 |
| **Payload** | | | | |
| | Thermal plasma monitor | 5 | 1 | 5 |
| | Mid energy particle monitor | 2 | 1 | 2 |
| | Magnetometer | 1,2 | 1 | 1,2 |
| | Waves A | 1,3 | 1 | 1,3 |
| | Waves B | 8 | 1 | 8 |
| | Neutral particle imager | 3 | 1 | 3 |
| **Total Spacecraft** | | | | **266,5** |

**Table 3-16 - RBM mass budget (dedicated launch)**

***Spacecraft Power Budget***

| **Subsystem** | | **Power (W)** |
|---|---|---|
| **OBDH** | | 15 |
| **Power** | | 10 |
| **Thermal** | | 10 |
| **AOCS** | | |
| | star trackers | 5 |
| | IMU | 10 |
| **Comm** | | |
| | TMCU electronics | 30 |
| | TTC | 10 |
| **Payload** | | |
| | Thermal plasma monitor | 8 |
| | Mid energy particle monitor | 4 |
| | Magnetometer | 2 |
| | Waves A | 1,2 |
| | Waves B | 0,6 |
| | Neutral particle imager | 3 |
| **Total spacecraft** | | **108,8** |

**Table 3-17 - RBM power budget (dedicated launch)**

### 3.3.3.4 Power subsystem

***Spacecraft Battery sizing***

| | |
|---|---|
| Eclipse duration (mn) | 130 |
| Orbit duration (mn) | 640 |
| DOD (%) | 40,00% |
| efficiency | 0,8 |
| Battery capacity (W.h) | 727,9 |
| Bus voltage (V) | 28 |
| **Battery capacity (A.h)** | **26,0** |

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| SA -> platform efficiency | 0,9 |
| Sunlight power demand | 149,9 |
| *Solar arrays* | |
| Cells technology | AsGa |
| Efficiency (EOL, 5 years) | 16% |
| Fill factor | 0,8 |
| Solar array surface (m²) | 0,8 |
| Number of arrays | 1 |
| **Required single array surface (m²)** | **0,8** |

**Table 3-18 - RBM power subsystem sizing**

A preliminary sizing of the spacecraft power subsystem has bee performed based on the worst eclipse time encountered in the orbit, and on the spacecraft power budget.

The calculated solar array surface is for an array perpendicular to the sun direction. For a spinning spacecraft with solar cells placed on the circumference, this represents the area of the projection of the illuminated panel on a plane perpendicular to the sun direction.

### 3.3.3.5 Propulsion subsystem

In order to perform the large insertion delta-V required to reach the final orbit, a bipropellant MON/MMH system is baselined.

It consists of

- A large thrust main engine for the insertion manoeuvre
- 8 low thrust attitude control thrusters
- 4 propellant tanks (2 for fuel, 2 for oxydizer)
- A pressurant tank (helium storage)
- Miscellaneous equipments (tubing, valves…)

***Spacecraft Propellant Budget***

| | |
|---|---|
| delta V | 2650 |
| Isp | 300 |
| Dry mass | 266,5 |
| **Propellant mass** | **389,27** |
| | |
| Fuel | MON/MMH |
| density | 1,16 |
| propellant volume | 335,58 |
| number of tanks | 4,00 |
| volume per tank | 83,90 |

**Table 3-19 - RBM propellant budget (dedicated launch)**

### 3.3.3.6 Launch

The three RBMs would be launched as a stack on a PSLV launcher.

The launch mass budget for the three spacecraft is as follows

***Launch Mass Budget***

| | |
|---|---|
| **S/C dry mass** | 266,5 |
| **propellant** | 389,3 |
| **launch wet mass (1 satellite)** | **655,8** |
| **launch wet mass (3 satellites)** | **1967,4** |
| **max launch mass** | 3400,0 |
| **margin** | **72,8%** |

**Table 3-20 - RBM dedicated launch mass budget**

This mass budget shows sufficient mass margin to have a good confidence on the outcome of a feasibility study on the Radiation Belt Monitors. It also shows that the instrumentation can be added on the spacecraft at little cost, as resources

## 3.3.4 ASAP5 launch option

### 3.3.4.1 Spacecraft configuration

According to the ASAP5 user's guide, the allocated volume for a microsatellite has a square base of 600x600 mm and a height of 710 mm.

Given the fact that the spacecraft power budget is, for preliminary sizing, identical to the one for the dedicated launch RBMs, it is not possible to have body-mounted solar panels, as the cross section of the allocated spacecraft volume is less than the required solar array surface.

An ACE-like configuration will therefore be proposed : spinning spacecraft with its spin axis directed towards the sun. The satellite will have a square base of 500x500 mm, leaving 50 mm to accommodate a deployable solar panel on each side. The satellite height is estimated to about 350 mm, but is in any case limited by the stability requirements associated with the spin stabilisation.

Given the very low resources available on this type of launch, both from the volume and mass points of view, no propulsion subsystem has been selected in the baseline. The main point of concern with this strategy is the atmospheric drag at perigee. The standard perigee altitude of an Ariane 5 GTO orbit is 560km. Atmospheric drag will cause a decrease of the apogee altitude over the mission lifetime. A precise analysis will be required to assess the apogee altitude loss over the considered lifetime (currently five years) and decide if it is compatible with the RBM mission and observation requirements.

### 3.3.4.2 System budgets

***Spacecraft Mass Budget***

| **Subsystem** | | **Mass** | **Number** | **Total mass** |
|---|---|---|---|---|
| **OBDH** | | 8 | 1 | 8 |
| **Power** | | | | |
| | solar arrays | 2 | 4 | 8 |
| | battery | 8 | 1 | 8 |
| | PCDU | 5 | 1 | 5 |
| **Structure** | | 12 | 1 | 12 |
| **Thermal** | | 5 | 1 | 5 |
| **AOCS** | | | | |
| | star trackers | 3 | 2 | 6 |
| | IMU | 2 | 2 | 4 |
| | sun sensors | 0,2 | 2 | 0,4 |
| **Comm** | | | | |
| | TMCU antennas | 1 | 2 | 2 |
| | TMCU electronics | 10 | 1 | 10 |
| | TTC equipment | 5 | 1 | 5 |
| **Payload** | | | | |
| | Thermal plasma monitor | 5 | 1 | 5 |
| | Mid energy particle monitor | 2 | 1 | 2 |
| | Magnetometer | 1,2 | 1 | 1,2 |
| | Waves A | 1,3 | 1 | 1,3 |
| | Waves B | 8 | 1 | 8 |
| | Neutral particle imager | 3 | 1 | 3 |
| **Total Spacecraft** | | | | **93,9** |

**Table 3-21 - Microsat RBM mass budget**

As previously mentioned, in this first cut the RBM power budget is taken to be the same for both the dedicated and ASAP5 launch.

***Spacecraft Power Budget***

| Subsystem | | Power (W) |
| --- | --- | --- |
| **OBDH** | | 15 |
| **Power** | | 10 |
| **Thermal** | | 10 |
| **AOCS** | | |
| | star trackers | 5 |
| | IMU | 10 |
| **Comm** | | |
| | TMCU electronics | 30 |
| | TTC | 10 |
| **Payload** | | |
| | Thermal plasma monitor | 8 |
| | Mid energy particle monitor | 4 |
| | Magnetometer | 2 |
| | Waves A | 1,2 |
| | Waves B | 0,6 |
| | Neutral particle imager | 3 |
| **Total spacecraft** | | **108,8** |

**Table 3-22 - Microsat RBM power budget**

### 3.3.4.3 Power subsystem

The battery capacity and required solar array surface of this RBM microsat concept are the same as the dedicated launch concept, as the power requirements are identical. The difference is the number of solar panels, as there are 4 deployable panels of the microsat RBM. Given the very limited surface available for solar panels, very high efficiency cells have been selected, with a moderate impact on the cost, as they are more expensive, but in small number.

