Space Weather Warning System

Summary of the Group Design Project
MSc in Astronautics and Space Engineering 2011–12
Cranfield University

College of Aeronautics Report SP001

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Abstract

Students of the MSc course in Astronautics and Space Engineering 2011–12 at Cranfield University studied a space weather warning mission for their group project. The mission was inspired by the Space Weather Diamond mission proposed by St. Cyr et al. in 2001. This report summarises the students' work and their findings.

The report consists of an overview and discussion of the technical work of the project and a compilation of the executive summaries which describe the specific contributions of each student.

Two mission concepts are developed to provide at least 2 hours' warning of severe space weather events. These are an augmented Space Weather Diamond mission and a Circular Heliocentric Constellation: both missions have advantages which justify further work.

College of Aeronautics reports relevant to space engineering are listed below. These are available from the library of Cranfield University and most are available electronically.

Report	Year	Title
SP001	2012	Space Weather Warning System, Summary of the Group Design Project, MSc in Astronautics and Space Engineering 2011–12
1001	2010	Debris Removal from Low Earth Orbit (DR LEO), Summary of the Group Design Project MSc in Astronautics and Space Engineering 2009/10
0703	2007	PRIMA, Precursor Rendezvous for Impact Mitigation of Asteroids. Summary of the Group Design Project MSc in Astronautics and Space Engineering 2006/07
0509	2006	GeoSAR: summary of the group design project, MSc in Astronautics and Space Engineering 2005/06
0502	2005	Mustang 0, A low-cost technology demonstration nanosatellite; summary of the group design project, MSc in Astronautics and Space Engineering 2004/05
0206	2003	Mustang 2001, Summary of the group design project, MSc in Astronautics and Space Engineering 2001/02
0205	2003	Lunar South pole mission, Summary of the group design project, MSc in Astronautics and Space Engineering 1996/97
9918	2000	ORWELL Demonstrator, Summary of the group design project, MSc in Astronautics and Space Engineering 1998/99
0019	2000	CUSTARD, A microsystem technology demonstrator nanosatellite. Summary of the group design project MSc in Astronautics and Space Engineering 1999-2000
9903	1999	Mars Xpress, summary of the group design project, MSc in Astronautics and Space Engineering, 1997/98
9603	1996	Linear mixture modelling solution methods for satellite remote sensing

This report is available electronically from Cranfield University's CERES service (`https://dspace.lib.cranfield.ac.uk/`).

Acknowledgements

The project is very much a team effort, and contributions from all those involved are much appreciated.

First of all, the work presented is primarily that of the MSc students (Salathun Allimoopan, Richard Arundal, Fanny Auneau, Guillaume Coutinho, Aurelien Hugon, Joshua Hull, Ander Iturri Torrea, Julia Leeson, Florian Maugein, Robert Meeks, Alan McLarney, Andrew Sheppard, Susana Soto Carlavilla, Sadhana Udayakumar and Alastair Wayman), who have each contributed about 600 hours.

Other members of staff in Cranfield's Space Research Centre and contacts in the space industry have often helped students by responding to queries or providing technical information. The time spent, help provided, and general encouragement is greatly appreciated.

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Chapter 1

Introduction

This report summarizes a group project of the MSc in Astronautics and Space Engineering for the academic year 2011–12 at Cranfield University. This chapter introduces the project's purpose and management and the roles taken by individual students in the project. The rest of the report includes a technical summary and discussion of the project, and then the full set of executive summaries from the individual reports written by each student.

1.1 MSc Group Project

Each year, students of the MSc in Astronautics and Space Engineering are given a current topic in the space industry as the theme for their group project. Students work in teams of typically 8–16 students on the project, which runs from October to the end of March. One of the projects for the year 2011-12 was a Space Weather Warning mission: this report summarises the project's aims, organisation, and findings.

1.1.1 Organisation of the Project

The project runs over the first two terms (October to Easter) of the year long MSc course in Astronautics and Space Engineering at Cranfield University. The students work as one team, organised into several subgroups, and each student contributes about 600 hours' effort to the project; the total resource represented by the project is approximately 9000 hours' work (over 5 man-years) for the academic year 2011-12.

Students are given responsibility for all technical aspects of the mission and over the 6 months of the project are required to develop a credible baseline mission. There are formal weekly progress meetings which staff supervisors attend, and two key milestones. The first is a System Requirements Review (SRR) presentation in early December and the second is the more formal Preliminary Design Review (PDR) in late March. The project runs in a similar manner to many industry projects and is intended to teach both technical and transferable skills to students.

Table 1.1 lists the students involved in the project and their technical responsibilities and Figure 1.1 shows the project work breakdown structure and the main work packages allocated.

The whole team met weekly to share progress and make key decisions about the mission design. Students in each of the sub-groups also met between the main meetings as they worked on their individual responsibilities - with the system engineers working hard to coordinate all the separate tasks.

1.1.2 Technical Overview

A mission concept called Space Weather Diamond (SWD) developed for NASA by St. Cyr et al. (2000) was the project's starting point. This mission uses four spacecraft in heliocentric orbits, each carefully phased so that they form a constellation which performs relative orbits about Earth at a distance of 0.1–0.2 AU.

Table 1.1: Space Weather Warning work package breakdown and allocation. The references are to the students' individual reports documenting their technical contributions.

WP	Description	Student
WP1000 System	Requirements, Risk	(Wayman, 2012)
	Baseline	(Leeson, 2012)
	Cost, budgets	(Allimoopan, 2012)
	Operations, S/ware	(Meeks, 2012)
WP2000 Mission	Orbit perturbations, propulsion	(Sheppard, 2012)
	Orbits, deployment	(Soto Carlavilla, 2012)
	Constellation	(Auneau, 2012)
	Launch	(Mclarney, 2012)
	AOCS	(Udayakumar, 2012)
WP3000 Mechanical	Configuration	(Maugein, 2012)
	Structure	(Arundal, 2012)
	Thermal	(Iturri Torrea, 2012)
WP4000 Electrical	Power	(Coutinho, 2012)
	OBDH	(Meeks, 2012)
	Communications	(Hugon, 2012)
WP5000 Payload	Requirements	(Hull, 2012)

After studying SWD for several months it became clear that initial deployment was very challenging. The team therefore started to study an alternative mission referred to here as the Circular Constellation, for which initial deployment seemed more straightforward. Both missions are discussed in this and the students' reports on the project.

1.2 Structure of this report

Following this Introduction, Chapter 2 and Appendix A give an overview of the technical work performed by the students and summarise their findings (e.g. tables for the mass, power, cost and propulsion budgets). This chapter also serves as an overview of the constraints the design had to meet. Chapter 3 is a brief discussion of the the project's findings with some suggestions for further work. The main content of the report is Appendix B where Executive Summaries from the students' reports are presented.

This report is based on the reports written by students describing their individual project responsibilities. The full reports are available from the School of Engineering, Cranfield University, and are summarised in Appendix B. *Readers should note that although gross errors in the individual reports should have been corrected, minor inconsistencies may remain in the detailed technical work presented.*

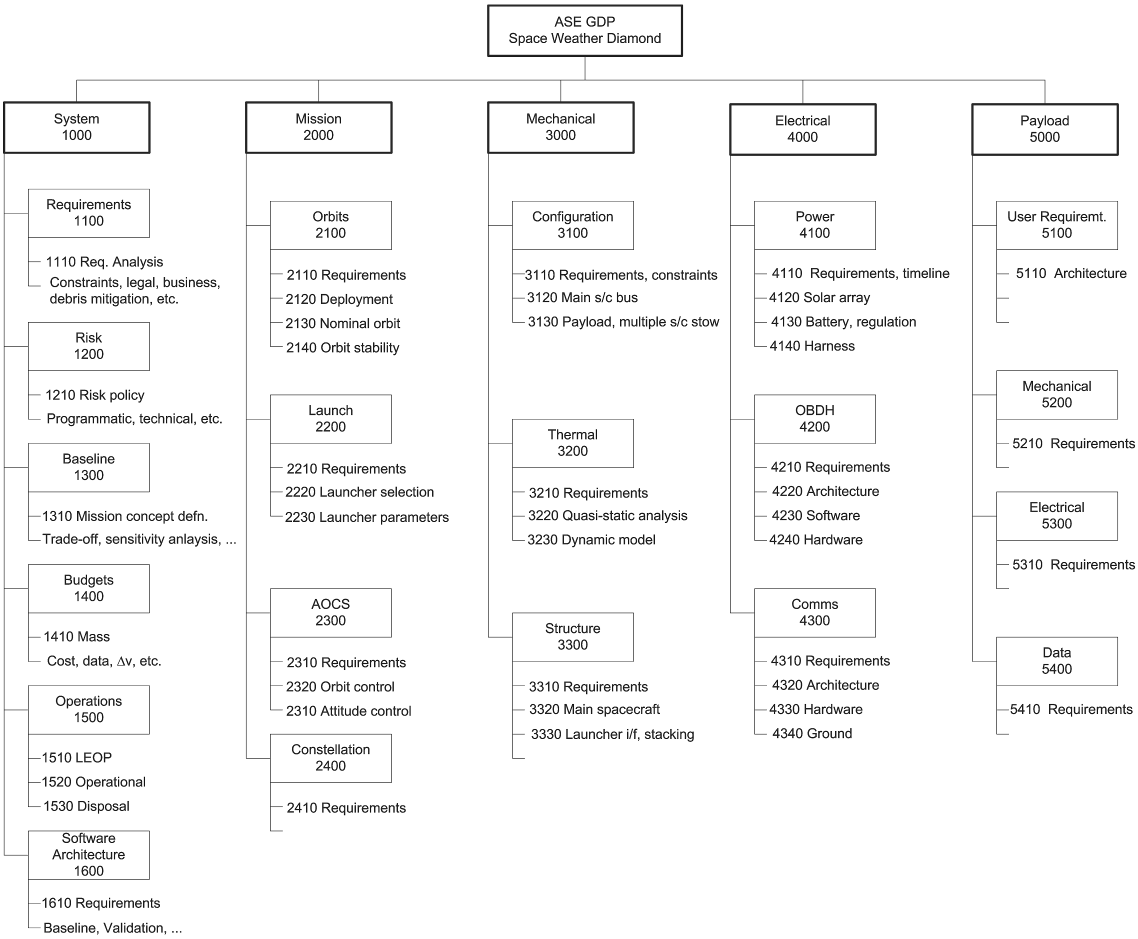

Figure 1.1: Initial work breakdown structure for the space weather warning mission (additional work packages for propulsion and mechanisms were defined as the roles were clarified).

Chapter 2

Technical Discussion

This chapter presents some of the technical areas studied within the project and gives an analysis of some of the top-level mission requirements.

2.1 Requirements Analysis

A fundamental aspect of this study is that its aim is to design an operational space weather warning system. The operational aspect brings several concerns which scientific missions are not generally concerned with.

2.1.1 Missed Detections

Two mission concepts were considered (both mentioned in St. Cyr et al. (2000)). The probability of detection of CMEs which hit Earth were calculated using a simple method illustrated in Figures 2.1 to 2.3. Contours of detection probability as a function of CME size and number of spacecraft in the two mission concepts are plotted in Figures 2.4 to 2.5.

Assumptions behind these results include that the CMEs travel radially from Sun to Earth and that the spacecraft are quasi-stationary, i.e. they do not move significantly during the time between the CME hitting the spacecraft and it hitting Earth.

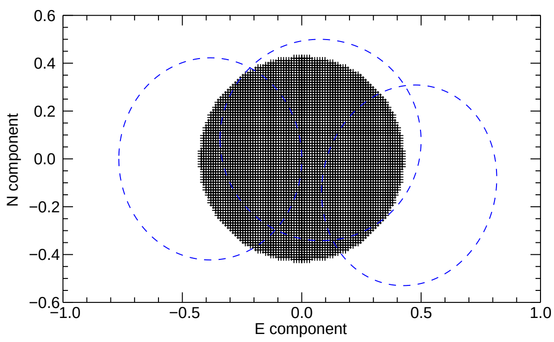

Figure 2.1: Regular grid of points used to represent possible CME directions which could impinge on Earth and the coverage provided by three spacecraft (blue dashed contours). A small set of directions near the bottom of the diagram represent CMEs which hit Earth but are not detected by any spacecraft.

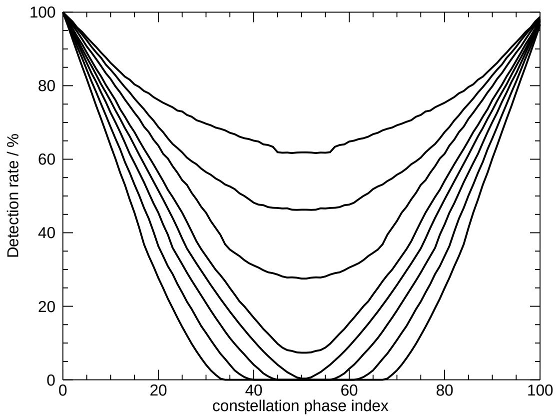

Figure 2.2: Variation in fraction of events detected as the constellation phase changes over one full cycle, for constellation sizes of 6-10, 12, 14 or 16 spacecraft.

2.1.2 False Alarms

False alarms can be costly and so the system design should minimise the rate of false alarms. If a SW event is detected by in situ sensors on one or more spacecraft which may be some distance from Earth a decision has to be made whether or not the event is a risk for Earth. If the spatial structure of the SW event is well understand then the warning can be given with some confidence. However, if the spatial structure is not known, and especially if only one spacecraft has detected it, then it is probably not possible to give a clear warning.

2.1.3 Service Quality

Operational services are characterised by specified levels of performance in areas like reliability, availability, integrity and timeliness. These are less important for scientific missions, but can influence system design (including the ground segment) significantly.

2.2 Technical

At first sight, the main technical challenges for a space weather warning system based on the SWD concept are:

- communications: link lengths are far greater than are found in Earth orbit, so data rates are likely to be low
- orbits / propulsion: manoeuvres into and between heliocentric orbits require high ΔV and are therefore likely to be an important part of mission design

Other areas such as structure, power, thermal can be based on missions already operational in Earth orbit, although there are likely to be several features which are specific to heliocentric missions and which have to be accounted for. Payload definition is always important and requires careful consideration to ensure that unnecessary requirements are not imposed on other sub-systems.

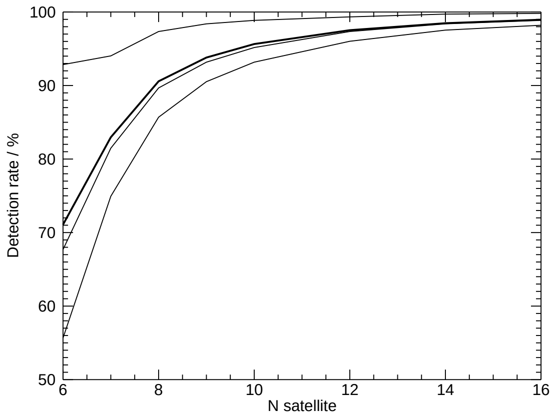

Figure 2.3: Detection probability as a function of constellation size for a CME with radius 25°(mean probability is in bold, other lines show 10th percentile, median and 90th percentile probabilities, e.g. for at least 10% of the year a 6 spacecraft constellation detects no more than 55% of CMEs even though the mean detection rate is over 70%).

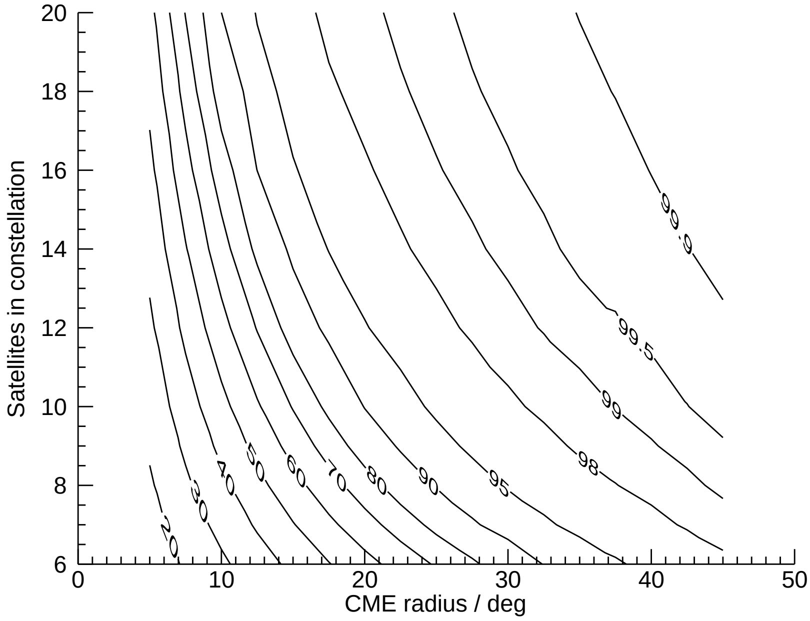

Figure 2.4: Contours of detection probability as a function of CME radius and number of spacecraft in the constellation for the Circular Constellation and CME radii from 5 to 45°(program swd6a.pro).

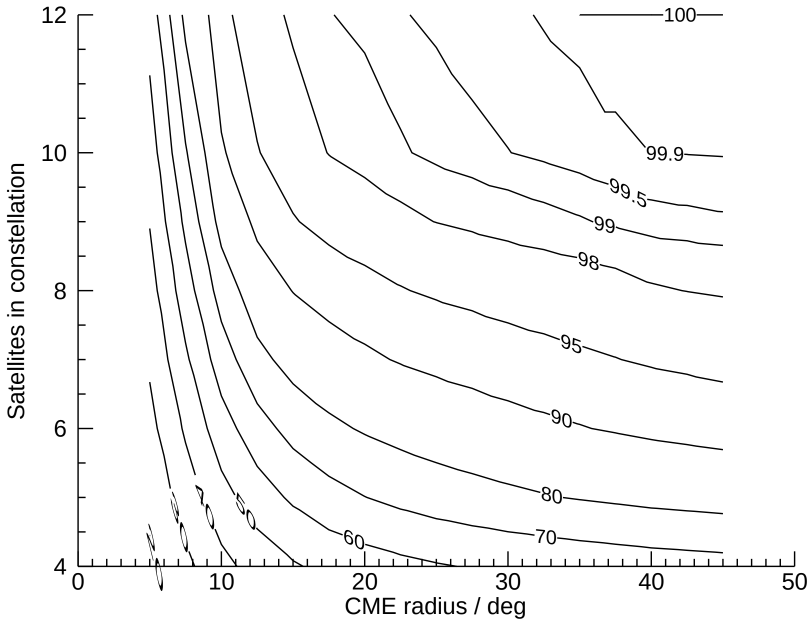

Figure 2.5: Contours of detection probability as a function of CME radius and number of spacecraft in the constellation for the SWD Constellation, CME radii from 5 to 45°, and a perigee of 0.8 AU and minimum detection time equivalent to 0.15 AU (program swd6b.pro).

Chapter 3

Discussion and Conclusions

This chapter provides brief discussion of the project's findings, a summary of its conclusions, and some suggestions for further work.