***Spacecraft Battery sizing***

| | |
|---|---|
| Eclipse duration (mn) | 130 |
| Orbit duration (mn) | 640 |
| DOD (%) | 40,00% |
| efficiency | 0,8 |
| Battery capacity (W.h) | 727,9 |
| Bus voltage (V) | 28 |
| **Battery capacity (A.h)** | **26,0** |

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| SA -> platform efficiency | 0,9 |
| Sunlight power demand | 149,9 |
| | |
| *Solar arrays* | |
| Cells technology | AsGa |
| Efficiency (EOL, 5 years) | 21% |
| Fill factor | 0,8 |
| | |
| Solar array surface (m²) | 0,6 |
| Number of arrays | 4 |
| **Required single array surface (m²)** | **0,2** |

**Table 3-23 - Microsat RBM power subsystem sizing**

#### 3.3.4.4 Launch

***Launch Mass Budget***

| | |
|---|---|
| **S/C dry mass** | 93,9 |
| **max launch mass** | 120,0 |
| **margin** | **27,8%** |

**Table 3-24 - Microsat RBM launch mass budget**

## 3.4 Ionospheric monitors

The Ionospheric/Thermospheric part of the full-scale space segment is presented in Figure 3-7.

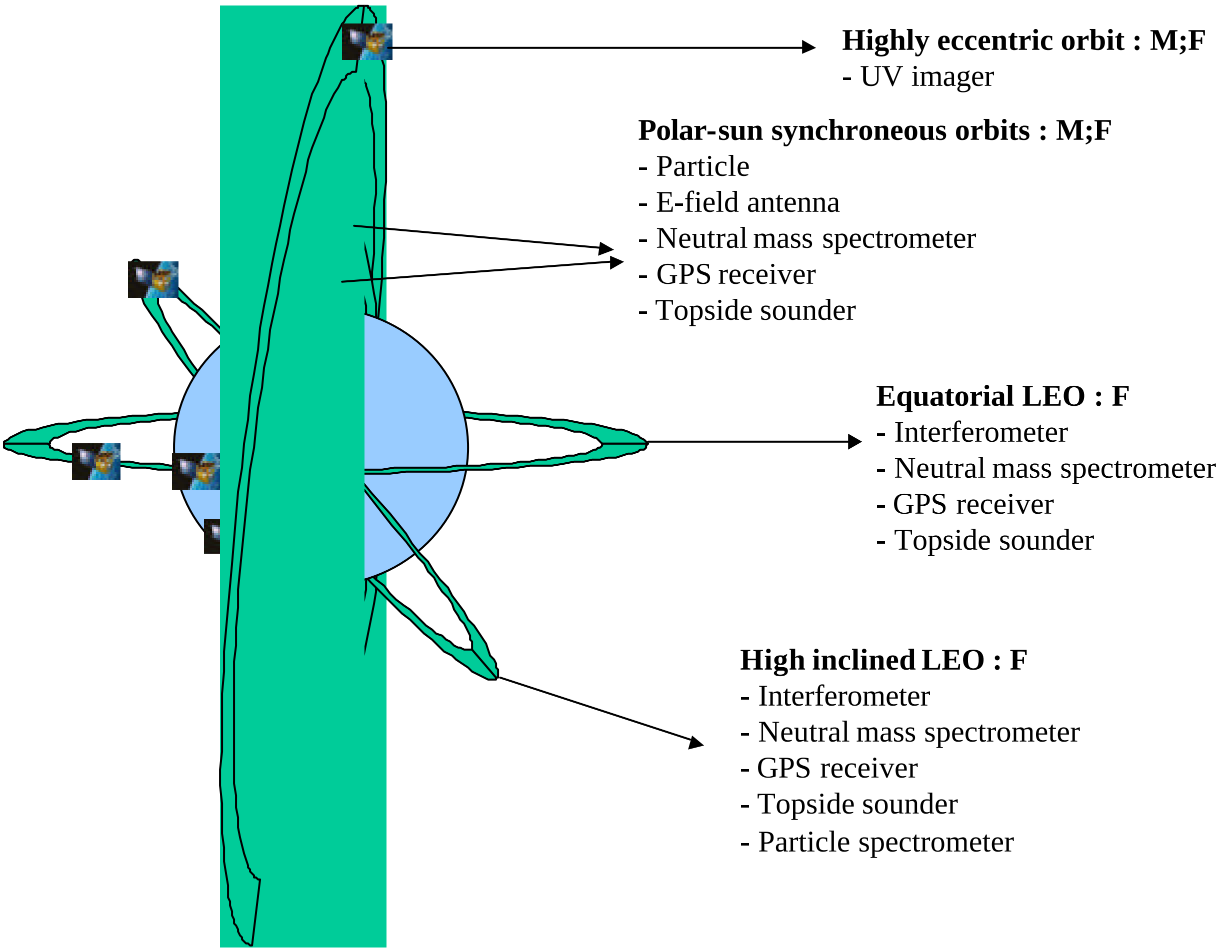



**Figure 3-7 - Ionospheric space segment (F=full scale; M=medium scale)**

The individual spacecraft launch and deployment strategies, as well as preliminary design, are addressed in the following sections.

## 3.4.1 LEO Sun-synchronous IM

### 3.4.1.1 Payload requirements

The SSO spacecraft payload requirements are presented in Table 3-25.

| WP2200-2300 ref. | Instrument | Mass (kg) | Power (W) | Telemetry (kbps) |
|---|---|---|---|---|
| 16 | E-field antenna | 2,7 | 0,5 | 2 |
| 18 | Low energy plasma monitor | 2 | 4 | 1 |
| 20 | Neutral mass spectrometer | 2,7 | 7,4 | 1 |
| 24 | GPS receiver | 8,5 | 12 | 0,1 |
| 25 | Topside sounder | 10 | 10 | 1 |
| **Total** | | **25,9** | **33,9** | **5,1** |

**Table 3-25 - SSO IM payload requirements**

#### 3.4.1.2 Launch and deployment strategy

The two sun-synchronous Ionospheric Monitors have orbits that have local times at equator crossing of 3h/15h and 9h/21h. The two orbital planes are separated by 90°. There are two possible strategies to deploy these satellites:

- Two separate launches (but which can be dual launches with other missions). In this case the satellites will have a very limited propulsion subsystem, for stationkeeping only.
- A single launch, with manoeuvres to reach the final orbits with the required local hours. In this strategy, it is not possible to perform the orbital plane separation using propulsion only, as this leads to an unacceptable amount of propellant required. The other way is to use a strategy similar to the one used for the RBM.

<u>Single launch deployment strategy</u>

The satellites would be launched into an intermediate orbit, a sun-synchronous 600km 6h/18h orbit. This orbit is in between the two desired orbits and has a 97.8° inclination. Then each satellite performs an inclination change of a few degrees (while keeping the altitude constant), in opposite directions, i.e. one of the satellites increases its inclination by a few degrees and the other one decreases it by the same amount. This introduces a drift of the orbit planes in opposite directions with respect to the initial orbit. When the desired angular separation is reached, the inclination is corrected back to the initial value of 97.8°.

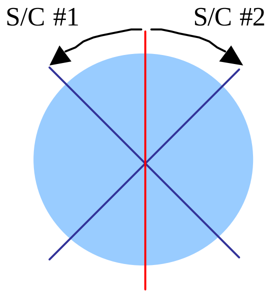



**Figure 3-8 : SSO IM Deployment strategy**

This deployment sequence starting with a 6h/18h orbit leads one of the satellites to have its ascending node on the sunlit side of the Earth, and the other on the nightside. If both ascending nodes a required to be on the same side of the planet the starting orbit for deployment shall be a noon/midnight orbit, leading to longer eclipse durations than during the nominal lifetime. The advantage of the 6h/18h is that there are no eclipses at the beginning of the deployment phase.

For large inclination changes the deployment sequence will be shorter, but the propellant mass required will be higher. Table 3-26 summarises the duration of the sequence for several inclination changes.

| Inclination change (°) | Total Delta V | Propellant mass (in % of s/c dry mass) | | Deployment duration (days) |
|---|---|---|---|---|
| | | Monopropellant (Isp = 210s) | Bipropellant (Isp = 310s) | |
| 3 | 791,4 | 46,8 | 29,7 | 119 |
| 4 | 1055,1 | 66,9 | 41,5 | 90 |
| 5 | 1318,7 | 89,7 | 54,3 | 72 |
| 6 | 1582,2 | 115,6 | 68,2 | 59 |
| 7 | 1845,6 | 144,9 | 83,5 | 51 |

**Table 3-26 : SSO IM deployment sequence duration**

Two propulsion subsystems types have been envisaged : monopropellant and bipropellant. The objective of this launch strategy being to save the cost of one launch, the objective is to build a spacecraft which cost does not exceed, by half the launch cost, the cost of the separate launch spacecraft (that has a bare minimum propulsion subsystem). It is not clear that the bipropellant system will satisfy this requirement.

For the purpose of this study a monopropellant system is taken as the baseline in order to limit the S/C cost. Should this option be studied at a later stage, the trade-off on the propulsion subsystem type shall be consolidated.

Taking as a baseline a dual Rockot launch (cost-effective, dual launch capability), the launcher capability in the required orbit is 970kg. Keeping a 20% system margin at launch, this means that for a dual launch each of the satellite masses shall not exceed 400kg including propellant.

Considering the monopropellant propulsion and the launch mass target, the strategy with a 4° inclination change will be selected. It should be noted that a provision of propellant for stationkeeping shall be added to the deployment delta-V requirement.

### 3.4.1.3 System budgets

***Spacecraft Mass Budget***

| Subsystem | | Mass | Number | Total mass |
|---|---|---|---|---|
| **OBDH** | | 10 | 1 | 10 |
| **Power** | | | | |
| | solar arrays | 8 | 2 | 16 |
| | battery | 10 | 1 | 10 |
| | PCDU | 5 | 1 | 5 |
| **Structure** | | 70 | 1 | 70 |
| **Thermal** | | 10 | 1 | 10 |
| **AOCS** | | | | |
| | star trackers | 3 | 2 | 6 |
| | IMU | 2 | 2 | 4 |
| | sun sensors | 0,5 | 2 | 1 |
| | reaction wheels | 5 | 4 | 20 |
| **Comm** | | | | |
| | TM equipment (ISL) | 5 | 1 | 5 |
| | TTC equipment | 5 | 1 | 5 |
| **Propulsion** | | | | |
| | tanks | 12 | 1 | 12 |
| | 10N thrusters | 0,7 | 8 | 5,6 |
| | main engine | 4 | 1 | 4 |
| | tubing | 4 | 1 | 4 |
| | miscellaneous equipment | 4 | 1 | 4 |
| **Payload** | | | | |
| | Low energy plasma monitor | 2,7 | 1 | 2,7 |
| | Interferometer | 2 | 1 | 2 |
| | Neutral mass spectrometer | 2,7 | 1 | 2,7 |
| | GPS receiver | 8,5 | 1 | 8,5 |
| | Topside sounder | 10 | 1 | 10 |
| **Total Spacecraft** | | | | **217,5** |

**Table 3-27 - SSO IM (full scale) mass budget**

***Spacecraft Power Budget***

| **Subsystem** | | **Power (W)** |
| --- | --- | --- |
| **OBDH** | | 15 |
| **Power** | | 15 |
| **Thermal** | | 20 |
| **AOCS** | | |
| | star trackers | 17 |
| | IMU | 20 |
| | reaction wheels | 20 |
| **Comm** | | |
| | TM electronics | 20 |
| | TTC | 10 |
| **Payload** | | |
| | E-field antenna | 0,5 |
| | Low energy plasma monitor | 4 |
| | Neutral mass spectrometer | 7,4 |
| | GPS receiver | 12 |
| | Topside sounder | 10 |
| **Total spacecraft** | | **170,9** |

**Table 3-28 - SSO IM (full scale) power budget**

### 3.4.1.4 Propulsion subsystem

As stated in the deployment strategy section, a monopropellant hydrazine propulsion subsystem has been selected. The porpellant budget is as follows, including the deployment delta-V and a 200m/s allocation for station keeping:

***Spacecraft Propellant Budget***

| | |
| --- | --- |
| delta V | 1255 |
| Isp | 210 |
| Dry mass | 217,5 |
| **Propellant mass** | **182,47** |
| | |
| Fuel | N2H4 |
| density | 1,00 |
| propellant volume | 182,47 |
| number of tanks | 1,00 |
| volume per tank | 182,47 |

**Table 3-29 - SSO IM (full scale) propellant budget**

### 3.4.1.5 Power subsystem

The power subsystem is sized according to the worst case eclipse time. The sun elevation with respect to the orbit plane is about 45° (with seasonal changes). Given the instrument requirements, a yaw steering strategy is possible (rotation of the spacecraft around the nadir axis), which, combined with the rotation of the solar arrays around their axis, ensures that the sun direction is always perpendicular to the arrays.

***Spacecraft Battery sizing***

| | |
|---|---|
| Eclipse duration (mn) | 30 |
| Orbit duration (mn) | 96,7 |
| DOD (%) | 25,00% |
| efficiency | 0,8 |
| Battery capacity (W.h) | 427,3 |
| Bus voltage (V) | 28 |
| **Battery capacity (A.h)** | **15,3** |

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| SA -> platform efficiency | 0,9 |
| Sunlight power demand | 275,3 |
| | |
| *Solar arrays* | |
| Cells technology | Si |
| Efficiency (EOL, 5 years) | 10% |
| Fill factor | 0,8 |
| | |
| Solar array surface (m²) | 2,5 |
| Number of arrays | 2 |
| **Required single array surface (m²)** | **1,2** |

**Table 3-30 - SSO IM (full scale) power subsystem sizing**

### 3.4.1.6 Launch mass budget

The launch mass budget is as follows, for a dual launch on Rockot:

***Launch Mass Budget***

| | |
|---|---|
| **S/C dry mass** | 217,5 |
| **propellant** | 182,5 |
| **launch wet mass (1 satellite)** | **400,0** |
| **launch wet mass (2 satellites)** | **799,9** |
| **max launch mass** | 960,0 |
| **margin** | **20,0%** |

**Table 3-31 - SSO IM (full scale) launch mass budget**

## 3.4.2 LEO high inclination IM

### 3.4.2.1 Payload requirements

The payload resources for the high-inclination IM are summarised in Table 3-32. The spacecraft orbit and attitude control mode (3 axis) are also driven by the observation requirements.

| WP2200-2300 ref. | Instrument | Mass (kg) | Power (W) | Telemetry (kbps) |
|---|---|---|---|---|
| 18 | Low energy plasma monitor | 2 | 4 | 1 |
| 19 | Interferometer | 42 | 19 | 1 |
| 20 | Neutral mass spectrometer | 2,7 | 7,4 | 1 |
| 24 | GPS receiver | 8,5 | 12 | 0,1 |
| 25 | Topside sounder | 10 | 10 | 1 |
| **Total** | | **65,2** | **52,4** | **4,1** |

**Table 3-32 - High inclination IM payload requirements**

### 3.4.2.2 Spacecraft configuration

The selected orbit is a 600km altitude, 75° inclination orbit. The sun elevation with respect to the orbit plane is not constant over the spacecraft lifetime (unlike a sun-synchronous orbit). The solar array design must take this into account, and two combined rotations of the solar arrays are required to properly orient the solar arrays towards the sun. A "yaw-steering" strategy could be used, one of the rotations being provided by a rotation of the spacecraft itself around the nadir axis. However the spacecraft must keep a constant along track attitude, due to the presence of the interferometer. Therefore two-axis steerable solar arrays are required.