3.1 Discussion

The project's real challenges only became clear after several months' work had been completed. The challenges were technical and organisational. The main technical concern was deployment of the constellations required for both concepts. To achieve a good detection probability and a long enough warning time for the faster space weather events, large constellations were required in both concepts. SWD requires fewer spacecraft but deployment seems very expensive using conventional methods. For the Circular Constellation, deployment is simpler but still difficult to achieve without a long phasing period.

Behind these practical engineering tasks, the fundamental mission requirements also pose challenges. Space weather can clearly have major impacts on terrestrial and space infrastructure, and therefore warrants the development of appropriate warning systems to mitigate harm. However, practical warnings can impose large costs on organisations, and so there needs to be careful consideration of how to manage the risks of missed detections and false alarms. Given the limited current knowledge of harmful space weather phenomena (e.g. the spatial and temporal structure of CMEs) it is difficult to design a cost-effective measurement / warning system able to identify dangerous events reliably. This justifies continuing, focussed research into SW phenomena and perhaps some pre-cursor missions before a fully operational system can be deployed.

The other challenge for the project team was how to cope with a project which seemed to be changing direction as it developed. This was as much an organisational challenge as a technical one, and it is to the students' credit that they managed to adapt their work to cover two mission concepts to enough depth to be able to present useful findings for both.

3.2 Conclusions

In brief, the project's conclusion is that with current technology it is feasible to build space weather warning systems able to provide much longer warning times than are possible from L1. However, the project has highlighted the technical challenges, and especially the issue of initial deployment. Using conventional deployment methods, both mission concepts developed require expensive orbit changes (in terms of propulsion) to achieve the final constellation.

3.3 Future Work

As with any feasibility study like this, there are areas of further work where more study would usefully improve the proposal. Some of the areas where we would like to see more work are listed below.

- Requirements: an operational space weather warning system brings particular demands in terms of reliability and integrity as well as system architecture. Further work is required to ensure that these are met appropriately.
- Deployment: several areas require further study. These include alternative deployment methods which can reduce the ΔV required and the feasibility of more mass efficient propulsion systems than conventional chemical engines. The difficulty of using conventional orbits suggests that non-conventional orbits (e.g. non-Keplerian orbits) should also be considered.
- Communications architecture: this is a central element of any operational system. The study has been able to evaluate a few options, but others should also be investigated, especially to understand the role of intersatellite links and means of downlinking data to Earth.

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Appendix A

Mission Baseline Technical Summary

This appendix summarizes the two mission baselines developed. Both use spacecraft in heliocentric orbits. The first mission concept uses spacecraft with orbits configured to achieve relative orbits about Earth (Space Weather Diamond) and the second concept uses a constellation in circular heliocentric orbits inside Earth's orbit about the Sun.

A.1 Baseline Mission Definition - Space Weather Diamond

The Space Weather Diamond (SWD) constellation provides quasi-continuous real-time space weather warning to Earth through in situ detection. It provides 2 hours warning time for the end users. The system is able to detect all Earth-bound Coronal Mass Ejections (CMEs) with an angular width greater than 14.61°. The constellation consists of 10 identical spacecraft, each in an inclined elliptical heliocentric orbit with a radius of perihelion of 0.821 AU and a period of exactly 1 year. The spacecraft are launched in pairs, requiring 5 launches in 2021. Due to the complex launch windows, the system has a 9 hour coverage gap per year if uncorrected. The system would be fully operational by 2023, enabling it to provide space weather warning for at least 11.1 years, nearly a full solar cycle. Each spacecraft is 3-axis stabilised, is powered by solar arrays with rechargeable batteries and has a liquid bipropellant propulsion system. The total loaded mass of each spacecraft is 699 kg. The spacecraft is illustrated in Figure A.1. A detailed baseline summary can be found in Tables A.1 to A.6.

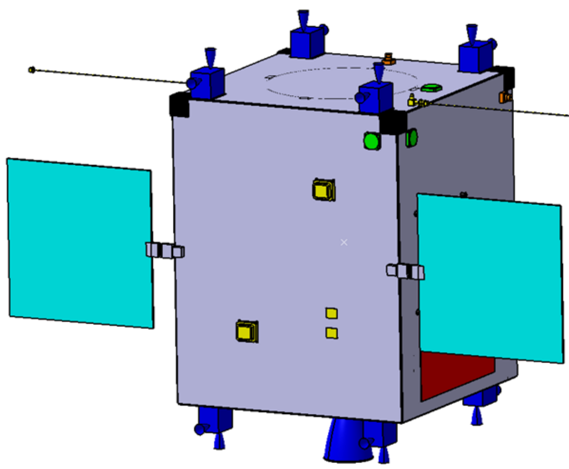


Figure A.1: SWD Spacecraft design (Maugein, 2012).

Notes for the following tables:

Table A.2	(1) Inclusive of 5% margin
Table A.6	(2) Charged Particle Spectrometer (CHaPS)

Table A.1: SWD Baseline: Systems

Parameter	Value	Unit	Notes
Systems			
Total mass	698.85	kg	
Dry mass	202.27	kg	Includes a 20% margin
Fuel mass	496.58	kg	(10.20 kg RCS, 486.38 kg for deployment burns)
Average power	171.96	W	
Peak power	237.87	W	When the spacecraft acquires the sun during deployment
Total mission cost	328.6	M €	FY11
Angular resolution	14.61	°	Width of each detection cone
OBDH			
Payload data	2.4	kbps	At a sampling rate of 10 Hz
Total housekeeping data	6.8	kbps	Peak value
Computer clock speed	1.7	MIPS	
Data storage	16.0	GB	

Table A.2: SWD Baseline: Mission

Parameter	Value	Unit	Notes
No. of spacecraft	10		All identical
Lost coverage (failure of one spacecraft)			
Without re-phasing	71.8	d	
With re-phasing	22.7	d	
1st Launch choice	Soyuz		Launched from French Guiana
2nd Launch choice	Falcon 9		Launched from Omelek Island
No. of launches	5		2 spacecraft attached to a dispenser
Total launch mass	1614.69	kg	Leaves 535 kg spare using a 217 kg dispenser
Launch date 1	20-Mar-21		Back-up launch date 6 months later for all launches
Launch date 2	25-Apr-21		
Launch date 3	01-Jun-21		
Launch date 4	07-Jul-21		
Launch date 5	11-Aug-21		
Deployment Method			Enables system to be operational by 2023
V infinity	-714.35	m s^{-1}	Provided by launcher
Deployment time	2.92	years	1 yr 11 mth (sic)
ΔV (max)	3714.76	m s^{-1}	Maximum amount required for 2nd spacecraft in each launch(1)
Operational Orbit			Heliocentric elliptical orbit
eccentricity, e	0.179		
inclination, i	0.37	°	
semi-major axis, a	1.000	AU	
period	1.000	year	
Radius of perigee	0.821	AU	
Control method: 3 axis stabilised			
AOCS - sensors			3 sun sensors 4 star trackers 2 inertial rate sensors
AOCS - actuators			4 reaction wheels 16 RCS thrusters

Table A.3: SWD Baseline: Propulsion

Parameter	Value	Unit	Notes
Liquid bipropellant system type	S400-12 Astrium		
Oxidiser	N2O4		
Fuel	MMH		
O:F ratio	1.65		
Specific Impulse, Isp	3120	m s^{-1}	
Thrust	420	N	

Table A.4: SWD Baseline: Mechanical

Parameter	Value	Unit	Notes
Configuration			
Size	2.42	m^3	Density of 288.6 $kg.m^{-3}$
Height	1.55	m	
Width	1.25	m	
Depth	1.25	m	
Thermal			
Multi-Layer Insulation	8.44	kg	
Radiators	0.66	m^2	Total area. In 2 locations
Heaters	0.38	kg	Required for RCS thrusters & apogee motor (9 in total)
Total Mass	11.00	kg	
Power	19.20	W	
Structure, all structure made from honeycomb composite			
Thrust tube diameter	0.76	m	Has propulsion tanks inside & height 1.55 m
4 side plates			
2 top and bottom plates			

Table A.5: SWD Baseline: Electrical

Parameter	Value	Unit	Notes
Power			
Solar Cells Deployed			
Type GaAs triple junction	29.5	%	efficiency
Size	1.12	m^2	
Power generated (BOL)	441.22	W	At 1 AU
Mass	9.41	kg	Including system & structure
Battery type: Li-ion			
Energy available	1000	W.h	
Mass	9	kg	2 x 4.5 kg SAFT batteries used
Total mass of power system	34.48	kg	Solar array, power system, structure and battery
Communications			
Frequency	2.2	GHz	
Inclination required	1.61	°	
High gain parabolic antenna	0.37	m	diameter
Ground station antenna	7.00	m	diameter
High gain data rate (minimum)	13.74	bps	In detection cones
High gain pointing requirement	4.15	°	HGA has its own pointing mechanism
Low gain antenna	6		patch antennae

Table A.6: SWD Baseline: Payload

Parameter	Value	Unit	Notes
Total mass	8.20	kg	With 20% margin included
Total power	10.64	W	No margin included
Instruments	4		
Magnetometer (MRMAG)	2		Placed on booms
Electron & ion spectrometer(1)	1		Sun viewing
Particle flux & species detector (HMRM)	1		Sun viewing
High energy particle detector (MuREM)	1		Perpendicular to CHaPs and HMRM

A.2 Baseline Mission Definition - Circular Constellation

The Circular baseline constellation provides continuous real-time space weather warning to Earth through in situ detection. It provides 2 hours warning time for the end users. The system is able to detect all Earth-bound Coronal Mass Ejections (CMEs) with an angular width greater than 30°. The constellation consists of 12 identical spacecraft, each in a circular heliocentric orbit with a semi-major axis of 0.853 AU and no inclination. The spacecraft are all launched together in December 2016. The system would be fully operational by 2023, enabling it to provide space weather warning for at least 7.2 years, covering the majority of a solar cycle. Each spacecraft is 3-axis stabilised, is powered by solar arrays with rechargeable batteries and has a liquid bipropellant propulsion system. The total loaded mass of each spacecraft is 330 kg. The spacecraft is illustrated in Figure A.2.

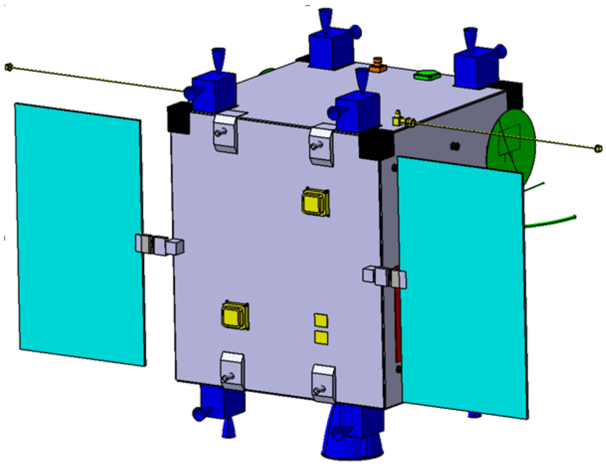


Figure A.2: Circular Spacecraft design (Maugein, 2012).

Notes for the following tables:
Table A.8 (1) Inclusive of 5% margin
Table A.12 (2) Charged Particle Spectrometer (CHaPS)

Table A.7: Circular Baseline: Systems

Parameter	Value	Unit	Notes
Systems			
Total mass	329.56	kg	
Dry mass	190.36	kg	Includes a 20% margin
Fuel mass	139.20	kg	(4.72 kg RCS, 134.48 kg for deployment burns)
Average power	188.49	W	
Peak power	228.27	W	When the spacecraft acquires the sun during deployment
Total mission cost	235.1	M €	FY11
Angular resolution	30.00	°	Width of each detection cone
OBDH			
Payload data	2.4	kbps	At a sampling rate of 10 Hz
Total housekeeping data	4.4	kbps	Peak value
Computer clock speed	1.8	MIPS	
Data storage	16.0	GB	

Table A.8: Circular Baseline: Mission

Parameter	Value	Unit	Notes
No. of spacecraft	12		All identical
Lost coverage (failure of one spacecraft)			
Without re-phasing	105.5	d	
With re-phasing	5.5	d	
1st Launch choice	Ariane 5		Launched from French Guiana
2nd Launch choice	Delta 4 M+(4,2)		Launched from Florida
No. of launches	1		12 spacecraft attached to a dispenser
Total launch mass	4454.76	kg	Leaves 1045 kg spare using a 500 kg dispenser
Launch date	21-Dec-16		Enables system to be operational by 2023
Deployment Method			Launches all spacecraft into a series of circular orbits
V infinity	-1206.04	m s^{-1}	Provided by launcher
Number of phasing orbits	6		Maximum
Deployment time	5.83	years	Limited by the 11th spacecraft
ΔV (max)	1635.68	m s^{-1}	Maximum amount required for 12th spacecraft(1)
Operational Orbit			Heliocentric circular orbit - the same for all spacecraft
eccentricity, e	0		
inclination, i	0	°	
semi-major axis, a	0.853	AU	
period	0.787	years	
Control method			3 axis stabilised
AOCS - sensors			3 sun sensors
			4 star trackers
			2 inertial rate sensors
			1 sun sensor
			2 star trackers
			1 inertial rate sensor for redundancy
AOCS - actuators			2 momentum wheels
			16 RCS thrusters
			1 momentum wheel
			4 RCS thrusters for redundancy

Table A.9: Circular Baseline: Propulsion

Parameter	Value	Unit	Notes
Liquid bipropellant system type		S400-12 Astrium	
Oxidiser	N2O4		
Fuel	MMH		
O:F ratio	1.65		
Specific Impulse, Isp	3120	m s^{-1}	
Thrust	420	N	

Table A.10: Circular Baseline: Mechanical

Parameter	Value	Unit	Notes
Configuration			
Size	1.10	m^3	Density of 299.6 $kg.m^{-3}$
Height	1.10	m	
Width	1.00	m	
Depth	1.00	m	
Thermal			
Multi-Layer Insulation	4.71	kg	
Radiators	0.62	m^2	Total area. In 2 locations
Heaters	0.38	kg	Required for RCS thrusters & apogee motor (9 in total)
Total Mass	7.14	kg	
Power	19.20	W	
Structure, all structure made from honeycomb Al-alloy			
Thrust tube diameter	0.49	m	Has propulsion tanks inside & height 1.1 m
Side plates	3		
radiation shielding	1		front plate, includes 4mm Al-alloy sheet for radiation protection
top and bottom plates	2		

Table A.11: Circular Baseline: Electrical

Parameter	Value	Unit	Notes
Power			
Solar Cells Deployed			
Type GaAs triple junction	29.5	%	efficiency
Size	1.12	m^2	
Power generated (BOL)	434.39	W	At 1 AU
Mass	9.41	kg	Including system & structure
Battery type: Li-ion			
Energy available	1000	W.h	
Mass	9	kg	2 x 4.5 kg SAFT batteries used
Total mass of power system	34.58	kg	Solar array, power system, structure and battery
Communications			
Frequency	2.2	GHz	
Inclination required	0.0	$^\circ$	
High gain parabolic antenna	0.70	m	diameter
Ground station antenna	7.00	m	diameter
High gain data rate (minimum)	13.8	bps	In detection cones
High gain pointing requirement	1.3	$^\circ$	HGA has its own pointing mechanism
Low gain antenna	2		patch antennae
Inter-satellite links			
Frequency	27	GHz	
Antenna diameter	0.34	m	2 per spacecraft
Pointing requirement	0.2	$^\circ$	Antennae have their own pointing mechanisms

Table A.12: Circular Baseline: Payload

Parameter	Value	Unit	Notes
Total mass	8.20	kg	With 20% margin included
Total power	10.64	W	No margin included
Instruments	4		
Magnetometer (MRMAG)	2		Placed on booms
Electron & ion spectrometer$^{(2)}$	1		Sun viewing
Particle flux & species detector (HMRM)	1		Sun viewing
High energy particle detector (MuREM)	1		Perpendicular to CHaPs and HMRM

Appendix B

Individual Report Executive Summaries

Executive summaries for all the project reports are given in this appendix. Full copies of the reports may be referred to at the School of Engineering, Cranfield University, UK.

The summaries presented here have been only lightly edited. Users of the summaries and reports should bear in mind that although efforts have been made to correct any significant errors, it is possible that some minor errors remain.

The reports are ordered alphabetically by author surname. Figure 1.1 shows the project work breakdown structure and students' individual responsibilities within the project.

Student	Work area
Salathun Allimoopan	Cost and budgets
Richard Arundal	Structure
Fanny Auneau	Constellation
Guillaume Coutinho	Electrical power
Aurelien Hugon	Operations and communications
Joshua Hull	Payload and ground segment
Ander Iturri Torrea	Thermal design
Julia Leeson	Baseline design
Florian Maugein	Configuration
Alan McLarney	Launchers
Robert Meeks	OBDH and software
Andrew Sheppard	Orbit perturbations and propulsion
Susana Soto Carlavilla	Orbits and deployment
Sadhana Udayakumar	AOCS
Alastair Wayman	Requirements and risk

Table B.1: Sub-system responsibilities for each student

B.1 Cost and Budgets: Salathun Allimoopan

Space weather is a storms of coronal mass ejection (CMEs) and solar flares caused by sun's plasma and its magnetic field, these storms spreads throughout the solar system, as it approaches the earth most of these storms miss out but still some interact with earth's atmosphere and affecting earth's environment. Particularly it is being a growing concern to power grid, airlines and astronauts in space.

The mission aim is to provide feasible baseline design for space weather. Hence, two different designs were studied, and one of them chosen to satisfy the mission requirements.

The objective of this report looks at providing mission cost with estimated budget of €300M and also identifying and providing acceptable mass, and power budget for the mission.

B.1.1 Mission Cost

Cost is the most crucial part of engineering discipline to know whether or not a project comes to fruition. In space Industry, cost estimation is analysed by software which is pre-programmed with industry-standard cost models such as Small Satellite Cost Model (SSCM) and Unmanned Space vehicle Cost Model (USCM).

In this project mission cost is worked on the parametric-based estimation and have estimated for two different missions- Space Weather Diamond Mission and Circular Mission, this has been summarised in the following sections.