The two spacecraft have the same orbit, only the phasing is different (i.e. they have an angular separation on the orbit). Therefore they can be launched on a single launch, and the deployment phase does not require large delta-Vs, only small altitude changes in order to introduce a drift between the two satellites.

### 3.4.2.3 System budgets

***Spacecraft Mass Budget***

| **Subsystem** | | **Mass** | **Number** | **Total mass** |
| --- | --- | --- | --- | --- |
| **OBDH** | | 10 | 1 | 10 |
| **Power** | | | | |
| | solar arrays | 8 | 2 | 16 |
| | battery | 10 | 1 | 10 |
| | PCDU | 5 | 1 | 5 |
| **Structure** | | 40 | 1 | 40 |
| **Thermal** | | 8 | 1 | 8 |
| **AOCS** | | | | |
| | star trackers | 3 | 2 | 6 |
| | IMU | 2 | 2 | 4 |
| | sun sensors | 0,5 | 2 | 1 |
| | reaction wheels | 5 | 4 | 20 |
| **Comm** | | | | |
| | TM equipment (ISL) | 5 | 1 | 5 |
| | TTC equipment | 5 | 1 | 5 |
| **Propulsion** | | | | |
| | tanks | 6 | 1 | 6 |
| | 10N thrusters | 0,7 | 8 | 5,6 |
| | tubing | 3 | 1 | 3 |
| | miscellaneous equipment | 3 | 1 | 3 |
| **Payload** | | | | |
| | Low energy plasma monitor | 2 | 1 | 2 |
| | Interferometer | 42 | 1 | 42 |
| | Neutral mass spectrometer | 2,7 | 1 | 2,7 |
| | GPS receiver | 8,5 | 1 | 8,5 |
| | Topside sounder | 10 | 1 | 10 |
| **Total Spacecraft** | | | | **212,8** |

**Table 3-33 - High inclination IM mass budget**

*Spacecraft Power Budget*

| Subsystem | | Power (W) |
|---|---|---|
| OBDH | | 15 |
| Power | | 15 |
| Thermal | | 20 |
| AOCS | | |
| | star trackers | 17 |
| | IMU | 20 |
| | reaction wheels | 20 |
| Comm | | |
| | TM electronics | 20 |
| | TTC | 10 |
| Payload | | |
| | Low energy plasma monitor | 4 |
| | Interferometer | 19 |
| | Neutral mass spectrometer | 7,4 |
| | GPS receiver | 12 |
| | Topside sounder | 10 |
| **Total spacecraft** | | **189,4** |

**Table 3-34 - High inclination IM power budget**

### 3.4.2.4 Propulsion subsystem

Given the low overall delta-V requirement, a simple monopropellant propulsion system is selected. An allocation of 200m/s has been done, including deployment and stationkeeping.

*Spacecraft Propellant Budget*

| | |
|---|---|
| delta V | 200 |
| Isp | 210 |
| Dry mass | 212,8 |
| **Propellant mass** | **21,70** |
| | |
| Fuel | N2H4 |
| density | 1,00 |
| propellant volume | 21,70 |
| number of tanks | 1,00 |
| volume per tank | 21,70 |

**Table 3-35 - High inclination IM propellant budget**

### 3.4.2.5 Power subsystem

The power subsystem sizing is as follows:

***Spacecraft Battery sizing***

| | |
|---|---|
| Eclipse duration (mn) | 35,5 |
| Orbit duration (mn) | 96,7 |
| DOD (%) | 25,00% |
| efficiency | 0,8 |
| Battery capacity (W.h) | 560,3 |
| Bus voltage (V) | 28 |
| **Battery capacity (A.h)** | **20,0** |

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| SA -> platform efficiency | 0,9 |
| Sunlight power demand | 332,5 |
| | |
| *Solar arrays* | |
| Cells technology | Si |
| Efficiency (EOL, 5 years) | 10% |
| Fill factor | 0,8 |
| | |
| Solar array surface (m²) | 3,0 |
| Number of arrays | 2 |
| **Required single array surface (m²)** | **1,5** |

**Table 3-36 - High inclination IM power subsystem sizing**

#### 3.4.2.6 Launch mass budget

***Launch Mass Budget***

| | |
|---|---|
| **S/C dry mass** | 212,8 |
| **propellant** | 21,7 |
| **launch wet mass (1 satellite)** | **234,5** |
| | |
| **launch wet mass (2 satellites)** | **469,0** |

**Table 3-37 - High inclination IM launch mass budget**

The selected launcher is PSLV, the main reasons being the low inclination requirement on the orbit, and the relatively low cost. The capability of this launcher for low inclinations is about 2900kg. There is therefore a very large margin at launch, and it could be envisaged to share the launch with another mission in a similar orbit.

## 3.4.3 Auroral and Polar cap monitor

### 3.4.3.1 Payload requirements

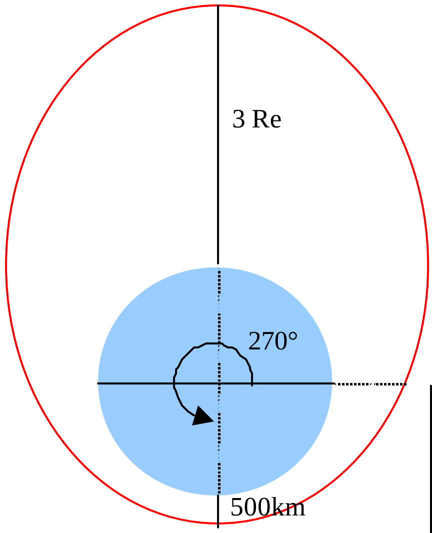
| WP2200-2300 ref. | Instrument | Mass (kg) | Power (W) | Telemetry (kbps) |
|---|---|---|---|---|
| 22 | UV imager | 20 | 10 | 10 |
| 23 | Visible imager | 29 | 10 | 10 |
| **Total** | | **49** | **20** | **20** |

**Table 3-38 - Polar caps monitor payload requirements**

### 3.4.3.2 Spacecraft configuration and launch strategy

The spacecraft must have 3-axis stabilisation, because of the presence of the two imagers. As its function is to image the north pole area, it will be pointed towards the north pole region during the phase where it is in view of the pole.

The selected orbit is as follows:




| Apogee | 3 Re |
|---|---|
| Perigee | ~500km |
| Inclination | 90° |
| Argument of perigee | 270° |

**Figure 3-9 - Polar cap monitor orbit characteristics**

The sun elevation with respect to the orbit plane will vary over time, as the orbit is not sun-synchronous. Two rotations of the solar arrays are required to be able to point them at the sun at any time. One can be provided by the rotation of the arrays around their symmetry axis. As the rotation of the spacecraft around the pointing (imaging) direction is not constrained, a "yaw steering" strategy is possible and will be used in this case, and this provides the second rotation.

The possible launch strategies are:

- Launch by a low-cost type LEO launcher (Rockot class). These launchers do not have the capability to launch directly into the required orbit (TBC on a case-by-case basis with launcher authority - the different user's guides are taken as reference in the frame of this study). Therefore the strategy is to launch into LEO and raise the apogee with onboard propulsion (delta-V on the order of 2000m/s).

  This strategy will require a dedicated single launch, considering the spacecraft and propellant mass required.

- Direct launch into the final orbit, which requires a more capable launcher (Soyuz class).

  This strategy will lead to a low spacecraft launch mass, well below the launcher capability in the case of Soyuz. A multiple launch with another mission destined to a similar orbit would be a very good opportunity in this case.

In the frame of this study the first strategy will be selected. However if a candidate mission for multiple launch is identified, one could reconsider this choice in favour of the second strategy.

#### 3.4.3.3 System budgets

The mass and power budgets for the polar cap monitor (for a launch into LEO and deployment using onboard propulsion) are presented herafter:

***Spacecraft Mass Budget***

| Subsystem | | Mass | Number | Total mass |
|---|---|---|---|---|
| **OBDH** | | 15 | 1 | 15 |
| **Power** | | | | |
| | solar arrays | 8 | 2 | 16 |
| | battery | 15 | 1 | 15 |
| | PCDU | 5 | 1 | 5 |
| **Structure** | | 80 | 1 | 80 |
| **Thermal** | | 15 | 1 | 15 |
| **AOCS** | | | | |
| | star trackers | 3 | 2 | 6 |
| | IMU | 2 | 2 | 4 |
| | sun sensors | 0,5 | 2 | 1 |
| | reaction wheels | 5 | 4 | 20 |
| **Comm** | | | | |
| | TM equipment (ISL) | 5 | 1 | 5 |
| | TTC equipment | 5 | 1 | 5 |
| **Propulsion** | | | | |
| | tanks | 10 | 2 | 20 |
| | pressurant tank | 8 | 1 | 8 |
| | 10N thrusters | 0,7 | 8 | 5,6 |
| | main engine | 4 | 1 | 4 |
| | tubing | 4 | 1 | 4 |
| | miscellaneous equipment | 5 | 1 | 5 |
| **Payload** | | | | |
| | UV imager | 20 | 1 | 20 |
| | Visible imager | 29 | 1 | 29 |
| **Total Spacecraft** | | | | **282,6** |

**Table 3-39 - Polar cap monitor mass budget**

***Spacecraft Power Budget***

| Subsystem | | Power (W) |
|---|---|---|
| **OBDH** | | 15 |
| **Power** | | 15 |
| **Thermal** | | 20 |
| **AOCS** | | |
| | star trackers | 17 |
| | IMU | 20 |
| | reaction wheels | 20 |
| **Comm** | | |
| | TM electronics | 20 |
| | TTC | 10 |
| **Payload** | | |
| | UV imager | 10 |
| | Visible imager | 10 |
| **Total spacecraft** | | **157** |

**Table 3-40 - Polar cap monitor power budget**

### 3.4.3.4 Propulsion subsystem

The required delta-V to reach the final orbit from a 500x500km orbit, with 90° inclination, is 1950m/s. The total allocation including stationkeeping will be 2150m/s.

A bipropellant subsystem is selected as the delta-V requirement is high. It will mainly consist of fuel and oxydizer tanks (one each), a pressurant tank (helium), a main thruster, and attitude control thrusters.

The propellant budget is shown hereafter:

***Spacecraft Propellant Budget***

| | |
|---|---|
| delta V | 2150 |
| Isp | 310 |
| Dry mass | 282,6 |
| **Propellant mass** | **290,47** |
| | |
| Fuel | MON/MMH |
| density | 1,16 |
| propellant volume | 250,41 |
| number of tanks | 2,00 |
| volume per tank | 125,20 |

**Table 3-41 - Polar cap monitor propellant budget**

### 3.4.3.5 Power subsystem

The power subsystem has been performed taking into account the worst case eclipse time (sun in the orbit plane) which is about 90mn, for an orbit period of 342mn.

***Spacecraft Battery sizing***

| | |
|---|---|
| Eclipse duration (mn) | 90 |
| Orbit duration (mn) | 342 |
| DOD (%) | 30,00% |
| efficiency | 0,8 |
| Battery capacity (W.h) | 981,3 |
| Bus voltage (V) | 28 |
| **Battery capacity (A.h)** | **35,0** |

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| SA -> platform efficiency | 0,9 |
| Sunlight power demand | 236,7 |
| | |
| *Solar arrays* | |
| Cells technology | Si |
| Efficiency (EOL, 5 years) | 10% |
| Fill factor | 0,8 |
| | |
| Solar array surface (m²) | 2,1 |
| Number of arrays | 2 |
| **Required single array surface (m²)** | **1,1** |

**Table 3-42 - Polar cap monitor power subsystem sizing**

### 3.4.3.6 Launch mass budget

| | |
|---|---|
| **S/C dry mass (kg)** | 282,6 |
| **propellant (kg)** | 290,5 |
| **launch wet mass (kg)** | **573,1** |

**Table 3-43 - Polar cap monitor launch mass budget**

The capability of the Rockot launcher into a 500x500km orbit for a 90° inclination is more than 1200kg. It would therefore be possible to share the launch with another mission. The characteristics of the initial orbit could be tailored to fit the requirements of the other satellite, as the constraint on the 90° inclination and on the perigee altitude of 500km are flexible.

# 4. ELEMENT DESIGN - MEDIUM SCALE SPACE SEGMENT

## 4.1 Combined Solar Observer - Upstream Monitor

In the medium scale space segment, the concept of data relay, which was the design driver for the full scale space segment, is not used. Each part of the space segment is independent from the other satellites in terms of instrument telemetry. For this reason the solar observation and upstream monitoring instruments can be grouped on a single spacecraft at L1.

### 4.1.1 Payload requirements

The payload of this combined SO/UM at L1 is a consists of a reduced capability set of the instruments of the full scale SO and UM.

| WP2200-2300 ref. | Instrument | Mass (kg) | Power (W) | Telemetry (kbps) |
|---|---|---|---|---|
| 2 | EUV imager | 28 | 20 | 28 |
| 3 | Magnetograph | 26 | 25 | 9,5 |
| 4 | Coronagraph | 25 | 25 | 50 |
| 5 | Halpha imager | 18 | 20 | TBD |
| 6 | Soft X-ray & UV flux monitor | 5 | 5 | 0,2 |
| 7 | Radio spectrograph | 12 | 6 | 0,5 |
| 8 | EUV spectrograph | 5 | 5 | 1 |
| 9 | Solar and galactic radiation monitor | 6 | 8 | 0,1 |
| 10 | Solar Wind monitor | 6 | 5 | 2 |
| 11 | Thermal plasma monitor | 6 | 8 | 2 |
| 12 | Mid energy particle monitor | 2 | 4 | 2 |
| 13 | Magnetometer | 1 | 2 | 0,2 |
| **Total** | | **140** | **133** | **95,5** |

**Table 4-1 - SO/UM payload summary**

### 4.1.2 Spacecraft configuration

The spacecraft configuration is driven by the solar observation instrument, that require 3 axis stabilisation and a sun-pointing attitude. The resulting configuration is comparable the the SoHO configuration except that this spacecraft has a lower launch mass.