SWD Mission Cost

In this mission we use constellation of 10 Spacecraft. 2 Spacecraft per launcher, therefore 5 launchers are used. Soyuz is used as primary launcher and Falcon9 used as backup launcher, cost of the mission with SOYUZ launcher is €341M and Cost of the mission with Falcon9 launcher is €371. Both does not meet allocated mission budget. Total mission cost is given in Table B.2

Table B.2: Space Weather Diamond Mission cost

Cost Component	Parameter	Unit	Cost (FY10$k)	RDT & E Cost (FY10$K)	Total Unit Cost for 10 S/C(FY10$K)
1.Payload	8.2	kg	9,753.14	5,851.89	19,618.25
2. Spacecraft Bus			24,382.86		
2.1 Structure	52.75	kg	4,444.24	3,110.97	7,815.68
2.2 Thermal	11	kg	1,024.70	512.35	2,320.28
2.3 Electrical Power System	34.48	kg	8,157.63	5,057.73	15,996.35
2.4 TT & C	4.26	kg	878.64	623.84	1,522.97
2.5 OBDH	2.5	kg	916.39	650.64	1,588.40
2.6 AOCS	22.02	kg	7,523.10	2,783.55	19,508.00
2.7 Propulsion	127.01	kg	1,438.15	719.08	3,256.48
MMH	8.1034004	Gallon	3,548.15		
N2O4	8.1205065	Gallon	266.27		
He	0.4873543	Gallon	0.01		
Total Cost of Spacecraft					108,182.33

Circular Mission Cost

In circular mission we use constellation of 12 Spacecraft, all the Spacecraft are stacked in one launcher. Arian 5 is the primary launcher and Delta 4 is the backup launcher. Total cost of the mission with Arian 5 for the financial year 2011 is €241M and total Mission cost with Delta4 launcher is €245M. The total mission cost can be seen in Table B.5

Table B.3: Space Weather Diamond Mission cost

Wraps	Parameter	Cost (FY10$k)	RDT & E Cost (FY10$K)	Total Unit Cost for 10 S/C(FY10$K)
3. Integration, Assembly & Test(IA&T)	Spacecraft total cost	8,134.12	0.00	28,702.86
4. Program Level	Spacecraft total cost	13,400.82	6,700.41	30,344.14
5. Ground Support Equipment (GSE)	Spacecraft total cost	3,862.25	3,862.25	3,862.25
6. Launch & Orbital Operations Support	Spacecraft total cost	3,569.65	0.00	12,596.22

Table B.4: Space Weather Diamond Mission cost

Insurance	Parameter	
Launch insurance	Spacecraft total cost	5,409.12
Mission insurance	Spacecraft total cost	5,409.12
Total		10,818.23

B.1.2 Mission Budget

The main purpose of the budgets is to monitor and resolve any key changes in the mission budgets; they are the Mass and the Power

Mass Budget

Mass budget of the spacecraft increased considerably throughout the project. An Initial allocation of 30% of margin was added for any contingency. As the design progressed margin has dropped down to 20%. As we have two different designs, Mass budget of the Space Weather Diamond is given in table 2.1a and Mass budget for Circular mission is given in table 2.1b

Power Budget

Preliminary power budget update was given by each subsystem engineers and power for different mission modes were worked by Guillaume. Initially body mounted solar cells were considered for the design, as it would not provide required power, we chose deployable solar panels. Moreover, the power required by individual subsystems varies according to the modes of operation. In circular mission the power budget is mainly driven by the communication subsystem, which is mainly because the communication uses inter-satellite links. The second key driver is being the power subsystem itself, because the power subsystem requires sufficient amount of power to provide for other subsystem and mechanism, for instance deployment mechanism and antenna pointing mechanism.

Table B.5: Circular Mission cost

Cost Component	Parameter	Unit	Cost (FY10$k)	RDT & E Cost (FY10$K)	Total Unit Cost of 12 S/C(FY10$K)
1.Payload	8.2	kg	9,684.05	5,810.43	19,577.59
2. Spacecraft Bus			24,210.13		
2.1 Structure	41.72	kg	3,411.17	2,387.82	6,024.89
2.2 Thermal	7.14	kg	625.58	312.79	1,424.48
2.3 Electrical Power System	34.58	kg	8,172.03	5,066.66	16,103.40
2.4 TT & C/DH	19.97	kg	3,646.85	2,589.26	6,348.00
2.5 OBDH	2.5	kg	916.39	650.64	1,595.15
2.5 ADCS	18.9	kg	6,029.36	2,230.86	15,731.02
2.6 Propulsion dry mass	124.81	kg	1,408.75	704.38	3,207.78
MMH	8.10340038	Gallon	130.97	1,571.65	
N2O4	8.12050645	Gallon	266.27	3,195.26	
He	0.48735425	Gallon	0.01	0.08	
Total Cost of Spacecraft				58,104.31	73,184.14

Table B.6: Circular Mission cost

Wraps	Parameter	Cost (FY10$k)	RDT & E Cost (FY10$K)	Total Unit Cost (FY10$K)
3. Integration, Assembly & Test(IA&T)	Spacecraft total cost	8,076.50	0.00	28,704.52
4. Program Level	Spacecraft total cost	13,305.89	6,652.94	30,298.04
5. Ground Support Equipment (GSE)	Spacecraft total cost	3,834.88	3,834.88	3,834.88
6. Launch & Orbital Operations Support	Spacecraft total cost	3,544.36	0.00	12,596.95

Table B.7: Circular Mission cost

Insurance	Parameter	Cost (FY10$k)
Launch insurance	Spacecraft total cost	3,659.21
Mission insurance	Spacecraft total cost	3,659.21
Total Cost (FY10$K)		7,318.41

Table B.8: Space Weather Diamond Mass Budget

Description	Total Mass	Units
Payload	8.20	kg
Power Subsystem	34.48	kg
Propulsion Subsystem	33.35	kg
Communications Subsystem	4.26	kg
OBDH	2.50	kg
Structure	52.75	kg
Thermal	11.00	kg
AOCS	22.02	kg
Spacecraft Dry Mass, Mdry	168.56	kg
Spacecraft Dry Mass (20% Margin)	202.27	kg
Spacecraft propellant mass, Mprop	496.58	kg
SPACECRAFT LOADED MASS Mload	698.85	kg
Total number of s/c per launcher (2)		
Total mass per launcher needed for s/c	1397.69	kg
Adapter mass	217.00	kg
Total launch mass	1614.69	
Total mass per launcher can be available	2150.00	
Launcher margin	535.31	

Table B.9: Circular Mission Budget

Description	Total	Units	Comments
Payload	8.20	kg	
Power Subsystem	34.58	kg	
Propulsion Subsystem	25.63	kg	
Communications Subsystem	19.97	kg	
OBDH	2.50	kg	
Structure	41.72	kg	
Thermal subsystem	7.14	kg	
AOCS subsystem	18.90	kg	
Spacecraft Dry Mass, Mdry	158.64	kg	
Spacecraft Dry Mass (20% Margin incl.)	190.36	kg	
Spacecraft propellant mass, Mprop	139.20	kg	
SPACECRAFT LOADED MASS Mload	329.57	kg	
Dispenser Mass	500.00	kg	Estimate from Globalstar dispenser mass
Number of Spacecraft per Launch 12			
TOTAL LAUNCH MASS	4454.76	kg	
Total launch mass available with Ariane 5	5500.00	kg	
Space available in launcher	1045.24	kg	

B.2 Structure: Richard Arundal

For this mission, the aim is to setup a constellation of spacecraft in an orbit that allows monitoring of solar activity and to be exposed to coronal mass ejections that approach the Earth with the intention of providing a warning system to terrestrial based organisations to take protective measures for assets and services (such as airports, communications etc.). The cost of the mission was set for a 300 Million Euro budget.

B.2.1 Primary Structure Overview

The monocoque sandwich cylinder was designed and sized using the forces and loads calculated in respect to the wet mass of the spacecraft and of the load multipliers in relation to the launch. The minimum thickness and masses required are summarised below. The Circular concept was modelled using the properties of Aluminium whilst the SWD concept was modelled using composites. The various reasons why these were chosen are discussed in the main body of the report.

Table B.10: Thrust Tube Sizes

Parameter	Value	Unit
Mass	7.350435	kg
Thickness	0.001299	m
Gauge of Sheet	0.000574	m
Mass	2.338507	kg
Thickness	0.001585	m
Gauge of Sheet	0.000226	m

Table B.11: Modelling radiation plate mass

Parameter	Value	Unit
Thickness of both plates	1.1481	mm
Extra thickness of Al required to meet 4 mm	2.8519	mm
Additional mass of Al	9.582	kg
Total mass for radiation plate	14.983	kg
Total mass replacing 1 side plate with radiation plate	40.7	kg

According to research by the ESA in a previous Space Weather project, there is a minimum thickness of aluminium that is required in order to protect a spacecraft from radiation effects. Table B.11 shows the impact of using a minimum of 4 mm Al shielding.

By returning to the design of the spacecraft panels for the Circular concept, the thickness is adjusted for one of the panels (namely the sun facing side) and the mass change was calculated accordingly. With the other 3 panels weighing 31.1 kg, this adds approximately 9.6 kg extra to each of the spacecraft, as shown in Table B.11.

Sizing of the panels for each concept are summarised below. A trade off comparison was generated and the group was invited to score the materials in accordance with their own knowledge and relevance to their work packages.

B.2.2 Manufacture Overview

Fig. B.1 illustrates the assembly process. The bottom plate, (1) will be fitted into a rig and form the foundation of the assembly, of which the nozzle will be then inserted from underneath. The thrust tube (with fuel tanks, etc.) (2) will then be fitted from above and secured to the bottom plate. The side plates (3) can then be attached in any order and at this stage can easily be detached and reattached in the event of modifications to the panel mounted components. The bottom plate

Table B.12: Space Weather Diamond Panel Sizing

Mass (Per plate)	2.724404	kg
If we use the sandwich panel idea, then:		
Thickness of both plates	0.001626	Metres
Minimum Face sheet spacing, h	0.000616	Metres
Assuming 4mm Face sheet spacing	0.004	Metres
2 x plate thickness + Face sheet spacing	0.005626	Metres
Mass of both plates	5.448808	kg
40% of this mass for honeycomb	2.179523	kg
Combined Mass *** Per SIDE Panel ***	7.63	kg
Combined Mass ** Per TOP/BOTTOM Panel **	6.57	kg
Total: Assuming 6 panels	43.661	kg

Table B.13: Circular Panel Sizing

Mass (Per plate)	1.928774	kg
If we use the sandwich panel idea, then:		
Thickness of both plates	0.001148	Metres
Minimum Face sheet spacing, h	0.00051	Metres
Assuming 4mm Face sheet spacing	0.004	Metres
2 x plate thickness + Face sheet spacing	0.005148	Metres
Mass of both plates	3.857549	kg
40% of this mass for honeycomb	1.54302	kg
Combined Mass *** Per SIDE Panel ***	5.4	kg
Combined Mass ** Per TOP/BOTTOM Panel **	4.76	kg
Total: Assuming 6 panels	31.118	kg

to side panel brackets would be fitted also. Once the side plates are secured, then the top plate (4) can be fitted and attached to the thrust tube, along with the corner brackets on the outside.

B.2.3 Mechanisms Overview

The following idea concept that was generated incorporates a very close rotating antenna using a drive and a control system to add to the inclination should it be required. Designing the mechanism to be able to rotate for the highest field of view value ensures that both concept requirements are satisfied. Initial considerations are that the slight inclination introduced may have some issues regarding the gears (i.e. teeth alignment) but further investigation into the impact of a ± 0.5 degree alteration would have to be considered. The lateral damping roller would be pressed against the outside of the halo, pushing it *into* the spacecraft, as such, whilst the axial damping rollers would maintain inclination. The damping of the rollers would assist for vibration and loads during the launch, but would have to be stiff enough to ensure that the pointing accuracy of the antenna was acceptable and to keep the rolling frame halo in position. Fig B.2 is the initial design of this antenna mechanism.

The uppermost dampers would be slightly stiffer in order to naturally push down the antenna to point at a lower inclination, as there is no gravity in space to pull the antenna down naturally. In addition, they would also have to flexible enough to allow the inclination changes introduced by the tension cables. As part of the interface fitting, consideration must go into the pins that will attach themselves to the interface within the launcher. The first consideration is actually to have rods that are attached to structurally robust parts of the structure. This will not only assist in reinforcing the structure but will provide support for the mounting. The pin fits within the dispenser, and the spring assist mechanism that comes as part of the dispenser pushes against the flat area beneath the attachment pin when required. The benefits of using this attachment shown in Fig B.3 is that it can be easily attached dependent on the concept chosen without drastically changing the structural design; it is a modular component.

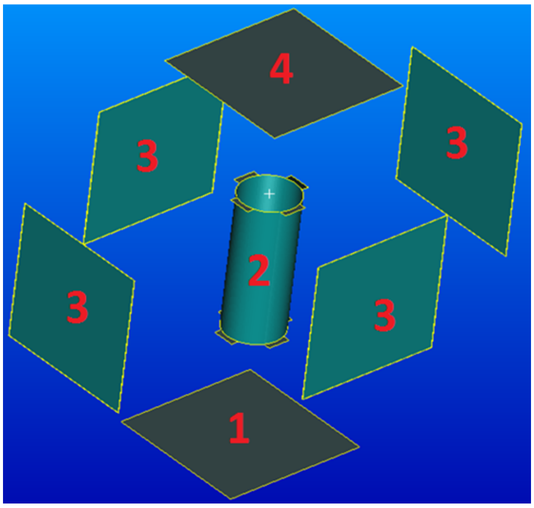


Figure B.1: Assembly process

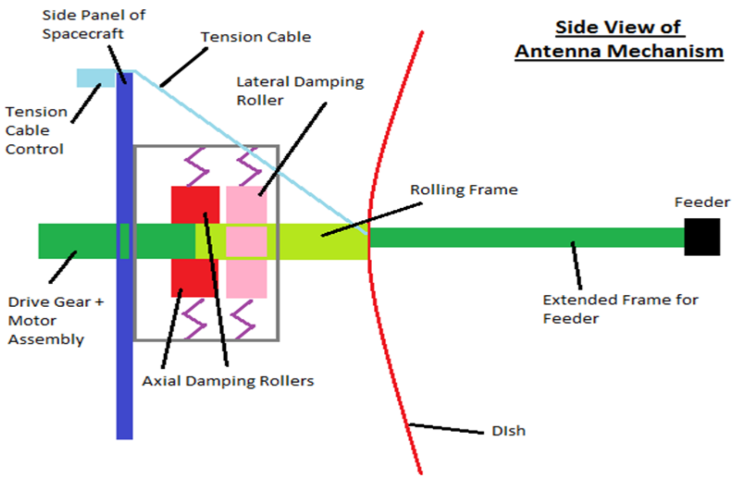


Figure B.2: Antenna mechanism concept

B.2.4 Surface Charging Overview

Surface charges are generated due to the effects of space plasma and photoelectric currents. As a satellite orbits between sunlight and shadow, a discharge with a high enough potential can be created. Charges can be more easily dissipated by using an electrically conductive material, hence why Aluminium is the ideal choice for spacecraft material, as used in the Circular concept. In the event of the satellite being within the cone of a CME, then these charges have the capability of reaching thousands of volts, possibly even millions.

B.2.5 Key References

Sarafin, (2007), Spacecraft Structures and Mechanisms, New York, Microcosm Press.

Steinmeyer et al., (1999), Patent 5,884,866 (Satellite Dispenser).

Wijker, J. J. (2008), Spacecraft structures, Springer, Berlin. ISBN 978-3642094774.

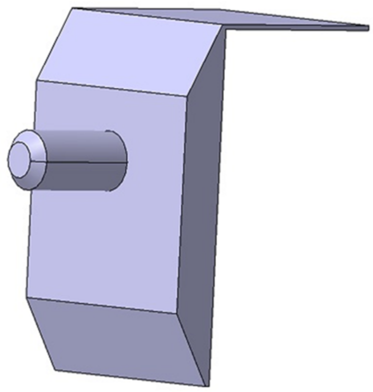


Figure B.3: Interface adapter

B.3 Constellations: Fanny Auneau

B.3.1 Background

Space Weather is becoming of greater concern nowadays. Especially Coronal Mass Ejections and the charged particles coming from the solar wind can have harmful effects on both humans and technologies on and in the vicinity of the Earth. As humans are more and more dependent on technologies (electricity, Internet and telecommunications), big solar events could have a devastating domino effect on our global economy.

To prevent such event happening, it is important to enable companies to take protective measures against these unstoppable solar events. The more warning time before a CME hits the Earth is given, the more time companies have to disconnect power plant lines from the network or reroute aircraft for example. Current Space Weather Missions like ACE and SOHO orbiting at the L1 Lagrange point, only provide 30 to 60 minutes warning time. As this is not enough for certain companies and the next solar maximum after 2013 is predicted to happen around 2024, a study was proposed to define a new warning system giving a better warning time and being fully operational for the next solar maximum.

The study focussed on 2 different designs. The first is called the "Space Weather Diamond Design" (named after the initial study led by St. Cyr and his team (St. Cyr, 2000)) and the second is named the "Circular Design". Both concepts are based on a constellation of spacecraft continuously monitoring the Sun. The "Space Weather Diamond Design" is based on a constellation using eccentric heliocentric orbits and the "Circular Design" constellation's uses a circular heliocentric orbit. When a CME is detected by the system to hit the Earth, a signal is sent back to Earth warning them a solar event is going to affect the Earth. The end users can then take protective measures (see Figure B.4).

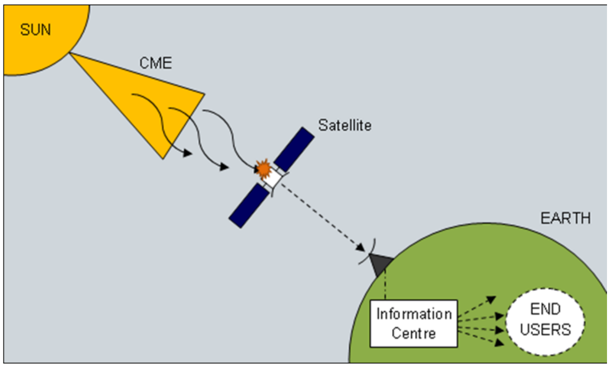


Figure B.4: Sub L1 Space Weather Warning Systems Concept.

B.3.2 Constellation Analysis

The Constellation Work Package's main aim was to determine the configuration of the spacecraft's constellation for both designs. It includes the number of satellites needed to satisfy the requirements and their positions relative to each other. The main objectives from a constellation point of view was to provide at least 2 hours warning time and be able to detect CME's wider than 35°. The number of spacecraft needed in a constellation depends on the operational orbit chosen for each satellite which depends on the set requirements (detection cones). The height of the detection cone determines the warning time provided and the cone angle determines the angular resolution of the system or the size of CME's the system can detect.

B.3.3 Space Weather Diamond Design

As the relation between all these parameters was unknown for the complex SWD constellation, a study has been done to choose a constellation and the operational orbit that would satisfy the requirements. A mission trade-off was performed including the DeltaV parameters (Soto Carlavilla, 2012). The DeltaV required for each spacecraft in the constellation could determine if it was possible to choose a specific orbit or not. The more eccentric the operational orbit is, the more propellant for each satellite is needed. The case of the loss of one spacecraft and its impacts on the constellation were also taken into account. Finally, a constellation of 10 spacecraft was chosen with the orbital parameters in Table B.14.

Table B.14: Main orbit / constellation parameters for the SWD design.