### 4.1.3 System budgets

***Spacecraft Mass Budget***

| Subsystem | | Mass | Number | Total mass |
|---|---|---|---|---|
| **OBDH** | | 20 | 1 | 20 |
| **Power** | | | | |
| | PCDU | 10 | 1 | 10 |
| | Solar array | 20 | 2 | 40 |
| | Batteries (Li-ion) | 5 | 1 | 5 |
| **Structure** | | 40 | 1 | 40 |
| **Thermal** | | 15 | 1 | 15 |
| **AOCS** | | | | |
| | reaction wheels | 5 | 4 | 20 |
| | star trackers | 3 | 2 | 6 |
| | IMU | 2 | 2 | 4 |
| | Sun sensors | 1 | 2 | 2 |
| **Comm** | | | | |
| | TM antenna | 8 | 1 | 8 |
| | TM electronics | 25 | 1 | 25 |
| | TTC equipment | 7 | 1 | 7 |
| **Propulsion** | | | | |
| | tanks | 10 | 1 | 10 |
| | 10N thrusters | 0,65 | 16 | 10,4 |
| | tubing | 5 | 1 | 5 |
| | miscellaneous equipment | 5 | 1 | 5 |
| **Payload** | | | | |
| | EUV imager | 28 | 1 | 28 |
| | Magnetograph | 26 | 1 | 26 |
| | Coronagraph | 25 | 1 | 25 |
| | Halpha imager | 18 | 1 | 18 |
| | Soft X-ray & UV flux monitor | 5 | 1 | 5 |
| | Radio spectrograph | 12 | 1 | 12 |
| | EUV spectrograph | 5 | 1 | 5 |
| | Solar and galactic radiation monitor | 6 | 1 | 6 |
| | Solar Wind monitor | 6 | 1 | 6 |
| | Thermal plasma monitor | 6 | 1 | 6 |
| | Mid energy particle monitor | 2 | 1 | 2 |
| | Magnetometer | 1 | 1 | 1 |
| **Total Spacecraft** | | | | **372,4** |

**Table 4-2 - SO/UM mass budget**

***Spacecraft Power Budget***

| Subsystem | | Power (W) |
|---|---|---|
| **OBDH** | | 25 |
| **Power** | | 20 |
| **Thermal** | | 15 |
| **AOCS** | | |
| | reaction wheels | 28 |
| | star trackers | 17 |
| | IMU | 20 |
| **Comm** | | |
| | TM electronics | 150 |
| | TTC | 20 |
| **Payload** | | |
| | EUV imager | 20 |
| | Magnetograph | 25 |
| | Coronagraph | 25 |
| | Halpha imager | 20 |
| | Soft X-ray & UV flux monitor | 5 |
| | Radio spectrograph | 6 |
| | EUV spectrograph | 5 |
| | Solar and galactic radiation monitor | 8 |
| | Solar Wind monitor | 5 |
| | Thermal plasma monitor | 8 |
| | Mid energy particle monitor | 4 |
| | Magnetometer | 2 |
| **Total spacecraft** | | **428** |

**Table 4-3 - SO/UM power budget**

### 4.1.4 Propulsion subsystem

A dedicated launch is necessary considering the expected spacecraft dry mass. The strategy is similar to the one used for Herschel/Planck at L2 : direct launch into a transfer orbit to L1, allowing to perform only trajectory corrections during the “cruise” phase and a low delta-V insertion manoeuvre.

For this purpose a monopropellant propulsion subsystem is sufficient. It allows to have a reasonably good performance (specific impulse of about 210s) associated with a low subsystem dry mass (only 1 low volume propellant tank).

The 10N thrusters used for stationkeeping, attitude control and wheels unloading during the spacecraft lifetime have been supposed to be enough to perform the course corrections and insertion manoeuvres. This needs to be confirmed, and 20N thrusters might be implemented is necessary.

The total delta-V allocation for course corrections, insertion and stationkeeping is taken to be 200m/s, based on Hershel/Planck similarities. Mission analysis will be required to confirm these values. A preliminary propellant budget is shown hereafter.

***Spacecraft Propellant Budget***

| | |
|---|---|
| delta V | 200 |
| Isp | 210 |
| Dry mass | 372,4 |
| **Propellant mass** | **37,97** |
| | |
| Fuel | N2H4 |
| density | 1,00 |
| propellant volume | 37,97 |
| tank diameter | 0,42 |

**Table 4-4 - SO/UM propellant budget**

### 4.1.5 Power subsystem

The spacecraft is always in view of the sun during its operational lifetime. The spacecraft battery capacity is not driven by the nominal operations, during which there are no eclipses, but by the launch and cruise phase, and by the safe mode requirements. The spacecraft might encounter an eclipse period shortly after launcher separation, depending on the ascent trajectory. During the operational lifetime, possible losses of attitude control must be taken into account, during which the solar arrays might not be illuminated. The batteries will have to power the spacecraft during the time necessary to recover the sun-pointing attitude.

This preliminary sizing is for the nominal operations only, i.e. the solar arrays are the only power source for the platform and payload.

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| SA -> platform efficiency | 0,8 |
| Sunlight power demand | 535,00 |
| | |
| *Solar arrays* | |
| Cells technology | AsGa |
| Efficiency (EOL, 11 years) | 13% |
| Fill factor | 0,8 |
| | |
| Solar array surface (m²) | 3,73 |
| Number of arrays | 2 |
| **Required single array surface (m²)** | **1,87** |

**Table 4-5 - SO/UM solar arrays sizing**

### 4.1.6 Launch mass budget

| ***Launch Mass Budget*** | |
|---|---|
| **S/C dry mass** | 372,4 |
| **propellant** | 38,0 |
| **launch wet mass** | **410,4** |

**Table 4-6 - Combined SO/UM launch mass budget**

As was already mentioned, extensive mission analysis will be required to determine the best trajectory for the transfer of the S/C to L1, and the launcher selection will be a consequence of the trajectory requirements. However the capability of the Soyuz launcher into Earth escape trajectories is clearly sufficient for this application (C3 > 36 $km^2/s^2$ for the considered launch mass).

## 4.2 Radiation Belt Monitors

In the medium scale space segment, the Radiation Belt Monitors instrumentation and orbit requirements are identical to the ones for the full scale space segment (see §3.3), except that the WAVES B instrument is not included. The only other difference in the design is the communications subsystem, as in the present case the data is not relayed by the Solar Observer, but is transmitted directly to the Earth.

The impact on the satellite is a communication subsystem requiring less resources than on the full scale RBM, as larger antennae can be used on ground, thus limiting the required RF power on the satellite.

The baseline launch strategy, however, remains the dedicated launch option on PSLV, for the programmatics constraints associated with the separate ASAP5 launches already mentioned.

### 4.2.1 System budgets

The system budgets are derived from the full scale RBM's budgets, with the required modifications on the communications subsystem.

***Spacecraft Mass Budget***

| **Subsystem** | | **Mass** | **Number** | **Total mass** |
|---|---|---|---|---|
| **OBDH** | | 15 | 1 | 15 |
| **Power** | | | | |
| | solar arrays | 8 | 1 | 8 |
| | battery | 10 | 1 | 10 |
| | PCDU | 4 | 1 | 4 |
| **Structure** | | 100 | 1 | 100 |
| **Thermal** | | 10 | 1 | 10 |
| **AOCS** | | | | |
| | star trackers | 3 | 2 | 6 |
| | IMU | 2 | 2 | 4 |
| | sun sensors | 0,5 | 2 | 1 |
| **Comm** | | | | |
| | TMTC | 8 | 1 | 8 |
| **Propulsion** | | | | |
| | tanks | 10 | 4 | 40 |
| | press tank | 10 | 1 | 10 |
| | 10N thrusters | 0,7 | 8 | 5,6 |
| | tubing | 5 | 1 | 5 |
| | apogee boost motor | 5 | 1 | 5 |
| | miscellaneous equipment | 5 | 1 | 5 |
| **Payload** | | | | |
| | Thermal plasma monitor | 5 | 1 | 5 |
| | Mid energy particle monitor | 2 | 1 | 2 |
| | Magnetometer | 1,2 | 1 | 1,2 |
| | Waves A | 1,3 | 1 | 1,3 |
| | Neutral particle imager | 3 | 1 | 3 |
| **Total Spacecraft** | | | | **249,1** |

**Table 4-7 - Medium scale RBM mass budget**

***Spacecraft Power Budget***

| **Subsystem** | | **Power (W)** |
|---|---|---|
| **OBDH** | | 15 |
| **Power** | | 10 |
| **Thermal** | | 10 |
| **AOCS** | | |
| | star trackers | 5 |
| | IMU | 10 |
| **Comm** | | |
| | TMTC electronics | 20 |
| **Payload** | | |
| | Thermal plasma monitor | 8 |
| | Mid energy particle monitor | 4 |
| | Magnetometer | 2 |
| | Waves A | 1,2 |
| | Neutral particle imager | 3 |
| **Total spacecraft** | | **88,2** |

**Table 4-8 - Medium scale RBM power budget**

### 4.2.2 Communication subsystem

The baseline is an S-band communication system, with a quasi-omnidirectional antenna and 3W RF power. The mass of the subsystem is estimated to less than 8kg, and the power requirement to 20W.

An optional system could be envisaged, consisting of an X-band transponder and antenna, allowing more flexibility in the transmissions. The power requirements would be 45W (with 10W RF power) and the mass less than 10kg.

### 4.2.3 Power subsystem

The power subsystem is identical to the full scale RBMs, taking into account the updated power requirement. The new sizing is shown hereafter:

***Spacecraft Battery sizing***

| | |
|---|---|
| Eclipse duration (mn) | 130 |
| Orbit duration (mn) | 640 |
| DOD (%) | 40,00% |
| efficiency | 0,8 |
| Battery capacity (W.h) | 592,4 |
| Bus voltage (V) | 28 |
| **Battery capacity (A.h)** | **21,2** |

***Spacecraft Solar Arrays sizing***

| | |
|---|---|
| SA -> platform efficiency | 0,9 |
| Sunlight power demand | 122,0 |
| | |
| *Solar arrays* | |
| Cells technology | AsGa |
| Efficiency (EOL, 5 years) | 16% |
| Fill factor | 0,8 |
| | |
| Solar array surface (m²) | 0,7 |
| Number of arrays | 1 |
| **Required single array surface (m²)** | **0,7** |

**Table 4-9 - Medium scale RBM power subsystem sizing**

### 4.2.4 Launch mass budget

***Launch Mass Budget***

| | |
|---|---|
| **S/C dry mass** | 249,1 |
| **propellant** | 363,9 |
| **launch wet mass (1 satellite)** | **614,4** |
| **launch wet mass (3 satellites)** | **1843,2** |
| **max launch mass** | 3400,0 |
| **margin** | **84,5%** |

**Table 4-10 - Medium scale RBM launch mass budget**

The medium scale RBM design leads to a launch mass margin slightly higher than in the full scale, due to the reduction of the telecommunication equipment resources. The margin is above 80% and gives very good confidence in the feasibility of the option.

## 4.3 Ionospheric Monitors

In the medium scale space segment, the sun-synchronous ionospheric monitors have been selected. The instrumentation and design are identical, the only difference being that the data is not transmitted through the GEO relay but directly to the ground. The communication equipment has to be replaced with the following characteritics : mass 8kg including antenna, power 20W (3W RF). This has very little impact on the overall design and none on the launch strategy.

# 5. ELEMENT DESIGN - MINIMUM SCALE SPACE SEGMENT

## 5.1 Solar Observer

Considering the objective of designing a cost effective minimal Solar Observer, the option of basing the spacecraft on a European recurrent platform will be considered here. The payload requirements are within the capability of the currently available or planned platforms (the mass and power requirements are quite low as compared to other planned missions of these platforms.

Special attention will be needed, however, for the pointing requirements, that more demanding than for most common Earth observation missions.

### 5.1.1 Payload requirements

| WP2200-2300 ref. | Instrument | Mass (kg) | Power (W) | Telemetry (kbps) |
|---|---|---|---|---|
| 2 | EUV imager | 28 | 20 | 28 |
| 4 | Coronagraph | 25 | 25 | 50 |
| 6 | Soft X-ray & UV flux monitor | 5 | 5 | 0,2 |
| 8 | EUV spectrograph | 5 | 5 | 1 |
| **Total** | | **63** | **55** | **79,2** |

**Table 5-1 - LEO SO payload summary**

### 5.1.2 Orbit selection

In order to maximise the sun observation time, the natural choice is a sun-synchronous dawn-dusk orbit (local time at equator crossing : 6h/18h). The altitude of the orbit has to be selected according to launch constraints, platform cost minimisation and observation time maximisation.

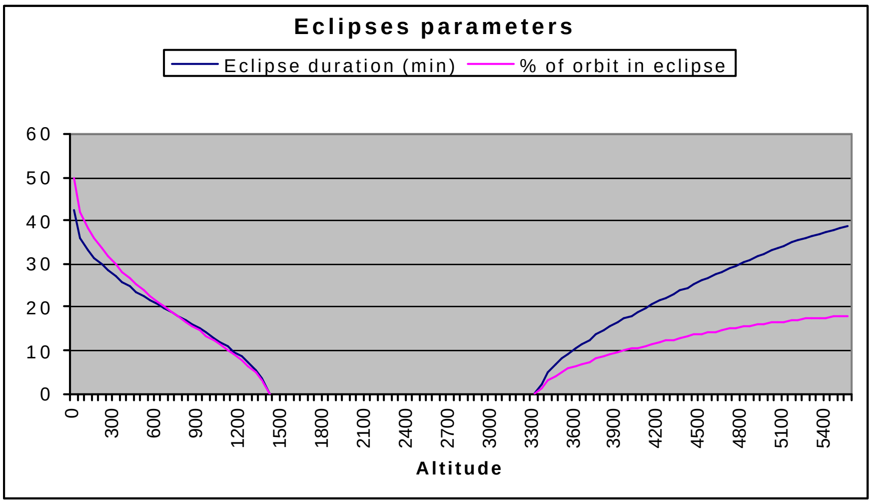



**Figure 5-1 - LEO SO eclipses vs. altitude**

The eclipse characteristics for dawn-dusk sun-synchronous orbits are plotted in table xx. This table shows that there are no eclipses for altitudes between 1500km and 3300km.

The capability of available (today and considering the mission horizon) low cost LEO launchers for this kind of orbit is on the order of 1000km (Rockot, COSMOS). Moreover, selecting a higher orbit altitude leads to an harsher radiation environment.

The baseline orbit altitude is 950km.

### 5.1.3 PROTEUS based Solar Observer

The PROTEUS capability for this mission can be summarised as follows

- Compatible with SSO orbits with altitudes of up to 1500km
- Compatible with inertial pointing
- Payload capability of up to 300kg, 300W

The principle of the implementation of a payload on PROTEUS is to build an independent payload module that has a square base of 1m x 1m, which is mounted on the platform. The platform design itself is recurrent and independent of the payload.