Type of orbit	Elliptical	
Number of spacecraft	10	
Semi-major axis (AU)	1.000	Same orbital period as Earth so each spacecraft describes a relative orbit around Earth.
Radius of perihelion (AU)	0.821	
Eccentricity (-)	0.179	
Inclination (degrees)	0.370	For communication purposes.
Difference in RAAN (between each satellite) (degrees)	36	To have a continuous coverage with a minimum number of satellites.
Warning time (hr)	2	Greater than with the initial SWD design.
Cone height (AU)	0.870	
Angular resolution (degrees)	14.6	Smaller than 35°, quite good angular resolution.

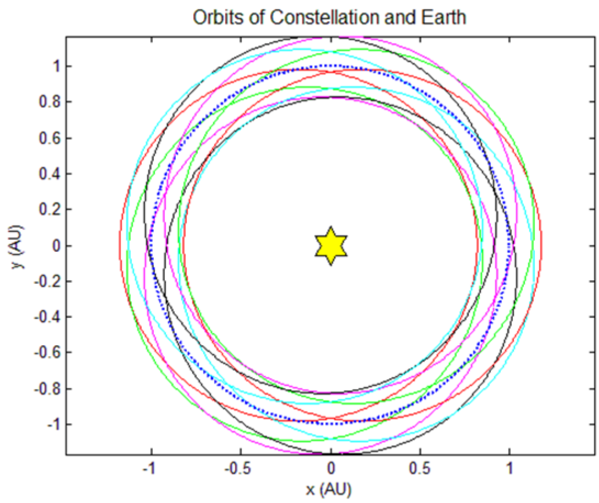


Figure B.5: Plot showing the Orbits of the SWD Constellation and the Earth with the Sun in the Centre.

B.3.4 Circular Design

The relation between the operational orbit chosen and the number of satellites needed in the constellation to cover it was known, the study done on this design was easier.

The final design chosen is a constellation of 12 spacecraft with the orbital parameters in Table B.15.

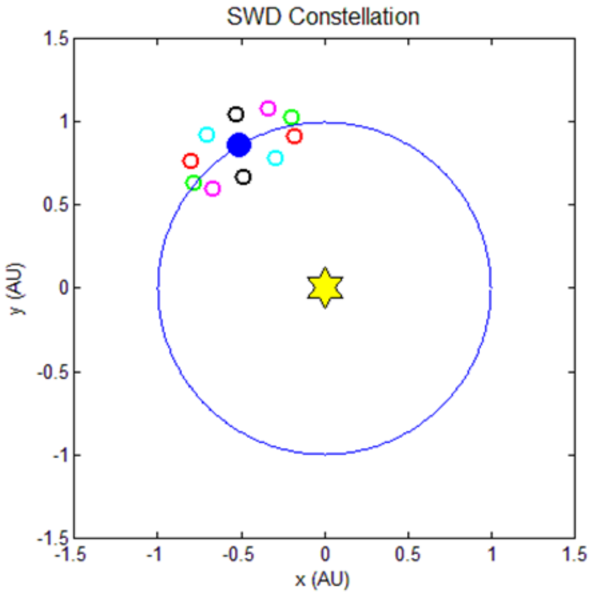


Figure B.6: Plot showing the SWD Constellation (Satellites represented by coloured circles) describing a Relative Orbit around the Earth (full dark blue circle, blue line represents Earth's Orbit) at one time.

Table B.15: Main Orbital/Constellation Parameters for the Circular Design.

Type of orbit	Circular	
Number of spacecraft	12	
Semi-major axis (AU)	0.853	Shorter orbital period than Earth, no relative orbit around Earth.
Radius of perihelion (AU)	0.853	
Eccentricity (-)	0	
Inclination (degrees)	0	Use of inter-satellite link.
Difference in RAAN (between each satellite) (degrees)	30	To have a continuous coverage with a minimum number of satellites.
Warning time (hr)	2	Greater than with the initial SWD design.
Cone height (AU)	0.853	
Angular resolution (degrees)	30	Smaller than 35°

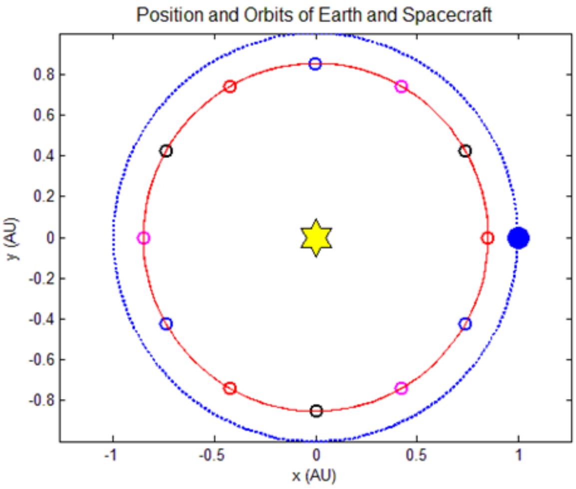


Figure B.7: Plot showing the Spacecraft's and the Earth's Orbits and Position at one specific time for the Circular Design.

B.4 Power and Payload Electrical system: Guillaume Coutinho

This appendix is a summary of the power and the electrical system of the project "Sub L1 Space Weather Warning Systems". Its aims is explaining in a short way the method to size these systems. The two designs are discussed but not in detail.

B.4.1 Introduction

Space Weather and its consequences on spacecraft, the human health and the near Earth environment are a topical issue, not only for the space industry but also for the airline industry and the power industry. In this context, our group of 15 students began the project "Sub L1 Space Weather Warning Systems".

This project consists of defining the baseline of a consistent mission with a better coverage and warning time than current systems. Two designs have emerged from our study and will be compared.

This report focuses on the power and the electrical systems which respectively has to provide the power required by other systems of the spacecraft without interruptions and has to manage and distribute this power to the other systems while protecting them from all damage caused by current or voltage anomalies.

B.4.2 Power budget and modes

When we talk about the power system, it is mandatory to study the power budget and modes because all the power system depends on the power budget and modes. Defining properly and clearly modes is one of the most important steps when we size the power system.

Table B.16: Average and peak power

	Average Power	Peak Power
SWD Design	171.96 W	237.87 W
Circular Design	203.33 W	242.67 W

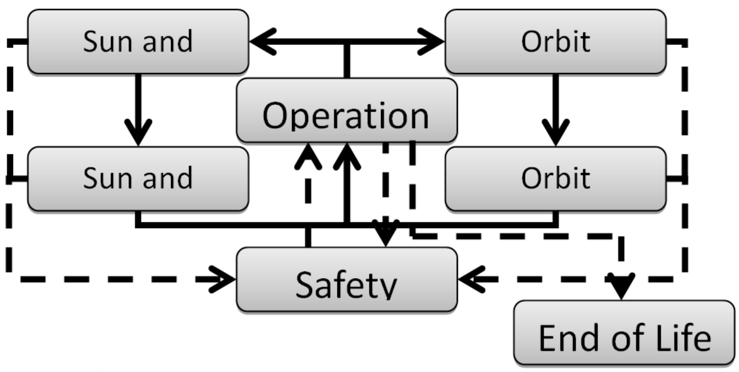


Figure B.8: Mode Diagram

B.4.3 Batteries

As in both designs there is no eclipse time in the Operational Orbit, the battery is sized to provide enough power during burns which allow the spacecraft changing its orbit by itself. Currently, the most efficient battery is the Lithium Ion battery.

Table B.17: Batteries

	Battery capacity	Number of batteries	Total mass
SWD Design	1002.60 W.hr	2	2x4.5 = 9 kg
Circular Design	983.42 W.hr	2	2x4.5 = 9 kg

B.4.4 Solar panels

Solar panels are used only during the Operational Mode. Thus, I used the power required in the Operational Mode to size the solar panels. The other key factors to size solar panels are the distance from the Sun, especially the maximum distance from the Sun and the lifetime which determine the degradation of solar panels over the time. The environment and particularly the radiation environment of solar panels have also an impact on their degradation. Moreover, due to the amount of power the spacecraft has to dissipate in the case of the use of body-mounted solar panels, deployable solar panels are used to be able to introduce cosines losses in order to produce only the power the spacecraft required. Currently the most efficient photovoltaic cells are the triple junction Gallium Arsenide cells which can reach en efficiency of 30% approximately. The standard size for the European cells is 20x40 cm^2.

Table B.18: Batteries

	Area calculated	Number of cells	Effective area	Total mass
SWD Design	1.07 m^2	2 panels of 7 cells	1.12 m^2	9.41 kg
Circular Design	1.12 m^2	2 panels of 7 cells	1.12 m^2	9.41 kg

B.4.5 Electrical system

The both designs use the same electrical diagram.

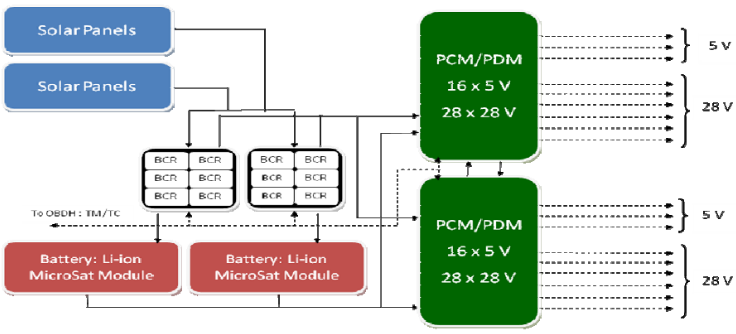


Figure B.9: Electrical diagram

Moreover, expect the fact that the Circular design needs two additional converters, all other electrical components are exactly identical.

B.4.6 The both designs

Due to the method to size the power and the electrical systems, the both designs provide the power required by the spacecraft for at least 13 years.

Moreover, since same components are chosen or both designs, the cost depends on the number of components. Expect the two additional converters for the Circular design the number of each component is identical for both designs. The remaining two requirements that we have to compare are the total mass of the power system and the area of solar panels.

In conclusion, the two designs seem almost identical even if other properties like the average power, the peak power and the battery capacity are different between.

Table B.19: Electrical system

	Total mass / kg		Area of solar panels
	Electrical system	Power system	m^2
SWD Design	11.75	34.48	1.12
Circular Design	11.85	34.58	1.12

B.5 Operations and Communications, Aurelien Hugon

This appendix presents the author's mission summary. Two work packages are described: operations and communications. Two levels of study are presented.

The project summary presents two space weather warning mission concepts: the space weather diamond and the circular missions. Both constellations are illustrated in Figure B.10.

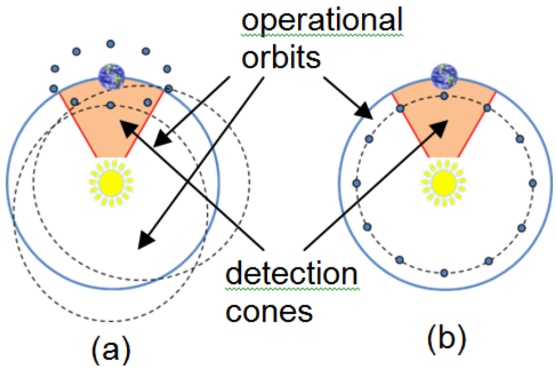


Figure B.10: Spacecraft constellations for (a) SWD and (b) Circular constellations.

B.5.1 Operations

Operational mechanism: how to forecast space weather?

It is important to remind how the warning mechanism occurs. The payload instruments generate data continuously in a region close to the Earth-Sun axis. This data is sent to Earth at the speed of light. Since the fastest CMEs travel only at 2600km/s, the difference of propagation speed allows the system to warn from the arrival of harmful particles.

Operational modes

The mission operational modes are defined in this paragraph.

- Operational Mode (OM): the spacecraft is travelling with the desired orientation (the shielded face pointing in the direction of the Sun), the battery is full and the solar panels have an optimum inclination with the sun vector. The payload instruments and the computer are working continuously. The OM mode is composed of two states:
  - Warning State (WS): The spacecraft is inside the detection cones and downlinking space weather warning data continuously.
  - Non-Warning State (NWS): the spacecraft is not in the WS mode. Each subsystem can be activated one after the other.
- Fully Operational Mode (FOM): the spacecraft is in OM mode and travelling on its operational orbit.
- Orbit Transfer Mode (OTM): The spacecraft is in the OTM mode when it receives a command to perform an orbit transfer manoeuvre. The spacecraft acquires an accurate orientation.
- Safety Mode (SM): The spacecraft turns into this mode when the battery reaches a certain level of discharge (or a serious anomaly is detected).
- Sun and Earth Tracking Mode (SETM): the spacecraft's battery charge reaches a predefined level.
- End of Life Mode (ELM): the spacecraft is in this mode if it is not possible to operate it to forecast space weather anymore, or for any other purpose.

SWD mission operations plan

Table B.20 describes the two mission operations plans.

Table B.20: Mission operations plans

Mission	Launch	Interplanetary transfer	Operational Orbit
SWD	- First launch date: 7th Jan 2021 - Launch vehicle: Soyuz - Five launches: ideally 36.5 days out of phase - 2 spacecraft per launch - Direct escape: 25min	- First orbit manoeuvre: 73.5 days after the launch -1 or 1.5 years to reach the spacecraft's operational orbit - Spacecraft expected operational	- Spacecraft expected fully operational for at least 10 years - 2h02min of warning for the quickest CMEs
Circular	- Latest date: 21st Dec 2016 - Launch vehicle: Ariane 5 - 12 spacecraft launched by the same launch	- First manoeuvres performed 163 days after the launch - 3.8 years after the launch, spacecraft expected in WS state - 5.8 years after the launch, constellation fully operational	- Phase expected fully operational for at least 7.2 years - 1h55min of space weather warning

B.5.2 Communications

This section lists the main constraints applied to the communications system, the solutions found to these concerns and the comparison of each design's performance.

Main constraints

- The solar conjunction which prevents from any communication downlink
- The cost: the large number of spacecraft potentially requires many ground stations

Ground stations network

Table B.21 recaps the ground stations selected, their location and their antennas diameter.

Table B.21: Ground stations selected

Location	Antennae Dish Size (diameter, m)	Location (Latitude, Longitude)
Kourou (French Guiana)	13, 13.5 or 15	+5°N, -52°E
Redu (Belgium)	13, 13.5 or 15	+50°N, +5°E
Perth (Australia)	13, 13.5 or 15	-31°N, +115°E
South Point (Hawaii)	>7	+19°N, -155°E

Theoretically, three ground stations are required to provide full sky coverage (24 hours coverage). However, since the cost has to be dramatically reduced, it has been decided to select existing ground stations from ESA's network instead of buying new ones.

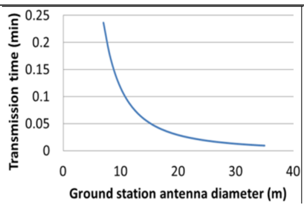
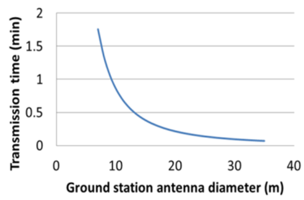
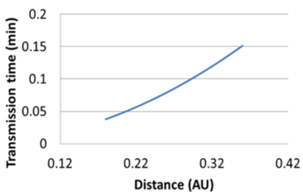
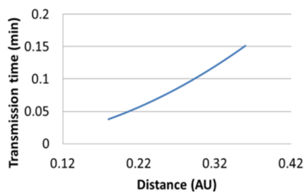
Links design

- Input parameter: message of 672 bits (contains payload and housekeeping data).

- Downlink: spacecraft situated at the furthest point inside the detection cones
- Circular crosslink: the distance separating the spacecraft is 0.44 AU.

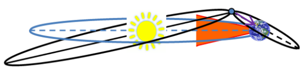
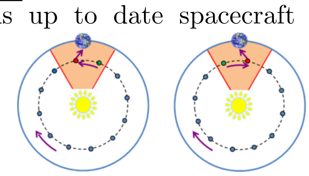
Downlink capabilities

Table B.22: High gain antennas transmission capabilities (without propagation time)

Downlink configuration	SWD	Circular
Housekeeping report (96 bits) to be transmitted from the longest distance from Earth (0.36 AU for SWD and 1.85 AU for the Circular)		
Payload report (320 bits) to be downlinked to a 12 m diameter ground antenna		

For SWD concept: downlinking using a patch antenna from the furthest place from Earth and a 12 m ground antenna takes around less than 3.2 min.

Subsystem comparison

SWD Orbit inclination	Circular Inter-satellite links
Principle: inclination of 1.7°on the orbit to allow the spacecraft to be outside of the interference cone at all times Advantages: less mass, less complex, longer warning time  Downlink: Reflector antenna aperture: 0.37m Pointing accuracy: 2.6° Warning time reduced by 3.24 min.	Principle: space weather warning data sent to a relay spacecraft before downlinking the payload report Advantages: Less propellant required, continuous up to date spacecraft in-  formation Downlink: Reflector antenna aperture: 0.7m Pointing accuracy: 1.4° Crosslink: Reflector antenna aperture: 0.34m Pointing accuracy: 0.23° Warning time reduced by 10.2 min.

B.5.3 Conclusion

The future work on this project would consist mostly in implementing inter-satellite link on the SWD concept (combining both solutions would be an asset), refining the link budget and the high gain antennas design, and probably increasing the overall system performance to downlink more information.

B.6 Payload and Ground Segment: Joshua Hull

The Payload and Ground Segment for the Sub L1 Space Weather Warning Systems are presented here.

The Sub L1 Space Weather Warning Systems Project has looked at the feasibility of implementing a warning system which can provide uninterrupted coverage on Space Weather Events which are headed for Earth. This warning system can be used by the space, airline and power industries to provide an early warning to protect their assets and save the industries millions in cost.

The Payload Work Package involved searching for the optimised solution for payload instruments in order to be able to observe the processes required to provide a space weather early warning system. Once this was completed, specifications of the subsystem had to be fed to the other subsystems to build a functional concept.

The Ground Segment Work Package, involved more research, this time on the location and specifications of the ground stations which make up ESA's ESTRACK Network, having ruled out NASA's corresponding network due to the requirement to keep as much of the infrastructure for the mission here in Europe as possible.

B.6.1 Payload Requirements

Having documented the types of space weather which would affect the local Earth environment and the infrastructure within it, the payload requirements were produced to reflect the characteristics observed.

From this the types of measurements required could be obtained. Both in-situ and remote sensing were viable options when it came to the types of instruments that could be used to measure the phenomena and their "interaction carrier particles" whether that be charged particles in the solar wind or photons in the EM spectrum.

B.6.2 Payload - First Iteration

Initially both in-situ and remote sensing instruments were considered for use to provide the SWW system. By having both types of instruments on board meant that the concept was a complete package. The ability to provide the data has no dependence on any other organisations which are partners in the project, therefore meaning there are fewer nodes where possible issues could arise in the dissemination of data.

The instruments which were discussed in this iteration were:

- Magnetometer
- Ion and Electron Spectrometer
- Vector Magnetograph
- X-ray Spectrometer
- Coronagraph

However looking at the requirements for each of these instrument types it quickly became clear that the instrument types were more suited to different nominal orbits. The magnetometer and the ion and electron spectrometer needed to be close to the Earth-Sun line to provide valuable data. On the other hand, the vector magnetograph and the x-ray spectrometer could be positioned either in the Earth-Sun line or away from the Earth-Sun line, in LEO for example. Finally the coronagraph would require a view point away from the Earth-Sun line due to the occulting disk required to block the solar disk.