The aspects that would have to be investigated more particularly in an accommodation study are:

- Thermal design of the payload module
- AOCS, pointing precision, stability

- Downlink strategy and equipment

## 5.2 Upstream Monitor

In the minimum scale system scenario, the Upstream Monitor is identical to the full scale design. See §3.2.

## 5.3 Radiation Belt Monitor

In this scenario there is only one RBM, and the following differences can be identified with respect to the full scale RBM design

- There is no more need for a precise orbit orientation, as there is only one "petal" instead of three that had to be evenly spaced
- The data transfer does not go through a GEO relay as in the full scale option, therefore the communication equipment will be lighter.
- The WAVES B instrument is not included.

Under these assumptions a cost effective launch and design can be baselined : Ariane ASAP5 micro.

The design is very similar to the one presented for the full-scale RBM, in §3.3.4. The only difference is the communication equipment, as the observation data is transmitted directly to the ground, instead of the GEO relay. (Comm subsystem characteristics in this case : 8kg, 20W).

# 6. POSSIBLE ADDITIONS AND FUTURE DEVELOPMENTS

## 6.1 Magnetospheric constellation

A total of 32 small satellites are distributed into several orbits, as discussed below.

Orbital characteristics

- Equatorial highly elliptical orbits

4 orbits with inclination 0°, perigee 2.5 Re, apogee 20 Re

line of apsides separated by 90°

6 evenly spaced satellites per orbit

- Polar orbit

1 orbit with inclination 90°, perigee 2.5 Re, apogee 15 Re

6 evenly spaced satellites in this orbit

- Molniya-type orbit

1 orbit with inclination 63.4°, perigee 2 Re, apogee 8 Re

2 evenly spaced satellites in this orbit

This is very similar to the SWARM (Space Weather Advanced Research Mission) that was proposed to ESA for a F2/F3 Flexi-Mission.

## 6.2 Solar wind monitor upstream of L1

A possibility of placing a solar wind monitor upstream of L1 using solar sail technologies can be considered.

To have a significant advantage over an L1 Upstream Monitor, such a spacecraft should significantly increase the warning time. We will here consider a monitor 3 million kilometers from the Earth, which doubles the warning time as compared to an L1 monitor.

The idea is to create an artificial libration point closer to the sun by adding an acceleration to the ones whose equilibrium leads to the existence of L1. The solar radiation pressure on a solar sail is the source of this acceleration.

For a distance of 3 million km, the necessary S/m ratio is on the order of 47, meaning that a $4700m^2$ sail is required for a 100kg spacecraft.

The most promising technologies for future solar sails are inflatable structures, which allow to have very large surfaces with a very low mass (basically the soft envelope and the gas used for inflation) .

However, considering the current state of development of solar sail technologies, this option will not be considered in the first system implementation that will be proposed. It might contribute to an extension of the system capacity and performance at a later stage.

## 6.3 Solar wind monitor downstream of L1

The principle is the same as for the spacecraft upstream of L1. However a solar sail cannot be used here as the artificial acceleration required is directed towards the sun. This spacecraft will have to rely on another propulsion device. As the propulsion needs to be low thrust and continuous, electric propulsion would be a natural choice.

A characteristic of electric propulsion systems is that they have a very high specific impulse (i.e. are very efficient propellant-wise) but that they require very high amounts of power to operate.

The main design challenge is expected to be the high power required by electric propulsion rather than the propellant mass, as it is contradictory to have a high power generation capability on a spacecraft that would otherwise be of the minisat or microsat class. Therefore the propulsion type shall have a high thrust/power ratio in addition of a high specific impulse.

We will examine here Arcjet, Hall Effect and Ion thrusters, and a spacecraft dry mass of 200 kg.

The assumptions and results are summarised in the following tables:

| **Propulsion type** | **Arcjet** | **Hall Effect** | **Ion thrusters** |
|---|---|---|---|
| Electric propulsion Isp (s) | 480 | 1600 | 2500 |
| Electric propulsion thrust/power ratio (mN/kW) | 130 | 55 | 35 |
| Electric propulsion specific mass (kg/kW) | 3.5 | 7 | 25 |

**Table 6-1 - Electric propulsion systems characteristics**

These values are typical values for electric propulsion systems, in order to derive orders of magnitude. However there are variations depending on the thrust range required. A more in depth study would take these variations into account.

The lifetime has to be selected so that it is compatible with the "operational" characteristic of the overall system:

- If it has only one thruster, the spacecraft will have to be replaced at the end of life
- If it has several thrusters, they can be used alternatively, but at the expense of the spacecraft dry mass.

For a two years lifetime, the results are presented in the following table.

| | | | | | |
|---|---|---|---|---|---|
| **Arcjet** | | | | | |
| Distance from Earth (km) | 1 000 000 | 1 100 000 | 1 200 000 | 1 300 000 | 1 400 000 |
| Required propellant mass (kg) | 8752 | 2781 | 1042 | 406 | 131 |
| Required maximum thrust (N) | 2,541 | 0,601 | 0,169 | 0,05 | 0,013 |
| Required power (kW) | 19,54 | 4,63 | 1,3 | 0,39 | 0,1 |
| **Hall Effect** | | | | | |
| Distance from Earth (km) | 1 000 000 | 1 100 000 | 1 200 000 | 1 300 000 | 1 400 000 |
| Required propellant mass (kg) | 425 | 249 | 145 | 78 | 32 |
| Required maximum thrust (N) | 0,178 | 0,091 | 0,047 | 0,023 | 0,009 |
| Required power (kW) | 3,23 | 1,65 | 0,86 | 0,42 | 0,16 |
| **Ion Thrusters** | | | | | |
| Distance from Earth (km) | 1 000 000 | 1 100 000 | 1 200 000 | 1 300 000 | 1 400 000 |
| Required propellant mass (kg) | 214,96 | 135,98 | 84,01 | 47,48 | 20,43 |
| Required maximum thrust (N) | 0,118 | 0,068 | 0,039 | 0,02 | 0,008 |
| Required power (kW) | 3,36 | 1,94 | 1,11 | 0,59 | 0,24 |

**Table 6-2 - UM using electric propulsion**

The thrust, power and mass results are directly proportional to the spacecraft dry mass. Therefore the problem has to be examined in terms of power/mass ratio (A spacecraft able to produce several kW will be more massive, and therefore will require more thrust and propellant to maintain its position, and finally more power). The limitation of this approach is that as the spacecraft becomes more massive, a higher proportion of its mass is dedicated to power generation.

The demonstrated lifetimes for electric propulsion systems are today on the order of 10000 hours, or about 1.2 years. Such a lifetime for an operational solar wind monitor is not acceptable as it would lead to building and launching a spacecraft each year. However the electric propulsion technologies are evolving rapidly, and sufficient lifetime and performance capabilities might become available soon.

Considering a 2 years mission, a spacecraft 100,000 km downstream of L1 (1,400,000 km from the Earth) using Hall Effect thrusters seems a reasonable option. It would require about 160W dedicated to propulsion (to be added to the payload and platform requirement), and about 32 kg of propellant (Xenon).

A spacecraft operational lifetime of more than two years will definitely require several electric thrusters due to lifetime issues.

It must be noted that the presented results suppose a constant thrust during the spacecraft lifetime. It might be possible to have an electric thruster operation only during given stationkeeping periods (a higher thrust would then be needed). However determining the actual thrust duration over a given lifetime requires extensive mission analysis that could not be performed here. This would an issue to be addressed in case this option is studied further at a later stage.

Considering the complexity of the implementation of a monitor downstream of L1, and the overall system complexity, this option is not considered a priority and will not appear in the baseline proposal. However it might be considered for an addition to the baseline system in a future time.