These initial findings and calculations for the data rates, power and mass requirements quickly showed a system; which in this configuration; that would evolve into a complex engineering solution which wouldn't be able to meet the requirements for the mission. These factors included ballooning values for mass, power, data rates etc. which in turn meant that the feasibility of multiple identical spacecraft became non-existent.

Therefore, in the second iteration a different approach was taken.

B.6.3 Payload - Second Iteration

For the second iteration, it was assumed that the remote sensing data could be obtained from another operational mission, which was either specifically there for space weather detection purposes or a scientific mission which could provide the information required. This removed the clash in requirements the instruments in the first iteration had.

It meant that the payload subsystem would be substantially smaller than the proposed subsystem in the first iteration. This therefore reduces the requirements from the other subsystems, which in turn reduces their parameter values significantly.

Further research was conducted into other instruments which could be placed into the sub L1 position to enhance the data on the oncoming space weather. The instruments selected for further study were:

- Magnetometer
- Ion and Electron Spectrometer
- High Energy Particle Detector
- Particle Flux and Species Detector

These provided the required measurements to detect the magnetic fields and charged particles produced in the solar wind, solar flares, CMEs, CIRs and Parker Spirals.

Research into current instruments which could measure the energy ranges specified in the requirements, see (Wayman , 2011), has been completed.

The for the magnetometer, there were two possible options: a science grade fluxgate magnetometer or a space weather grade magnetoresistive magnetometer which were provided by ICL. Due to the need for keeping the mass, power etc. budgets as low as possible the magnetresistive magnetometer has been chosen due to the fact that it can detect the ranges of measurements required.

For the ion and electron spectrometer, one option was looked at, the charged particle spectrometer, produced by MSSL. The energy range which it covered was not ideally suited to the mission. However to provide an initial value for the subsystem, which in turn allowed the rest of the team to produce a workable solution, this instrument will surface. This instrument measures the charge, mass and kinetic energy of charged particles within the solar wind.

The High Energy Particle Detector also only had one option looked at. This instrument MuREM, produced by the SSC, differs from the ion and electron spectrometer in that it has a higher energy detection range and is facing not at the Sun but perpendicular to the Earth-Sun line into the plane of the ecliptic to observe the particles travelling in the parker spirals. Although these are moving radially from the Sun slower than the other phenomena, they carry significant energy due to the velocity the particles are travelling at.

The Particle Flux and Species Detector, with the HMRM detector the instrument looked at in more detail, has been produced by RAL and ICL. It measures the total radiation dose received by the spacecraft by measuring the flux and energy of the incoming particles.

Further iterations to get a more suitable energy detection range is required although the solution provided is a workable solution.

B.6.4 Ground Segment

The Ground Segment work involved researching the ground stations which were available for use. ESA's ESTRACK network has been chosen due to the requirement to keep as much of the infrastructure and operations within Europe, therefore NASA's equivalent network has been discarded.

ESA's ESTRACK Network is made up of three subsidiary networks: the Core Network which includes ESA's DSN, the Cooperative Network which enables ESA to use NASA, JAXA, etc. dishes in return for providing time for them on ESA's dishes. Finally there's the Augmented Network where ESA has to buy time on the dishes of companies such as the Swedish Space Corporation.

In order to satisfy the requirements of the communications subsystem, dishes with diameters greater than or equal to 7 metres are required. Therefore, part of the research required was to find out the diameter and number of dishes each ground station had. Another requirement was that the ground stations needed to be able to transmit in S-Band.

The ground stations were not restricted to higher latitudes due to the orbit parameters, therefore a greater range of sites which have dishes could be considered for the primary and back up ground stations.

These parameters were fed to the communications work package. Between this and the communications work packages, it has been shown that a minimum of five ground stations will be needed per day to provide continuous coverage.

The ground stations used in this analysis is for a worst case scenario where the larger dishes are unavailable. By showing that using 7 metre diameter dishes from the Augmented Network full coverage can be achieved provides flexibility when issues arise with other missions and the larger dishes are required, as it has been proven that the mission will not go off-line in these situations. This ability is a necessary measure if the Sub L1 Space Weather Warning Systems being proposed are to be successful.

B.7 Thermal Control System: Ander Iturri Torrea

B.7.1 Requirements

The requirements that the TCS are to maintain the temperature of the spacecraft in specified ranges guaranteeing the optimum performance of the subsystems. The requirements impose limitations on temperature range and temperature variation during the mission lifetime.

The critical requirements on each area can be found in the next table:

Table B.23: SWD Requirements Summary

Requirement	Component	Value
Temperature Range	MuREM, CHaPS, HRMR	10°C to 40°C
Temperature Gradients	Structural Element	<2°C/m
Temperature Stability	Electronic Elements	< 5°C/h
Lifetime	All	13 years of duration

B.7.2 Hot and Cold Cases Definition

The TCS design will require a mathematical analysis of the missions suggested. This analysis will be made for two critical situations: the hottest and the coldest cases. It is assumed that every other thermal situation will be included between those two cases. The cases will be defined for the two concepts suggested: the Space Weather Diamond concept, and the Circular concept. The SWD concept will require 10 spacecraft in an elliptical orbit about the Sun, whereas the Circular concept will require 12 spacecraft in a circular orbit around the Sun. The Earth influence is neglected when selecting these cases, as both concepts are far enough from it to make this assumption. The definition of these two cases can be found in the next table:

Table B.24: Hot and Cold Cases Summary

Concept	Hottest case	Coldest case
SWD	0.821 AU from the Sun at BOL	1.179 AU from the Sun at EOL
Circular	0.853 AU from the Sun at BOL	1 AU from the Sun at EOL

One of the faces of the spacecraft will be permanently pointing towards the Sun and the heat dissipated by the subsystems will be estimated as the power consumed. The analysis will consist of computing the steady-state heat equation on each face of the spacecraft:

$$\epsilon_i A_i F_{si} \sigma (T_i^4 - T_s^4) = \alpha_i A_i F_{si} q_s + Q_i + Q_{radij} + Q_{condij}$$

B.7.3 Materials Used

The thermal control will be possible with some specified materials that will be placed in the spacecraft. The most important material that will be used is going to be the MLI. Not only will it cover the spacecraft shells, but it will also cover the propellant tanks or the subsystems electronic boxes. The MLI used in both concepts will be composed of Teflon coated and backed, Aluminized Mylar, Nomex and Aluminized Nomex. Two versions of this MLI will be manufactured: one “wide” version (24 internal layers) for the face pointing towards the Sun and one “thin” version (10 internal layers) for the other components. Radiators will be used to dissipate heat and it will be made of Optical Solar Reflector. Two radiators will be placed in the faces where the subsystems will be allocated, the two lateral faces. Patch heaters will be used to maintain appropriate temperatures inside the thrusters and the big booster of the spacecraft. The materials used for both concepts can be summarized in the next table:

B.7.4 Results

The results of the analysis are obtained for the two critical cases considered and using a mathematical model. The results can be seen in the next table and list, ordered by requirements:

Table B.25: Materials Summary

Material	Properties	SWD	Circular
MLI	Teflon, Mylar, Nomex	10.22 m^2	5.78 m^2
	"wide" $\alpha = 0.0013, \epsilon = 0.0027$ "thin" $\alpha = 0.0035, \epsilon = 0.0150$	8.44 kg	4.71 kg
Radiators	OSR: $\alpha = 0.20$, $\epsilon = 0.80$	2 x 0.33 m^2 2.18 kg	2 x 0.33 m^2 2.05 kg
Heaters	Patch heaters: 9 pairs 8 in thrusters 1 in booster	19.20 W 0.38 kg	19.20 W 0.38 kg

- Temperature Range: (see Table B.26)
- Temperature Gradients: The maximum gradient within the spacecraft will be 4°C/m in SWD concept and 6°C/m in the Circular concept.
- Temperature Stability: No eclipse or sudden changes in the thermal environment is expected, so the quasi-stationary hypothesis is applied and the variation of temperature through time is very small.
- Lifetime: The analysis includes the degradation of the materials during the mission lifetime to ensure the temperature requirements are fulfilled in all that time.

Table B.26: Temperature Range Results Summary

Component	SWD concept	Circular concept
MuREM	12.6450°C to 36.7509°C	18.2210°C to 37.5332°C
CHaPS	12.6231°C to 36.7290°C	18.1991°C to 37.5113°C
HRMR	12.6137°C to 36.7196°C	18.1897°C to 37.5019°C

B.7.5 Conclusion

Both concepts do fulfil the requirement of maintain the temperature variation within some safe limits. The maximum temperature gradients for structural elements are too high, being higher than the demanded value. Temperature stability is ensured and the lifetime requirement is taken into account in the analysis.

The TCS of the SWD concept is heavier (11.00 kg) than the TCS of the Circular concept (7.14 kg). the obtained mass remains approximately in the range of values expected. However, the obtained power value is higher than expected for both concepts.

As a conclusive summary, the TCS designed for both the SWD concept and the Circular concept are active systems and they both consist of MLI (Teflon, Aluminized Mylar, Nomex and aluminized Nomex), radiators (OSR) and heaters (patch heaters)

B.8 Baseline Design: Julia Leeson

This Executive Summary provides information on sub L1 space weather warning systems designed by students enrolled in the MSc in Astronautics and Space Engineering at Cranfield University in 2011/12. The project was initially based on St. Cyr's Space Weather Diamond concept (St. Cyr et al., 2000). Two different baselines were investigated thoroughly and each was found to have its own merits. Both baselines have been designed to meet the same set of requirements by Wayman (2012).

B.8.1 Space Weather Diamond Concept Baseline

The Space Weather Diamond constellation provides quasi-continuous real-time space weather warning to earth through in situ detection. It provides two hours warning time for the end users. The system is able to detect all earth-bound Coronal Mass Ejections (CMEs) with an angular width greater than 14.61°. The constellation consists of ten identical spacecraft, each in an inclined elliptical heliocentric orbit with a radius of perihelion of 0.821 AU and a period of exactly one year. The spacecraft are launched in pairs, requiring five launches in 2021. Due to the complex launch windows, the system has a nine hour coverage gap per year if uncorrected. The system would be fully operational by 2023, enabling it to provide space weather warning for at least 11.1 years, a full solar cycle. Each spacecraft is three-axis stabilised, is powered by solar arrays with rechargeable batteries and has a liquid bipropellant propulsion system. The total loaded mass of each spacecraft is 699 kg. The spacecraft is illustrated in Figure B.11. A concise baseline summary with key information can be found in Table B.27.

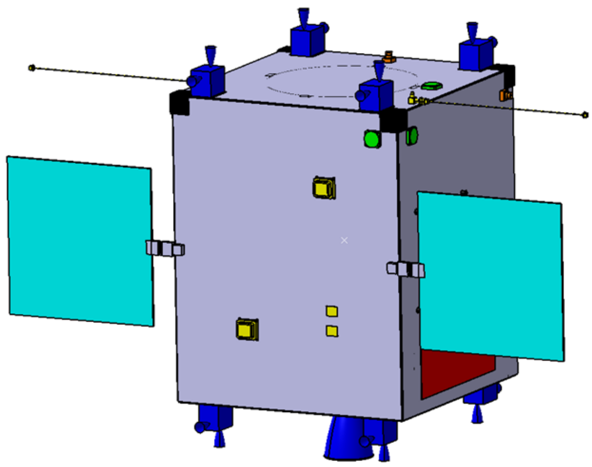


Figure B.11: SWD Spacecraft design (Maugein, 2012).

B.8.2 Circular Concept Baseline

The Circular baseline constellation provides continuous real-time space weather warning to earth through in situ detection. It provides two hours warning time for the end users. The system is able to detect all earth-bound Coronal Mass Ejections (CMEs) with an angular width greater than 30°. The constellation consists of 12 identical spacecraft, each in a circular heliocentric orbit with a semi-major axis of 0.853 AU and no inclination. The spacecraft are all launched together in December 2016. The system would be fully operational by 2023, enabling it to provide space weather warning for at least 7.2 years, covering the majority of a solar cycle. Each spacecraft is three-axis stabilised, is powered by solar arrays with rechargeable batteries and has a liquid bipropellant propulsion system. The total loaded mass of each spacecraft is 330 kg. The spacecraft is illustrated in Figure B.12. A detailed baseline summary can be found in Table B.28.

Table B.27: SWD Baseline Summary

Mission Lifetime	Thirteen years (operational for 11.1 years)
Number of Spacecraft	Ten
Operational Orbit	Elliptical heliocentric orbits with semi-major axis 1.000 AU and radius of perihelion 0.821 AU (0.37°inclination)
Deployment	Two spacecraft per launch. Each spacecraft does two or three Hohmann transfers to apoapsis in the operational orbit. Total deployment time of one year and 11 months
Max. Delta V	3714 m s^{-1}
Launch Date	Five Soyuz launches in the first half of 2021
Payload	Magnetometers, ion & electron spectrometer, high energy particle detector & particle flux species detector
AOCS	Three-axis stabilised, sun pointing
Power Subsystem	GaAs triple junction solar cells, deployed array area 1.12 m^2 & Li-ion batteries
Communications	HGA 0.37 m, 2.2 GHz
Propulsion	Astrium S400-12 bipropellant system (486.38 kg propellant)
Size	1.55 m x 1.25 m x 1.25 m
Thermal	Active (heater, radiator & MLI)
Redundancy	Replication, diverse & functional
Mass	698.85 kg (wet) & 202.27 kg (dry with 20% margin)
Average Power	171.96 W
Total Estimated Mission Cost	€340.7 million

B.8.3 Baseline Comparison

The two different baselines both had their own advantages and disadvantages. As the requirements for both designs were the same, the key system drivers were similar for both designs. These included:

- System warning time;
- Detection cone angle;
- Number of spacecraft;
- Operational orbit;
- Deployment method;
- AOCS control method; and,
- Whether inter-satellite links were required or not.

For the SWD concept, the delta V required for each spacecraft and the number of launches were key drivers of the deployment method and similarly for the Circular concept, the deployment method was driven by the total deployment time.

Both designs were also compared with St. Cyr's original Space Weather Diamond concept. The SWD and Circular concepts both provided an improvement in warning time over St. Cyr.

The Circular concept was identified as the better system due to the simpler deployment method, however the SWD baseline provided a better warning system.

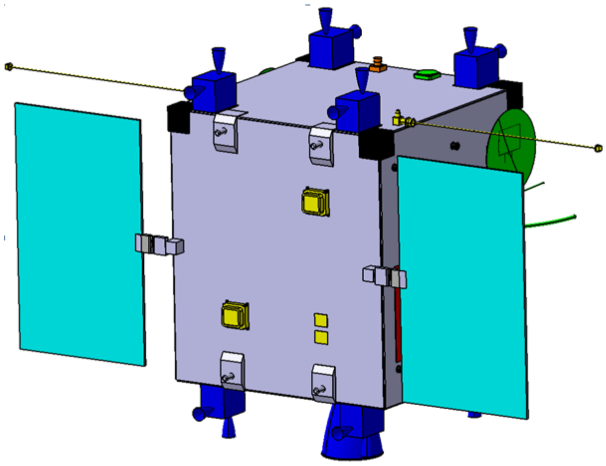


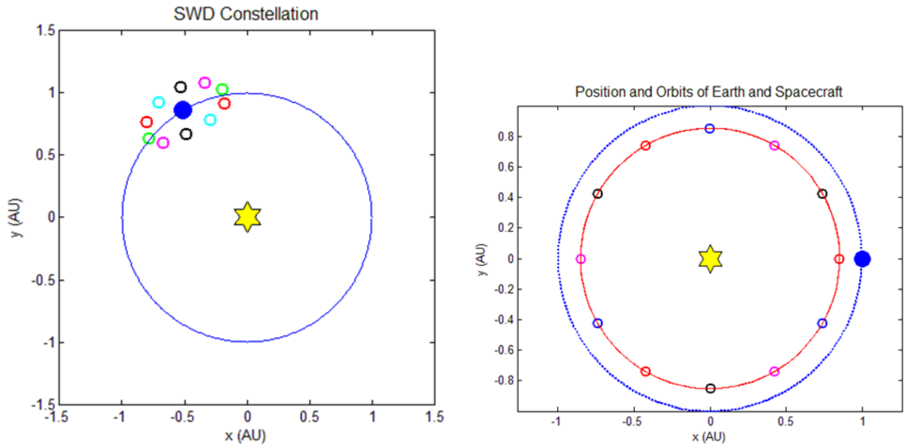
Figure B.12: Circular Spacecraft design (Maugein, 2012).

Table B.28: Circular Baseline Summary

Mission Lifetime	Thirteen years (operational for 7.2 years)
Number of Spacecraft	Twelve
Operational Orbit	Circular heliocentric orbit with semi-major axis of 0.853 AU (not inclined)
Deployment	Twelve spacecraft in one launch, 5.8 years deployment time
Max. Delta V	1636 m s^{-1}
Launch Date	One Ariane 5 launch on 21 Dec 16
Payload	Magnetometers, ion & electron spectrometer, high energy particle detector & particle flux species detector
AOCS	Three-axis stabilised, sun pointing
Power Subsystem	GaAs triple junction solar cells, deployed array area 1.12 m^2 & Li-ion batteries
Communications	HGA 0.70 m, 2.2 GHz, Ka-band inter-satellite links
Propulsion	Astrium S400-12 bipropellant system (139.20 kg propellant)
Size	1 m x 1 m x 1.1 m
Thermal	Active (heater, radiator & MLI)
Redundancy	Replication, diverse & functional
Mass	329.56 kg (wet) & 190.36 kg (dry with 20% margin)
Operational Power	188.49 W
Total Estimated Mission Cost	€240.8 million

B.9 Configuration: Florian Maugein

The aim of the project was to develop a space weather warning system. The project was based on the Space Weather Diamond paper published in 1999 by NASA (O.C. St. Cyr, 1999). Two concepts have actually been studied: the Space Weather Diamond concept and the Circular concept.



(a) SWD (b) Circular

Figure B.13: Illustration of the SWD and the Circular concept orbits (AUNEAU, 2012).

B.9.1 Space Weather Diamond concept

The Space Weather Diamond concept is a constellation of ten identical spacecraft. Each spacecraft is a 700 kg cuboid spacecraft with the dimensions 1.25 m x 1.25 m x 1.55 m. Each spacecraft is orbiting on its own heliocentric orbit and has a relative orbit around the Earth. The spacecraft is sometimes between the Earth and the Sun, sometimes behind the Earth.

Configuration

Because of payload's requirements, each spacecraft of the constellation must have a face constantly pointing at the Sun. The face opposite to the Sun-pointing face supports a high gain antenna, with a conceptual mechanism design (ARUNDAL, 2012).

Two solar panels are mounted on the sides of the spacecraft. A total area of 1.12 m^2 is provided, and a mechanism allows the panels to rotate in order to create a cosine loss. Each spacecraft requires 16 thrusters. They are mounted, by clusters of two, on each corner of the spacecraft. A main thruster is mounted on the bottom face of the spacecraft. The main thruster is aligned with the two propellant tanks required. Each tank is 229 L. They are mounted one at the top of the other to avoid the centre of gravity from shifting in another axis than the axis of the spacecraft during the burns. The two propellant tanks are supported by an inner thrust tube. A pressurant tank is also required. It is mounted outside of the thrust tube.

The inside components are mounted on two faces to facilitate the thermal subsystem. They are mounted on the side walls of the spacecraft, and at the bottom to avoid bending moments caused by lateral loads at launch. There are power components with two batteries, payloads' electronics, communication components, AOCS components with four reaction wheels and a computer and data recorder.

Stacking

The Space Weather Diamond deployment method allows having two spacecraft per launcher. The launcher used for this concept is a Soyuz. The two spacecraft are stacked one next to the other, using a platform to increase the diameter of the launcher interface.

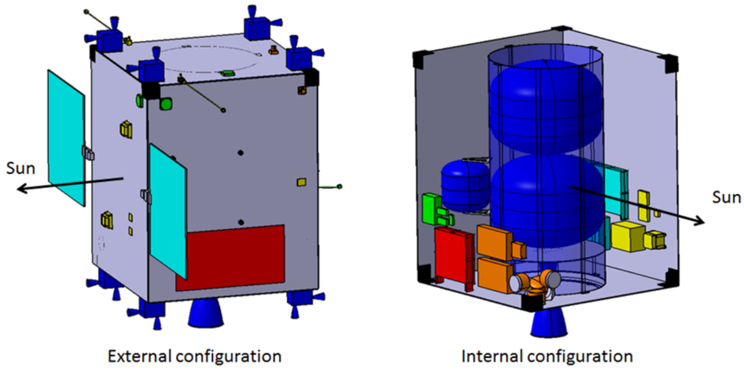


Figure B.14: Space Weather Diamond configuration.

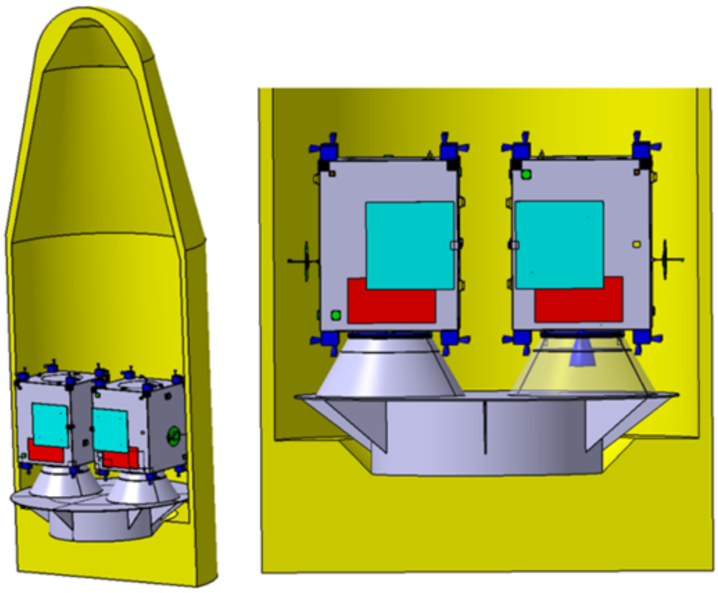


Figure B.15: Space Weather Diamond Stacking.

B.9.2 Circular concept

The Circular concept is a constellation of twelve identical spacecraft. Each spacecraft is a 329 kg cuboid spacecraft with the dimensions 1 m x 1 m x 1.1 m. Each spacecraft is orbiting on the same heliocentric orbit, smaller than the Earth's orbit.

Configuration

Because the components are roughly the same as for the Space Weather Diamond concept, the configuration is similar. The area of solar array required is the same as for the SWD concept. The two propellant tanks are smaller: 64 L each. The main difference is the presence of inter-satellite links. More components are required such as two antennas, mounted on the side of the spacecraft to allow a communication between the spacecraft of the constellation.

Stacking

The deployment for the Circular concept is much easier than for the SWD. The twelve spacecraft of the constellation can be stacked into one launcher, an Ariane 5. A dispenser has been used, similar to one used in the Globalstar mission where six spacecraft have been launched. A central tube supports some fittings (in red). Each spacecraft is fixed in four points to the fittings.

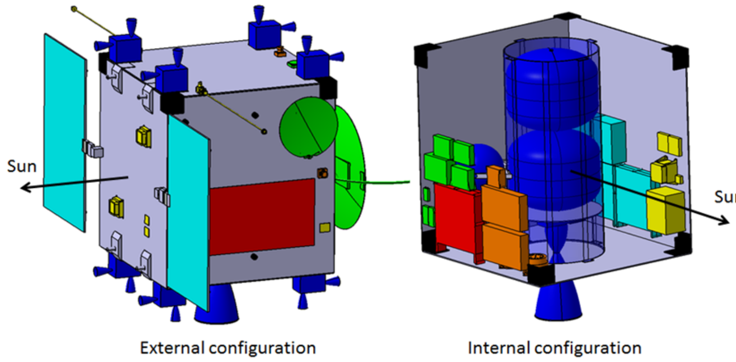


Figure B.16: Circular concept configuration.

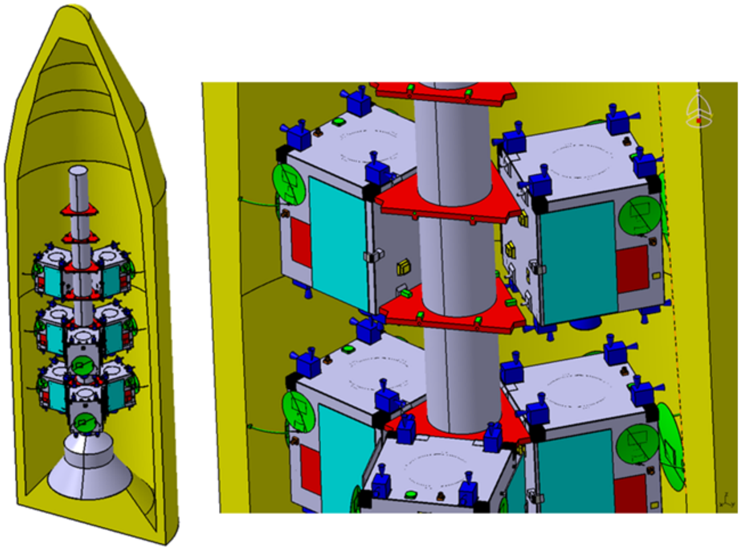


Figure B.17: Circular concept stacking.

B.9.3 Conclusion

The configuration is similar for both concepts. The main differences are the size of each spacecraft and the presence of inter-satellite links in the Circular concept. However, the way of stacking is completely different. The possibility to stack all the spacecraft into one launcher is a great advantage for the Circular design in terms of simplicity of deployment, risks and cost. The spacecraft of the Circular concept are much smaller: their wet mass is less than the half of the SWD concept's wet mass. Even if some components are added for inter-satellite links, the spacecraft of the Circular concept are more efficient in terms of configuration. The Circular concept is also better from a stacking point of view. But the choice of a concept is not driven by the configuration of the spacecraft. Many other parameters must be taken into account such as the feasibility of the whole mission, the cost or the requirements met.

B.10 Launchers: Alan McLarney

The assignment looked at two methods of creating a space weather warning system; each had their own method of deployment that would affect the launcher choice and associated campaign.

The study looked at two methods to create a warning constellation. The first (Space Weather Diamond) was based on a NASA paper from 2000 by St. Cyr et al. and the other was a simpler heliocentric orbit concept.

B.10.1 Space Weather Diamond - SWD

The SWD deployment method - 2 spacecraft were launched together on directly injected escape trajectory into heliocentric orbit (as shown by the red line as shown in fig B.18) with a V_∞ of -717.92 m/s. This launch was required to be repeated every 36.5 days for 5 launches (10 spacecraft in total to create the constellation) in its entirety.

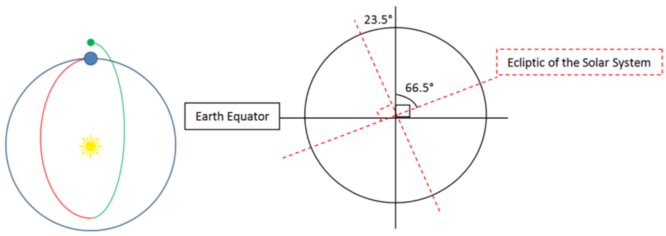


Figure B.18: SWD deployment.

To accomplish this an azimuth launch of 66.5°is needed from launch sites with a latitude of less than +/- 23.5°(the angle of the Earth versus the ecliptic) to launch directly onto the ecliptic. The European launch site at Kourou (French Guiana) was chosen as the primary launch site and the proposed Omelek Island site in the central Pacific as the secondary (as the only other real choice available). Both of these sites either offered or had no obvious reasons why the required launch azimuth could not be met.

The spacecraft design general specifications for SWD are as follows:

Wet mass = 700.92 kg - Dimensions = 1.25m x1.25m x1.55m (see Fig. B.19). Total Launch mass = 1615 kg

- The primary launcher chosen for SWD was the Soyuz ST from Kourou. It offered 2150 kg capacity to V_∞, > 700 kg excess capacity.
- The secondary launcher chosen for SWD was the Falcon 9 from Omelek Island. It offered 2500 kg capacity to the V_∞ required

The constellation has to be launched no later than 2021 to meet the operational deadline of 2023.

As can be seen in Fig. B.20 the ideal dates (that being the dates 36.5 days apart) as required by the orbital work package does not exactly line up with the launch windows available from either Kourou or Omelek Island. If launch dates of closest fit were taken from just Kourou or a mix of Kourou and Omelek then there would be a 9 hour or 8.5 hour “blackout” gap respectively from bunching up of the spacecraft. This would not meet the requirement of constant coverage and would need to be corrected.

Problems

From the launcher perspective the SWD offers several problems in its implementation:

1. SWD required 5x launches with very specific launch windows (to the minute). This would be in practice extremely difficult if at all possible to accomplish due to human factors and unforeseen events.

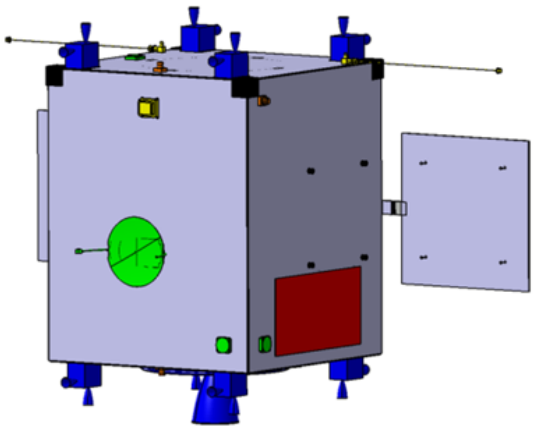


Figure B.19: SWD spacecraft configuration.

Ideal date (GMT) – 36.5 days apart	Kourou – GMT	Omelek Is. – GMT
7th Jan @ 04:28:20	7th Jan @ 21:11:28	7th Jan @ 07:01:56
12th Feb @ 10:28:20	12th Feb @ 18:49:56	12th Feb @ 04:39:20
20th Mar @ 16:28:20	20th Mar @ 16:28:20	21st Mar @ 02:14:16
25th Apr @ 22:28:20	25th Apr @ 14:03:20	25th Apr @ 23:52:20
1st Jun @ 04:28:20	1st Jun @ 11:39:20	31st May @ 21:32:20
7th Jul @ 10:28:20	7th Jul @ 09:16:20	7th Jul @ 19:06:20
11th Aug @ 16:28:20	11th Aug @ 07:00:20	11th Aug @ 16:47:20
16th Sep @ 22:28:20	17th Sept @ 04:33:20	16th Sept @ 14:26:20
22nd Oct @ 04:28:20	22nd Oct @ 02:13:20	22nd Oct @ 12:03:20
27th Nov @ 16:28:20	27th Nov @ 23:50:20	27th Nov @ 09:44:20
2nd Jan @ 22:28:20	2nd Jan @ 21:29:20	3rd Jan @ 07:17:20

Figure B.20: Ideal launch times and opportunities at chosen launch sites

2. The launch site at Omelek Island is only a proposed site - both it and the Falcon 9 are both unproven as suitable and thus present risk to the project.

3. The launch windows do not line up exactly with the required 36.5 day sequence. This would leave a minimum 8.5 hour blackout in the system as the constellation phasing would be bunched up - total coverage was expressly stated in the requirements as needed

Conclusion

The SWD approach from the launcher perspective is too complex and impractical to be realised. The second approach, the Circular Concept (CC), is a response to this impracticality and has been designed to be more desirable for the launcher work package.

B.10.2 Circular Concept - CC

The Circular Concept (CC) is designed to create a detection "barrier" between the Sun and the Earth at all times. The constellation of 12 spacecraft is designed to be launched on one launcher and deployed in space with the final configuration as seen in Fig B.21.

As we only have one launch the most efficient time can be chosen; the deployment of the CC design takes almost 6 years and so the launch year needs to be no later than the commencement of 2017. Thus the winter solstice of 21st December 2016 is the most ideal date for launch and can be accomplished at Earth equatorial azimuth to a V_∞ of 1206.04 m/s.

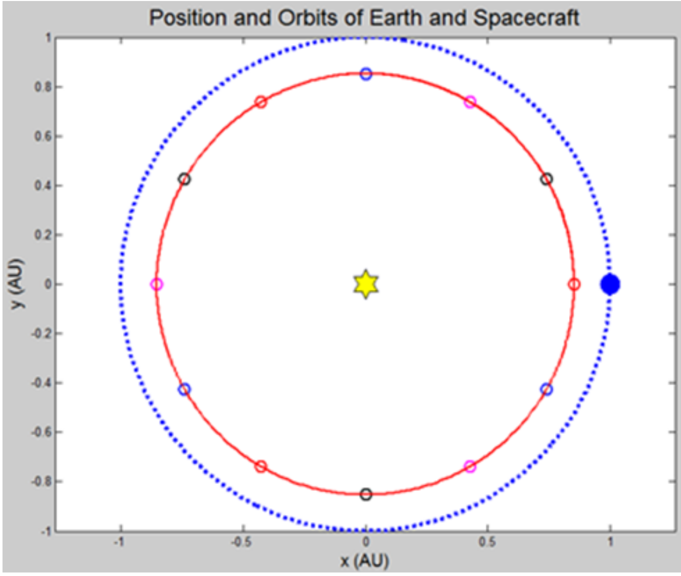


Figure B.21: CC orbits

The CC spacecraft is to have the following general specifications: Wet mass: 321.84kg - Dimensions: 1m x1mx 1.1m (see Fig. B.22). Total launch mass of constellation = 4455 kg

- The primary launcher chosen is the Ariane 5 from Kourou. This launcher offers 5500 kg capacity to lift the 4455 kg load.

- The secondary launcher chosen is the Delta 4 M+(4,2) from Florida. This launcher offers a capacity of 4500 kg.

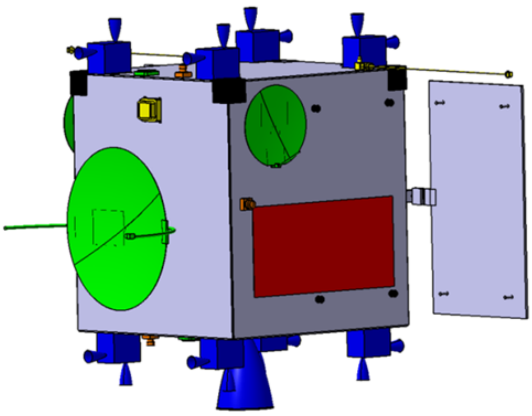


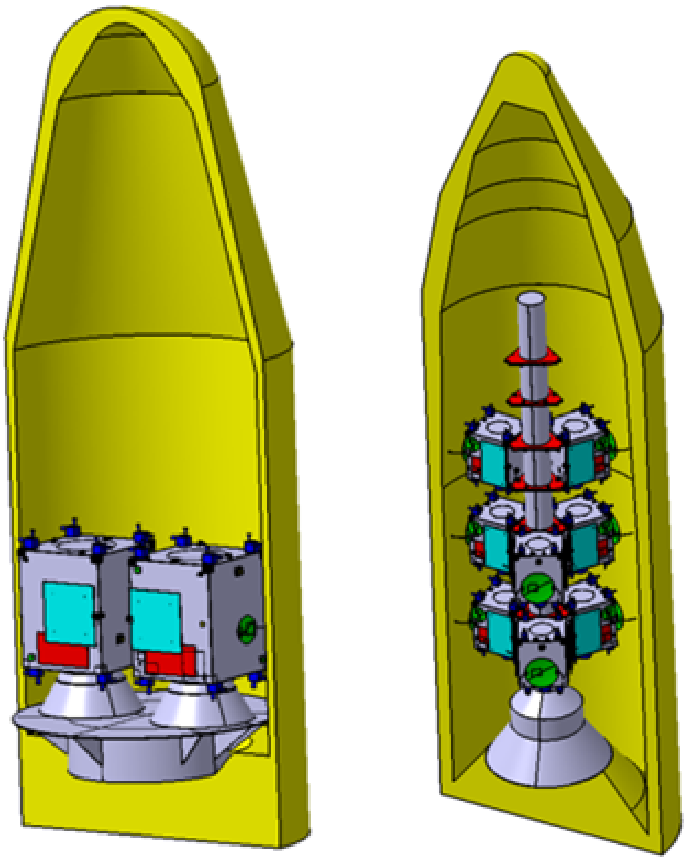
Figure B.22: CC spacecraft configuration

Problems

From the launcher perspective there are minimal problems with the CC design when compared with the SWD. However: the launch capacity of the Delta 4 is approaching the limit of the design. It is recommended that the Delta 4 M+(5,4) variant be considered if the M+(4,2) model is found to be insufficient.

Conclusion

Data and input has been given out to many other work packages based upon the information given here. It is acknowledged that the SWD method does offer better services in some ways over the CC method when taking the system as a whole; however unless a better method is devised to deploy the SWD constellation the simpler approach of the CC constellation is the recommended method at this time.



(a) SWD spacecraft in the Soyuz ST fairing

(b) CC constellation in the Ariane 5 fairing

Figure B.23: Satellites stowed in the launcher fairing.

B.11 Software Architecture and On-Board Data Handling: Robert Meeks

In this report two work packages were designed, the Software Architecture and On-Board Data Handling. Although there were slight differences for each work package between the two mission architectures (Space Weather Diamond and Circular) that were designed in this project, most of the key features are the same. The only distinct difference between the architectures is the number of states of operation there are in the Fully Operational Mode (FOM).

B.11.1 Software Architecture

The software architecture portion of the report begins with an initial estimation of the complexity of the system. This had two purposes. The first was so that initial size, mass and power requirements could be estimated so that these parameters could be used by other subsystems to use for their own calculations. The second purpose was to gain more of an idea of the type of architecture that may be required for each of the mission architectures. The complexity of the system was found to be at the simple end of the typical range in a table used from Wertz and Larson (1999). This along with parameters such as robustness redundancy and system processing ability for hardware and software were used to conduct a trade-off. The trade-off concluded that the Star or Centralised architecture was most suited to the mission being designed.

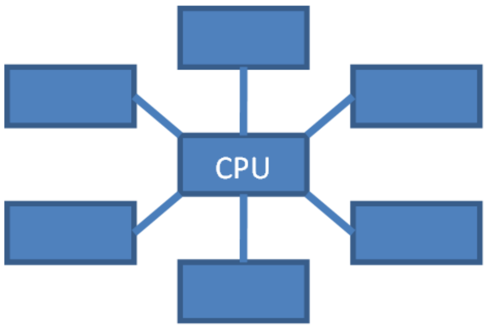


Figure B.24: Star Architecture.

With the engineer in charge of the operations work package operational modes were agreed. The Fully Operational Mode then had warning states defined. For Space Weather Diamond (SWD) this meant having two states, Warning and Non Warning. Circular required more because of having inter-satellite links which overcame an interference cone at the centre of the detection cone. How the spacecraft would switch between states for both mission architectures has been discussed. Figures B.25 and B.26 show the mode/state diagrams for each of the architectures.

B.11.2 On-Board Data Handling

The design of the OBDH system focuses on the data flow from payload to communication array. The selection of appropriate off the shelf hardware has been made according to the results calculated.

The primary objective is to send high integrity warning data back to Earth so that appropriate measures can be taken to weather an approaching CME. The secondary objective is to provide scientific data so that CME's may be better understood. It was realised early on that for both architectures a workable design that would satisfy the primary objective would not necessarily satisfy the secondary objective fully. The final design reflects this; the report that contains the data on the CME (the payload report) is an average and standard deviation of data collected from each of the four different payloads over one minute. It is believed that this would provide enough information about the CME.

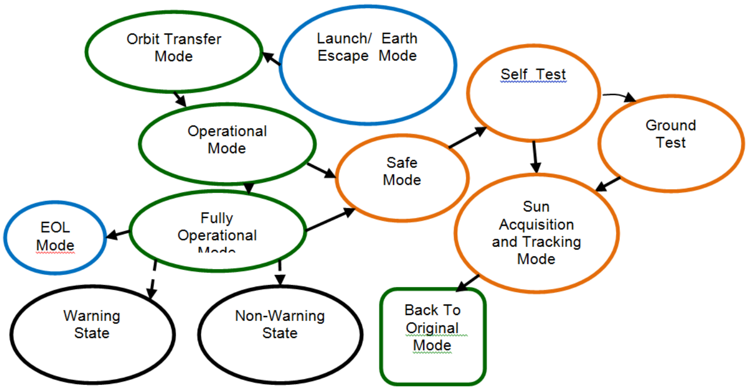


Figure B.25: SWD Mode/State Diagram.

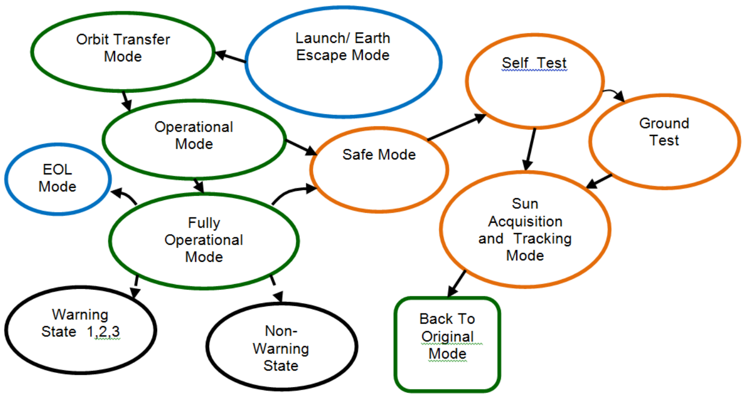


Figure B.26: Circular Mode/State Diagram.

Industry standards were used and researched to better understand the components that would be expected in data structure. Although only some of the main components were used it is thought that the design of the report structure is adequate for this level of design (CCSDS, 1992).

For integrity the data sent in each payload report is sent with a category rating. The OBDH categorizes the data collected according to the intensity of the CME. This can then be calculated again at the ground station to check. The payload is also sent twice as a default to further add integrity.

The housekeeping report has been made as small as possible, only providing an indication of nominal or otherwise operation of all the subsystems. The rest of the heath monitoring is processed on-board.

The requirements of the computer were also calculated in units of Instruction Per Second or IPS. The method for calculating this was found in Wertz and Larson (1999). The calculated value for the processor was for SWD 3.93 MIPS and 3.89 MIPS for Circular.

The selected hardware, the Radiation Tolerant Flight Computer (SSTL, 2009) from SSTL was selected apart from having the required processing power it was also chosen because it had acceptable:

- Radiation tolerance
- Operational life
- Power requirements
- Mass

There was also a need for storing data to make sure the data reached Earth without error. There may also be a reason later for storing data for science purposes. For these reasons a data

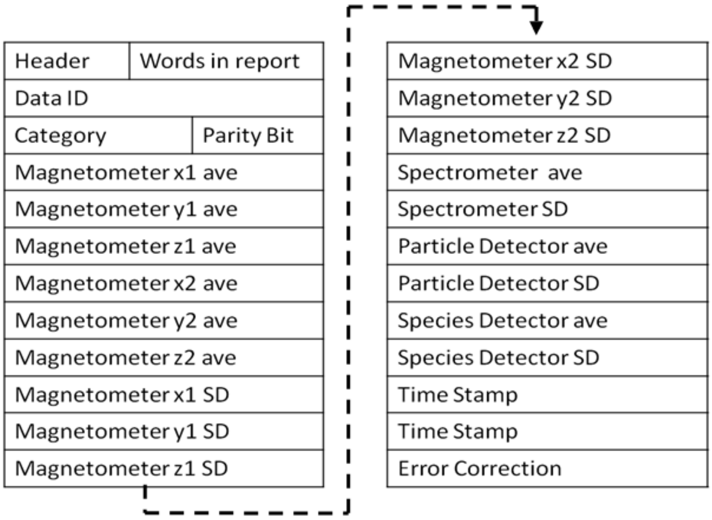


Figure B.27: Payload Report.

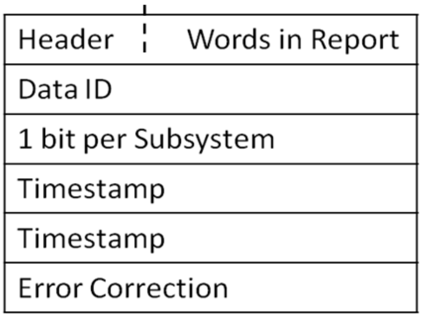


Figure B.28: Housekeeping Report.

recorder was added to the system (SSTL, 2003). It again is a product of SSTL and has 16 GB of storage. This will provide more than enough storage and make the system more robust because the computer will be able to back itself up on to the data recorder.

B.12 Orbital Perturbations and Propulsion: Andrew Sheppard

The goal was to design a space weather warning system that would improve upon the current L1 solar monitoring satellites. A space weather constellation design from NASA, the space diamond, was used as an initial baseline (St. Cyr et al., 2000). Two designs were chosen for further study and design iterations, the space weather diamond (SWD) with ten satellites in eccentric, heliocentric orbits, and the circular concept with twelve satellites in a circular, heliocentric orbit. This study is concerned with the effects of perturbations on the nominal orbits of the two models as well as the sizing of the propulsion subsystem.

B.12.1 Perturbations

As both mission concepts have a long operational life, in carefully phased constellations, the perturbations on the nominal orbit of each satellite have to be analysed and potentially corrected for.

Method

Satellite Tool Kit (STK) was used to model the constellations. STK numerically integrates the motion of a satellite over a desired time frame. A simulation campaign was followed to determine the effects of solar radiation pressure and third-body interactions on the satellites.

SWD Results

The perturbation of Venus was the only noticeable interaction with the SWD constellation, causing the satellite's orbits to deviate over the lifetime of the mission, shown in Figure B.29. It did not affect the nominal orbit in the satellite's operational zone and so was left as an inherent perturbation in the system as correction would be costly in terms of fuel.

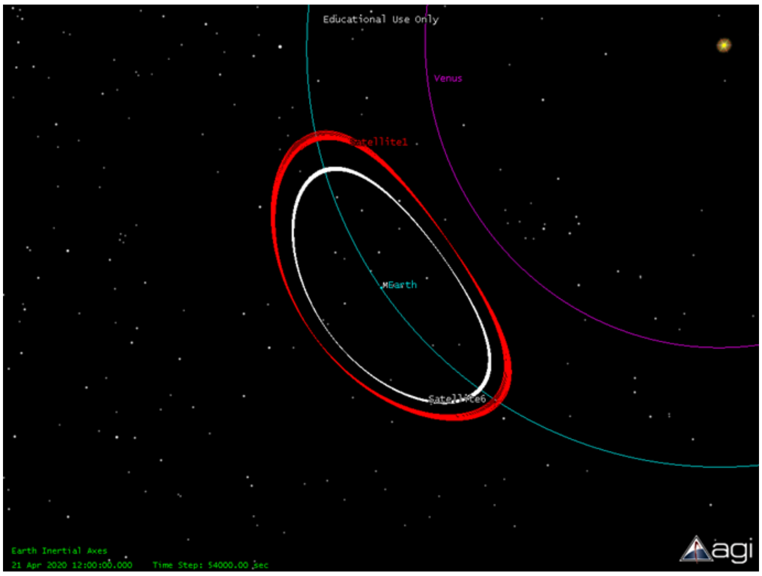


Figure B.29: The effect of Venus' perturbation on two satellites in the SWD.

Circular Results

The circular concept appears unperturbed by solar radiation pressure and third-body interactions, this is thought to be due to the symmetry of its circular orbit with that of Venus and Earth. Neither concept includes perturbations which cause deviation from the nominal orbit such that the top-level requirements of the system are not met, and it was decided not to include orbital corrections.

B.12.2 Propulsion

The propulsion subsystem was driven in its design by the deployment method from the orbits work package.

Type

As multiple manoeuvres are required for deployment the use of solid motors was not appropriate. The careful phasing of the satellites in their intermediate trajectories also excluded the use of electric propulsion. The high delta Vs necessary for orbital insertion required a liquid chemical bipropellant system.

Engine Selection

The highest performing European engine, Astrium's S400-15 was too long to fit our launch adapter and so a concession was made and the Astrium S400-12 engine chosen. This resulted in a mass increase of the order of a few kilograms. This engine uses N_2O_4 as the oxidiser and MMH as the fuel in a ratio of 1.65:1. Its specific impulse is 3120 m s^{-1} and it has a thrust of 420 N.

Sizing and Configuration

The propellant tanks are based on those designed for the Astrium Hotbird communications satellites. They have ellipsoidal end-sections and cylindrical mid-sections in order to bring down the height of the system. Table B.29 shows the propellant tank characteristics for both concepts.

Table B.29: Subsystem sizing for both concepts

Propellant Tank Characteristic	SWD	Circular
Diameter	0.753 m	0.480 m
Height	0.62 m	0.42 m
Mass	8.53 kg	5.31 kg
PMD	Bubble trap	Bubble trap
Tank Capacity (each)	222 L	62.3 L
Dry Mass	202.3 kg	190.4 kg
Propellant Mass	486.4 kg +10.2 kg RCS	134.5 kg + 4.72 kg RCS
Component Mass	13.74 kg	13.7 4kg
Subsystem Mass	38.97 kg	28.09 kg

There are two tanks in each system with an in-line configuration. A CATIA model representing the propulsion module for the SWD satellites is shown in figure B.30.

The propulsion module for the SWD became a mission driver as it determined the structural size and mass of the satellites, bringing us close to the maximum launcher capacity. To reduce the scale of the system redundancy had to be removed in terms of components, rendering it less reliable. In direct contrast to this the circular concept had a modest propulsion system and all twelve satellites could easily fit inside the Ariane V launcher. In future iterations a separate propulsion schematic could be drawn up for the circular design with increased redundancy in the propulsion system, making it even more robust and increasing its advantages over the SWD design.

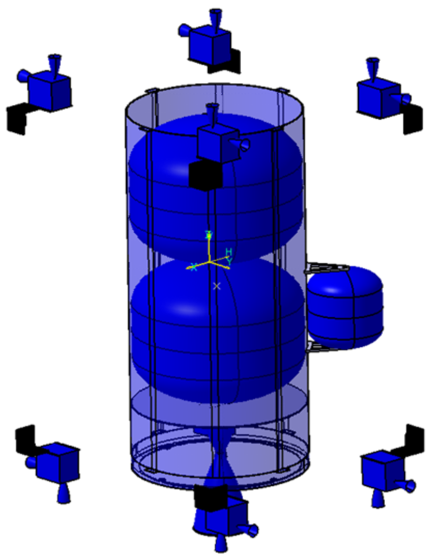


Figure B.30: A CATIA representation of the propulsion module for the SWD satellites (Maugein, 2012).

B.13 Orbits and Deployment of the Spacecraft: Susana Soto Carlavilla

As part of the MSc in Astronautics & Space Engineering at Cranfield University, a Group Design Project was developed by a team of 15 students. The objective of this project was to create a feasible baseline for a space weather warning system based on the Space Weather Diamond (SWD) concept proposed by St. Cyr in 1999. This document explains the deployment method proposed for each mission concept.

B.13.1 Space Weather Diamond Concept

This concept consists on a constellation of 10 spacecraft in identical heliocentric orbits with the same period as the Earth's orbit but different eccentricity. The system would provide almost continuous 2 hours warning time during 11.1 years and it will be capable of detecting CMEs of a width greater than 14.6°. The spacecraft are identical, with a total mass of 699 kg each and the same payload. The payload consists in two magnetometers, one electron and ion spectrometer, one particle flux and species detector and one high energy particle detector. In order to fulfill the requirements of the warning time, there was stated the constraint that the spacecraft should not be at a distance from the Sun greater than 0.86967 A.U. According to the requirements and in collaboration with the constellation subsystem, the operational orbits parameters were calculated and are presented in the following table:

Table B.30: Operational orbit parameters for the SWD concept

OPERATIONAL ORBIT	PARAMETERS
Semi-major axis	1 AU
Eccentricity	0.179
Inclination	0.37°
Argument of perigee	90°
Right Ascension Of Ascending Nodes	Every 36°

Deployment of the constellation

The specific requirements for the deployment of the spacecraft include the deployment of a constellation of ten spacecraft in their respective orbits with the lower number of launches as possible, optimising at the same time the deployment time of the whole constellation and the delta V that should be provided by each spacecraft. Apart from that, the simplicity of the deployment method should be maintained in the design process.

Different possibilities were studied, as utilising a parking orbit around the Earth. In this case, the delta V that should proportionate each spacecraft for escaping from the gravitational field of the Earth is too high (more than 3km/sec). Another possibility studied was the use of a heliocentric parking orbit, but a common one for every spacecraft was not found. Any attempt of launching more than two spacecraft in the same launcher lead to trajectories with path angles changes, which is expensive in terms of delta V.

Finally, the only feasible method found for deploying the constellation consists on performing 5 launches every 36.5 days with two spacecraft per launch. For this method, it is assumed that the launches could be performed at any time during the year and that the launch vehicle could be launched at any inclination to get the desired trajectory. Besides, the Earth's orbit has been assumed circular for the calculations

The Soyuz is the launch vehicle chosen by the launch subsystem. The launch vehicle would inject the two spacecraft straight into a heliocentric transfer orbit, outside of the gravitational field of the Earth, with the help of an upper stage. The trajectories followed by each spacecraft are represented in the next figure and are Hohmann transfers, as they require less energy than other transfer trajectories.

Both spacecraft would be released from the Fregat in the first transfer orbit (TO1). Once they arrive at the periapsis of this orbit, a different burn would be applied to each of them, so

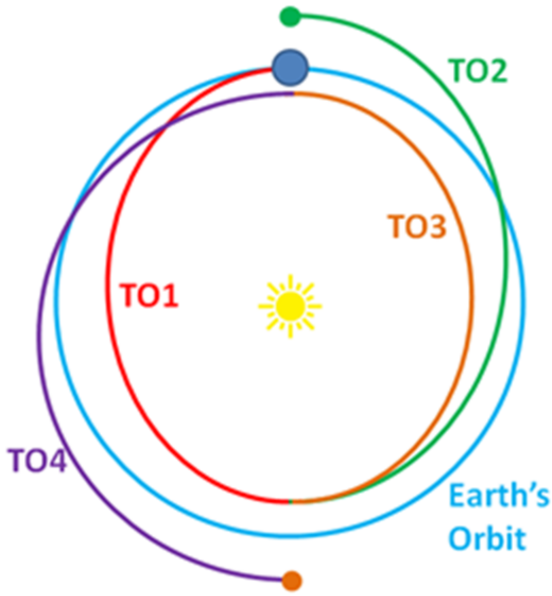


Figure B.31: Deployment trajectory for the SWD concept.

they will follow different trajectories. The first spacecraft will cover two transfer orbits (TO1 and TO2) in total, by performing two burns, arriving at its final orbit one year after the launch. On the other hand, the second spacecraft will cover three transfer orbits (TO1, TO3 and TO4) in one year and a half, performing three burns. Both of them will arrive at the apoapsis of their operational orbits. An extra burn is done by each spacecraft for changing the inclination, as they are launched into the ecliptic plane and the operational orbit has an inclination of 0.37 degrees for communication purposes. The change of inclination must be performed when the ecliptic plane and the plane containing the operational orbit are crossing. For that reason and doing a study of the different possibilities, it is decided to perform this extra burn in the first transfer orbit when the true anomaly is 90°.

The results for this method are presented in the table A.2. The second spacecraft requires a higher delta V than the first one, but both spacecraft have the same design for reducing the complexity of the mission baseline. The necessity of 5 launches and the high delta V required by each spacecraft raise the cost of the mission. For this and other reasons, the deployment of the spacecraft became a key factor. As a more efficient deployment method was not found for the SWD concept, it was necessary to study another constellation whose deployment was simpler.

B.13.2 Circular Concept

The circular concept is formed by 12 identical spacecraft with the same equipment and payload that the spacecraft of the previous concept. In this case, the total mass of each spacecraft will be 330 kg. The spacecraft will be all put in the same circular orbit around the Sun with a radius of 0.853 AU equally separated between them. The mission will be fully operational by 2023 for 7.2 years. In this case, the warning time will be of at least two hours as well but the CMEs will be detected for widths greater than 20 degrees.

Deployment of the constellation

The requirements are the same as the ones exposed in the section A.1.1 but in this case the spacecraft must be deployed in a circular constellation with a radius of 0.85297 AU. For this deployment method, every spacecraft will be put in a dispenser so all of them can be launched in the same launch vehicle, which in this case will be the Arian 5. The upper stage of the launcher will put the dispenser straight into a heliocentric transfer orbit. Once the dispenser has escaped from the gravitational field of the Earth, the spacecraft are realised, travelling one behind the other. This transfer orbit is designed by the next constraints; its apohelion must match with the Earth orbit and its radius of periapsis must coincide with the radius of the operational orbit.

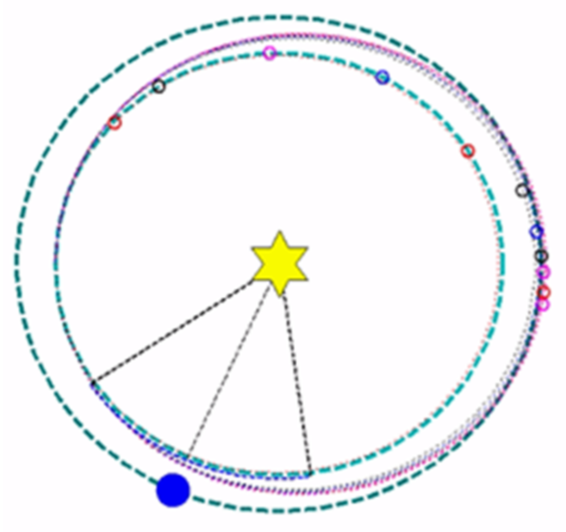


Figure B.32: Deployment trajectory for the Circular concept (from Coutinho, 2012).

Once the spacecraft arrive to the periapsis of the transfer orbit, the first one will apply a burn to get straight into the operational orbit. The others would apply different burn in that point in order to follow different phasing trajectories until the operational orbit, so they will arrive at this last one at the right moment. Different phasing orbits have been designed for each spacecraft by changing the size of the orbit and the number of rounds that the spacecraft must turn around them, so the delta V and the deployment time have been optimized as much as possible. Even if each spacecraft need different Delta Vs, they have been all designed identically to provide the highest delta V required for simplifying the design, as before. In this case, the total mass that can be lifted by the launch vehicle restricts the total delta V provided by each spacecraft to 1.8 km/s. For that value, the deployment time needed is 5 years and 10 months. In order to use the most efficient trajectories, the ones that require the less delta V possible for that deployment time have been chosen, obtaining the results presented in the table A.2.

B.13.3 Conclusions and Further Work

The next table contains the escape velocities needed in each case, the maximum delta V that must be provided for the spacecraft for each concept with a 5% of margin, as well as the deployment time.

Table B.31: Comparison of the escape velocity and delta Vs needed for each concept

Mission Concept	SWD	CIRCULAR
V infinity (m/s)	-717.92	-1212.07
Highest delta V (m/s)	3714.76	1635.68
Total deployment time	1 year & 11 months	5 years & 10 months

For the SWD concept, the delta V that must be performed by each spacecraft is higher than for the circular concept. Besides, 5 launches are required, which increase a lot the cost, complexity and risk of the mission. It addition, it was assumed for the deployment study that the launches could be made at any time, but launch windows, as well as launcher facilities availability must be taken into account. For instance, the launch windows would imply a warning gap of 9 hours per year. All these factors make the mission almost unfeasible.

For these reasons, it was proposed the circular concept, which requires just one launch. Nevertheless, the deployment time in this case is too high (5 years and 10 months) although it was verified that most of the satellites of the constellation would spend less time in their transfer trajectories. What is more, the constellation is completely operational from the fourth year, as during the last year of the deployment, the Earth will be facing only the spacecraft that have occupied their final position in the operational orbit. Still, alternatives could be searched to reduce the deployment time, as performing a second launch. By this way, each launch vehicle could carry half of the satellites and by phasing them correctly, the deployment time could be reduced significantly.

Apart from this, other deployment methods could be studied in the future for reducing the delta V and the number of launches. For instance, the use of the moon as a parking orbit or for performing a fly-by could be a good possibility, even more if weak stability boundaries were used.

As the main problem for finding an efficient deployment method was the number of satellites of the constellation, it would be worth to study another totally different concept of mission using the solar sails. For this concept, only one spacecraft would be needed as it would always be in the Earth-Sun line, positioned in a pseudo- Lagrangian point closer to the Sun than L1.

B.14 AOCS: Sadhana Udayakumar

B.14.1 Introduction

The report discusses the Attitude and Orbit Control System (AOCS) of spacecraft for both mission concepts namely, Space Weather Diamond (SWD) concept and the Circular concept, in the interests of studying space weather at sub L1 locations to provide adequate warning time.

The report focuses mainly on the selection and configuration of the components that make up the AOCS. An attempt in designing a control system using Matlab and Simulink has also been made.

The following steps were followed in order to arrive at the final design.

1. Identification of main requirements and derived requirements of the mission.

2. Discussion of mission modes and the selection of the most suitable control method.

3. Selection of components i.e. the sensors and actuators, followed by their sizing and configuration.

On arriving at the base design having fulfilled the above points,

1. the fuel required by the AOCS was quantified.

2. a control system using a PID controller to develop attitude and rate control systems was designed.

B.14.2 AOCS Requirements

Pointing knowledge Pointing accuracy	Continuous pointing at the Sun for payload and power. Greater than +/- 1.4°
Control of orbital parameters	In order to be confident of orbital transfer manoeuvres that would place spacecraft in the right destination.
Mission duration	Provide sufficient redundancy to components in order to fulfil service for a 13-year long mission.
Fuel	Allocate fuel for AOCS applications (including margin) and unforeseeable situations (assumption).
Mass	Must be as light as possible

B.14.3 Mission Modes

Phase	Actions
Upper Stage Drop-Off Mode	De-tumble Acquire Sun Solar Array Deployment
Orbit transfer mode (OTM)	Orient spacecraft by performing a 90°yaw followed by a 90°nose-down pitch manoeuvre. Spin-up to 60 rpm in the duration of the burn. Reorient spacecraft to Sun-pointing.
Operational orbit mode (OM)	Always maintain Sun-pointing Slew 1°per day about the spacecraft- Sun line.
Safety mode (SM)	Similar to the Upper Stage Drop-Off Mode. Acquire Sun by controlling the attitude about the X, Y axes and the rate about the Z-axis. (Hardacre, 2012)
End of life mode (ELM)	Once the spacecraft has served the mission lifetime or is incapable of continuing due to any reason, it enters ELM. The fuel is used up, the batteries are drained and the computers are powered off in order to avoid any explosions due to collisions.

Chosen control method: 3-axis control stabilization

Main disturbance sources: SRP (Solar radiation pressure) and thruster misalignment during burns

The following components were chosen to sense and control the movements of the spacecraft. If the components selections differ between the two concepts, it has been mentioned.

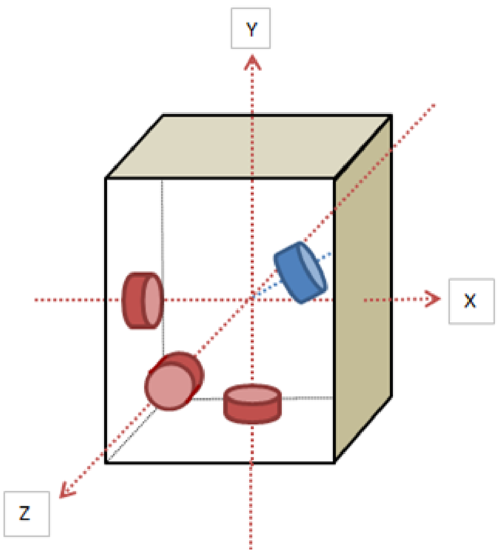
Item	Name	Supplier	Mass kg	Power W	Size mm	FoV	Pointing accuracy	Lifetime	Quantity
Inertial rate sensor (dual redundancy)	MIRAS 01	SSTL	2.8	5	324x191x54	N/A	N/A	7.5	2
Sun sensor	CSS	Bradford Eng. BV	0.365	0.2	1.8x106x49	128x128	<1°On-axis <2°Off-axis	13+	3
Star tracker	Micro ASC	Univ. Denmark	0.5	19	100x100x45 (DPU) 5x5x5 (CHU)	16.4x16.4	0.0016° EOL	30+	4

List of actuators

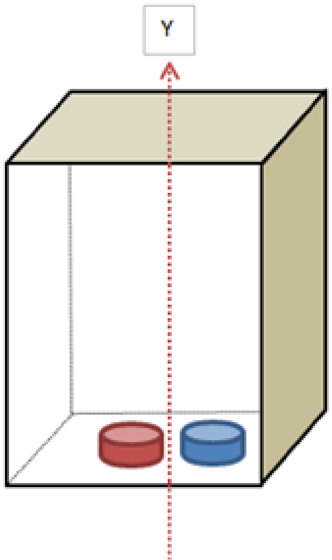
Concept	Item	Name	Supplier	Mass kg	Power W	Quantity
SWD	Reaction wheel	Microwheel 100SP-M	SSTL	2	5	4
Circ	Momentum wheel	RSI 1.6-33/60A	Rockwell Collins	$\simeq$2	17	2
Both	RCS thruster	10 N biprop	EADS Astrium	0.65	40	16

Configuration of components

See Figures B.33 and B.34 for illustrations of the configuration proposed for the AOCS actuators.



(a) SWD concept



(b) Circular concept

Figure B.33: Reaction/ Momentum wheel configuration (locations have been exaggerated in the above figures).

B.14.4 Fuel

The fuel has been quantified to perform momentum dumping and slew maneuvers. An estimate has not been included in terms of the spin up maneuver during the OTM. However due to the large margin used and exaggerated number of unpredicted attitude losses used to calculate the fuel, it should most likely be covered within the current fuel budget.

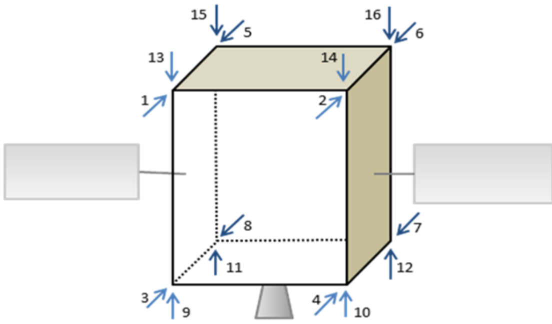


Figure B.34: Thruster configuration (same for both concepts, 16 thrusters, 5 DoF).

The number of thruster pulses include the pulses required for momentum dumping and the number of pulses that may be required to correct unexpected attitude losses.

Concept	Dist from Sun	SRP	No. thruster pulses	Momentum dumping (1 s pulse)	Slew man-oeuvres ($\simeq$6 s pulses)	Fuel
	AU	μN m		kg	kg	kg
SWD	0.821	33.1	2700+33000	0.51	8.0	10.3
Circ	0.853	22.2	2200 + 22000	0.64	3.3	4.7

B.14.5 Conclusion

The current AOCS system design is capable of fulfilling the requirements and perform effectively during the different modes. The control method selected is 3-axis stabilisation for both SWD and Circular concepts. The hardware components were sized based on the external disturbances. With the current selection of sensors and actuators, the spacecraft is capable of having knowledge and control over its attitude and orientation, and the orbital parameters including its location in orbit.

The components are COTS which have been flown and tested in previous missions. However the distance from the Earth ($>$ 0.15 AU) puts the spacecraft in a environment unprotected from radiation. The components are radiation hardened or have some set radiation dose limits. However in order to optimise the selection of hardware, more knowledge on the radiation environment is essential, which is part of the future work for the entire group.

Redundancy has been added to all components which should ensure service during the mission lifetime of at least 13 years.

The launch is scheduled in approximately 10 years time (2021). SWD is constrained by mass and cost. A few years can see a boost in micro technology with components having lower mass and using lesser power, with better radiation endurance and overall a compact, robust and high performing parts to make up the AOCS.

B.15 Requirements and Risk: Alastair Wayman

B.15.1 Space Weather

NASA defines space weather as "Conditions on the Sun and in the solar wind, magnetosphere, ionosphere and thermosphere that can influence the performance and reliability of space-borne and ground-based technological systems and can endanger human life or health (NASA, 2012b)." Usually, the effects of space weather are tolerable. However, periods of intense space weather activity can lead to serious implications on Earth. The most extreme of these space weather events are coronal mass ejections (CMEs): ejections of billions of tons of plasma from the surface of the Sun caused by the re-arrangement of the Sun's magnetic field.

Space Weather Risks

CMEs travel outward through space at speeds of up to 2500 km/s (Zhang et al., 2004). When directed towards the Earth they can take anywhere between a few hours and a few days to arrive. Once here, the plasma interacts with the near-Earth environment in several ways, all of which have an impact upon the systems and infrastructure we utilise. Protons and electrons can interact with spacecraft introducing single event effects that can ultimately lead to the failure of a spacecraft. These electrons and protons can also lead to similar effects on systems within the atmosphere as they are channelled down towards the poles by the Earth's magnetic field. They can upset the operation of avionics on polar flights as well as pose a radiation risk to the crew on board. As the protons travel down through the atmosphere they can collide with the atoms present liberating energetic neutrons. These neutrons also pose a radiation risk to crews on polar flights, leading to them being classed as radiation workers by the European Union since 2000 (EU Council, 1996).

As well as liberating neutrons, the collisions may also ionise the atoms, particularly in the ionosphere. This increase in the ionisation level of the ionosphere can cause an atmospheric layer to form that blocks all HF radio frequencies at the poles. This is of particular concern for airlines, as FAA regulations require them to maintain contact with air traffic control at all times. At low latitudes this is not a problem, as a geostationary SATCOM system can be used. This cannot be accessed north of 82°and so these Polar Cap Absorptions can lead to costly rerouting and diversions of aircraft over a period of several days (Baker et al., 2008). This increased ionisation can also be caused by increased levels of UV radiation incident upon the Earth, such as can be caused by solar flares. When the radiation from a large solar flare is directed at the Earth it can lead to the blocking of all HF radio communications on the entire sunlit side of the Earth for several minutes. As well as blocking communications, this ionisation has a further effect upon Global Satellite Navigation Systems (GNSS) users. The ionisation releases an electron into the atmosphere, increasing the total electron content (TEC). This increase in the TEC causes scintillations as well as propagation delays that lead to the signals from GNSS systems being degraded and large position errors introduced into their output.

Perhaps the most damaging and worrying effect from space weather comes from the magnetic field CMEs carry with them. When they reach the Earth, this magnetic field interacts with the Earth's own magnetic field causing it to rapidly vary. This varying magnetic field can generate huge currents in the long infrastructure of railways, pipelines and power distribution networks. The geomagnetically induced currents (GICs) generated are DC currents, and so power distribution networks are not designed to manage them. As such, they can cause huge amounts of damage. In 1989 a CME hit the Earth causing the magnetic field to drop at a rate of 480 nT/min in the Quebec region. Within 92 seconds induced currents in the power grid left 5 million people without power for over 9 hours. The total economic impact of this event is estimated at CDN$2Bn (Baker et al., 2008).

The 1989 event was small, with geological records showing an event leading to a change of 5000nT/min in 1921 (Kappenman, 2006). The potential is thus there for much more severe effects if a similar level of event were to occur today. Due to the increasing dependency of society on technology as well as the complex links between different segments, the effects of such an event would be devastating.

The effects of space weather against all of these systems can be mitigated with adequate early warning. 2 hours of warning would allow for the advanced rescheduling of polar flights, the rescheduling of operations reliant upon accurate GNSS signals, the reorientation or hibernation of spacecraft as well as the redistribution of power generation to less vulnerable areas (Baker et al., 2008).

Current Space Weather Monitoring Systems

Current systems exist to monitor space weather, however they all have serious limitations. The GOES, POES and Cluster spacecraft are currently operational in GEO and LEO. Due to their close proximity to the Earth they provide no warning, only current conditions, or now-casts. The STEREO system, comprising spacecraft both ahead and behind the Earth are imaging CMEs in 3d, allowing their velocity to be resolved. This offers fantastic warning time, but no measure of the severity of a CME. Currently, all space weather warning data is provided to Noaa by the ACE spacecraft positioned at L1. ACE is, however, approaching the end of its design life and, even at L1, provides no more than 1 hour of warning time.

The Space Weather Diamond

In response to these limitations both Nasa and Esa commissioned studies into space weather warning systems. The Nasa study was led by Dr. O. C. St. Cyr. It proposed a constellation of 4 spacecraft, all in eccentric, heliocentric orbits with a period of exactly 1 year. With an eccentricity of 0.1 they would have a radius of perihelion of 0.9 A.U., giving a 10x improvement in the warning provided in comparison to L1. The spacecraft were carefully phased such that their perihelions were equally spaced and all laid on the Earth-Sun line. This led to the constellation producing an apparent orbit (Distant Retrograde Orbit) about the Earth. Doing this maintains a spacecraft within 8 of the Earth-Sun line at all times with a minimum improvement in warning time of 6.5x compared to L1 (St. Cyr et al., 2000).

With this as an initial idea, the project team followed the aim set out by Hobbs (2011) to "develop a credible baseline design for a mission to provide useful warning of hazardous SW events based on the Space Weather Diamond concept".

B.15.2 Requirements

Analysis of the user needs allowed for a detailed set of requirements to be generated. These defined the need for a system that shall:

- Provide at least 2 hours warning for all hazardous, Earth-bound space weather
- Detect all Earth-bound CMEs with an angular width greater than 35°
- Identify CMEs through the use of in situ detection of particulate radiation and the interplanetary magnetic field
- Provide warning data to end users with a communication and processing delay not worse than 5 minutes
- Minimise the occurrence of false alerts through the use of integrity data in the form of standard deviations and ranges for all data products
- Cost less than €300M

It was identified that remote sensing data would be particularly useful to the system. However, the requirements for remote sensing would drive the cost of the system and so it was chosen to source this data via alternative means. The requirement to provide 2 hours warning time for the fastest CMEs drove the design of the constellation, requiring one spacecraft to be in each of the cones defined in Figure B.35 at all times. This required a constellation of 10 identical spacecraft giving the system an angular resolution of 14.60°. Several limitations existed for this system, including the large delta-V required as well as the need for 5 launches within a 12 month period. This led to the design of a second system consisting of 12 spacecraft, equally spaced around a circular orbit, 0.85297 A.U. from the Sun.

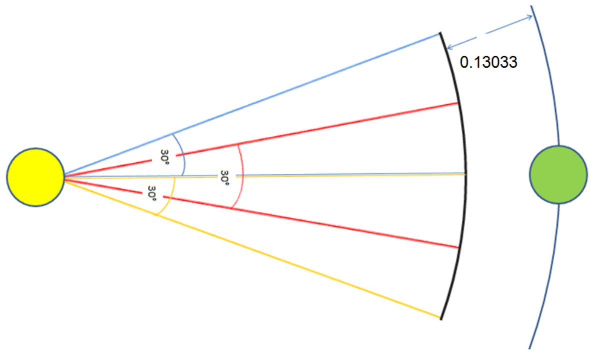


Figure B.35: Detection cones used to define mission requirements.

B.15.3 Risk

The risks to both systems were analysed, with the main risks being the number of launches required for the St. Cyr concept as well as the radiation environment for both. This required the use of 4 mm of aluminium shielding on the sun-facing side of the spacecraft.

B.15.4 References

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